HiPER: a Roadmap for Future Space Exploration with innovative Electric Propulsion Technologies

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Electric Propulsion could play a very important role in future international space exploration programmes by enabling more affordable and sustainable space-to-space missions. HiPER is a project funded by the European Union aimed at laying the technical and programmatic foundations for the development of innovative Electric Propulsion technologies to fulfill future space transportation and space exploration needs. First year of activities has been mainly devoted to select and study the near term and long term scenarios which could benefit from the increase of the operational power level of the Electric Propulsion system. The paper presents results from such activities and describes the most relevant mission and transportation scenarios so far addressed.

Nomenclature

AOCS	=	Attitude&Orbit Control Subsystem
BOL	=	Begin Of Life
EML1	=	Earth-Moon lagrangian point L1
EOL	=	End Of Life
EP	=	Electric Propulsion
GTO	=	Geostationary Tranfer Orbit
LEO	=	Low Earth Orbit
MSR	=	Mars Sample Return
NEO	=	Near Earth Object
NEP	=	Nuclear Electric Power
PPU	=	Power Processing Unit
SEL2	=	Sun-Earth lagrangian point L2
SEP	=	Solar Electric Power
TPBVP	=	Two Points Boundary Value Problem

I. Introduction

Space exploration is crucial to answer some key questions about the evolution of the solar system and life beyond Earth. Both robotic and human exploration in the Earth-Moon system and beyond is necessary to accomplish these goals.

However, a sustainable and affordable space exploration programme is a challenge that no single nation can do on its own. Even in the United States, that still represents the pillar of space exploration, internal discussion has recently emerged about current NASA budget as it cannot afford the space exploration plan originally envisioned for

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the ambitious "Constellation program" mainly focused on replacing the aging Space Shuttle and returning humans to the lunar surface. Therefore, a future international cooperation on globally coordinated space exploration programs, sharing challenging and peaceful goals seems to be necessary[1-4].

Apart from contributing to the expansion of scientific knowledge and expertise in further exploration, a global-scale space exploration can also benefit society by driving technological innovation for a better tenor of living and by enabling new business opportunities. Indeed, an economic expansion for all that companies that will provide commercial services (e.g. transportation, telecommunications and navigation systems and, in a longer run, resource extraction and processing capabilities) will be generated by the potential opportunities of a new space age. Nevertheless, focal point in the implementation of this ambitious and enduring space programme is the identification of clear long-term goals and then the improvement of the capability for in-space servicing and space transportation to reach the selected destinations.

In this perspective, a global-scale and multi-decade space exploration program will be not just another big and difficult space programme but it will be a sustainable audacious exploration venture and it will stay well beyond the 21st century.

II. HiPER Reference Application Scenario and Objective

Europe's long term space exploration goals include missions to Near Earth Objects (NEOs), the operation of large observatories at the Sun-Earth L2 (SEL2) Lagrange point and - in the longer run - a manned mission to Mars. However, such ambitious objectives will only be made possible by a complex plan of pre-cursor robotic missions and will necessarily involve cooperation with NASA and other space agencies.

Electric Propulsion could play a very important role for the implementation of this scenario, by enabling more affordable and sustainable space-to-space missions. Today's state-of-the-art Electric Propulsion has achieved the capability to enable high efficiency "one shot" interplanetary missions, such as those developed or under development for the European SMART-1 and Bepi Colombo spacecraft, for the American Deep-Space 1 and DAWN, and for the Japanese Hayabusa. What is needed next is the development of EP systems intended for the much larger payloads and re-use capability required by the future mission scenario outlined above.

Present European capability tops at about 5 kW thruster power level; however, most future mission scenario for EP-enhanced space transportation of large payloads foresee the use of 10 to 25 kW thrusters. Re-use capability can be achieved by improving the reliability of key components and technologies, and by a proper thruster design taking into consideration mission requirements (such as long term in-space storage and similar) which have not traditionally been addressed in past EP designs.

Nevertheless, the ability to fully realize the transportation and economic benefits deriving from the application of advanced electric propulsion is also dependent upon the development of suitable electrical power sources. At present, these are essentially based on photo-voltaic solar arrays, although some early attempts at nuclear EP propelled missions to the outer planets are under study. In the near and medium term, the commercial application of large scale EP space transportation, the robotic missions foreseen for the exploration programmes and the unmanned space science observatories will be able to fulfill their requirements relying on new concepts in solar power.

However, in the longer term, as larger payloads and more powerful science observatories will be needed, a new generation of power sources will become essential to power the spacecrafts: these will be most likely based on nuclear technology. Once developed, they will also provide a most cost effective EP capability for larger manned or cargo missions.

HiPER ("High Power Electric propulsion: a Roadmap for the future") is a project funded by the European Union (EU) under the Space Theme of the 7th Framework Programme, and aimed at laying the technical and programmatic foundations for the development of innovative Electric Propulsion technologies (and of the related power generation systems) to fulfill future space transportation and space exploration needs.

It is an ambitious, 3-year collaborative activity involving 19 partners from 6 European Countries under the co-ordination of Alta. HiPER's objective will be pursued by conceiving and substantiating a long term vision for mission-driven Electric Propulsion development, considering realistic advances in state of the art of EP related technology and performing basic research and proof-of-concept experiments on some of the key concepts identified by such a vision.

Three different EP concepts (and related solar electric and nuclear power generators) have received particular attention in recent years and are at present considered as the candidates with the highest application potential: Hall Effect Thrusters (HET), Gridded Ion Engines (GIE) and MagnetoPlasmaDynamic Thrusters (MPDT). Each of the three has characteristics which make it particularly suitable to a specific class of mission.

III. Future Mission Classes enabled by Electric Propulsion

Thanks to the high thrust efficiency and lifetime, modern EP technologies enable mass savings, launch flexibility, long interplanetary journeys and faster missions with no gravity assist constraints. This also open the way to transferring large payloads through the solar system in a much more affordable way than in the past. Besides, larger payloads can be achieved by increasing the operational power level of the propulsion systems.

According to political, social and economic aspects related to a sustainable and affordable space exploration scenario, during the first year of activities, the HiPER Mission Analysis team was devoted to define some near and long term mission and transportation scenarios which could benefit from high power EP.

The HiPER preliminary list of mission classes includes: orbit transfer in the Earth-Moon system, NEOs exploration, exploitation and risk mitigation, Mars sample return, interplanetary transportation (e.g. Mars), outer solar system and beyond robotic exploration, and constellation reseeding.

Some of them (e.g. transportation in the Earth-Moon system or Mars sample return mission) can be grouped as near term missions that, apart from the development of high power EP thrusters, rely on the assumption of Solar Electric Propulsion (SEP). SEP missions are designed employing solar panels to generate the required on-board power and their mission architectures are generally applicable to a wide range of missions. However, due to the rapid decrease of solar radiation flux that is an inverse relationship with the Sun distance squared, SEP missions can be considered inappropriate for missions that go beyond Mars orbit[5]. Therefore, an alternative source of energy is mandatory for outer solar system exploration.

