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Concept Study of a Beamed Energy Propulsion Craft as Workhorse of a Future Space Transportation Architecture

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This paper defines three different scenarios, describing the potential future commercial space transportation market. Based on the discussed scenarios boundary conditions for future space transportation systems are established. The paper gives a brief overview over different types of future launcher systems with Beamed Energy Propulsion (BEP) engines. It concentrates on thermal and plasma propulsion and analyses a launcher system in terms of performance in the individual scenarios. A cost estimation based on the results of an ascent trajectory analysis serves to evaluate the system's performance.

Nomenclature

α	angle of attack, rad.
Δv	required velocity change, m/s
ϵ	nozzle expansion ratio, -
η_{pl}	payload scaling factor, -
γ	path angle, rad.
ξ	mass fraction, -
A	cross sectional area, m^2
C_D	drag ratio, -
C_m	momentum coupling coefficient, N/W
c_{eff}	effective exhaust velocity, m/s
E	energy (beam pulse), J
F	thrust, N
g	gravitational acceleration, m/s^2
h	altitude, m
I	specific impulse, s
m	mass, kg
P	power (beam), W
p_c	pressure (chamber), Pa
R_E	radius (Earth), m
t	time, s
v	velocity, m/s

I. Introduction

The high specific cost for access to space currently forms a bottleneck preventing the expansion of humanity into space. Extrapolations of the development of specific cost of transportation, based on the experience of the past 40 years, does not suggest any foreseeable changes.[1] A reduction of the specific cost for transportation may however open up multiple new business opportunities in space, enable a sustained lunar outpost and may ultimately lead to the colonization of space. The BEP launcher is an innovative concept, where the power source for heating of the propellant is not carried in the vehicle, but is left on ground. The vehicle carries only the propellant itself, and in case of atmospheric operation, it may even utilize the ambient air as propellant. The technical feasibility of the concept has been demonstrated in the past for small scale crafts in laboratories throughout the world.

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II. Launch Scenarios

A. Scenario Description

When evaluating the future on the basis of scenarios it is not important to define a scenario that actually reflects the real future, as it will evolve. The main task is to define different scenarios, which together span a realm of potential futures, including the real future. One can imagine the scenarios to be the extreme corner points, with the actual future to be found in the shaded area between the scenarios. Therefore the actual future will possibly include elements of many scenarios. The following scenarios and associated transportation demand (Earth to orbit) are considered:

1. Scenario I: Business as Usual (Baseline Scenario)

According to the current launch estimation of the US Federal Aviation Administration the trend in Geostationary Orbit (GEO) satellite masses is no longer increasing.[2] The average mass of GEO satellites is around 4.5 metric tons and the annual launch demand is roughly 100 metric tons. The annual commercial launch demand for Low Earth Orbit (LEO) is about 30 metric tons. Figure 1 shows the history of commercial launches and the current estimations for future launch demand. Based on this data, the first scenario assumes a constant market for commercial payload delivery to LEO and GEO.

