7th International Symposium on Launcher Technologies, Barcelona, Spain, 02-05 April 2007

A COMPARISON OF SEP AND NEP FOR A MAIN BELT ASTEROID SAMPLE RETURN MISSION

Bernd Dachwald[1], **Wolfgang Seboldt**[2], **Horst W. Loeb**[3], and **Karl-Heinz Schartner**[3]

[1]DLR, Oberpfaffenhofen, Germany
German Space Operations Center, Mission Operations Section
Phone: +49-8153-28 2772, E-Mail: bernd.dachwald@dlr.de

[2]DLR, Cologne, Germany
Phone: +49-2203-601 3028, E-Mail: wolfgang.seboldt@dlr.de

[3]University of Giessen, Giessen, Germany
I. Physics Institute
Phone: +49-641-9933 {130|140}, E-Mail: {horst.w.loeb|karl-heinz.schartner}@exp1.physik.uni-giessen.de

Innovative interplanetary deep space missions, like a main belt asteroid sample return mission, require ever larger velocity increments (ΔVs) and thus ever more demanding propulsion capabilities. Providing much larger exhaust velocities than chemical high-thrust systems, electric low-thrust space-propulsion systems can significantly enhance or even enable such high-energy missions. In 1995, a European-Russian Joint Study Group (JSG) presented a study report on "Advanced Interplanetary Missions Using Nuclear-Electric Propulsion" (NEP). One of the investigated reference missions was a sample return (SR) from the main belt asteroid (19) Fortuna. The envisaged nuclear power plant, Topaz-25, however, could not be realized and also the worldwide developments in space reactor hardware stalled. In this paper, we investigate, whether such a mission is also feasible using a solar electric propulsion (SEP) system and compare our SEP results to corresponding NEP results.

1 Introduction

In 1995, a European-Russian Joint Study Group (JSG) presented a study report on "Advanced Interplanetary Missions Using Nuclear-Electric Propulsion" (NEP) [1]. One of the investigated reference missions was a sample return (SR) from the main belt asteroid (19) Fortuna, another one was a mission to Mercury. The envisaged nuclear power plant, Topaz-25, however, could not be realized and also the worldwide developments in space reactor hardware stalled. In the meantime, the Mercury mission became an ESA cornerstone mission, BepiColombo, based, however, on solar electric propulsion (SEP). Consequently, the question arose whether SEP might also be profitable for missions into the outer regions of the solar system.[1] Therefore, the German Space Agency placed a study contract to the University of Giessen, with a sub-contract to the German Aerospace Center, to assess the prospects of SEP for missions to the outer regions of the solar system. This paper describes the results of this study for the SEP Fortuna-mission and compares them to corresponding NEP results.

[1]NASA's discovery class mission Dawn already demonstrates the feasibility of a *non*-SR mission to the main asteroid belt using SEP [2, 3].

Innovative interplanetary deep space missions require ever larger velocity increments (ΔVs) for accelerating and breaking spacecraft and thus ever more demanding propulsion capabilities. The so-called rocket equation gives the ΔV that spacecraft can gain as

$$\Delta V = V_e \cdot \ln(m_0/m_f) \quad (1)$$

where V_e is the exhaust velocity of the propellant, m_0 is the initial spacecraft mass and m_f is the final spacecraft mass. Due to the energy barrier inherent in chemical combustion, chemical high-thrust propulsion systems (rocket engines) have a limited V_e and thus a limited ΔV-capability. Therefore, the state-of-the-art technique for high-energy missions uses chemical high-thrust propulsion systems in combination with planetary gravity-assist maneuvers to achieve larger ΔVs. This technique, however, results in long, complicated, and inflexible mission profiles, as the delays of ESA's Rosetta-mission [4] and NASA's MESSENGER-mission [5] have shown. For a sample return mission to Fortuna, the overall required ΔV is at least 13.7 km/s, which renders this mission very difficult for any chemically propelled spacecraft.

Electric low-thrust space-propulsion systems can significantly enhance or even enable high-energy missions because – providing much larger exhaust velocities than chemical high-thrust systems – they use the propellant more efficiently. Consequently, they permit significantly larger ΔVs, larger payloads, and/or cheaper launch vehicles. At the same time they allow direct trajectories with simpler mission profiles, flexible launch windows, and mostly even reduced flight times, as compared to gravity-assist trajectories.

2 Mission Outline

The asteroid (19) Fortuna is one of the largest asteroids in the main belt. The diameter of Fortuna's nearly circular shape, as it has been measured by the Hubble Space Telescope, is about 225 km [7]. It has a darkly colored low-albedo surface and is likely composed of primitive carbon compounds with aqueous altered surface material. Fortuna is a good candidate for a sample return mission into the asteroid belt because the study of the aqueous alteration process can give important information on the chemical and thermal evolution of the earliest solar system [6]. The classical orbital elements of Fortuna in the J2000 heliocentric ecliptic reference frame are [8]:

$$\begin{aligned}
\text{Epoch} &= 53800 \\
a &= 2.4417549\,\text{AU} \\
e &= 0.15934995 \\
i &= 1.57343\,\text{deg} \\
\omega &= 182.00272\,\text{deg} \\
\Omega &= 211.37213\,\text{deg} \\
M &= 37.6395577\,\text{deg}
\end{aligned}$$

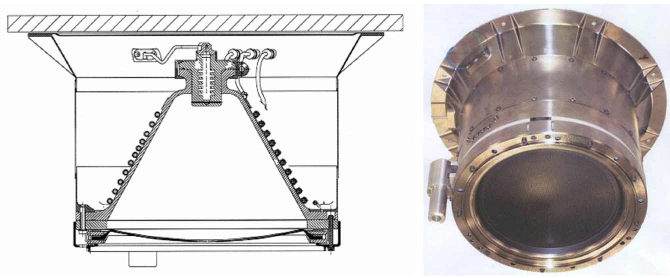
For both the SEP and the NEP propulsion option, two sample return mission concepts have been considered, a combined EP/chemical concept and an all-EP sample return mission concept:

- **CR (chemical return):** The "payload" that the EP transfer vehicle has to deliver to the asteroid comprises a 175-kg lander and a 565-kg (wet) two-stage chemical sample return vehicle (SRV), which returns the collected samples to Earth.[2] The EP transfer vehicle stays at the asteroid. This scenario was chosen by the JSG in [1] for the NEP Fortuna SR-mission.

- **ER (electric return):** The EP vehicle samples the asteroid with its own devices (it drops a small autonomous surface package) and returns the collected samples to Earth, where it releases a reentry capsule. The estimated "payload" mass of this option is 225 kg.

Thus there are four EP mission architectures to be investigated, **SCR** (solar electric with chemical return), **SER** (all solar electric), **NCR** (nuclear electric with chemical return), and **NER** (all nuclear electric).