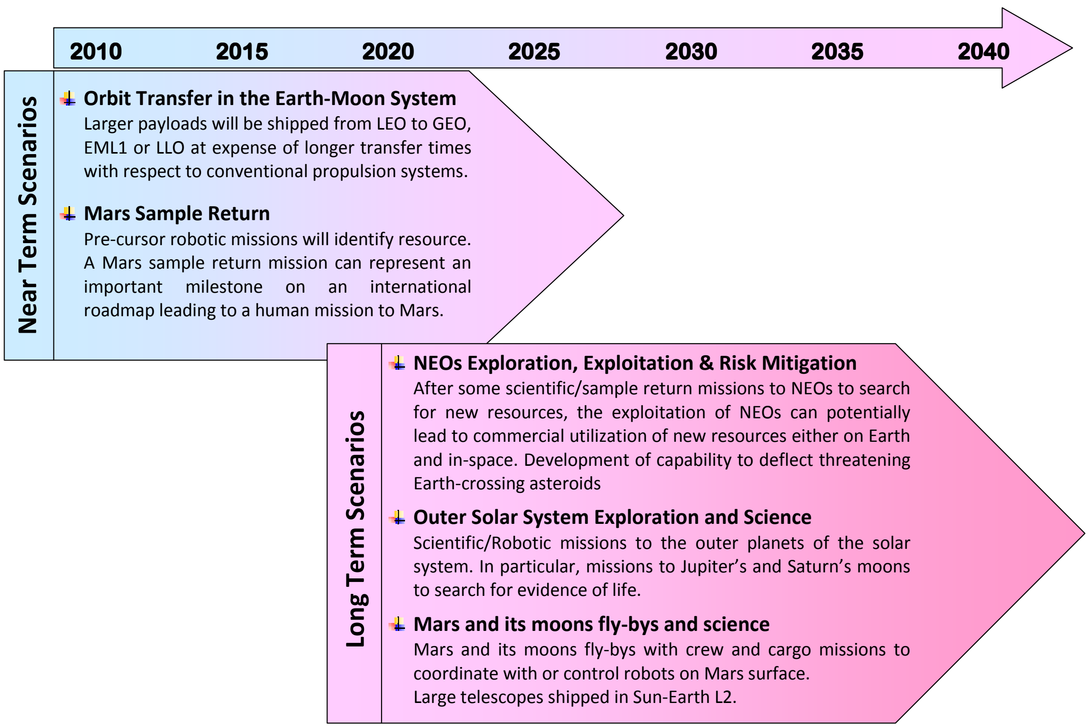


Figure 3. HiPER proposed near term and long term high power EP mission scenarios

Nevertheless, in a longer run, a nuclear power generation source coupled with high power EP thrusters will allow a wider range of mission classes as benefits deriving from NEP tend to grow with the distance travelled, especially away from the sun. Besides, a NEP system provides surplus power for on-board use. For instance, comfortable living conditions to the crew, redundant safety systems and active protection against solar and cosmic radiations can be assured.

IV. Solar and Nuclear Power Generation

A preliminary analysis has been carried out on the main characteristics of power generation subsystems. Hereafter, solar and nuclear power generation preliminary requirements are outlined.

A. Solar Power Generation

Solar array systems are the most common method to provide electric power to the spacecraft by solar illumination of solar cells. Even if their power conversion efficiency from sunlight to electricity is relatively low (from 10% to 30% at BOL depending on the technology), the flexibility and variability of the many solar array types and configurations allow their application in many mission classes and different space environments.

For instance, in Earth science missions solar arrays require shielding from radiation and their weight is not a critical aspect. For commercial applications, solar arrays design is the result of a trade-off aimed at maximizing the on-board power level maintaining cost and mass efficiency. Finally, in interplanetary missions, lightweight solar arrays able to be stowed into compact launch volumes are preferred.

Nevertheless, the increasing electrical power demand on-board telecommunication spacecrafts and the more aggressive and demanding mission requirements have led the major manufacturers to develop solar arrays more and more efficient. This was possible thanks to the improvements of the major solar array technologies at subsystem level such as innovative structural platforms, deployment systems, novel mechanisms and of course higher efficiency solar cells.

Table 1. Efficiencies of main cell technologies

Cell Technology	BOL Efficiency	EOL efficiency (1E15 MeV eq. rad. Dam.)	Notes
Crystalline Silicon	17%	12%	Flight Proven
GaAs/Ge TJ (With different Ge substrate thickness)	28%	22%	Flight Proven
GaAs/Ge TJ next generation thin cells	>30%	>25%	Prototype
CIGS (thin film)	13%	11%	Prototype
Refractive Concentrator	>27%	24%	Prototype (SCARLET flew on DS-1 but it was less efficient)

About solar cells, the first single-crystal-silicon widely used in the space industry from sixties had a 10% BOL efficiency that was improved up to 17% in the early eighties. Then, in the late eighties and nineties, the single junction gallium arsenide replaced the use of silicon cells exceeding the 19% of efficiency. But only with the recent manufacturing of rigid solar array using triple-junction solar cells (GaInP2/GaAs/Ge) with a Germanium substrate BOL efficiency has grown up to 30%[6].

Table 2. Some figures of main solar array technologies

Solar Array	BOL Power Density [W/m^2]	BOL specific mass [α=kg/kW]	Stowed Power Density [kW/m^3]
Rigid panel technology with GaAs TJ SOA cells	>300	15 – 20	8 – 12
Ultra light rigid panels with GaAs TJ SOA cells	>300	6 – 8	10 – 14
Thin film solar array	>120	12 – 14	40 – 60
Refractive concentrator with GaAs TJ SOA cells (Stretched lens array)	> 300	< 4	80 – 100

Because of their light weight, low cost and high radiation hardness with respect to rigid arrays, thin film solar cells were also developed. The family of thin film solar cells includes devices based on amorphous silicon (a-Si), cadmium telluride (CdTe), Copper-Indium-Selenide (CIS) and Copper-Indium-Gallium-Selenide (CIGS). At the moment, this technology has been only used for terrestrial applications. Furthermore, even if BOL efficiencies (13% for market application and up to 19% in laboratory) are significantly less than triple-junction solar cells, their potential mass and cost savings are very interesting.

Refractive concentrator solar arrays have been also developed in order to reduce the use of solar cells with a consequent mass reduction. This allows the cover glass thickness to be increased with a lower mass penalty than with non-concentrator arrays. Recently, NASA, ENTECH Inc. and other U.S. team members have developed an ultra-light version of the flight-proven SCARLET array (Deep Space 1 mission), but with much better performance metrics[7]. Efficiencies greater than 27% have been obtained and, thanks to the ticker cover glass and to the additional protection that form the concentrator, an increased tolerance to radiation has been analytically demonstrated (recent studies showed a radiation degradation that could withstand 13 flow spiral transits of the Earth's Van Allen belts[8]).

In Table 1 and Table 2, the efficiencies and some figures inherent to main cell and solar array technologies are shown.

B. Nuclear Power Generation

The majority of NEP experience is in LEO being the Russian RORSAT programme. Some early US studies and projects were aimed at adapting the Russian technology for a demonstration mission in a higher LEO orbit (the Nuclear Electric Propulsion Space Technology Programme (NEPSTP). Others were aimed at developing a new, higher power reactor, the SP-100.

