ADVANCED PROPULSION SYSTEMS ENGINEERING REPORT

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ABSTRACT

This report together with [RD1] and [RD2] shall give an overview about future nuclear electric and nuclear thermal propulsion and their application for interplanetary missions. In [D23.1] an overview about the three dominating electric propulsion mechanisms is given as also a short overview about successful thruster examples. Furthermore block diagrams for NEP and NTP have been defined. Out of several possible interplanetary scenarios a set of six missions has been motivated. An overview over system and mission analysis tools that are needed to investigate the suggested missions is then given. Finally an evaluation system for NEP and NTP thrusters as presented in [RD1] and [RD2] is developed. This should then result in easy to judge values of (mission) parameters in [RD3] for each suggested mission.

KEY WORDS: DiPoP, Electric Propulsion, Space Missions, Mission Analysis Tools, Nuclear Electric Propulsion, Evaluation Matrix.

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List of acronyms and abbreviations

EP	-	Electric Propulsion
EPPU	-	Electric Power Processing Unit
NEP	-	Nuclear Electric Propulsion
NTP	-	Nuclear Thermal Propulsion
OBC	-	On Board Computer
PCDU	-	Power Conditioning and Distribution Unit
TPPU	-	Thermal Power Processing Unit
TT	-	Thermal Thruster
HIPARC	-	High Power ARCjet
MPD	-	Magneto-Plasma-Dynamic
MPDT	-	Magneto-Plasma-Dynamic Thruster
AF-MPDT	-	Applied-Field Magneto-Plasma-Dynamic
SF-MPDT	-	Self-Field Magneto-Plasma-Dynamic
HET	-	Hall Effect Thruster
GIT	-	Gridded Ion Thruster
LEO	-	Low Earth Orbit
GEO	-	Geo Stationary Orbit
GTO	-	Geo Transfer Orbit
KBO	-	Kuiper-belt Objects
NEO	-	Near Earth Objects

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2 Introduction

This report together with [RD1] and [RD2] shall give an overview about future nuclear electric and nuclear thermal propulsion and their application for interplanetary missions. In this document, an overview about the three dominating electric propulsion mechanisms (electro thermal, electromagnetic and electrostatic) is given as also a short overview about successful thruster examples. More details on electric propulsion, their performance and their power processing units shall be found in [RD1]. Details on nuclear thermal propulsion mechanisms and thermal power processing units can be found in [RD2]. Furthermore block diagrams for NEP and NTP have been defined. Out of several possible interplanetary scenarios a set of six missions have been motivated briefly (NEOs, Mars science, manned missions to Mars, Mars rescue and return, Jupiter moons, Titan science). An overview over system and mission analysis tools (STK, STO, STA, Gesop & Astos, Orbiter, GMAT) that are needed to investigate the suggested missions is then given. Finally an evaluation system for NEP and NTP thrusters as presented in [RD1] and [RD2] is developed. This should then result in easy to judge values of (mission) parameters in [RD3] for each suggested mission.

3 Theory of Nuclear Powered Propulsion

3.1 Nuclear Electric Propulsion (NEP)

The following chapter gives a short overview about electric propulsion mechanisms suitable for interplanetary space flight and nuclear power sources. The generators used for these propulsion systems are described in [RD4] and [RD5].

Unlike chemical propulsion, where the maximal energy amount is bounded within the propellant (energy limitation), all electric propulsion applies additional external electric energy to the propellant. Therefore higher specific energies and higher exhaust velocities can be achieved. In this chapter introduced thrusters are well known fundamental concepts excluding hybrid and highly advanced thrusters. However, hybrid systems are respective mixtures of the following acceleration concepts.

3.1.1 Electrothermal Thrusters

Typical electrothermal thrusters are arcjet thrusters. As chemical thrusters, arcjet thrusters heat the propellant and then expand and accelerate the hot fluid medium in a divergent nozzle. But instead of using exothermal chemical reactions, the propellant is partially ionized and heated by an arc between the electrodes (anode and cathode).

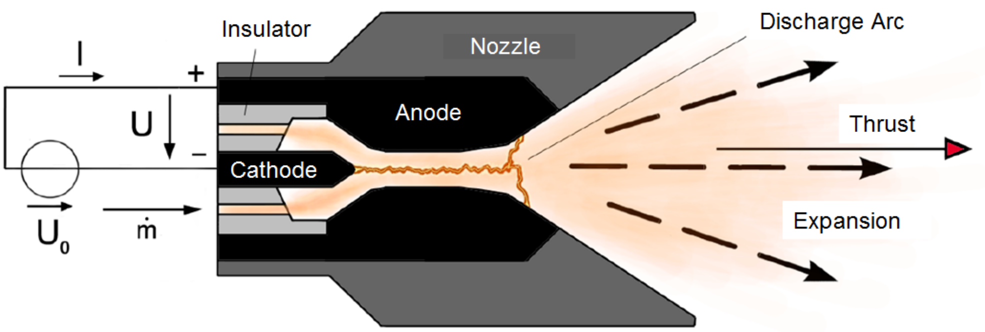

Figure 1: Schematic view of function principle of common arcjet thruster.

There is a great variety of propellant for thermal arcjets. But due to its thermal acceleration mechanism the propellant should have a light molecular weight which is e.g. the case for the hydrogen [R3]-[R5]. Because of its storage issues, chemical compounds rich in hydrogen like hydrazine or ammonia are used as alternative propellants [R4], [R5]. The thrust efficiency of arcjets thrusters is about 30 - 50 %. Up to 60 % of the losses are caused by the ionization energy and the frozen flow [R3]-[R5]. For this reason the ionization level should be minimized to achieve higher efficiency.

An arcjet thruster consists of a Laval nozzle that serves as an anode and a cathode coaxially mounted upstream of the anode. Both are typically machined out of tungsten alloys due to its high melting temperatures and they are isolated against each other by a high temperature insulator. The arc is typically ignited by a short high voltage pulse (several µs) of 1 - 4 kV. Then the voltage between the electrodes drops to around 30 - 50 V with an arc burning between the tip of the cathode and the converging section of the nozzle – this is the so called low voltage mode. In this mode the arc attaches due to high pressure in the convergent section to a single point with high thermal loads followed by significant erosion. After a short transition (some ms to s) the thruster operates in a desired state. The arc stabilizes in the entrance zone of the diverging section of the nozzle with voltages of 80 - 160 V – the so called the high voltage mode. In this mode the pressure in the divergent section drops and the arc attaches diffuse with reduced local heat load, where almost no erosion occurs.

Arcjets up to 100 kW were developed and operated under laboratory conditions. In 1999 the well-known ESEX flight experiment demonstrated an in orbit operational 26 kW arcjet with a total thrust of 2 N and exit velocity of about 7,8 km/s [R6]. Next to the American ESEX project there was only little development in the last 20 years on high power arcjets, one was an attempt of NASA to rebuild the GSC engine and the other is the development of the HIPARC thruster system at IRS with a total thrust of up to 4 N and exit velocity of up to 19,6 km/s [R7], [R8]. Aerojet qualified and operate arcjets with up to 2,2 kW for several thousand hours [R6].

The concept of the so called resisto-jets is well-known, reliable and cheap. However, due to the rather low specific impulse they do not cover primary propulsion tasks.

3.1.2 Electromagnetic Thrusters

Magnetoplasmadynamic thrusters known as MPD thrusters are typically electromagnetic arc thrusters operating at low pressures and with low mass flow rate respectively by using $j \times B$ electromagnetic forces to accelerate fully ionized gas [R3], [R4]. MPD thrusters use mostly gaseous propellants like hydrogen (H2), nitrogen (N2), helium (He), neon (Ne), argon (Ar), krypton (Kr) and xenon (Xe). Lithium is also very common as propellant due to low ionization energy of first ionization degree and high ionization of second ionization degree. However, because of contamination problems most of research teams use commonly Argon as propellant [R3], [R4], [R9], [R10].

Beside to electro-thermal thrust the following interactions occur in SF-MPD (SF = self-field) thrusters (cylindrical coordinates):

- B_θ und j_r → Acceleration force,
- B_θ und j_z → Pressure increase towards z-axis.

In addition to these interactions the following mechanisms occur for AF-MPD (AF = applied-field) thrusters due to external magnetic field:

- B_z und j_r → Hall current j_θ,
- B_r und j_θ → Acceleration force,
- B_z und j_θ → Pressure increase towards z-axis.

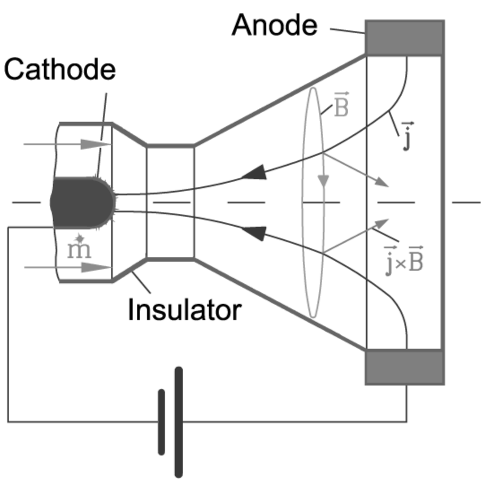

Figure 2: Schematic view of function principle of common SF-MPD thruster.

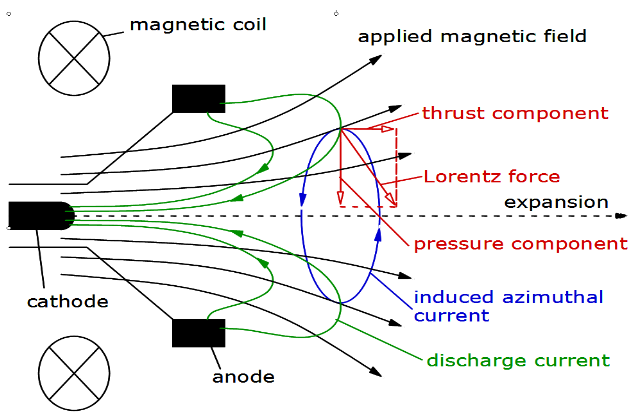

Figure 3: Schematic view of function principle of common AF-MPD thruster.

Magnetoplasmadynamic thrusters are under consideration for steady-state, quasi-steady and pulsed mode. Furthermore MPD thrusters are distinguished between SF- (Self-Field) and AF-MPD (Applied-Field MPD) thrusters. Self-field MPD thrusters consist of coaxial nozzle shaped anode and cathode in the middle, while Applied-field MPD thrusters have additionally external magnetic field generated by solenoid coil or permanent magnet allowing this way further acceleration mechanism due to interaction of induced Hall current and external magnetic field [R3], [R4].

While permanent magnets tend to be heavy and to have low operational temperature (< 200 °C) solenoid coils can have relative high losses depending on the coil size and current level. In order to increase the efficiency of AF-MPD thrusters High Temperature Super Conducting (HTSC) coils should be considered in application case [R11].

Self-field MPD thrusters require high currents in a range of more than 2 kA for electric power in some hundred kW range [R3], [R12]. This means low power SF-MPD thrusters lack of sufficient efficiency due to a too low self-induced azimuthal magnetic self-fields. By applying external magnetic fields the thrust efficiency in case of AF-MPD thrusters can be increased even at low power levels e.g. bellow 100 kW. Furthermore the required discharge current decreases to 300 - 2000 A depending on applied magnetic flux density for a 100 kW device. However, the discharge voltage increases for specific power range approximately up to 50 - 350 V compared to SF-MPD thrusters with discharge voltage of 20 - 100 V (~100 kW device).

The erosion rate (measured in mass loss per charge) of the cathode is about 0,5 ng/C to 0,2 µg/C for steady-state thrusters and 2-3 orders of magnitude larger i.e. 0,2 - 60 µg/C for quasi-steady state thrusters and the cathode is the life limited part of MPD thruster [R3], [R13]. Here numerous investigations on hollow and multichannel hollow cathode were done in order to increase the life time of the cathode [R10], [R14]. However, the multichannel systems are typically justified for the Li-driven devices only.

The arc of MPD thrusters will be ignited typically with a high voltage ramp between 500 V and some kV depending on mass flow rate and the potentially used applied magnetic flux density. High applied magnetic fields e.g. increase the initial ignition voltage. However, AF-MPD thrusters can still be ignited at low applied magnetic flux density.

