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DESIGN OF LOW-THRUST TRAJECTORIES FOR THE EXPLORATION OF THE OUTER SOLAR SYSTEM

Massimiliano Vasile, Andres Galvez, Leopold Summerer, Franco Ongaro

ESA – ESTEC, Advanced Concepts Team
Keplerlaan 1, 2200 AZ, Nordwijk ZH, The Netherlands
Email: Massimiliano.Vasile@esa.int

ABSTRACT

The paper analyses the use of a Nuclear Electric Propulsion systems for missions to the outer part of the solar system and points out the possibility to deploy a probe while performing a gravity assist manoeuvre in the vicinity of Jupiter or to insert the spacecraft in a highly elliptical orbit about Pluto.

The design of the trajectories has been performed with a direct transcription method by finite elements in time. As the problem presents quite a number of possible solutions dependent on launch window, transfer time and the use of gravity assist manoeuvres, a global optimization strategy has been used to procure sets of promising initial guesses. These initial guesses have been subsequently optimized using direct transcription and NLP.

INTRODUCTION

Currently most of our knowledge about Pluto and its moon Charon comes from indirect clues. No spacecraft has ever visited either of them, and from the Earth (or its proximity) its angular size is close about resolving limit of the most capable ground and space-based observatories. However, there is unanimity on the scientific interest of the Pluto-Charon system. Therefore a mission to the Pluto-Charon system and eventually to a Kuiper Belt Object (KBO) will significantly increase our knowledge of the formation and evolution of the Solar System as well as the origin of volatiles and organic molecules that enabled the appearance of life on our own planet. The mission therefore has a strong exobiological interest, which could be increased exponentially by adding new elements like a Europa or Titan microprobe deployment on route to the final destination, taking advantage of the opportunities provided by the gravity assist at the giant planets.

In this paper some possible mission scenarii for a mission to Pluto and the Kuiper belt are proposed: these include the utilization of advanced propulsion systems (nuclear electric propulsion) and power technologies and the possibility to deploy a probe while a gravity assist maneuver in the vicinity of Jupiter is performed. If the Jupiter option is selected the possibility of a swing-by of one of the moons is investigated. In particular a swing-by of Ganymede can be performed to brake the probe while the main spacecraft continues on its way to Pluto. In addition the possibility of advanced missions using chemical propulsion ad gravity or aero-gravity assist maneuvers have been studied.

Analyses available in the literature[1] propose to carry out a very quick flyby of Pluto and Charon with a large relative velocity, thus enabling a limited science return. Another option is therefore to study the possibility to alter the mission analysis concept in such a way that the flyby velocity can be reduced.

The design of the NEP trajectories has been performed with a direct transcription method by finite elements in time[2]. However the problem presents quite e number of possible solutions dependent on launch window, transfer time and combination of planetary encounters, therefore in order to find favourable launch windows and the optimal sequence of swing-bys a global optimization strategy has been used to procure sets of promising initial guesses. Then, these initial guesses have been optimized using direct transcription and NLP[3].

TECHNOLOGICAL AND SCIENTIFIC OBJECTIVES

A mission to the Pluto-Charon system would significantly increase our knowledge of the formation and evolution of the Solar System. Of particular interest in the Pluto-Charon system is the atmospheric transfer of methane between these bodies and their compositional difference.

Some of the driving scientific objectives of such a mission would include:

- Surface chemical composition
- Surface morphology
- Atmospheric chemistry
- Gravimetry

A strawman payload to achieve the scientific objectives within the allocated mass limits include:

- Imaging X-ray Spectrometer
- Wide / Narrow Field Imager
- IR-Spectrometer
- Radio Science Experiment

The available payload mass obviously depends on the mission scenario. However it is reasonable to assume that even a limited payload mass value (e.g. 20 kg), would be sufficient to meet a significant part of the scientific goals. This should be achievable even with current technology and considering the heritage of other planetary exploration missions like SMART-1[4]. The scientific return from a Pluto mission would increase tremendously if either the fly-by would occur at a small relative velocity or if the spacecraft could go into orbit around the Pluto-Charon system. This would not only increase the coverage but also the accuracy of the scientific investigations because of the low signal to noise ratio for certain instruments partly due to the large distance from the sun.

From a technological point of view, inserting a probe in orbit around such a distant planet at a reasonable propellant expenditure, poses considerable issues both in terms of propulsion and power. Furthermore the long travel results in a long waiting time before any result can be obtained. Since access to distant targets as Pluto can be effectively achieved through a swing-by of Jupiter, the delivery of a probe in the jovian system will significantly improve the scientific return of the mission offering some intermediate results while waiting for the analysis of the Pluto-Charon system.

