FINAL REPORT: ADVANCED PROPULSION SYSTEMS & POWER PROCESSING UNIT

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ABSTRACT

The present report summarises efforts and achievements in advanced space propulsion and concentrates the content of the reports D23.1, D23.2 and D23.3 submitted in the frame of the DiPoP project. The report lists parameters and technical readiness of various advanced concepts and subsystems. In the second part of the report, mission analysis based on the collected data are documented and summarised. The results are listed in tables and depicted in respective diagrams and indicate that for rapid mass efficient transfer, solid core nuclear fission systems for thermal propulsion are the disruptive technology of choice for missions to Mars and the near Earth asteroids. As for nuclear electric concepts, various approaches appear attractive, but that there is a trade off between thrust and exhaust velocity demanding diligent optimisation and case studies. The document concludes with recommendations.

KEY WORDS: DiPoP, Advanced Space Propulsion, Disruptive Systems, Nuclear Rockets, Nuclear Power in Space, Nuclear Thermal Propulsion, Nuclear Electric Propulsion, Mission Analysis, Interplanetary Propulsion Capabilities, Overview, System Engineering

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1. Introduction

The DiPoP project seeks to identify potentially disruptive technologies with power and propulsion in focus, as these subjects pose serious challenges to the sustainable development of the solar system. For example, power is needed to operate instruments and gear. However, current space power provision mostly relies on solar power which is scarcely available in the more distant regions of the solar system. Sufficient propulsion is prerequisite to more mass efficient and rapid space transportation. Current chemical and solar electric systems still impose strict limitations on reach, on mission duration and on the delivered mass fractions. All of these issues entail systemic challenges: The small delivered mass fractions which are equivalent for excessive propellant masses force “bespoke” space craft designs with individual development and certification costs per unit and with little opportunity to serialise. This makes robotic exploration relatively expensive and rigid in the sense that missions need to be planned long in advance while their execution may take some more years. This rules short term activities out. The currently lengthy mission durations are particularly troublesome for inhabited missions. At least during the transfer, the crew is exposed to hazardous solar particle radiation, physiologic issues of low gravity and psychological ones of seclusion.

In chapter 2 of report [R 4], a rationale to enhance transportation performance by indefinite raise of mass specific power was derived, making a cause for high power propulsion systems. By doing so, it may be possible to augment both specific acceleration and exhaust velocity while current state of the art thrusters enable only one or the other enhancement. The most interesting propulsion systems from this point of view appear to be those relying on nuclear power sources.

There are two fundamental ways applying nuclear power to a Newtonian reaction engine with variable system mass: Nuclear Electric and Nuclear Thermal propulsion. The first one consists in generating electricity from the nuclear process heat and feed an electric thruster with this electricity. This is covered in [R 3]. The second one reaps the process heat directly to warm up a propellant. A general overview on many western concepts based on this principle is given in [R 4]. This report is however poor in information about Russian counterparts since there is little reliable documentation available in the languages of the DiPoP-team, i.e. English, French, German or Italian.

The basic engineering report [R 2] summarises propulsion theory, system architectures and mission analysis objectives and instruments. The present report D 23.4 synthesises these items with the data collected in [R 3] and [R 4] which are recapitulated in chapter 2. Chapter 3 documents the mission analysis. It explains the final rationale for the selected scenarios, presents the applied tools, and summarises the results before concluding in chapter 4.

2. Recapitulation: Disruptive Space Power and Propulsion

2.1. Generalities

With respect to the mission architecture, the Power Provision System and the Propulsion System are the two most important ones among the classical subsystems of a spacecraft. This is a rather intuitive statement in both cases, as (electrical) power is typically needed to operate the other subsystems and as there would hardly be any mission without propulsion. Enabling enhanced mission architectures therefore demands above all more capable power and propulsion technologies. The current state of the art power and propulsion basically confines mankind on Earth and imposes prohibitive system challenges, which are collected in figure 1 for propulsion. Only very modest missions are currently sustainably feasible. The situation can be metaphorically described as a wall; and the power and propulsion technologies enabling more audacious missions and thus breaching this wall can be coined "*disruptive*" technologies.

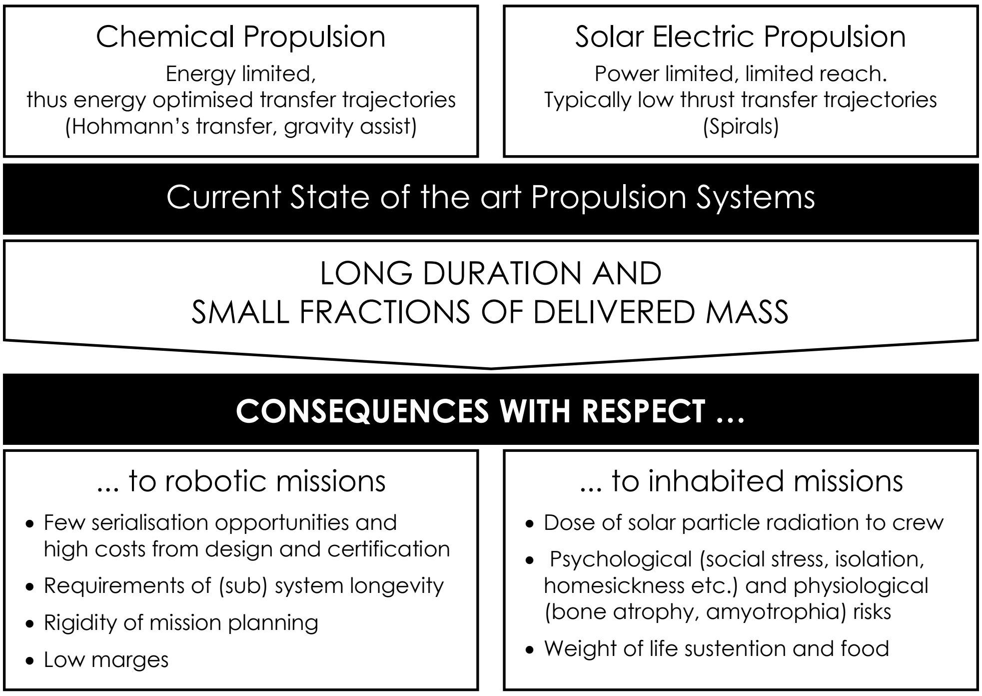


Figure 1– Current state of the art propulsion features and consequences overview.

A disruptive propulsion technology would resolve one or all of the issues related to current state of the art space propulsion, namely the lengthy voyage duration and the rather small delivered mass fraction. Recalling that space flights are characterised by their total velocity increment which can be obtained from the classical Tsiolkovsky equation

$$\Delta v = c_e \ln\left(\frac{m_0}{m_f}\right), \tag{1}$$

in which the exhaust velocity c_e relates it to the ratio of initial mass m_0 to final mass m_f, it can be stated that the delivered mass fraction can be augmented through raising c_e. However, the velocity increment is also generated by the acceleration a over the burn duration τ_B:

$$\Delta v = \int_{T_B} a \, dt . \tag{2}$$

This indicates that there is also a crucial interest in raising the acceleration, and the more so if the proposed mission architectures stipulates finite burns during important segments of the trajectory. Recapitulating from [R 4] that the mass specific jet power of a thruster is

$$\alpha = \frac{1}{2} a \, c_e , \tag{3}$$

the rationale to raise it indefinitely and thus for the development of more powerful power sources is obtained. In the same report, nuclear power sources have been identified as the most interesting ones based on a comparison of the atomic mass specific power densities of various power sources. This is valid as this value is an intrinsic upper limit. The actual data is smaller than this because systems mass specific data can vary a lot with system architecture, with the sizing and with the selection of components. It is one of the purposes of the present report to collect the respective values and relate them with other criteria of disruptiveness as defined in [R 2].

The advantage of the nuclear power system above the solar power system also depends on distance of mission target from the sun. Already beyond Mars the solar constant drops below 50 % of the value of 1360 W/m² in Earth orbit. This means the area of solar arrays should be increased to magnitude of 2 in order to generate the same power as in Earth orbit. However, the comparison of the *mass specific power α* of solar arrays and nuclear power systems enables the estimation of a transition distance in which both are approximately similar. This can serve as a selection criteria between Solar Electrical Propulsion (SEP) and NEP systems for specific mission scenario. The decreasing power of solar power generators especially from the asteroid belt and beyond with less than 10 % of the Earth solar power generation rules out SEP concepts for missions to the outer solar system. The mass specific power, and its inverse, the power specific mass, for nuclear and solar power generation are shown in table 1.

In general, there are two thinkable approaches to apply nuclear power for propulsion purposes, either as a Nuclear Thermal Propulsion, in which the process heat is converted into thrust power directly, or Nuclear Electrical Propulsion, in which this is done through an additional stage of conversion. Both concepts are outlined in the following two sections.

Power generation type	el. Power kW	Power sp. mass kg/kW	Mass sp. power W/kg	Mass sp. power (norm.)
Solar electric (typical)	*1*	*33.3*	*30*	*0.4*
Solar electric (concept)	*5000*	*13.3*	*75*	*1*
Solar thin film (concept)	*5000*	*10.0*	*100*	*1.33*
Solar electric (SLA)	*3.75*	*2.7*	*375*	*5*
SNAP-10A	0.5	*909.0*	1.1	0.015
Bouk	3.5	*263.0*	3.8	0.05
Topaz-1	6	*166.7*	6	0.08
SNAP-8	30	*133.0*	7.5	0.1
SP-100	100	*54.0*	18.5	0.25
(concept)	200	*40.0*	25	0.33
Brayton (concept)	5000	*13.3*	75	1

Table 1 – Overview on the mass specific electric power of different power sources.

2.2. Nuclear Thermal Concept Overview

The report [R 4] gives an overview on nuclear thermal propulsion systems such as depicted below in figure 2. The propulsion power is provided as heat by the system's core which is a nuclear power source. The heat is then used to warm up a working medium which is finally exhausted as a propellant. A thrust force is generated by the principle of conservation of momentum. Typically, hydrogen is selected as a working medium for its light atomic mass and hence high heat capacity making it an outstanding coolant. Its atomic mass also makes it an easily accelerated and thus excellent propellant, as explained in the annex of [R 4]. However, the cause of its unsurpassable assets is also the cause of its drawbacks. Hydrogen can easily dissipate due to the small size of its atoms and is therefore difficult to store.

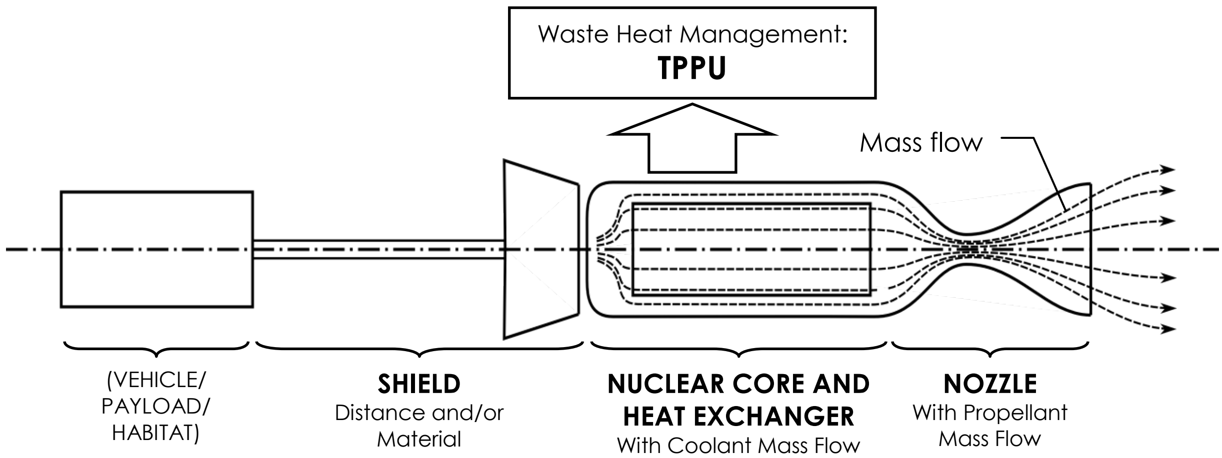


Figure 2 – Schematic presentation of Nuclear Thermal Propulsion Systems.

It is thinkable to use a core based on nuclear decay, in which case the system is a Radioisotope Heated Thermal Propulsion (RHTP) thruster, based on nuclear fission, in which case the system is a Nuclear Thermal Fission Propulsion (NTFP) system (see table 2 for reactor strategies). Both of them can be considered as state of the art principles as energy has already been drawn technically from the respective processes. In addition, there are not only many theoretical studies on NTFP carried out, but also several experiments and ground tests. It is possible to hybridise NTFP both electrically and chemically [R 4]. Due to neutronic and ionising radiation affecting both hardware and an eventual crew, a shielding system composed of radiation attenuating material or distance can be necessary.

The realisation of RHTP and NTFP can be more or less advanced. The less advanced ones rely on a solid core, i.e. the nuclear power source is present as a solid body. A little more advanced are designs stipulating a liquid core. In the case of liquid core RHTP, the radioisotope is a molten body. In the case of liquid fission propulsion, the core can also be a watery solution of a fission fuel. A further enhancement of NTFP consists in allowing for gaseous cores stabilised by coolant pressure, toroidal vertex flow and finally quartz containment. The purpose of shifting from solid to liquid and from liquid to gaseous cores is to achieve higher exhaust velocities as they are limited by the core temperature. Liquid and gaseous cores enable higher temperatures and thus higher exhaust velocities. More exotic approaches to propulsion are offered by the Nuclear Salt Water Rocket and by devices such as the Orion concept, which appear to be too audacious [R 4].

Type	Description	Concepts	Issues	Data
Heterogeneous thermal reactor	Uses relatively lowly enriched fuel. Needs moderators to obtain thermal neutrons for fission. Separate moderator body.	Many	Corrosion, life time	Theory
Homogeneous thermal reactor	Similar to heterogeneous thermal reactors. Fuel and moderator in a shared body.	Some	Heat loads, coolant fit	Speculative [R 5]
Fast reactors	Uses highly enriched fuel. No moderator needed.	e.g. SNAP-10A	Heat loads	Resilient

Table 2 – Types of reactor strategies for solid body NTFP

Even higher exhaust velocities can be obtained from more advanced propulsion systems such as fusion propulsion which often entails magnetic confinement and consequently magnetic nozzles but also from anti-matter propulsion. Never the less, this final option of propulsion is far from any economic and technical feasibility as the necessary antiprotons are generated in heavy collider experiments with respective costs and low half life [R 5]. The outlook for fusion propulsion is better, as a proof of fusion power can likely occur during the ITER experiments [R 6].

Report [R 4] proposes sets of criteria for RHTP and NTFP, as summarised in the following two sections. In addition to that, the present document amends criteria relevant for fusion propulsion. Section 2.2.4 concentrates the information found during the literature survey conducted in the frame of WP 23 of the DiPoP project [R 2, 3, 4].

2.2.1. Radioisotope Heated Thermal Propulsion (RHTP) criteria

From system oriented considerations as developed in [R 4], it is possible to derive some technical radioisotope selection criteria for RHTP systems which are similar to the criteria for Radioisotope Thermal Generators (RTG). Note that criteria for RHTP working media are – as in general – besides low chemical noxiousness high heat capacity and easy accelerability; thus small molecular masses.

The RHTP criteria group as follows and lead to the necessity of tradeoffs:

- **Propulsion criteria**
  - Low thrust variation, thus long half life
  - High energy yield, thus short half life or large *E*
  - Efficient power transfer to coolant/propellant:
    high cross section in medium, short thermalisation reach, thus alpha radiation
  - Mass specific thrust parameters: light weight of system, thus alpha radiation
- **Chemical safety criteria**
  - Crew/technician safety: substance non poisonous
  - Material safety: substance non corrosive
  - Space craft safety: decay products chemically safe
- **Physical safety criteria**
  - Crew/technician safety: radioactivity limited or well shielded
  - Material safety: limited yield of power, must withstand lack of cooling
  - Space craft safety: decay products lowly radioactive
- **Economic considerations / synergies**
  - ... use as an RHU (radioisotope heating unit) after propulsion use
  - Availability of radioisotope and
  - Cost of processing and handling

2.2.2. Nuclear Thermal Fission Propulsion (NTFP) criteria

Primordial technical criteria for NTFP systems can be established analogously:

- **Propulsion criteria**
  - High energy yield of the core and efficient power transfer to working medium
  - Avoidance of excess propellant saturation with heavy fission products or educts
    - Thus: low fuel vaporisation or erosion rates
  - Mass specific thrust parameters: light weight of system and/or compact volume
- **Chemical safety criteria**
  - Crew/technician safety: substances non poisonous
  - Material safety: substance non corrosive or susceptible to corrosion
  - Space craft safety: fission educts and products chemically safe
- **Physical safety criteria**
  - Crew/technician safety: radioactivity limited or well shielded
  - Fission safety: avoidance of prompt criticality, only controlled criticality
  - Material safety: limited yield of power, must withstand lack of cooling
  - Space craft safety: core ejecta lowly radioactive or injected in escape trajectory
- **Economic considerations / synergies**
  - ... use as a power plant after propulsion use (bimodal concept)
  - Availability of fissile material and
  - Low rates of fuel loss
  - Cost of processing and handling
  - Longevity: Low lifetime maintenance efforts; sustainability

2.2.3. Magnetically Confined Fusion Propulsion (MCFP) criteria

As for NTFP and RHTP, there are some primordial criteria for fusion propulsion systems. They can be collected from the available literature:

- **Propulsion criteria**
  - High thrust and high exhaust velocity
  - Efficient power transfer to working medium
- **Fusion reaction criteria**
  - Avoidance of plasma poisoning with the working medium or first wall material
  - Large cross sections
  - Availability of fusion fuel, or in-situ availability
- **Radiological safety criteria**
  - Low neutronic reactions preferable for
    - Avoidance of activation
    - Avoidance of neutron and secondary radiation
    - Thus low shield masses
  - Avoidance of tritium
- **Reactor criteria**
  - Low masses and compact structures
  - Low maintenance, low degradation
  - Longevity

2.2.4. Tables

The Technological Readiness Levels (TRL) of NTP relevant subsystems are found in table 3. The data of disruptive nuclear rockets collected for [R 4] are concentrated in table 4, while the estimations of the more advanced systems' characteristics are noted in table 5. Among these, there are the NSWR as an example of a system relying on prompt fission and a few representative fusion propulsion systems.

Item	TRL	Country	Ref.	Comments
Subsystem	6 - 7	USA	[R 7, 8]	*NERVA,*
	5	Europe	[R 9]	*10 MW reactor for Europa 3, MAPS (300 MW), state during the 1980s*
Reactor	6	Russia		*RD-0410 (CADB)*
	6 - 7	USA	[R 7, 8]	*NERVA, Nuclear furnace*
	5	Europe	[R 9]	*10 MW reactor for Europa 3, MAPS (300 MW), state during the 1980s*
Reactor fuel (pilot plant production)	7	USA	[R 7, 8]	*UN, BISO and TRISO (UO2)*
	7	Europe	[R 9]	*BISO and TRISO particles (70's)*
Hydrogen Turbo Pump	7	Europe	[R 9]	*Expander cycle, similar to VINCI LH2 TP*
C-C nozzle extension)	9	Europe	[R 9]	*RL10B2 extension*
LH2 tank	9	Europe	[R 9]	*Ariane 5*
Long term LH2 storage (ZBO)	5 - 6	USA	[R 7, 8]	*NASA tests*
	4	Europe	[R 9]	*On-going FP7 ISP programme*

Table 3 – TRL levels of NTP relevant subsystems.