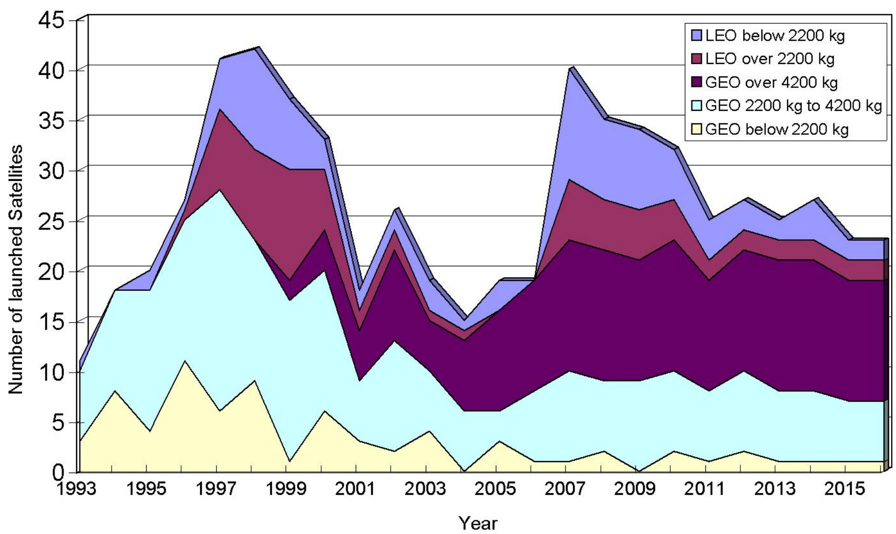

Figure 1. Historical and future development of commercial satellite launches

2. Scenario II: Microcosm

Moore's law predicts doubling of transistors per integrated circuit every 2 years. This trend can be approximately confirmed since 1970 and is currently believed to continue until at least 2024.[3] The constantly increasing computational power allows for ever smaller, more autonomous electronic devices, as we can witness in our daily lives. Several contemporary studies predict a variety of future space applications to be fulfilled by swarms of micro- (up to 100 kg) nano- (up to 10 kg) or pico-satellites (up to 1 kg). The baseline assumption is 10% of the market demand, in terms of total payload mass, as in the first scenario (*Business as Usual*). The typical payload mass is assumed to be 10 kg.

3. *Scenario III: Space Opera*

Currently space agencies throughout the world are developing plans for a manned return to the moon in order to establish a permanent human presence on the moon by mid of this century. Since the 1960s several studies about lunar base design and lunar development have been published. The studies differ substantially in the estimated transportation demand, mostly because of the difference in size of the projected lunar base concepts. From the studies a median value for a lunar base design can be assumed to be at least 100 Mg payload delivery to the lunar surface per year with a size of the lunar modules of at least 10 Mg.[4,5] This is assumed to last for 25 years or more. Given the Δv requirements for the transport from LEO to Low Lunar Orbit (LLO) and the descent from LLO to the lunar surface a mass increase by a factor of 10 (i.e. 1 kg on the Moon requires 10 kg in LEO) seems reasonable. It is arbitrarily assumed that additional communication infrastructure to support the exploration activities will require 50% of today's launch demand to GEO or energetically similar orbits.

B. Scenario Implications

Given the description of the assumed scenarios above, the transportation demand listed in Table 1 is required. The payload increment defines the typical mass of a payload. Higher or lower payload mass delivered by the launcher system will reduce the efficiency due to required rendezvous or clustering mechanisms. This is represented by the payload efficiency η_{pl} as shown in Table 2. Obviously manned access to space does not allow payload clustering or separation. It is assumed that in each scenario separate man-rated launcher-systems exist and are operated. For transport of astronauts, reliability is obviously of much higher concern than the specific cost of transportation. Table 1 lists the launch demand for a period of 10 years, which will be considered as the life-cycle-time for the purpose of the cost estimation in section IV.

	Transportation demand LEO	**LEO Payload increment**	**Transportation demand GEO**	**GEO Payload increment**
Business as Usual	30 Mg	2 Mg	100 Mg	5 Mg
Microcosm	3 Mg	0.01 Mg	10 Mg	0.01 Mg
Space Opera	1000 Mg	100 Mg	150 Mg	5 Mg

Table 1. Scenario annual transportation demand and payload increment (10 years)

The listed payload increment refers to the standard mass of payload. If the launcher system delivers a higher payload, additional mass will be required for payload clustering and separation mechanisms. The listed efficiency $\eta_{pl} = 0.9$ is based on the payload mass difference of the *Ariane 5* launcher, when comparing single and dual launches. If on the other hand the delivered payload mass is lower than the system mass, the system has to be delivered in batches. This will require additional mass for docking, rendezvous etc. The efficiency $\eta_{pl} = 0.5$ is conservatively based on the ratio of payload and fuel to structure of the ATV vehicle. This assumes the payload and fuel brought to the International Space Station (ISS) to be the nominal payload and the structure to be the required mass for rendezvous, docking etc.

	payload efficiency
High payload mass ($m_{pl} >> m_{incr}$)	$\eta_{pl} = 0.9$
Low payload mass ($m_{pl} << m_{incr}$)	$\eta_{pl} = 0.5$

Table 2. Payload efficiency

III. BEP Launcher

A. Different Types of BEP Launchers

A variety of different concepts for space transportation systems with BEP engines were proposed in past studies by several different authors. Figure 2 provides a rough overview.