MISSION ANALYSIS

Two missions have been studied: a double probe to Jupiter and Pluto, a small probe to Europa. In both cases the spacecrafts are equipped with nuclear electric propulsion (NEP) engines and will perform a number of gravity assist manoeuvres to reach the final destination.

Initial Guess Generation

The aim is to find an optimal sequence of transfers from the Earth to Pluto or from Earth to Jupiter passing by a predefined number of intermediate stops (actually swingbys). Even though the propulsion system is electric and not chemical, the trajectory, which minimizes the overall cost in terms of Δv, is regarded as optimal for both means of propulsion since a further optimization with a better model of electric propulsion will be performed using DITAN[3] (a software tool for the design of gravity assist low-thrust trajectories, developed by Politecnico di Milano under ESA contract).

Each arc connecting two subsequent bodies has a deep space Δv manoeuvre at an unknown point in time and space, each swingby is modelled collapsing the sphere of influence in one single point in space with radius r_p linking the transfer arcs r_i before and r_o after the swing by, thus the following relation must hold:

$$\mathbf{r}_i = \mathbf{r}_o = \mathbf{r}_p \tag{1}$$

Since the swingby is un-powered the following relationships between the incoming and the outgoing velocities must hold:

$$\tilde{v}_i = \tilde{v}_o \tag{2}$$

Furthermore, the outgoing relative velocity vector is rotated, due to gravity, by an angle π-2β with respect to the incoming velocity vector and therefore the following relation must hold:

$$\tilde{v}_o^T \tilde{v}_i = -\cos(2\beta)\,\tilde{v}_i^2 \tag{3}$$

where, if μ is the gravity constant of the planet, the complementary angle of rotation of the velocity is computed as:

$$\beta = \operatorname{acos}\left(\frac{\mu}{\tilde{v}_i^2 \tilde{r}_p + \mu}\right) \tag{4}$$

All quantities with a tilde are relative to the swing-by planet and $\tilde{r}_p$ is the periapsis radius of the swing-by hyperbola. Constraints given by equation (1) can be explicitly solved while constraint on the velocity requires the rotation of the velocity vector $\tilde{\mathbf{v}}_i$ of an angle equal to δ=π-2β in the orbit plane of the hyperbola, which is unknown. Therefore we introduce another parameter ω, which represent the rotation angle of a plane around the vector $\tilde{\mathbf{v}}_i$

$$\mathbf{n}_\omega = Q(\tilde{\mathbf{v}}_i)\mathbf{n}_i \tag{5}$$

$$\tilde{\mathbf{v}}_o = Q(\mathbf{n}_\omega)\tilde{\mathbf{v}}_i \tag{6}$$

where Q($\tilde{\mathbf{v}}_i$) and Q($\mathbf{n}_\omega$) are the two rotation matrices defined by the quaternions:

$$\mathbf{q} = \left[\mathbf{v}_i sin\frac{\omega}{2}, \cos\frac{\omega}{2}\right]^T \tag{7}$$

and

$$\mathbf{q} = \left[\mathbf{n}_\omega sin\frac{\delta}{2}, \cos\frac{\delta}{2}\right]^T \tag{8}$$

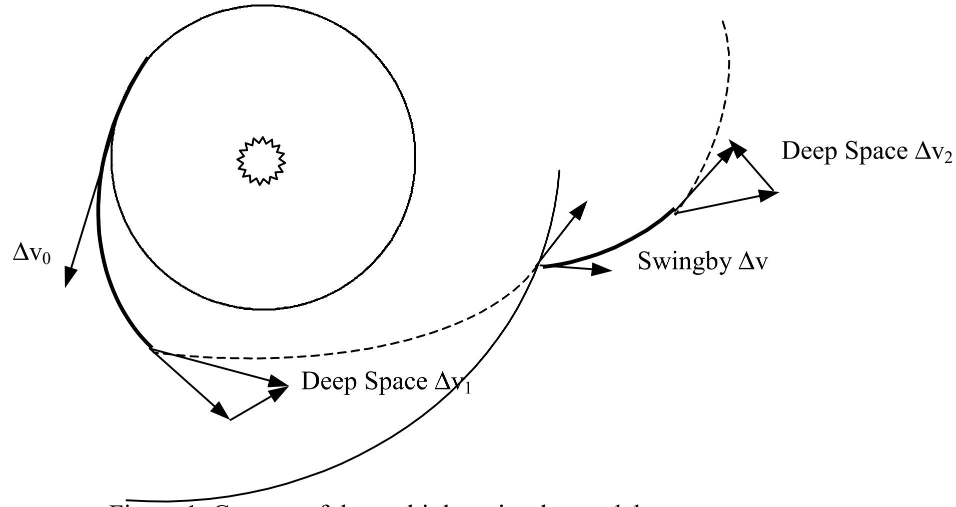

Figure 1. Cartoon of the multiple swing-by model

respectively and n_i is the normal to the projection of the incoming vector onto the xy plane (see Fig.2)..