[2] JSG-design by Lavochkin



(a) Cross section (b) Photo

Figure 1: The RIT-22 ion thruster (image courtesy of EADS Astrium)

Table 1: Nominal performance data of the RIT-22 ion thruster

Beam voltage	V_{beam}	2.100 kV
Specific impulse	I_{sp}	4763 s
Power consumption	P_{max}	6.209 kW
Thrust	F_{max}	175.0 mN
Thrust-to-power ratio	F_{max}/P_{max}	28.18 mN/kW
Propellant mass flow	$\dot{m}$	3.719 mg/s
Propellant mass flow per power	$\dot{m}/P$	0.6034 mg/kWs

Within our study, a launch between 01 Jan 2012 and 31 Dec 2015 was foreseen. To allow a better comparison of the results, direct interplanetary insertion with zero hyperbolic excess energy ($C_3 = 0\,\text{km}^2/\text{s}^2$) was assumed and gravity assists were not foreseen, although they might be beneficial for the real mission. For the Fortuna-Earth return leg, the hyperbolic arrival velocity has been limited to 6.4 km/s because the gravitational acceleration of Earth adds another 11.2 km/s, so that the Earth entry velocity is $\sqrt{6.4^2 + 11.2^2}\,\text{km/s} = 12.9\,\text{km/s}$, which is equivalent to the re-entry of NASA's Stardust capsule [9, 10].

3 Spacecraft Design

3.1 The RIT-22 Ion Thruster

After having reviewed the existing EP hardware in Europe, we have decided to employ for our proposed mission on the RIT-22 ion thruster, which is based on the existing flight-proven RIT-10 ion thruster and currently under qualification at EADS-ST and the University of Giessen. A RIT-22 thrust unit consists of the thruster (7.9 kg), the rf-generator (RFG, 1.9 kg), the neutralizer (0.4 kg), the flow control unit (FCU, 0.4 kg), and the power supply converter unit (PSCU, 17.9 kg). The total mass of a RIT-22 thrust unit is therefore 28.5 kg. Figure 1 shows the cross section and a photo of the RIT-22 thruster. The performance data of the RIT-22 is given in Table 1. The RIT-22 has the following key advantages:

1. As a gridded ion thruster, it is able to run with specific impulses up to 7000 s with high efficiencies.

Table 2: SRV mass budget (all values are in kg)

Liquid propellant booster structure	50
Harness	4
Margin	8
First stage dry	**62**
Propellant	400
First stage wet	**462**
Onboard control complex	20
Onboard radio complex	10
Antennas & feeders	2
Power supply subsystem	6
Structural body with tanks	5
Propulsion subsystem	6
Thermal control subsystem	3
Structure	3
Harness	4
Margin	10
Re-entry capsule	21
Second stage dry with re-entry capsule	**90**
Propellant	13
Second stage wet with re-entry capsule	**103**
Sample return vehicle (two stages wet)	**565**

2. The absence of discharge electrodes in the ionizer chamber guarantees reliable operation for extended lifetimes, being only limited by grid erosion. In addition, the discharge electronics could be on ground potential, even at high beam voltages.

3. Regulating only rf-discharge power at a given propellant flow rate provides simple and easy thrust control.

For a power level $0.65 \leq P/P_{\text{max}} \leq 1$, it can be assumed that $F \propto P$ and $\dot{m} \propto P$, whereas the specific impulse is constant.

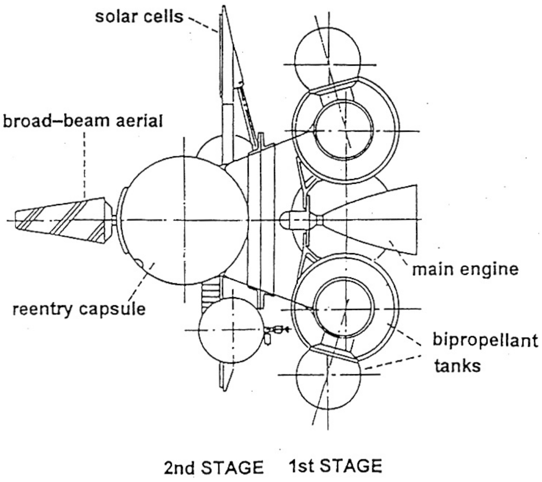
3.2 Chemical Sample Return Vehicle

For mission architectures SCR and NCR, the SRV is used for the transfer from Fortuna back to Earth. It comprises two stages:

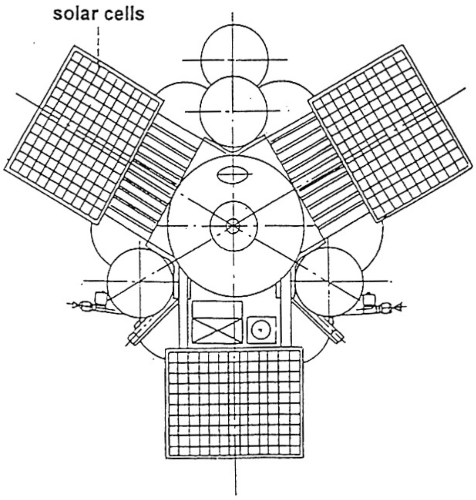
1. The first stage is intended for the insertion of the spacecraft into the Fortuna-Earth return trajectory. It is equipped with liquid propellant boosters (hydrazine + nitrogen tetroxide, $I_{sp} = 315\,\text{s}$).

2. The second stage is intended for trajectory corrections of the spacecraft during cruise and before Earth return.

Figure 2 shows two cross sections of the SRV. The mass budget of the SRV is given in Table 2. The required mass of the SRV depends strongly on the actual Fortuna-Earth return trajectory. Assuming a Hohmann-transfer from Fortuna's aphelion to Earth, the minimal ΔV that is required for this phase would be 3.44 km/s. Taking, however, the actual constellations of Fortuna and Earth in the return launch window (01 Dec 2012 – 31 Dec 2019) into account, the minimal ΔV is 3.64 km/s for a launch at Fortuna on 07 Sep 2015 (arrival at Earth on 06 Nov 2016, 1.17



(a)



(b)

Figure 2: Two cross sections of the sample return vehicle (from [1])

years transfer duration). Figure 3 shows the minimal ΔVs within this launch window and the associated transfer times.

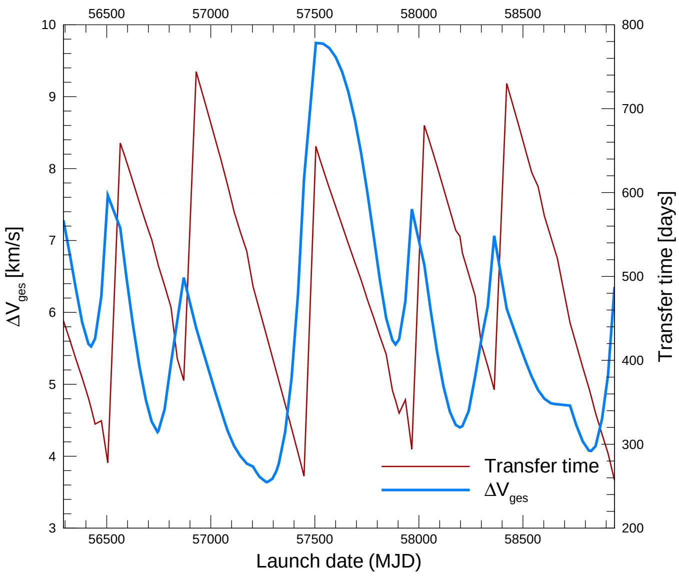


Figure 3: ΔVs and transfer times for the Fortuna-Earth return trajectory

7th International Symposium on Launcher Technologies, Barcelona, Spain, 02-05 April 2007