SF-MPD thrusters achieve relative high thrust efficiencies up to 30 % for gas-fed steady state devices. For Li-driven devices thrust efficiencies up to 50 % are reported [R3], [R4], [R12]. The thrust efficiency of SF-MPD thrusters is strongly power dependant due to required current level as the magnetic fraction of the thrust is directly proportional to the second power of the current. This means the thrust efficiency increases with power level. The exit velocity of SF-MPD thrusters depends on mass flow rate and propellant and can vary between 10 and 60 km/s [R4], [R12]. At low power levels AF-MPD thruster can achieve higher thrust efficiency compared to SF-MPD thrusters. A state of the art gas-fed steady-state AF-MPD thrusters is DLR's X16, which reached thrust efficiency up to 39 % with exit velocity of 35 km/s at 11,6 kW discharge power, low discharge current of 80 A and high applied magnetic field of 0,6 T [R9], [R10]. In contrast to X16 a Li-fed steady-state 130 kW AF-MPD thruster at RIAME MAI institute performed thrust efficiencies up to 47 % with exit velocity up to 35 km/s at a comparably low applied magnetic field of 90 mT and high discharge current between 1700 and 2100 A [R15].

3.1.3 Electrostatic Thrusters

Electrostatic thrusters also known as ion thrusters accelerate the ionized propellant in an electrostatic field [R3], [R4], [R17]. For this reason the propellant has to be highly ionized and finally neutralized. For this kind of thrusters the propellant should have low ionization energy and high molecular weight e.g. Xenon. The three concepts of ion thrusters are Hall Effect Thruster (HET), Gridded Ion Thrusters (GIT) and the patented High Efficient Multistage Plasma Thrusters (HEMP-T) [R3], [R4], [R17], [R19].

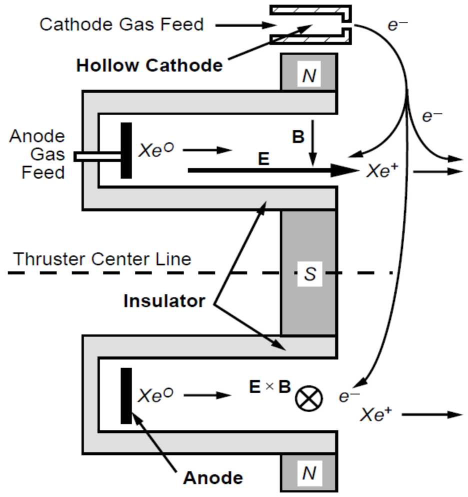

Figure 4: Schematic view of function principle of common HET thruster [R17].

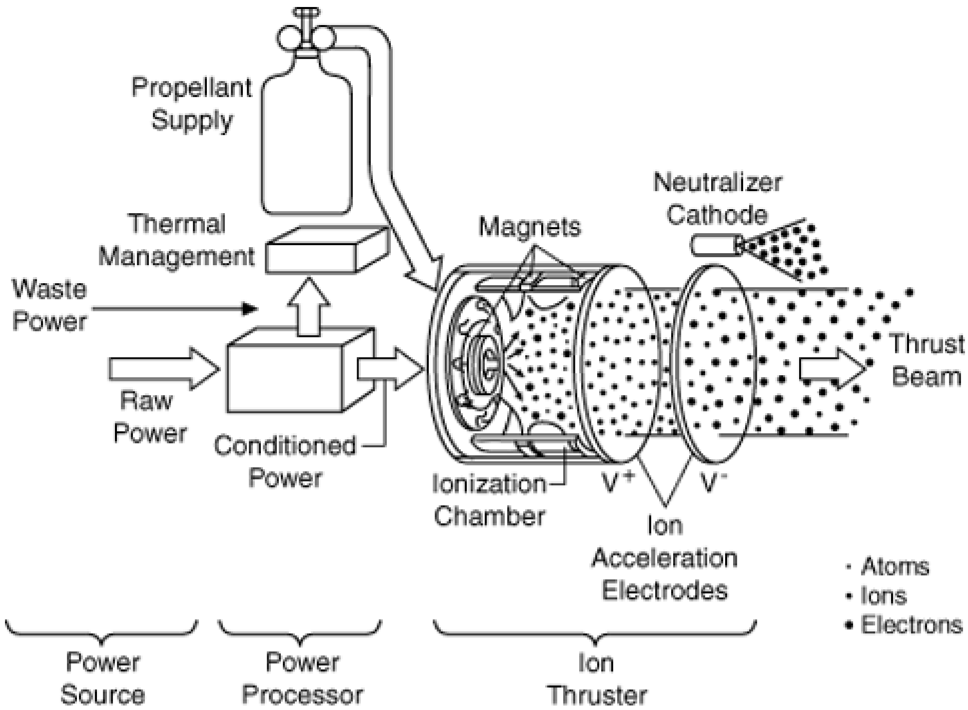

Figure 5: Schematic view of function principle of common GIT thruster [R18] .

HET consists of a cylindrical channel with an interior anode, a magnet coil that generates a radial magnet field across the channel and a cathode extern to the channel. The propellant is injected into the channel from the anode region and is ionized by the counter flowing electrons coming from the extern cathode. In the channel an axial electric field is established between the anode and the hollow-cathode plasma produced outside of the thruster channel. The radial magnetic field prevents electrons from this cathode plasma from streaming directly to the anode. Instead, the electrons spiral along the magnetic field lines and in the azimuthal direction both orthogonal to E and B around the channel, and they diffuse by collision processes and electrostatic fluctuations to the anode and channel walls. The plasma discharge generated by the electrons in the crossed electric and magnetic fields ionizes the propellant injected into the channel from the anode region. The reduced axial electron mobility produced by the transverse magnetic field permits the applied discharge voltage to be distributed along the channel axis in the quasi-neutral plasma, resulting in an axial electric field in the channel that accelerates the ions to form the thrust beam. The external hollow cathode plasma is not only the source of the electrons for the discharge, but it also provides the electrons to neutralize the ion beam. The gas density has to be low enough to ensure collisionless ion flow. As the result HETs have larger exit geometries compared to an arcjet with same power level.

HETs achieve thrust efficiencies in the 45 - 65% range with average exit velocities up to 30 km/s. The life time of Hall thrusters in terms of hours of operation amounts to over 10000 h [R16], [R17]. HETs have been tested with a variety of propellant gases such as argon and krypton, but xenon is the present standard for space applications. High power HETs with up to 96 kW were developed and demonstrated by the NASA Glenn Research Center. An exit velocity of 35 km/s, a thrust of 3,3 N and an efficiency of 58 % were achieved [R20].

A GID consists of basically three components: the plasma generation, the accelerator grid system and the neutralizer cathode. Modern GITs utilize direct current (DC) electron discharges, radio frequency (RF) discharges or microwave (ECR) discharges to ionize a large fraction of the propellant, in most cases xenon, in the ionization chamber [R3], [R4], [R17]. At the end of this chamber it is covered by a double-grid structure across which the ion acceleration voltage up to 10 kV is applied. The screen grid which comes first must extract the ions from the discharge plasma and focus them through the accelerator grid which is the second one. Thereby the grids must minimize ion impingement on the screen grid and extract the maximum number of the ions that are delivered by the plasma discharge to the screen grid surface. In addition, the grids must minimize neutral atom loss out of the ionization chamber to maximize the mass utilization efficiency of the thruster. High ion transparency and low neutral transparency drives the grid design toward larger screen grid holes and smaller accelerator grid holes. The neutralizer cathode is positioned outside the thruster and provides electrons at the same rate as the ions to avoid charge imbalance with the spacecraft. The RF skin depth problem in ionization chamber limits the plasma density, mass flow rate and thrust density, which means higher thrust and power level, can only be achieved by an increase of the thruster's outlet [R17].

GITs achieve exit velocities up to 70 km/s with efficiencies in 50 - 70 % range. The lifetime amounts to several 10000 operating hours [R17]. High Power GITs operating with 20 kW are now considered at the NASA Glenn Research Center and at the Jet Propulsion Laboratory with the HiPEP and NEXIS program [R21]. The Dual-Stage-4-Grid thruster is currently developed by the ESA. The design operating power is at 250 kW class. At low power the prototype of the thrusters demonstrated the exit velocity of 140 km/s and a total efficiency of 70 % [R22], [R23].

3.2 Nuclear Thermal Propulsion

Nuclear Thermal Propulsion is a special case of *Thermal Propulsion* which is a family of *Newtonian Reaction Engines* for propulsion in space. Their working principle is based upon the conservation of momentum. In the case of time variant system mass, they are commonly called *rockets*. A rocket is accelerated by ejecting a propellant with an exhaust velocity which is depending on the energy fed into the propellant. In the case of thermal propulsion, this consists in heat emerging from any given source of power, here: nuclear power.

A system draft of an NTP is shown in figure 6: An NTP consists of a nuclear power source – called *(nuclear) core* in this report – a heat exchange system in which the heat yield of the core is fed to a medium acting as a coolant to the core before being ejected as a propellant through a suitable nozzle. This synergetic use of a working medium is called *regenerative cooling*. In a generalised view, these subsystems are so far similar to those of other thermal propulsion systems. Other than that, NTP systematically require a shield due to the core's expected radioactivity that can be noxious to both the vessel's hardware and an eventual crew. For the latter, elevated doses of radiation constitute an important health risk. The detrimental effects of radiation may be addressed by appropriate shielding which can consist of radiation attenuating material or distance. Since there will always occur an immense flux of waste heat during propulsion – due to high mass specific power even despite high efficiency – or after-heat in *idle mode*, i.e. when no propulsion is performed, a heat flux sink needs to be implemented, called *Thermal Power Processing Unit* (TPPU). The TPPU can both be employed to harvest the heat flux to generate electrical energy for the space craft, and to dump the ultimate waste heat via radiators to space.

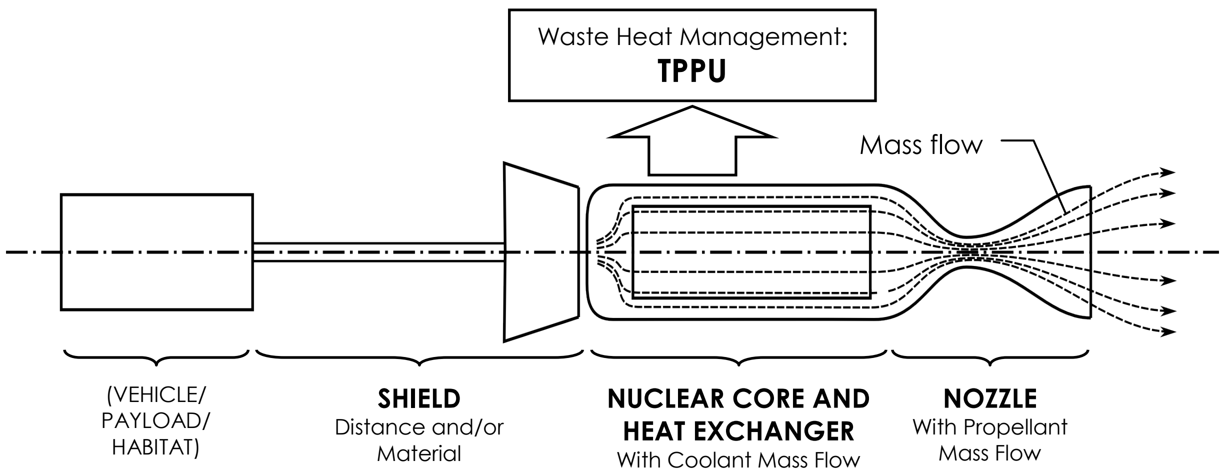

Figure 6: Schematic presentation of Nuclear Thermal Propulsion Systems.

The basic motivation for the consideration of nuclear thermal systems consists in the eventually indefinite raise in mass specific power they can enable. The benefit of doing so can be derived from fundamental rocket equations. One important space flight parameter is the velocity increment Δv.

To learn more about NTP, please refer to [RD2].

4 System Architecture Analysis

For further investigation in terms of conceptual modelling overview system architecture for different spacecraft propulsion concepts should be defined and clarified. At this point the distinction between NEP and NTP propulsion concept takes on a system plane more detailed shape. Figure 7 shows the overview of possible propulsion concepts for the next decades and the scope of DiPoP project. The more detailed system architecture is given in each specific system block diagram. Additionally a combination of both concepts is also possible allowing this way high thrust with thermal thruster and high exit velocities with electric propulsion.