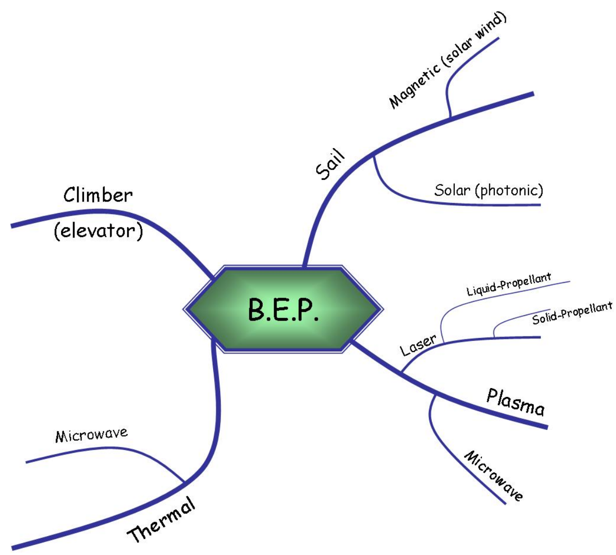

Figure 2. Different types of BEP vehicle concepts

The main branches, depicted in Figure 2 are:

Climber: The climber type is a vehicle for a *space elevator* type transportation system. The climber will convert beamed energy into mechanical energy used for climbing the tether to the orbital destination station.

Sail: The beamed energy is reflected by a sail. The resulting radiation pressure exacts a force on the vehicle. The *Solar Sail* is a specific concept of this category, using the pressure of the solar light for propellantless propulsion.

Thermal: A heat exchanger in the vehicle is heated by beamed energy. The heat is transferred to on-board propellant, which is ejected in a classical convergent-divergent nozzle.

Plasma: The incoming beamed energy is focused by a mirror system. In the focal point on-board propellant or ambient air is converted into plasma. The ejected plasma propels the vehicle by momentum exchange.

While the *Sail* is a feasible approach for in-space applications only, the *Climber* is a radically different concept and difficult to compare with traditional launch systems. This study concentrates on *Plasma* propulsion. The basic results and conclusions are also valid for *Thermal* propulsion, due to the similarity in performance and hardware requirements.

B. Launcher Concept

Figure 3 shows the propulsion concept of the discussed vehicle. The electromagnetic beam is focused by the reflecting parabolic contour of the vehicle. The propellant, which may be ambient air, is converted to plasma by the energy deposit (state **2** in Figure 3). The plasma front propagates through the nozzle tube and continuously absorbs energy from the ongoing microwave pulse (state **3**,**4** in Figure 3). Subsequent to the plasma ejection the nozzle tube is refilled by propellant.

The technologies for both the laser BEP and the microwave BEP launch systems are currently at a similar Technology Readiness Level (TRL) of about 3.[6] Different to laser BEP, microwave emitters can be obtained with a significantly lower financial investment.[7] The use of phased arrays allows coupling of low power, low cost microwave emitters to create a focused high power microwave beam for BEP.

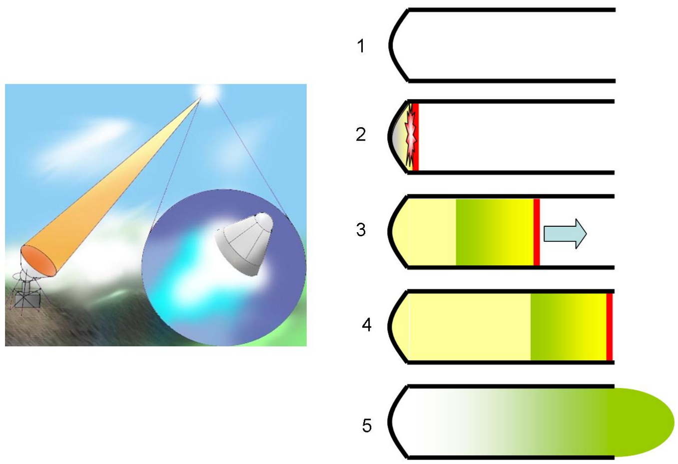

Figure 3. Propulsion concept

1. Vehicle Data

Table 3 lists the parameters used to define the vehicle and beam base for the trajectory analysis and cost estimation.

