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OPTIONS FOR A MISSION TO PLUTO AND BEYOND

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Abstract

In this paper several options for a potential mission to the limits of the solar system are presented. Starting from a direct launch to Jupiter used to boost the spacecraft toward Pluto the paper analyses several other possibilities using both chemical and nuclear electric propulsion, multiple gravity assist maneuvers and aero-gravity assist maneuvers. Some scenarios include a combined mission to a closer system or a return mission. The wide range of possibilities has been obtained using two different global approaches to span the solution domain for a possible transfer to Pluto and beyond. The most promising solutions have been refined and fully optimized using a direct method.

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 and that of the Kuiper Belt Objects, to which they probably belong or are closely related to. The latter is a family of bodies outside of Neptune's orbit that have been identified to be a source of short-period comets. Currently many KBO have been observed, and some of them have extremely interesting features variability (e.g. Chiron), strong absorption of blue light (maybe indicating that they are completely covered by complex organic molecules), diversity, etc. A mission to the Pluto-Charon system and 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 favorable 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[2]. On the other hand, orbits using no thrust arcs or impulsive shots can quickly be assessed by means of 'C_3 matching' and a simple enumerative search. A number of alternative chemical propulsion orbits using a swing-by around Jupiter and other bodies are given. To decrease the enormous launch energy required some alternatives using aero-gravity assists are also shown.

SCIENCE OBJECITVES

A mission to the Pluto-Charon system and to a Kuiper Belt Object (KBO) 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.

The KBO is a family of bodies outside of Neptune's orbit that have been identified to be a source of short-period comets. Currently many KBO have been observed, and some of them have extremely interesting features variability (e.g. Chiron), strong absorption of blue light (maybe indicating the presence complex organic molecules), diversity, etc. This would also have strong exobiological interest, provide important clues on the origin of volatiles and organic molecules that enabled the appearance of life on our own planet.

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

1. Surface chemical composition
2. Surface morphology
3. Atmospheric chemistry
4. Gravimetry

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

1. Imaging X-ray Spectrometer
2. Wide / Narrow Field Imager
3. IR-Spectrometer
4. 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.

MISSION ANALYSIS

Two different classes of options have been investigated, namely chemical propulsion options with aero-gravity assist (AGA) maneuvers and nuclear electric propulsion (NEP) options with gravity assist maneuvers. The two classes of problems have been solved with two completely different approaches; however a comparison between the two is out of the scope of this paper since our main interest is to show how a broad variety of demanding trajectory design problems can be solved with either of the two of the approaches presently under development at ESTEC. As it will be discussed later, both methods perform well in the related field of application providing interesting and valuable results.

CHEMICAL AND AGA OPTIONS

Orbits using no thrust arcs or impulsive shots can quickly be assessed by means of 'C_3 matching' i.e. the arrival C_3 before swing-by equals the departure C_3 after the swing-by (taking into account the constraint of maximum deflection angle). Planet departure/arrival/swing-by times are optimized using a simple enumerative search. Trajectories are calculated using Lambert Solving routines and solutions are stored for which the incoming C_3 of a swing-by matches the outgoing C_3. This method has proven to be very effective for feasibility studies since it gives solutions of reasonable accuracy in short time (order of minutes). Furthermore it allows for easy implementation of constraints (maximum launch C_3 and/or declination, maximum transfer time etc.) and avoids tuning of optimization parameters. A software tool called Swing-by

Calculator (SBC) was developed implementing these features[7].

A number of data-bases has been implemented, including different versions of Ariane 5. These versions include the Ariane 5 ECA, ECB and 'Initiative 2010'. ECA is the current version while ECB is to have its maiden launch in 2006 and Initiative 2010 its first flight in 2010. If we consider a launch in the timeframe of 2016-2018, the choice of the Ariane 5 2010 is the most obvious one. This launcher has the objective of having a 43% increase in perfromance over Ariane 5 ECB (GTO performance increases from 11.2 to 16 tons). The performance of ECA and ECB for a wide variety of C_3 and declination of the escape asymptote is know from Biesbroek & Ancarola[5]. The '2010' performance is taken as ECB performance (multi-injection case) plus 43% and is shown Fig.2. Note that a 10% margin is subtracted. The performance is symmetrical with respect to the 0° declination due to the use of a circular parking orbit. The maximum performance can be found for 5.2° declination which is the latitude of the launch site (Kourou, French Guyana).

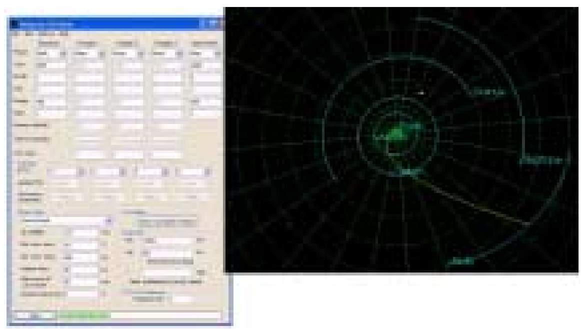

Figure 1: Overview of the SBC for non-SEP mission design showing an Earth-Jupiter-Pluto result

It can be seen from Fig.2 that the performance decreases rapidly above a C_3 of 95 km^2/s^2. It would probably be wiser to use a high-elliptic parking orbit, detach from the Ariane 5 upper-stage and use an extra solid kick-stage. This would give similar performance as, for example, a Delta-IV with Star48V solid motor. However to stay on the conservative side we use the data as shown in Fig.2.

In order to have some satellite-mass left after launch the departure C_3 should be restricted to 95 km^2/s^2, which excludes the use of a direct transfer to Pluto. Figure 3 shows the launch and arrival C3 for a launch in the year 2016, and ranging the travel time from 8 to 20 years. A launch in 2017 and 2018 gives almost exactly the same results as the figure for 2016.

A minimum launch C_3 appears for a transfer time of 12 years (launch in 2016/2017/2018 and arrival in 2027/2028/2029 respectively) which is equal to 204 km^2/s^2 for 2016, 211 km^2/s^2 for 2017 and 216 km^2/s^2 for 2018). Therefore, the minimum C3 for direct transfer is still more than twice the available launch energy.

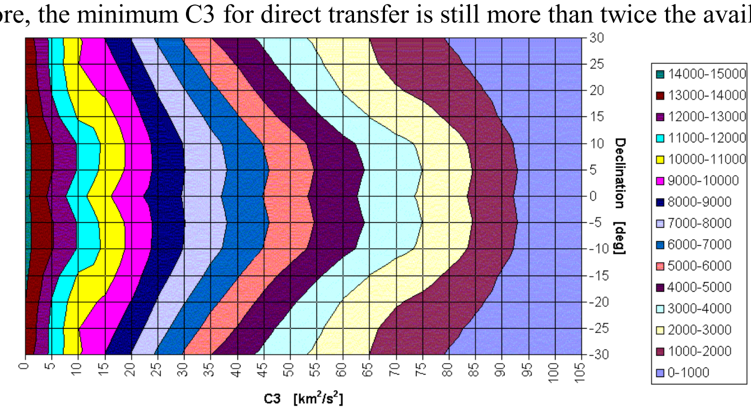

Figure 2: Estimated Ariane 5 'Intiative 2010' performance. Taken from Biesbroek[6]

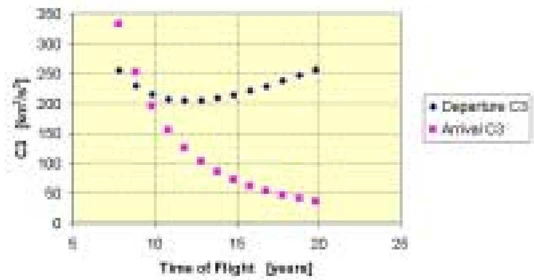

Figure 3: Launch and arrival C3 for direct transfer to Pluto

EJP un-powered

The use of a swing-by around a heavy planet could considerably decrease the launch energy. In particular a Jupiter swing-by has played a crucial role in mission feasibility for the New Horizons Pluto-Kuiper mission[1] and is an obvious choice for any trajectory design to Pluto.