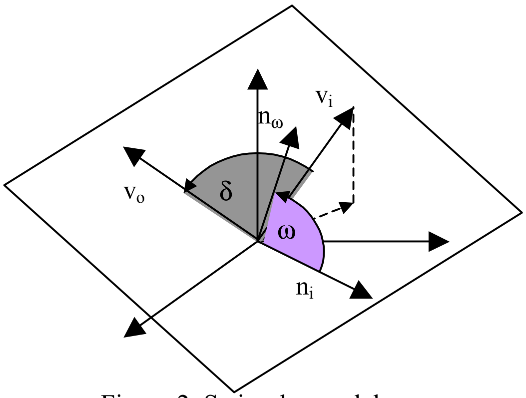

Figure 2. Swing-by model

The outgoing conditions are then propagated for a time t_i up to the deep space maneuver from that point on a coast arc of length T_i is computed solving a Lambert's problem from the maneuver point to the destination planet (see Fig.1). Therefore starting from a planet or a generic point in space it is possible to reach a desired point in space passing by a number of swingbys and providing a corresponding number of Δv maneuvers. The problem can then be written in the following form:

$$\min \quad f(\mathbf{y}) = \sum_{i=0}^{N} \Delta v_i \qquad (9)$$

$$\mathbf{y} \in D$$

where N is the number of encounters after departure and the vector y is defined as:

$$\mathbf{y} = [t_0, \Delta v_0, t_1, T_1, \omega_1, \tilde{r}_{p1}, \ldots, t_i, T_i, \omega_i, \tilde{r}_{pi}, \ldots, t_N, T_N]^T \qquad (10)$$

The problem in this form is amenable to a solution with an algorithm for unconstrained global optimization.

For completely ballistic transfers (with no deep space manoeuvres) with multiple swingbys a simplified and faster 2D model has been used.

Trajectories departing from Earth with e certain Δv_0 at a given epoch have been propagated analytically up to an intersection with the orbit of the first swingby planet. Then a linked-conic model of the swing-by is used to computed post-swingby velocity and again the trajectory is propagated analytically up to the intersection with the orbit of the following swingby planet. The process is repeated till the final target is reached. The miss distance Δr is defined as the difference between each intersection point and the actual position of the planet on its orbit. The problem becomes to minimise the miss distance and at the same time the departure and arrival Δvs:

$$\min \quad f(\mathbf{y}) = \Delta v_0 + \sum_{j=1}^{N} \Delta r_j \qquad (11)$$

$$\mathbf{y} \in D$$

where N is now the number of planetary encounters and the solution vector is:

$$\mathbf{y} = [p_1, \ldots p_i, \ldots p_N, t_0, \Delta v_0, h_{p1}, \ldots,, h_{pi}, \ldots, h_{pN-1}]^T \quad (12)$$

where p_i represent planet's identification number and h_p is the pericentre altitude during swingby.

DFET Optimisation

The problem of designing a trajectory visiting more than one planet on its way to Pluto has been translated into an optimization problem in which the sequence and the date of the encounters are unknown and must be optimized. The problem is a mixed inter-nonliner programming problem with multiple solutions. Due to its nature it can not be solved by a local optimizer based on gradient methods like common NLP solver used in direct collocation and a global technique should be used instead. The global search of the solution space has been carried out using a mixed systematic-stochastic method, combining evolution programming and branching technique[5]. The most interesting preliminary solutions found by the global step have been optimized using DITAN.

POWER SOURCES

Due to the duration and the distance to the sun, only nuclear power sources are currently able to power such a mission. Since 1960s, the US develop and fly RTGs. The Soviet Union and subsequently Russia focused their efforts on space nuclear fission reactors. Thus, the two reliable and flight proven NPS currently available are the US GPHS-RTG, delivering per unit of 56 kg 4.264 kW_{th}/0.285 kW_e and the Russian TOPAZ-1 fission reactor, providing per unit of 980 kg 150 kW_{th}/5 kW_e.[6] For both systems, the mass/power ratio is about 196 kg/kW_e. The Soviet RTGs used onboard of Mars96 are delivering only hundreds of mW and are not considered here[7]. The follow-up soviet reactor version, TOPAZ-2 presented some improvements leading to 1061 kg 135 kW_{th}/5.5 kW_e (193 kg/kW_e).[8] TOPAZ-2 was extensively tested, including nuclear ground tests in the Soviet Union until 1988, but is not yet space-proven[9]