Vehicle cross section area	A	$=$	0.5 m^2
Vehicle liftoff mass	m_0	$=$	250 kg
Vehicle structure mass	m_{str}	$=$	10 kg
Mass flow (*rocket-mode*)	$\dot{m}$	$=$	1 kg/s
Specific impulse	I_{sp}	$=$	600 s
Beam repetition frequency	f	$=$	1000 Hz
Beam pulse duration	τ	$=$	350 μs
Beam power	P	$=$	1 GW

Table 3. BEP vehicle and beam base parameters

C. Launch Strategy

Figure 4 shows the proposed ascent trajectory as discussed in detail in a previous study.[8] The vehicle will launch on a vertical trajectory to allow visibility from the station throughout the ascent phase. The passive stabilization of the vehicle will allow it to keep it's position centered on the beam, even when the increasing altitude would cause a shift due to the faster angular rotation of the base on the surface. This effect will provide a lateral velocity increase for the vehicle.

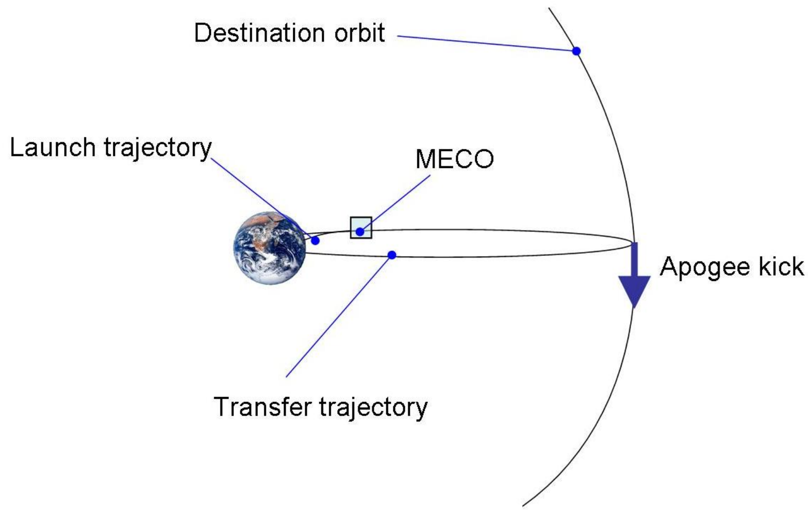

Figure 4. Launch trajectory

D. Trajectory Analysis

Based on previous studies[9] the vehicle is considered to operate in two flight modes: the *air-breathing-mode* and the *rocket-mode*. In the *air-breathing-mode*, air is used as propellant and the nozzle tube is filled by ambient air before each pulsed operation. The *rocket-mode*, using on-board propellant, starts during atmospheric flight, when the ambient air density does no longer support an efficient propulsion. In this study the switching point between *air-breathing-mode* and *rocket-mode* occurs at a flight altitude of $h = 12$ km.

An important parameter for BEP engine performance evaluation is the momentum coupling coefficient (C_m), as defined in Equation (1). For the atmospheric flight of this analysis the momentum coupling coefficient varies between $C_m = 280$ N/MW (at sea level) and $C_m = 150$ N/MW (at $h = 12$ km), depending on the ambient pressure.

$$C_m = \frac{\int_0^t \ F \ dt}{E} \tag{1}$$

The launch trajectory is simulated by numerical integration of the governing equations. The numerical analysis uses the Euler method for integration. Equations (2) and (3) are integrated for the trajectory simulation:

$$\frac{dv_x}{dt} = \cos(\alpha + \gamma) \ (I(i,h) \ g_0 \ \dot{m} + P \ C_m(i,h)) - \cos(\gamma)\frac{\rho(h)}{2}({v_x}^2 + {v_y}^2) \ C_D(i) \ A(i) \tag{2}$$

$$\begin{aligned} \frac{dv_y}{dt} &= \sin(\alpha + \gamma) \ (I(i,h) \ g_0 \ \dot{m} + P \ C_m(i,h)) - \sin(\gamma)\frac{\rho(h)}{2}({v_x}^2 + {v_y}^2) \ C_D(i) \ A(i) \\ &\quad + \left(\frac{(v_x + v_{ground})^2}{R_E + h} - g\right) \ m \end{aligned} \tag{3}$$

The equations describe propulsion, drag and gravity forces acting on the vehicle. The index i is the number of the stage/phase. The thrust force consists of two terms, the first one being used in during *rocket-mode* operation, the second one during *air-breathing-mode*. To correctly model the thrust just one term is used in each flight mode (i.e. I is set to 0 during the *air-breathing-mode* and C_m is set to 0 during the *rocket-mode*).