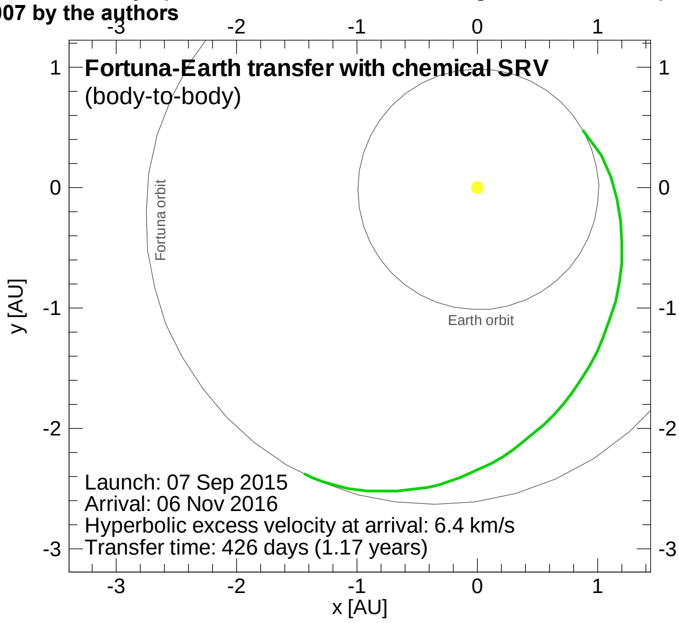


Figure 4: Fortuna-Earth body-to-body transfer trajectory for the chemical SRV.

Using $m_{pR1} = 400$ kg of propellant for the first stage and $m_{pR2} = 13$ kg for the second stage, the ΔV-capability is 3.8 km/s for the first stage and 0.37 km/s for the second stage. With this ΔV-capability, there exists a single Earth return window (see Fig. 3), where $\Delta V \leq 3.8$ km/s. It opens at 12 Jul 2015, closes at 20 Oct 2015, and has thus a width of 100 days.

3.3 SEP Spacecraft Design

For the SEP power subsystem, conventional Si HI-ETA solar cells are used. They achieve at 1 AU conversion efficiencies of 15.4% and provide, given a 90% covering density, an area specific power of 190 W/m^2. With an area specific mass of 2.5 kg/m^2, this yields a mass specific power of 76 W/kg. Our solar electric power subsystem is designed to yield at BOM (begin of mission) the power that is required by three RIT-22 ion engines at full thrust, 18.629 kW. This yields a solar panel area of 98 m^2 and a mass of 245 kg. This can be realized with two wings, each consisting of eight 2×3.06 m panels. Therefore, the length of one panel is 16 m.

For the degradation of the solar cells, we have used a simple model that gives a degradation of 7.5% at Fortuna and a degradation of 15% back at Earth (for mission architecture SER). In our simple model, the degradation of the solar cells does only depend on solar distance r. Without degradation, the power output of the solar cells would be $P(r) = P_{\mathrm{BOM}}(r_0/r)^{1.7}$ ($r_0 = 1$ AU, the exponent of 1.7 is typically used to describe the better efficiency of solar cells at lower temperatures), where P_{BOM} is the power output at BOM. To get a by 7.5% reduced power output at the target (semi-major axis is 2.44175 AU), we use $P(r) = P_{\mathrm{BOM}}(r_0/r)^{1.78733}$ for the Earth-Fortuna leg and to get a by 15% reduced power output back at

Table 3: Masses, mass fractions, and accelerations for the two SEP mission architectures

		SCR	SER
"Payload" mass	[kg]	740	225
Bus mass	[kg]	415	415
Spacecraft dry mass without EP module	[kg]	1155	640
Power unit mass	[kg]	245	245
Propulsion units mass	[kg]	85	85
Structure + tank + gimbal system mass	[kg]	147	148
EP module dry mass	[kg]	477	478
Spacecraft dry mass	[kg]	1632	1118
Xe propellant mass (Earth-Fortuna)	[kg]	421	317
Xe propellant mass (Fortuna-Earth)	[kg]	-	108
Spacecraft launch mass	[kg]	2053	1543
"Payload" + bus mass fraction	[%]	56.3	41.5
EP module mass fraction	[%]	23.2	31.0
Propellant mass fraction	[%]	20.5	27.5
Acceleration at launch	[mm/s^2]	0.2557	0.3402
Acceleration at Fortuna arrival	[mm/s^2]	0.0652	0.0868
Acceleration at Earth return	[mm/s^2]	-	0.3992

Earth, we use $P(r) = 0.85 P_{\mathrm{BOM}}(r_0/r)^{1.60528}$ for the Fortuna-Earth leg.

Figure 5 sketches the SCR-vehicle with the attached lander and SRV. Three thrusters are used. At

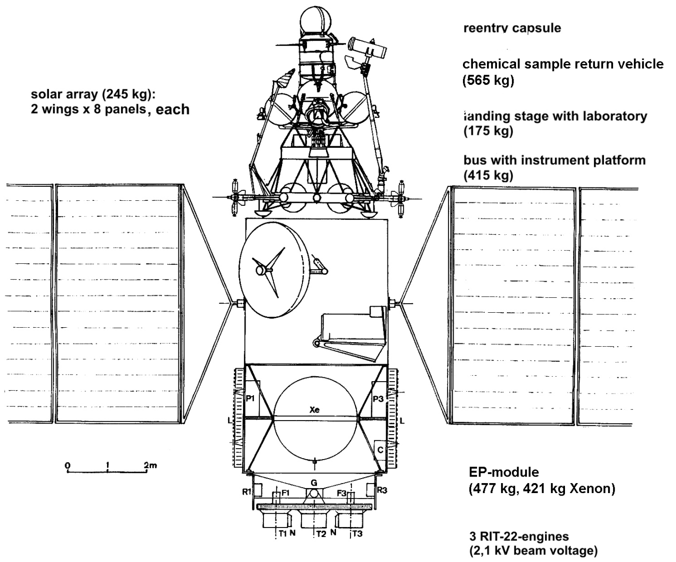


Figure 5: Sketch of the SCR-vehicle (length 6.5 m, launch mass 2.05 mt) with 3 RIT-22 engines. The SER-vehicle is similar, but without the chemical SRV. Legend: Xe: Xenon propellant tank, T: RIT-22 thrusters, N: neutralizers, F: flow control units, R: rf-generators, P: power supply & control units, C: cluster control unit, L: thermal louvers, G: gimbal system

the maximum thrust of $3F_{\max} = 0.525$ N, their total power consumption is $3P_{\max} = 18.627$ kW and their propellant consumption is $3\dot{m}_{\max} = 11.240$ mg/s. Table 3 shows the masses, mass fractions, and achieved accelerations for the two SEP mission architectures. The propellant masses have been estimated with the classical equation for the ΔV-requirement of electric propulsion systems for transfers between co-planar circular orbits on a logarithmic spiral trajectory [1]:

$$\Delta V = v_0 \left(1 - \sqrt{R_0/R_T}\right) \tag{2}$$

where v_0 is the velocity of the initial body (29.8 km/s for Earth), R_0 is the orbital radius of the initial body (1 AU for Earth), and R_T is the orbital radius of the target body (2.44 AU for Fortuna). Thus for Earth and Fortuna, $\Delta V = 10.72\,\mathrm{km/s}$.