Plotting the two space proven systems with some of the data available from studies on small space reactors shows the gap between power levels of currently existing RTGs and the ones of reactors. (Fig.3) One possibility to fill the gap is to use RTGs in parallel, as has been done for the Cassini spacecraft[6,10] However, from a practical point of view and especially to allow the radiators to function properly, more than 6 units, providing 1.7 kW_e seem unrealistic. The grey line in Fig. 3 symbolizes a hypothetical extension to up to 12 units. Such a combination would deliver 3.4 kW_e with a mass of 672 kg based on the current US GPHS RTG specifications. The ongoing efforts to implement new more efficient conversion systems (e.g Stirling, TPV, MHD) would ameliorate the performances. A programme for a dynamic radioisotope power system (DIPS) existed in the US until 1980. It made use of an organic fluid Rankine system with total power levels of about 1.3 kW_e (min. 0.5; max 2.0 kW_e) at a mass of 215 kg. The 18.3% efficiency would have reduced the mass to power ratio to 165 kg/kW_e.

A minimum size is necessary for reactors to become efficient in terms of mass/power ratios. The only available flight-proven system, the Russian TOPAZ-1 seems to be at the lower end of this scale. The predecessors of TOPAZ-1 and the only flown US space reactor (SNAP-10A) had significantly worse mass/power ratios. (669 kg/kW_e for SNAP-10A)

Despite their conceptual technical differences, Fig. 4 and 5 try to compare different proposed space reactor concepts in terms of kg/kW_e. The advantage of reactors in terms of mass/power ratio becomes clearer by including concepts for space reactors, that extend up into the hundreds of kW_e region.(Fig.4) In this category, the most detailed available study seems to be the US SP-100 reactor concept, delivering 105 kW_e at a mass of about 4.6 tons.[11,12] More advanced concepts, such as the French ERATO and the UK UKSR, delivering about 200 kWe at 7 tons and 8.2 tons respectively, make use of dynamic energy conversions systems (Brayton cycle with He/Xe at estimated 18-20%) instead of the thermoelectrically obtained 4.6% for the SP-100 concept.

One of the disadvantages of reactors compared to RTGs is their usually limited lifetime. While the first reactors were designed only for a one year lifetime, the TOPAZ versions have nominal lifetimes of 1 to 3 years. The SP-100 system was intended to provide 7 years of continuous power within a 10 years frame. Recently proposed planetary surface reactors to power robotic and/or human outposts on Mars are also designed to function for 10 years without maintenance.

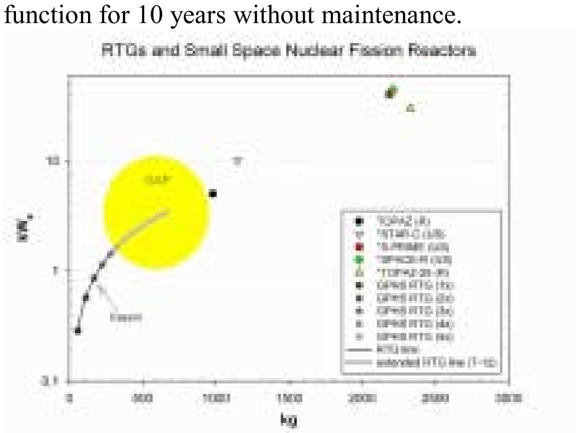

Figure 3: Mass to power plot for RTGs and small space reactors (* mark concepts).

Despite the huge differences in concepts, ranging from the nuclear fuel to the conversion system, the direct comparison of several available proposed and realised systems shows that a fairly good linear relationship between the mass and the delivered power can be found. (Fig.5) The relationship could be expressed as

$$P_{kW_e} \approx 0.0293 \cdot M_{kg} - 25.63 , \qquad (13)$$

where P is the delivered power in kW_e and M is the total mass of the reactor system. According to this relation, the lower mass limit of reasonably proportioned reactor systems would be about at 860 kg, in good agreement with the characteristics of TOPAZ-1.

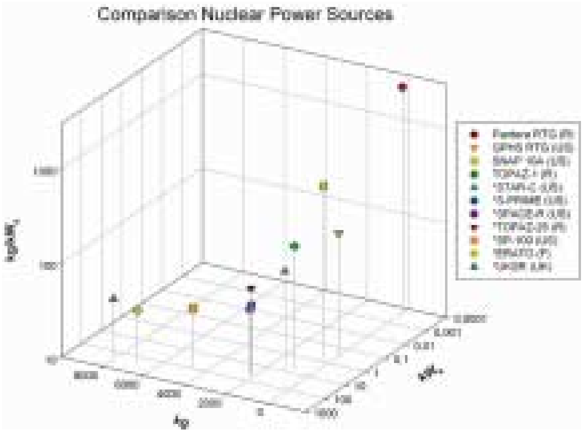

Figure 4: Mass/Power ratio comparison for different NPS.