The reactor and associated systems technology varies with the power required. The only in-orbit experience to date is with reactors generating up to 5KWe (from 130KWth) with thermionic conversion. The abandoned US SP-100 project envisaged a 100 KWe from 2500 KWth reactor with liquid metal energy conversion. Studies were made to increase this to 750 KWe with Brayton cycle technology. For the MWe range of power gas cooled reactors are seen as the best option but the magnitude of the power and propellant budgets becomes very challenging.

There is a very significant difference between the specific mass achieved with the most advanced Russian flight demonstrated reactor (TOPAZ) of 120 kg/KWe compared to a goal of 10 kg/KWe for a 10 MWe power plant. The baseline SP-100 demonstrated about 50 kg/KWe and was thought to be capable of being developed to operate in the 25-30 kg/KWe range. This is compared with a solar array specific mass of about 10 kg/KWe. The more optimistic targets for an SP-100 development are in the 25-30 kg/KWe.

Table 3. Specific mass breakdown for a 30 kg/kWe nuclear power generation system for space applications

Reactor System including: ▪ Thermal to electrical conversion system ▪ Shielding ▪ Cooling	22 kg/kWe
Power Management System including: ▪ Conditioning and distribuition ▪ Cables and connectors ▪ Thermal management	3 kg/kWe
Electric Propulsion System including: ▪ Thrusters ▪ Power Processing Units ▪ (Dry) Propellant storage and feed system ▪ Pointing mechanisms	3 kg/kWe
Spacecraft Systems including: ▪ Structure ▪ Communications ▪ AOCS ▪ Environmental protection and management	2 kg/kWe

There is recognition that <10 kg/KWe is needed for a manned mission to Mars but the technical development still looks extremely challenging. For these reasons, the HiPER Nuclear Power Generation Team proposes a target of 30 kg/KWe recognizing that achieving even 35 kg/KWe in the foreseeable future may stretch the boundaries of feasibility in the medium term. To meet the most demanding missions it will be seen however that only a breakthrough to 10 kg/KWe will make these possible and there needs to be a development route from the medium to the long term.

The principal features of the realized space nuclear fission power projects and major studies are summarized in Table 4. None were used for nuclear electric propulsion although subsequent studies were based on the use of Topaz-2 for this purpose. The most successful was the Buk reactor which powered the radar payload in the USSR's

RORSAT programme. The Topaz-2 programme was looking encouraging in both robustness and increasing lifetime when it was cancelled. However it took until the SP-100 experiments to demonstrate specific mass and lifetimes consistent with mission aspirations. All the systems used thermoelectric or thermoemission conversion machinery. Studies indicated the need to move to Brayton cycle technology to achieve better efficiency and specific mass but developing the necessary lifetimes remained unproven.

Table 4. Space Nuclear Fission Power Generator Projects

Project	Romashka	Buk	Topaz-1	Topaz-2	SNAP-10A	STAR-R	SP-100	SAFE-400
Country	Russia	Russia	Russia	Russia	USA	USA	USA	USA
Development Status	Prototype	32 Flights	2 Flights	Ground test	Flight test	Study	Prototype	Study
Timescale	1961-66	1970-78	1987	1992	1965	1972	1992	2007
Reactor Type	Fast	Fast	Thermal	Thermal	Thermal	Fast	Thermal	Fast
Fuel	UC_2	UO_2	UO_2	UO_2	U-ZrHx	UO_2+W or Re	UN	UN
Conversion Type	Thermoelectric	Thermoelectric	Thermoemission	Thermoemission	Thermoemission	Thermoemmision	Thermoelectric	Thermoelectric
Thermal Power (kW)	40	100	150	135	45.5	N/K	2000	400
Electrical Power (kWe)	0.5 - 0.8	3.5	6	5.5	0.5	10 - 15	100	100
Efficiency %		~3.5	~4	~3.7	~1	N/K	~5	~25
System Mass (kg)	N/K	930	980	1061	435	~600	5422	N/K
Specific Mass (kg/KWe)	N/K	266	163	193	870	N/K	54	N/K
Design operating Years	0.5	1	1	3	~1	N/A	≤10	10
Achieved operating Years	0.17	0.5	0.96	1.5	0.12	N/A	N/A	N/A

V. Mission Analysis of proposed scenarios: Preliminary Results

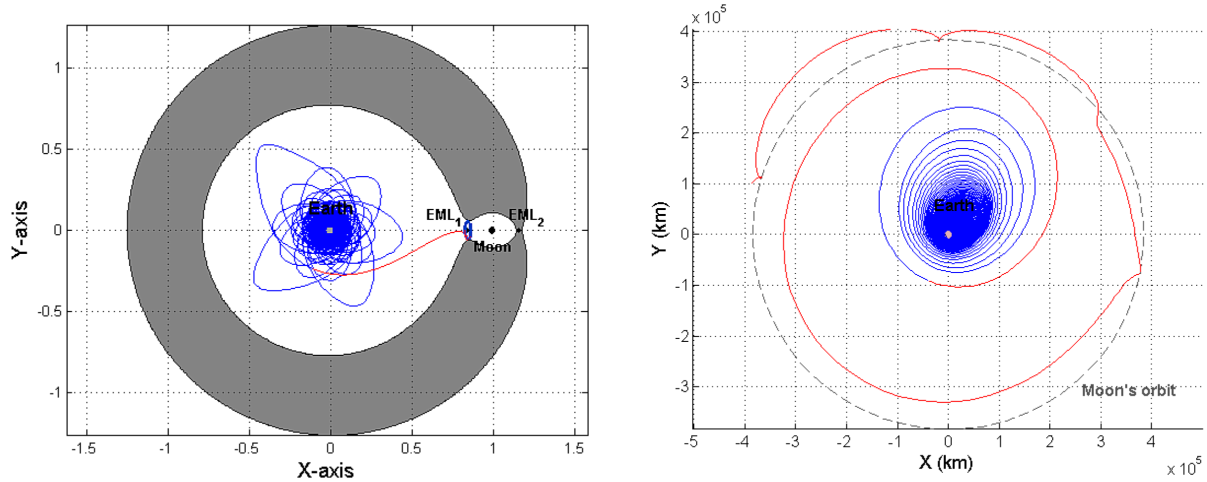
The preliminary results obtained from mission analysis of proposed mission and transportation scenarios provide the initial principal transfer parameters that constitute the basis for future global mission analysis optimization activities. Besides, these results also define a broad range of performance (thrust, specific impulse, power level, throughput mass, firing time, etc.) useful to the HiPER technological teams involved in the development of high power EP thrusters and power generation systems.

A. SEP Orbit Transfer in the Earth-Moon System

This near term scenario was studied combining Electric Propulsion (low-thrust) with the Circular Restricted Three-Body model (circular orbits of both Earth and Moon). In particular, a LEO to the Earth-Moon Lagrangian point L1 (EML1) transfer was studied. A low-energy transfer was designed and thus a significant amount of propellant mass was saved. This was possible because such trajectories are modeled by making use of gravity as much as possible; by simply exploiting the simultaneous gravitational fields of two attractors (the Earth and the Moon), the ΔV needed to perform the transfer is reduced. Apart from a reduction of the transfer ΔV, preliminary design of low-energy trajectories provides a more accurate solution. In particular in the region where the gravitational fields of Earth and Moon are comparable (e.g. in EML1), the spacecraft experiences gravitational forces from both the Earth and the Moon and thus only this kind of modelization could offer accurate results.