System	Type	c_e / m/s	F / N	P_{th} / kW	P_{jet} / kW	m / kg	Ref.
POODLE	RHTP	< 6940	< 1.1	5.6 - 4.8	3.78	23.6	[R 10]
NRX-A6	NTFP (solid)	8310	216000	1.17e6	8.97e5	N. A. (ed.guess: 9000)	[R 11, 12]
NERVA-1	NTFP (solid)	8093	334000	1.57e6	1.35e6	9000	[R 11, 12]
PHOEBUS-2A	NTFP (solid)	7900	922600	4.08e6	3.64e6	N. A. (ed.guess: 9000)	[R 11, 12]
NERVA-2	NTFP (solid)	8110	867000	N.A.	3.52e6	34000	[R 11, 12]
Timberwind 45	NTFP (solid)	9830	441000	N.A.	2.17e6	1500	[R 13]
Timberwind 75	NTFP (solid)	9830	736000	N.A.	3.62e6	2500	[R 13]
Timberwind 230	NTFP (solid)	9830	2452000	N.A.	1.21e7	8300	[R 13]
RD-0410	NTFP (solid)	8920	35500	1.96e5	1.58e5	2000	[R 14, 15]
IRGIT	NTFP (solid)	N.A.	N.A.	N.A.	N.A.	N.A.	N.A.
LCR 1	NTFP (liquid)	15000	680	N.A.	5.10e3	N.A.	[R 16]
LCR 2	NTFP (liquid)	13500	920	N.A.	6.21e3	N.A.	[R 16]
LCR 3	NTFP (liquid)	12000	1090	N.A.	6.54e3	N.A.	[R 16]
GCR 1	NTFP (gas.)	15000 – 25000	1000000	N.A.	7.5e6 – 1.3e7	45000	[R 16]
GCR 2	NTFP (gas.)	30000 – 70000	67000	N.A.	1e6 – 2.3e6	60000 - 200000	[R 16]
NLBR	NTFP (gas.)	20000	400000	N.A.	4e6	32000	[R 17]

Table 4 – Propulsion parameters of RHTP and NTFP thrusters (ed. guess: educated guess)

System	Type	c_e / m/s	F / N	P_{th} / kW	P_{jet} / kW	m / kg	Ref.
NSWR	Prompt	66000	12930000	(4.27e11)	4.27e11	N. A. (ed.guess: 500)	[R 18]
D^3He WGD 10	MCFP	180000	20000	1.90e6	1.80e6	100000	[R 19]
^{11}Bp WGD 10	MCFP	28000	13000000	1.91e8	1.82e8	3200000	[R 19]
GDM	Gasdynamic	1136000	50000	5.50e7	2.84e7	400000	[R 20]
Discovery 2	Spher. torus	350000	28000	4.90e6	4.90e6	360000	[R 21]

Table 5 – Propulsion parameters of more advanced NTP thrusters

2.3. Nuclear Electric Concept Overview

As mentioned above, another option would consist in using a nuclear power source to generate electricity operating an electric propulsion (EP) system. The interest of EP is the option to use electrical acceleration mechanisms that can achieve an even higher exhaust velocity than thermal propulsion concepts without necessarily requiring higher hardware temperatures. Many of these concepts rely only partially on the thermal heating of the propellant if at all, and an important part of the kinetic energy in the exhausted propellant has been fed in electrically, for example using electrostatic acceleration as in ion thrusters or by electromagnetic acceleration as in Magneto Plasma Dynamic (MPD) thrusters. A plethora of concepts and interesting hybridisation approaches have been suggested and studied. An overview on EP principles is summarised in [R 2, 3, 22] and described in detail in [R 23].

While one of the features of NEP is that the electricity can be used to operate various electrical thrusters, it is also a feature that various nuclear power sources can be used to generate the necessary power. Hence, there is a relatively large liberty in selection. The power source can be selected independently and so can the thruster. While this may seem to enable arbitrary compositions only needing to respect the high-c_e-high-a-rationale, there has been done considerable work to identify optimum regimes with respect to the payload fraction [R 22] and to the voyage duration [R 23]. A generic setup of a space craft with fission based NEP is drafted in figure 3. Observe that the fission core is now at the bow of the craft, as it is not necessary to have it situated between the payload and the electrical thruster. Above that, not being exposed to

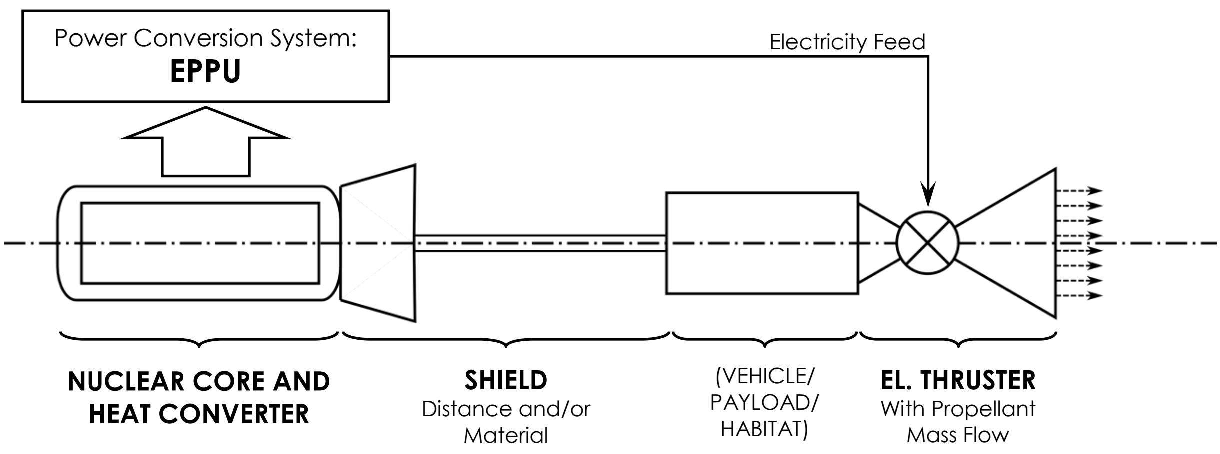


Figure 3 – Schematic presentation of generic fission based Nuclear Electric Propulsion Systems.

eventual neutron radiation is beneficial for the longevity of the thruster. This architecture is not necessary for low radiation nuclear cores such as RTG which generally feature alpha radiation, or low to aneutronic fusion reactors.

2.3.1. Power Conversion System Overview

From all of the above, it can be concluded that while electricity generation from the waste heat of an NTP otherwise radiated into space like the core's residual or eventual decay heat is – albeit beneficial to the overall system efficiency – an optional subsystem, electricity generation from the nuclear process heat is an absolute necessity to NEP concepts. However, the power conversion system (PCS) will feature many similarities among both cases.

The first similarity is found in the approach: The currently available radio isotope and fission technology provides a so called process heat which is yielded as heat radiation or through conduction. To convert this heat into electricity, the same physical principles among which the *Seebeck*, the *Peltier* and the *Thompson* effect, but also the *Alkali Metal Thermo Electrical Conversion* (AM-TEC), the *Thermo Photovoltaic Conversion* (TPC), the *Thermionic Conversion* apply. In systems using turbo pumps, electrical power can be obtained from a small *Alternator* on the turbo pump shaft. This alternative is the minimal realisation. A third alternative would consist in using *thermo dynamic cycle processes* using the heat to drive a mechanical machine whose motion is converted into electricity with a generator. Among these, there are the *Stirling* and the *Brayton Cycle* which appear to be the most promising options for large scale electricity provision around 100 kW_e. An inconvenience is the reliance on moving parts which is challenging in the space environment and entails longevity requirements for robotic space craft. In the case of inhabited space craft, it is thinkable to have the crew conducting maintenance. However, this can encompass other challenges such as extra vehicular activities or radiation dose monitoring.

These systems are also applicable to fusion reactors, especially for the expected neutronic first generation reactors fed with Deuterium-Tritium fuelling. If low to aneutronic reactors are available, *Magneto Hydro Dynamic* (MHD) approaches become an additional attractive alternative because then the fusion power is primarily distributed on charged particles able to drive MHD generators. MHD generators are also ranked as thermo-electric converters.

The generators driven by thermodynamic cycles intrinsically provide *alternating current* (AC) at an output voltage tailored to the needs of the load while *direct current* (DC) is obtained from thermo-electric converters, typically at 70 V. Thus, depending on the necessary condition of the electricity more or less heavy secondary processing units are required. Some details can be found in [R 2] and [R 3]. It is possible to envisage a use of both approaches. In this case, the DC output can be used to compensate the losses of the *Power Distribution and Conditioning Unit* (PDCU).

In all cases, substantial energy storage shall be provided for initial operation and starting transient and reactor shut down. This amounts to several kilo watts over a period of a couple of minutes. Typical values in the range of 1 to 10 kWh are comparable to the order of magnitude of energy storage aboard telecommunication satellites.

2.3.2. Electric Power Processing Unit (EPPU)

While in the prior section on NEP it was stated that combinations of Nuclear Electric Generators and electrical thrusters were to be laid out with respect to optimisation results from [R 22] and [R 23], system and mission analysis reveals that no single electric propulsion technology is likely to be compatible with any arbitrary NEP generator. Higher exhaust velocities tend to be advantageous for the missions to the outer solar system where nuclear power appears as a unique enabling technology due to the faint solar power beyond the Asteroid Main Belt (AMB). However lower exhaust velocities are probably more suitable for applications closer to Earth. It is likely that once the technology is available, these could become more numerous. Also whereas the lift capacity of Ariane 5 is easily compatible with power levels in the 200 kWe class, it also suits 25kW Gridded Ion Thrusters (GIT), Hall Effect Thrusters (HET) and High Efficiency Multistage Plasma Thruster Thrusters (HEMPT). A 2 MWe NEP generator may operate efficiently with 100 kWe Magneto Plasma Dynamic Thrusters (MPD). Equally a 30 kWe NEP generator, for example, could operate with existing GITs and HETs or even simple arcjet systems.

In principle it is possible to design the NEP generator to be compatible with a range of EP technologies and the key to this is the interface which is described as the PCDU. This depends on many factors. It must deliver power of the required quality to the EP systems and protect the NEP generator from sudden un-programmed load changes. It must have access to the energy for in-orbit commissioning and cold re-starts as well as controlled power up, power down and safe standby or low power operations. As some thrusters may require high frequency AC causing severe electrical loads and electromagnetic radiation noise disturbing the space craft's electronics, the harness mass is likely to be a significant contribution to the propulsion system mass. Both this mass and the harness efficiency are important criteria of an electrical propulsion concept for the laying of the space craft. Further, the PDCU is influenced by the selection of AC or DC power distribution, NEP turbo-alternator and EP PPU design and the architecture determining the distance between the NEP generator and the EP systems. The requirements per electrical propulsion concept are detailed in the next section.

2.3.3. Electric Propulsion Input Requirements

Gridded Ion Thrusters

For nuclear electric applications GITs may be considered in the 5 kW / c_e = 30000-50000 m/s and the 20-25 kW / c_e = 100000-150000 m/s range. For the former, the PPU generates high voltage in the 1.6-1.8 kV and low voltage in the 30A / 35V range as for example Kaufman's thruster or an RF generator of similar power level (RIT). For higher power and higher exhaust velocity the high voltage increases to 6-8 kV. Normally, a power supply input stability of ~ ± 3% will be specified. However the processing to generate high and low voltages will normally take care of the power supply variation. In principle, power input can be AC or DC and be anywhere in a range between hundreds of Volts and several kilo Volt. Since the beam power is by far the largest part of the power, it is advisable to use a direct rectified source for this function eliminating heavy transformers. On the other hand, the weight penalty is acceptable for the discharge voltage and auxiliary functions.

High Efficiency Multistage Plasma Thruster (HEMPT)

HEMPTs' discharge characteristic is very similar to the HETs' one, but the discharge voltage is higher, typically 1 kV instead of 400 V. Up to now, operation has been demonstrate unto a power level of 5 kW. For the purposes of comparison in general terms, the HEMPT will be considered a part of the HET family.

Hall Effect Thruster (HET)

For nuclear electric applications HETs may be operated around 5kW / c_e = 20000 – 30000 m/s and 20 - 25 kW / c_e = 25000 – 35000 m/s. The latter is valid for xenon. Works are under way to allow operation with krypton and argon leading to higher exhaust velocities for a given discharge voltage. The input circuitry generates 200 V to 400 V for the lower power range and 400 V to 800 V for the higher. The engine is tolerant of operation with an unregulated power supply but in practice a limit of about ± 3% is advisable to limit thrust fluctuation. The discharge characteristic being "vertical", a voltage source is naturally stable with the discharge. Ideally the input power voltage will be DC and match that of the main power supplied to the thruster, i.e. 200 – 800 V, depending on the point selected for the thruster's operation.

As in the case of ion propulsion, it is advisable to set the alternator voltage to suit the discharge voltage and use insulating transformers for the other functions, such as magnet supply if any, cathodes, valves.

Magneto-Plasma-Dynamic (MPD) thrusters

For nuclear electric applications MPDs may be in the range of 100 kW to 1000 kW. The main power supply to the thruster is at about 50 V to 150 V respectively. The need for power input stability is not fully defined at this stage, however, an assumption common to that for GITs and HETs seems prudent. In principle, the PPU will be simpler if the input voltage is DC and matches the main 50 V or 150 V supplies to the thruster. A regulator may be required in this case as the MPD discharge characteristic may not be compatible with the source characteristic. This regulator may for example perform a current limitation with an efficiency as high as 97 %. Bearing in mind that the discharge current range may span 500 – 2000 A depending on the applied magnetic field for the 100 kW case, it could be necessary to split the regulator into several parallel units according to the available hardware.

Applied-Field type MPD will need a different power supply due to needed power distribution for the electromagnet which needs to generate magnetic flux densities between 0.1 and 0.6 Tesla. Supra conduction can be considered for a reduction of Ohmic losses in the electromagnet but will require additional cooling and additional thermal insulation.

2.3.4. TRL levels of NEP

The following tables 6 and 7 provide the TRL for NEP relevant technologies such as reactor, fuelling, converters, radiators, power processing units, electric thrusters and propellant storage. The TRL is noted according to the state of national programs in the USA, Europe and Russia.

Item	TRL	Country	Ref.	Comments
HET	9	EU, USA	[R 29, 30]	*P ≤1.5 kW*
	9	USA	[R 29, 30]	*P ≤ 5 kW*
	6	Europe	[R 29, 30]	*P ≤ 5 kW*
	5	USA	[R 29, 30]	*P ≤ 70 kW*
	5	Europe	[R 29, 30]	*P ≤ 21 kW*
GIE	8 - 9	EU, USA	[R 29, 30]	*P ≤6 kW*
	5	USA	[R 29, 30]	*P ≤ 20 kW*
Li AF MPD	5	USA, Ru.		*P ≤ 250 kW*
Ar AF MPD	4 - 5	Europe	[R 29, 30]	*P ≤ 150 kW, IRS (D , Alta (I)*
VASIMR	5	USA	[R 29, 30]	*P ≤ 200 kW*

Table 6 – High power EP for NEP: TRL Levels

Item	TRL	Country	Ref.	Comments
Subsystem	5 - 6	USA	[R 7, 8]	*Prometheus,*
	5	Europe	[R 25 -28]	*ERATO, NEP / SEP ESA study*
Reactor	9	Russia	[R 24]	*TOPAZ, BUK*
	6	USA	[R 7, 8]	*SP 100, SAFE*
	5	Europe	[R 25 -28]	*OPUS*
Reactor fuel	9	Russia	[R 24]	*TOPAZ 93 % UO2 fuel pills*
	7	Europe	[R 25 -28]	*BISO and TRISO particles (70's)*
Brayton: 30-100 kW APU (airbreathing)	9	EU, USA	[R 7, 8]	*Single shft turbo Brayton*
He Xe turbo Brayton	6	USA	[R 7, 8]	*Ariane 5*
He Xe turbo Brayton Rankine / Hirn conversion	4	Europe	[R 25 -28]	*NASA tests*
	3	Europe	[R 25 -28]	*On-going FP7 ISP programme*
Stirling conversion	4	Europe	[R 25 -28]	*50 000 h tests*
Thermionic conversion	6 (70's)	Europe	[R 25 -28]	*Detailed design 100 kW*
Thermocouple conversion	6	Europe	[R 25 -28]	*Ground applications*
AMTEC	4	USA	[R 7, 8]	*Lifetime issues*
PDCU	9	Europe	[R 25 -28]	*Up to 25 kW, 50 or 100 V DC*
PPU	9	USA	[R 7, 8]	*5 kW (GIE, HET), 30kW (arcjet)*
	9	Europe	[R 25 -28]	*2.5 kW (GIE, HET)*
	8	Europe	[R 25 -28]	*5 kW (GIE, HET)*

Table 7 – TRL levels of NEP relevant subsystems.

2.3.5. Overview of Electrical Propulsion

Table 8 contains an overview of relevant electrical propulsion systems surveyed in the frame of DiPoP. These thrusters are considered for the mission analysis in the following sections.

		GIT		**HET**		**ARCjet**		**AF-MPDT**	
Data:	*Unit:*	*NEXT*	*NEXIS*	*173Mv1*	*457Mv2*	*GSC 141*	*HiPARC*	*X16*	*MAI*
Electric power	kW	6.86	20.4	7.56	25.2	30	100	11.6	118.86
Thrust efficiency	%	70	75	61	63	54	28	38	43
Thrust power	kW	4.86	15.46	4.61	16.38	16.2	28	4.408	52.44
Thrust	mN	237	446	391	1280	3350	2900	250	3140
Mass flow rate	mg/s	5.8	6.5	16.5	50	50	150	7	94
Exhaust velocity	km/s	40.910	69.160	23.690	24.720	9.908	20.000	35.714	33.400
Specific impulse	s	4170	7050	2415	2520	1010	2038	3640	3405
Specific thrust	mN/kW	34.548	21.863	51.719	50.793	111.666	29	21.638	26.417
Propellant	-	Xenon	Xenon	Xenon	Xenon	Hydrogen	Hydrogen	Argon	Lithium
Reference	-	AIAA 2012-4023	IEPC 2005-281	Hofer R. Dissertation	IEPC 2011-339	AIAA Journal Vol. 3 No. 1	IEPC 97-007	AIAA 75-417	IEPC 97-117

Table 8 – Overview of selected electric thrusters.

3. Mission analysis

In the frame of the DiPoP project, possible mission targets have been proposed in [R 2]. During the conduction of respective analysis, some of the missions' destinations were changed for the sake of offering more representative cases. This is documented in this section as well as changes in the selection of the tools used to perform the mission analysis. NEP missions were analysed based on a numerical approach while NTP missions were assessed using an analytic high-thrust approximation verified in academic studies conducted by one of the DiPoP consortium's members. In the final section, the relevant results are concentrated.

3.1. Proposed Missions

The purpose of the mission analysis conducted in the frame of DiPoP consists in assessing the disruptiveness of various nuclear electrical and nuclear thermal propulsion systems. This assessment follows a two sided rationale: On one hand, the systems' capability to perform given enhanced mission goals has been evaluated in simplified cases representing various classes of NEP – listed in table 8 – and NTP – as listed above in table 4 and 5. On the other hand, the potential impact of the surveyed disruptive technologies has been derived from the selected missions which had to be representative for certain classes of mission in turn. The missions were selected with respect to three criteria:

- *Relevance:* The mission consists in a significant activity in the domains of exploration, science or commerce and entails a respective impact. It serves near to medium term space exploration plans and human needs.
- *Ambition:* The mission is more exigent than any prior mission. It makes therefore an exemplary application for disruptive approaches.
- *Representation:* The mission is representative for a class of similar missions sharing essential characteristics.

