A VEGA Dedicated Electric Propulsion Transfer Module To The Moon

IEPC-2007-306

Presented at the 30th International Electric Propulsion Conference, Florence, Italy
September 17-20, 2007

C. Casaregola[†], K. Geurts[†], P.Pergola[†], L. Biagioni[*], M. Andrenucci[‡]
Alta S.p.A., Pisa, 56121, Italy

Scientific missions to the Moon represent essential steps for human space exploration as they can provide unique insight into a range of fundamental questions about the history of our solar system. Moreover, the recent successful conclusion of the SMART-1 Moon mission based on an Electric Propulsion system has generated considerable interest in the possibility of more efficient access to the Moon by advanced spacecraft with a larger payload ratio. In this context a high performance mission architecture is being studied by the Italian Space Agency (ASI), for which Alta was assigned responsibility for the coordination of a team of Italian industrial and academic institutions and for the definition of an Electric Propulsion Orbital Transfer Module. The paper presents results from phase 1 of this study and describes the most relevant mission aspects so far addressed. An optimized low-thrust orbit transfer strategy is shown to be capable of delivering a significant payload in Moon orbit with a VEGA launch, with mission flexibility and agility characteristics which could not be achieved by a conventional fully chemical transfer. Several options are explored, identifying technologies that are presently available and that could already allow for this type of mission to be successfully implemented.

Acronyms

ASI	= Italian Space Agency
BOL	= Begin Of Life
BPRU	= Bang-Bang Pressure Regulation Unit
EFU	= Electric Filter Unit
EOL	= End Of Life
EPDP	= Electric Propulsion Diagnostic Package
EPS	= Electric Propulsion Subsystem
GTO	= Geostationary Transfer Orbit
LEO	= Low Earth Orbit
LLO	= Low Lunar Orbit
LO	= Lunar Orbiter
LS	= Lunar Surface
MDLM	= Moon Descending and Landing Module
MRRH	= Maximize the Rate of change of the Relative angular momentum vector (H)
OBDH	= On Board Data Handling

[†] Ph. D. Student, Department of Aerospace Engineering, University of Pisa; Research Engineer, Alta S.p.A.; Member AIAA
[*] Associate Fellow AIAA
[‡] CEO, Alta S.p.A.; Professor, Department of Aerospace Engineering, University of Pisa; Senior Member AIAA.

OTM = Orbital Transfer module
P/L = Payload
PPU = Power Processing Unit
PRE = Pressure Regulation Unit
RAAN = Right Anomaly Ascending Node
RW = Reaction Wheel
TRL = Technology Readiness Level
TT = Tangential Thrust
XFC = Xenon Flow Controller

I. Introduction

The Moon is potentially a unique museum of the history of the solar system. Scientific missions to the Moon undoubtedly can provide unique insights into a range of fundamental questions. Those of most obvious interest and importance are related to the origins and history of the solar system, whether life is unique to Earth and how life on Earth began. According to this growing interest in Moon exploration, many Earth-Moon transfer strategies with associated mission scenarios have been proposed and analysed in the past few decades.[1,2,3,4]
While most of these studies were based on conventional propulsion systems, alternative solutions based on low thrust, high efficiency propulsion technology (i.e. Electric Propulsion) have gained increasing attention in recent years, also in relation with the successful completion of the SMART-1 mission.[5,6,7,8]
Other studies have looked at the possibility to deliver materials to a Lunar orbit for Moon observation or for the development and construction of a lunar base.[9] Results of such studies generally agree on the benefits in terms of LEO launch mass reduction and associated launch-cost savings achievable through the use of Electric Propulsion whenever mission transfer times do not represent a priority.

An Italian Vision for Moon Exploration has been outlined by several studies committed by the Italian Space Agency (ASI). In this context, Alta was assigned responsibility for the coordination of a team of Italian industrial and academic institutions and for the definition of an Electric Propulsion Orbital Transfer Module (OTM) capable of delivery a significant payload in Low Lunar Orbit (LLO) with a VEGA launch, with mission flexibility and agility characteristics which could not be achieved by a conventional fully chemical transfer.

Mission analysis aspects relevant to the preliminary design definition of the OTM are presented, with the purpose of defining the overall scenario within which the study was performed. A central aspect and the starting point of this feasibility study is the preliminary design of the low-thrust Earth-Moon Transfer trajectory from a Low Earth Orbit (LEO) to a Low Lunar Orbit (LLO). Details about the Electric Propulsion Subsystem (EPS) are also given.

This paper is organized as follows. Section II gives a brief overview of the study objectives and of the mission scenarios analyzed. A preliminary sensitivity analysis on preliminary LEO-LLO low-thrust transfers in a specified range of power levels and specific impulses is discussed in Section III. In Section IV the trajectory design is summarized, results and discussion outlined. In Section V OTM Electric Propulsion Subsystem is described and, in the end, conclusions and plans for future research are outlined in Section VI.

II. Study Objectives and Mission Scenarios

Primary objective of this feasibility study is the definition of a VEGA dedicated Electric Propulsion Orbital Transfer Module (OTM) able to allow an efficient access to the Moon and compatible with the mass, dimensions and volume constraints imposed by the national launcher.