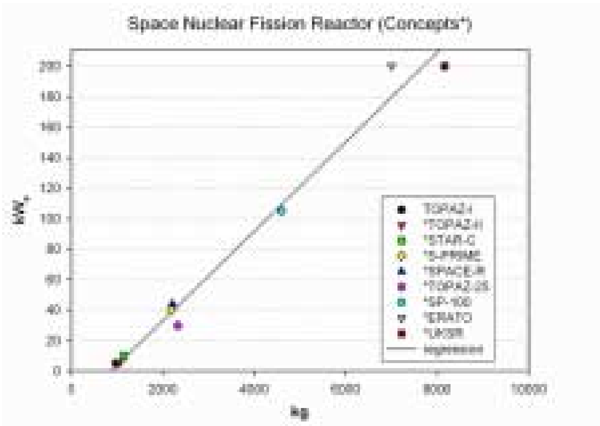

Figure 5: Power to Mass relation of nuclear reactors

EARTH-PLUTO MISSION

Any mission to the furthest planet of the solar system requires a high Δv and a considerable travel time. In a previous study[13] it has been considered the possibility of sending a small probe (about 600kg of wet mass) to Pluto using nuclear electric propulsion, via a swing-by of Jupiter.

This previous study demonstrated the possibility of injecting some 400 kg into a highly elliptical orbit around the distant planet provided a very low thrust propulsion system (about 0.038 N) powered by a cluster of four RTGs.

Due to the long trip to Pluto in the same study it was investigated the possibility to deliver a probe around Jupiter during the swing-by. The encouraging preliminary results opened the way to a more detailed analysis. Here, a slightly different scenario is proposed: a carrier spacecraft carries a piggy-back probe from Earth to Jupiter, then the probe is released and continues its course towards Pluto, while the carrier spacecraft is injected into an orbit around Jupiter. In this scenario the probe is equipped with RTGs and a very low-thrust electric engines (0.05N) and low I_{sp} requiring 1kW of power, while the carrier is equipped with a cluster of high thrust high specific impulse electric engines powered by a 50 kW_e. fission reactor with a total mass of about 2.5 tons including shielding. The mission duration and the Jovian radiation environment are favourable for such a choice. A first approximation of the total fractional dose accumulated during the different manoeuvres until the final Europa orbit by using the simplified periapsis dependent function described in [14] and [15] gives a fractional dose of 1.06.

A summary of the characteristics of the carrier and of the probe for both scenarios can be found in Table 1.

Table.1. Main characteristics of the two spacecrafts

S/C	POWER SOURCE	THRUST (N)	I_{SP} (S)	MASS (KG)
Probe	RTGs	0.05	3000	1e3
Carrier	Reactor	1.5	6000	5e3

First of all the opportunity to use a swingby of the Earth or of another planets of the inner solar system has been investigated. The vector **y** is then extended to include a combination of possible planetary encounters:

$$\mathbf{y} = [p_0,\dots,p_i,\dots,p_N,t_0,\Delta v_0,t_1,T_1,\omega_1,\tilde{r}_{p1},\dots,t_i,T_i,\omega_i,\tilde{r}_{pi},\dots,t_N,T_N]^T \qquad (14)$$

where p_i is the reference number of planet i-th. Now considering a departure from the Earth and two possible encounters before Pluto we take $p_0=3, p_N=9$ (with N=3) , p_1 and p_2 in the interval [1,9]. After 3000 evaluations of the function f with the combined evolution-branching algorithms we obtain a number of interesting solutions. It is remarkable that all the best solutions found have a sequence $p=[3,3,5,9]^T$ confirming that a direct launch to Jupiter is an optimal strategy. The final transfer time T_N has un upper limit of 3000 days because we want a fast transfer to Pluto, some interesting alternative and the launch date has a lower limit 5000MJD because the departure must be in the range [5000,8000]. In this interval there are many different possible launch windows with different characteristics. The best solution found for the sequence [3,3,5,9] is summarised in Tab 2..