Obviously, the latter is a rather vague criterion, as the characteristics can be defined with respect to various aspects such as the distance and the type of the target and many others. In the scope of this document, missions are defined by the type of payload (inhabited or robotic), distance, transfer time, and masses.

In a first instance, the missions described in table 9 were selected [R 2].

Mission	NEOs (Human)	Mars Science	Mars Human	Mars Rapid	Jupiter	Neptune (Triton)
Distance / AU	0.91	1.52	1.52	1.52	5.2	30.1
Δv / km/s	4 (114 d)	5.6 (365 d)	6.7 (123 d)	9.1 (93 d)	16.7	>24.4
Initial mass / t	7	7	140	30	7	7
Thrust / N	2.5	(0.5-3)	100	10.7	(0.5-1)	(0.2-0.5)
Time / d	114	730	123	83	< 1000	<3650
Reference	Zimmer	-	Schmidt	Schmidt	-	-

Table 9 – Overview of mission requirements for the chosen missions.

From this table it can be seen that in the mid term, the Jupiter mission taking up to 1000 days but at least the trip time to Mars is most likely too lengthy to be inhabited, while not exigent enough to be served by a robotic probed using disruptive propulsion. In contrast, a robotic mission to Neptune's moon Triton appeared to be sufficiently exigent and in the same time a more representative mission and was maintained. To summarise, the following solar system objects have been considered for analysis of mission scenario:

- *NEO-Asteroid:* Most of the NEOs can be reached with chemical propulsion; NTP concepts may however allow a more rapid and mass-efficient transfer. Eventually this may be more sustainable due to the lower propellant consumption caused by the higher exhaust velocity. Because of the relatively low thrust level the NEP concept is less suitable compared to NTP.
- *Mars:* For the further investigation of the feasibility of NEP concepts, Mars can be an interesting destination. This is even more true for NTP approaches as their high specific acceleration might enable rather straight transfers beneficial to inhabited purposes.
- *Triton (Neptune):* Due to the far distance to Neptune the transfer time for any mission is very high possibly requiring a more detailed gravity assist manoeuvre. While this is still envisageable for robotic probes, no significant enhancement compared to Cassini or Galileo can be expected. This would however be different for NTP missions.

Other than that, a frame was defined to govern the envisaged mission analysis. First, launch options are selected. Current heavy launch systems available today are *Ariane 5*, *Proton-M* and *Delta 4-H* with transport capability of around 21 t to *Low Earth Orbit* (LEO), 9 t to *Geo Transfer Orbit* (GTO) and around 7 t to interplanetary transfer orbit [R 31, 32]. *Energia* could allow for up to 100 t into LEO, but is not available these days. This means interplanetary spacecrafts with payload masses above 7 t need to be launched piecewise and assembled in orbit before being cast off. The whole mission spacecraft should consist of predefined modules of less than 20 t with low risk, standardized docking ports and sub-elements.

In the frame of the present mission analysis, for each mission only the transfer from Earth's orbit around sun to the target's heliocentric orbit are considered. The transfer from earth to an interplanetary orbit is considered for neither NEP nor NTP and is assumed to be executed by external kick stages. However, chemical and nuclear thermal kick stages should be assessed in upcoming projects.

As far as masses are concerned in the mission analysis, payloads of 100 t have been considered for low thrust NEP scenarios. The respective space craft was assumed to contain a further mass margin of 50 or 100 t. In the case of high thrust systems, typically NTP, a dual approach was taken. On one hand, the travel duration for a given payload mass encompassing the space craft's subsystems with the exception of the separately considered propulsion system have been determined. On the other hand, it was estimated how much mass could be delivered at the destination in a given travel duration – which is probably more indicative for a propulsion systems capabilities.

3.2. Mission Analysis Tools

The selection of software (SW) tools that can be utilized for the mission analysis is summarized and revised in this section.

A general assessment can only take universal tool functions into consideration as is the case in the study presented here. An important issue for tool selection is the reliability of the results. The complexity of the trajectory extraction process implies that a professional solution can only be obtained by investing substantial funding into the method. This can be either for in-house development of application specific customer-furnished-software (CFS) involving extensive testing and evaluation time or for procurement and integration of a customer-off-the-shelf (COTS) product. A good commercial indicator for a professional tool is market penetration. However, since the financial obstacle of introducing a professional tool to the market is important, they are rather scarcely spread in the academic sector. On the other hand, the quality of a self-built tool is often driven by a more in-depth treatment of a desired single aspect of trajectory creation but has in no case yet reached the complexity and completeness of a commercial product.

The overview in table 10 already shows the selection of tools that were presented in detail. But information about the tools origin, availability, compatibility with other software and databases and a herein defined readiness level are already summed up at this point to display again the different approaches, maturities and considerable factors of a tool assortment. If a tool's source is closed or open depends on the accessibility. STK or GESOP & ASTOS, for example, are commercial software that has to be purchased although an academic software release is obtainable for universities. These are marked with closed source. Open source software as e.g. STA or GMAT is accessible for anybody and doesn't have to be purchased.

The practical experiences made in the frame of DiPoP showed, that the ownership of licences is indeed the most important aspect of mission analysis tools. Many fully evolved mission analysis tools which appeared to be suitable for the scope and purpose of DiPoP cannot be used for the project as the licences – especially for a commercial venture such as DiPoP – are excessive amounting several thousand Euros. A licence for STK-Astrogator was initially assumed to be

Tool	Origin	Source	Prog. Language	Database	Rendering Engine	GUI	Rdns. Level
STK	AGI	closed	In-house	NORAD	In-house	In-house	I
STO	EADS Astrium	closed	MatLab	HORIZONS (NASA), SPICE (NASA)	MatLab	MatLab	II
STA	ESA	open	C++	SPICE (NASA), HIPPARCOS (ESA)	Celestia, OpenGL	Qt	II
GESOP/ ASTOS	Astos Solutions	closed	In-house	In-house	In-house	In-house	II
Orbiter	Martin Schweiger	closed	n/a	n/a	n/a	n/a	III
GMAT	NASA	open	C++	SPICE (NASA)	OpenGL	wxWidgets	II

Readiness Level (Rdns. Level) Definition	
I	*User level interaction only with full tool function coverage and company support, stand-alone solutions*
II	*Ready to use solution expandable via interfaces, editors and user generated packages*
III	*Fully customer designed software tool, may include freeware code and COTS-interfaces*

Table 10 – Analysis Tool Overview.

owned by the IRS as a member of the DiPoP consortium but it was ultimately found out to be limited to academic usage making it unavailable for the project. Other low cost tools proved not to be suitable for the analysis proposed for this report making a case for the in-house development mentioned above. The used frame of the development was Matlab of which licences are owned by the University of Stuttgart as a member of the DiPoP consortium. Two approaches were implemented and preliminarily tested in an academic evaluation using STK [R 33, 34]. While the first one is suitable for high thrust systems providing for enough thrust to allow the neglect of the Solar gravitational acceleration, such as typical NTP, the other one is a classical numerical tool approximating the analytically insoluble powered motion in a central force field. Both are detailed in the following.

3.2.1. High thrust approximation for NTP

With the data of NTP retrieved in the reports [R 3] and concentrated in table 4 and table 5, mission analysis was conducted relying on an in-house algebraic mission analysis tool. The tool is documented in an internal report at the IRS [R 33] and introduces approximating equations for a field free assumption as proposed by Williams in [R 35]. The internal report [R 33] which is of an educational nature also contains a chapter on the verification of the equations for some relevant cases with STK.

A majority of NTP systems appear to fulfil the high-α-rationale mentioned in section 2.1 as they have both a relatively high thrust to weight ratio and relatively high exhaust velocities. While the latter indicates an improvement in deliverable masses, the former insinuates a more rapid acceleration. This can enable disruptive transfer architectures consisting in rapid trajectories which are opposed to Hohmann's lengthy transfers enforced by current state-of-the art propulsion concepts[1]. An exemplary Hohmann's transfer is shown on the left of figure 4 while the novel transfer trajectory is *drafted* on its left. It can be characterised as a *continuous-burn-rendez-vous* in which the thruster is accelerating towards the destination for a finite duration – as opposed to the impulsive burns enacted in Hohmann's transfers – before it is used to decelerate

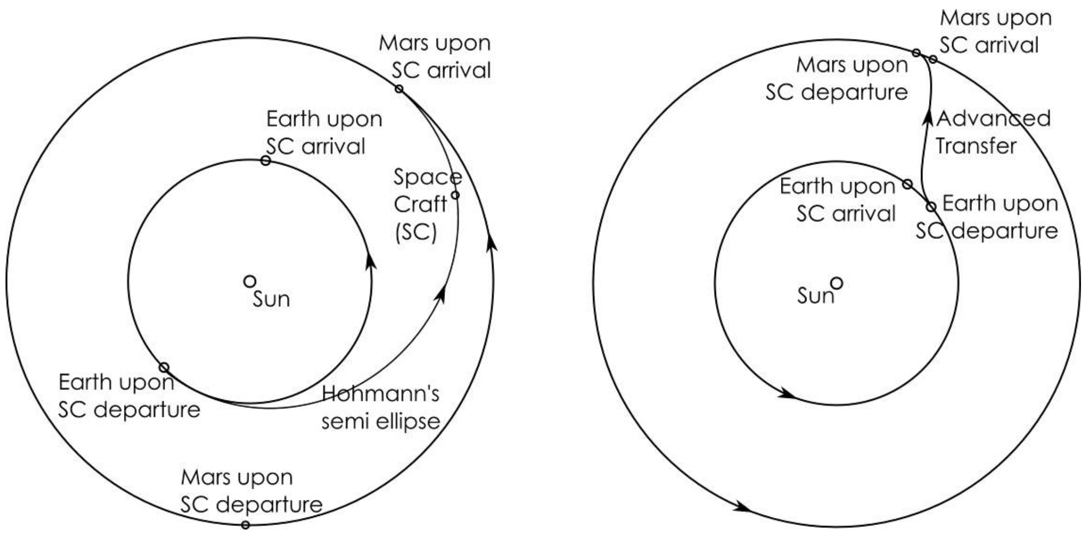


Figure 4 – Schematic of Hohmann's transfer (left) and the continuous-burn-rendez-vous (right).

[1] Spiral transfers are an established alternative for low thrust systems of very high exhaust velocities but may be even lengthier than Hohmann's transfers. Another alternative consists in bi-elliptical transfers which are also lengthy.

in the opposite direction. Another difference to the classical transfer approach consists in the complete absence of any thrust free coasting leg which is taking the whole duration of the classical approach. The last distinction consists in the neglect of the solar gravity. It is an essential factor for Hohmann's transfers but can be neglected with NTP systems of sufficient thrust.

Equations modelling this continuous-burn-rendez-vous are obtained by integrating parameterised equations of motion in a gravity free scenario before simplifying them and separating known propulsion parameters and interplanetary data from unknown values like mass or travel time. The initial acceleration is

$$a_0 = \frac{F}{m_0} = \frac{c_e}{\tau}\left(1 - \frac{m_f}{m_0}\right) \tag{4}$$

in which m_0 is the initial and m_f the final mass of the space craft. The latter is composed of the masses of the propulsion system and the payload encompassing the space craft's subsystems. Further, F and c_e are fixed parameters in the present case. From this equation and the relation of the given distance D to the voyage duration τ which can be seen as an effective "cruising speed"

$$\frac{D}{\tau} = c_e\left(\sqrt{\frac{m_f}{m_0}} - 1\right)^2\left(1 - \frac{m_f}{m_0}\right)^{-1} \tag{5}$$

being a function of the mass fraction and c_e, a set of equations can be derived. It yields a final mass solution for a given travel time

$$m_f(\tau) = F\frac{\tau}{c_e}\left(1 - \left(\frac{c_e\tau - D}{c_e\tau + D}\right)^2\right) \tag{6}$$

and the necessary propellant mass $m_p = m_0 - m_f$. These equations can be inverted to obtain a solution consisting in the voyage duration as a function of the final mass. Note that while the system would in general yield mass specific data enabling scaling, absolute numbers are fixed by imposing absolute propulsion parameters through a_0. Another interesting result obtained is the moment to switch from acceleration to deceleration:

$$\frac{\tau_{acc}}{\tau} = \left(1 + \sqrt{\frac{m_f}{m_0}}\right)^{-1}. \tag{7}$$

Note that the remaining time τ_{dec} can be obtained by simply subtracting this time from the voyage duration τ.

The results per space craft are given in section 3.4.1.

3.2.2. Low thrust solution for NEP

The mission analyses for NEP have been done with an in-house developed Matlab-Code which solves the equation of motion with the differential equation solver *ode45* provided by Matlab. The equation of motion given through

$$\vec{a} = \ddot{\vec{r}} = \frac{\mu}{r^2}\vec{r} + \frac{1}{m}\vec{F} \tag{8}$$

where $\vec{a}$ is the acceleration vector, $\vec{r}$ is the position vector, μ is the gravitational parameter, m is the mass of the vehicle and $\vec{F}$ is the thrust vector. In order to solve this differential equation of the second order, it has to be transformed into a differential equation system of first order, which has the form of

$$\dot{\vec{y}} = \vec{y} = f(\vec{r}, \vec{v}, m) \tag{9}$$

with

$$\vec{y} = \left[r_x, r_y, r_z, v_x, v_y, v_z, m\right], \\ \dot{\vec{y}} = \left[v_x, v_y, v_z, a_x, a_y, a_z, \dot{m}\right]. \tag{10}$$

The results per space craft are given in section 3.4.2.

3.2.3. Mission Analysis Process (MAP)

All the objectives of the mission analysis can be combined by performing two objectives in one mission scenario. Furthermore, the existing global space infrastructure needs to be considered as well as possible *In Situ Resources Utilisation* (ISRU). Figure 5 shows a simplified evaluation flow diagram of space vehicle for a specific mission scenario and the relation to mission analysis. However, this is usually an iteratively converging process.

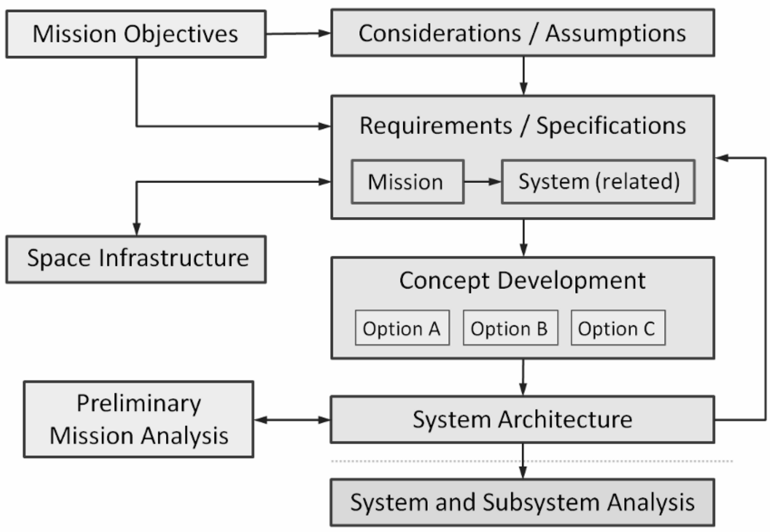


Figure 5 – Simplified evaluation flow diagram of space vehicle.

After selecting mission destinations and defining mission purposes, the mission and system based requirements and specifications can be extrapolated. Here, concepts need to be defined for basic system architecture of proposed NTP and NEP.

A basic breakdown of relevant contributions to the total mass of the space craft is

$$m_{tot}(t) = m_{PLD} + m_{PS} + m_{PL}(t), \tag{11}$$

with m_{tot} the total mass, m_{PLD} the payload's mass, m_{PS} the propulsion system's mass and m_{PL} the propellant's mass which varies over time. While further details can be considered, these masses are the most pertinent ones for typical mission analysis and thus sufficient. Figure 6 on the next page shows a simplified mass related system architecture scheme of generic spacecraft powered by both NTP and NEP. The payload mass of the space craft is mission related and will be assumed with respect to each specific mission scenario.

In the framework of this mission analysis some structural masses were integrated into corresponding masses. For example, the propellant tanks' mass is assumed to be a part of total propellant mass and is constant. As argued above, some propellants with smaller atomic mass can entail higher tank masses as it is the case for Hydrogen and Helium. Heavy gases such as Xenon can be stored in simpler tanks. However, the use of different propellant types is out of

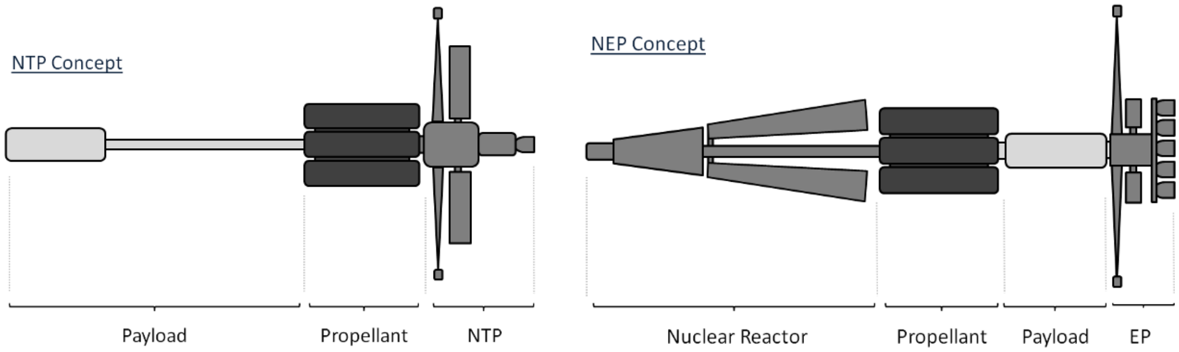


Figure 6 – Mass related system architecture of NTP and NEP based space vehicle concepts.

scope and has therefore been neglected. It is recommended to consider them in upcoming mission analysis, especially as far as thermal propulsion systems are concerned, which rely on light gases as a propellant as indicated in the Annex of [R 4].

The propellant mass decrease during the thrusters' burn depends on the mass flow rates of stored propellant. In the case of variable exhaust velocity, a saving of propellant mass can be expected [R 36]. However in this mission analysis the propellant mass flow rate distribution will be considered as constant that leads to

$$m_{PL}(t) = m_{PLT} + m_{PL,0} - \dot{m}_{PL} t \,, \tag{12}$$

with m_{PLT} the mass of the propellant tank, $m_{PL,0}$ the initial mass of propellant, and $\dot{m}_{PL}$ the propellant mass flow rate.

Propulsion mass of NTP consists of nuclear power source (NPS), thermal power processing unit (TPPU), power conditioning and distribution unit (PCDU), radiators (R), thermal thruster (TP), position control thrusters (PCT) and structural mounting mast (S) leading to:

$$m_{PS} = m_{NPS} + m_{TPPU} + m_{PCDU} + m_R + m_{PCT} + m_S + m_{TP} \,, \tag{13}$$

In case of NEP, the thermal thruster (TP) will be replaced by electric propulsion (EP). NEP and NTP concepts have been introduced in [R 2, 3 and 4]. An overview of selected electric thrusters in this mission analysis is given in table 8 on page 20.

For power scaling of NEP power plant the Brayton cycle based NEP reactor can be used, which according to [R 3] in the power range between 100 kW and 5 MW varies in the mass specific power α between 20 and 50 W/kg, and the power specific mass α^{-1} respectively between 50 kg/kW and 20 kg/kW. For scaling of nuclear power source in mission analysis the power density was assumed with 20 W/kg as worst case scenario. This value corresponds with other data found in literature [R 3, 22 and 23].