3.4 NEP Spacecraft Design

For the NEP spacecraft power subsystem, the Topaz-25 nuclear reactor is used. It is an up-scaled design version of the space-qualified Topaz reactor that was build by Krasnaya Zvesda, Moscow, in the late 1980s. The Topaz-25 reactor provides $P_{NR} = 30\,\mathrm{kW_e}$ of electric power.

Figure 6 sketches the NCR-vehicle with the attached lander and SRV. Five RIT-22 ion thrusters are

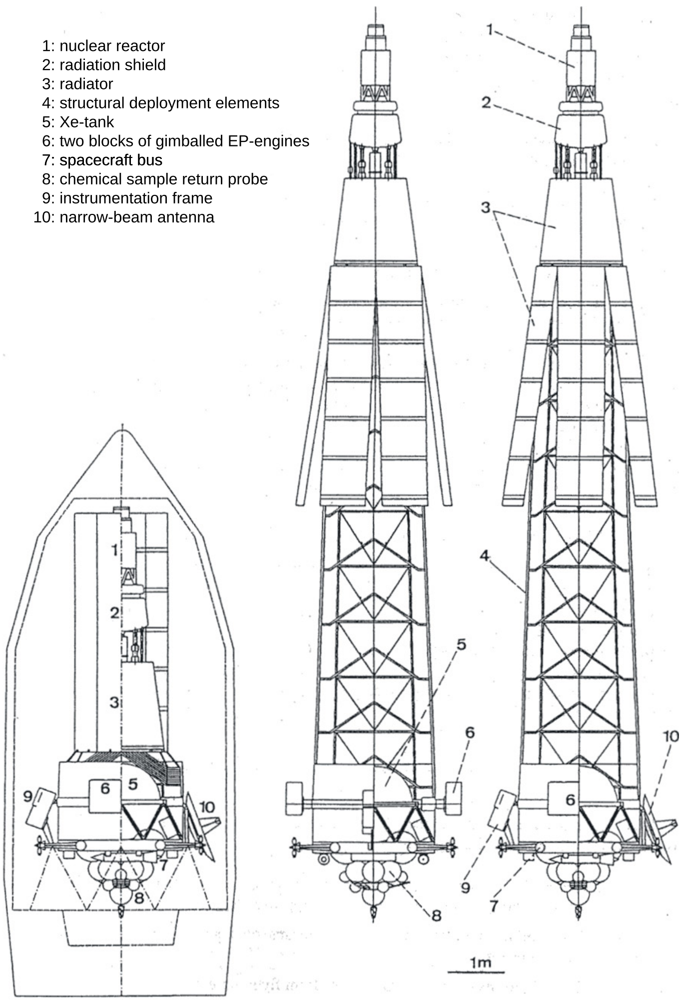


Figure 6: Sketch of the NCR-vehicle (length 16.7 m, launch mass 5.36 mt) with 5 RIT-22 engines. The NER-vehicle is similar, but without the chemical SRV. (In the original JSG-study, 8 ESA-XX thrusters have been used because the RIT-22 did not exist at that time.)

used. At the maximum thrust of $5F_\mathrm{max} = 875\,\mathrm{mN}$, their total power consumption is $5P_\mathrm{max} = 31.045\,\mathrm{kW}$ and their propellant consumption is $5\dot{m}_\mathrm{max} = 18.733\,\mathrm{mg/s}$. Because $P_{NR} < 5P_\mathrm{max}$, the NEP spacecraft runs at a reduced thrust level of 846 mN, where $\dot{m} = 18.102\,\mathrm{mg/s}$. For a power level $0.65 \leq P/P_\mathrm{max} \leq 1$, it can be assumed that $F \propto P$ and $\dot{m} \propto P$, whereas the specific impulse is constant. Table 4 shows the masses, mass fractions, and achieved accelerations for the NEP mission architecture.

Table 4: Masses, mass fractions, and accelerations for the two NEP mission architectures

		NCR	NER
"Payload" mass	[kg]	740	225
Bus mass	[kg]	415	415
Spacecraft dry mass without EP module	[kg]	1155	640
Power unit mass	[kg]	2330	2330
Propulsion units mass	[kg]	142	142
Structure + tank + gimbal system mass	[kg]	637	689
EP module dry mass	[kg]	3109	3161
Spacecraft dry mass	[kg]	4264	3801
Xe propellant mass (Earth-Fortuna)	[kg]	1100	1076
Xe propellant mass (Fortuna-Earth)	[kg]	-	369
Spacecraft launch mass	[kg]	5364	5246
"Payload" + bus mass fraction	[%]	21.5	12.2
EP module mass fraction	[%]	58.0	60.3
Propellant mass fraction	[%]	20.5	27.5
Acceleration at launch	$[\mathrm{mm/s^2}]$	0.1576	0.1612
Acceleration at Fortuna arrival	$[\mathrm{mm/s^2}]$	0.1983	0.2028
Acceleration at Earth return	$[\mathrm{mm/s^2}]$	-	0.2225

3.5 Comparison of the SEP and NEP Spacecraft Design

Figure 7 makes a graphical comparison of the launch masses of the four mission architectures. One can

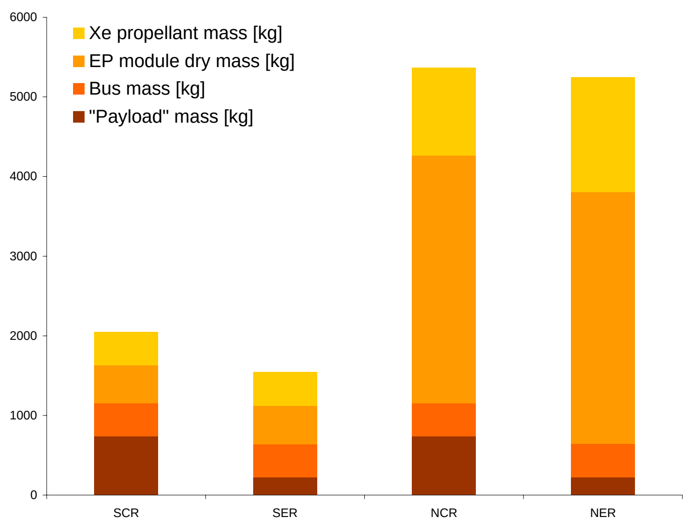


Figure 7: Comparison of the SEP and NEP mission architectures.

clearly see that the launch masses for the SEP mission architectures are much smaller than for the NEP architectures. This would result in large cost reductions for the SEP architectures, because a smaller and thus cheaper launch vehicle could be used. Further SEP cost advantages come from the fact that solar cells are easily available of the shelf, whereas a nuclear reactor is still to be developed and so the first missions would have to bear a large part of the development costs. Besides technical arguments, another advantage of SEP over NEP is the political acceptance, which should not be discussed here further.