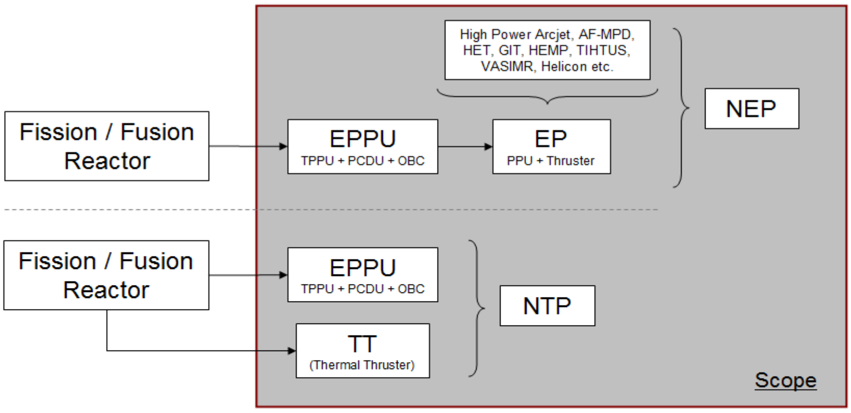

Figure 7: Simplified schematic overview of propulsion concepts SEP, NTP and NEP.

4.1 System Block Diagram NEP

The NEP concept consists of fission or fusion based nuclear reactor, electric power processing unit (EPPU) and electric propulsion (EP). Figure 8 shows the further details of NEP concept in a block diagram.

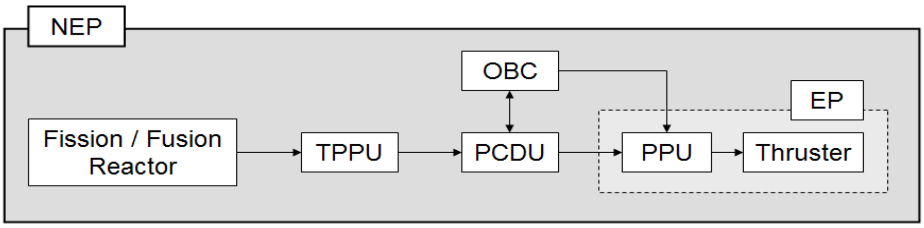

Figure 8: System block diagram of NEP concept.

The EPPU include thermal power processing unit (TPPU) or converter, power conditioning and distribution unit (PCDU) connected to the on board computer. Depending on clustering number, redundancy of EP and also on size and weight of power processing unit (PPU), the number of thruster specific PPUs could be as high as number of thrusters. For further investigations more detailed characterisation of propulsion concept is required.

4.2 System Block Diagram NTP

The NTP concept simplifies the consideration from the systematic point of view to nuclear reactor and respective thermal thruster. To provide the crew's life support systems and the spacecraft with power a TTPU and PCDU will be required leading to bimodal system (see Figure 9). In case of NTP requirements on EPPU are given by life support systems and subsystems of the spacecraft.

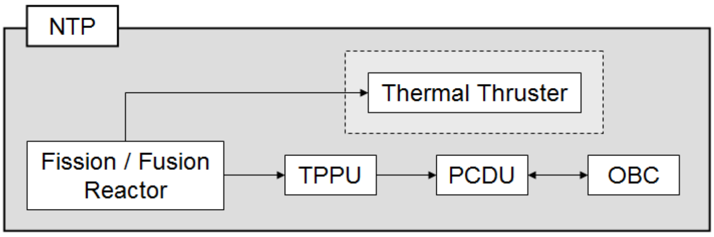

Figure 9: System block diagram of bimodal NTP concept.

5 Proposed Missions

To guarantee the success of specific missions, objectives must appeal to public, political leaders and also they must be essential for scientists with respect to long term plans and to sustainability.

Moreover, future needs and possibilities for humanity should be covered [RD6]. However the biggest limitations are given through mission costs, available electrical power sources for space structures and vehicles which lead us to limited and costly space launch capabilities. Moreover, a certain group of missions e.g. the ones that make use of nuclear power systems need to be accompanied by a strong risk minimization and safety approach [RD6].

One of the current heavy launch transport systems which are 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 [R24]-[R26]. This means interplanetary spacecrafts with payload masses above 7 t need to be assembled in LEO and then boosted with a separate interplanetary kick stage in order to enable respective missions that require payload masses significantly above 7 t. Therefore, the whole mission spacecraft should consist of predefined modules of less than 20 t with low risk, standardized docking ports and subelements.

One of the main criteria of possible missions is the distance to the object from Earth and the purpose of the mission. Further definitions and distinctions of mission scope lead to more detailed requirements for the mission spacecraft. Some of characteristic criteria are listed below:

- distinction between manned and automatic mission,
- definition of transfer time (very important in case of manned missions),
- definition of mission payload.

In this chapter an overview of possible mission scenarios is given. Furthermore the distances to the objects and further requirements on exemplary NEP/NTP systems were presented. Here for a very simplified consideration Hohmann and spiral orbit transfer were assessed in order to compare requirements of total Δv for specific missions. Furthermore additional information found in literature of extensive mission analysis is also presented and added.

5.1 General Assessment

The advantage of the nuclear power system above the solar power system depends on distance of mission target from the sun. Already beyond Mars the solar constant E_e drops below 50 % compared to the value of 1360 W/m² in Earth orbit [R27], [R28]. 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 specific power of solar arrays and nuclear power systems defines the transition distance and selection criteria between SEP and NEP systems for specific mission scenario. Table 1 shows the decreasing power of solar power generators especially from the asteroid belt and beyond with less than 10 % of the Earth solar power generation. Additionally specific power for nuclear and solar power generation is shown in Table 2. By this data a break even power was defined in Table 1 using a 75 W/kg solar power generator as norm. This break even power can then be interpreted as a minimum thrust for this mission depending on the used propulsion system with typical specific thrust values of 20 - 100 N/MW[R5].

Mission target	Distance from Sun / AU	Norm. power / -	Break Even Power kW	Min. Δv / km/s	Spiral Δv / km/s	Transfer Time / d	Transfer time – real / d
Mercury	0,39	6,6	-	17,0	17,9	105	146 (Flyby) 2400 (Orbit)
Venus	0,72	1,9	-	5,3	5,3	145	150-200 (Orbit)
Earth	1	1	-	-	-	-	-
Mars	1,52	0,43	> 200	5,6	5,6	260	80 d (Flyby) 210 d (Orbit)
Asteroids	2,80 (2-3,4)	0,13	> 30	11,3	12,0	480	1400 (Orbit)
Jupiter	5,20	0,037	> 3,5	14,4	16,7	1000	400 (Flyby) 2000 (Orbit)
Saturn	9,54	0,011	> 0,5	15,7	20,1	2200	1100 (Flyby) 2450 (Orbit)
Uranus	19,2	0,0027	> 0,5	16,0	23,0	5850	2000-3000 (Flyby)
Neptune	30,1	0,0011	> 0,5	15,7	24,4	11200	4400 (Flyby)
Pluto	39,5	0,00064	> 0,5	15,5	25,0	16600	3500 (Flyby)
Eris	67,7	0,00022	> 0,5	15,0	26,2	36700	-

Table 1: Properties of possible mission targets.

Power generation type	el. Power kW	Power density W/kg	Power density (norm.) -	Reference
Solar electric (typical)	1	30	0,4	IRS,FLP
Solar electric (concept)	5.000	75	1	[R29]
Solar thin film (concept)	5.000	100	1,33	[R29]
Solar electric (SLA)	3,75	375	5	[R30]
SNAP-10A	0,5	1,1	0,015	[RD4]
Bouk	3,5	3,8	0,05	[RD4]
Topaz-1	6	6	0,08	[RD4]
SNAP-8	30	7,5	0,1	[R31]
SP-100	100	18,5	0,25	[RD4]
(concept)	200	25	0,33	[R32]
Brayton (concept)	5.000	75	2,5	[R29]

Table 2: Overview over electric power densities of different power sources.

5.2 Possible Missions

5.2.1 Mercury / Venus

Mercury and Venus missions are not considered in this report as the solar power generation due to the low distance to the sun is far superior to nuclear power generation is out of scope of this document. Note however, that solar power units experience an increased degradation due to solar particle emission.

5.2.2 Earth / Moon

There are several possible missions in earth orbit where solar electric, nuclear electric and nuclear thermal propulsion is intended for maximizing payload weight of communication satellites, telescopes and other scientific missions. A reusable transfer stage between low earth orbit and geo stationary orbit with standard qualified docking ports is one of the favourite ideas. The same counts for transfer missions for building up an infrastructure for space exploration for example by establishing the Moon base.

However the public acceptance of a fleet of nuclear powered transport systems in low and medium Earth orbit will be very difficult and the solar power generation is still a good alternative with similar or even higher power densities. Thus nuclear powered vehicles for earth orbit are neither recommended nor considered [R33].

5.2.3 NEO (Near Earth Objects)

Near earth object (NEOs) or near earth asteroids (NEAs) are a very interesting mission goals. On scientific perspective their composition is similar to objects created at the birth of our solar system thus holding more information on its creation process. Additionally there are several of them classified as potentially hazardous asteroids (PHAs) which cross earth orbit and once may hit the earth. This even increases the need to investigate the characteristics of the asteroids. Knowledge about their composition and structural integrity is needed for development of proper counter measures to deflect or destroy PHAs.

Furthermore the low distance with a maximum perihelion distance of 1,3 AU results in low velocity increments and low mission duration. Thus NEOs may be a first step for human space flight out of the Earth-Moon system before moving to the Mars. Especially for human missions propulsion systems with higher thrust and thus higher power budgets are required. However due to the low distance from Sun, NEO missions still prefer solar electric propulsion over nuclear powered propulsion concepts.

As missions to near earth objects are considered as next step before reaching to the mars, a short term mission with no need of LEO clustering is recommend reducing the maximum mass to less than 7 t.
Table 3 shows an example mission derived from [R34] to “1999 AO10” with a total mission velocity increment after injection into interplanetary orbit of 4 km/s, a total mission duration of 121 days with 7 days of stay at the asteroid. Due to the orbit of 1999 AO10 this mission may be accomplished in the year 2025.

Target	Thrust / N	Δv / km/s	Travel time / d	Initial mass / t	Sun distance / AU
1999 AO10	(2,5)	4	114	7	0,91

Table 3 : Example NEO mission

5.2.4 MARS

Numerous mission scenarios to Mars have been proposed: the transport of orbiters, landers and rovers, a sample return mission, a manned orbiter with robotic landers, the first manned Mars base, the following transport missions to and from this base and so on. Latest mission have been Mars Global Surveyor, Mars Odyssey, Mars Express, Mars Exploration Rover, Mars Science Laboratory, ExoMars etc. Each of these mission scenarios will benefit from a propulsion system with increased exit velocity enabling higher payload masses or faster transports at same payloads. Due to the increasing distance from Sun nuclear power propulsion get attractive for a mission to Mars requiring constant high power in Mars orbit. This is especially true for manned or heavy transport missions requiring several Newtons of thrust. A more detailed analysis on the thrust to velocity increment ratio and the specific route to mars has been done by [R29] and is shown in the tables below and figure 10. These investigations are used as baseline for three different reference missions to the Mars.

The first mission is be a robotic mission as the scientific missions flown before with a spacecraft of less than 7 tons of initial mass carrying orbiters, landers and/or rovers. The transfer time is not critical and may be in the same order as for the Hohmann transfer.

Target	Thrust / N	Δv / km/s	Travel time / d	Initial mass / t	Sun distance / AU
Mars - robotic	(1)	5660	365	7	1,52
Mars - manned	100	6710	(123)	140	1,52
Mars - rescue	(64)	9130	93	30	1,52

Table 4 : Overview of different requirements for specific mars mission scenarios.

Thrust level @ 140 / 30 t N	Trajectory type A			Trajectory type B		
	C-A			C-B		
	Velocity increment m/s	Travel time d	Burn Time D	Velocity change m/s	Travel time D	Burn time d
Impulse	5600	>250	<1	-	-	-
Spirals	5660			-	-	-
50 / 10,7	5430	158	110	8560	320	185
100 / 21,4	6710	123	70	7580	252	80
300 / 64,3	9130	93	33	11690	171	50
500 / 107	10740	83	24	9100	168	20
Direct	11500					

Table 5: Velocity change requirements for different travel times and constellations to Mars for a 30 or 140 t vehicle.