A halo orbit around EML1 with an arbitrary value of 1250 km out of plane amplitude has been chosen as the final orbit of the transfer. In order to maximize launcher capability, a circular orbit at 1000 km of altitude has been selected. This value represents the minimum altitude at which atmospheric drag can be neglected. In fact, as a large solar array surface is needed to obtain high power levels (hundreds of square meters), at lower altitudes the spacecraft exposed cross sectional area and therefore its ballistic coefficient could represent a limiting factor for orbit raising. An initial mass of 20 metric tons has been assumed as it is compatible with state-of-the-art performance of heavy launch vehicles. A preliminary mission analysis was carried out varying the power level available for the thrusters (assumed to be the 81% of the power produced from solar array/nuclear reactor due to the power losses). In particular the values of 80 kW, 100 kW and 200 kW were assumed. Three specific impulse values were also investigated (2000 s, 5000 s and 10000 s). Known the power available at thruster level and the specific impulse, the thrust level can be easily found if thrust efficiency η_T is also known. In particular, the value η_T=0.5, 0.65 and 0.75 was assumed for Isp=2000 s, 5000 s and 10000 s respectively.

The overall LEO-EML1 implemented transfer is composed of a ‘forward’ and a ‘backward’ strategy. The ‘backward’ phase starts from the final halo. Asymptotic solutions of the CR3BP equations associated to the periodic halo orbit were generated by a ballistic propagation of a small initial perturbation in the direction of the diverging eigenvectors and the asymptotic orbit closest to the Earth was selected. Then, the backward integration in time of the CR3BP equations of motion (i.e. the backward ballistic propagation) on the selected asymptotic orbit was stopped when the closest point to the Earth is reached.

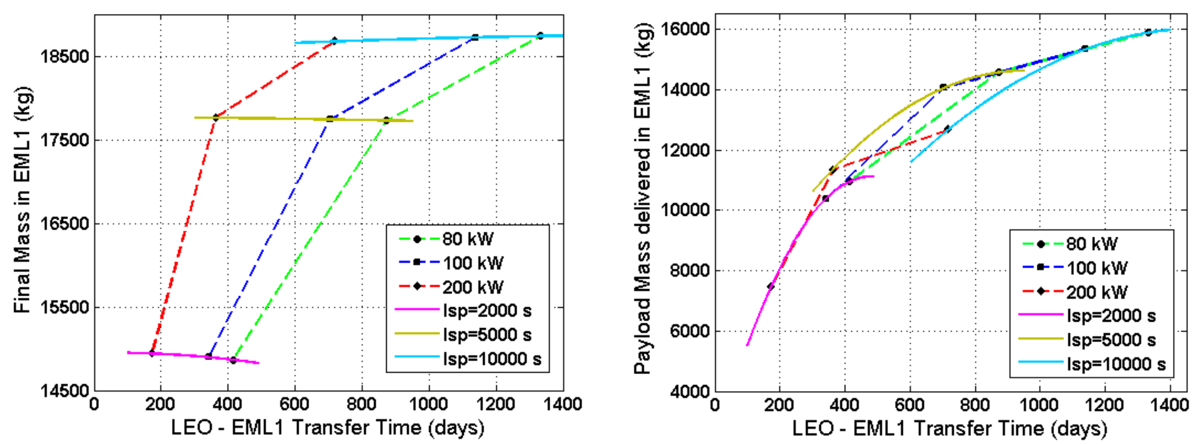


a) LEO-EML1 transfer in the CR3BP synodic system of reference

b) LEO-EML1 transfer in an inertial geocentric system of reference

Figure 4. LEO-EML1 overall transfer in both in a synodic frame and in an inertial geocentric system of reference

The ‘forward’ phase starts from LEO and ends in the end point of the ‘backward’ phase where thrusters are switched off and then the spacecraft ‘ballistically’ moves towards its final halo orbit in EML1. The ‘forward’ phase was divided into three parts: in the first part (from LEO up to about 24000 km of altitude), a continuous tangential thrust was applied as we want to cross Van Allen belts as fast as possible. Then, a tangential thrust only around the perigee was applied to increase the apogee altitude until a selected eccentricity is reached. These two parts were analyzed using the DOrbit software. Finally, an optimized phase was implemented to solve a TPBVP to match the two final states. Eclipse effects and sun perturbations have been considered. As the power levels considered are very high, no thrust during eclipse was assumed. The overall LEO-EML1 EP transfer for a reference case is shown in Fig. 4 both in a rotating and in an inertial system of reference.



a) Final mass delivered in EML1 vs Transfer Time

b) Payload mass delivered in EML1 vs Transfer Time

Figure 5. Final Spacecraft Mass and Payload Mass delivered in EML1 vs Transfer Time in a LEO– EML1 EP transfer

For preliminary mass budgets, we assumed the dry mass to be composed of solar arrays (specific power of 50 W/kg and 300 W/m^2), PPUs and thrusters (100 kg/N), platform mass (500 kg), tanks and related structures (20% of the total propellant mass) and payload mass (not known a priori). About the solar electric generator mass, it is worth noting that the state-of-the-art value of 50 W/kg can be improved over the next five-ten years up to > 300 W/kg with a stowed power density of about 80-100 kW/m^3 (see Table 2).

The final mass obtained in EML1 (and thus the propellant mass needed to perform the overall transfer) for all the cases analyzed is shown in Fig. 5-a). Furthermore, taking into account the propellant mass needed for the transfer and the spacecraft dry mass breakdown above discussed, it is possible to compute the payload mass delivered in EML1. Results are shown in Fig. 5-b).

B. SEP Mars Sample Return

Considered as a key mission by all space agencies and a priority for Europe, Mars Sample Return (MSR) can represent a very good candidate to exploit the benefits of Electric Propulsion (in combination with conventional propulsion) within a reasonable time-scale. In addition to the high scientific value deriving by the analysis of the sample (composition, assessment of evidence of prebiotic processes/past life, analysis of geological processes, etc..), it is recognized that MSR also would have a big impact on the public opinion, just the way it happened when Apollo samples returned from the Moon.

Many studies have been carried out on MSR missions by the most important space agencies. Since 2003, ESA has included MSR as a flagship mission within its Aurora Exploration program and has conducted several internal and industrial assessment studies on this topic. However, most of these studies[9,10] assume as baseline solution the use of conventional propulsion. Therefore, to fit within current and near-future technologies and launch vehicle capabilities, two flight elements (a lander and an orbiter) put in orbit with two distinct launches have been considered as reference mission architecture. Alternatively, the use of Electric Propulsion at expense of trip time and dry mass, could allow performing an end-to-end mission from a single launch. However, as there are still some areas in which chemical propulsion is preferred (e.g. release manoeuvres, docking/rendez-vous manoeuvres, fast AOCS manoeuvres) the SEP MSR mission shall be integrated with conventional thrusters.

An Ariane 5 ECA launcher was assumed. It is able to deliver 20 metric tons on a 200 km altitude circular orbit, 10 metric tons on a GTO and 5.5 metric tons on a departure hyperbola with V_{inf}=2.3 km/s. Several high-power SEP MSR missions have been studied varying the specific impulse and the thrust level (and thus the power level) used during the overall transfer.