The summary of relevant NTPs that was tested in the past is shown in table 4 on page 15. Additionally an extrapolated thruster with 25 MW thermal power and 3.6 kN thrust is presented for comparison with existed thruster models. The masses of propulsion system and propellant relative to initial total mass has been optimized for specific mission and propulsion concept in order to achieve shortest orbit transfer times from Earth to Mars.

The launch costs of current launch systems limit the initial total mass of space vehicle. Considering payload masses the initial total masses for selected targets and mission scenarios has been assumed and summarised in table 11 on the next page with related data for selected mission targets. The number of modules has been considered with transfer to GTO by Ariane 5 ECB with a maximum payload mass of ~ 10 metric tons.

Mission data:	Unit	Mars I (20)	Mars I (50)	Mars II (50)
Mission Type	-	*Mars Cargo Cargo Tug*	*Mars Cargo Cargo Tug*	*Manned / Heavy Cargo*
Return to Earth	-	Yes	Yes	Yes
Initial mass	10^3 kg	150	150	200
Modules (A5 ECA, 20 t each)	-	3 + 5	3 + 5	5 + 5
Payload	10^3 kg	100	100	100
Available mass	10^3 kg	50	50	100
NEP / NTP *	%	10 - 90	10 - 90	10 - 90
Propellant *	%	90 - 10	90 - 10	90 - 10
Specific power	W/kg	20	50	50

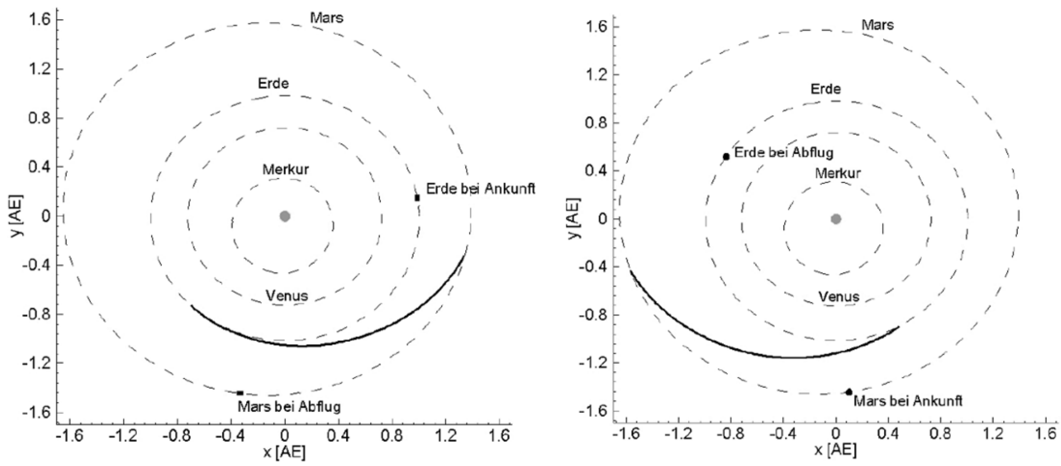
Table 11 – Overview of selected mission scenarios and respective system related parameters.
NEP /NTP and propellant masses have been optimized for specific thruster type.

3.3. Mission analysis procedure

For further mission analysis the basic analysis procedure, considerations and respective assumptions is explained below.

3.3.1. Assumptions

The vehicle is assumed to start from one of the libration points of Earth-Lunar system. It can further be assumed that the vehicle is located out of Earth's sphere of influence, on Earth orbit and has the Earth's velocity. The following consideration is to give an overview of the capability of different thrusters and to determine approximately the velocity increment Δv and travel time using different thrusters. Therefore gravitational and atmospherically perturbations are neglected. The orbits of Earth and the other planets are assumed to be coplanar and circular in order to reduce the complexity of the calculation and thus the computing time. By the example of Mars mission three different types of trajectory can be devised which are shown in figures 7, 8 and 9 [R 38]. These possible trajectories have been analyzed by Schmidt explicitly for manned Mars missions. In order to compare different propulsion types and respective transfer times A-type trajectories will be used for different mission targets in the framework of DiPoP project.



Outward Trajectory Type A Return

Figure 7 – Schematic view of A-type trajectory from orbit 0 to orbit 1 (Earth-Mars orbit transfer) [R 38].

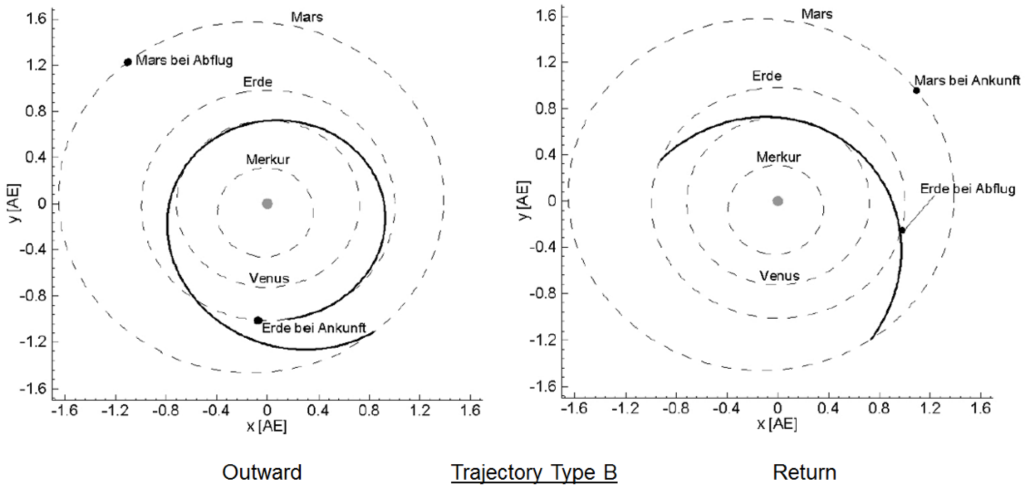


Figure 8 – Schematic view of B-type trajectory from orbit 0 to orbit 1 (Earth-Mars orbit transfer) [R 38].

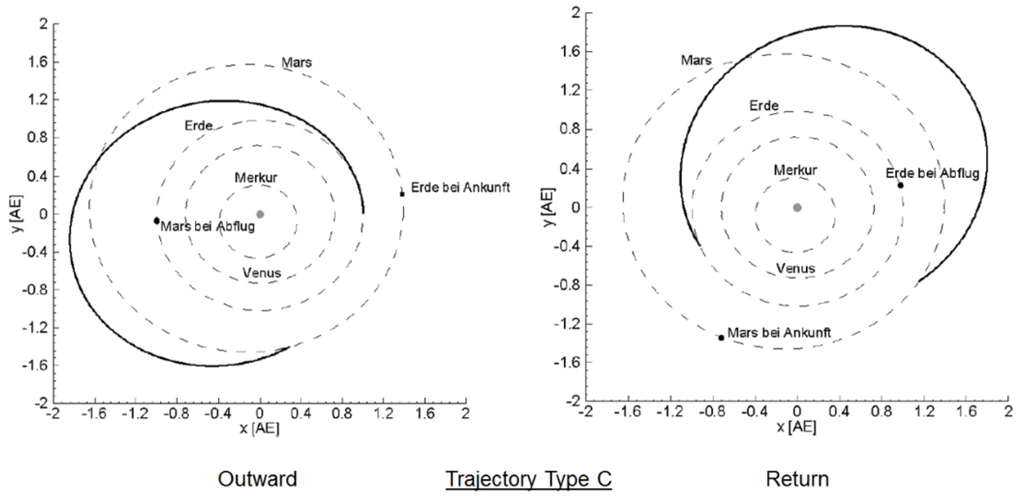


Figure 9 – Schematic view of C-type trajectory from orbit 0 to orbit 1 (Earth-Mars orbit transfer) [R 38].

A more detailed schematic of an A-type mission trajectory is shown in figure 10 on the next page. Here the starting orbit is r_0 in blue and the target orbit is r_1 in green. The thrust phase is indicated with red colour and coasting phase without thrust in turquoise. Thrusters can be fired in different directions with respect to the position vector. In the present simplified 2D problem, the angle between the thrust and the velocity vector are defined by α_T and represent the ratio between angular and radial acceleration. Low α_T is more applicable for low thrust propulsion concept while for higher thrust levels of 500 N and more depending on available Δv higher alphas may be investigated.

Further determination need to be made by magnitude of main thrust, where depending on type of propulsion concept the provided thrust can be one of two opposite extreme thrust levels: low-thrust and high-thrust. The low-thrust orbit transfer considered below is similar to a Homann orbit transfer. The first thrust phase raises/reduces the apoapsis/periapsis near to the targets orbit, following by a coasting phase without thrust. After the coasting phase the last burn raises/reduces the periapsis and the apoapsis to the target targets orbit. This type of orbit transfer is generally used by NEP driven vehicles.

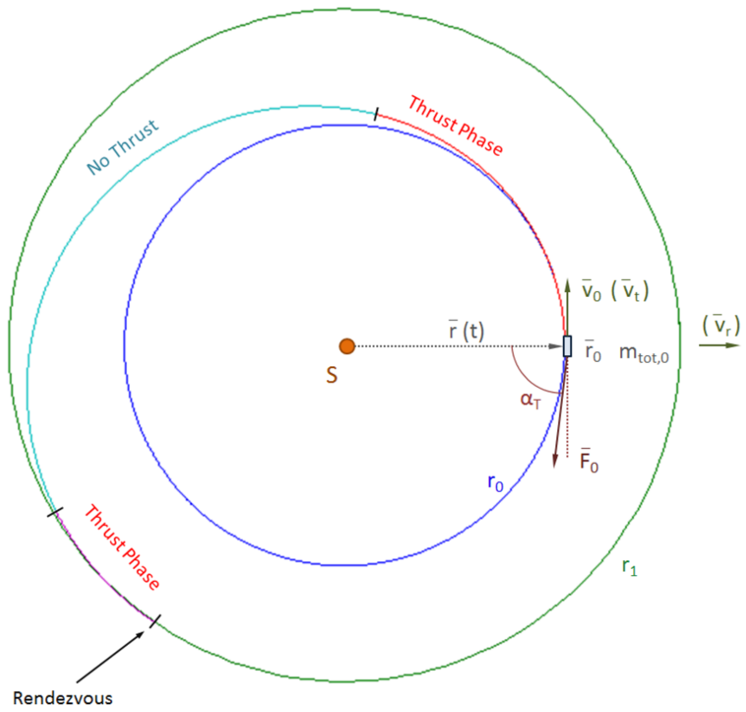


Figure 10 – Schematic view of trajectory from orbit 0 to orbit 1 evaluated by MATLAB mission analysis tool.

During high-thrust orbit transfer the thrust phases take place while near position to start and end orbit. In opposite to low-thrust transfer the space vehicle has hyperbolic trajectory. That means the thrust vector near to target orbit is showing in opposite direction compared to velocity vector. However such thrust intensive transfer manoeuvre requires high thrust levels at relative high exhaust velocities. In this case only NTP or Fusion type propulsion systems can be considered.

The figures 11 and 12 show different initial trajectories from start orbit of 400 km around Earth at different thrust levels with specific impulse of 3000 s and at different exhaust velocities with thrust level of 100 / 300 N [R 38]. Nevertheless as example an EP propulsion with clustered thrust of 300 N and specific impulse of 3000 s equals to 4.4 MW of thrust power that even in case of high efficient EP (thrust efficiency of 70%) will require a nuclear power plant with net power output of 6.3 MW and power plant mass of ca. 130 metric tons.

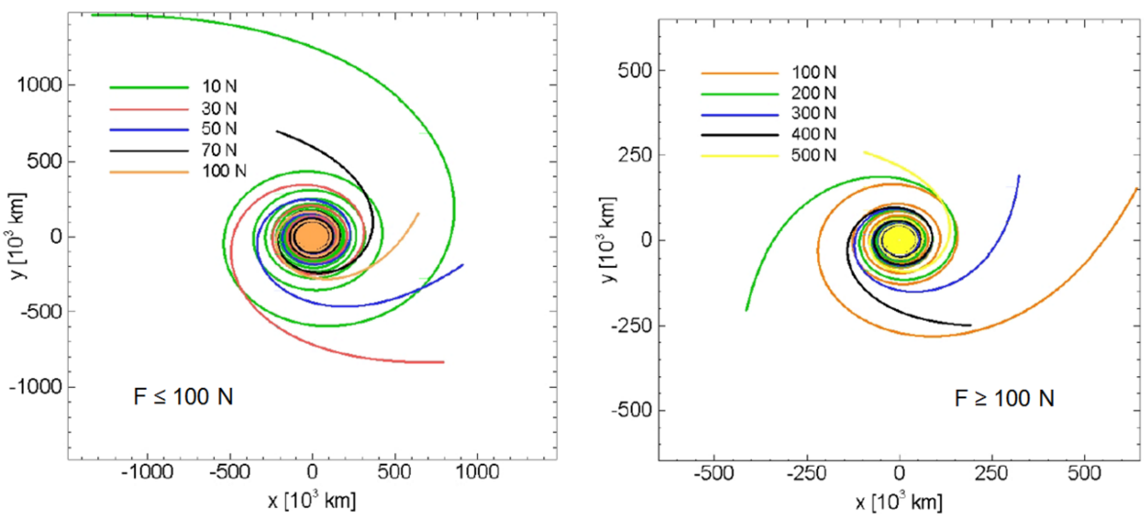


Figure 11 – Trajectories at different thrust levels and respective specific impulse of 3000 s [R 38].

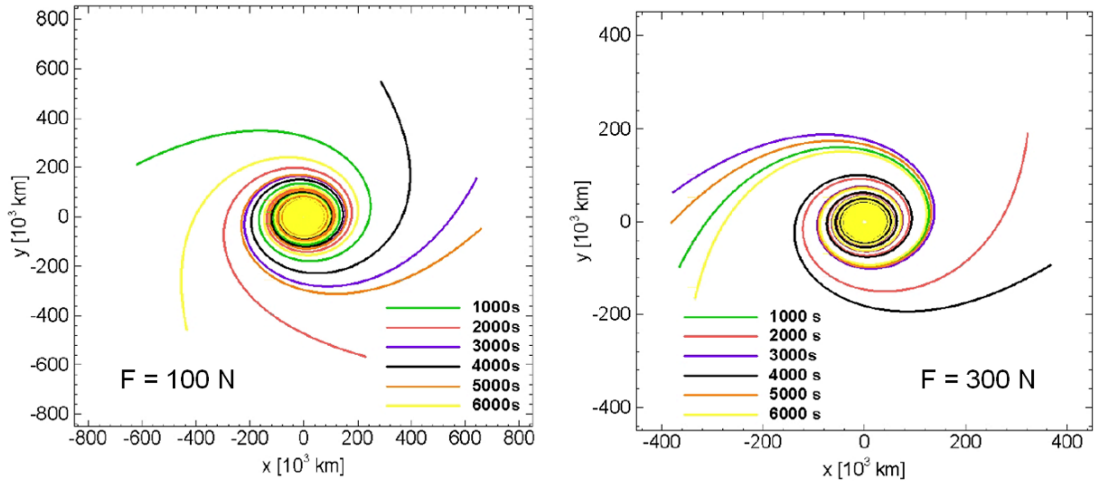


Figure 12 – Trajectories at different specific impulse levels and respective thrusts of 100 / 300 N [R 38]

3.3.2. Procedure

A simplified flow chart of MATLAB tool's structure is shown in figure 13 below. This flow chart shows mission analysis procedure for all scenarios. The output of each propulsion configuration will be analysed in case of overall feasibility and compared with concepts with respective data such as travel time, required Δv, complexity mass of propulsion system and propellant mass.

Different strategies can be used for orbit transfer to a destination target. There are no tools which can find the time or Δv-optimal trajectory. The difficulty is to find an optimal strategy finding the trajectories. Here experiences and expert knowledge is required. For the following consideration a trajectory similar to Hohmann orbit transfer is chosen to travel from Earth to Mars. The first maneuver will raise the apoapsis near to the Mars' orbit. After that the vehicle will coast a certain time. Similar to Hohmann orbit transfer a second and a last burn will increase the semi major and eliminate the eccentricity simultaneous. At the end of this phase the vehicle enters

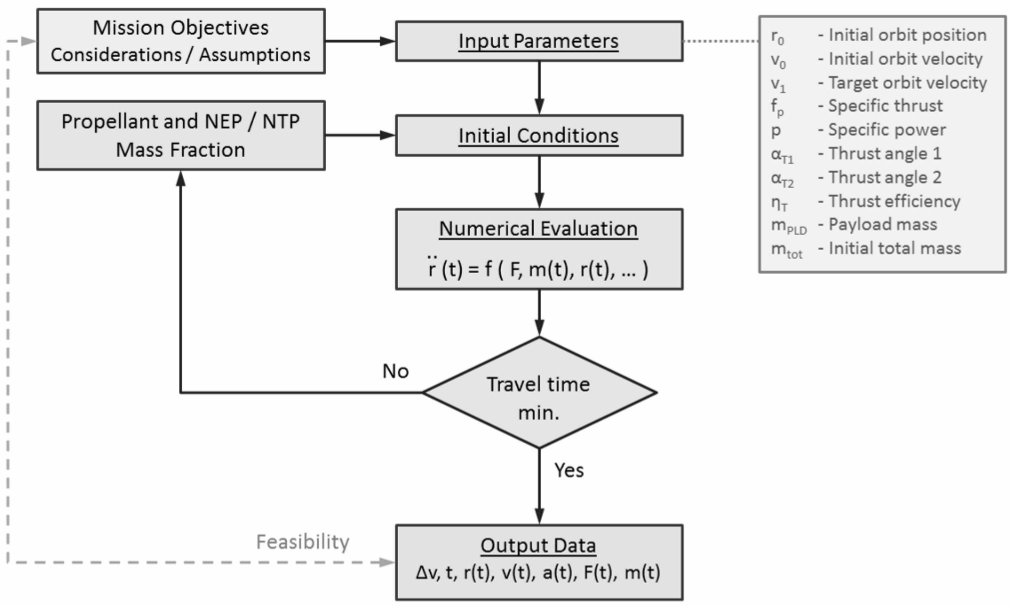


Figure 13 – Flowchart of MATLAB mission analysis tool.

Mars' sphere of influence. Here the vehicle is on a parabolic orbit respective to Mars. Hence a capture maneuver has to be done, which will be not considered further.

For mission to Kuiper-Belt objects or Saturn moons other strategies have to be chosen. With the strategy mentioned above the travel time will be much higher than travel time of mission with gravity assist. Due to its complexity interplanetary gravity assist manoeuvres are out of scope and require further extensive analysis.

3.4. Analysed scenarios

The following section gathers the results of the mission analysis as conducted above for a trip to Mars at shortest distance. A section concerning the high thrust mission scenarios will be followed by a section concentrating the low thrust mission scenarios.

3.4.1. Missions using NTP Systems

An evaluation of the field free approximation for the continuous-burn-rendez-vous from outlined in section 3.2.1 has been conducted for all of the NTP from table 4 and table 5 except for RHTP proved not to fit at all to the field free assumption. The estimation yielded table 12 on pages 33 and 34 and table 13 on pages **Erreur ! Signet non défini.** to 38 for both fixed transfer durations and fixed payload masses respectively. The destination is Mars when it is nearest to Earth (approximately 0.525 AU).