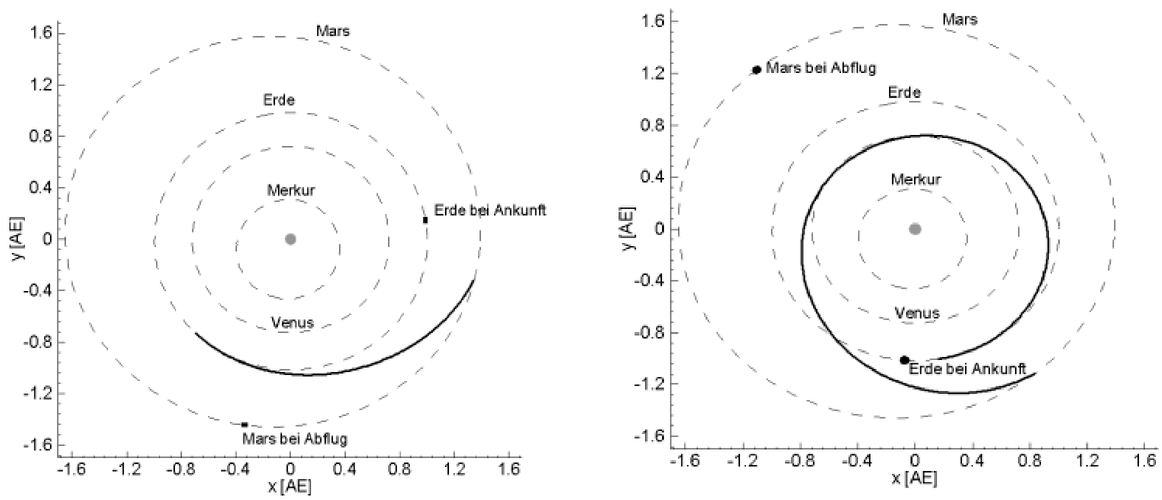

Figure 10: Example trajectories to mars depending on earth mars constellation
(left – Constellation C-A, right Constellation C-B) [R29].

The second mission is a crew and cargo transport mission for a Martian base. This has been assessed by Schmidt with an initial mass starting from interplanetary orbit of 140 t considering especially the habitat, life support and power supply. Thrust levels of 50 to 500 N were investigated for different earth-mars constellations resulting in different velocity increment requirements and different transfer times. As reference a one way mission on trajectory type A with a thrust of 100 N and a transfer time of 4 months has been chosen.

A rescue mission derived from the same study serves as third mission. Here highest thrust ratio is required to get humans back from mars as quick as possible. Because of this the velocity increment from increases above 9 km/s while the travel time reduced to 3 months with an initial mass of the return vehicle of only 30 t.

5.2.5 Jupiter and Saturn Moons

There are several mission scenarios to the Jupiter's and to the Saturn's moons. The Jupiter is orbited by 66 conformed moons and Saturn by around 62 moons that open many doors for exploration and also for understanding of geological and/or astronomic aspects in early age of Solar System.

One of the exemplary reference missions was performed by Galileo spacecraft sent by NASA in 1989 to study Jupiter and its moons with a total mass of 2230 kg including a descent module with mass of 339 kg for the entry in Jupiter's atmosphere [R35]. The flight path of Galileo to Jupiter's orbit was a VEEGA (Venus-Earth-Earth Gravity Assist) a multiple flyby manoeuvre.

Another a joint NASA/ESA/ASI spacecraft mission is Cassini Huygens that was launched in 1997 and is studying the Saturn and its natural satellites since 2004 [R36]. With the launch mass of 5712 kg including orbiter, propellant and 320 kg heavy Huygens descent probe the spacecraft was sent via VVEJGA (Venus-Venus-Earth-Jupiter Gravity-Assist) trajectory towards Saturn and its moons. As in case of Galileo such gravity assist flight path for Cassini Huygens spacecraft was required in order to acquire enough speed to reach Saturn with the limited launch mass. Otherwise the spacecraft wouldn't have enough propellant left over to allow braking manoeuvres for orbit insertion around the Saturn.

These introduced exemplary reference missions would benefit from electric propulsion and nuclear power generation systems. Due to very low solar constant with a fraction of 1-3 % relative to solar constant given in Earth orbit at distances between 5,2 AU and 9,58 AU to the Sun the NEP concept is superior compared to SEP. With nuclear power generation higher power output will be possible allowing providing the spacecraft or electric propulsion with electrical power in a kW range. By using of electric propulsion with high exhaust velocity the effective payload of scientific spacecraft can be increased. For this approximately required Δv to Jupiter are about 14,4 km/s with Hohmann and about 16,7 km/s for spiral orbit transfer. With Saturn as target destination a Δv of 15,7 km/s will be required for Hohmann and about 20,1 km/s for spiral orbit transfer.

As example a science probe mission (7 t) to Jupiter with a velocity increment of 16,7 km/s was chosen. To get an enhancement over general swing by manoeuvres with travel times of around 2000 d the travel time for this mission is set to 1000 d.

Target	Thrust / N	Δv / km/s	Travel time / d	Initial mass / t	Sun distance / AU
Jupiter	(0,5-1)	> 16,7	1000	7	5,2

Table 6 : Example Jupiter mission.

5.2.6 KBO (Kuiper-Belt Objects)

The extension of Kuiper-belt is about 30-55 AU. There are two different types of KBOs such as resonant objects (twotinos and plutinos) like Pluto, which are locked in orbital resonance with Neptune, and classical Kuiper-belt objects (cubewanos) moving almost in circular orbits like Quaoar. One of possible KBO missions could be to planetoid Pluto but also Neptune's moon Triton is one of possible mission targets of Kuiper-belt objects.
In 2006 a New Horizon spacecraft was sent by NASA with fly past planet Pluto and is after JGA (Jupiter Gravity Assist) approximately in a halfway between Earth and Pluto on heliocentric trajectory after 6,3 years (see Figure 11). The total flight time will be about 9,5 years or 3462 days. The goal of this mission is to study Pluto and its moons Charon, Hydra and Nix.

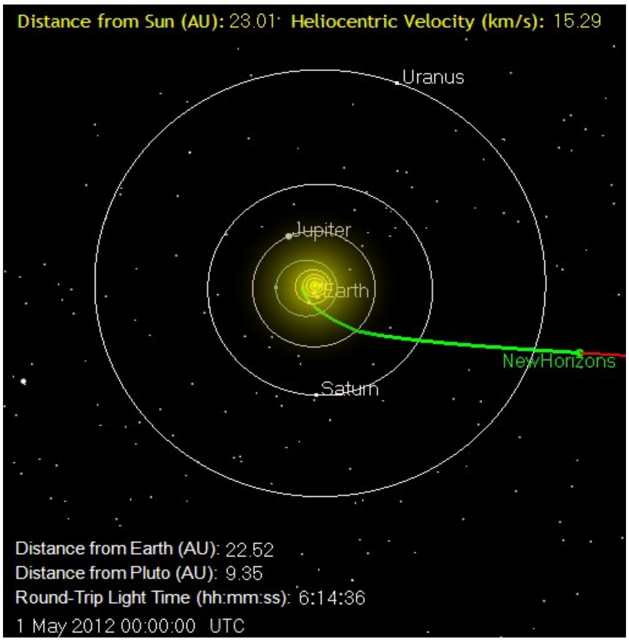

Figure 11: Current position and trajectory of New Horizon spacecraft (01.05.2012) [R37].

As in case of Jupiter and Saturn all KBOs are too far away from the Sun making SEP concept less viable compared to NEP. For Hohmann orbit transfer a minimum Δv of 15,5 km/s will be required and in case of spiral orbit transfer about 25 km/s. Electric propulsion could decrease travel time or increase scientific payload allowing this way more scientific instruments for exploration.

As example a science probe mission (7 t) to Triton with a velocity increment of 25 km/s was chosen. For a realistic operational mission the total mission transfer time is assumed as less than 10 years.

Target	Thrust / N	Δv / km/s	Travel time / d	Initial mass / t	Sun distance / AU
Triton	(0,2-0,5)	> 24,4	3650	7	30,1

Table 7 : Example KPO mission.

5.3 Kick Stages

For each mission only the transfer from earth-sun orbit to target-sun orbit are considered. The transfer from earth to an interplanetary orbit is not considered for NEP or NTP and is done by chemical kick stages. However a comparison between chemical boosters and nuclear thermal propulsion should be assessed in future.

5.4 Mission Overview

Table 8 shows an overview over the requirements of the missions chosen in chapter 4.

Mission	NEOs (Human)	Mars			Gas Giants (Jupiter)	KPO (Triton)
		Science	Human	Rescue Return		
Distance / AU	0,91	1,52	1,52	1,52	5,2	30,1
Delta v / 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 8: Overview of mission requirements for the chosen missions.

Considering power density up to 75 kg/kW of advanced solar electric power generation such as SLA and relative high solar constant at distances up to 1,5 AU the SEP concept offers a viable safe simple propulsion concept. The NEP powered spacecraft became more advantageous at distances approximately beyond 1,5 AU. In case of NTP concept the travel time may play an exceptional role allowing this way fast rescue and return missions for manned spacecraft due to high thrust levels. An optimized combination of NEP and NTP for specific mission and also by considering synergy aspects of subsystems could be also viable even at distances beyond 1 AU due to high exit velocities or high thrust levels.

6 Mission Analysis Tools

This subsection assesses a selection of software (SW) tools that can be utilized as part of a method to support a systematic approach to mission design for NEP and NTP. In the first part the tools are introduced and discussed with respect to their features. After that the second part describes approaches for the selection of suitable tools to meet the demands of mission design within the scope of this study.

The referred to method generates and optimizes a mission trajectory from a sequential routine, i.e. definition, integration, iterative optimization, evaluation and visualization. The method itself will hereinafter be referred to as a tool, albeit in fact often one or more SW-sub-tools are combined to form the design tool that translates into the method. The goal is to determine flight trajectory solutions to best fit the various requirements arising from economic, technical, political, social, scientific, astronomic and ecologic factors. These factors represent design drivers. Therefore, they not only shape the mission but also the design tool as an instrument of achieving them. In simple terms, an ideal mission trajectory must found from the pool of all feasible trajectories by means of an extraction process. The quality of a tool however goes even further and is benchmarked by attributes that can be considered a measuring tape, as only few of the available tools actually go the full distance in reality. The attributes are:

- Calculation running time
- Mission scenario input time
- SW-Modularity
- HW/SW-Platform independence
- Calculation Accuracy
- Price
- Tool availability and support
- Data exchange versatility

Due to contradicting resource demands, these attributes are always subject to a design trade-off with respect to the tool and its inherent functions. A breakdown of the aforementioned factors and attributes reveals a huge number of functions for potential implementation into an applicable mission trajectory tool.

To illustrate the overall tool-framework structure derived from the requirements, the schematic in

Figure 12 shows a breakdown of the method into functional blocks. The framework is not limited to a single tool, but often represents a construct of multiple sub-tools which can share the framework. This is not accounted for in the

Figure 12 as the sub-tools can penetrate inside of the functional blocks. Also note that processes of multiple sub-tools may be running sequentially or in parallel at any given time. However, complexity is added to the framework in many ways. The arrows between the blocks signify the flow of information up to the point of data output and visualization. Groups of blocks are highlighted by coloured background to express the modularity of the framework. Some tools offer “plug and play”-modules to increase benefits from this structure. The background colour indicates whether the included blocks are usually part of the basic module, of extension modules or of both.

Scenario Definition: This represents the interaction of the design-engineer with the trajectory tool for input of the mission scenario and constrains. Today the common form for comfortable data input is an intuitive graphical user interface based on a Windows environment. It usually offers point and click selection capability, easy access tabs and options as well as context sensitive input fields. In some cases this is complemented by wizards to guide through certain stages of data input or import from external sources.