Two opportunities for a Earth-Jupiter-Pluto (EJP) trajectory without deep-space manoeuvres exist in the 2016-2018 launch time-frame namely launching in December 2016 or January 2018. Jupiter swing-by should be in 2018/2019 respectively. For both opportunities, three different transfer times were taken into account: 10, 15 and 20 years. Table 1 below shows an overview of the minimum launch C3 trajectories found for these six possibilities. Taking into account the performance of Ariane 5, we see that only the solutions of 2016 comply to the performance constraints, and that the transfer time should be at least 15 years. The declination of the escape asymptote is relatively low (between –10° and +10°) which is favourable for a launch from Kourou. The arrival velocity decreases with increasing transfer time, leading to a minimum of 6 km/s for a 20 year transfer. Table 1 below also shows the launch masses for the six possibilities, after subtracting an Ariane 5 adapter mass of 145 kg:

Table 1: Overview of EJP transfers for 2016/2018 launch windows

Launch date	Jupiter fly-by	Pluto arrival	Transfer time	C3	declination	Arrival vel.	Launch Mass
17/12/2016	30/06/2018	01/01/2028	11 years	103.6	8.17	14.2	0
17/12/2016	07/08/2018	01/01/2032	15 years	97	9.43	9.4	248
16/12/2016	31/08/2018	01/01/2037	20 years	94	10.08	6.3	635
01/02/2018	09/04/2019	01/01/2028	10 years	145	-7.60	15.0	0
13/01/2018	24/05/2019	01/01/2033	15 years	118	-9.53	8.9	0
13/01/2018	13/06/2019	01/01/2038	20 years	111	-9.00	6.0	0

Table 2: Overview of the 2016 20-year EJP transfer

Launch Date	16 Dec 2016
Launch Mass	635 kg
Launch C3	94 km^2/s^2
ΔV for mid-course manoeuvre	50 m/s
Jupiter swing-by	31 Aug 2018
ΔV for mid-course manoeuvre	50 m/s
Pluto arrival	1 Jan 2037
Pluto arrival velocity	6.3 km/s
Pluto orbit insertion ΔV	5.726 km/s
Total propellant mass	533 kg
Tank mass (assumed 10% of prop. mass)	53 kg
Satellite dry mass excluding tanks	49 kg
Total mass in 100 km Pluto orbit	102 kg

Unfortunately, this does not look very positive. The 20-year transfer of 2016 seems the most favorable but still the launch mass is too low to establish an orbit around Pluto. Table 2 highlights the features of such a scenario, using a specific impulse of 325 seconds and a final circular orbit of 100 km altitude.

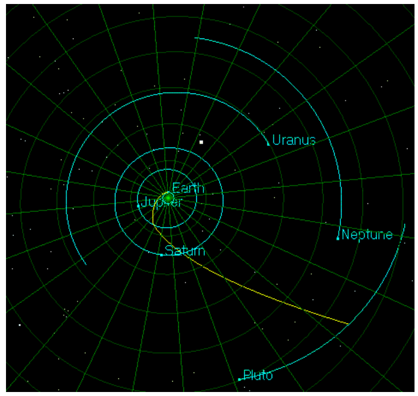

Figure 4: EJP (un-powered) 20-year transfer with launch in 2016

A 102 kg dry mass of which 53 kg consists of tanks is a complicated system design, but not necessarily unfeasible for a design of 2016. Still, we need to consider alternative solutions with higher mass. Figure4 shows the trajectory for this scenario.

EJP powered

The situation may be improved by using a powered swing-by around Jupiter. Due to the enormous sphere of influence of Jupiter, the swing-by itself takes more than a month to complete (time between entering and leaving the sphere of influence) which gives the opportunity for applying thrust during the swing-by. The same 6 possibilities were computed again, now constraining the launch C_3 to 100 km^2/s^2 and allowing a burn during the Jupiter swing-by. The ΔV's mentioned are the ones required for the powered swing-by. The final mass is calculated using also 2 times a mid-course maneuver of 50 m/s.

Table 3: Overview of EJP transfers using power Jupiter swing-by (I_{sp} = 325 s)

Launch year	Jupiter fly-by	Pluto Arrival	Transfer time [yrs]	C3 [km^2/s^2]	ΔV [km/s]	Launch mass [kg]	Fly-by mass [kg]	Mass in 100 km-orbit [kg]
16/12/2016	07/11/2018	01/01/2027	10	88	3.430	1359	449	0
15/12/2016	26/10/2018	01/01/2032	15	88	1.645	1273	736	0
15/12/2016	24/10/2018	01/01/2037	20	89	1.121	1256	856	141
16/01/2018	30/07/2019	01/01/2028	10	98	4.997	147	0	0
15/01/2018	07/10/2019	01/01/2033	15	88	4.817	1423	304	0
15/01/2018	08/10/2019	01/01/2038	20	88	3.931	1436	405	0

Even though the situation has improved for a Pluto fly-by mission (mass increases from 635 kg to 856 kg for a 20-year mission), the situation is less favourable for a Pluto orbit mission: the 20-year 2016 mission gives a solution of 141 kg mass in Pluto orbit, of which 112 kg is tank mass (assuming a tank relation of 10% of propellant mass) In total this gives 29 kg of mass excluding tanks. This decrease is due to the fact that the powered swing-by does decrease the launch C_3, but it also increases the arrival C_3 leading to lower mass in orbit. Figure 1 shows the EJP trajectory for this 20-year transfer launched in 2016.

AGA OPTIONS

An interesting alternative approach may be applied when using 'Aero-Gravity Assists' (AGA's). This principle is based on the fact that when using aero-braking, the deflection angle of a gravity assist can become much larger than a conventional gravity assist. From the optimiser we get the deflection angle for each gravity assist from which, assuming a certain swing-by altitude, we can calculate the required L/D ratio and compare it to a value that is supplied by the user.

A typical AGA trajectory to Pluto is the Earth-Venus-Mars-Pluto (EVMP) trajectory, as studied by Bonfiglio and Longuski[15]. A launch opportunity for such a trajectory exists in February 2017. The Venus AGA will then be at end of August / beginning of September 2017, Mars AGA in December 2017 and the

Pluto arrival date can be varied to comply to a 10, 15 or 20-year transfer. Figure 5 below shows the launch mass for a range of different trajectories: the transfer time is varying between 10 and 20 years and the L/D ratio is ranging from 4 to 10. The swing-by altitude for both the Venus and Mars AGA is set to 80 km.

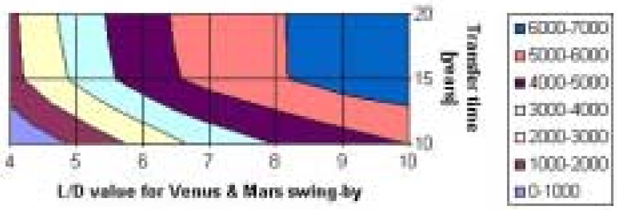

Figure 5: Launch mass (excluding 145 kg adapter) for EVMP transfers using AGA

A number of features can be seen from this surface plot. First of all the performance goes up with increasing L/D ratio. No solutions were found for L/D ratios below 4. Present technology of 'standard' satellites limits the L/D ratio to 2 however we may assume higher numbers for a launch in 2017. Secondly, the launch mass increases with increasing transfer time, however the difference in launch mass between a 15-year and 20-year transfer is relatively small. The launch mass is quite high compared to the EJP transfers (ranging from 1159 kg for a 10-year transfer with L/D = 5, to 6734 kg for a 15-year transfer with L/D = 10). The C_3 ranges from 30 to 80 km^2/s^2.Using a specific impulse of 325 seconds, and assuming 50 m/s mid-course manoeuvres for each transfer leg, we can calculate the mass at Pluto arrival:

Table 4: Satellite mass at Pluto arrival for EVMP transfers using AGA

		Transfer time [years]		
		10	**15**	**20**
L/D	**4**	0	1618	1727
	5	1106	3049	3317
	6	2177	4340	4452
	7	3228	5091	5254
	8	3839	5621	5628
	9	4340	6136	6224
	10	4752	6424	6164

Unfortunately the arrival velocity for the AGA transfer is very high; about 15.7 km/s for the 10-year transfer, 9.2 km/s for the 15-year transfer, and 6.3 km/s for the 20-year transfer. If we want to establish planetary orbit, this would lead to very high insertion ΔV and therefore a very high propellant mass and tank mass. The AGA therefore allows lowering the launch C_3, but in turn it speeds up the trajectory leading to a high arrival velocity. The following two figures show the satellite mass in orbit, subtracted by 10% tank mass. Two final orbits are chosen: one circular 100 km orbit and one 10-day orbit with periapsis altitude of 100 km and apoapsis altitude 51252 km.