The question now is whether these mass advantages come with large penalties in transfer times / mission duration and/or mission flexibility.

4 Results

4.1 Results for Orbit-To-Orbit Transfers

For an unbiased comparison of both SEP and NEP, it is necessary to compare the propulsion *capabilities* of the different mission architectures rather than the performance for specific mission alternatives, which are defined by the actual constellations of Earth and Fortuna during the considered launch window. Therefore, we have first optimized the trajectories for an orbit-to-orbit (not body-to-body) transfer. This process yields the absolute transfer time minima, irrespective of the actual constellation of Earth and Fortuna at launch. A body-to-body transfer for mission architecture SER is given later in section 4.2.

For trajectory calculation and optimization, we have used InTrance, a program that uses evolutionary neurocontrol to calculate near-globally optimal trajectories. This method is based on a combination of artificial neural networks (ANNs) with evolutionary algorithms (EAs). ENC attacks low-thrust trajectory optimization problems from the perspective of artificial intelligence and machine learning. Here, it can only be sketched how this method is used to search for optimal low-thrust trajectories. The reader who is interested in the details of the method is referred to Refs. [11, 12]. The problem of searching an optimal low-thrust trajectory $\boldsymbol{x}^\star[t] = (\boldsymbol{r}^\star[t], \dot{\boldsymbol{r}}^\star[t])$ – where the symbol "$[t]$" denotes the time history of the preceding variable and the symbol "$\star$" denotes its optimal value – is equivalent to the problem of searching an optimal thrust vector history $\boldsymbol{F}^\star[t]$. Within the context of machine learning, a trajectory is regarded as the result of a spacecraft steering strategy S that maps the problem relevant variables (the spacecraft state $\boldsymbol{x}$ and the target state $\boldsymbol{x}_\mathrm{T}$) onto the thrust vector, $\mathsf{S} : \{\boldsymbol{x}, \boldsymbol{x}_\mathrm{T}\} \subset \mathbb{R}^{12} \mapsto \{\boldsymbol{F}\} \subset \mathbb{R}^3$. This way, the problem of searching $\boldsymbol{x}^\star[t]$ is equivalent to the problem of searching (or learning) the optimal spacecraft steering strategy $\mathsf{S}^\star$. An ANN may be used as a so-called neurocontroller (NC) to implement spacecraft steering strategies. It can be regarded as a parameterized function $\mathsf{N}_{\boldsymbol{\pi}}$ (the network function) that is – for a fixed network topology – completely defined by the internal parameter set $\boldsymbol{\pi}$ of the ANN. Therefore, each $\boldsymbol{\pi}$ defines a spacecraft steering strategy $\mathsf{S}_{\boldsymbol{\pi}}$. The problem of searching $\boldsymbol{x}^\star[t]$ is therefore equivalent to the problem of searching the optimal NC parameter set $\boldsymbol{\pi}^\star$. EAs that work on a population of strings can be used for finding $\boldsymbol{\pi}^\star$ because $\boldsymbol{\pi}$ can be mapped onto a string $\boldsymbol{\xi}$ (also called chromosome or individual). The trajectory optimization problem is solved when the optimal chromosome $\boldsymbol{\xi}^\star$ is found. An evolutionary neurocontroller (ENC) is a NC that employs an EA for learning (or breeding) $\boldsymbol{\pi}^\star$. ENC was implemented within a low-thrust trajectory optimization program called InTrance, which stands for **In**telligent **Tra**jectory optimization using **n**euro**c**ontroller **e**volution. InTrance is a smart global trajectory optimization method that requires only the target body/state and intervals for the initial conditions (e.g., launch date, hyperbolic excess velocity, initial propellant mass, etc.) as input to find a near-globally optimal trajectory for the specified problem. It works without an initial guess and does not require the attendance of a trajectory optimization expert. However, because InTrance (as it is characteristic of evolutionary algorithms) does not use any gradient information, it does not permit a fast convergence in the nearness of an optimum and can therefore not locate optimal solutions exactly, but only near-optimal solutions.

4.1.1 SEP Results

Mission Architecture SCR: For the SEP mission architecture with the chemical return, SCR, only the Earth-Fortuna transfer has to be calculated (see Fig. 4 for the Fortuna-Earth return trajectory with the chemical SRV). The transfer trajectory is shown in Fig. 8. The transfer takes 1083 days and consumes

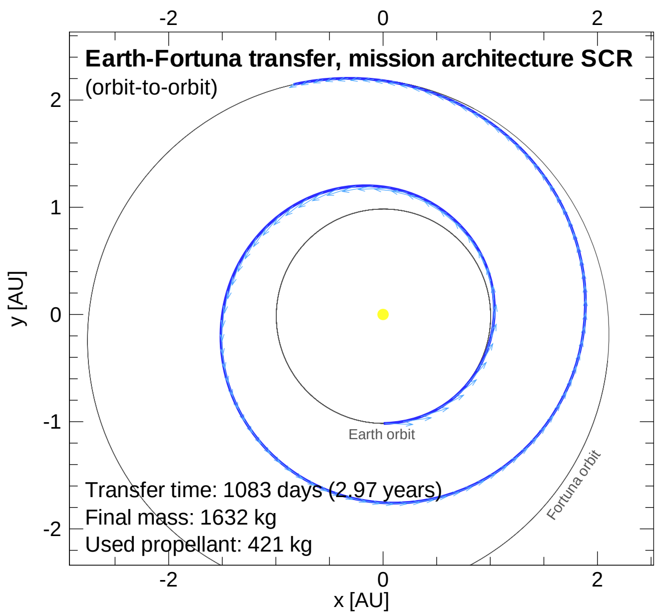


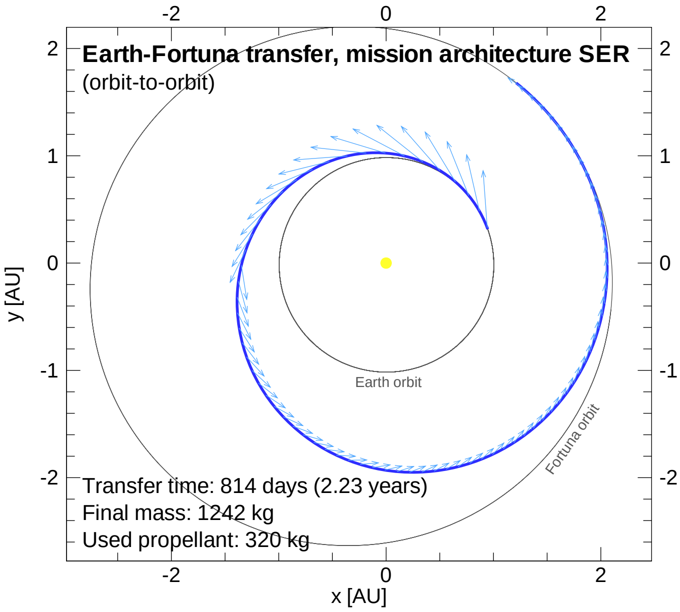
Figure 8: Earth-Fortuna orbit-to-orbit transfer trajectory for mission architecture SCR.