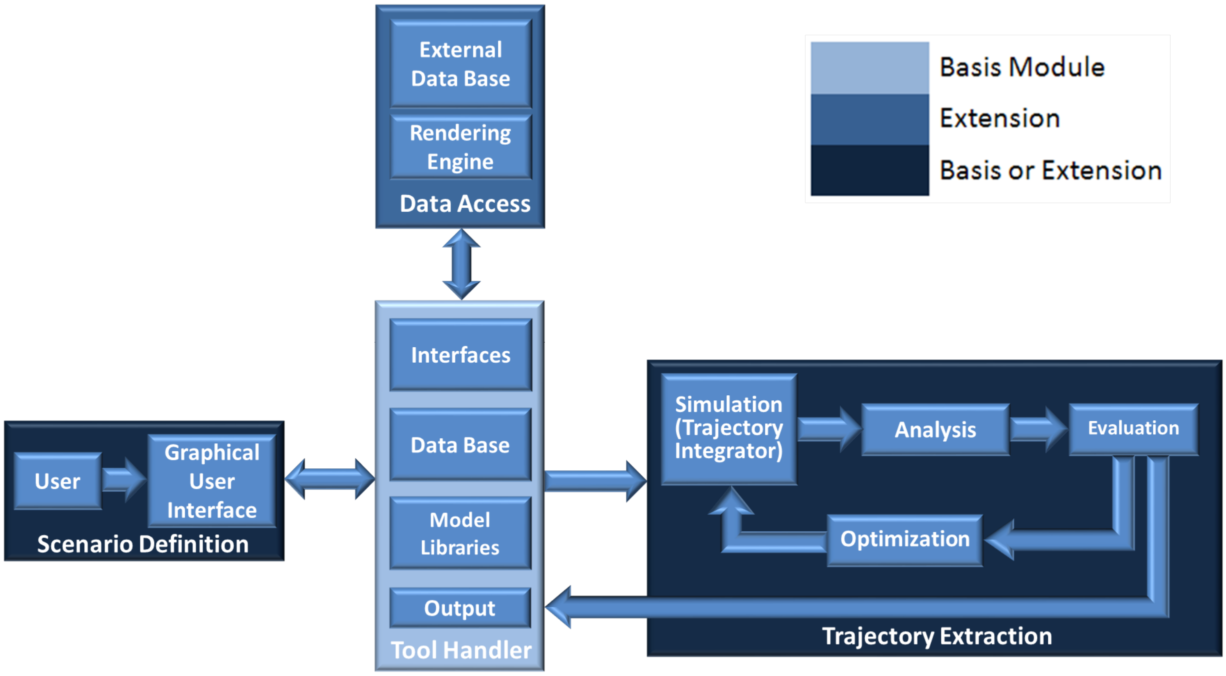

Figure 12: Trajectory Design Tool Framework.

Tool Handler: This module stores all the required scenario information for the simulation of the desired mission phases like ascent, travel manoeuvres and idle, descent, landing, swing-by and fly-by to name a few. The boundaries specified herein limit the pool of all possible associated trajectories. Also SW-interfaces and external output capabilities are provided. The output can be linked to external sources, e.g. in the form of a certain library or file-type for data exchange or external processing. Important parts of the handler are the model libraries. They basically contain a complete scientific description of the laws of movement and interaction of different types of bodies in the solar system.

Trajectory Extraction: The trajectory extraction is realized by an iterative routine. At first, the simulation combines all available data on the mission, scientific models, boundary and starting conditions, optimization target values and exit criteria etc. to numerically compute a trajectory. This trajectory is simply an enormous set of data including all computed parameters for each time step of the calculation. The trajectory is then analyzed to calculate the optimization target values. A subsequent evaluation process confirms if the acquired solution represents a feasible trajectory and if it matches the exit criteria. If either of these tests fails, the process will optimize

the starting conditions within the predefined boundaries and restart the simulation to create a new trajectory. A lot of sophisticated intelligent optimizing approaches exist with varying efficiency, even including neural networks. The general rule is that the number of required iterations rises with the complexity of the mission scenario.
Data Access: This includes links to 3D-animation rendering engines for graphical visualization of the trajectory data set. The capabilities of the engines normally allow for a very realistic access to the results and thorough data analysis by the design engineer. Sometimes this capability is available even during trajectory extraction for preliminary trial runs and debugging purposes. Access to the data set can also be possible through external data base tools with extended visualization capabilities.

However, it should be noted, that the number of tools and methods available today is as high as the sum of approaches, internal processes and user/customer specific demands on the market. This induces functions that are unique to single tools only and complicates a tool comparison. Further, no standards for mission analysis tools are available whatsoever. The investigation of a generalized approach for a universal and international applicable design method would be beyond the scope of this study. The common workaround is to create cross platform tools and provide SW flexibility by means of interfaces. The term cross platform refers to methods that are implemented and inter-operate on multiple computer platforms like the Windows x86 architecture or PowerPC.

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 is important, they are rather scarcely spread in the academic sector. On the other hand, the quality of a self-built tool is often given 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 9 already shows the selection of tools that will be presented in detail in the following chapter. 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 programming language is important in terms of extensibility if customer specified demands are not yet considered in the tool database or model libraries. Commercial software extension abilities in this field are usually more limited than in case of open source software. Moreover the integrability of different databases selectable by the customer is a factor that can be taken into account. User friendly interfaces e.g. the GUI as the input window and the rendering machine as

graphical output window are considered here as well. But after all the software readiness level is usually the driving criterion for the selection process. Level I indicates a fully developed COTS software that already includes a huge variety of models and databases to select and that is applicable as it is. If the analysis tool is fully developed by the supplier but still can be extended and fitted to the customer specific requirements level II is appointed. In the case of mission requirements or customer demands that can't be fulfilled by level I or level II software tools the customer has to develop an analysis tool tailored to a specific problem. Depending on the time frame, financial resources and mission requirements any of the mentioned criterions in Table 9 may dominate.

Tool	Origin	Source	Prog. Language	Database	Rendering Engine	GUI	Readi-ness 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 Definition	
I	User level interaction only with full tool function coverage and company support, maybe 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 9: Analysis Tool Overview.

6.1 Analysis Tool Options

6.1.1 STK (Satellite Tool Kit)

Satellite Toolkit (STK) [R38] is a software framework by Analytical Graphics, Inc.1 which has been designed to solve complex dynamic problems in aerospace. It can be used for analysis in the field of orbits, trajectories, thermal (single node only), communications, radar, radar jamming, debris, radiation and others.

At the core of STK is a geometry engine that is designed to determine the time-dynamic position and attitude of a subject, determining dynamic spatial relationships among all of the objects under consideration including the quality of those relationships or accesses given a number of complex, simultaneous constraining conditions. STK has been developed as a commercial off the shelf software tool. It is widely used by NASA and industry for the following purposes:

- Shuttle Mission Control to show the attitude and position of the Shuttle in real time while on orbit.
- Visualization of deep space missions.
- Earth orbital mission analysis.
- by industry for analysis and visualization of proposed satellite constellations
- high fidelity images and simulations of spacecraft and orbits, Orbit determination and visualization, Spacecraft attitude determination and visualization, Real time trajectory visualization from the output of trajectory optimization programs, Orbit interaction analysis and visualization.

The size and the capabilities of STK require some time in to learn and understand the concept and the possibilities of this powerful, yet expensive (50 - 90 k€ depending on license), software suite. Although it is a closed source commercial program, it has interfaces allowing for integration into customer-designed products. The scripting interface named Connect is language independent and enables STK to act within a proprietary Transmission Control Protocol (TCP)/Internet Protocol (IP) socket or a Component Object Model link, requiring the STK/Integration add-on package.

The interface to STK is a standard GUI display with customizable toolbars and dockable maps and 3D viewports. Each analysis or design space within STK is called a *scenario*. Within each scenario any number of satellites, communications systems or other objects can be created. Each scenario defines the default temporal limits to the child objects, as well as the base unit selection and properties. All of these properties can be overridden for each child object individually, as necessary. For each object within a scenario, various reports and graphics (both static and dynamic) may be created. Relative parameters, between one object and another can also be reported and the effect of real-world restrictions (*constraints*) enabled so that more accurate reporting is obtained. Through the use of the *constellation* and *chains* objects, multiple child objects may be grouped together and the multipath interactions between them can be investigated.

STK Basic is free to all users and is also the core module for all other STK modules. It allows access calculations to be performed between satellites and fixed points on the Earth's surface (or between satellites). There is also the ability to import satellites from the NORAD public satellite database (which can be updated online from within STK). As of STK 8.0, users of the free version can also publish their scenarios to the VDF format. This means that scenarios can then be opened using the free AGI Viewer product.

One of the essential modules that need to be obtained in case of a mission analysis approach with STK is the Astrogator. This module offers the following features:

- Finite and impulsive burn modelling
- Targeted manoeuvre sequences

- Fuel-usage calculations
- Sophisticated event detection capability
- Deep space mission design and analysis
- Industry-standard atmospheric density models-exponential, Harris-Priester, Jacchia 1971, Jacchia-Roberts, Cira 72, MSIS, US Standard
- Custom central bodies and force models
- High precision numerical integrators Runge-Kutta-Fehlberg, Runge-Kutta-Verner, Burlisch-Stoer, Gauss-Jackson
- Parameter optimization

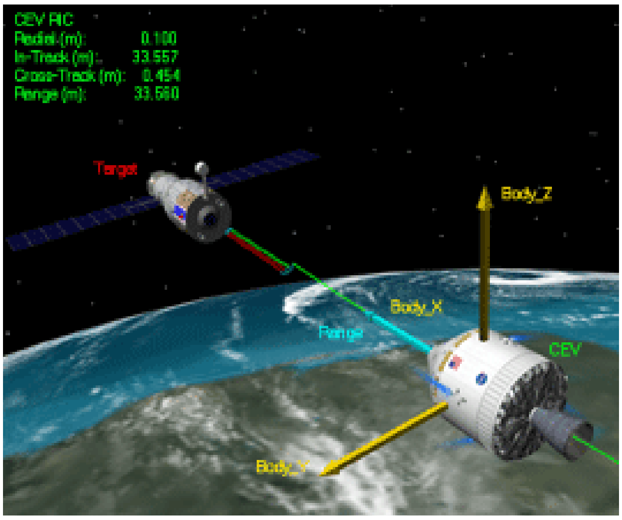

Figure 13: STK Astrogator sample screen shot.

6.1.2 GMAT (General Mission Analysis Tool)

GMAT is a software system for trajectory optimization and mission analysis. Analysts use GMAT to design spacecraft trajectories, optimize manoeuvres, visualize and communicate mission parameters, and understand a mission's trade space. The GMAT project is a collaborative effort between NASA, the space community, and the open source community. To maximize technology transfer GMAT is open source and is licensed under the NASA Open Source Agreement. This tool is intended both for real-world engineering design studies and as a tool for education and public engagement in the spirit of the NASA Charter.

The current release of GMAT is a snapshot of the system halfway through the current development cycle. GMAT was released at this time to begin the technology transfer process and to foster new collaborative relationships. While GMAT has undergone extensive testing and is mature software, the system is to be considered in beta form on some platforms and alpha form on others. The GMAT team considers the system to be in an alpha release state as long as the system is functional, but has not been rigorously tested for numerical precision and functional stability. Once testing has been performed and stability verified, the system becomes a beta product if the number of open issues is considered small and the system is usable for most problems [R39]. GMAT has the following status on the supported platforms:

- Windows: Beta version.
- Mac: Alpha version
- Linux: Alpha version

Since this tool is still under development and testing, the following explanations concerning structure, design and components are not yet all implemented.

There are several high level requirements for GMAT that drove the design of the system. These requirements can be summarized in five broad categories: MATLAB Accessibility, Extensibility, Formation Modelling, Parallel Processing, and Open Source Availability. The system is designed to run on Macintosh, Windows, and variants of Unix (including Linux) through a recompilation of the source. GMAT, by design, performs detailed orbit and attitude modelling, providing an engine that can be called from MATLAB for tasks that present performance issues when built in the MATLAB language. One prime driver for the development of GMAT was to provide a tool that allows users to try new components and models in the system without rebuilding it from scratch. This capability is partially satisfied by the MATLAB interface described above. Components of GMAT can also be added to the system by writing new code that can be compiled into shared libraries and incorporated into the system at run time. All of the operating systems GMAT supports provide native methods for this capability, and the system is designed to make the addition of new components simple using these capabilities.

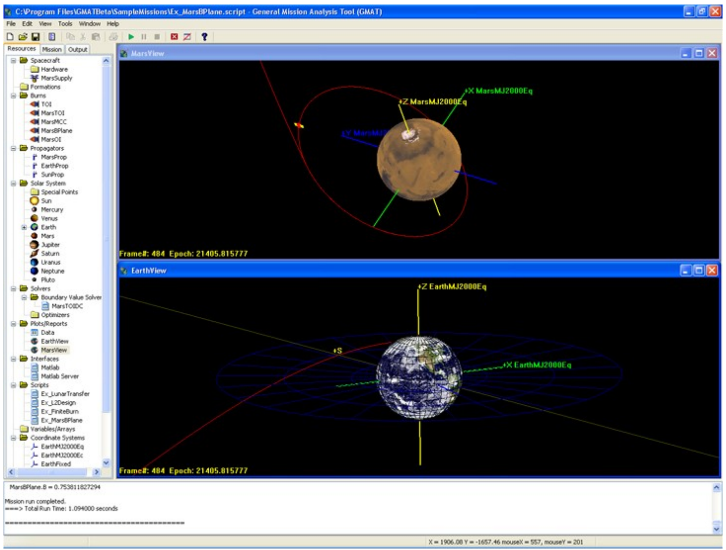

Figure 14: GMAT GUI.