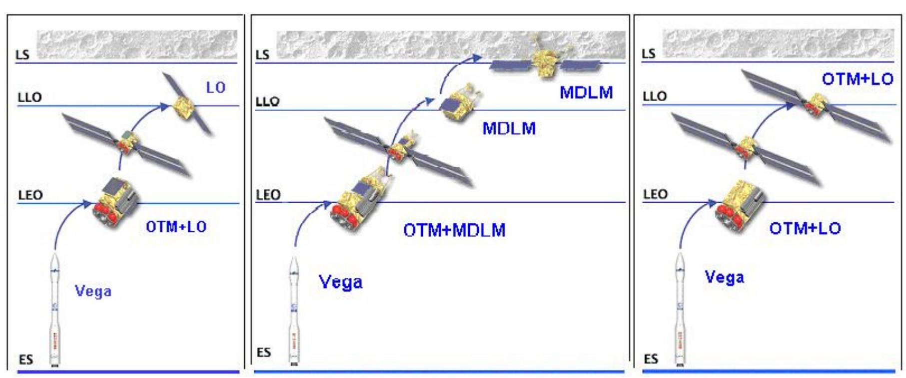

Figure 1. Mission scenarios

Thus, according to the scientific objectives of the ASI program "Italian Vision for Moon Exploration", the design of a low-thrust Earth-Moon transfer trajectory from LEO to LLO has been carried out to allow the Electric Propulsion OTM to place a Lunar Orbiter (LO) and a Moon Descending and Landing Module (MDLM) in the required orbit around the Moon.

The Electric Propulsion orbit raising and Moon capture manoeuvre will call into play the following principal and mutually interrelated parameters:

1) Mass ratio
2) Thrust and Power levels
3) Transfer time

The achievable mass ratio will mainly depend on the propulsion technology selected (and relevant specific impulse) and on the details of the thrusting and steering strategy. The transfer time will mainly depend on the available thrust level, which will be essentially limited by power constraints. The purpose of the analysis and optimization activities included in this study was to assess possible transfer strategies and their implications in order to obtain the best trade off among the above factors.

Accordingly, several scenarios were investigated, and three different mission profiles were finally selected for further consideration (Fig. 1):

1) Launch with VEGA, OTM+LO starts the EP thrusting phase, OTM releases LO in required LLO.
2) Launch with VEGA, OTM+MDLM starts the EP thrusting phase, OTM releases MDLM in required LLO.
3) Launch with VEGA, OTM+LO starts the EP thrusting phase, OTM+LO operates in required LLO.

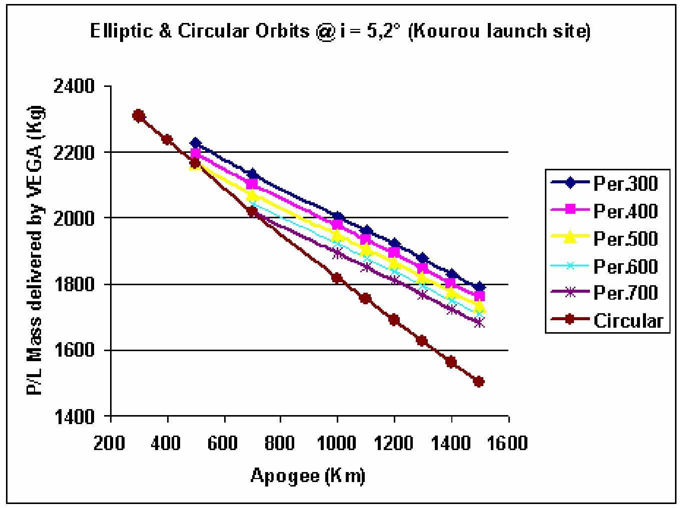

Figure 2. P/L mass performance of VEGA launcher

A preliminary trade-off analysis was performed to determine the best OTM initial orbit after VEGA release. As the OTM's primary objective is to reach the required LLO orbit with a relatively large P/L mass in spite of the mass constraints imposed by the VEGA launch vehicle (Fig. 2), an initial circular orbit at 400 km with an inclination of 5.2° (Kourou launch site latitude) was chosen. This orbit maximizes the initial OTM mass even if a longer transfer time is needed, but no time constraints were imposed by ASI.

III. Preliminary Sensitivity Analysis

Preliminary LEO-LLO low-thrust transfers with constant thruster power levels (5 kW, 10 kW and 15 kW) in a range of specific impulses varying from 1500 s to 5000 s were investigated. Results obtained were analyzed in two steps:

1) in the 1st step propellant mass needed to perform the orbit transfer was computed without taking into account the Power Subsystem mass.

2) in the 2nd step results were integrated also taking into account the Power Subsystem mass (Fig. 3).

In this second step, three different solar array technologies were analyzed:

1) Flexible blanket based on CIGS cells.
2) Ultra Light rigid panels with GaAs TJ s.o.a. cells.

	Specific Mass (kg/m^2)	Power density @ launch (W/m^2)	Specific Power @ launch (W/kg)	Stowed Power density @ launch (kW/m^3)	BOL efficiency	EOL Efficiency (1E15 MeV eq.rad.dam.)	TRL
Technology #1	1.6	130	80	45 – 60	12%	10%	3
Technology #2	1.7	300	180	8 – 15	28%	22%	9
Technology #3	1.65	330	200	60 – 100	24% (33%)	20% (26%)	4 (3)

Thruster Power Level (kW)	Batteries (kg)	PCDU (kg)	PSU (kg)	Solar Array (kg)	Power Sub. Total Mass (kg)
5	25	20^{*}	30^{*}	Tec.#1: 88	163
				Tec.#2: 39	114
				Tec.#3: 35	110
10	25	30	50	Tec.#1: 162	267
				Tec.#2: 72	177
				Tec.#3: 65	170
15	25	40^{*}	70^{*}	Tec.#1: 250	385
				Tec.#2: 111	246
				Tec.#3: 100	235

Table 1. Candidate technologies for S/A and preliminary Power Subsystem Mass Budget

3)Flexible blanket based on GaAs ultra thin cells.