421 kg of Xe-propellant. Thus, to deliver the final mass of 1632 kg, a launch mass of 2053 kg is required.

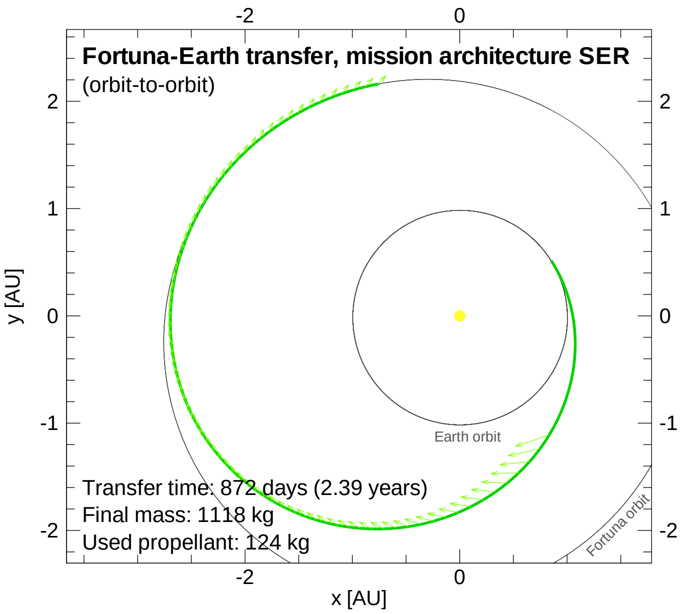
Mission Architecture SER: For the all-SEP mission architecture, SER, both transfer legs have to be

7th International Symposium on Launcher Technologies, Barcelona, Spain, 02-05 April 2007

calculated. The first leg is shown in Fig. 9(a) and the return leg is shown in Fig. 9(b). The transfer



(a) Earth-Fortuna transfer



(b) Fortuna-Earth transfer

Figure 9: Orbit-to-orbit transfer trajectories for mission architecture SER.

from Earth to Fortuna takes 814 days and consumes 320 kg of Xe-propellant. This is only 0.9% more than the propellant mass estimated with the approximation of Eq. (2). To deliver the final mass of 1242 kg (1118 kg spacecraft dry mass and 124 kg propellant for Earth return, which is 14.8% more than the propellant mass estimated with the approximation of Eq. (2)), a launch mass of 1562 kg is necessary. No rendezvous is required at Earth but the spacecraft is allowed to arrive with an hyperbolic excess velocity of 6.4 km/s, so that the re-entry velocity is less than 12.9 km/s. The actual hyperbolic excess velocity at Earth return is less, 5.54 km/s.

4.1.2 NEP Results

Mission Architecture NCR: For the NEP mission architecture with the chemical return, NCR, again only the Earth-Fortuna transfer has to be calculated (see Fig. 4 for the Fortuna-Earth return trajectory with the chemical SRV). The transfer trajectory is shown in Fig. 10. The transfer takes 762 days and

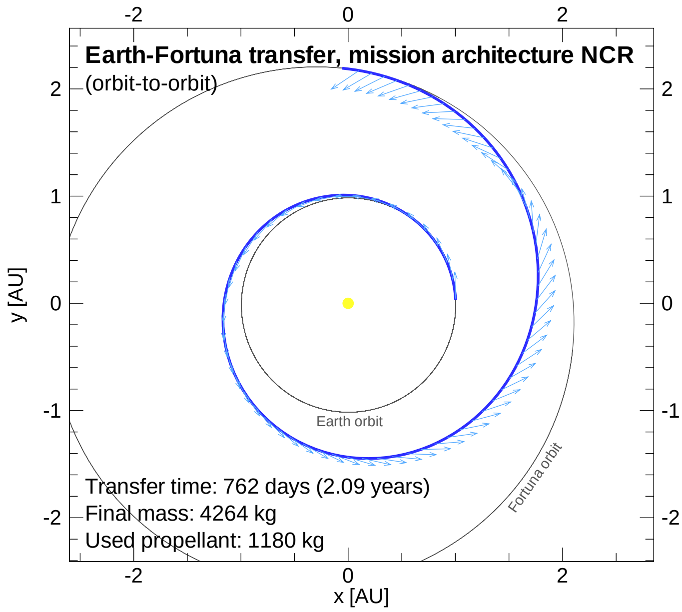


Figure 10: Earth-Fortuna orbit-to-orbit transfer trajectory for mission architecture NCR.

consumes 1180 kg of Xe-propellant. This is 7.3% more than the propellant mass estimated with the approximation of Eq. (2). To deliver the final mass of 4264 kg, a launch mass of 5444 kg is required.

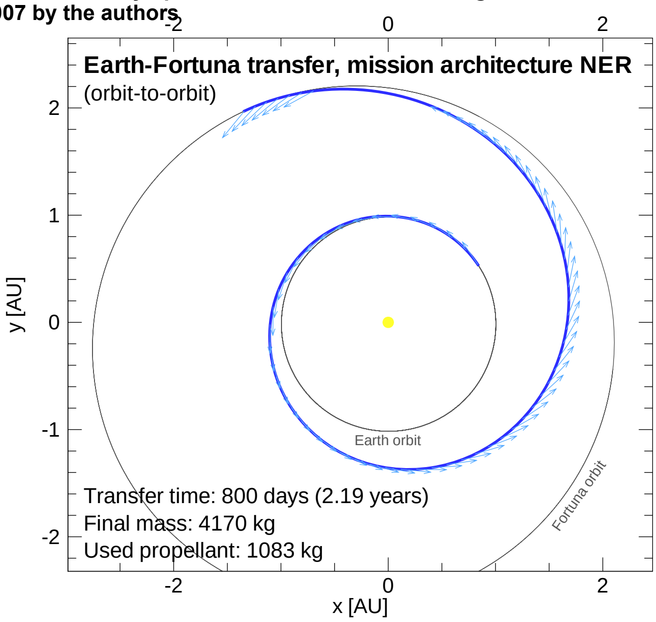
Mission Architecture NER: For the all-NEP mission architecture, NER, again both transfer legs have to be calculated. The first leg is shown in Fig. 11(a) and the return leg is shown in Fig. 11(b). The transfer from Earth to Fortuna takes 800 days and consumes 1083 kg of Xe-propellant. This is only 0.7% more than the propellant mass estimated with the approximation of Eq. (2). To deliver the final mass of 4170 kg (3801 kg spacecraft dry mass and 369 kg propellant for Earth return), a launch mass of 5253 kg is necessary. The hyperbolic excess velocity at Earth return is 6.34 km/s.