1. Orbit Transfer

Since the vehicle is at Main Engine Cut-Off (MECO) on a ballistic impact trajectory, a kick-maneuver is necessary to reach the destination orbit. It has been suggested to use a different on-board propulsion system to obtain the final orbit.[9] Figure 5 shows a possible trajectory for orbit insertion. In this case the apogee of ballistic trajectory lies beyond the destination orbit. A bi-elliptical transfer is used to reach the destination orbit in an efficient way.

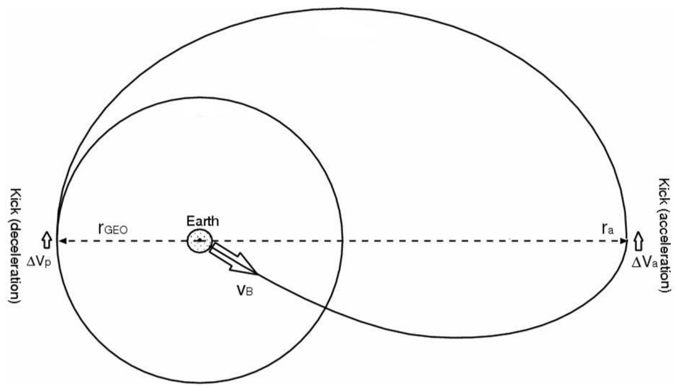

Figure 5. Orbital overshoot and bi-elliptical transfer[9]

Due to the OBERTH effect this trajectory can help to increase the payload, at the cost of an increased flight time. Figure 6 shows the resulting required Δv and payload for a 250 kg vehicle with a specific impulse of $I_{vac} = 600$ s. The trade-off between orbital overshoot and required trajectory change Δv is of high relevance for a system using different propulsion concepts (with different specific impulses) for main propulsion and orbit insertion. The system discussed here uses the same BEP propulsion for both orbit insertion and main propulsion, in order to keep the vehicle design as simple and cost effective as possible. In order to better compare the performance for LEO and GEO orbit, a trajectory with no orbital overshoot is considered to estimate the system performance. Therefore the proposed vehicle has a design reserve, which can be tapped into by further optimizing the trajectory.

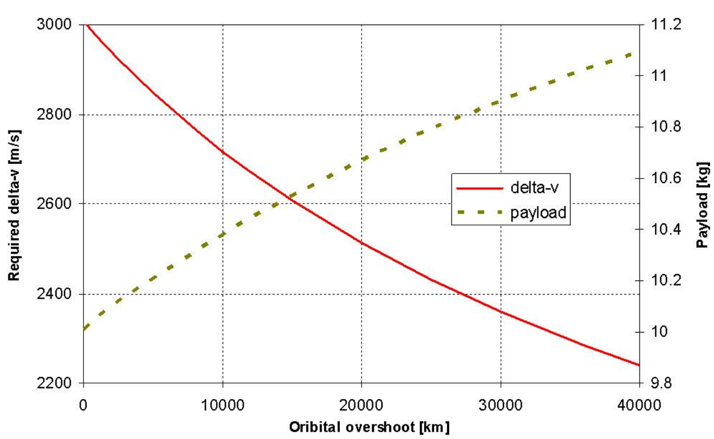

Figure 6. Required Δv and resulting payload vs. orbital overshoot for a GEO trajectory

E. System Performance

Table 4 shows the achieved payload mass for a system as specified in Table 3.

	LEO	GEO
Orbit altitude	200 km	35786 km
Required Δv (orbital insertion)	7468 m/s	3014 m/s
Total power consumption	42.81 MWh	47.93 MWh
Payload mass	36.3 kg	10 kg

Table 4. System performance for different destination orbits

IV. Cost Estimation

This section offers a cost comparison with conventional launcher systems. Different scenarios produce different cost effectiveness results. For the analysis a total life cycle of 10 years is assumed. The cost analysis results in an average value of cost per flight for the considered life cycle. The actual cost is therefore initially higher and decreases significantly towards the end of the time interval. Some baseline assumptions are made for the cost comparison:

1. The required cost for research and development are not included in the cost analysis. They are assumed to be an institutionalized investment, as in the case of conventional launcher systems.
2. Similarly the cost for construction of the required infrastructure is not part of the cost estimate. It is also assumed to be an institutionalized investment.
3. The launch cost will reflect the required maintenance and refurbishment related to the infrastructure.