For table 12, calculations for four months (τ = 120 d), three months (τ = 90 d), two months (τ = 60 d), one and a half months (τ = 45 d), one month (τ = 30 d), and half a month (τ = 15 d) have been performed. The results concentrated in table 12 encompass the dry mass fraction ($\varepsilon = M_B/M_0$) of mass after the burn vs. mass at the beginning, propellant mass M_P, the duration of both the acceleration (τ_{acc}) and the deceleration leg (τ_{dec}), and initial acceleration a_0 both in SI units and compared to the gravitational acceleration exerted by the Sun at the distance of Earth, i.e. a_E = 5.9e-3 m/s². Especially the latter is valuable as it can point out how realistic the approximation is. Since the basic premise of this tool is a negligible Solar gravitational acceleration, results below $a_0/a_E = 2$ appear to indicate relatively unreliable data, below 1 even inapplicable. The former are hence highlighted in red italic letters to indicate that the respective propulsions systems are potentially inapt for the selected mission and the latter in a warning bold italic red. Since none of the thrusters achieves $a_0/a_E > 1$ for the 120 and 90 d cases, the respective results have been discarded altogether and can be found in Annex B. Further, the other values in lines with $a_0/a_E < 2$ are faded grey – with exception of the *payload margin* (PLM).

The PLM describes how many more times a generic payload of 40 t (twice the mass of MIR's basic module) can be included in M_B after burn besides the propulsion system. If this value is larger than 20, it is highlighted in bold green pointing out economically highly interesting thruster applications. In many cases, they appear with low a_0/a_E making a case for more detailed mission simulations. If the PLM is however smaller than one it determines critical applications and is printed in bold red. While values between one and zero (i.e. less than 40 t in the present estimation) may still be interesting for contingency robotic short term applications, a PLM of zero – only the propulsion system is delivered – or below indicates purposeless missions. Values appearing in the line of the respective thruster are also faded grey.

A third criterion is a high ε: The fraction varies from zero to one and values closer to one represent very mass efficient transfer configurations. Values larger than two thirds are thus highlighted in bold green. However, certain systems yield even fractions between 0.95 and 1 which are printed in bold yellow for being an extreme result of the estimation. Its reliability for these systems has to be confirmed in a more detailed mission simulation. Also, values less than 0.2 are printed in bold yellow: Other than indicating relatively bad mass efficiency, little ε often correlate with prohibitive absolute propellant masses.

Not meeting these three criteria allows ruling out certain propulsion applications which are faded out except for the failing value. Among the remaining lot, exceptionally attractive configurations making a mark by having a good mass fraction ε are highlighted with a grey background. Summarising the findings, it can be stated that with consideration for a field free assumption the solid body NTFP stemming from the ROVER and Timberwind programmes are able to perform transfers of about 30 to 45 days with remaining dry mass fractions from 69 % to 90 %. The respective masses after burn M_B range from approximately 150 to 450 tons. Since these thrusters have in some extent already experienced ground testing, they can consequently be qualified as disruptive technologies. A similar statement could also be made for certain GCRs, were not the fissile containment rather challenging.

					τ = 60 d				
	ε / -	a_0 / m/s²	a_0 / a_E / -	M_0 / t	M_B / t	PLM. / -	M_P / t	τ_{acc} / d	τ_{dec} / d
NRX-A6	**0.76**	1.02e-2	*1.73*	814.5	615.1	15.15	199.4	32.1	27.9
NERVA-1	**0.83**	1.07e-2	*1.81*	757	631.4	15.56	125.6	31.4	28.6
PHOEBUS-2A	**0.94**	1.13e-2	*1.92*	698.6	654.2	15.51	44.4	30.5	29.5
NERVA-2	**0.93**	1.13e-2	*1.91*	718.7	670.2	15.90	48.5	30.5	29.5
Timberwind 45	**0.87**	1.09e-2	*1.85*	900.2	784.6	19.58	115.6	31.0	29.0
Timberwind 75	**0.92**	1.12e-2	*1.90*	876.4	807.2	**20.12**	69.2	30.6	29.4
Timberwind 230	**0.98**	1.15e-2	*1.96*	851.9	831.1	**20.57**	20.8	30.2	29.8
RD-0410	**0.16**	5.74e-3	*0.97*	1553.7	251.1	6.23	1302.6	42.8	17.2
GCR [R20] a	**0.94**	1.13e-2	*1.92*	1323.1	1245.4	**30.01**	77.8	30.5	29.5
GCR [R20] b	**0.94**	1.13e-2	*1.92*	2205.2	2075.6	**50.77**	129.6	30.5	29.5
GCR [R13] a	0.40	7.77e-3	*1.32*	3859.7	1538.5	**36.96**	2321.2	36.8	23.2
GCR [R13] b	0.40	7.77e-3	*1.32*	9005.9	3589.8	**84.75**	5416.1	36.8	23.2
NLBR	**0.86**	1.08e-2	*1.84*	1844	1584.8	**38.82**	259.2	31.1	28.9
NSWR	**1.00**	1.17e-2	*1.98*	5662.7	5636.3	**140.89**	26.5	30.0	30.0
D3He WGD 10	**0.02**	3.78e-3	*0.64*	47565.6	909.6	**20.24**	46656	52.7	7.3
11Bp WGD 10	**1.00**	1.17e-2	*1.98*	2402.3	2391.2	**-20.22**	11.2	30.0	30.0
GDM [R60]	0.29	6.88e-3	*1.17*	165046.2	47265.7	**1171.64**	117780.5	39.1	20.9
Discovery 2	**0.09**	4.92e-3	*0.83*	71119.3	6319.3	**148.98**	64800	46.2	13.8
					τ = 45 d				
	ε / -	a_0 / m/s²	a_0 / a_E / -	M_0 / t	M_B / t	PLM. / -	M_P / t	τ_{acc} / d	τ_{dec} / d
NRX-A6	**0.69**	1.74e-2	2.94	478.4	328.8	8.01	149.6	24.6	20.4
NERVA-1	**0.79**	1.85e-2	3.13	438.2	344	8.37	94.2	23.9	21.1
PHOEBUS-2A	**0.92**	1.99e-2	3.37	397.2	363.9	8.25	33.3	23.0	22.0
NERVA-2	**0.91**	1.98e-2	3.36	408.9	372.5	8.46	36.4	23.0	22.0
Timberwind 45	**0.83**	1.90e-2	3.22	517.6	431	10.74	86.7	23.5	21.5
Timberwind 75	**0.90**	1.97e-2	3.33	499.6	447.7	11.13	51.9	23.1	21.9
Timberwind 230	**0.97**	2.04e-2	3.46	481.1	465.5	11.43	15.6	22.7	22.3

Table 12 – Mission results for the continuous-burn-rendez-vous at fixed transfer durations.

	ε / -	a_0 / m/s²	a_0 / a_E / -	M_0 / t	M_B / t	PLM. / -	M_P / t	τ_{acc} / d	τ_{dec} / d
τ = 45 d									
RD-0410	**0.08**	8.44e-3	*1.43*	1056.8	79.9	1.95	976.9	35.3	9.7
GCR [R20] a	**0.92**	2.00e-2	3.38	751.7	693.4	16.21	58.3	23.0	22.0
GCR [R20] b	**0.92**	2.00e-2	3.38	1252.8	1155.6	**27.77**	97.2	23.0	22.0
GCR [R13] a	*0.29*	*1.23e-2*	*2.08*	*2446.1*	*705.2*	*16.13*	*1740.9*	*29.3*	*15.7*
GCR [R13] b	*0.29*	*1.23e-2*	*2.08*	*5707.5*	*1645.4*	***36.13***	*4062.1*	*29.3*	*15.7*
NLBR	**0.82**	1.88e-2	3.19	1062.6	868.2	**20.91**	194.4	23.6	21.4
NSWR	**0.99**	2.07e-2	3.51	3187.8	3167.9	**79.19**	19.8	22.5	22.5
D3He WGD 10	**0.00**	5.14e-3	*0.87*	34992.8	0.8	**-2.48**	34992	44.8	0.2
11Bp WGD 10	**0.99**	2.07e-2	3.51	1352.4	1344	**-46.40**	8.4	22.5	22.5
GDM [R60]	**0.18**	1.05e-2	*1.79*	107781.3	19445.9	**476.15**	88335.4	31.6	13.4
Discovery 2	**0.03**	7.01e-3	*1.19*	49912	1312	**23.80**	48600	38.7	6.3
τ = 30 d									
NRX-A6	0.57	3.59e-2	6.09	231.2	131.5	3.06	99.7	17.1	12.9
NERVA-1	**0.70**	3.93e-2	6.66	206	143.2	3.36	62.8	16.4	13.6
PHOEBUS-2A	**0.88**	4.38e-2	7.42	180.3	158.1	3.10	22.2	15.5	14.5
NERVA-2	**0.87**	4.36e-2	7.39	185.9	161.6	3.19	24.2	15.5	14.5
Timberwind 45	**0.76**	4.09e-2	6.94	240.2	182.5	4.52	57.8	16.0	14.0
Timberwind 75	**0.85**	4.31e-2	7.31	228	193.4	4.77	34.6	15.6	14.4
Timberwind 230	**0.95**	4.56e-2	7.73	215.6	205.2	4.92	10.4	15.2	14.8
RD-0410	**0.01**	1.36e-2	2.31	655.4	4.1	**0.05**	651.3	27.8	2.2
GCR [R20] a	**0.89**	4.40e-2	7.46	340.7	301.8	6.42	38.9	15.5	14.5
GCR [R20] b	**0.89**	4.40e-2	7.46	567.9	503.1	11.45	64.8	15.5	14.5
GCR [R13] a	***0.14***	*2.22e-2*	*3.76*	*1353.4*	*192.8*	*3.32*	*1160.6*	*21.8*	*8.2*
GCR [R13] b	***0.14***	*2.22e-2*	*3.76*	*3158*	*449.9*	*6.25*	*2708.1*	*21.8*	*8.2*
NLBR	**0.74**	4.04e-2	6.84	495.2	365.6	8.34	129.6	16.1	13.9
NSWR	**0.99**	4.65e-2	7.88	1419	1405.8	**35.13**	13.2	15.0	15.0
D3He WGD 10	**0.04**	7.39e-3	*1.25*	24345.9	1017.9	**22.95**	23328	24.9	5.1
11Bp WGD 10	**0.99**	4.65e-2	7.88	602.0	596.4	**-65.09**	5.6	15.0	15.0
GDM [R60]	**0.06**	1.81e-2	3.07	62671.3	3781.1	**84.53**	58890.2	24.1	5.9
Discovery 2	**0.00**	1.08e-2	*1.83*	32449.7	49.7	**-7.76**	32400	28.9	1.1
τ = 15 d									
NRX-A6	0.32	1.14e-1	19.33	72.9	23	**0.35**	49.9	9.6	5.4
NERVA-1	0.48	1.34e-1	22.70	60.4	29	**0.50**	31.4	8.9	6.1
PHOEBUS-2A	**0.77**	1.65e-1	27.90	48	36.9	**0.07**	11.1	8.0	7.0
NERVA-2	**0.76**	1.63e-1	27.68	49.7	37.5	**0.09**	12.1	8.0	7.0
RD-0410	***0.07***	*2.55e-2*	*4.33*	*349.4*	*23.8*	***0.54***	*325.6*	*11.9*	*3.1*

Erreur ! Source du renvoi introuvable. Table 12 (ctd.) – Mission results for the continuous-burn-rendez-vous at fixed transfer durations.

	τ = 15 d								
	ε / -	a_0 / m/s²	a_0 / a_E / -	M_0 / t	M_B / t	PLM. / -	M_P / t	τ_{acc} / d	τ_{dec} / d
Timberwind 45	0.58	1.45e-1	24.49	68	39.1	0.94	28.9	8.5	6.5
Timberwind 75	0.72	1.60e-1	27.05	61.6	44.3	1.04	17.3	8.1	6.9
Timberwind 230	0.91	1.78e-1	30.17	55.2	50	1.04	5.2	7.7	7.3
GCR [R20] a	0.78	1.66e-1	28.17	90.3	70.8	0.65	19.4	8.0	7.0
GCR [R20] b	0.78	1.66e-1	28.17	150.4	118	1.83	32.4	8.0	7.0
GCR [R13] a	*0.00*	*5.16e-2*	*8.74*	*581.8*	*1.5*	*-1.46*	*580.3*	*14.3*	*0.7*
GCR [R13] b	*0.00*	*5.16e-2*	*8.74*	*1357.5*	*3.5*	*-4.91*	*1354*	*14.3*	*0.7*
NLBR	*0.54*	*1.41e-1*	*23.90*	*141.9*	*77.1*	*1.13*	*64.8*	*8.6*	*6.4*
NSWR	0.98	1.85e-1	31.39	356.4	349.8	8.73	6.6	7.5	7.5
D3He WGD 10	0.25	1.15e-2	*1.95*	15624.9	3960.9	96.52	11664	10.0	5.0
11Bp WGD 10	0.98	1.85e-1	31.39	151.2	148.4	-76.29	2.8	7.5	7.5
GDM [R60]	0.01	3.82e-2	6.48	29716.3	271.2	-3.22	29445.1	13.7	1.3
Discovery 2	*0.14*	*1.87e-2*	*3.17*	*18732.3*	*2532.3*	*54.31*	*16200*	*11.0*	*4.0*

Erreur ! Source du renvoi introuvable. Table 12 (ctd.) – Mission results for the continuous-burn-rendez-vous at fixed transfer durations.

It is further notable, that the initial acceleration meets the field free criterion the better, the shorter the transfer duration is set to be and that shortening the transfer duration also decreases the total mass budget including the propellant mass, which is reasonable: With a fixed thrust force, smaller masses are more easily accelerated. Consequently, thrust legs take less time resulting in a reduced propellant consumption, too. This trend competes however with a decreasing mass fraction ε indicating a potential trade off among travel time and propellant economy.

Table 13 concentrates field free approximations for fixed payload masses M_{PL}. Calculations have been performed for 10, 25, 50, 100, 150 and 200 tons. The criteria concerning the mass fraction ε and a_0/a_E already used in table 12 have been likewise applied to table 13. Similar to the preceding table, outstanding results among the applications meeting the criteria are marked with a grey background. The observations for fixed payload masses back those for fixed travel times. NTFP developed during the NERVA and Timberwind projects are able to resolve the proposed missions. Also some GCR may be considered. The transfer durations take from approximately 20 to 40 days being similar to the ones identified as particularly interesting in the fixed durations cases. The respective propellant masses range from 4 to 70 tons for the smaller payloads – i.e. 50 t or less – and from 10 to 150 tons for payloads of 100 t or more.

Excluding NSWR applications, the space craft with extreme masses are a 22 tons craft driven by a Timberwind 230 thruster with 3 t of propellant and 10 t of payload, and a 332 t craft propelled by an NRX-A6-like thruster with 122 t of propellant and 200 t of payload. Note however, that NSWR could even provide lighter space craft adding negligible propulsion and propellant masses to the payload if an actual NSWR appeared feasible in the next decades. Also note that operations using an RD-0410 thruster require relatively large propellant masses due to the small characteristic acceleration.

The mass fraction ε appears not to follow any particular trend in correlation with the payload. This reveals a need for further optimisation studies for a given payload size. Finally, it is also noteworthy that in neither of both tables 12 and 13 fusion propulsion devices can compete with NTFP systems which are less advanced.

				M_{PL} = 10 t			
	τ / d	a_0 / m/s²	a_0/a_E / -	M_0 / t	M_B / t	ε / -	M_P / t
NRX-A6	14.0	0.13	21.48	65.6	19	0.29	46.6
NERVA-1	12.7	0.18	30.15	45.5	19	0.42	26.5
PHOEBUS-2A	16.3	0.14	23.89	56.1	44	**0.78**	12.1
NERVA-2	16.2	0.14	24.09	57.1	44	**0.77**	13.1
Timberwind 45	9.1	0.34	57.50	29.0	11.5	0.40	17.5
Timberwind 75	8.5	0.44	74.50	22.4	12.5	0.56	9.9
Timberwind 230	9.2	0.46	77.52	21.5	18.3	**0.85**	3.2
RD-0410	*33.1*	*0.01*	2.07	*730.8*	*12*	***0.02***	*718.8*
GCR [R20] a	13.3	0.21	35.18	72.3	55	**0.76**	17.3
GCR [R20] b	10.5	0.32	54.51	77.7	55	**0.71**	22.7
GCR [R13] a	*23.5*	*0.03*	*5.20*	*977.8*	*70*	***0.07***	*907.8*
GCR [R13] b	*24.8*	*0.03*	*4.85*	*2448*	*210*	***0.09***	*2238*
NLBR	11.7	0.22	36.68	92.4	42	0.45	50.4
NSWR	2.7	5.65	958.42	11.7	10.5	**0.90**	1.2
D3He WGD 10	50.5	0.00	*0.77*	39372	110	**0.00**	39262
11Bp WGD 10	69.5	0.01	*1.47*	3223	3210	**1.00**	12.9
GDM [R60]	22.1	0.03	4.40	43723	410	**0.01**	43313
Discovery 2	39.1	0.01	*1.39*	42611	370	**0.01**	42241

				M_{PL} = 25 t			
	τ / d	a_0 / m/s²	a_0/a_E / -	M_0 / t	M_B / t	ε / -	M_P / t
NRX-A6	17.3	0.09	15.38	91.6	34	0.37	57.6
NERVA-1	16.0	0.12	20.31	67.5	34	0.50	33.5
PHOEBUS-2A	18.7	0.11	18.38	72.8	59	**0.81**	13.8
NERVA-2	18.5	0.11	18.58	74.0	59	**0.80**	15.0
Timberwind 45	12.7	0.19	32.69	51.0	26.5	0.52	24.5
Timberwind 75	12.1	0.24	40.20	41.4	27.5	0.66	13.9
Timberwind 230	12.3	0.26	44.36	37.6	33.3	**0.89**	4.3
RD-0410	*36.9*	*0.01*	*1.83*	*827.5*	*27*	***0.03***	*800.5*
GCR [R20] a	14.9	0.17	28.46	89.3	70	**0.78**	19.3
GCR [R20] b	11.8	0.26	44.42	95.4	70	**0.73**	25.4
GCR [R13] a	*24.5*	*0.03*	*4.93*	*1032*	*85*	***0.08***	*946.8*
GCR [R13] b	*25.2*	*0.03*	*4.75*	*2498*	*225*	***0.09***	*2273*
NLBR	13.2	0.18	29.71	114.1	57	0.50	57.1
NSWR	4.1	2.42	409.63	27.3	25.5	**0.93**	1.8
D3He WGD 10	50.8	0.00	*0.77*	39647	125	**0.00**	39522
11Bp WGD 10	69.7	0.01	*1.47*	3238	3225	**1.00**	13.0
GDM [R60]	22.1	0.03	4.39	43877	425	**0.01**	43452
Discovery 2	39.2	0.01	*1.39*	42770	385	**0.01**	42385

Table 13 – Mission results for the continuous-burn-rendez-vous at fixed payload masses.