After the orbit injection by the launcher, the vehicle has to reach escape conditions to start the Earth-Mars interplanetary transfer. No Moon fly-by option was considered at this stage. During the escape phase, constant tangential thrust and no out-of-plane corrections were assumed. Depending on the thrust level and initial mass assumed, the escape time from Earth can take quite long (see Table 5). In order to give some credibility to the MSR mission, all the cases analyzed with an escape phase longer than two years were neglected.

Table 5. SEP MSR Earth escape phase from different initial orbits and conditions

Case	Vehicle mass (kg)	Injection periapsis altitude (km)	Injection apoapsis altitude (km)	Isp (s)	Thrust (N)	Electric Power from SA (kW)	Escape duration (days)	Escape duration (years)	Propellant mass (kg)
Case 1	20000	200	200	2500	2.44	70	588	1.61	5056
Case 2	20000	200	200	2500	4.88	140	296	0.81	5096
Case 3	20000	200	200	5000	1.22	57	1325	3.63	2683
Case 4	20000	200	200	5000	2.44	114	635	1.74	2400
Case 5	20000	200	200	5000	4.88	228	318	0.87	2739
Case 6	20000	200	200	10000	0.61	50	3161	8.66	1783
Case 7	10000	200	36000	2500	1.22	35	310	0.85	1333
Case 8	10000	200	36000	2500	2.44	70	159	0.44	1370
Case 9	10000	200	36000	5000	1.22	57	325	0.89	701
Case 10	10000	200	36000	10000	2.44	197	173	0.47	373
Case 11	10000	200	36000	15000	2.44	277	167	0.46	239

To reduce mission duration, most of SEP MSR transfer trajectories were designed optimizing the transfer time. The main drawback of this choice is to qualify thrusters for long continuous firing time. Most of the cases analyzed were also designed optimizing the propellant mass consumption.

For preliminary mass budgets, we assumed the dry mass to be composed of solar arrays (specific power of 50 W/kg and 300 W/m^2 at 1 AU), PPUs and thrusters (100 kg/N), platform mass (500 kg), tanks and related structures (20% of the total propellant mass) and payload mass. A payload mass of 3000 kg was assumed. It is composed of a lander, rover and ascent vehicle able to put 20 kg of Mars samples into Low Mars Orbit (LMO). Mars insertion manoeuvre and Earth reentry were assumed to start from hyperbolic trajectories.

In Table 6, the preliminary results obtained for the SEP MSR mission are shown. Simulations were performed using the orbit propagator software PSIMU for departure and insertion trajectories and the software ETOPH for the optimized interplanetary trajectories. A negative margin means that a SEP MSR with that combination of initial mass, specific impulse and thrust is not feasible. Some case have a positive margin, but only those cases in which a positive margin is associated to an acceptable transfer time (e.g. Case 5) should be considered as interesting combinations. However, these preliminary figures has only the objective to identify the most interesting thrust and specific impulse ranges that will be investigated more in depth in a following global optimization analysis.

Table 6. SEP MSR mission: summary of the resultsof the cases analyzed

Case	Initial mass (kg)	Initial orbit (km)	Specific Impulse (s)	Thrust (N)	Transfer Optimization criteria	SA Power (kW)	Transfer time (years)	Final Mass margin (kg)
Case 1.1	20000	200 x 200	2500	2.44	Minimum Time	70	6.93	**-4889**
Case 1.2	20000	200 x 200	2500	2.44	Minimum Consumption	70	9.26	**+430**
Case 2.1	20000	200 x 200	2500	4.88	Minimum Time	140	4.91	**-5075**
Case 2.2	20000	200 x 200	2500	4.88	Minimum Consumption	140	5.06	**-1033**
Case 4	20000	200 x 200	5000	2.44	Minimum Time	114	8.45	**+4489**
Case 5	20000	200 x 200	5000	4.88	Minimum Time	228	4.55	**+1498**
Case 7.1	10000	200 x 36000	2500	1.22	Minimum Time	35	6.06	**-1088**
Case 7.2	10000	200 x 36000	2500	1.22	Minimum Consumption	35	6.55	**-16**
Case 8.1	10000	200 x 36000	2500	2.44	Minimum Time	70	4.08	**-2481**
Case 8.2	10000	200 x 36000	2500	2.44	Minimum Consumption	70	3.99	**-1166**
Case 9	10000	200 x 36000	5000	1.22	Minimum Time	57	7.71	**+1042**
Case 10.1	10000	200 x 36000	10000	2.44	Minimum Time	197	5.48	**-763**
Case 10.2	10000	200 x 36000	10000	2.44	Minimum Consumption	197	4.97	**-196**
Case 11	10000	200 x 36000	15000	2.44	Minimum Time	277	4.72	**-925**

C. SEP/NEP Near Earth Objects Exploration, Exploitation and Risk mitigation

Near-Earth Objects (NEOs) are comets and asteroids orbiting in the Earth's neighborhood. The scientific interest in comets and asteroids is threefold.

First of all, they represent an important record of the solar system formation process. In fact, the most common hypothesis is that some 4.6 billion years ago, outer planets (Jupiter, Saturn, Uranus and Neptune) formed from an agglomeration of billions of comets while inner planets (Mercury, Venus, Earth and Mars) from an agglomeration of asteroids. This means that studying the composition of comets and asteroids we can have some information on the primordial mixture from which the planets formed.

Secondly, as they are so close to the Earth, they could be most easily exploited for their materials. In fact, the mining of useful space resources can potentially lead to in-space utilization addressed to the development of space structures or to the generation of fuel for the exploration and colonization of our solar system.

Finally, asteroids are also potentially the most hazardous objects in space for the Earth. In fact, with an average interval of about 100 years, rocky or iron asteroids larger than about 50 meters would be expected to reach the Earth's surface and cause local disasters. But, on an average of every few hundred thousand years or so, asteroids larger than a kilometer could cause global disasters. However, in this case, given several years warning time, existing technology could be used to deflect the threatening object away from Earth.

Moreover, as many NEOs are relatively close to Earth, these missions could be accomplished by extending and validating the capabilities of the interplanetary transportation and human support systems of currently planned crew and cargo systems.

For this preliminary analysis, an asteroid database containing about 20000 catalogued and not-catalogued objects was established from the data available in the software EPOCH[11]. Main characteristics of the objects in the database are the semi-major axis (in AU), the inclination and the orbital eccentricity.

Due to the very large number of objects in the database, two representative and attractive classes of NEOs were selected: the first class, hereafter called "close to the Earth", refers to NEOs with a semi-major axis between 1 and 1.5 AU (their orbits are between Earth and Mars); the second class, hereafter called "farther from Earth", refers to NEOs with a semi-major axis between 2 and 3 AU (their orbits are between Mars and Jupiter). Within each class of NEOs, a reference object was chosen to perform the transfer. This selection was performed in the catalogued objects (with recognizable international numbering).

The mean values and standard deviation of the main orbital elements of both NEOs classes are shown in Table 7 and Table 8 .

Table 7. Mean values and standard deviation of "close to Earth" NEOs

	Semi-major axis (AU)	Eccentricity (-)	Orbital Inclination wrt Ecliptic (°)
Mean values	1.2984	0.3946	18
Standard deviation	0.156	0.196	14.68

Table 8. Mean values and standard deviation of "farther from Earth" NEOs

	Semi-major axis (AU)	Eccentricity (-)	Orbital Inclination wrt Ecliptic (°)
Mean values	2.503	0.163	6.942
Standard deviation	0.224	0.078	5.224

In the first class, the NEO whose orbital elements are closest to the mean values is the asteroid n° 1685-Toro. Its main orbital characteristics are shown in Table 9. In the second class, the NEO whose orbital elements are closest to the mean values is the asteroid n° 3218- Delphine whose orbital characteristics are shown in Table 10.