Baseline comparison to evaluate the BEP launcher performance is the conventional launcher of today (with a specific launch cost of 10000 $/kg to LEO and 20000 $/kg to GEO). The following elements are considered for the cost estimation:

Electricity: A cost of 0.06 $ / kWh is assumed, based on present cost of electricity.

Propellant: A cost of 1 $ / kg is assumed, including fuel storage and handling. The study assumes the propellant to be liquid argon.[8]

Installation refurbishment: The beam base for an average power transmitted to the vehicle of 1 GW is assumed to be an investment of 10 Million $.[6] It is assumed that for each mission 0.01 % of the installation cost will have to be spent for refurbishment, spare parts etc.

Vehicle manufacturing: Cost estimation is based on the *Transcost*-method.[10] As Cost Estimation Relation (CER) for the first unit cost of the BEP launcher the CER for simple ballistic stages has been chosen. This results in first unit cost of 834000 $. It is assumed that mass production benefits will reduce the cost of an increased number of produced items below the initial manufacturing cost of the first items. A standard approach is the assumption of a cost reduction by a specific amount for each doubling of produced items. The n$^{\text{th}}$ item will incur the fractional cost: $f = n^{\frac{\ln 0.75}{\ln 2}}$

Launch and mission operation: Cost estimation is also based on the *Transcost*-method. The CER takes vehicle lift-off mass, launch rate, and vehicle complexity into account.

A. Vehicle Performance Evaluation

Table 5 and Table 6 show a cost breakdown for the launch to LEO and GEO respectively. In all scenarios the launch cost is about one order of magnitude less than the conventional cost. Figure 7 shows a comparison of conventional and estimated BEP transportation cost. Most notably the total cost of the scenario *Space Opera*, using BEP is lower than the conventional cost of the baseline scenario (*Business as Usual*). This indicates that a technology breakthrough, enabling the discussed BEP system, may easily lead to more ambitious space endeavors.

	Business as Usual	*Microcosm*	*Space Opera*
Electricity:	2569 $	2569 $	2569 $
Propellant:	204 $	204 $	204 $
Installation refurbishment:	1000 $	1000 $	1000 $
Vehicle manufacturing:	8694 $	29889 $	4919 $
Launch and mission operation:	581 $	13364 $	137 $
Specific cost	**362 $/kg**	**1306 $/kg**	**245 $/kg**

Table 5. Cost per flight, LEO destination

	Business as Usual	*Microcosm*	*Space Opera*
Electricity:	2876 $	2876 $	2876 $
Propellant:	230 $	230 $	230 $
Installation refurbishment:	1000 $	1000 $	1000 $
Vehicle manufacturing:	8694 $	29889 $	4919 $
Launch and mission operation:	581 $	13364 $	137 $
Specific cost	**1338 $/kg**	**4736 $/kg**	**916 $/kg**

Table 6. Cost per flight, GEO destination

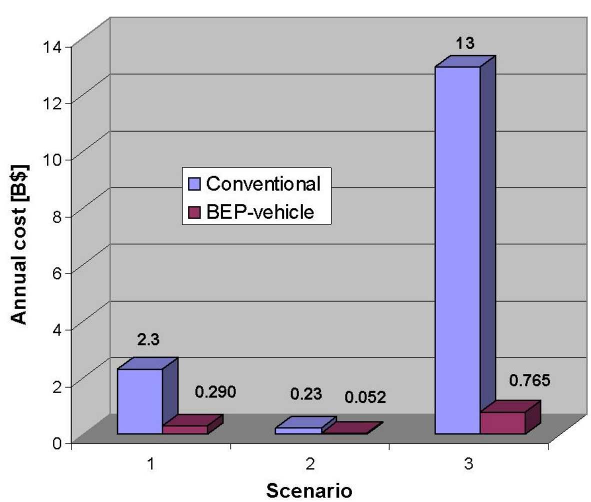

Figure 7. Total annual transportation cost

V. Far Future Applications

An application with high relevance for Scenario III (*Space Opera*) is the shuttle service between a LLO space station and a lunar surface installation. Several modern studies of lunar installations rely on the use of In-Situ Resource Utilization (ISRU) for supply of lunar installations. The lunar soil contains a substantial amount of oxygen. Samples taken by the missions Apollo 11-17 and Luna 16 and 20 found 39.7 % to 44.6 % of oxygen contained in the soil.[11] Given the abundance of oxygen, it is worth being considered as propellant. Figure 8 shows a future space transportation architecture, arbitrarily envisaged for the year 2100, using a BEP lunar shuttle.