4.1.3 Comparison of the SEP and NEP Results

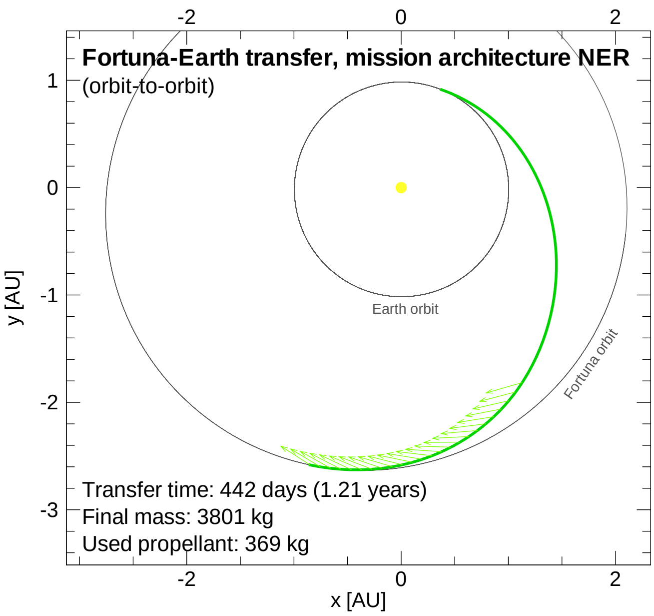
Table 5 shows the resulting orbit-to-orbit masses and transfer times for the four investigated mission architectures. One can see that the masses do not greatly differ from the analytically approximated values in Tables 3 and 4.

Due to the reduced solar power at larger solar distances and with increasing degradation, the total transfer time for mission architecture SCR is 27%

7th International Symposium on Launcher Technologies, Barcelona, Spain, 02-05 April 2007



(a) Earth-Fortuna transfer



(b) Fortuna-Earth transfer

Figure 11: Orbit-to-orbit transfer trajectories for mission architecture NER.

longer than for the fastest NEP architecture (NCR) and the total transfer time for option SER is 42% longer. Due to the large mass of the nuclear reactor, however, the launch mass for mission architecture NCR is 2.5 times larger than for the lightest SEP architecture (SEP) and the launch mass for option NER is 3.4 times larger. Because the launch mass *and* the total transfer time are subject to optimization, we are dealing here with a multi-objective optimization problem. Probably the best way to tackle this kind of problems is to borrow the concept of Pareto-optimality from the economical sciences. According to this concept, every solution is Pareto-optimal that is not dominated by some other solution that is better in *all* objectives. Thus a Pareto-optimal solution can only be improved with respect to some single objective at the expense of at least one other objective.

Table 5: Masses and transfer durations for the four mission architectures

		SCR	SER	NCR	NER
Spacecraft dry mass	[kg]	1632	1118	4264	3801
Xe propellant mass (Earth-Fortuna)	[kg]	421	320	1180	1083
Xe propellant mass (Fortuna-Earth)	[kg]	-	124	-	369
Spacecraft launch mass	[kg]	2053	1562	5444	5253
Transfer time (Earth-Fortuna)	[days]	1083	814	762	800
Transfer time (Fortuna-Earth)	[days]	426	872	426	442
Total transfer time	[days]	1509	1686	1188	1242

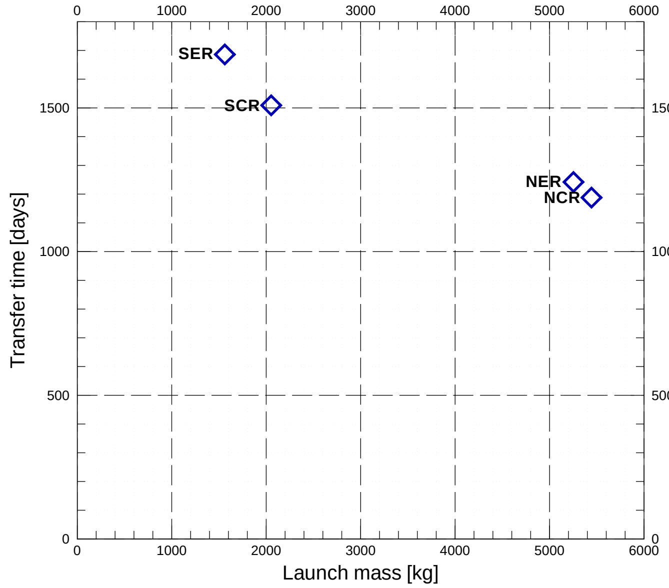


Figure 12: Comparison of the launch masses and the total transfer times for the four mission architectures.

Of course objectives could be treated against each other by defining a cost function Q, e.g. Q = mission duration $\times$ launch mass. The result of such an objective reduction technique is a single solution, that does, however, typically not reflect the possible compromises between the different objectives. One can see from Fig. 12 that no mission architecture is dominated by one of the other architectures and thus all are optimal in the Pareto-sense. To select a "the best" solution, further analysis would have to be performed with respect to cost, which is not within the scope of this paper (generally, larger launch masses tend to increase launch costs and longer transfer times tend to increase ground operation costs). Nevertheless, we can state here from our preceding analysis that the SEP mission architectures tend to yield longer flight times but also much lower launch masses.

We remember here that we have optimized the preceding trajectories for orbit-to-orbit transfers to get an unbiased comparison irrespective of the actual constellation of Earth and Fortuna at the given launch window. For the real mission, body-to-body transfers have to be calculated, which give penalties in terms of propellant and transfer time to account for the actual (non-optimal) phasing of Earth-and Fortuna within the selected launch window. This will be shown in the next section for mission architecture SER.

4.2 Results for Body-To-Body Transfers

The optimization of a mission for each of the four architectures within the given launch window (01 Jan 2012 – 31 Dec 2015) is not within the scope of *this* paper but left to further research. Basically, the Pareto-optimal front of solutions has to be calculated for all four architectures. The mission optimization task, however, is much more difficult than before in section 4.1 because, in contrast to the orbit-to-orbit transfers, the combined propellant mass and the total mission duration have to be minimized. One trajectory leg can not be optimized without considering the other leg and "the penalty" in terms of additional propellant and additional transfer time has to be optimally distributed between both transfer legs. Here, we only want to show that an all-SEP mission is feasible, without claiming that our scenario is optimal or even near-optimal. The trajectories for our proposed potential mission scenario are shown in Fig. 13 and the mass breakdown is given in Table 6.

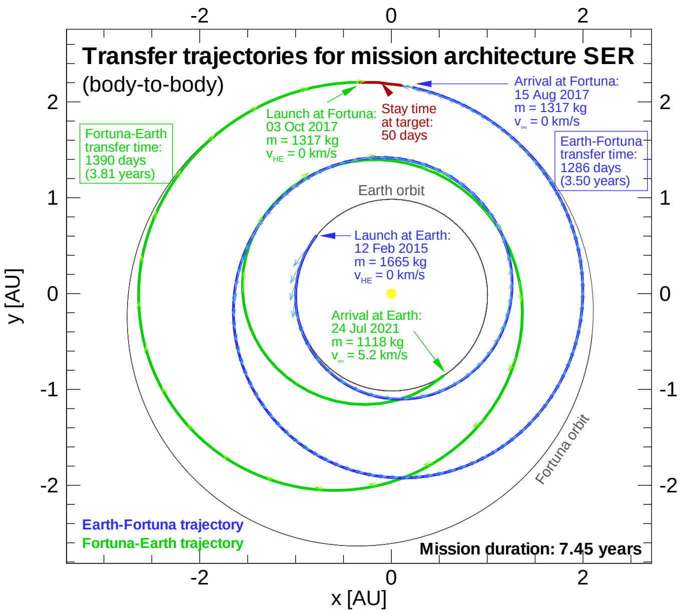


Figure 13: Body-to-body transfer trajectories for mission architecture SER.