Table 9. Main orbital parameters of "close to Earth" object 1685-Toro

Semi-major axis (AU)	Eccentricity (-)	Orbital Inclination wrt Ecliptic (°)	RAAN (deg)	Argument of perigee (deg)	Mean anomaly (deg)	Julian Day
1.367	0.436	9.375	274.5	126.8	60.08	2448700.5

Table 10. Main orbital parameters of "farther from Earth" object 3218-Delphine

Semi-major axis (AU)	Eccentricity (-)	Orbital Inclination wrt Ecliptic (°)	RAAN (deg)	Argument of perigee (deg)	Mean anomaly (deg)	Julian Day
2.521	0.218	2.705	183.6	240.5	266.5	2448700.5

EPOCH was used to perform continuous thrusting from Earth escape (just a parabolic trajectory from Earth) to the rendez-vous with the selected object. A feature is included in the tool for an enhanced starting with an excess hyperbolic velocity at the Earth escape.

The optimizer tool selects by itself the optimal specific impulse at each time of the mission in a so-called "Power Limited" optimization process. Therefore the electrical power (P_e) is kept constant to perform a balance between the thrust force (F) and the specific impulse (I_{sp}) during a continuous thrusting transfer. The "constant power" strategy of course is representative in the case of NEP. On the other hand, assuming a SEP spacecraft, the decrease of power available must be taken into account as spacecraft moves away from the Sun.

The parametric study of a rendezvous from Earth escape (with zero m/s of excess hyperbolic velocity) to asteroid 1685-Toro and to asteroid 3218-Delphine was carried out in order to assess the ΔV needed to perform the interplanetary mission while considering a set of mission duration and a set of date of departure from Earth escape. An initial mass of 50 metric tons at Earth escape and an on-board power of 300 kW were assumed. Several solutions were obtained assuming a transfer time between 400 to 800 days. As regards the "close to the Earth" objects class, in most of the cases studied the transfer ΔV obtained is lower than 11000 m/s.

As a general rule, it is worth noting that "close to Earth" NEOs missions with shortest duration are more expensive in terms of ΔV. In general, a mission with about have a ΔV≈10 km/s can be considered representative of "Close to Earth" NEOs missions.

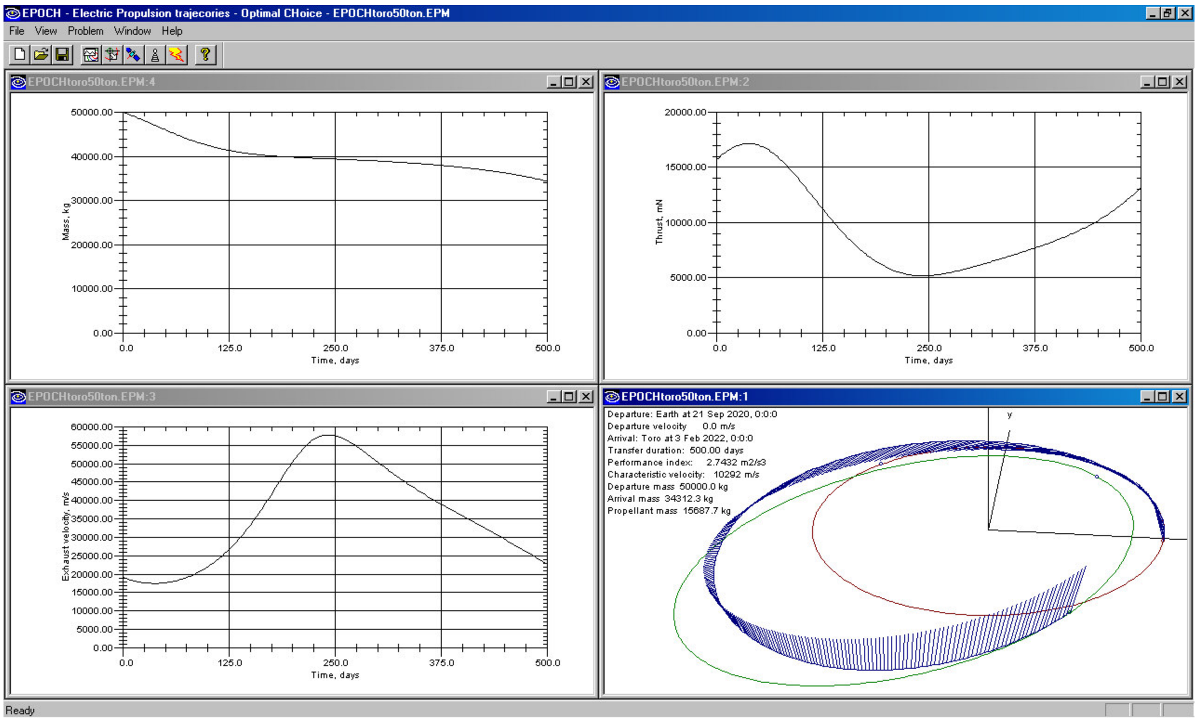


Figure 6. Heliocentric transfer of the reference NEP mission to the "close to Earth" object 1685-Toro

Concerning "farther from Earth" objects class, in most of the cases studied the transfer ΔV obtained is lower than 12000 m/s. Also in this case, as a general rule, the short duration missions are more expensive in terms of ΔV. In general, mission with ΔV≈11 km/s can be considered representative of "farther from Earth" NEOs missions.

Figure 6 shows the representative NEP mission from Earth to the selected asteroid 1685-Toro. The transfer takes about 500 days in the heliocentric phase, with the departure date fixed on September 21st 2020 (departure date not optimized). The transfer ΔV is about 10.3 km/s. The decrease of the spacecraft mass is also shown, the thrust ranges between 5 N and 17 N and the exhaust velocity (i.e. the product 'g_0Isp') ranges between 17000 m/s and 60000 m/s. This means that specific impulse ranges between 1700 s and 6000 s.

A feature in the EPOCH software also allows the user to take into account the change of the available electrical power as the distance from the Sun increases (as for the SEP case). The law selected is here is inversely proportional to the square of the distance to the Sun ($1/R^2$). Figure 7 shows a representative SEP mission from Earth to the selected "Close to the Earth" asteroid 1685-Toro. The power level at 1 AU was fixed at 300 kW as for the NEP case. The transfer obtained takes about 500 days in the heliocentric phase, with the departure date fixed on September 21st 2020. The transfer ΔV is 10031 m/s, the initial mass is 50 metric tons and the final mass is 13908 kg. The decrease of the spacecraft mass is also shown, the thrust ranges between 3 N and 20 N and the exhaust velocity (i.e. the product 'g_0Isp') ranges between 15000 m/s and 34000 m/s. This means that specific impulse ranges between 1500 s and 3450 s.

The main difference with respect to the previous NEP similar case is the drastic decrease of final mass of 13.9 metric tons only, which is mainly due to the decrease of the electrical power at the arrival.