	τ / d	a_0 / m/s²	a_0/a_E / -	M_0 / t	M_B / t	ε / -	M_P / t
				MPL = 50 t			
NRX-A6	21.5	0.06	10.80	130.4	59	0.45	71.4
NERVA-1	20.2	0.08	13.53	101.4	59	0.58	42.4
PHOEBUS-2A	22.1	0.08	13.34	100.4	84	**0.84**	16.4
NERVA-2	21.9	0.08	13.51	101.7	84	**0.83**	17.7
Timberwind 45	16.9	0.12	19.82	84.1	51.5	0.61	32.6
Timberwind 75	16.2	0.14	23.39	71.2	52.5	**0.74**	18.7
Timberwind 230	16.2	0.15	26.07	63.9	58.3	**0.91**	5.6
RD-0410	*41.2*	*0.01*	*1.60*	*947.5*	*52*	*0.05*	*895.5*
GCR [R20] a	17.2	0.13	21.67	117.3	95	**0.81**	22.3
GCR [R20] b	13.6	0.20	34.10	124.3	95	**0.76**	29.3
GCR [R13] a	*26.0*	*0.03*	*4.56*	*1115*	*110*	*0.10*	*1005*
GCR [R13] b	*25.8*	*0.03*	*4.60*	*2580*	*250*	*0.10*	*2330*
NLBR	15.4	0.13	22.82	148.5	82	0.55	66.5
NSWR	5.7	1.24	210.93	53.0	50.5	**0.95**	2.5
D3He WGD 10	51.3	0.00	0.76	40073	150	**0.00**	39923
11Bp WGD 10	69.9	0.01	1.45	3263	3250	**1.00**	13.0
GDM [R60]	22.3	0.03	4.36	44127	450	**0.01**	43677
Discovery 2	39.5	0.01	1.38	43030	410	**0.01**	42620
				M_{PL} = 100 t			
	τ / d	a_0 / m/s²	a_0/a_E / -	M_0 / t	M_B / t	ε / -	M_P / t
NRX-A6	27.7	0.04	7.01	201.1	109	0.54	92.1
NERVA-1	26.5	0.05	8.34	164.5	109	0.66	55.5
PHOEBUS-2A	27.7	0.05	8.67	154.5	134	**0.87**	20.5
NERVA-2	27.4	0.05	8.80	156.2	134	**0.86**	22.2
Timberwind 45	22.9	0.07	11.44	145.6	101.5	**0.70**	44.1
Timberwind 75	22.2	0.08	13.01	128.1	102.5	**0.80**	25.6
Timberwind 230	21.9	0.08	14.38	115.9	108.3	**0.93**	7.6
RD-0410	*47.5*	*0.01*	*1.33*	*1134*	*102*	*0.09*	*1032*
GCR [R20] a	21.1	0.09	14.75	172.3	145	**0.84**	27.3
GCR [R20] b	16.5	0.14	23.45	180.7	145	**0.80**	35.7
GCR [R13] a	*28.5*	*0.02*	*4.02*	*1264*	*160*	*0.13*	*1104*
GCR [R13] b	*27.0*	*0.03*	*4.34*	*2736*	*300*	*0.11*	*2436*
NLBR	18.9	0.09	15.86	213.8	132	0.62	81.8
NSWR	8.1	0.63	107.50	104.1	100.5	**0.97**	3.6
D3He WGD 10	52.3	0.00	*0.75*	40835	200	**0.00**	40635
11Bp WGD 10	*70.5*	0.01	*1.43*	3313	3300	**1.00**	13.1
GDM [R60]	22.5	0.03	4.32	44611	500	**0.01**	44111
Discovery 2	39.9	0.01	*1.36*	43529	460	**0.01**	43069

Table 13 (ctd.) – Mission results for the continuous-burn-rendez-vous at fixed payload masses.

	M_{PL} = 150 t						
	τ / d	a_0 / m/s²	a_0/a_E / -	M_0 / t	M_B / t	ε / -	M_P / t
NRX-A6	32.6	0.03	5.27	267.3	159	0.59	108.3
NERVA-1	31.5	0.04	6.10	224.9	159	0.71	65.9
PHOEBUS-2A	32.3	0.04	6.44	207.9	184	0.89	23.9
NERVA-2	31.9	0.04	6.55	209.8	184	0.88	25.8
Timberwind 45	27.5	0.05	8.15	204.5	151.5	0.74	53.0
Timberwind 75	26.8	0.05	9.08	183.4	152.5	0.83	30.9
Timberwind 230	26.4	0.06	9.95	167.4	158.3	0.95	9.1
RD-0410	*52.4*	*0.01*	*1.17*	*1289*	*152*	*0.12*	*1137*
GCR [R20] a	24.3	0.07	11.23	226.5	195	0.86	31.5
GCR [R20] b	19.0	0.11	17.95	236.1	195	0.83	41.1
GCR [R13] a	*30.7*	*0.02*	*3.64*	*1398*	*210*	*0.15*	*1188*
GCR [R13] b	*28.1*	*0.02*	*4.12*	*2883*	*350*	*0.12*	*2533*
NLBR	*21.8*	*0.07*	*12.27*	*276.3*	*182*	*0.66*	*94.3*
NSWR	*9.9*	*0.43*	*72.24*	*154.8*	*150.5*	*0.97*	*4.3*
D3He WGD 10	*53.1*	*0.00*	*0.73*	*41513*	*250*	*0.01*	*41263*
11Bp WGD 10	*71.0*	*0.01*	*1.41*	*3363*	*3350*	*1.00*	*13.2*
GDM [R60]	*22.7*	*0.03*	*4.27*	*45073*	*550*	*0.01*	*44523*
Discovery 2	*40.3*	*0.01*	*1.35*	*44004*	*510*	*0.01*	*43494*
	M_{PL} = 200 t						
	τ / d	a_0 / m/s²	a_0/a_E / -	M_0 / t	M_B / t	ε / -	M_P / t
NRX-A6	36.7	0.03	4.25	331.1	209	0.63	122.1
NERVA-1	35.7	0.03	4.84	283.7	209	0.74	74.7
PHOEBUS-2A	36.3	0.03	5.13	260.8	234	0.90	26.8
NERVA-2	35.9	0.03	5.23	263.0	234	0.89	29.0
Timberwind 45	31.4	0.04	6.36	262.0	201.5	0.77	60.5
Timberwind 75	30.7	0.04	7.00	237.9	202.5	0.85	35.4
Timberwind 230	30.2	0.04	7.62	218.8	208.3	0.95	10.5
RD-0410	*56.5*	*0.01*	*1.06*	*1428*	*202*	*0.14*	*1226*
GCR [R20] a	27.1	0.05	9.08	280.1	245	0.87	35.1
GCR [R20] b	21.2	0.09	14.57	290.8	245	0.84	45.8
GCR [R13] a	*32.7*	*0.02*	*3.34*	*1523*	*260*	*0.17*	*1263*
GCR [R13] b	*29.1*	*0.02*	*3.92*	*3023*	*400*	*0.13*	*2623*
NLBR	24.4	0.06	10.05	337.2	232	0.69	105.2
NSWR	11.4	0.32	54.43	205.5	200.5	0.98	5.0
D3He WGD 10	53.8	0.00	*0.72*	42130	300	0.01	41830
11Bp WGD 10	*71.5*	*0.01*	*1.39*	*3413*	*3400*	*1.00*	*13.3*
GDM [R60]	22.9	0.02	4.23	45517	600	0.01	44917
Discovery 2	40.6	0.01	*1.33*	44459	560	0.01	43899

Table 13 (ctd.) – Mission results for the continuous-burn-rendez-vous at fixed payload masses.

3.4.2. Missions using NEP Systems

The analysis procedure introduced in section 3.3 with the in-house MATLAB tool has been used to evaluate *Δv*, transfer times and respective masses for given mission scenarios with low thrust propulsion systems. In this section, the results for all EP based and analyzed propulsion concepts have been summarised in figures 14 to figure 20 and in the tables 14, 15 and 16.

	Unit	NEXT	NEXIS	173Mv1	457Mv2	GSC	HiPARC	X16	MAI Li
c_e	km/s	40.91	69.16	23.69	24.72	9.908	20	35.714	33.4
Thrust eff.	-	0.7	0.75	0.61	0.63	0.54	0.28	0.38	0.43
F/P	mN/kW	34,548	21,863	51.719	50.793	111.666	29	21,638	26.417
F	N	20.99	16.58	18.73	19.75		7.31	11.93	13.72
NEP mass	t	30.67	38.23	18.18	19.37	not	13.05	28.04	26.65
M_P	t	19.33	11.77	31.82	30.63	feasible	36.95	21.96	23.35
Δv	km/s	5.64	5.65	5.65	5.65		5.66	5.65	5.65
Travel time	d	539.12	743.79	592.94	557.14		1317.66	887.73	821.46
	m	17.97	24.79	19.76	18.57		43.92	29.59	27.38
	a	1.48	2.04	1.62	1.53		3.61	2.43	2.25

Table 14 – Simulation result of Mars I mission scenario with NEP concept for specific EP system and power density of 20 W/kg for nuclear power (150 t initial mass and 100 t payload mass).

The initial mass of the space craft in Mars I mission scenario is about 150 t including a payload mass of 100 t. The trajectory for a low mass specific power *α* of 20 W/kg is shown in figure 14 all EP transfer times from Earth to Mars vary from 540 to 1318 days at full propellant consumption. Due to the similarity of the trajectory to a Hohmann transfer the used propellant mass primarily depends on the exhaust velocity of the used EP system cluster. Very low exhaust velocities GSC arcjets based on NEP are not feasible in this case. However, additional variation of the thrust angle may change this result due to higher thrust level comparable to other EP systems.

While 20 W/kg based NEPs have relatively long transfer times depending on thrust efficiency, in case of 50 W/kg based nuclear power sources the situation changes, where most NEP concepts except of HiPARC (597 d) have transfer times in the range between 348 and 422 days (see figure 15). As in case of 20 W/kg the used propellant mass depends on exhaust velocity of used EP system, where complete propellant has been used. Here NEXIS and NEXT based EP systems show lowest propellant consumption followed by AF-MPD thrusters (X16 and MAI). Hall Effect thrusters allow faster thruster times due to high thrust efficiency and thrust level and need more propellant for given initial and payload mass.

	Unit	NEXT	NEXIS	173Mv1	457Mv2	GSC	HiPARC	X16	MAI Li
c_e	km/s	40.91	69.16	23.69	24.72	9.908	20	35.714	33.4
Thrust eff.	-	0.7	0.75	0.61	0.63	0.54	0.28	0.38	0.43
F/P	mN/kW	34,548	21,863	51.719	50.793	111.666	29	21,638	26.417
F	N	52.67	41.54	47.30	49.82		18.33	29.93	34.46
NEP mass	t	30.78	38.30	18.37	19.55	not	13.10	28.13	26.76
M_P	t	19.22	11.70	31.63	30.45	feasible	36.90	21.87	23.24
Δv	km/s	5.61	5.62	5.61	5.61		5.65	5.63	5.62
Travel time	d	347.99	376.21	354.24	349.58		596.21	421.84	397.04
	m	11.60	12.54	11.81	11.65		19.87	14.06	13.23
	a	0.95	1.03	0.97	0.96		1.63	1.16	1.09

	Unit	NEXT	NEXIS	173Mv1	457Mv2	GSC	HiPARC	X16	MAI Li
c_e	km/s	40.91	69.16	23.69	24.72	9.908	20	35.714	33.4
Thrust eff.	-	0.7	0.75	0.61	0.63	0.54	0.28	0.38	0.43
F/P	mN/kW	34,548	21,863	51.719	50.793	111.666	29	21,638	26.417
F	N	127.33	91.56	149.09	151.55	74.08	71.54	75.48	88.97
NEP mass	t	74.41	84.43	57.90	59.47	13.59	51.10	70.94	69.11
M_P	t	25.59	15.57	42.10	40.53	86.41	48.90	29.06	30.89
Δv	km/s	5.60	5.60	5.60	5.60	5.61	5.61	5.61	5.60
Travel time	d	307.28	328.42	298.29	297.81	328.92	340.86	340.82	327.70
	m	10.24	10.95	9.94	9.93	10.96	11.36	11.36	10.92
	a	0.84	0.90	0.82	0.82	0.90	0.93	0.93	0.90

Table 16 – Simulation result of Mars I mission scenario with NEP concept for specific EP system and power density of 50 W/kg for nuclear power (200 t initial mass and 100 t payload mass).

Table 15 – Simulation result of Mars I mission scenario with NEP concept for specific EP system and power density of 50 W/kg for nuclear power (150 t initial mass and 100 t payload mass).

Exemplary trajectories of HiPARC and NEXT based NEP are shown in figure 17 and figure 18. The proposed trajectory strategy A (figure 7, page 28) in case of HiPARC based NEP with more than 1318 travel days is too long. Other trajectory types such as B or C (figures 8 and 9, page 29) require additional extensive mission analysis.

A comparison between NEP, NTP, D3He and chemical propulsion for given payload of 100 t and initial mass of 150 t is shown in figure 16. Due to similarity of Hohmann transfer and higher thrust levels all results of non-EP based propulsion systems were evaluated with Hohmann equations. These results of non-EP based propulsion systems need to be considered as comparable rough estimations, where initial masses has been evaluated by the need of required propellant mass for a given Δv. Nevertheless, the capability results of different NTP systems based propulsion concepts are summarized in table 12 and 13. The results shows the mandatory of high power density for nuclear power source of NEP concepts, which would allow shorter transfer times or higher payload masses.

Additional mission analysis of Mars II scenario has been done for low NEP concepts with nuclear power source power density of 50W/kg, 200 t initial mass and inclusive payload mass of 100 t. As in Mars I scenario the simulated data include one-way trajectories only with consumption of complete propellant storage.

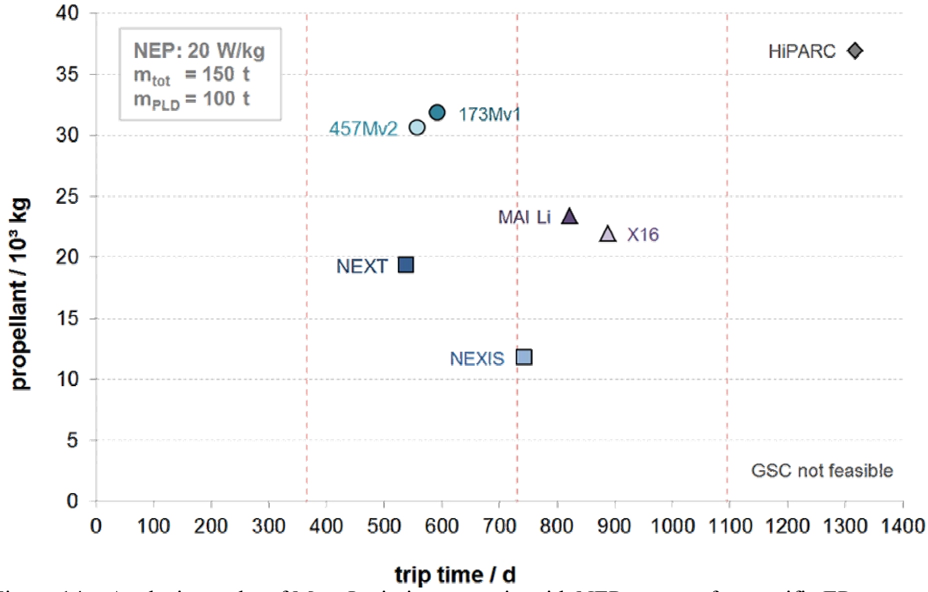


Figure 14 – Analysis results of Mars I mission scenario with NEP concept for specific EP system

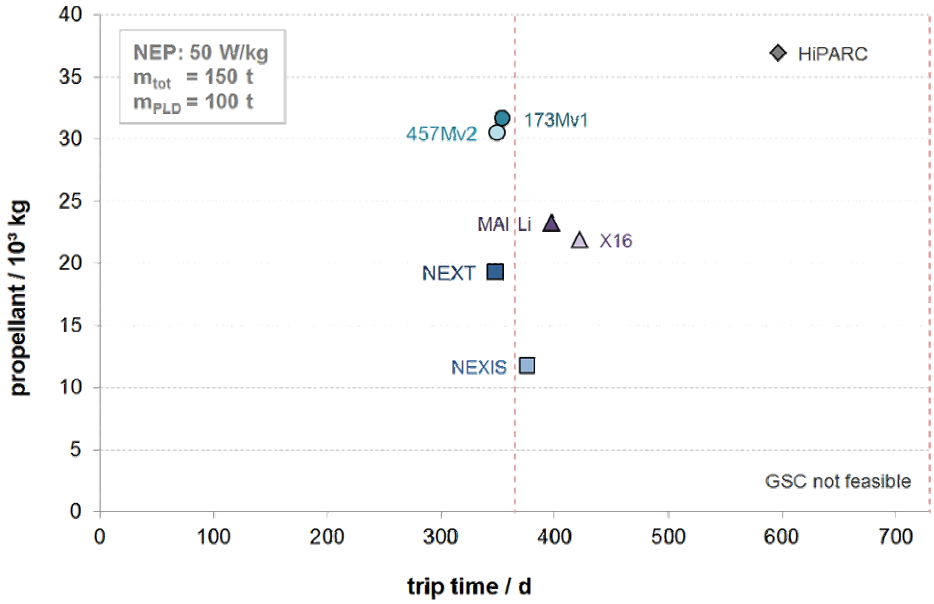


Figure 15 – Analysis results of Mars I mission scenario with NEP concept for specific EP system and power density of 50 W/kg for nuclear power (150 t initial mass and 100 t payload mass).

Due to higher NEP mass higher thrust levels has been achieved with evaluated transfer times between 300 and 340 days. The GSC arcjet concept became feasible in this scenario case. However, the propellant consumption is relatively high compared to NEXT, NEXIS, X16 and MAI AF-MPD. The tendency of propellant consumption is here similar to the Mars I scenario with power source density of 50 W/kg, which rely on exhaust velocity and thrust efficiency of the EP system. While higher exhaust velocities towards NEXIS thruster level decrease propellant consumption, the higher thrust efficiency decrease the travel time. Additional comparison with rough estimated high thrust propulsion concepts from Mars II scenario is given in figure 19 and 20. Since required propellant and thruster mass need to be increased for non-EP based propulsion concept leading to higher initial mass. However, the Hohmann transfer for high thrust propulsion concepts is questionable and require more direct trajectory, which is less efficient but still able to provide with faster transfer times.

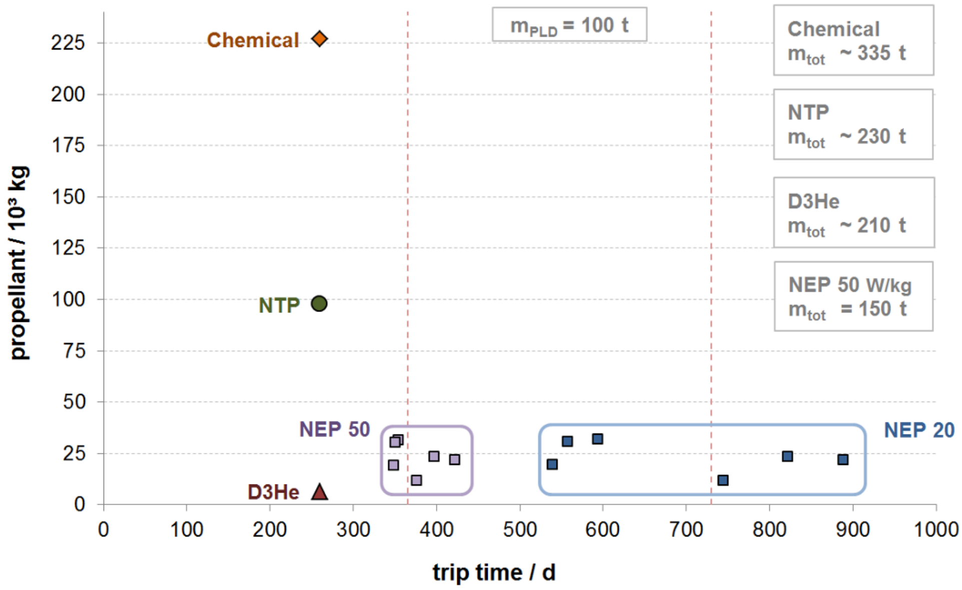


Figure 16 – Comparison of analysis results of Mars I mission scenario with NEP / NTP / NFTP concept and chemical propulsion. (NTP/D3He and chemical propulsion results based on impulsive Hohmann transfer).