Table 5: Satellite mass (excluding tanks) in 100 km Pluto orbit for EVMP transfer

		Transfer time [years]		
		10	**15**	**20**
L/D	**4**	0	0	131
	5	0	0	252
	6	0	0	337
	7	0	0	399
	8	0	0	428
	9	0	0	473
	10	0	0	468

Table 6: Satellite mass (excluding tanks) in 10-day Pluto orbit for EVMP transfer

		Transfer time [years]		
		10	**15**	**20**
L/D	**4**	0	0	156
	5	0	0	299
	6	0	0	400
	7	0	0	473
	8	0	0	507
	9	0	0	561
	10	0	0	555

We can observe two major aspects: first of all the transfer time needs to be 20 years. For the 10-year and 15-year transfer the mass of the tanks is simply to large and requires more than the available mass. Even though the launch energy is not much different between the 15-year and 20-year transfer, the difference in orbit insertion is very high, making the 15-year transfer unfeasible for orbit insertion. Secondly, the difference in payload mass between a 100 km orbit and 10-day orbit is about 19%. Different scenarios were tested with higher apoapsis, however the decrease in orbit insertion ΔV was minimal. The total mass in orbit is shown below for the 20-year transfer. Note that the maximum is found for a L/D = 9.

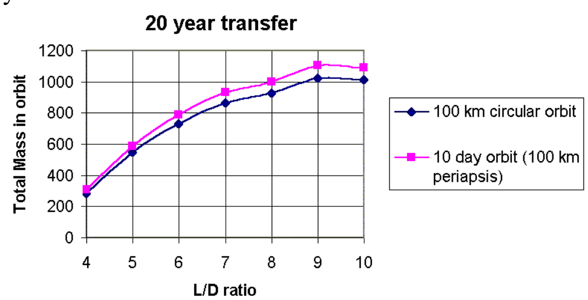

Figure 6: Total mass in orbit for 20-year EVMP transfer

We can select a baseline mission by assuming a L/D ratio of 4 and a 20-year transfer time. Table 7 highlights the features of such a scenario, using a specific impulse of 325 seconds and a final circular orbit of 100 km altitude: The trajectory is shown in Fig.7 with on the left the transfer until Mars and on the right the transfer from Mars to Pluto.

Table 7: Overview of the 2017 20-year EVMP transfer using AGA (L/D = 4)

Launch Date	8 Feb 2017
Launch Mass	1810 kg
Launch C3	72 km^2/s^2
ΔV for mid-course manoeuvre	50 m/s
Venus AGA	7 Sep 2017
ΔV for mid-course manoeuvre	50 m/s
Mars AGA	17 Dec 2017
ΔV for mid-course manoeuvre	50 m/s
Pluto arrival	1 Jan 2037
Pluto arrival velocity	6.3 km/s
Pluto orbit insertion ΔV	5.844 km/s
Total propellant mass	1526 kg
Tank mass (assumed 10% of prop. mass)	153 kg
Satellite dry mass excluding tanks	131 kg
Total mass in 100 km Pluto orbit	284 kg

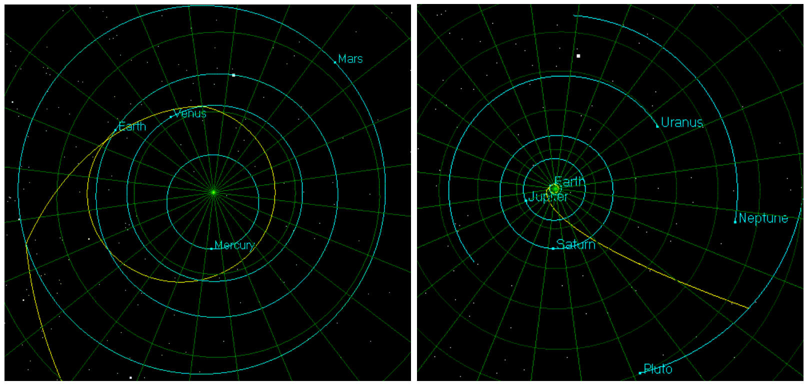

Figure 7: EVMP 20-year trajectory

AGA options using Jupiter swing-by

As mentioned in the abstract of this paper, a preference goes to a fly-by or swing-by around Jupiter during the transfer, to allow for a probe to be ejected to Europa. A search was performed to find EVMJP trajectories (Earth-Venus-Mars-Jupiter-Pluto) using an AGA at Venus and Mars (swing-by altitude 80 km) and a powered swing-by at Jupiter (minimum swing-by altitude 650,000 km) however Jupiter is, in the launch frame 2016-2018 not in a perfect position to exploit this feature. As a result, a Jupiter swing-by has a negative impact on the arrival mass and in particular the mass in Pluto orbit, see Fig.8. The launch mass is almost twice as low as the EVMP transfers.

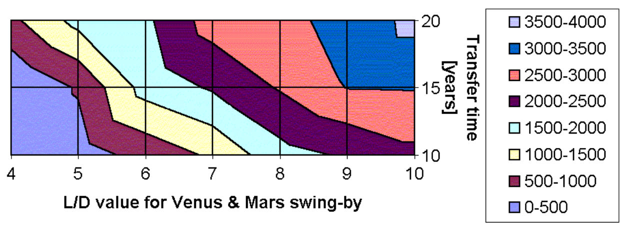

Figure 8: Launch mass (excluding 145 kg adapter) for EVMJP transfers using AGA

Table 8 gives an overview of the total satellite mass when arriving at Pluto. The typical transfer scenario is a launch in January / February 2017, a Venus AGA in August / September 2017, Mars AGA in December 2017. The Jupiter swing-by occurs in December 2018 for a 10-year transfer, in January / February 2019 for a 15-year transfer and April / May 2019 for a 20-year transfer. Using a circular final orbit of 100 km, and subtracting tank masses (10% of propellant), we get the masses in orbit reported in Tab.9:

We see that, in order to have some mass left in orbit, we are obliged to use high values of the L/D ratio (at least 7). Also note that even though a solution is found for L/D of 5, no solution is found for L/D = 6. No solutions were found for 10-year and 15-year transfers. Some of the solutions were using powered Jupiter gravity assists, others converged to un-powered gravity assists. A high sensitivity of the final mass was found for different swing-by dates and for swing-by altitude of the AGA's. It may be that other solutions exist using a different altitude. We can select a baseline mission by assuming a L/D ratio of 10 and a 20-year transfer time. The following table highlights the features of such a scenario, using a specific impulse of 325 seconds and a final circular orbit of 100 km altitude. The trajectory is shown in Fig. 9.

Table 8: Satellite mass at Pluto arrival for EVMJP transfers using AGA

		Transfer time [years]		
		10	**15**	**20**
L/D	**4**	0	0	499
	5	185	374	1041
	6	503	1593	996
	7	890	1935	2022
	8	1709	2097	2740
	9	1516	2829	1790
	10	1989	2846	3445

Table 9: Satellite mass (excluding tanks) in 100 km Pluto orbit for EVMJP transfer

		Transfer time [years]		
		10	**15**	**20**
L/D	**4**	0	0	0
	5	0	0	15
	6	0	0	0
	7	0	0	82
	8	0	0	189
	9	0	0	198
	10	0	0	239

Table 10 Overview of the 2017 20-year EVMJMP transfer using AGA (L/D=10)

Launch Date	9 Feb 2017
Launch Mass	3667 kg
Launch C3	51.44 km^2/s^2
ΔV for mid-course manoeuvre	50 m/s
Venus AGA	3 Sep 2017
ΔV for mid-course manoeuvre	50 m/s
Mars AGA	13 Dec 2017
ΔV for mid-course manoeuvre	50 m/s
Jupiter swing-by	20 Mar 2019
ΔV for mid-course manoeuvre	50 m/s
Pluto arrival	1 Jan 2037
Pluto arrival velocity	6.4 km/s
Pluto orbit insertion ΔV	5.606 km/s
Total propellant mass	3073 kg
Tank mass (assumed 10% of prop. mass)	307 kg
Satellite dry mass excluding tanks	286 kg
Total mass in 100 km Pluto orbit	593 kg