Technical characteristics of each technology are shown in Table 1.

Besides, for each power level, battery mass, PCDU and PSU mass were also included in the Power Subsystem mass budget. Some of these data (the ones with * in Table 1) were appropriately scaled to have tentative values. Solar arrays were sized for the three different power levels in order to obtain a BOL installed power of 7, 13 and 20 kW. In Table 1 the values computed for each power level are summarized.

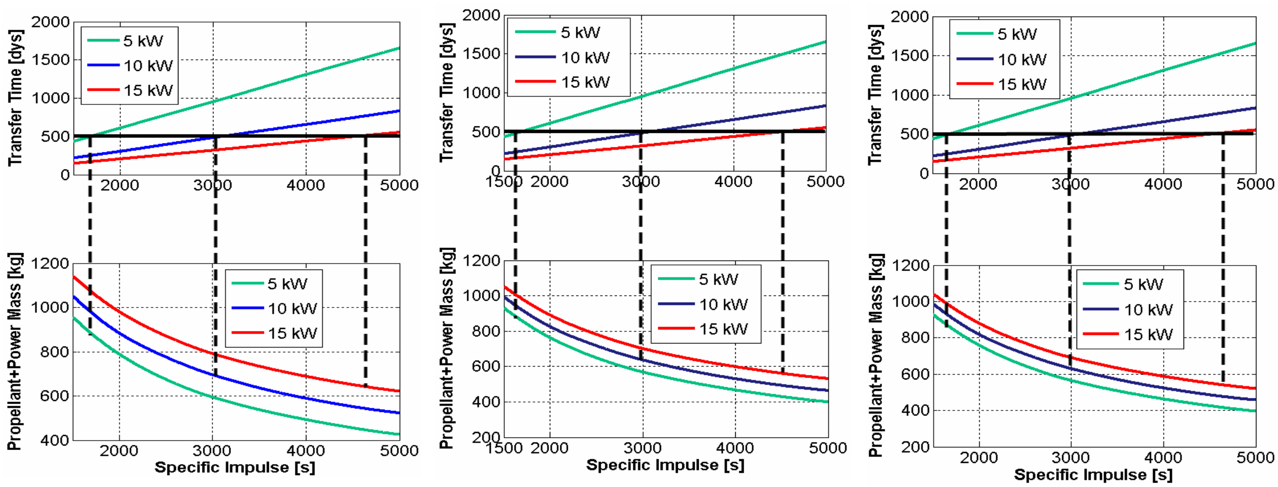

Figure 3. (From left to right) 2nd step preliminary results in Technology #1, Technology#2 and Technology#3

Figure 3 shows the plots obtained in the 2nd step for each selected technology. It can be observed that, for all technologies investigated, for a fixed transfer time and for different power levels, the sum of Propellant mass plus Power Subsystem mass has a minimum value at higher values of the specific impulse (in technology#1 this is not exactly true as the values obtained for the 10 kW and 15 kW curves are almost equal).

However, in order to have a propellant plus power subsystem mass not higher than 800 kg and an acceptable transfer time (approximately 3 years), a specific impulse higher than 2500 s and a power level equal or larger than 10 kW were necessary.

IV. Mission Analysis

A. Trajectory Design

A geocentric inertial Cartesian coordinate system was used as system of reference for the Earth-Moon low-thrust transfer design. A circular LEO starting from 400 km of altitude was assumed as departure orbit and a VEGA deliverable mass of 2240 kg (Fig. 2) considered (assumed as the initial OTM mass). Thrusting was assumed to be switched off during the eclipses and, according to the previous analysis, a constant available power P= 10 kW and an I_{sp}=3000 s were considered for the Electric Propulsion operation. The thrust available was derived by the relation:

$$T = \frac{2 \cdot \eta_T \cdot P}{I_{sp} \cdot g_0} \tag{1}$$

where g_0 is the gravitational constant and η_T the propulsion system thrust efficiency.

During the entire trajectory, Cowell's method was used for the integration of the equations of motion with the inclusion of different perturbative terms to account for the deviation from a perfect two body environment. In particular, perturbations due to Sun, Moon, Earth's harmonics (J2, J3, J4, J5, J6), atmospheric drag and solar radiation pressure were included. Accelerations due to the Electric Propulsion System operation were also included in the perturbative term.

Launch window considerations were also done according to ASI reference launch date (2011). The time frame explored was limited to two years only within this phase of the study. However no significant advantages were found from this analysis in the time frame considered. The best launch date was found to be the 12/13/2012. The initial RAAN was assumed to be the same as the Moon's. In fact with this assumption the orientation of the initial orbital plane and the Moon's orbital plane are the same and only a change in the plane inclination must be performed.