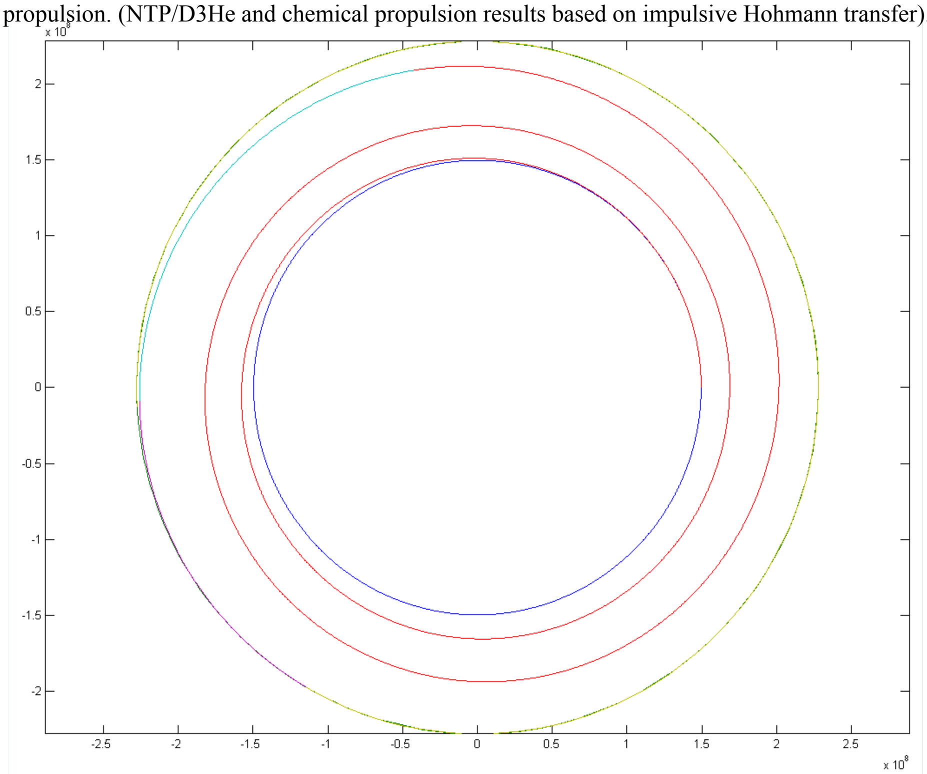


Figure 17 – Trajectory of Mars I mission scenario with HiPARC NEP and power density of 20 W/kg (150 t initial mass and 100 t payload mass).

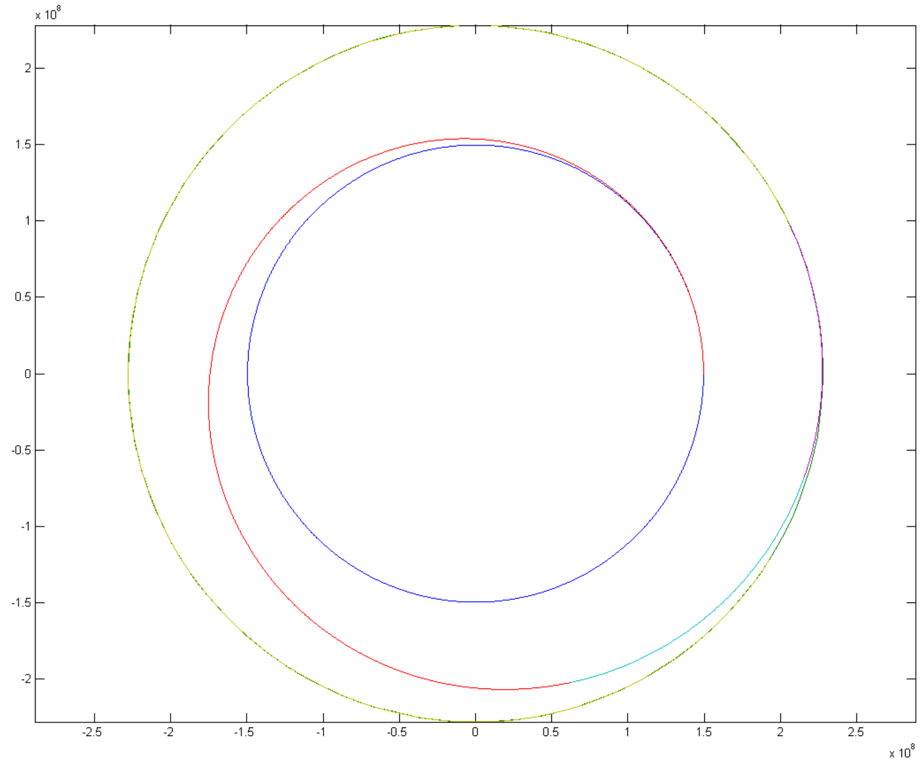


Figure 18 – Trajectory of Mars I mission scenario with NEXT NEP and power density of 20 W/kg (150 t initial mass and 100 t payload mass).

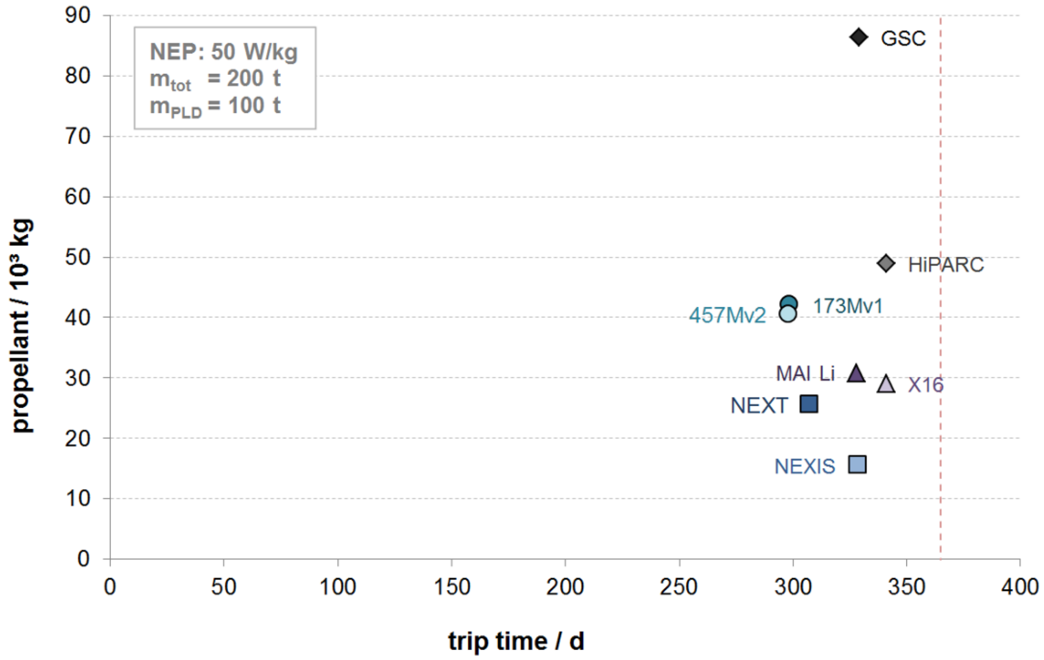


Figure 19 – Analysis results of Mars II mission scenario with NEP concept for specific EP system and power density of 50 W/kg for nuclear power (200 t initial mass and 100 t payload mass).

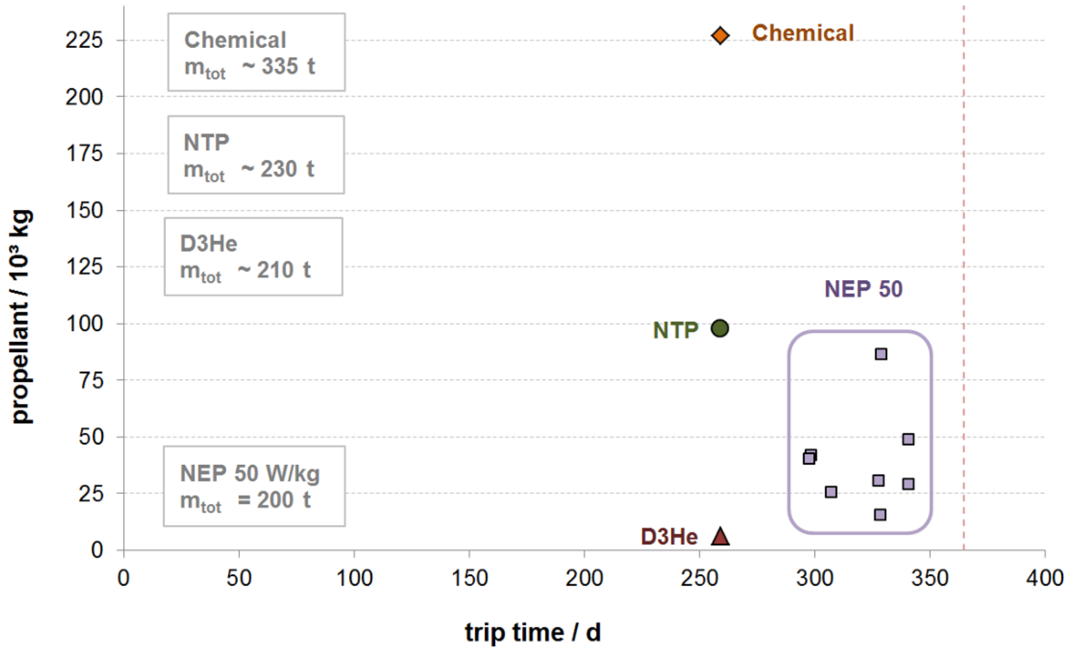


Figure 20 – Comparison of analysis results of Mars II mission scenario with NEP / NTP / NFTP concept and chemical propulsion. (NTP / NFTP and chemical propulsion results based on impulsive Hohmann transfer).

4. Conclusions

The present report recapitulates the activities of the work package 23 of the DiPoP project. It covers the results of the surveys on both Nuclear Electric Propulsion and on Nuclear Thermal Propulsion and respective power conversion, summarises primordial approaches and presents a study of representative mission scenarios.

The assessment of the mission analysis results shows that even with technology available in the short or medium term significant breakthrough ventures are possible. Already rather modest NEP systems enable significantly enhanced interplanetary transportation. Even more audacious cargo and inhabited missions are feasible in the medium term if high power NTP approaches could be enacted. An exemplary voyage to Mars could consist in a space craft of about 150 metric tons departing from an infrastructure in Earth's orbit or a Lagrange Point space port and reaching the destination in approximately one month with an estimated propellant consumption of 30 tons. As tables 12 and 13 insinuate, this is well feasible with systems like NERVA which were already ground tested by no later than the 1960s. This is also spotted to be in accordance with some national space flight strategies.

The costs of the vehicle increases however with higher initial mass due to relatively high costs of launch systems and their limitation of payload transfer with current technology. This can be an obstacle for several more advanced NTP systems which appear extremely attractive for missions heading farther given their higher thrust levels potentially reducing the travel time. Further evolution of these concepts is thus necessary; and a system level tradeoff between cost and travel time has to be made for each specific mission.

At a lower power level, Nuclear Electric Propulsion may offer similar transport capacities, albeit entailing near to conventional transfer durations of at least 200 days for Mars. For the chosen mission from Earth to Mars with 100 tons of payload using NEP, the analyses show that propulsion systems with high specific impulse can save a lot of propellant. This feature is the great benefit of NEP for heavy cargo missions above conventional chemical and disruptive nuclear thermal thrusters. However, the analyses concentrated in section 3.4.2 also reveal how important high mass specific power sources are for EP. It was found that the thrust level is a critical parameter when it comes to continuous burns. For example, NEXIS cluster driven vehicles have a longer transfer duration than NEXT cluster driven ones. Also, HETs have a better trip performance than HiPARCs. This was expectable. However, there are hints at an additional propellant economy issue. In fact, a longer transfer duration is also a longer burn duration, and therefore more propellant is consumed. All of this correlates with the power feed to the thruster. Hence, it can be summarized, that increasing the power specific mass of power plants reduces both travel time *and* propellant consumption. This result makes a case for further development of light weight nuclear power sources.

While this outlook to open new, significant classes of space flight missions makes a certain case for the application of nuclear power in space and while the available data and review on Disruptive Power and Propulsion appears to sustain the technical aspect of such projects, a consideration of present days Public Perception of the enacted concepts reveals other challenges. The major impediment may be seen in the public's reluctance to implement nuclear technologies which can in essence be characterised as a feeble solidarity towards project responsible.

Annex A: Notes and reminders

Exhaust velocity implications

The exhaust velocity of thermal propulsion systems is approximately

$$c_e = \sqrt{\frac{2\gamma}{\gamma - 1} \cdot \frac{R\,T}{M_A}}\,, \tag{A}$$

in which γ is the ratio of heats, R the gas constant, M_A the molecular mass of the propellant and T its temperature. The equation reveals that higher temperatures are better for augmented exhaust velocities as well as minimum molecular masses. It can be shown that the influence of γ is not as important. Further, it can be concluded that hydrogen is the optimal propellant. Hydrogen has also the best heat properties making it the best coolant as well. However, it is difficult to store and will likely make a case for cryogenic tanks.

A note on specific dimensions

This document often refers to the mass specific power

$$\alpha = \frac{P}{m}, \tag{B}$$

which relates a power P – be it electrical, or thermal, or of the jet – to a mass m. For the calculation, the latter typically is the mass of the total system or of a subsystem of reference. Cautiously used, this dimension enables to compare how power dense systems of various sizes are. Another way to put it is its inverse, the power specific mass

$$\alpha^{-1} = \frac{1}{\alpha} = \frac{m}{P}\,, \tag{C}$$

which can be used to indicate how much mass of a given system is needed to provide a give reference portion of power.

It is noteworthy, that literature prior to the 1980s like [R 22] preferred the power specific mass for comparisons, while contemporary documents like [R 23] and [R 35], which is characteristic for current AIAA literature utilise the mass specific power.

Types of radiation

There are four types of radiation [R 4]:

- **Alpha radiation**, which removes four nucleons forming an helium-core consisting of two protons and two neutrons, which has low shielding requirements
- **Beta radiation**, which may be an electron or a positron, which is the same mass with an opposite charge. Electrons come from a neutron transmuting into a proton. Positrons appear if a proton is changed into a neutron, which has medium shielding requirements
- **Gamma radiation**, which is a highly energetic photon from a core relaxation, which has high shielding requirements
- **Neutron radiation**, which consists in the core's loss of a neutron, which has high shielding requirements and which risks the space craft's activation.

Annex B: Discarded results from the NTP mission estimation

As mentioned above, in section 3.4.1, some thrusters were critically not able to fulfil the field free condition for certain fixed voyage durations. Never the less, they are listed here for the sake of transparency.

					τ = 120 d				
	ε / -	a_0 / m/s²	a_0 / a_E / -	M_0 / t	M_B / t	PLM. / -	M_P / t	τ_{acc} / d	τ_{dec} / d
NRX-A6	**0.87**	2.73e-3	*0.46*	3048	2649	**66.01**	398.9	62.1	57.9
NERVA-1	**0.91**	2.79e-3	*0.47*	2898	2647	**65.95**	251.2	61.4	58.6
PHOEBUS-2A	**0.97**	2.87e-3	*0.49*	2749	2661	**65.67**	88.8	60.5	59.5
NERVA-2	**0.97**	2.87e-3	*0.49*	2826	2729	**67.36**	97	60.5	59.5
Timberwind 45	**0.93**	2.82e-3	*0.48*	3482	3251	**81.24**	231.1	61.0	59.0
Timberwind 75	**0.96**	2.86e-3	*0.48*	3435	3297	**82.36**	138.5	60.6	59.4
Timberwind 230	**0.99**	2.90e-3	*0.49*	3387	3345	**83.42**	41.6	60.2	59.8
RD-0410	0.42	1.98e-3	*0.34*	4496	1890	**47.21**	2605	72.8	47.2
GCR [R20] a	**0.97**	2.88e-3	*0.49*	5214	5058	**125.3**	155.5	60.5	59.5
GCR [R20] b	**0.97**	2.88e-3	*0.49*	8690	8431	**209.6**	259.2	60.5	59.5
GCR [R13] a	0.64	2.36e-3	*0.40*	12724	8082	**200.5**	4642	66.8	53.2
GCR [R13] b	0.64	2.36e-3	*0.40*	29690	18857	**466.4**	10832	66.8	53.2
NLBR	**0.93**	2.81e-3	*0.48*	7110	6591	**164**	518.4	61.1	58.9
NSWR	**1.00**	2.92e-3	*0.49*	22624	22572	**564.3**	52.9	60.0	60.0
D3He WGD 10	0.20	1.54e-3	*0.26*	117117	23805	**592.6**	93312	82.7	37.3
11Bp WGD 10	**1.00**	2.92e-3	*0.49*	9598	9576	**159.4**	22.3	60.0	60.0
GDM [R60]	0.54	2.20e-3	*0.37*	515655	280095	**6992**	235561	69.1	50.9
Discovery 2	0.33	1.81e-3	*0.31*	193397	63797	**1586**	129600	76.2	43.8
					τ = 90 d				
	ε / -	a_0 / m/s²	a_0 / a_E / -	M_0 / t	M_B / t	PLM. / -	M_P / t	τ_{acc} / d	τ_{dec} / d
NRX-A6	**0.83**	4.74e-3	*0.80*	1754	1454	**36.13**	299.2	47.1	42.9
NERVA-1	**0.89**	4.89e-3	*0.83*	1654	1466	**36.42**	188.4	46.4	43.6
PHOEBUS-2A	**0.96**	5.08e-3	*0.86*	1555	1488	**36.36**	66.6	45.5	44.5
NERVA-2	**0.95**	5.07e-3	*0.86*	1599	1526	**37.29**	72.7	45.5	44.5
Timberwind 45	**0.91**	4.96e-3	*0.84*	1981	1808	**45.15**	173.3	46.0	44.0
Timberwind 75	**0.95**	5.05e-3	*0.86*	1946	1842	**45.98**	103.9	45.6	44.4
Timberwind 230	**0.98**	5.15e-3	*0.87*	1909	1878	**46.73**	31.2	45.2	44.8
RD-0410	0.31	3.15e-3	*0.53*	2834	879.9	**21.95**	1954	57.8	32.2
GCR [R20] a	**0.96**	5.09e-3	*0.86*	2948	2831	**69.65**	116.6	45.5	44.5
GCR [R20] b	**0.96**	5.09e-3	*0.86*	4913	4718	**116.8**	194.4	45.5	44.5
GCR [R13] a	0.54	3.92e-3	*0.66*	7650	4168	**102.7**	3482	51.8	38.2
GCR [R13] b	0.54	3.92e-3	*0.66*	17850	9726	**238.1**	8124	51.8	38.2
NLBR	**0.90**	4.94e-3	*0.84*	4048.8	3660	**90.70**	388.8	46.1	43.9
NSWR	**1.00**	5.18e-3	*0.88*	12731.2	12691.5	**317.3**	39.7	45.0	45.0
D3He WGD 10	**0.11**	2.29e-3	*0.39*	78489.2	8505.2	**210.1**	69984	67.7	22.3
11Bp WGD 10	**1.00**	5.18e-3	*0.88*	5401.1	5384.3	**54.61**	16.7	45.0	45.0
GDM [R60]	0.44	3.59e-3	*0.61*	316040.9	139370.2	**3474**	176670.7	54.1	35.9
Discovery 2	0.22	2.81e-3	*0.48*	124768.5	27568.5	**680.2**	97200	61.2	28.8

Table 17 – Discarded mission results for the continuous-burn-rendez-vous at fixed transfer durations.
(Discarded due to $a_0/a_E < 1$ inadequateness to field free premise.)