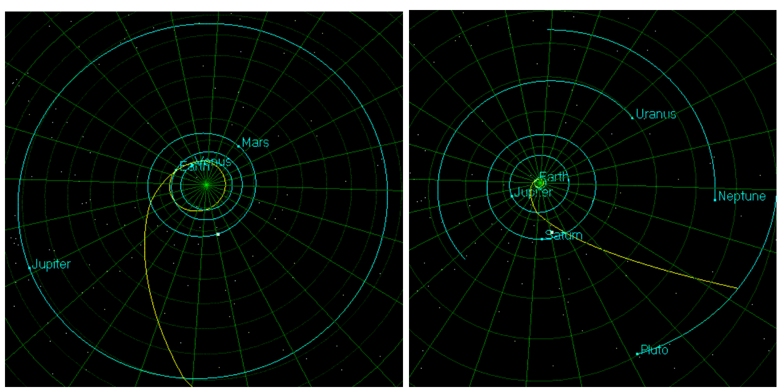

Figure 9: EVMJP 2017 20-year transfer

NEP OPTIONS

Advanced kinds of propulsion systems can provide valuable alternative ways to reach far destinations like Pluto. Electric propulsion is an option even though it is quite demanding in terms of power. Solar power however is already not sufficient at Jupiter distance and therefore the source of power must be found among nuclear devices. In the following chapter a more detail analysis of power requirements will be presented. Here we investigate an alternative option of the NASA new horizon mission design for the Pluto-Kupier belt mission using electric propulsion. The launch date must be around 2006 with an arrival in a date between 2015 and 2020 with direct launch from Earth to Jupiter, which provides a meaningful Δv to boost the spacecraft to Pluto. The early launch date does not allow to think of advanced systems not yet developed and following the NASA philosophy the mission must be as cheap as possible.

We selected a 600kg class spacecraft equipped with a state of the art electric propulsion engine; several different trajectories have been designed using either ion engines or SPT with different thrust levels (from 34mN to 40mN) or Isp (from 1700s to 4100s). A launch capability of 12 km/s is expected from the evolution of A5 for a low declination therefore we forced a C_3 of 144 km^2/s^{2}and a declination of less than 12 degrees. Following the same strategy of the NASA mission the propellant mass has been optimized to reach Pluto with different levels of arrival velocity from a fast flyby to a parabolic insertion. The results are reported in Fig. 10 for the two extreme cases of a fast flyby and of a parabolic capture. In table 11 mission profile is summarized for the fast flyby and for the capture with three different engines.

The trajectory has been designed and optimized using a direct collocation method based on a finite element transcription implemented in the software DITAN[3]. Since we want an arrival date before 2020 due to science reasons an upper bound on the Jupiter-Pluto transfer has been fixed at 4000 days,. however the brake solution can be improved relaxing this bound constraint as can be read in Tab.11.

Table 11. Alternative Options for EJP mission in 2006

	Fast Transit	Brake(0.04N)	Brake(0.04Nex)	Brake(0.034N)	Brake SPT(0.04N)
C_3	144 km^2/s^2	144 km^2/s^2	144 km^2/s^2	144 km^2/s^2	144 km^2/s^2
Launch Date	19/01/2006	19/01/2006	09/01/2006	11/01/2006	08/01/2006
Jupiter Encounter	23/02/2007	11/03/2007	29/03/2007	28/03/2007	01/04/2007
Pluto arrival	04/10/2014	21/02/2018	05/12/2020	04/12/2020	08/12/2020
Pluto v_∞	15.337 km/s	50 m/s	50 m/s	50 m/s	50 m/s
Mass at departure	600 kg	600 kg	600 kg	600 kg	600 kg
Mass at arrival	565.5 kg	368.64 kg	403.86kg	441.3kg	279.9kg
Total thrust time	6600 hours	46968 hours	39122 hours	49836.7 hours	33338 hours

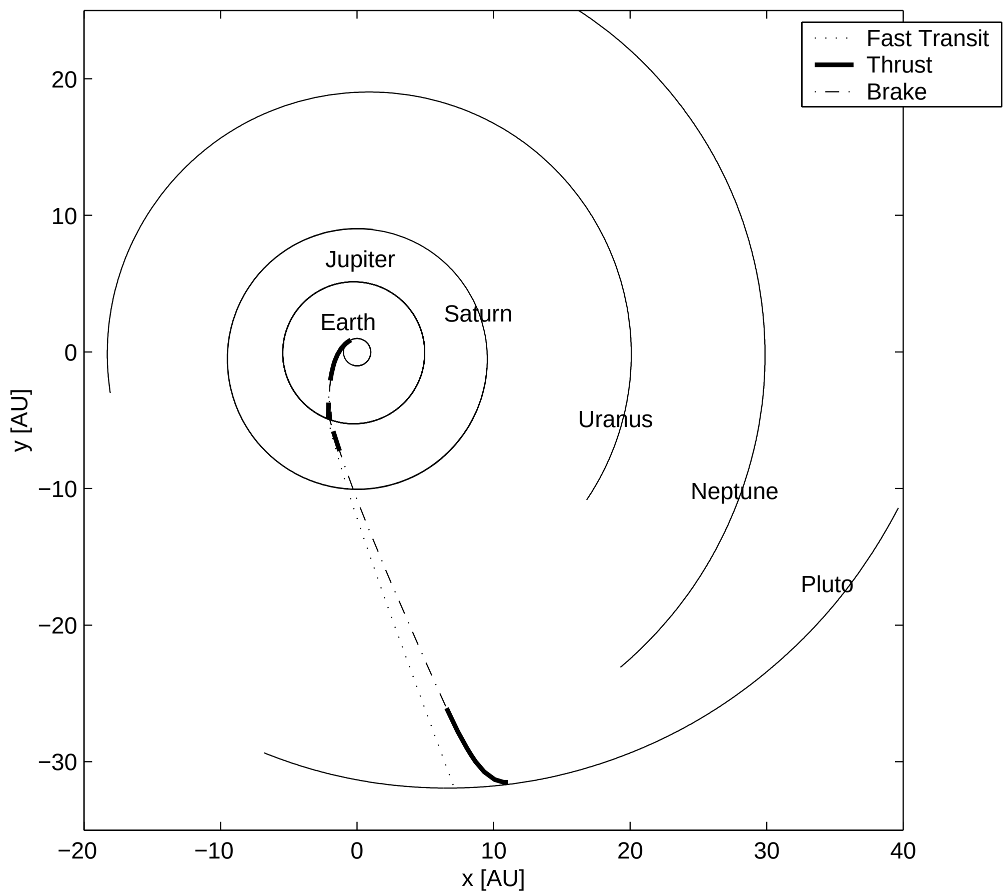

Figure 10.Earth-Jupiter-Pluto Option in 2006 with electric propulsion

Advanced Options

If the departure date is delayed, more advanced options can be thought involving the use of a nuclear reactor and a more advanced electric propulsion system.. In this case a completely different philosophy has been used to design the mission: the arrival date is no more considered an issue but what can be done during the trip to Pluto or after the arrival at Pluto are regarded as driving issues.

The first question is if Jupiter is the most favorable gate to the outer part of the solar system and if so what can be done during the swingby of the giant planet. A second possibility investigated is to look for trajectories that can visit more than one celestial body and then move to Pluto as final destination. The possibility of having a Voyager kind of mission, which could include Pluto and come back, has been investigated.

Optimization Approach

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.

Problem Formulation

The aim is to find an optimal sequence of transfers from the Earth to Pluto 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.

Each arc connecting two subsequent bodies has a deep space Δv maneuver at an unknown point in time and space, each swingby is modeled collapsing the sphere of influence in one single point in space with radius $\mathbf{r}_p$ linking the transfer arcs $\mathbf{r}_i$ before and $\mathbf{r}_o$ 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:

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

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 $\delta=\pi-2\beta$ 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

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

respectively and n_i is the normal to the projection of the incoming vector onto the xy plane.