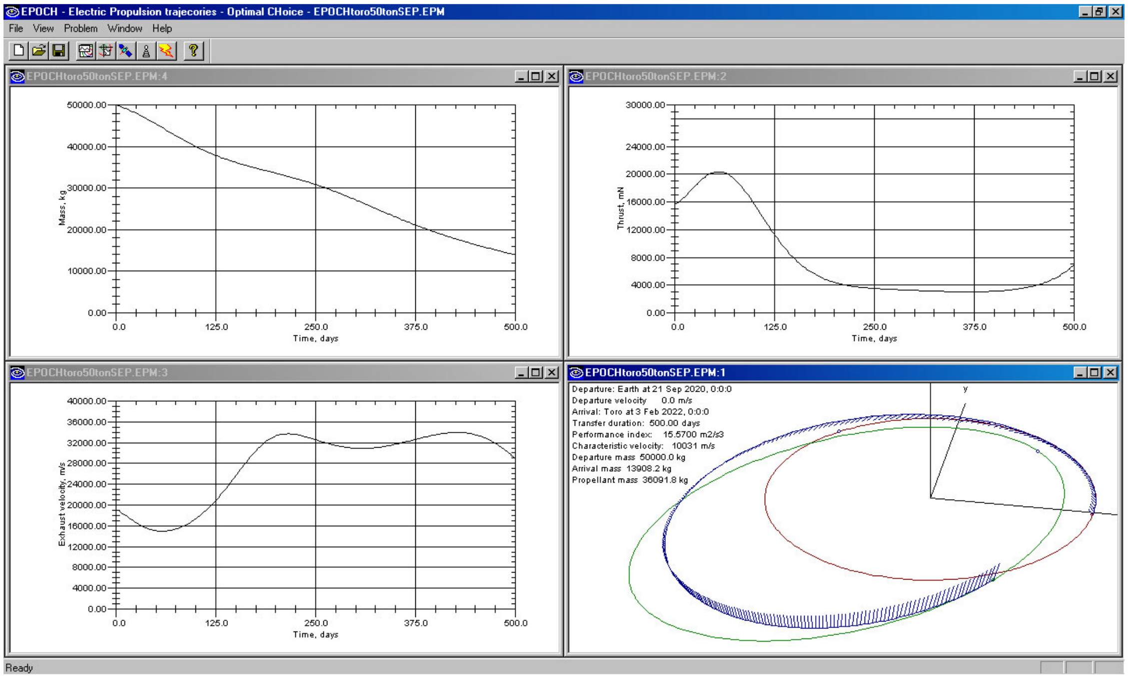


Figure 7. Heliocentric transfer of reference SEP mission to the "Close to Earth" object 1685-Toro

D. NEP Missions to Outer Planets

Main objective of this preliminary analysis was to assess the preliminary figures that can be obtained going from EML1 to the Outer planets (e.g. Jupiter and Saturn) without gravity assist manoeuvres. Therefore, the spacecraft mass budget and the mission duration were studied as they represent the system drivers. When the total mass of the power generation NEP system plus the electric propulsion subsystem was too huge to prevent a satisfactory payload mass to accomplish the selected mission, mission parameters were revised in order to obtain an acceptable payload mass within acceptable transfer times.

As direct transfers have been assumed, the optimization of the propellant mass consumption has been implemented. Besides, an initial mass of 10 metric tons from EML1 has been assumed. This includes the propellant mass, a 500 kg platform mass, the NEP system mass, PPUs, EP thrusters and the mass of tanks&structure that is proportional to the propellant mass consumed.

As regards the NEP system, at this stage of the study a specific mass of 27 kg/kW has been considered for power generation levels higher than 100 kWe that use a Brayton cycle technology. A specific mass of 35 kg/kW has been assumed for a power level between 90 and 100 kWe and 50 kg/kWe for lower power levels (obtained by means of thermoelectric/thermoionic technology).

Tables 11 and 12 summarize the different results of NEP missions to Jupiter and Saturn respectively. Each mission is supposed to start from EML1as we assumed that the vehicle is be able to be assembled and launched from a station located in EML1 where escape conditions can be easily reached. An average value of 18 kg of propellant has been computed to reach escape conditions (escape phase varies from few days to less than one month depending on the selected thrusters). However, at this stage of the study, the variation of the propellant mass and of the escape time to escape Earth's orbit can be considered negligible especially if compared to the overall transfer.

As regards Jupiter transfers, the elliptic capture orbit at arrival has got a perigee radius Rp_i=200 000 km and an apogee radius of Ra_i=12 000 000 km. A slow down manoeuvre lasting from 10 to 20 days was therefore necessary to obtain capture conditions. In this way, spacecraft arrived at Jupiter with a null infinite velocity. As periapsis velocity for the selected altitude is 35593 km/s, a ΔV=580 m/s was necessary to lower the apoapsis.

A departure date around March 1st 2018 was selected (on a few months window, the departure date has not a lot of impact on the mission duration and ΔV). As already said, an initial mass of 10 metric tons was assumed and most of the cases analyzed give positive mass margin. This guarantees mission feasibility from the mass budget point of view. Case 3.2 in Table 11 is the best margin case.

Table 11. Preliminary Results of NEP missions from EML1 to Jupiter.

Optim. criteria	Thrust (N)	Isp (s)	Electric Power @ engine (kW)	NEP device mass (kg)	Transfer time (years)	Mass After Jupiter insertion (kg)	(1) Propellant Mass (kg)	(2) Tanks+ Struct. (kg)	(3) Thrusters + PPUs + Platform + NEP Mass (kg)	(1)+(2)+(3) Minimum Initial Mass (kg)	Mass Margin (kg)
Min. time	2.44	2500	56.4	3479	2.6	1770	8230	1646	4223	14099	**-4099**
Min. cons	2.44	2500	56.4	3479	4.45	5157	4843	969	4223	10035	**-35**
Min. time	1	5000	37.8	2331	6.63	5649	4351	871	2931	8153	**1847**
Min. time	1.22	5000	46.1	2844	5.29	5760	4240	848	3466	8554	**1446**
Min. cons	1.22	5000	46.1	2844	7.21	7141	2859	572	3466	5589	**4411**
Min. time	1.50	5000	56.7	3497	5.29	4815	5185	1037	4147	10369	**-369**
Min. cons	1.50	5000	56.7	3497	6.31	6946	3054	611	4147	7812	**2188**
Min. time	1.83	5000	69.1	4266	3.68	5566	4434	887	4949	10270	**-270**
Min. cons	1.83	5000	69.1	4266	4.95	7090	2910	582	4949	8441	**1559**
Min. time	2	5000	75.5	3264	4.06	4703	5297	1059	3964	10320	**-320**
Min. cons	2	5000	75.5	3264	4.83	7108	2892	578	3964	7434	**2566**
Min. time	2	10000	130.9	4363	4.99	6726	3274	655	5064	8993	**1007**
Min. cons	2	10000	130.9	4363	4.96	8416	1584	317	5064	6965	**3035**
Min. cons	2	10000	130.9	4363	4.65	7716	2284	457	5064	7805	**2195**
Min. time	3	10000	196.4	6545	3.05	6999	3001	600	7345	10946	**-946**
Min. cons	3	10000	196.4	6545	3.29	7833	2167	433	7345	9945	**55**
Min. cons	1.75	15000	160.9	5365	4.16	8394	1606	321	6040	7967	**2033**
Min. cons	1.75	15000	160.9	5365	5.48	8841	1159	232	6040	7431	**2569**
Min. cons	2	15000	183.9	6131	3.86	8311	1689	338	6831	8858	**1142**
Min. cons	2	15000	183.9	6131	5.00	8907	1093	219	6831	8143	**1857**

Table 12. Preliminary Results of NEP missions from EML1 to Saturn.