Due to the beta status of the software no reliable information can be obtained from it as of yet. However depending on tool support, future reference applications and the implemented degree of thruster, system and mission configuration flexibility of the trajectory integrator it could be feasible as a cheap and more available addition to the STK-software-suite.

6.1.3 GESOP & ASTOS

ASTOS Solutions [R40] offers solutions for many different applications like feasibility studies, mission planning & analysis, reference trajectories, performance calculations, vehicle design, safety & risk assessment, verification and validation. The proposed software used in the trajectory optimization field is GESOP, which stands for Graphical Environment for Simulation and Optimization. It is “a software system for trajectory analysis and design of user defined, controllable dynamic systems governed by a set of ordinary nonlinear differential equations, associated boundary conditions, path constraints and cost functions” [R41]. GESOP itself is not really a space trajectory optimization program; rather, it uses an application called ASTOS (Aerospace Trajectory Optimization Software) for the actual trajectory optimization, which supports mission types as launcher, reentry, in-orbit and interplanetary missions. If the wanted mission is not included in ASTOS it is possible to program an own optimization problem in Fortran 77, C, Ada 95 or Matlab scripts.

In-orbit and interplanetary mission profiles could be relevant for DiPoP. A typical in-orbit problem is a transfer from the geostationary transfer orbit (GTO) to the geostationary orbit (GEO). One possibility to do so is the use of an electric propulsion system. In this case the transfer can last up to several months. During the transfer the spacecraft is spiraling from the GTO to the final GEO in dozens or even hundreds of revolutions. Another in-orbit application are close proximity operations at small asteroids that are orbiting in a photo-gravitational orbit, hovering, flyovers and landings. Services provided include general feasibility, delta-v and reference trajectories. For interplanetary applications impulsive or low thrust transfers, general transfer opportunities (for impulsive and low thrust transfer) with genetic algorithms and analysis of launch windows are considered. Moreover vehicle design adjustments due to mission studies that consider mission objectives can be done. Sensitivity analyses will provide information about necessary system modifications. Various subsystem designs, like different propellant types, can be easily benchmarked as necessary modifications of the tanks, engines and shape can be considered. Last a Risk Analysis Module (RAM) for the computation of the casualty and fatality probability according to the latest ESA and NASA safety guidelines is available. Through a semi-automatic procedure, a Monte Carlo analysis can be performed to compute the potential debris foot print. Advanced mission analysis is possible through the export filter to STK.

GESOP includes four optimization programs which are PROMIS (University of Illinois-Urbana Champaign) [R42], SOCS (Sparse optimal Control Software, Boeing Research & Technology) [R43], TROPIC (Trajectory Optimization by Direct Collocation, DLR/MBB/ESA-ESTEC) [R44] and CAMTOS (Collocation and Multiple Shooting Trajectory Optimization Software, University of Stuttgart) [R45].

In past years, analyses for missions like SMART-1, ConeXpress and SMART-OLEV were conducted from feasibility studies to operational purpose with increasing sophistication. ASTOS is being extensively used at ESA and aerospace industry community to calculate optimal launch and entry trajectories and was one of the tools used by ESA to assess the risk due to the ATV 'Jules Verne' re-entry.

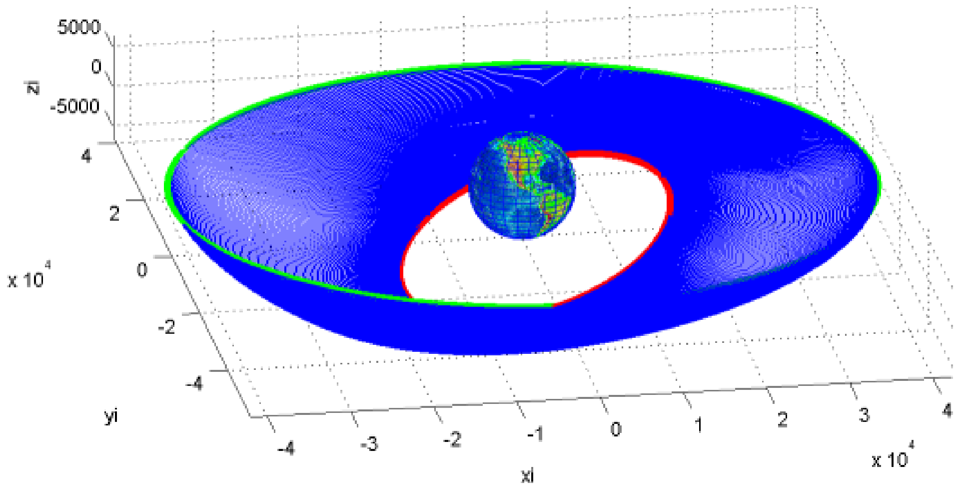

Figure 15: ASTOS Orbit Transfer.

6.1.4 STA (Space Trajectory Analysis)

STA is an open source astrodynamics suite initiated and supported by ESA. Conceived as an education and research tool to support the analysis phase of a space mission, STA is able to visualize a wide range of space trajectories. The functionality of the current STA version allows creating space scenarios, computing trajectories, creating plots, 2D and 3D visualizations, producing a system level analysis of a space mission, and storing data. Further the current version includes the following mission arcs: re-entry trajectories, descent and landing trajectories, Lagrange point trajectories, loitering trajectories around planets and moons, and system engineering analysis [R46].

STA provides calculations in the fields of spacecraft tracking, attitude analysis, coverage and visibility analysis, orbit determination, position and velocity of solar system bodies, etc. This software tool implements the concept of "space scenario", similar to STK, composed of Solar system bodies, spacecraft, ground stations, pads, etc. It is able to propagate the orbit of a spacecraft where orbital propagators are included. This software cannot yet calculate manoeuvres, although this will probably be possible in future versions. Trajectories are not propagated interactively, but instead a trajectory plan is built and then calculated by the propagator.

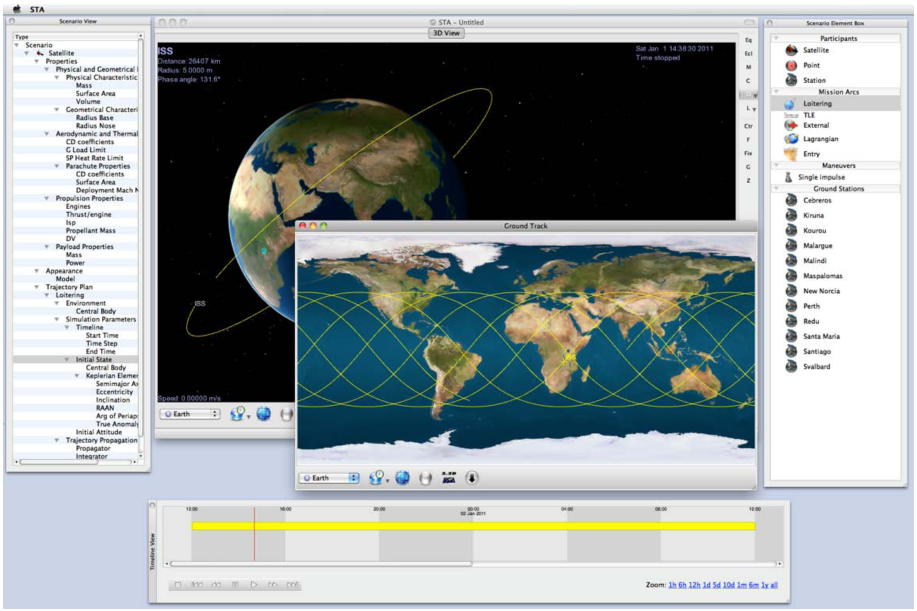

Figure 16: STA GUI.

All code of STA is open source. STA uses also open source codes provided by others. Firstly, this tool uses the Celestia code and Celestia data [R47]. In particular, STA uses Celestia's rendering engine for the 3D view window, some of its catalogs of stellar and planetary information, and some of its texture maps and 3D models of solar system bodies. STA also uses the Eigen libraries. Eigen is a C++ template library for linear algebra that includes vectors, matrices, and related algorithms. For plotting, STA also makes use of Curveplot. This is a library for high precision plotting of piecewise cubic curves in OpenGL [R48]. The GLEW library is used to access advanced OpenGL features used in the 3D visualization. STA contains as well the QwPlot3D programming library that provides 3D-widgets for plotting. And finally STA uses the optimizer NSGA-II from the Kanpur Genetic Algorithms Laboratory (KanGAL) of the Indian Institute of Technology Kanpur. Moreover STA is compatible with the NASA SPICE system for spacecraft and planetary information. SPICE was developed by Caltech/JPL under contract to NASA [R49]. The star database (stars.dat) is derived from the ESA's HIPPARCOS star catalogue and it is the same that Celestia uses.

6.1.6 TriaXOrbitaL

The program has been designed for computing and viewing dynamically in three dimensions the spacecraft trajectories with thrust periods and cruise (ballistic) periods. The program allows computing and viewing the secular orbital perturbations as well as their short period evolution. Interplanetary trajectories with or without thrust arcs are managed as well. The main rule for the use of the program being the traceability, each computation parameters is saved in a customized database. This tool is available for free at www.kopoos.com.

Some examples of Maneuvers:
The following examples are part of those provided with the genuine database included in the program package.

- Continuous orbit transfer strategy from GTO inclined 28° to GEO : standard views and and cubic view

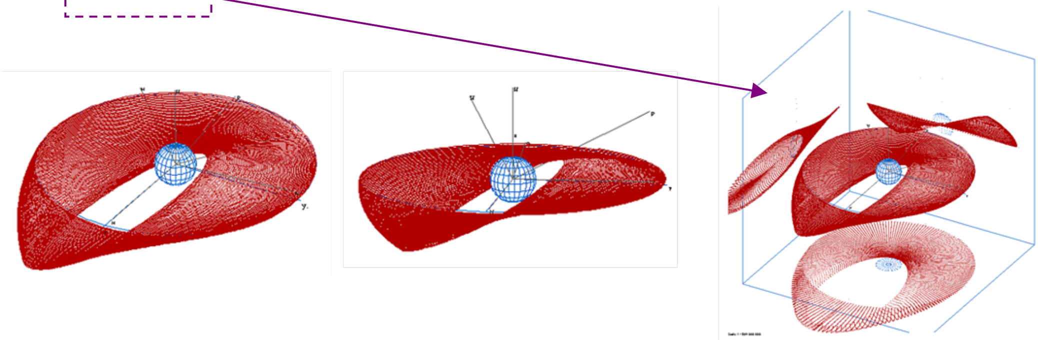

Figure 17: Maneuvers according to a Snecma C. R. Koppel patent

- Moon flyby standard views and zoom view of the flyby event and free Earth return (ref., Kaplan Apollo 11 back-up trajectory)

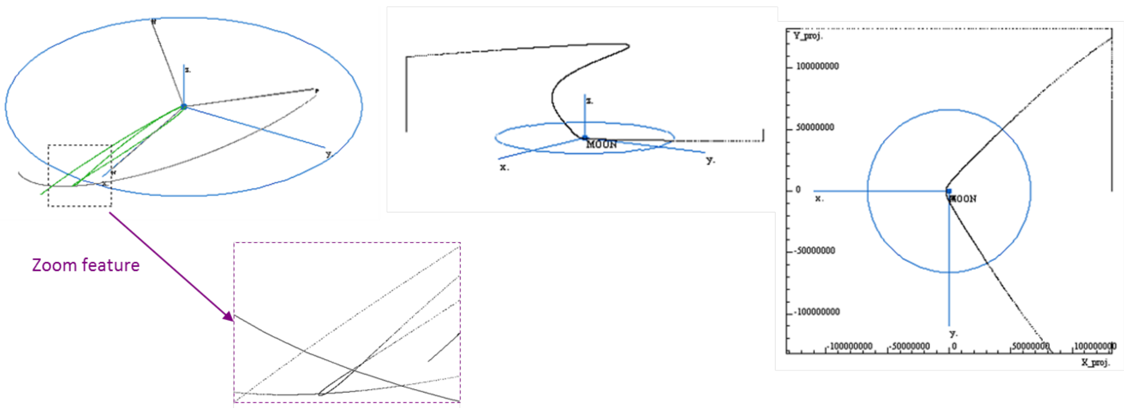

Figure 18: Interplanetary trajectory with a Moon flyby and a Venus flyby and Landing on the Moon south pole