The complete transfer was modelled using three distinct phases:

1) Geocentric spiral-out and plane adjustment (Phase 1).
2) Geocentric Moon phasing and lunar capture (Phase 2).
3) Selenocentric final orbit acquisition (Phase 3).

Phase 1

Phase 1 was used to raise from LEO, escape Van Allen Belts as soon as possible and obtain an inclination equal to the Moon's orbital inclination. It was divided in a first phase in which a Tangential Thrust (TT) is used to increase the geocentric radius. By applying this strategy, the rate of change of semi-major axis (and hence of the orbital energy) is maximized.

The second part of Phase 1 is then started to Maximize the Rate of change of the Relative angular momentum vector $\vec{H}$ (MRRH) between the spacecraft and the Moon orbit. This MRRH thrusting strategy[10] has the unique property to be defined in such a way to include target conditions, in our case the angular momentum vector of the Moon. This approach is based on the maximization of the functional:

$$F = \left[\left(\vec{H} - \vec{H}_f\right) \times \vec{R}\right] \cdot \vec{a}_{Thrust} = \left[\varDelta\vec{H} \times \vec{R}\right] \cdot \vec{a}_{Thrust} \tag{2}$$

where $\varDelta\vec{H}$ is the relative angular momentum vector.

The MRRH strategy, therefore, maximizes the rate of change of the angle between the two orbital planes (called "Wedge Angle"). Moreover, this strategy offers the possibility to employ different weights (w_i) for each thrust component. In fact, the thrust can be written as[11]:

$$\vec{T} = \left|\vec{T}\right| \cdot \frac{(w_C \cdot C_C \cdot \vec{u}_C + w_P \cdot C_P \cdot \vec{u}_P)}{\sqrt{(w_C \cdot C_C)^2 + (w_P \cdot C_P)^2}} \tag{5}$$

Thus the thrusting strategy in Phase 1 provides a tangential thrust until the geocentric radius is equal to R_{switch}, then a MRRH strategy is applied with the in plane weighting factor (w_C) and the out of plane one (w_P) with a ratio of 1:5. A sensitivity analysis of the solution, especially in terms of fuel mass consumption, was performed using different R_{switch} and I_{sp}.

In the end, a R_{switch}= 40000 km was chosen to assure the complete Van Allen belts transit. Nevertheless, no significant dependence of the results from this value was observed in the range considered. Phase 1 trajectory is shown in Fig. 4.

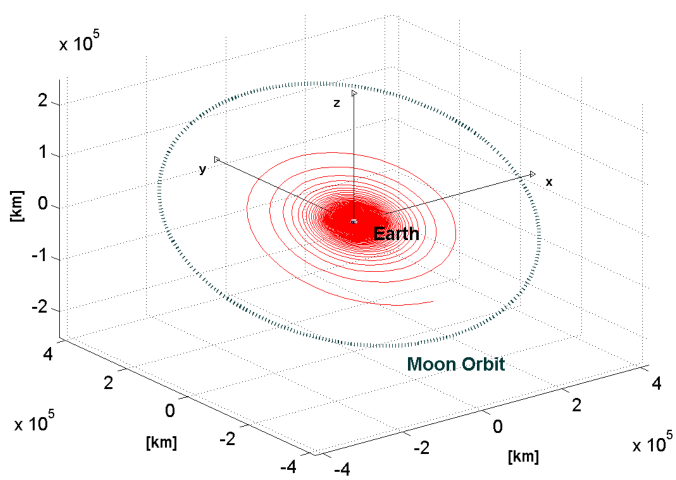

Figure 4. Phase 1 of OTM Earth-Moon low-thrust

Phase 2

Phase 2 was implemented in order to achieve the lunar encounter with the proper characteristics. The purpose of this trajectory segment was to obtain a lunar capture with the minimum fuel consumption. In order to achieve this goal an optimized phase was implemented.

In this phase the spacecraft's orbit plane is the same as the Moon's one, hence the capture problem is reduced to a coplanar problem. Accordingly, only the in-plane thrust angle and the thrust modulus are considered as control parameters. The entire trajectory was evaluated in a geocentric system of reference with the same axis definition and perturbation employed in Phase 1.

The optimization technique follows a direct approach with the use of a Symplex algorithm to solve the non linear unconstrained optimization problem. This method is advantageous in that computational time is shorter than for other techniques, it does not involve the use of numerical or analytic gradients and can also handle discontinuities, but it strongly depends on the initial guess.

The optimized phase 2 requires about 19 days of transfer time with a fuel mass consumption of about 9 kg. The capture orbit so obtained is “stable” for about 50 days and, within this period, the final manoeuvre of phase 3 progressively reaches the required operative LLO. The geocentric and selenocentric representation of the Phase 2 is shown in Fig. 5.