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. 11). 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:

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

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

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

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

$$\mathbf{y} = [p_0, ..., p_i, ..., p_{N,} t_0, \Delta v_0, t_1, T_1, \omega_1, \tilde{r}_{p1}, ..., t_i, T_i, \omega_i, \tilde{r}_{pi}, ..., t_N, T_N]^T \tag{11}$$

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 iinteresting solutions. It is remarkable that all the best solutions found have a sequence $\mathbf{p}=[3,3,5,9]^T$ 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 summiresed in Tab. 12 along with the trajectory optimised by DITAN with electric propulsion. Figure 12 show the two best intial guesses.

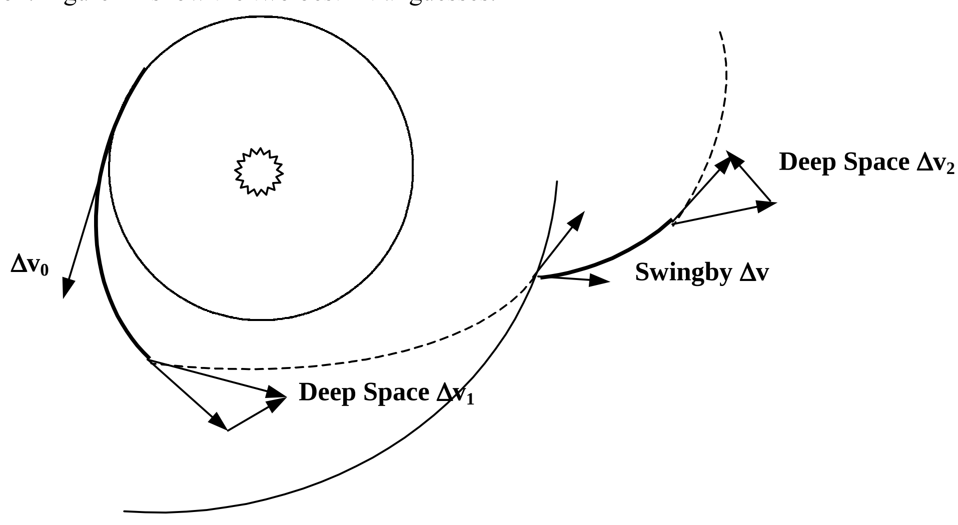

Figure 11. Cartoon of the trajectory model

If the petit tour option is considered and we look at the swingby of Jupiter we can think about having a combined mission to Pluto and to one of the moon of Jupiter (this scenario will be called split option in the following). In this case a 2500 kg composite spacecraft is launched within the same launch window of the electric propulsion optimised petit tour, equipped with a 0.2 N engine with an I_{sp} of 6000s. While entering the sphere of influence of Jupiter a 500 kg probe is delivered and sent to encounter Ganymede while the main spacecraft continues its course to Pluto. The resulting trajectory is represented in Fig.13 for the EJP transfer and in Fig.14 for a closeup of the trajectory insidet the sphere of influence. The effect of Ganymede,as can be seen, is to brake and insert the probe into a high syncronous elliptical orbit around Jupiter periodically encountering Genymede to lower the apocenter, a summary of the result for the split electric propulsion option is reported in Tab. 12.

Table12. Petit Tour Options

Value	Best Solution	Electric Propulsion	Split option
C_3	139.85 km^2/s^2	139.85 km^2/s^2	81 km^2/s^2
Launch Date	22/01/2018	22/01/2018	12/12/2016
Jupiter Encounter	30/03/2019	01/04/2019	02/07/2018
Swingby pericenter	9.9371e5 km	8.999e5 km	1.841e6km
Δv maneuver	1.373e-3 km/s	-	-
Pluto Arrival	24/09/2027	10/09/2027	22/03/2035
Mass at Launch	-	2500 kg	2500kg
Mass at Pluto	-	2498.54 kg	1961.8 kg

Another interesting possibility as mentioned above is to look for multiple encounters on the way to Pluto visiting more than one planet of the outer solar system. Looking for sequences containing either Jupiter,Saturn,Neptune,Pluto or Jupiter, Uranus,Neptune,Pluto we found three intriguing possible transfers. The three trajectories, here referred to as grand tour, are represented in Fig. 15 and their characteristics are summarized Tab. 13. Even in this case the spacecraft is equipped with a 0.2N engine with an I_{sp} of 6000s.