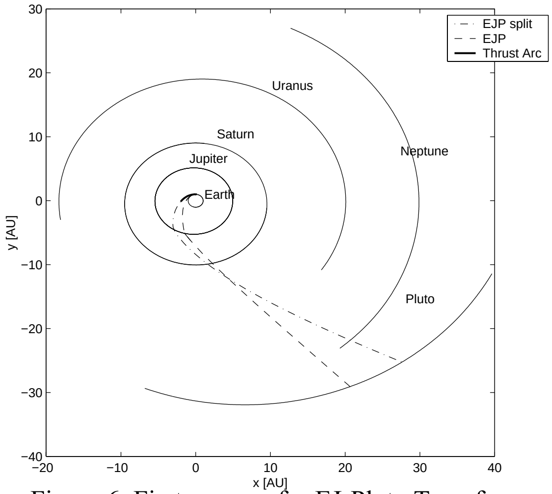

Figure 6. First guesses for EJ-Pluto Transfers

Table 2. First guess for the EJ-Pluto transfer

VALUE	SPLIT OPTION
C_3	81 km^2/s^2
Launch Date	12/12/2016
Jupiter Encounter	02/07/2018
Swingby pericenter	1.841e6km
Δv maneuver	-
Pluto Arrival	22/03/2035

Then first guess solution of Tab. 2 has been fully optimised reducing the C_3 at launch down to 36 km^2/s^2. The resulting solution has been plot in Fig. 7. For the first leg of the trajectory the two spacecrafts are together and it has been assumed that only the engines of the carrier are used. From Jupiter to Pluto the probe continues with its own engines.

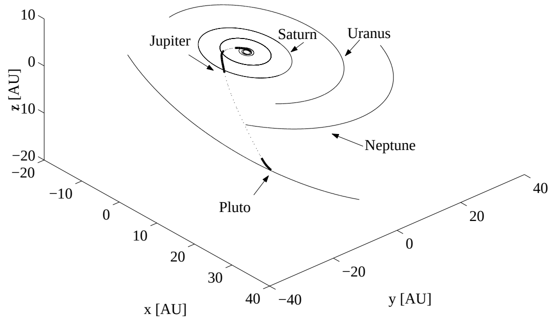

Figure 7. Earth-Pluto optimised transfer

The probe reaches the sphere of influence of Pluto with a relative velocity of 2.185 km/s and brakes to reach a distance of 1e6 km with a velocity of 0.141 km/s and a mass of 681.5 kg, then the engines are injected again to put the spacecraft into an high elliptical orbit. In the meanwhile the carrier uses a first swing-by of Ganymede to be captured into the jovian system (see Fig. 8)-

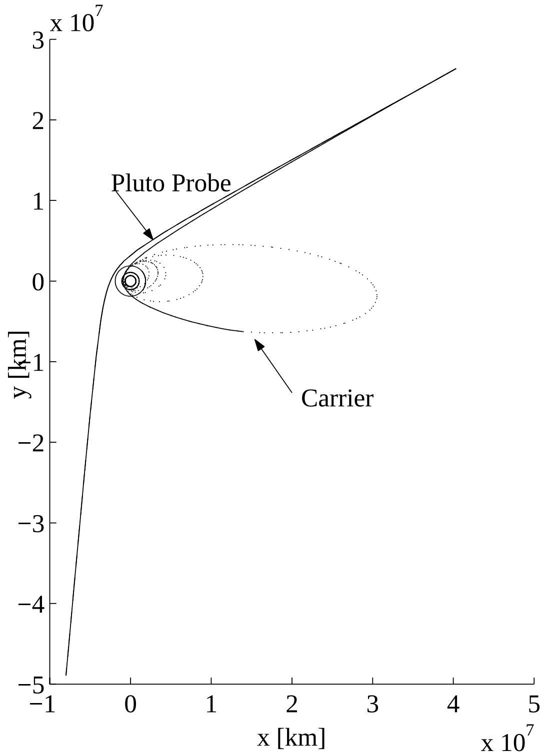

Figure 8. Trajectory within Jupiter's sphere of influence. Solid lines represent thrust arcs while cost arcs are represented by doted lines.

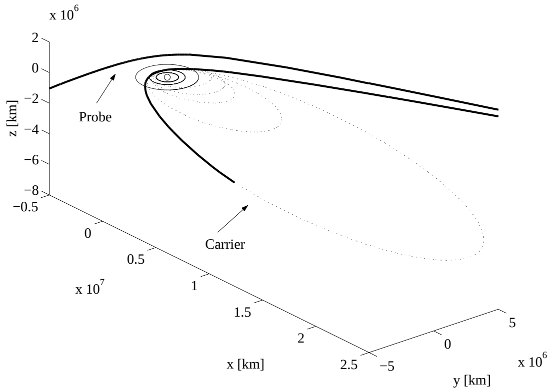

Figure 9. Close up of the trajectory within the sphere of influence of Jupiter.