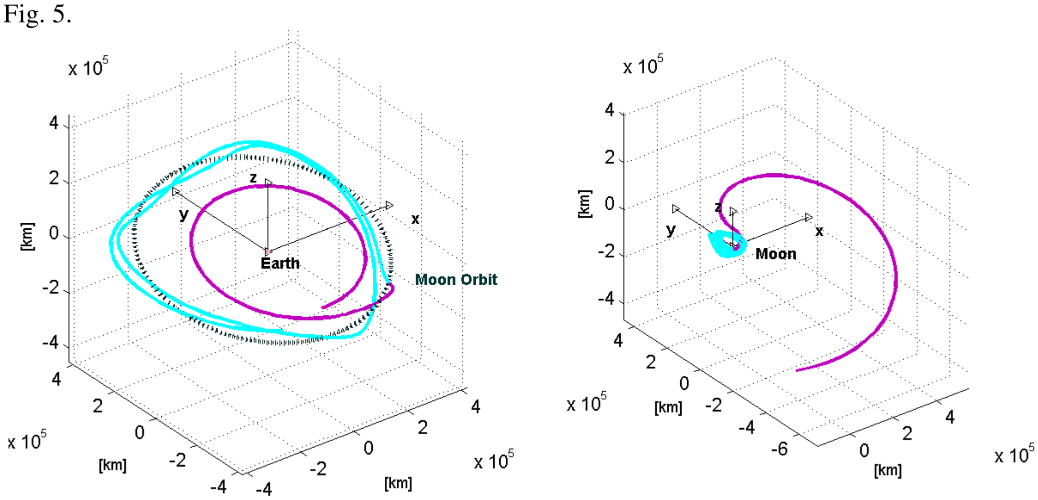

Figure 5. Phase 2 of OTM Earth-Moon low-thrust transfer in the geocentric (on the left) and selenocentric system of reference.

Phase 3

After the lunar capture, the final polar, circular orbit at 100 km of altitude is obtained. A selenocentric inertial frame of reference is considered centered in the Moon’s center with the same axis direction of Phase 1. Perturbation due to the Earth, the Sun, the solar radiation pressure and the Moon J2 are duly accounted for. As the required final capture orbit was to be polar, the thrusting strategy had to include:

1) Orbital plane change
2) Re-circularization
3) Orbit altitude decrease

Again, an hybrid MRRH/anti-TT strategy was employed with a circular polar LLO at 100 km of altitude as target condition, but limiting the thrust phase only to the neighborhood of the perigee, with free final argument, in order to obtain also a reduction of eccentricity. The orbit obtained is shown in Fig. 6.

In this case both the circumferential component of the MRRH strategy and the TT were directed along the negative velocity unit vector. A suitable combination of the in-plane and out-of-plane weighing factors for MRRH thrusting strategy led to complete as soon as possible the plane change, then, when the orbit was almost circular, the strategy was switched to an anti-TT strategy to complete the altitude reduction.

As shown in Table 2, different values of thrust angle were explored in this phase. Upon considering fuel mass consumption, a final value of thrust angle between ± 100 deg around the argument of perigee was selected.

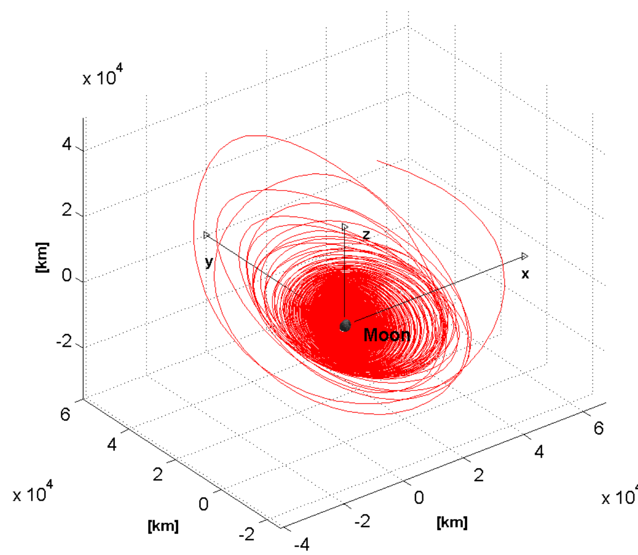

Figure 6. Phase 3 of OTM Earth-Moon low-thrust transfer

Thrust Angle [deg]	Considerations
< 50°	No target orbit obtainable
60°	Δm = 187.6 kg
80°	Δm = 175.5 kg
100°	Δm = 171.7 kg
110°	Δm = 196.1 kg
> 110°	Inclination tend to diverge

Table 2. Propellant budgets for different thrust angles in Phase 3

B. Results and Discussion

The main results for the reference trajectory design with a VEGA launch are summarized in Table 3. As it can be observed, a total transfer time of about 3 years was computed with a propellant mass fraction of about 0.28 and a total ΔV of about 9.9 km/s.

In addition to the complete reference trajectory design previously described, further analyses were carried out taking into account both the baseline VEGA launcher and an alternative scenario with the Soyuz launcher to make a comparison. In both cases, payload mass delivered by Electric OTM in a circular

Table 3. Results of Preliminary LEO-LLO low-thrust trajectory design

Simulation Input Data	Departure Date	13/12/2012
	Initial Altitude (km)	400
	Initial inclination (deg)	5.2
	Initial eccentricity	0
	Initial RAAN (deg)	345
	Initial Mass (kg)	2240
	Specific impulse (constant) (s)	3000
	Thrust efficiency	0.55
	EP Power (constant) (kW)	10
Phase 1	Total Transfer Time (days)	478
	Propellant Mass consumption (kg)	455
	Total ΔV (km/s)	6.682
Phase 2	Total Transfer Time (days)	17
	Propellant Mass consumption (kg)	10
	Total ΔV (km/s)	0.165
Phase 3	Total Transfer Time (days)	600
	Propellant Mass consumption (kg)	175
	Total ΔV (km/s)	3.054
Overall LEO-LLO transfer	Total Transfer Time (days)	1095
	Propellant Mass consumption (kg)	640
	Total ΔV (km/s)	9.902