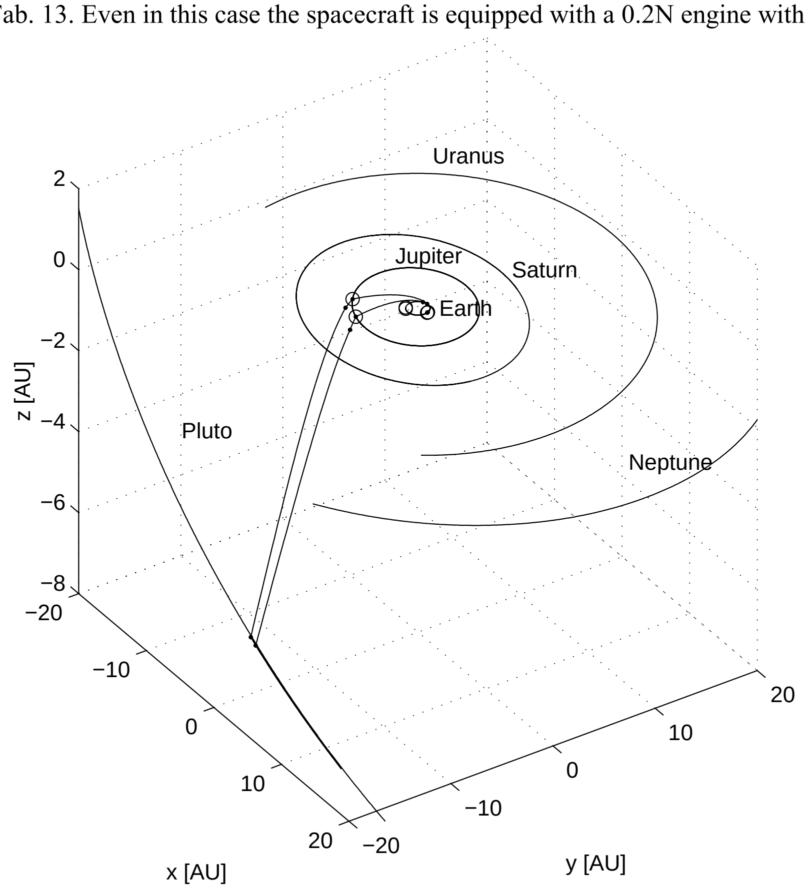

Figure 12. Petit tour optimal first guesses

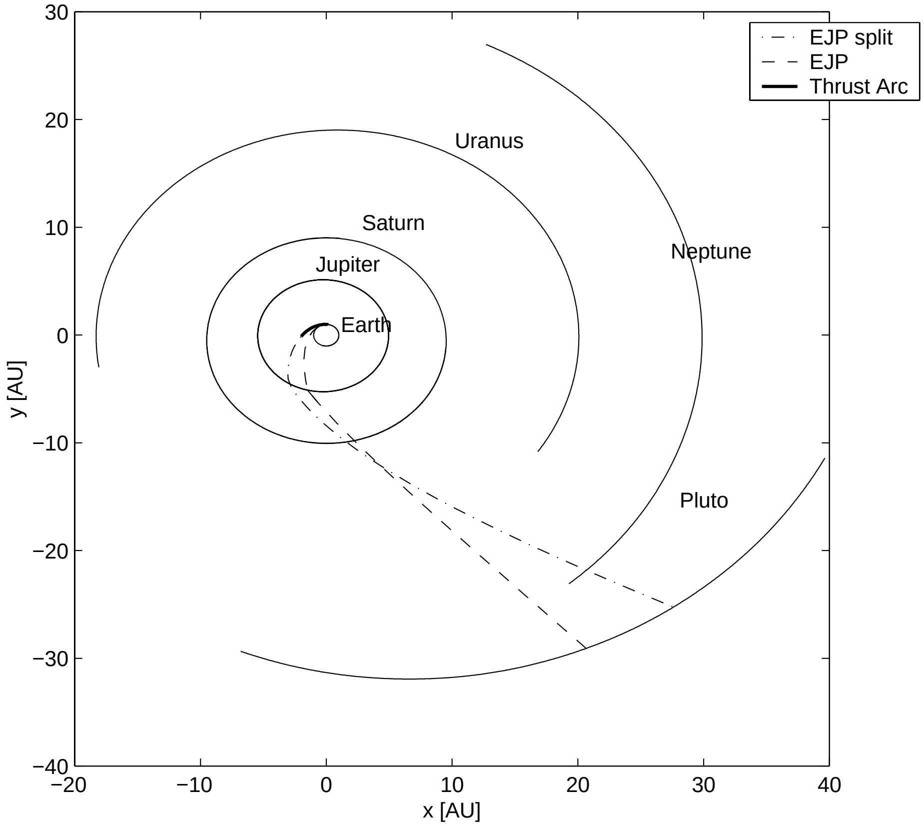

Figure 13. Optimised version of the Petit Tour with electric propulsion

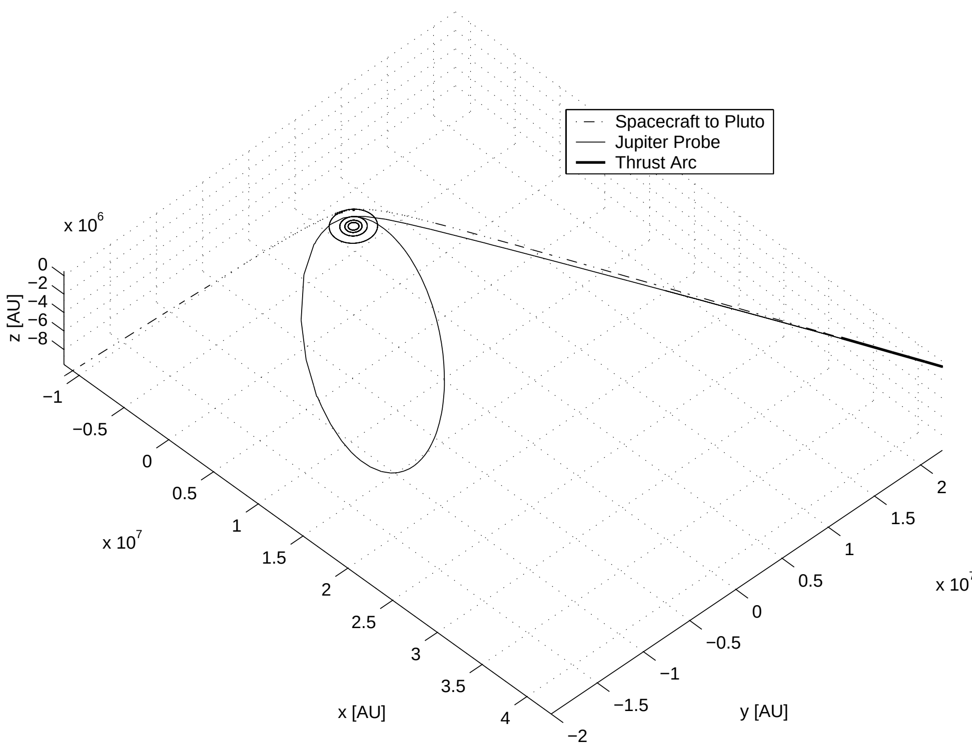

Figure 14. Close-up of Jupitr swing-by with capture of the microprobe

Table 13. Grand Tour Options

Value	Option 1	Option 2	Option 3
C_3	104.49 km^2/s^2	104.49 km^2/s^2	104.49 km^2/s^2
Launch Date	25/09/2013	12/09/2013	04/01/2006
Jupiter Encounter	08/01/2017	13/12/2015	26/05/2008
Swingby altitude	3e5 km	5e5 km	5e5 km
Saturn Encounter	29/10/2020	-	-
Swingby altitude	3e6 km	-	-
Uranus Encounter	-	-	16/11/2013
Swingby altitude	-	-	3500 km
Neptune Encounter	08/02/2027	08/02/2027	09/11/2021
Swingby altitude	3500 km	3500 km	527662 km
Arrival Date	24/12/2044	21/01/2042	24/07/2041
Mass at Launch	2000 kg	2000kg	2000kg
Mass at Pluto	1358.086 kg	1792.65 kg	1794.38 kg
C_3 at Pluto	80.71 km^2/s^2	111.89 km^2/s^2	57 km^2/s^2

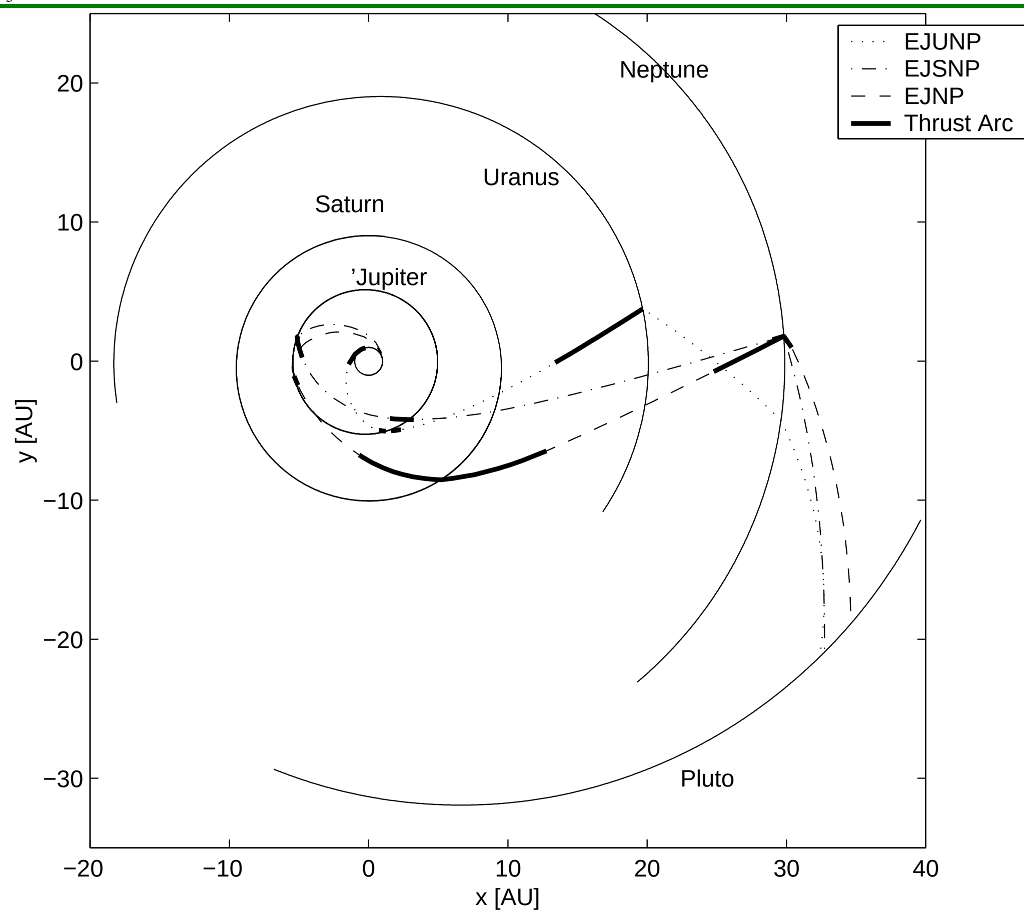

Figure 15.Grand Tour with return opportunity

NEP spacecraft system considerations

In addition to discussing the interest of the trajectory design options, it is deemed necessary to complete the analysis with a preliminary assessment of the technical feasibility of the proposed NEP concept. Provided that a launcher with the capability to deliver the required performance is available, even in a very preliminary analysis would make it immediately clear that the two more critical aspects for the feasibility of the mission are the choice of the power source and propulsion system, in order of priority. The former should deliver enough power for the latter to provide the required thrust for a significant period of time necessary to meet the goals. An adequate choice of components and design of the propulsion system should find the right balance between a subsystem dry mass, fuel mass and system reliability. On the other hand the power system should also provide electrical power for the spacecraft to operate and survive in the harsh conditions of the outer solar system.

These two aspects, power and propulsion, are interrelated and should be considered together when performing the trade-off analysis and selecting the optimal configuration. Based on current technology a mission to Pluto would inherently have a very long duration, ruling out the use of conventional power sources. The distance to the sun furthermore reduces the power choice to nuclear power sources.