At this point a sequence of swingbys of Ganymede are used to reduce the apocentre . The sequence, represented in Fig 9 and summarised in Tab. 4, is stopped when the spacecraft intersect the orbit of Europa..

Table 3. Summary of EJ-Pluto Transfer

PLANET	DATE	H_P (KM)	V_∞ (KM/S)
Earth	21/11/2016	/	6
Jupiter	07/08/2018	2.52e7	12.71
Pluto	28/04/2036	1e6	0.141

Table 4. Summary of Jupiter Tour

MOON	DATE	HP (KM)
Ganymede	20/10/2018	200
Ganymede	25/11/2019	203.7
Ganymede	04/02/2020	200.0
Ganymede	04/03/2020	200.0
Ganymede	25/03/2020	200.0
Ganymede	08/04/2020	/

The final mass of the carrier when intersecting the orbit of Europa is 4100 kg and can start its primary mission with a considerable payload. This scenario requires quite advanced technologies both for the carrier and for the launcher. It has been assumed that an Ariane 5 'Initiative 2010' will be used and that by 2016 the required propulsion technology will be available.

EARTH-JUPITER MISSION

The idea of a small probe equipped with electric propulsion and RTGs can be interesting even for a mission to a less distant target: Europa. In this section the possibility of injecting a 600 kg probe around Europa is investigated.

Since the attempt is to design a relatively low cost mission, Soyuz has been taken as reference launcher. However the C_3 capability of Soyuz is not sufficient for a direct injection into an Earth-Jupiter transfer. Therefore a preliminary search for interesting launch windows involving a sequence of swing-bys of the Earth and Venus have been carried out using a global approach.

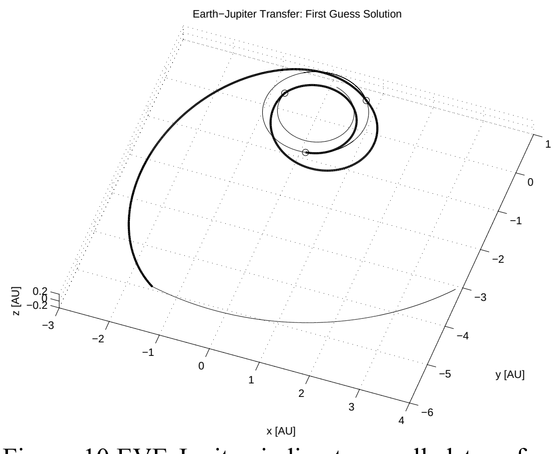

Figure 10.EVE-Jupiter indirect propelled transfer: first guess

Preliminary global search have been performed at first looking for short transfers with a limited sequence of swingby. The optimal one, represented in Fig. 10, makes use of a swing-by of Venus and a single swing-by of the Earth. However this sequence with an escape velocity of 3.6 km/s would require for a deep space manoeuvre of 4 km/s for a total transfer time of less than 4 years.

Therefore we looked for a better, even though slower, option (Fig. 11). If the time is not an issue an EVEEJ (Earth-Venus-Earth-Earth-Jupiter) transfer gives a considerable flexibility in terms of launch window and launch opportunities. Moreover even if the v_∞ is slightly higher, 3.8 km/s, no deep space manoeuvres are required and the arrival velocity at Jupiter is relatively slow, just 5.71 km/s.