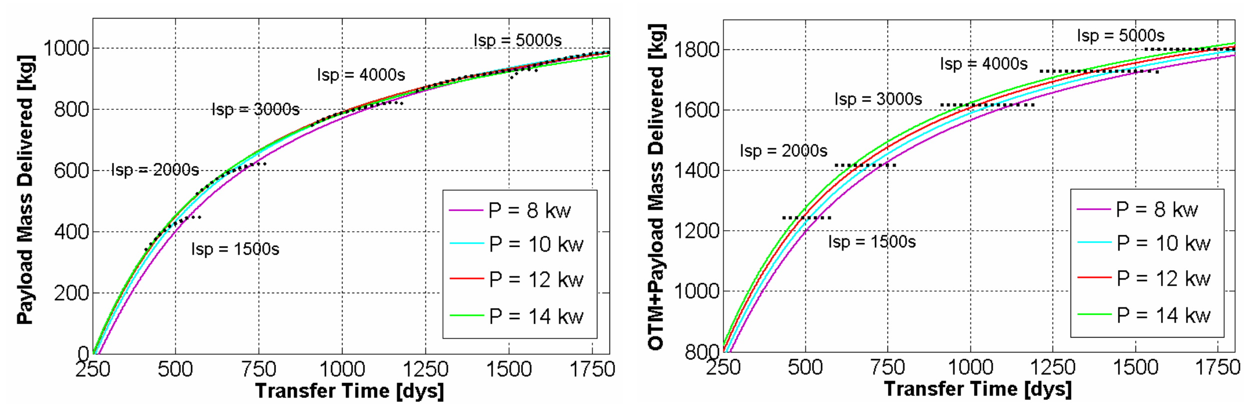

Figure 7. Parametric analysis of P/L and OTM+P/L mass delivered in a polar 100 x 100 km LLO with VEGA Launch

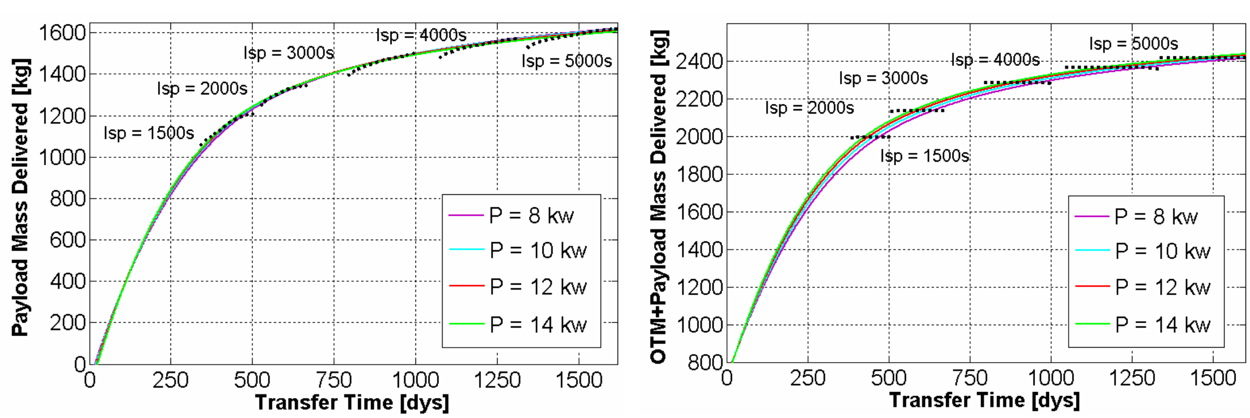

Figure 8. Parametric analysis of P/L and OTM+P/L mass delivered in a polar 100 x 100 km LLO with Soyuz Launch in GTO

polar LLO at 100 km of altitude versus transfer time and for different power levels were analyzed.

As regards the Soyuz launch, assuming use of the Kourou launch site, and of the Fregat upper stage, a P/L mass of 2760 kg can be delivered in GTO[12].

Results obtained in both cases are shown in Fig. 7 and Fig. 8. The P/L mass delivered in LLO without taking into account the OTM mass is plotted on the left of Fig. 7 and Fig. 8 while on the right the corresponding results including also the OTM mass are shown.