Optim. criteria	Thrust (N)	Isp (s)	Electric Power @ engine (kW)	NEP device mass (kg)	Transfer time (years)	Mass after insertion (kg)	(1) Propellant Mass (kg)	(2) Tanks+ Struct. (kg)	(3) Thrusters + PPUs + Platform + NEP Mass (kg)	(1)+(2)+(3) Minimum Initial Mass (kg)	Mass Margin (kg)
Min. time	1.2	5000	46.1	2844	6,9	4535	5465	1093	3466	10024	**-24**
Min. time	2	10000	130.9	4363	5,4	6479	3521	704	5064	9289	**711**
Min. time	2	15000	183.9	6131	4,3	7227	2773	555	6831	10159	**-159**
Min. cons	2	15000	183.9	6131	8,5	8766	1234	247	6831	8312	**1688**
Min. cons	2	15000	183.9	6131	4,9	8227	1773	355	6831	8959	**1041**

In addition, the best results in terms of mass margin are obviously obtained assuming the minimum consumption optimization criteria; in these cases, the transfer time is longer, from few months to 2 years with respect to the minimum time transfers.

As regards Saturn transfers, the elliptic capture orbit at arrival has got a perigee radius Rp_i=100 000 km and an apogee radius of Ra_i=12 000 000 km. Results obtained are show in Table 12 where three of the five cases studied show higher positive mass margins and acceptable transfer times. In particular, Cases 2 and 5 where a high specific impulse was assumed, take 5 years to perform the interplanetary transfer.

With reference to the results obtained, it is worth noting that NEP mission durations are often shorter than the classical chemical transfer that takes about 6 years to Jupiter and 8 years to Saturn with multiple swing-bys.

E. NEP Earth-Mars transfers

The last long term scenario investigated is inherent to NEP Earth-Mars transfers aimed at crew missions orbiting the red planet to coordinate with or control robots on Mars surface or cargo missions for future infrastructures. This can represent an important milestone on an international roadmap leading to a human mission to Mars surface.

As the study was based on the assumption of NEP systems, constant power throughout the transfer was assumed. The software EPOCH was used to compute the optimized transfers and rendezvous from Earth to Mars. The results of a typical optimized EP transfer for a vehicle having an initial mass of 100 tons and powered from a 300 kW NEP system are shown in Fig. 8. Pitch and yaw thrust angles during the transfer are shown in Fig. 9.

A 400 days transfer was obtained assuming a zero excess velocity at Earth escape and at Mars rendezvous (V_{inf}=0). The overall system efficiency - including line losses, PPU efficiency and thrust efficiency – was assumed to be 50%. As shown in Fig. 8, thrust ranges between about 4 N and 45 N, while Isp ranges between about 900 s and 8000 s. A final $\Delta V \approx 8.7$ km/s was obtained for this reference transfer.

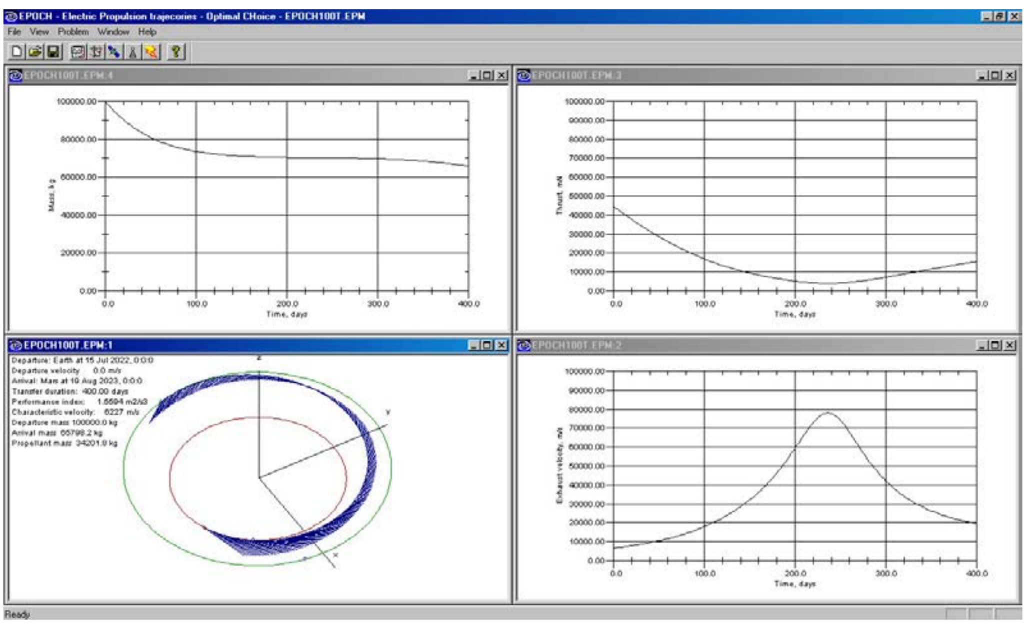


Figure 8: Typical optimized NEP Earth-Mars transfer

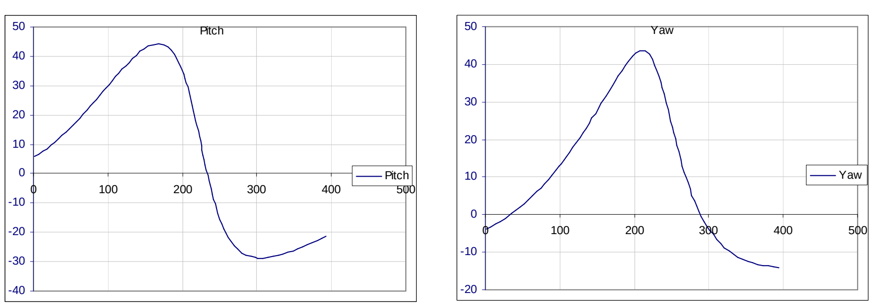


Figure 9: Pitch and yaw thrust angles of the NEP Earth-Mars transfer

VI. Conclusion

Future Electric Propulsion technologies combined with Solar Electric or Nuclear Electric high power generation can enable a wide range of mission classes. Based on the selection of specific near term and long term mission and transportation scenarios, the impact of the use EP in future missions has been evaluated and preliminary results from mission analysis show the potential advantages obtainable by using this option. Generally, larger payloads can be achieved with respect to conventional propulsion missions; furthermore, by increasing the operational power level of the propulsion system, in some cases, transfer times are comparable or even shorter with respect to missions using chemical propulsion.

The preliminary results so far obtained within the HiPER programme also define a broad range of thruster performance requirements such as thrust, specific impulse, throughput mass, firing time, lifetime, etc. that constitutes the basis for the development of future Electric Propulsion and power generation technologies to be addressed in the remaining phases of the programme.

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