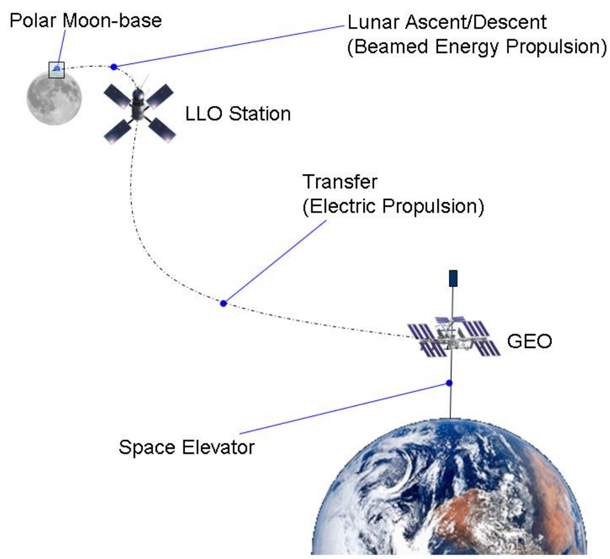

Figure 8. Space transportation architecture 2100

By use of the code Chemical Equilibrium with Applications (CEA)[12] the performance of heated fluids in a typical convergent-divergent nozzle can be calculated. This is relevant for a *Thermal* propulsion system. Figure 9 shows the specific impulse and required absorbed power for such a vehicle propelled by oxygen. A specific impulse of $I_{vac} = 250$ s seems feasible. A study using the LENOIR-cycle to model performance of the *Microwave Rocket* shows that similar performance can be achieved with such a system.[8]

For a required $\Delta v = 2069\ \frac{\mathrm{m}}{\mathrm{s}}$ for lunar ascent and $\Delta v = 1900\ \frac{\mathrm{m}}{\mathrm{s}}$ for lunar descent to/from 100 km LLO,[4] the basic rocket equation (4) leads to fuel fractions as shown in Table 7.

specific impulse	**one-way (ascent only)**	**return-trip (without refueling)**
$I = 200$ s	$\xi_{fuel} = 65.2$ %	$\xi_{fuel} = 86.8$ %
$I = 225$ s	$\xi_{fuel} = 60.8$ %	$\xi_{fuel} = 83.4$ %
$I = 250$ s	$\xi_{fuel} = 57$ %	$\xi_{fuel} = 80.2$ %
$I = 275$ s	$\xi_{fuel} = 53.6$ %	$\xi_{fuel} = 77$ %
$I = 300$ s	$\xi_{fuel} = 50.5$ %	$\xi_{fuel} = 74$ %