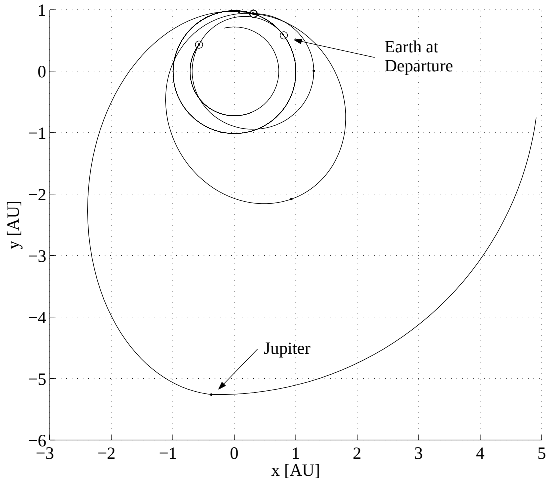

Figure 11. First Guess for the EVEE-Jupiter unpropelled transfer

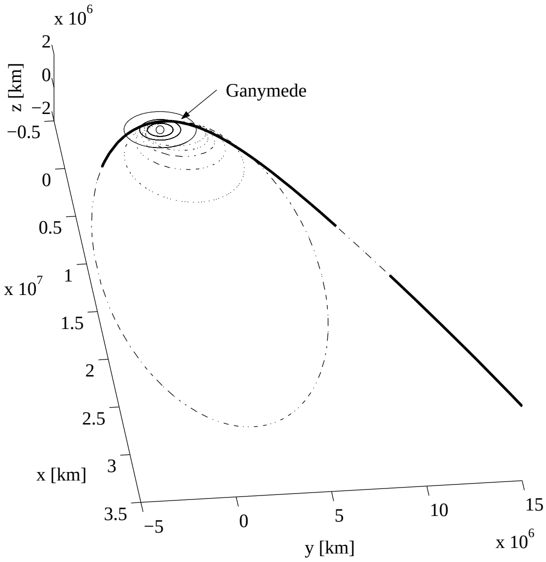

Figure 12. Optimised capture and descent to Europa. Solid lines represent thrust arcs while dashed lines represent coast arcs.

Table5 Main characteristics of the spacecrafts

POWER	THRUST (N)	I_{SP}(S)	MASS (KG)
RTGs	0.05	3000	600

This second first guess was then optimised with electric propulsion and the resulting trajectory has been plot in Fig. 13 and the swingby sequence summarised in Tab. 6. This mini satellite is assumed to be equipped with 4 ion engines operated 2 at times and 4 RTGs delivering 1.2 kW, the main characteristics of the spacecraft are summarised in Tab. 5.

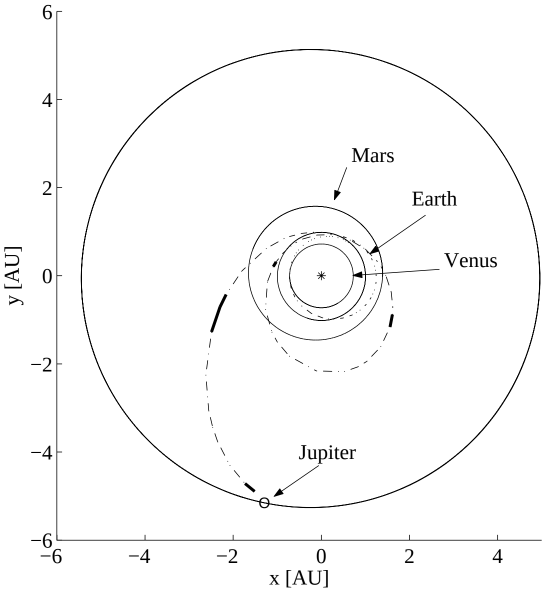

Figure 13. Optimised EVEE-Jupiter transfer. Solid lines represent thrust arcs and dashed lines coast arcs.

Once the probe is at Jupiter the same strategy as for the previous scenario is used to insert the spacecraft into a high elliptical stable orbit around the giant planet. Again a sequence of swingbys of Ganymede are used to reduce the apocentre. This sequence is here stopped at the first encounter with Europa.

Table 6. Summary of EVEEJ transfer

PLANET	DATE	H_P (KM)	V_∞ (KM/S)
Earth	26/10/2013	/	3.8
Venus	19/02/2014	282.35	4.39
Earth	29/11/2014	726.4	8.12
Earth	24/11/2016	814.31	9.1
Jupiter	29/05/2019	/	4.91

Table 7 Summary of Jupiter Tour

MOON	DATE	HP (KM)
Ganymede	29/05/2019	200.0
Ganymede	04/10/2020	200.0
Ganymede	22/12/2020	237.3
Ganymede	27/01/2021	233.8
Ganymede	17/02/2021	222.6
Ganymede	03/03/2021	217.6
Europa	20/03/2021	300.0

A summary of the descent toward Europa is reported in Tab.7 and the trajectory represented in Fig. 12. The spacecraft reaches its final destination with a residual mass of 540 kg. In this second scenario the technology required both for the launch and for the propulsion system is less demanding demonstrating the feasibility of a small mission to the outer planets with nuclear electric propulsion.

CONCLUSIONS

In this paper two options, one for an advanced combined mission to Pluto and to Jupiter, the other for a cheap mission to Europa have been analyzed. Starting from a previous study the possibility of delivering a spacecraft in orbit around Jupiter during the swingby required to send a small probe to Pluto has been studied.

Although the technology required for this type of mission appears to be quite advanced never the less with a single launch of two spacecrafts this study demonstrated that a 681 kg probe could be put in orbit around Pluto using electric propulsion and RTGs while a second 4100kg spacecraft, equipped with a nuclear reactor, is operating in Jupiter orbit.

Then,the idea of a small spacecraft propelled by very low-thrust ion engine and powered by RTGs has been applied to a mission to Europa demonstrating the feasibility of this option.

Both analysis will be extended further, in a future work, considering the final descent to Europa for both spacecrafts in orbit around Jupiter.

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