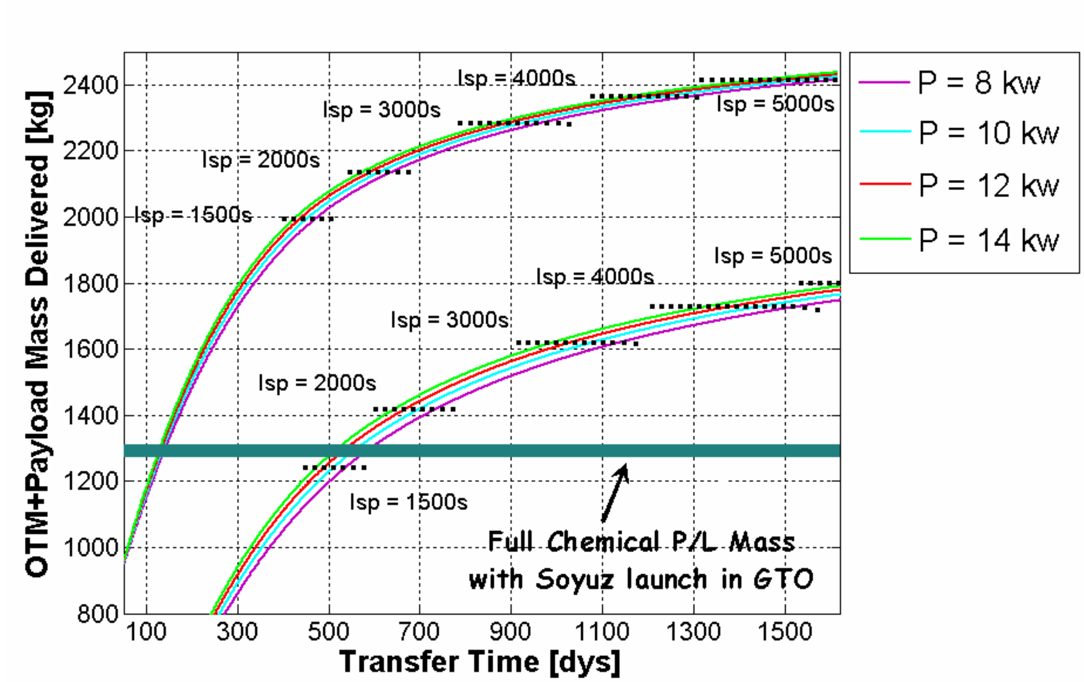

Figure 9. MTO+P/L Mass in LLO vs. transfer time for a full Chemical transfer and an Electric OTM launched by VEGA and Soyuz

A diagram summarizing the results obtained for the MTO+P/L mass is shown in Fig. 9. Here the results obtained by an Electric OTM launched by VEGA in LEO and by an Electric OTM launched by Soyuz in GTO are shown together and compared to a P/L mass delivered by a Soyuz launch in GTO with a full chemical orbit transfer in LLO. This last mass of about 1300 kg was computed assuming the same initial mass in GTO (2760 kg) and a Hohmann transfer to LLO with a $\Delta V = 2300$ m/s and an $I_{sp} = 320$ s.

Two "crossing points" can be identified in this diagram. The first one, between the full chemical P/L mass delivered to LLO and MTO+P/L mass delivered by an Electric OTM launched with Soyuz, occurs at about 130 days of transfer time. The second one, between the full chemical P/L mass delivered in LLO and MTO+P/L mass delivered by an Electric OTM launched with VEGA, occurs at about 550 days.

It turns out that an Electric OTM is a more efficient solution with respect to a full chemical transfer to LLO if we consider transfer times longer than 130 days with a Soyuz launch or longer than 550 days with a VEGA launch. Moreover, the longer the transfer time, the larger the ΔP/L mass.

V. OTM Electric Propulsion Subsystem Overview

Electric Propulsion Subsystem (EPS) preliminary design is based on the SMART-1's flight experience[13] even if OTM's power level is much higher. A preliminary EPS functional diagram is shown in Fig. 10.

The OTM power is supplied by the two solar panels enabling the plasma thruster to operate at 10 kW max. The main functions of the EPS are:

1) to provide the electrical power to the subsystem.
2) to store the propellant and deliver its rated flow to the thruster.
3) to interface with the S/C OBDH and Communication subsystems.

The first function is accomplished by means of an electrical power system composed of a main power transformer called Power Processing Unit (PPU) which transforms the input S/C bus (100 Volts) into the voltage required by the thruster (~800 Volts). An Electrical Filter Unit (FU) is between the PPU and the thruster anode/cathodes to reduce the electrical thruster oscillations and to protect the PPU electronics.

A Xenon system fulfills the second function. It is composed of a main Tank which stores the gas under high pressure, a pressure regulator called Bang-Bang Pressure Regulation Unit (BPRU) which has the task to deliver the Xenon under constant pressure to the adjustable flow regulator called Xenon Flow Controller (XFC). Then, the XFC delivers the rated flow to the thruster anode and cathode.

The digital interface between the EPS and the S/C OBDH and Communication subsystems is composed of a Pressure REgulation (PRE) Unit whose micro processor has the task to control the constant pressure level of the BPRU by means of a simple and robust algorithm loop. Moreover, in the PRE Unit, telemetry and commands dealing with the PPU and the thruster are first converted into simpler messages and then sent to the PPU's digital interface for further processing into PPU's micro processor where a simple and robust algorithm loop is integrated to generate the analogic control signal to the XFC.

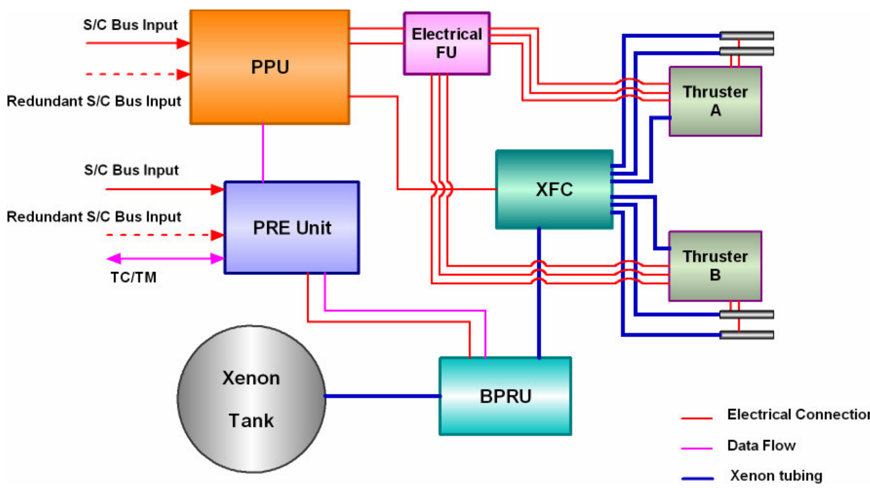

Figure 10. EPS functional diagram

For redundancy reasons, EPS architecture foresees a fully redunded Xenon feeding system able to feed one of the two thrusters according to the switch unit located in the PPU.