Table 7. Required fuel fractions for lunar ascent/descent

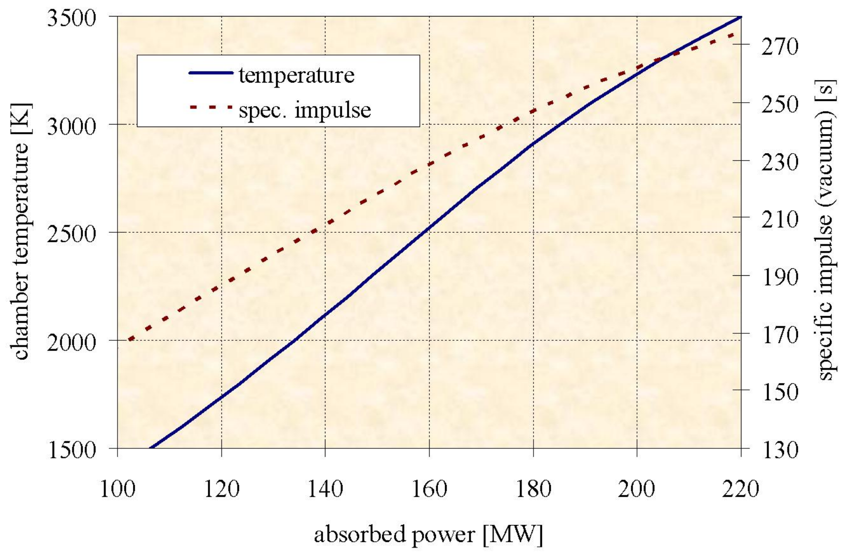

Figure 9. Specific impulse of heated oxygen ($p_c = 10$ MPa and $\epsilon = 100$)

$$\Delta v = c_{eff} \ \ln \frac{m_0}{m_0 - m_{fuel}} \tag{4}$$

Local applications on Mars, using oxygen generated from the atmosphere as propellant, seem equally interesting, in case of an established ISRU infrastructure on Mars.

VI. Conclusion and Outlook

Three different scenarios have been defined. The baseline scenario (*Business as Usual*) is based on current launch demand. The scenario (*Microcosm*) is a minimalistic scenario, assuming a shift towards very small payload masses and an overall decrease of the launch demand. The optimistic scenario (*Space Opera*) foresees large lunar installations and the required associated transportation demand. The performance of a BEP vehicle for launch from Earth to LEO has been analyzed. In all discussed scenarios utilization of a BEP vehicle can lead to a significant reduction in launch cost. The suitability of a BEP engine using ISRU oxygen for local transportation in the lunar environment has been shown.

Future work will concentrate on BEP vehicle definition. A preliminary feasibility study on a ceramic heat exchanger for a BEP thermal propulsion system is currently being initiated in the Institute of Structures and Design of the German Aerospace Center (DLR).

References

[1] Parkin, K. L. G., *The Microwave Thermal Thruster and its Application to the Launch Problem*, Ph.D. thesis, California Institute of Technology, Pasadena, CA, May 2006.

[2] N.N., “2007 Commercial Space Transportation Forecasts,” Tech. rep., Federal Aviation Administration, May 2007, HQ-013107.INDD.

[3] Izydorczyk, J., “Bounds on Minimal Energy Spent on Computations,” *7th IEEE International Symposium on Signal Processing and Information Technology*, Cairo, Dec. 2007.

[4] Eckart, P., *The Lunar Base Handbook*, McGraw-Hill, 2nd ed., 2006.

[5] Hufenbach, B., editor, *Moon: the 8th continent (Human Spaceflight 2025)*, ESA-ESTEC, Noordwijk, Dec. 2003.

[6] Kare, J. T. and Parkin, K. L. G., “A Comparison of Laser and Microwave Approaches to CW Beamed Energy Launch,” *Fourth International Symposium on Beamed Energy Propulsion*, edited by K. Komurasaki, AIP, Nov. 2005.

[7]Oda, Y., Nakagawa, T., Komurasaki, K., Takahashi, K., Kasugai, A., Sakamoto, K., and Imai, T., "An Observation of Plasma Inside of Microwave Boosted Thruster," *2nd International Symposium on Beamed Energy Propulsion*, Sendai, Oct. 2003.

[8]Herbertz, A., Oda, Y., and Komurasaki, K., "Competitiveness-Analysis of a Future Space Transportation System: The Microwave Rocket," *5th International Symposium on Beamed Energy Propulsion*, Nov. 2007.

[9]Katsurayama, H., Komurasaki, K., and Arakawa, Y., "Feasibility for the Orbital Launch by Pulse Laser Propulsion," *Journal of Space Technology and Science*, Vol. 20, No. 2, 2004, pp. 32–42.

[10]Koelle, D. E., *Handbook of Cost Engineering for Space Transportation System*, TransCostSystems, Ottobrunn, Feb. 2003, Transcost 7.1.

[11]Freitas, R. and Gilbreath, W., editors, *Advanced Automation for Space Missions*, NASA/ASEE Summer Study, Santa Clara, USA, Aug. 1980.

[12]Gordon, S. and McBride, B. J., *Computer Program for Calculation of Complex Chemical Equilibrium Compositions and Applications Vol.II: Users Manual and Program Description*, NASA Lewis Research Center, June 1996, NASA RP-1311.