The insertion of the SEP into its interplanetary trajectory takes place on 12 Feb 2015. After its 3.5-year cruise, the spacecraft arrives at Fortuna on 15 Aug 2017. There, the surface package is dropped and the ground operations team has 50 days to sample the asteroid and to bring the samples back to the spacecraft. The spacecraft leaves the asteroid on 03 Oct 2017 to return the collected samples to Earth. After its 3.8-year cruise, it arrives at Earth with an hyperbolic excess velocity of 5.2 km/s and releases its 21-kg re-entry capsule, which enters the atmosphere with a velocity of $\sqrt{5.2^2 + 11.2^2}$ km/s = 12.35 km/s,

Table 6: Comparison of masses and transfer durations for mission architecture SER (o2o = orbit-to-orbit, b2b = body-to-body)

		o2o	**b2b**
"Payload" mass	[kg]	225	225
Bus mass	[kg]	415	415
Spacecraft dry mass without EP module	[kg]	640	640
Power unit mass	[kg]	245	245
Propulsion units mass	[kg]	85	85
Structure + tank + gimbal system mass	[kg]	148	148
EP module dry mass	[kg]	478	478
Spacecraft dry mass	[kg]	1118	1118
Xe propellant mass (Earth-Fortuna)	[kg]	320	348
Xe propellant mass (Fortuna-Earth)	[kg]	124	199
Spacecraft launch mass	[kg]	1562	1665
Transfer time (Earth-Fortuna)	[days]	814	1286
Transfer time (Fortuna-Earth)	[days]	872	1390
Total transfer time	[days]	1686	2676

less than NASA's Stardust capsule [9, 10].

Fig. 13 and Table 6 show that for the body-to-body transfer the mass penalty is moderate (7%) whereas the transfer time penalty is considerable (59%). The mass that has to be injected is 1665 kg. The current capability of the Soyuz/Fregat for $C_3 = 0\,\mathrm{km}^2/\mathrm{s}^2$ is 1600 kg. It can be expected that until 2015 an upgraded version of the Soyuz/Fregat with a larger launch capacity is available and/or that the mass of our SEP spacecraft can be reduced by using more advanced light-weight technologies. Another option would be to use the SEP system also for the final Earth escape phase, so that the C_3 from the launcher's upper stage can be $< 0\,\mathrm{km}^2/\mathrm{s}^2$.

5 Summary and Conclusions

For a sample return mission to the main belt asteroid (19) Fortuna, the overall required ΔV is at least 13.7 km/s, which renders this mission difficult for any chemically propelled spacecraft. Considering both a nuclear and a solar electric power source, we have investigated the prospects of different electric propulsion architectures. The solar electric concepts are based on existing hardware, conventional silicon solar cells and the RIT-22 ion thruster. The nuclear electric concepts are based on the envisaged Russian Topaz-25 nuclear reactor and the RIT-22. For both power sources, we have investigated an all-electric mission architecture and an architecture with a chemical sample return vehicle. From basic ΔV-considerations and optimized orbit-to-orbit transfers, we have found that the SEP mission architectures tend to yield longer flight times (more than 4.1 years instead of less than 3.5 years) but also much lower launch masses (less than 2.1 tons instead of more than 5.2 tons). In addition, the chemical return architectures had slightly larger launch masses but also slightly faster transfer times than the all-electric architectures. To demonstrate the feasibility of the all-solar electric mission, given the fact that a nuclear propulsion system is currently not available in Europe, we have also considered the actual constellations of the planetary bodies for a launch between 2012 and 2015. Our resulting spacecraft has a mass of 1665 kg, which has to be injected into an interplanetary transfer trajectory with zero hyperbolic excess velocity. The mission duration is 7.45 years. Our results show that a main belt asteroid sample return mission with a solar electric propelled spacecraft is feasible, despite the decreased solar power availability far from the sun.

6 Acknowledgements

The authors thank Ralph Kahle (German Space Operations Center, Flight Dynamics Group) for providing the code for the calculation of the chemical transfers.

References

[1] Advanced interplanetary missions using nuclear-electric propulsion. Technical report, Joint Study Group on Advanced Interplanetary Missions Using Nuclear-Electric Propulsion, University of Giessen, 1995.

[2] C. T. Russell, A. Coradini, U. Christensen, M. C. De Sanctis, W. C. Feldman, R. Jaumann, H. U. Keller, A. S. Konopliv, T. B. McCord, L. A. McFadden, H. Y. McSween, S. Mottola, G. Neukum, C. M. Pieters, T. H. Prettyman, C. A. Raymond, D. E. Smith, M. V. Sykes, B. G. Williams, J. Wise, and M. T. Zuber. Dawn: a journey in space and time. *Planetary and Space Science*, 52:341–365, 2004.

[3] M. D. Rayman, T. C. Fraschetti, C. A. Raymond, and C. T. Russell. Dawn: A mission in development for exploration of main belt asteroids Vesta and Ceres. *Acta Astronautica*, 58(11):605–616, 2006.

[4] P. Ferri. Mission operations for the new Rosetta. Bremen, Germany, September/October 2003. 54th International Astronautical Congress. IAC-03-Q.5.01.

[5] R. L. McNutt, S. C. Solomon, R. E. Gold, and the MESSENGER Team Leary, J. C. The MESSENGER mission to Mercury: Development history and early mission status. *Advances in Space Research*, 38(4):564–571, 2006.

[6] S. Fornasier, M. Lazzarin, C. Barbieri, and M. A. Barucci. Spectroscopic comparison of aqueous altered asteroids with CM2 carbonaceous chondrite meteorites. *Astron. Astrophys. Suppl. Ser.*, 135:65–73, 1999.

[7] A. Storrs, B. Weiss, B. Zellner, W. Burleson, R. Sichitiu, E. Wells, C. Kowal, and D. Tholen. Imaging observations of asteroids with Hubble Space Telescope. *Icarus*, 137:260–268, 1999.

[8] JPL solar system dynamics group website.

[9] P. N. Desai, R. A. Mitcheltree, and F. McNeil Cheatwood. Entry trajectory issues for the Stardust sample return capsule. Arcachon, France, March 1999. International Symposium on Atmospheric Reentry Vehicles and Systems.

[10] P. N. Desai, R. A. Mitcheltree, and F. McNeil Cheatwood. Sample return missions in the coming decade. Rio de Janeiro, Brazil, 2000. 51st International Astronautical Congress. IAF-00-Q.2.04.

[11] B. Dachwald. Evolutionary neurocontrol: A smart method for global optimization of low-thrust trajectories. Providence, USA, August 2004. AIAA/AAS Astrodynamics Specialist Conference. AIAA Paper 2004-5405.

[12] B. Dachwald. Optimization of very-low-thrust trajectories using evolutionary neurocontrol. *Acta Astronautica*, 57(2-8):175–185, 2005.