Annex B: Updated NTP evaluation matrix

The mission estimations in section 3.4.1 allowed to update the evaluation matrix established in reference [R 4]. A new line has been added in the bottom of the table. The mission estimation indicates a trade off between the excellent mission potential of solid core NTFP designs and the expected advantaged of fusion propulsion concepts.

	Solid RHTP	Liquid RHTP	Solid NTFP	Liquid NTFP	Gaseous NTFP	ORION	NSWR	MCFP (WGD)	GDM
Relative Technological Readiness	***	*	***	*	*	***	*	**	*
Available information	**	**	***	**	***	**	*	**	**
Principle	***	***	***	***	**	**	*	**	**
Scaling models	**	*	**	*	***	**	***	***	***
Experimental data	**	*	***	*	**	**	-	-	-
Project documentation	**	-	***	*	**	**	-	-	-
Safety									
Power controllability	-	-	***	**	**	**	*	***	***
Passive accident prevention	-	-	**	*	*	-	**	***	***
Radiologic safety									
Avoidance of loss of radiologic inventory	**	*	**	*	*	-	-	**	**
Low severeness of radiation	*	*	*	*	*	*	**	***	***
Low health issues	**	**	**	*	*	*	**	***	***
Low chemical risks	***	***	**	*	*	**	**	***	***
System safety									
Insusceptible to single point failures	***	*	**	*	*	*	*	**	**
Safety means									
Shield	***	***	*	*	*	*	**	***	***
Distance	***	***	**	*	*	-	*	***	***
Containment	***	*	**	*	*	-	**	***	***
Low maintainance	***	***	**	**	**	*	*	***	***
System degradation	-	-	**	*	*	**	*	***	***
Fueling									
Cost	**	**	***	**	**	*	**	*	*
Availability	**	**	**	**	**	**	**	*	*
Fuel readiness	***	***	***	**	**	*	**	**	**
In Situ Resources	*	*	**	**	**	*	*	***	***
Propulsion characteristics									
High exhaust velocity	*	**	*	**	***	***	***	***	***
High characteristic acceleration	*	*	**	**	***	***	***	***	**
High mass specific power	*	*	**	***	***	***	***	***	***
Parameter invariance	-	-	***	**	**	**	***	***	***
Parameter controllability	-	-	**	**	*	**	*	**	**
Mission capability *(Destination Mars, Near Earth Asteroids...)*	-	-	***	n/a	***	n/a	***	*	*

Table 18 – Updated evaluation Matrix of concepts of NTP
(- - very bad or intrinsically impossible; * - relatively bad; ** - average; *** - good)

Annex C: Nuclear Thermal Propulsion Roadmap & summary

A roadmap for the development of a European Nuclear Thermal Propulsion System is illustrated in diagram below. As established in [the present document], solid fuel Nuclear Thermal Fission Propulsion (NTFP) thrusters appear to be the disruptive technology of choice to introduce a game change into the sustainable development of space.

The two main arguments for this statement are:

- Solid NTFP technology is already proven to be developable and has even experienced ground testing in the second half of the 20st century.
- Preliminary mission estimation show that rapid heavy transfer is most likely performed in a propellant economic manner using solid NTFP thrusters.

The basic premises for the proposed roadmap are

- Europe's international space partners with nuclear space experience share respective knowledge to boost the development.
- Public support for space nuclear programmes can be obtained in the timeframe of the NTFP development.

Upon this foundation, the first stage objectives are hence:

- To **establish practical experience** in current space and relevant terrestrial fission nuclear power generation, possibly through collaboration in Russia's space Megawatt Nuclear Power and Propulsion System (NPPS), European Generation IV civil power programmes, and similar US-American initiatives.
- To gather an expert group prior to 2015 to orchestrate various projects. This group should constitute first in 2013 and **provide by 2014 definitions for technology research and development tasks**
    - *for high temperature, mass-efficient space fission nuclear reactors,*
    - *for solid nuclear fuel with respect to exposure to hot and reactive media,*
    - *for reliable propellant storage and regenerative cooling,*

  in the frame of the Horizon 2020 research programme and compile a data base of relevant European capabilities.
- To **identify the most promising candidate missions** and define mission requirements.

The result of the activities to achieve these objectives should provide for the conception and realisation of an actual European demo mission relying on NTFP systems.

Second stage objectives are likely:

- Analytical selection of a candidate first European solid NTFP space mission
- Development of the candidate mission programme taking advantage of existing experience, and any synergies with related programmes and the adaptation of existing infrastructure where this is cost effective
- Establishing a sustainable programme based on mission analysis and the lessons learned

The proposed Horizon 2020 research includes:

- High temperature reactors, reactor control systems, long-life fuel and shielding,
- Fuel material sciences with respect to high temperature reactive media,
- Regenerative cooling based on effusion and transpiration transport in porous media,
- Mass efficient power and waste heat management and distribution,
- High temperature, low mass fixed and low temperature deployable radiators (also for NEP),
- Architecture, commissioning and safety design compatible with launch and vehicle constraints,
- Concept and comparative systems studies on more advanced propulsion approaches encompassing Gas Core Reactor technologies and Fusion propulsion,
- Social and cultural research and development concerning the public support of the programme with respect to aspects of the social implementation and the public understanding of the technology and its regulatory background.

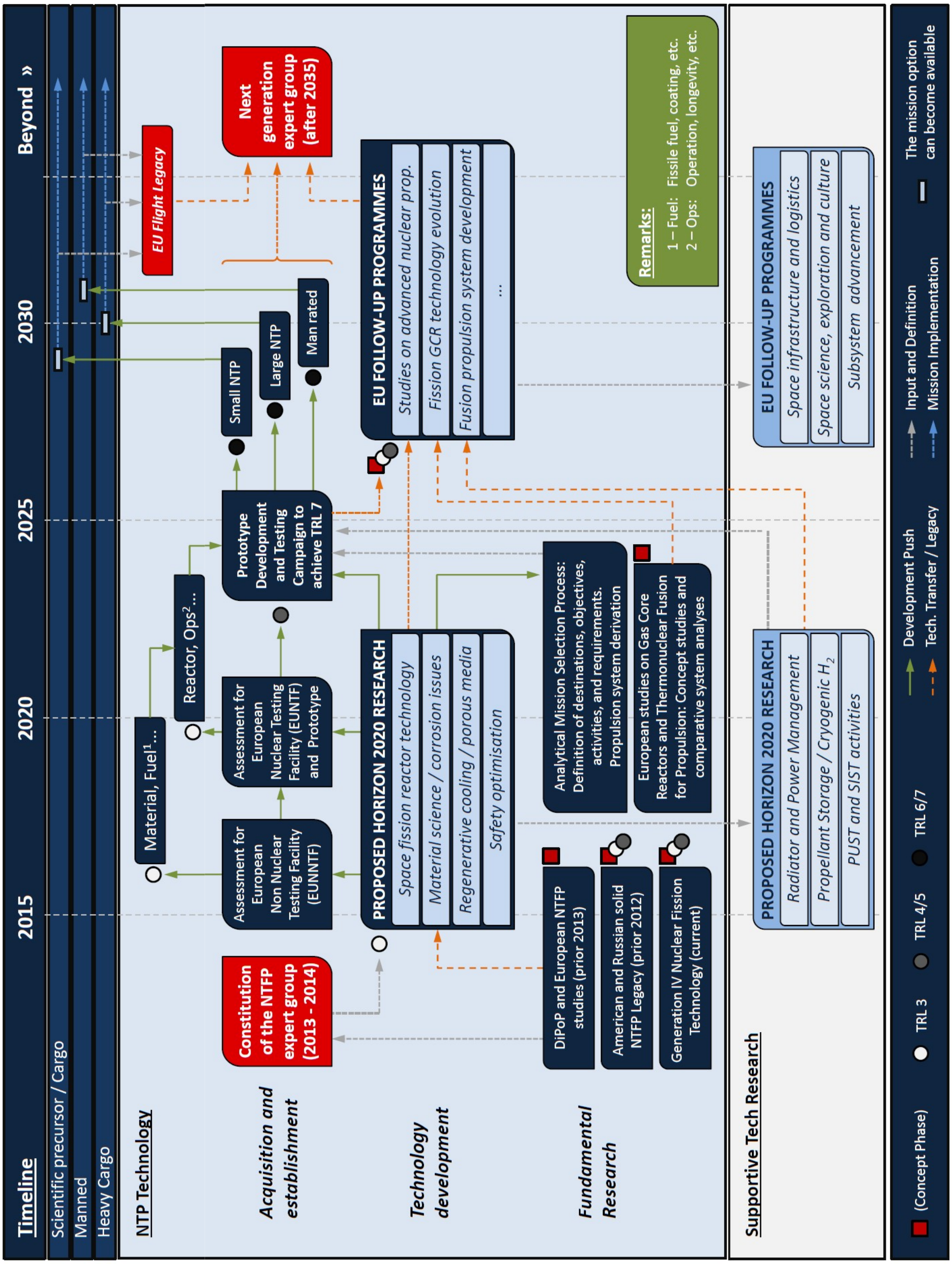


Figure 21 – Nuclear Thermal Propulsion Roadmap

Annex D: Disruptive Electric Propulsion Roadmap & summary

Proposed Roadmap Procedures

The roadmap of proposed development for Electric Propulsion (EP) is illustrated in the diagram below. One of the first proposed steps is a formation of European expertise group for primary electric propulsion. The main task of this group is coordination of non-political primary EP development in Europe. The agenda of this group is:

- Coordination of the development strategy and implementation
- Cooperation with mission/scenario analysis groups for the definition of EP requirements of each specific mission
- Cooperation with fission nuclear power generation groups with respect to Power Conditioning and Distribution Unit (PCDU) and synergy aspects, heat management and heat rejection for the definition of power processing unit (PPU) requirements for each specific EP type
- Extended cooperation with fission nuclear power generation working groups towards flight demonstration of considered nuclear electric propulsion (NEP) systems

With respect to possible mission scenarios and their requirements the primary objectives for development of primary EP are:

- review of existing and possible concepts of primary electric propulsion,
- considering estimated performance, complexity, sustainability, propellant availability, alternative propellants, feasible long life-cycle and current TRL level,
- identification of most promising EP type for scientific/robotic missions with power range below 100 kW,
- identification of most promising EP type for manned/cargo missions with high output power up to some MW,
- candidate selection depending on scalability of power level, estimated life time sustainability, propellant type, IRSU and possible technology transfer and
- mission related development considering clustering, redundancy, heritage and TRL level.

Therefore a basic development of each EP system should consist of following sub steps:

- proof of concept of EP types under consideration,
- characterization of scaling effects and optimization for higher efficiency with respect to thrust density or specific impulse and long life-cycle,
- investigation of alternative propellants with respect to storage, feeding systems, performance, availability and IRSU,
- further development towards engineering EP system models (including PPU and other additional subsystems),
- duration test for qualification towards flying EP system and

- implementation/qualification of NEP system and flight demonstration.

The following tasks would be required and need to be considered on the subsystem level:

- Implementation of superconducting materials in coils for magnetic field (B-field) generation (applied field magneto plasma-dynamic thrusters (AF-MPDT), Helicon thrusters, electron cyclotron resonance (ECR) and ion cyclotron resonance (ICR) heating)
- Cooling systems and thermal insulation for superconducting coils
- PPU development with respect to requirements of specific EP system, power scaling and PCDU development for fission nuclear power sources
- High power EP will require extensive cooling and heat rejection systems
- Development of heat rejection systems with respect to possible synergy aspects of NEP concepts.

Electric Propulsion Considerations for Roadmap

The proposed EP systems in the roadmap are separated in two main groups that are differentiated according to the operational power level. EP systems with a power range up to 100 kW are best suited for automatic scientific missions. Here, the non-disruptive Gridded Ion Thruster is one of the candidates with good efficiency, high specific impulse and long-life cycle, but with generally low plasma density making the thrusters larger and heavier than other propulsion for providing the same thrust (relevant for Kuiper belt exploration or other very high ΔV missions). However, a possible use on a manned space vehicle with a power output level up to 200 kW can be realized via clustering thus providing additional redundancy and additional use of chemical propulsion as a boost stages. Advanced disruptive Hall-Effect thrusters (HET) may be also investigated providing also long-life cycles comparable to state of the art Gridded Ion Thrusters (GIT). Compared to existing GIT current non-disruptive HET produce lower specific impulse in the range between 1500 and 3000 seconds. This can be modified by using alternative propellants with lower atomic mass. They offer higher thrust density but lower efficiency and can be operated up to 100 kW. However, current non-disruptive GIT and HET are operating with Xenon, of which the production yield is limited to a few tons per year and which has very high costs. Hence, Xenon can be justified for robotic science missions (for example with power output up to 30 kW). Such missions would require relatively small amounts of propellant. Nevertheless, in the range of 30 -200 kW, alternative propellants such as Argon or Krypton for primary EP depending on mission duration need to be investigated.

At high power ranges up to some MW, the use of alternative propellants and EP systems with higher thrust density become more important due to the fact of the required amount of propellant and number of clustered EP units. Additionally, this is strongly motivated by heavy cargo transport, manned missions e.g. to Mars or by station keeping requirements of manned space stations. An establishment of Lunar or Martian base will require a sustainable infrastructure which implies the transportation of crew, propellant, life support important supplies and modules. Here, magnetoplasmadynamic thrusters (MPDT) provide relatively high thrust densities at power levels up to some MW, specific impulses in range between 3000 and 5000 seconds, estimated thrust efficiency up to 50 %, and the use of alternative propellants such as Argon, Lithium or Hydrogen.

Furthermore, the Martian Atmosphere allows an ISRU of Carbon dioxide or Argon for propellant. The first one is available in large quantities and can be used in high power hybrid thrusters. Depending on the hybridization concept, thrusters could also allow the use of wastes such as used water. Moreover, they can combine different acceleration processes allowing for a potential further improvement of system flexibility and thrusters relevant parameters. The hybrid thrusters TIHTUS and VASIMR are good examples for possible realization. While TIHTUS consist of electrothermal arc jet as first stage and inductive RF heating as second stage, VASIMR uses a Helicon plasma source as first stage, an ICR heating as second stage and a magnetic nozzle as a final acceleration stage. However, hybrid thrusters consist of multiple stages and are highly complex. Thus, further research and development is necessary. Hybrid thrusters can be optimized in terms of efficient plasma generation and acceleration with respect to the use of different propellants, high efficiency and variable specific impulse. Different concepts and combinations can be used such as inductive radio-frequency (RF) energy coupling, ECR/ICR, Helicon, Field Reversed Configuration (FRC) via Rotating Magnetic Field (RMF), theta pinch acceleration, etc. A possible development of hybrid thrusters for the European Space Flight is considered for Horizon 2020 programs and outlined in illustrated roadmap.

In the domain of high specific impulse propulsion, more advanced concepts can be devised which partly derive from fusion plasma containment technology, such as the Inertial Electrostatic Confinement (IEC). Note however, that it is currently not envisaged to operate an IEC device at a net power gain, a situation which is abbreviated as "$Q < 1$", with Q the fraction of gained power relative to the heating power. The heating power is mainly used to accelerate ions in the device which can escape through a Voltage breach at extremely high exhaust velocities estimated to be significantly larger than those obtained by nowadays ion thrusters. In long term fusion based primary electric propulsion systems promise new ways of solar system exploration and new paradigm in space transportation.

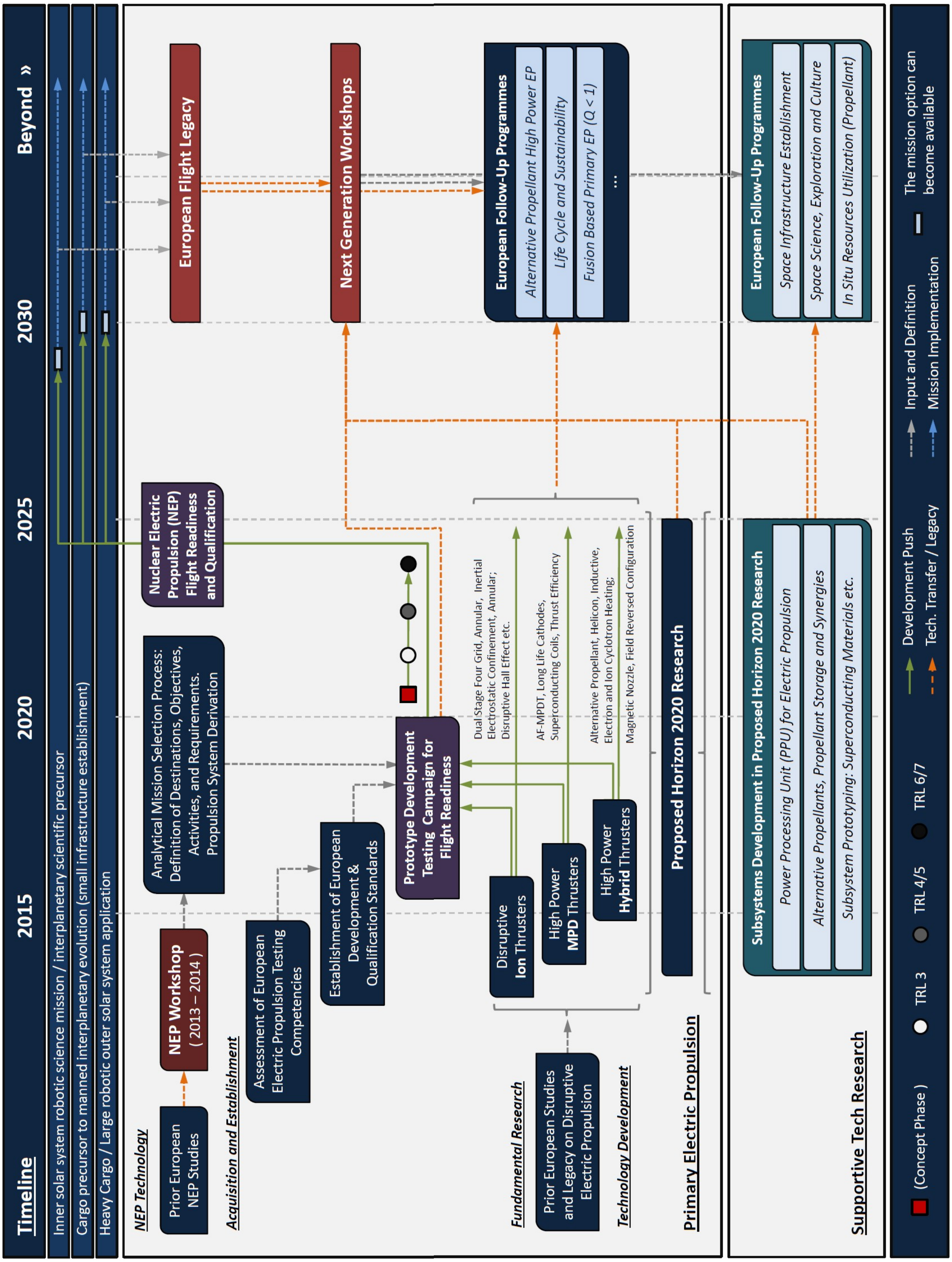


Figure 22 – Disruptive Electric Propulsion Roadmap