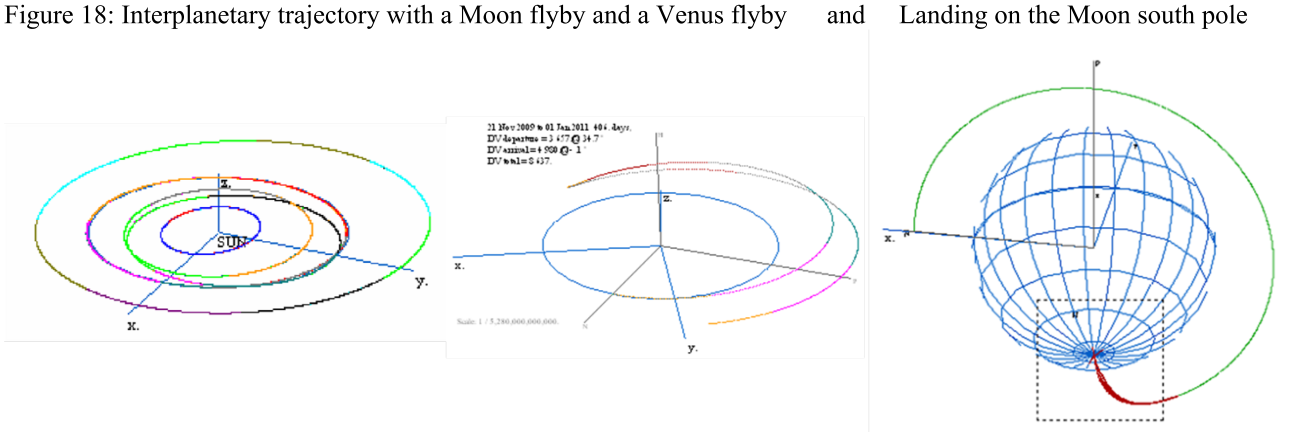

Figure 19: The tool includes an impressive Lambert porkchop feature for interplanetary trajectory without thrust

6.1.5 Orbiter

The software program Orbiter is a real time space flight simulator for Windows PC programmed and distributed by Dr. Martin Schweiger of UCL [R50]. It basically provides physical models for orbit and deep space flight mechanics, atmospheres as well as an environment for simulation of spacecrafts and their flight control system functions, i.e. cockpit environment (HUD, MFD). The ascent and descent flight phases are included. Several user defined packages are available for download to extend the range of simulated bodies, including ships, instruments, bases, orbital stations and planets.

Figure 20: Orbiter simulation of docking approach.

A relatively large and active internet community is contributing new models through web add-on repositories, fixing model related issues and providing input for future improvement of the simulator. An open-source development of graphic engines for orbiter is the Orbiter Visualization Project although this adds no functional value as of yet. The general purpose of Orbiter is personal use, education and other non-commercial applications. A closed source basic software package is provided free of charge for download under the Orbiter Freeware License Terms.

Although providing some of the required functionalities, the program was not designed as a mission-design engineering-tool but as a functional simulator of space vehicles. It is listed here anyway due to recent application attempts. No information or comparison of the actual accuracy of the simulation is currently available nor is there any interface to or utilization of common data bases. Further due to its closed source very limited tool support is available for professional utilization of any kind. A complete mission analysis framework would have to be implemented manually. Consideration of the simulator appears feasible in terms of a draft platform for a mission analysis tool still keeping in mind, that substantial temporal and financial efforts would be required for tool development and testing. This would be mandatory to achieve user friendly application and implementation into any kind of design process. However the economic feasibility of the development is limited by demands towards tool complexity but in turn could be sustained by the goal of achieving an independent in-house software solution if so desired.

6.1.6 STO - Spacecraft Trajectory Optimizer

Originally intended to investigate lunar transfer trajectories, the German EADS Astrium in Bremen within its Guidance Navigation and onboard Software department has developed the trajectory calculation software Luna 2 for mission analysis purposes. It allows for calculation of a flight path including predefined manoeuvre sequences.

Special emphasis was put on platform independence to keep the code simple, robust and fast. In addition 32bit Windows compatibility in combination with the sleak code allows for fast calculations and most desired applicable results. The tool was originally written in the FORTRAN language but has now been ported into a MatLab environment. The Fortran-tool lagged behind because of its initial moon trajectory application heritage. Since 2007 efforts are being put into adding new functions and extend the original program, i.e. graphical user interface, 3D-graphical visualization, database support, interfaces to supplemental tools and highly-modular internal structure. To reflect the numerous improvements, the new tool has been named STO. The program structure is presented in
Figure 21.

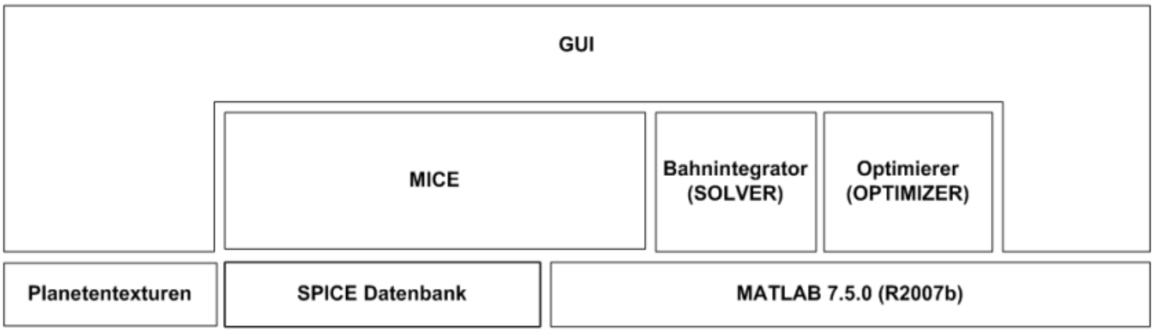

Figure 21: STO program structure.

The goal to develop the software into a full scale mission design, analysis and optimization tool with export capabilities for the results has successfully been achieved. For evaluation a comparison with the commercial software STK by AGI has been performed yielding a deviation of the results from both tools of less than three percent. As of current, the following main features are available through STO:

- User definition and input of a high number of celestial bodies and spacecrafts,
- Fully customizable thrusters and thruster configurations,
- User definable optimization criteria and optimization target values,
- Utilization of highly accurate solar system data (SPICE, HORIZONS),
- Intuitive and easy to use MatLab GUI,
- Definition of staging and course correction events,
- Fast and accurate trajectory integration algorithm,
- Iterative trajectory analyser and optimizer (currently Earth-Moon trajectories only),
- Detailed output of parameters over mission time for manual analysis.

As a result of the numerous implemented improvements, the STO tool offers a wide field of application. For example, the new software is currently also being used for educational purposes as part of the Student Space Station Design Workshop at the Institute of Space Systems in Stuttgart. The issues remaining with the software are:

- Rather limited MatLab graphical rendering engine for data display,
- Only high thrust level optimization implemented,
- No ascent/descent mission phase,
- No extraction of trajectory specific information (time of eclipse, contact window, etc.),
- Only gravitational influences on trajectory simulated,
- Dependence on MatLab with Optimization Toolbox,
- No official tool support.

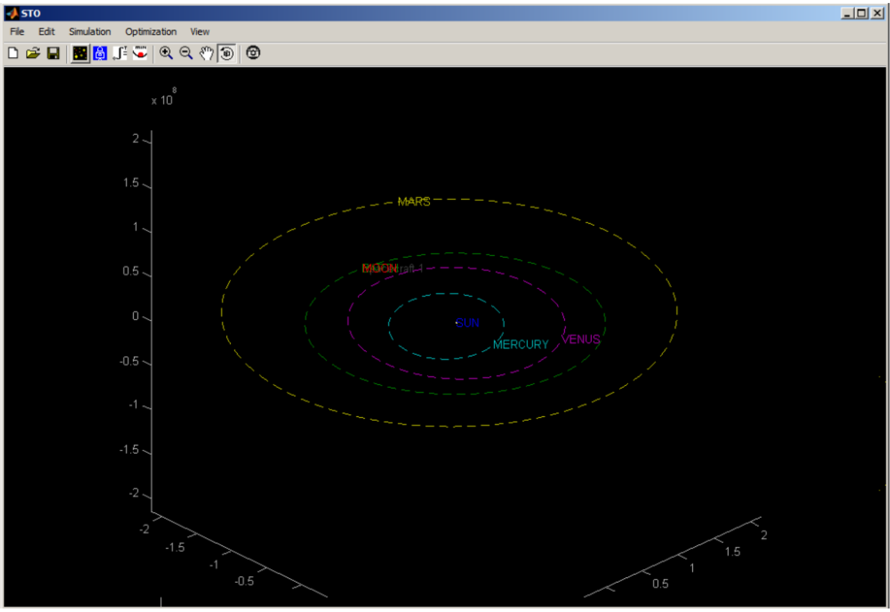

Figure 22: STO GUI main window.

The
Figure 22 illustrates the still rather minimalistic data displays which in no way compares to the graphics of rendering engines used by tools like Celestia or STK. In terms of only high level thrust optimization a workaround can be achieved by implementing additional routines into an external library. The problem is that low and continuous thrust level orbit transfers are not supported at present. The external library is supposed to keep the calculation times at acceptable levels.

In conclusion the STO-Tool is a very good option for generating fast results as part of the preliminary design process or to speed up the design eliminating unwanted mission trajectory scenarios through iterative pre-optimization runs. It offers a very good platform for future development due to utilization of the well supported and widely used MatLab-interpreter language. However, if reliable results are required including additional celestial perturbations and mission design specific trajectory parameters the use of more sophisticated tools like STK is mandatory.

6.2 Software Tool Selection Approach

The ultimate goal of the presented selection approach is the proposal of three options for a trajectory design method within this study. Each of the options includes several SW-tools. A complete and in-depth benchmark of all SW-tools in question is primarily limited by the complexity of the overall systematic mission design engineering process itself. Although the trajectory method as a tool is stand-alone, it is closely tied to the requirements of the process and the final way of implementation. This leaves room for interpretation in terms of what the tool needs to look like and what its functional portfolio needs to be. One way around this would be to consider all features and define weighing impact factors in order to comprehensively answer the question. However, this would require the procurement and substantial testing efforts which is beyond the scope of this study. It is also assuming that the tool evaluation can be completely separated from knowledge of the design process structure, which poses a high development risk.

According to this, the three options presented within this study have to be considered a proposal subject to the availability of all required information. To compensate for this, two general criteria are identified for evaluation and a simplified approach to tool selection. The criteria are universal, qualitative and establish a link between quality of the results and projected cost intensity. For the tool selection approach they will be defined on the tool and customer level respectively.

The first criterion is the quality of the extracted trajectory solution, i.e. the trajectory data set. It refers to the tool level. The quality can be broken down into accuracy and complexity. The accuracy is based on the amount and reliability of predefined data, the model library as well as the accuracy of the applied numerical solvers. The complexity represents the simulation depth and level of detail for the mission scenario, e.g. manoeuvres, body count, margins and constrains. For some scenarios also handling of the integrator and optimizer can pose difficulties. When the worst case scenario has been reached by the optimizer, a more complex solution is desirable to increase quality but always subject to a trade-off with calculation time.

The second criterion is cost control on a customer level, which refers to cost factors that can be expected for tool implementation. The expected costs are primarily controlled by customer involvement in the tool development process. A high amount of customer contributions to the design tool typically lowers costs. The general rule applies that cost cuttings are paid for in time investments. An economic break-even point is set by the equality of costs for COTS-license procurement and CFS development costs. Assuming the full availability of facilities and manpower for SW-development, higher contributions are beneficial in terms of in-house tool expertise (closed source) and tool customization for seaming less implementation into the mission design process. A decisive factor here can also be the level of outsourcing desired by the customer. On the contrary, the drawbacks of required resources and a usually long term development are often critical. The common keyword for this is time-to-market (TTM), which refers to smart product development cycles depending heavily on high-end simulation COTS products. An international trend for CFS is to create open source tools incorporating the worldwide scientific community to benefit from its extensive pool while keeping the TTM in perspective via possible spin-offs.