The operation of the EPS throughout a long duration mission like the transfer from LEO to LLO needs to be carefully verified and constantly monitored for avoiding any potential risk that would impair or even jeopardize the space mission successful accomplishment. According to that, an Electric Propulsion Diagnostic Package (EPDP) including diagnostic systems based on probes/sensors suitable to characterize and verify the thruster health monitoring was also added. A preliminary mass budget is shown in Table 4.

Table 4. EPS mass budget

Item	Unit Weight (kg)	Quantity	Weight (kg)
Anode Unit + TSD	19	2	38
Cathode	0,5	4	2
XFC	6	1	6
BPRU	3	1	3
PRE Unit	1.5	1	1.5
PPU	50	1	50
EFU	1	1	1
Tank	50	1	50
EPDP	3	1	3
Piping, Ducting & Harness	10	-	10
Margin (20%)			33
EP Subsystem Total Weight			**197.5**

VI. Conclusions

A high performance Electric Propulsion OTM launched to LEO with VEGA and then transferred from LEO to LLO by means of an optimized low-thrust strategy has been shown capable of delivering a significant payload in

LLO with greater mission flexibility with respect to a conventional fully chemical transfer. From the comparison of the low-thrust transfer results obtained in a limited range of power levels and I_{sp}, a VEGA dedicated Electric Propulsion Orbital Transfer Module proves more efficient for access to the Moon if a short transfer time is not required. Similar considerations can be done considering a Soyuz dedicated Electric Propulsion OTM. Nevertheless, for what concerns mission analysis, future work will be needed to address the possibility of a low-thrust transfer with variable I_{sp} and the capability of LLO capture strategies including Moon's resonance effects in order to evaluate their potential benefits with respect to the basic strategy.

Acknowledgements

This study is funded by the Italian Space Agency. The authors would like to thank Mr. D. Barbagallo and Mr. L. Bisi of the ASI whose comments and useful contribution were fundamental in the good outcome of this work.

References

[1]Belbruno, E. A., "Lunar Capture Orbits, a Method of Constructing Earth Moon Trajectories and the Lunar Gas Mission, " 19th AIAA/DGLR/JSASS International Electric Propulsion Conference, 1987, Colorado Springs, Colorado.

[2]Belbruno, E. A., Carrico, J. P., "Calculation of weak stability boundary ballistic lunar transfer trajectories," AIAA/AAS Astrodynamic Specialist Conference 14-17 August 2000, Denver, Colorado.

[3]Koon, W. S., Lo, M. W., Marsden, J. E., Ross S. D., "Shoot the Moon," AAS 00-166.

[4]Koon, W. S., Lo, M. W., Marsden, J. E., Ross S. D., "Low Energy Transfer to the Moon," Celestial Mechanics and Dynamical Astronomy 81: 63–73, 2001.

[5]Jahn, R., Cano, J.-L., "Optimum Low-Thrust Transfer Between Two Orbits," ESOC Mission Analysis Section Working Paper N. 414, March 1999.

[6]Schoenmaekers, J., Pulido, J., Cano J.-L., "SMART-1 Moon Mission: Trajectory design using the Moon Gravity," S1-ESC-RP-5501 issue 1, ESOC, May 1999.

[7]Cano J.-L., Schoenmaekers, J., Jehn, R., Hechler, M., "SMART-1 Mission Analysis: Collection of Notes on the Moon Mission," ESOC, Mission Analysis Section Working Paper N. 417, August 1999.

[8]Cano J.-L., Hechler, M., Horas, D., Khan, M., Pulido, J., Schoenmaekers, J., Jehn, R., "SMART-1: Correlated report on Mission Analysis," S1-ESC-RP-5506 issue 1.2, ESOC July 2001.

[9]Palaszewski, B., "Electric Propulsion for Lunar exploration and Lunar Base development," URL: http://www.nss.org/settlement/moon/library/LB2-105-ElectricPropulsion.pdf

[10]Leipold, M., Pagel, G., DeNiem, D., "Electric Propulsion Transfer to the Moon for Smart-1", Technical Report DLR-TB 426-98/01.

[11] Casaregola, C., Geurts, K., Pergola, P., Biagioni, L., Andrenucci, M., "Mission Analysis and Architecture Definition for a Small Electric Propulsion Transfer Module to the Moon", 43rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, July 8-11, 2007, Cincinnati, OH

[12]"Soyuz from the Guiana Space centre user's manual," Arianespace, June 06, issue 1, rev 0.

[13] Koppel, C.R., Estublier, D., "SMART-1 Primary Electric Propulsion Subsystem the Flight Model", 28th IEPC, March 17-21, 2003, Toulouse, France.