Nuclear Power Sources (NPS)

Traditionally, NPS providing electrical energy are divided into RTGs and nuclear reactors. Since 1960s, the US develop and fly RTGs. The Soviet Union and subsequently Russia focused their efforts on space 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.[9] 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[10]. The follow-up soviet reactor version, TOPAZ-2, purchased by the US in the early 90s for technology transfer and testing presented some improvements leading to 1061 kg 135 kW_{th}/5.5 kW_e and thus 193 kg/kW_e. [11] TOPAZ-2 was extensively tested, including nuclear ground tests in the Soviet Union until 1988, but is not yet space-proven. [12]

Plotting the two space proven systems with some of the data available from the studies on small space reactors shows the gap between existing RTG power levels and the power levels of small reactors. (Fig.16) One possibility to fill the gap is to use RTGs in parallel, as has been done for the Cassini spacecraft.[9][13] 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 not realistic. The grey line in Fig. 16 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 a new, probably dynamic (Stirling) conversion system would ameliorate the performances. A similar 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)

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.17) In this category, the most detailed study seems to be the US SP-100 reactor concept, delivering 105 kW_e at a mass of about 4.6 tons. [14][15]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 a nominal lifetime of 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.

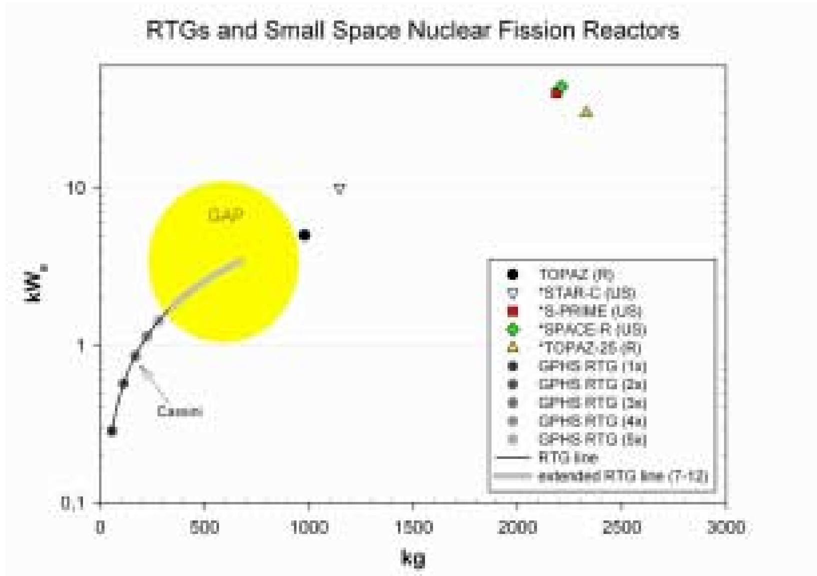

Figure 16: Mass to power plot for RTGs and small space reactors (* mark concepts).

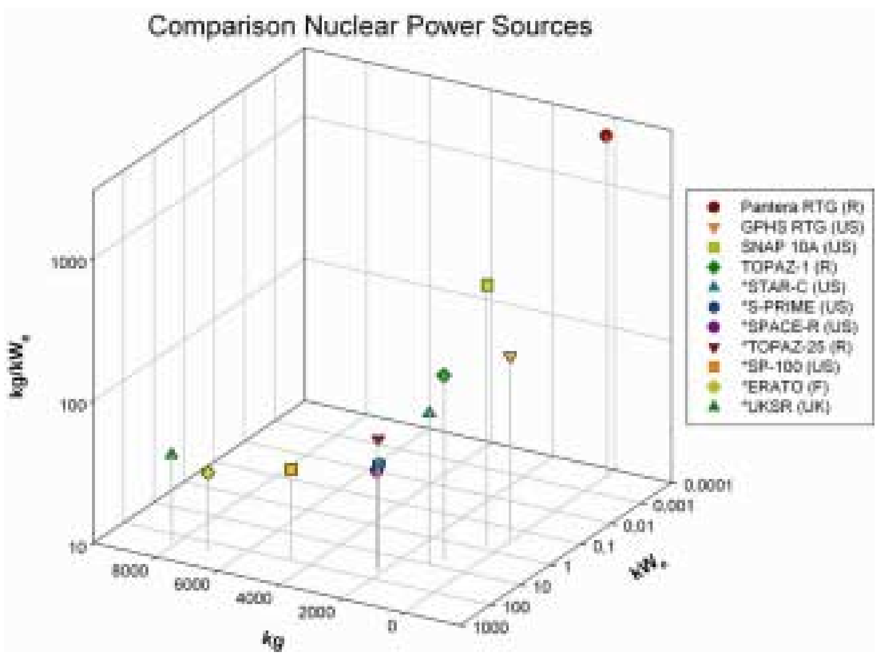

Figure 17: Mass/Power ratio comparison for different NPS.

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.18) The relationship can be expressed as

$$P_{kW_e} \approx 0.0293 \cdot M_{kg} - 25.63\,, \tag{12}$$

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.

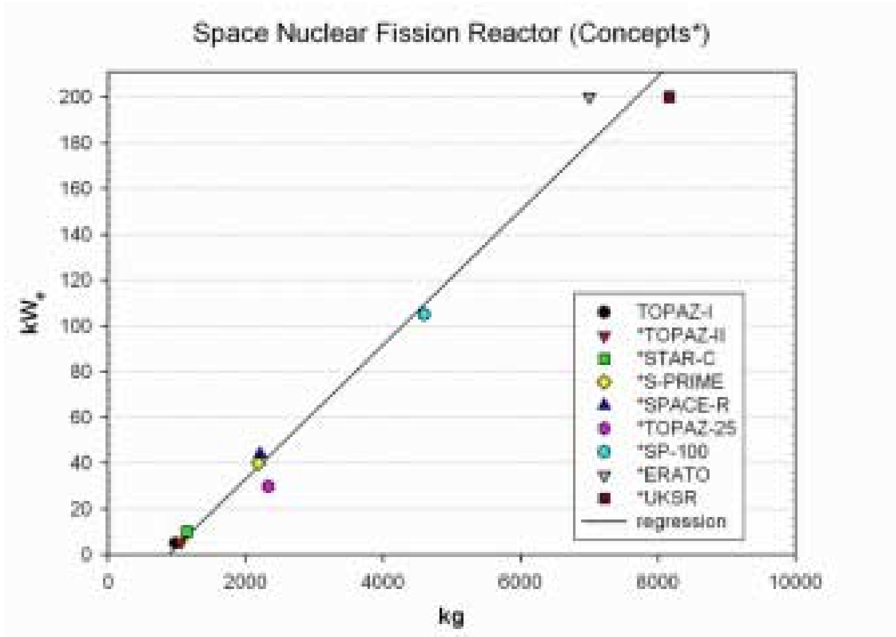

Figure 18: Power to Mass relation of nuclear reactors.

The petit tour and the grand tour option with power requirements of about 8 kW surpass the power levels current RTGs can reasonably supply. Among the reactor concepts proposed until now, only the US STAR-C concepts could theoretically fulfil the power and mass requirements of these missions. The conservative approach, the use of the space-proven TOPAZ reactor, would not satisfy the power demand in its current configuration. The STAR-C concept on the other hand looks quite optimistic, especially given the development needs and the foreseen launch dates. Nevertheless, this concept could be taken as a baseline for the current assessment. Its UC_2/ZrC based reactor core with a mass of 408 kg would provide about 74 kW_{th}, converted in a thermionic conversion system at 13.5% efficiency into about 10 kW_e. The total system mass was foreseen to be only 1148 kg, which leaves about one ton to the structure, propulsion system and instrumentation of the spacecraft.[18] About 2 kW_e would remain for extra-propulsion purposes during the trust arches of the trajectory.

Meeting the Power Requirements for Pluto Missions

Based on the available power systems, the mission has to be either in the hundreds of kg class, requiring about 1 kW_e of power, or in the mass range of several tons with a power necessity of 5 to 10 kW_e. Higher masses would require multiple starts and orbital rendez-vous and docking manoeuvres not considered in this paper. In between these two options, the power delivery via nuclear power sources seems not possible based on available systems. Considering furthermore the total mass limitation of about 2.5 tons at launch, some minimal structure and payload mass of a few hundred kg and the required propellant as outlined in previous sections of this paper, the possible mass and power rages of Pluto spacecraft can be limited to the areas highlighted in Fig.19.