In Figure 23 a breakdown of the two criteria and their respective levels centred on the approach of a simplified tool selection is illustrated. Altogether three levels of quality are depicted bottom up in the form of blue coloured pyramid levels. It shows that within the method higher tool levels build on the basis of lower ones. The structure mirrors a deductive, result-oriented strategy as well as the interdependence of the results of each tool level.

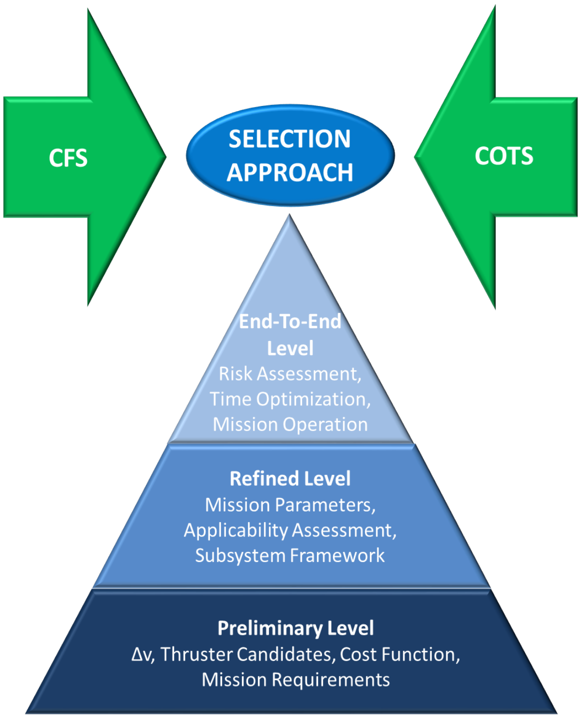

Figure 23: Tool Selection Approach Pyramid.

For each quality step the main progress of the result is added below the level description. The paramount of the pyramid carries the selection approach. The second criterion of cost control incorporates two choices for the customer depicted by the two green arrows to either side of the approach. While the criterion of quality of the results is inherent for all three presented options, the options themselves differ in their projected cost. Two of the options result from a focus on utilizing customer-of-the-shelf (COTS) or customer furbished software (CFS) tools only. The third option represents a hybrid of SW-tools, facilitating COTS but incorporating CFS where possible to cut costs and keep own contributions to the point.

With the prescribed approach it is now possible to determine the design tool options by matching each SW-tool to the levels of the two criteria. Each of the three method options has to involve all of the tool levels. In Table 10 the three by three field matrix shows the assignment of SW-tools as a result of the selection process. The table interprets the two selection criteria horizontally and vertically. The rows show the customer level representing cost, while the lines show the tool level, representing quality. Therefore, each of the rows stands for one of the three distinct options.

Method / Tool Level	CFS	Hybrid	COTS
Preliminary	Excel	Orbiter & Simulink / Excel	STK
Refined	MATLAB / Simulink / STO	STA / GMAT	STK
End-To-End	ANSYS / CATIA	ANSYS / CATIA	GESOP & ASTOS / ANSYS

Table 10: Software Tool Selection.

The COTS option lacks an extra preliminary level as the STK tool is an all-in-one solution. Preliminary estimations for trial runs of simplified scenarios or for general scenario adjustment ahead of the trajectory extraction are possible by deactivating the optimizer, limiting the level of detail or choosing basic trajectory solvers. The end-to-end level still requires coupling of other professional tools to complement the trajectory data down to hardware level, e.g. simulation of thermal-vacuum and mechanical hardware characteristics.

For the CFS option the tools are generally more basic yet still potentially powerful especially from the refined level on. Simple scenario iterations can be done by building a suitable Excel-table. This is also generally sufficient in terms of accuracy especially with respect to the many assumptions at this design stage. The STO-tool is fully customer designed and allows for accurate and fast trajectory extraction. It should be complemented by Simulink to include the investigation subsystems along the trajectory. The high penetration of the ANSYS and CATIA smart development tools recommends them for customer designed add-on development. This would only require procurement of the most basic program modules keeping costs at a minimum.

The hybrid solution benefits heavily from open source development, therefore combining customizability with the already existing extensive pool of functions. Feasible results can even be achieved on a purely user level interaction without having to put in expert knowledge. Cost savings can be expected with regard to the active community and its constant tool support and maintenance. This makes this option very versatile and flexible, especially in terms of reaction time for implementation of modifications. However, the risk of buggy functions, program errors and incompatibility is heightened compared to both, CFS as well as COTS. Also the overall engineering mission design process is exposed which can be flagged as classified. The perspective of creating a commercial tool from this option are increased compared to since a natural market penetration and awareness work towards creating a brand with an extensive network of followers.

7 Preparation of Evaluation Matrix

7.1 Criteria and Weighting

There are two different ratings of the thrusters done by this report. A general evaluation is done on behalf of the specific thruster parameters and additionally a mission specific evaluation illustrates the viability of the thrusters for these specific missions. The general thruster evaluation is done by the following criteria:

- General
  - TRL — As defined by ESA (see Table 11)
  - Complexity — Accounts for the number of subsystems and regulation task
  - Lifetime — The total system lifetime
  - Maintainability — How many parts have to be exchanged for extended lifetime?
  - Safety — How hazardous is the thruster or its propellant?

- Performance and Scalability
  - Nominal power — Not rated but needed for clustering assessment
  - Power specific mass — The weight of the total propulsion system divided by the power
  - Area specific thrust — The thruster diameter divided by the thrust
  - Power specific thrust — The electric input power divided by the thrust
  - Exit velocity — A nominal exit velocity
  - Thrust efficiency — As defined by $\eta = F^2 / 2\dot{m}P_{el.}$
  - Throttle ability — Minimum operation power divided by maximum operation power
  - Clustering ability — The ease of clustering
  - Costs — An estimation of the cost of the total propulsion system

The NTP thrusters will get a separate rating for power specific mass, power specific thrust and thrust efficiency as they convert the thermal energy of nuclear reactor directly into thrust power.

- Propellant (see Table 12)
  - Availability — Mass of propellant produced each year worldwide
  - Tank mass — Mass of tank system divided by total mass of tank and propellant at a reference point of 1000 t propellant mass
  - Costs — Estimated cost of the propellant
  - Evaporation — Is evaporation or melting of the propellant needed for conveying?
  - Contamination — Does the propellant contaminate the spacecraft?

The missions proposed before then deliver mission requirements (see Table 3 for example) to be fulfilled by nuclear electric (D23.2) or nuclear thermal (D23.3) propulsion. Thus each mission has a given velocity change Δv, a start or end mass m_0 or m_e, a maximum transfer time t (or a thrust F requirement) and a power density of the reactor (for NEP - given by D31 or D32). From these requirements following mission characteristics are then rated:

- Mission
  - Cluster Size

    How many thrusters are needed for the required thrust
  - Payload ratio

    The payload ratio µL will be calculated by considering the mass of the propellant, the propellant tank, power generator and the thruster system itself. While propellant budget will be received just by the solving the Ziolkowski equation, the thruster system mass can be estimated with the mass of the plain thruster times the cluster number and the system mass scaling with plus 50 % per additional thruster. The tank mass can be estimated by extrapolating reference tank masses (
    Table 12) with the exponent 2/3.
  - Cost per Payload

    The cost of 1 kg payload considers launch cost, power cost, propellant cost and hardware cost as defined by Table 13
  - Propellant Availability

    The needed propellant for this mission divided by the total world production per year
  - Life time usage

    The life time of the thruster divided by the mission duration time

TRL	Level description
1	Basic principles observed and reported
2	Technology concept and/or application formulated
3	Analytical & experimental critical function and/or characteristic proof-of-concept
4	Component and/or breadboard validation in laboratory environment
5	Component and/or breadboard validation in relevant environment
6	System/subsystem model or prototype demonstration in a relevant environment (ground or space)
7	System prototype demonstration in a space environment
8	Actual system completed and "Flight qualified" through test and demonstration (ground or space)
9	Actual system "Flight proven" through successful mission operations

Table 11: Technology readiness level description.

Propellant	Word production / t/a	Cost / €/kg	Ref. tank weight kg/kg	Reference
Xenon	60	1200	7,16/44	[R51], [R52], [R53]
Krypton	320	330	-	[R51], [R52]
Neon	3400	330	-	[R51], [R52]
Helium	30.000	50	12,7/3,7	[R52], [R53]
Hydrazine	100.000	20?	9,5/95	[R54], [R55]
Argon	750.000	5	12,7/37	[R52], [R56], [R53]
Ammonia	100.000.000	<1	11,2/55	[R54], [R58]
Hydrogen	55.000.000.000	120	4740/9200	[R57]

Table 12: Propellant properties and world production capacity (*t/a: tons per year*).

Component	Power	Hardware	Launch (interplanetary)
Cost	1,5 k€/W	50 k€/kg	30 k€/kg

Table 13: Cost estimation [R60], [R61].

Weighting		1	2	3	4	5
Criteria	**Factor**	**Very good**	**good**	**fair**	**bad**	**Very bad**
Cluster Size -	3	1-5	5-20	20-50	50-100	100+
Payload ratio µL %	5	70-100	50-70	30-50	10-30	0-10
Cost per Payload k€/kg	5	0-50	50-100	100-150	150-200	200+
Prop. Availability %	1	<0,01	>0,01	>0,1	>1	>10
Life time %	3	<10	10-20	20-50	50-100	100+

Table 14: Mission specific thruster evaluation.

Weighting		1	2	3	4	5
Criteria	**Factor**	**Very good**	**good**	**fair**	**bad**	**Very bad**
General						
TRL -	5	9	7-8	5-6	3-4	1-2
Complexity -	1	Few Subsystems & Regulation	-	Some Subsystems & Regulation	-	A lot of Subsystems & Regulation
Lifetime kh	3	>20	10-20	5-10	1-5	< 1
Maintainability -	1	Single Parts	-	Whole Thruster	-	Whole System
Safety -	2	Not Toxic nor Explosive nor Radioactive	-	Toxic or Explosive or Radioactive	-	Toxic and Explosive and Radioactive
Performance and Scalability						
Spec. mass kg/kW	3	< 1	1-10	10-30	30-100	100+
Spec. thrust/area N/m²	3	>1000	100-1000	10-100	1-10	<1
Spec. thrust/power N/MW	5	100+	75-100	50-75	25-50	< 25
Exit velocity km/s	5	10+	5+	2,5+	1+	< 1
Thrust efficiency %	5	80-100	60-80	40-60	20-40	0-20
Throttle ability %	2	1	10	25	50	75+
Clustering ability -	2	No Problem	-	Some Problems	-	Serious Problems
Costs k€/kW	1	< 10	10-25	25-50	50-100	100+
Propellant						
Availability t/a	2	> 100.000	< 100.000	< 10.000	< 1000	<100
Tank mass %	3	<5	5-10	10-25	25-50	50+
Costs €/kg	1	<1	1-10	10-100	100-1000	1000+
Evaporation -	1	Not Needed	-	Evaporation Needed	-	Melting Needed
Contamination -	2	No Problems	-	Some Problems	-	Serious Problems

Table 15: General thruster evaluation.

7.2 Evaluation Matrix

For the final evaluation matrix the rating for every thruster is multiplied with the factor of the criteria and a mean rating is defined for general missions and specific to the suggested missions. The example in table 16 is suggest for [RD3]. The documents [RD1] and [RD2] contain adapted evaluation matrices for NEP and NTP each.

Thruster type	General	NEO	Mars probe	Mars manned	Mars rescue	Gas planets	Kuiper belt obejcts
Thruster 1	…	…	…	…	…	…	…
Thruster 2	…	…	…	…	…	…	…
…	…	…	…	…	…	…	…

Table 16: Exemplary matrix: Final evaluation of propulsion systems.

8 Conclusion

The present document provides information on work package 23. First electric thrusters which can be used in Nuclear Electric Propulsion are introduced before a generalised overview on Nuclear Thermal Propulsion is given. Both concepts are developed in respective documents [RD1], [RD2]. For preliminary analysis a comparison of both approaches is enabled through a set of block diagrams.

In the second part of the report, both interesting generic mission scenarios are developed as well as an overview of state of the art mission analysis tools introduced. Modelling and simulation of the same mission goals but with different propulsion systems can reveal the best approach to fulfil more specific missions with goals similar to the generic scenarios developed here.

The document concludes with a suggestion for evaluation matrices for NEP and NTP based on missions. In the final document [RD3], this can be combined with the respective system evaluation matrices from [RD1] and [RD2]. Thus, it can be possible to develop a more mature evaluation upon which educated decisions on a rational development can be made.