The second important technical constraint, the lifetime necessary for a Pluto mission is currently only fulfilled by RTGs. Nuclear reactors are inherently more complex systems requiring moving parts. Reactor powered Pluto missions require systems that could operate for periods extending to decades without human intervention. Furthermore the operation at multiple power levels as well as several shut-down and re-ignite cycles should be possible.

The electric propulsion options retained in this paper require considerable amount of power for long periods during thrust-arcs. Several options retained for the spacecraft in the Pluto-Charon parabolic insertion scenario would require less power than provided by 4 current GPHS-RTS at end of life for propulsion purposes. These 4 RTGs will deliver after 10 years at Pluto about 1050 W, leaving little power for the rest of the spacecraft operation during the final approach.

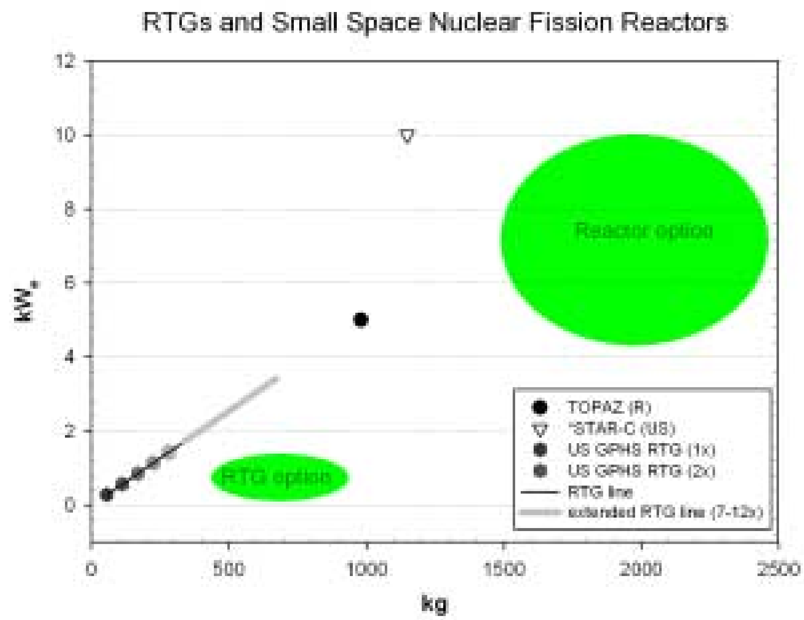

Figure 19: Available power systems and corresponding s/c mass and power limits.

Table 14. Power mass budget

Power source	US GPHS-RTG
Number of units	4
Mass per unit (kg)	56
Electrical power per unit, BOL	0.285
Thermal power per unit	4.264
Total electric power (kW)	1.140
Total thermal power (kW)	17.056
Electric total power, EOL (kW)	1.050
Total power source mass (kg)	224

The petit tour and the grand tour option with power requirements of about 8 kW surpass the power levels current RTGs can reasonably supply. Among the reactor concepts proposed until now, only the US STAR-C concepts could theoretically fulfil the power and mass requirements of these missions. The conservative approach, the use of the space-proven TOPAZ reactor, would not satisfy the power demand in its current configuration. The STAR-C concept on the other hand looks quite optimistic, especially given the development needs and the foreseen launch dates. Nevertheless, this concept could be taken as a baseline for the current assessment. Its UC_2/ZrC based reactor core with a mass of 408 kg would provide about 74 kW$_{th}$, converted in a thermionic conversion system at 13.5% efficiency into about 10 kW$_e$. The total system mass was foreseen to be only 1148 kg, which leaves about one ton to the structure, propulsion system and instrumentation of the spacecraft.[18] About 2 kW$_e$ would remain for extra-propulsion purposes during the trust arches of the trajectory.

In conclusion, the only currently availably power source fulfilling the requirements of outer planet missions are US RTGs. They fulfil the power needs of the less demanding mission scenarios as well as the lifetime constraints. For larger missions, as the petit and the grand tour missions described in this paper, past concepts for space nuclear fission reactors would have to be developed. In its current form, the Russian TOPAZ reactor would not satisfy the mass to power and life time requirement. Theoretically, for the presented petit and grand tour missions, the most appropriate concept described in literature is the US STAR-C reactor concept.

Electric Propulsion system

As a proof of concept the selection of an appropriate thruster model has also been carried out for the Pluto-Charon system parabolic insertion scenario. It takes into account the available power levels on arrival to the target orbit, the dry mass of the system, its reliability and performance. The main characteristics of the system are given in Tab.15 along with the hypothetical ion thruster and SPT engine used for the other break options.

Table 15. Propulsion budget

Thruster	Ion	SPT	QinetiQ T5
Thruster type	Ion gridded	Hall effect	Ion gridded
Thrust level per thruster (mN)	20	20	17
Specific impulse (s)	3100	1700	4100
Specific Power (W/mN)	24	16	30.5
No. Units	4	4	4
Active / redundant units	2/2	2/2	2 / 2
Thruster mass per unit (kg)	2	1.5	1.7
Mass for PPU + propellant feed system (kg)	13.0	10.0	13.6
Total power requirement (kW)	960	720	1.037
Total propulsion ss dry mass in kg (Xe tank excluded)	60	46	61.2

One of the challenging issues that should be further addressed is the thruster lifetime. According to the supplier the predicted theoretical lifetime is about 20000 h, based on engine performance modeling and grid wear-out rates from testing. However and the total engine on-time value for each of the two redundant pairs of thrusters, extensive qualification would still be needed to ensure the feasibility of this operation strategy.

General system considerations

It is estimated that in this scenario the available dry mass of the spacecraft in orbit around Pluto can be as high as 441 kg. This figure might seem enough, but after subtracting the mass of the four RTGs (224 kg) and that of the propulsion subsystem (61.2 + mass of tank!) and considering all other spacecraft subsystems, a payload of about 20 kgs and a system margin of 20% a first estimation gives a mass of about 502 kgs. Some of the peculiarities that have to be considered in the mass budget includes an structure capable of supporting the RTGs, external shunts required to dissipate the excess power when all engines are off, and a demanding communications design to meet both engineering and scientific goals.

As stated above our interest was solely to assess the technical feasibility of a NEP to the outer reaches of the Solar System. Obviously there are several programmatic and cost issues that make such concept very unlikely to be implemented in the proposed configuration. For instance, the requirement to be powered by four RTGs would make the mission extremely costly; as a matter of fact the model of RTGs that has been considered in our analysis might not be available at all in the future. However, theoretically they could make the mission possible even considering only present day technology. As discussed before, developments are expected in the field of power generation with significantly improved mass to electrical power ratios, bridging the existing gap between today's RTGs and space-qualified nuclear reactors. It is therefore likely that these developments would improve the chances of a NEP spacecraft for the exploration of the Outer Solar System becoming a reality.

CONCLUSIONS

In this paper several possible options for a mission to Pluto have been analyzed. Starting from the New Horizon mission alternative options have been investigated resorting to innovative technologies. In particular the possibility of using aero-gravity assist maneuvers and nuclear electric propulsion have been studied aiming either to put a spacecraft in orbit around Pluto or to visit several planets on the way to Pluto. A number of trajectory options using chemical propulsion has been presented. The use of chemical propulsion restricts the transfer time to 20 years for orbit insertion. Using a powered swing-by around Jupiter, a Pluto fly-by mass of 856 kg can be achieved whereas using an un-powered swing-by around

Jupiter, a total mass of 141 kg could be inserted in a 100-km Pluto orbit. The use of aero-gravity assists improves the solution; using an Earth-Venus-Mars-Pluto trajectory with L/D ratio of 4 results in 284 kg mass in 100-km Pluto orbit.

Other trajectory options using electric propulsion appear to be quite interesting. In particular the 2006 window offers the possibility to have a wide range of arrival conditions included a parabolic capture with a small spacecraft equipped with RTGs and a low thrust engine. Always in this launch window a grand tour trajectory is possible visiting several planets before the arrival at Pluto. More challenging possibilities includes other grand tours in the 2013 window anyway these options pose the problem of having a considerable power source which should last for tens of years more than state of the art reactors. Another extremely interesting option is to have a composite mission with a probe delivered in Juppiter orbit. Even in this case an advanced power source is required. In particular filling the gap between RTGs and nuclear reactors would be desirable.

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