

Design of a Pipeline for Satellite-Aided Capture at the Giant Planets of the Solar System

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Abstract—For orbiters aiming at the outer planets of our solar system, most of the ΔV cost is associated with the final insertion at the targeted planet. An efficient way of reducing this cost is using flybys of the moons of the planet to reduce the energy of the orbit at arrival, called satellite-aided capture.

Designing a full transfer from the Earth to an outer planet, including multiple gravity assists and satellite-aided capture raises important issues that must be addressed. One of them is the multiple control parameters that are required to compute such a trajectory. These parameters must be varied over a large array of values to guarantee that all possibilities are covered with enough precision, ensuring that the final trajectory is the best possible. Current approaches on satellite-aided capture mainly focus on designing trajectories inside the sphere of influence of the targeted planet, with no or minimal focus on linking it to the interplanetary trajectory. However, it remains to create a full pipeline to compute a transfer trajectory from the Earth to an outer planet of the solar system using multiple gravity assists and satellite-aided capture.

This paper will focus on creating such a pipeline for orbiters targeting Jupiter, Saturn, Uranus, and Neptune. First, different multiple gravity assists sequences are computed, allowing to make a choice based on transfer time and ΔV cost. This is obtained with a multi-objective dynamic programming exploration, allowing to capture optimal Pareto fronts of ΔV and time of flight in limited computational effort. This transfer sets initial conditions of the satellite-aided capture. Multiple capture sequences are computed around these initial conditions allowing to choose the one minimizing the insertion ΔV . Finally, the last branch of the interplanetary transfer is modified to meet the updated initial conditions of the interplanetary transfer. To compute the satellite-aided capture, the branches between the moons of the sequences are simulated using Lambert arcs. The flybys are approximated as discrete events and are computed to meet the conditions set by the previous and following branches. This pipeline is capable of reproducing scenarios of previous missions to Jupiter and Saturn, ensuring proper functioning of the code. It can also be used to design new trajectories for orbiter at Uranus and Neptune, which have only been visited by Voyager 2 during flybys.

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1. INTRODUCTION

When Galileo performed the first insertion maneuver at Jupiter at the end of the year 1995, it faced the same issue that will face all future orbiters of the outer planets: the importance of the ΔV cost for the orbital insertion. To minimize this cost, the mission design included a flyby of Io at an altitude of 897 km before performing the insertion maneuver: this was the first use of satellite-aided capture in flight [1]. Without the Io flyby, the insertion maneuver would have had a ΔV of 839 m/s, which was instead reduced by 175 m/s to 664 m/s.

Longman [2] was the first to simulate one and two-flyby satellite-aided capture at Jupiter by using energetic considerations. This method allowed to compute energy variations following moon flybys, but no trajectory nor ΔV for orbital insertion was simulated. Nock and Uphoff [3] conducted a survey of the capabilities of single satellite-aided capture at the Earth, Jupiter, Saturn, Uranus and Neptune, but the results suffer from imprecision caused by the limited knowledge of the moons at that time. Two-flyby sequences at Jupiter were later investigated more precisely by Macdonald and McInnes [4], and by Lynam et al. [5] under an ephemeris-free model. Triple and quadruple sequences were also studied by Lynam et al. [6], and by Lynam [7] [8] [9]. In the case of sequences with more than two flybys, solving Lambert's problem was deemed too complex and instead Laplace resonances of Jupiter's moons and phase angle considerations are used. Although triple and quadruple sequences are more efficient and quite elegant solutions, their rarity and the constraints they causes on navigation make them impractical in real scenarios.

Bellome et al. [10] created a pipeline to design MGA-DSM (Multiple Gravity Assists - Deep Space Maneuvers) transfers between any two planets of the Solar system. This is achieved by coupling Tisserand graph exploration and multi-objective dynamic programming to identify the possible MGA sequences and optimize the trajectory. Direct transfers from the Earth to Jupiter constrained by two-flyby satellite-aided capture were studied by Lynam and Longuski [11], and by Lynam [12]. Although the direct trajectory can be linked to an MGA transfer, the usable trajectories are limited to the ones with the Earth in the penultimate position. Scott et al. [13]

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developed a methodology to design missions to Jupiter with an MGA transfer and single or double satellite-aided capture. Phase-free satellite-aided capture solutions are found, before searching for the epochs meeting the phasing requirements. The capture is then connected to the desired transfer.

A clear gap in the literature can be identified, as satellite-aided capture is mainly investigated for Jupiter and rarely connected to a realistic interplanetary transfer. Planets like Saturn, Uranus and Neptune are often excluded from the discussions, and when the transfer is included, it is connected to a predetermined satellite-aided capture solution rather than finding a solution that matches the desired transfer. Hence, it remains to create a tool capable of designing interplanetary transfer and matching satellite-aided capture to every planet where this type of capture is applicable.

In this paper, we combine the different methodologies used for each step of an interplanetary mission to make them work together in a single pipeline. By applying the methodologies to Saturn, Uranus and Neptune in addition to Jupiter, this pipeline is capable of simulating a MGA-DSM transfer and performing satellite-aided capture for any of the planets of the outer Solar system.

2. MULTI-OBJECTIVE DYNAMIC PROGRAMMING FOR INTERPLANETARY TRAJECTORIES

The first step in designing the trajectory of an orbiter targeting another planet is to conceive the interplanetary transfer. This is achieved by solving a multi-objective optimisation problem via dynamic programming, following the work of Bellome et al. on the subject [10].

Problem formulation

Designing MGA trajectories is a global optimization problem, as multiple local optimums can exist for a given planetary sequence. This is also a multi-objective optimization problem, as most typical space missions aim to minimize both the transfer time and the ΔV . As these parameters are usually conflicting (the faster we are, the sooner we get there), the output of this optimization problem will be a Pareto front, a surface that gives the minimum transfer time for a given amount of ΔV , and vice versa. This problem aims to minimize a function $F(X,y)$ depending on integer X and continuous-varying variables y . The function F encodes the objectives of the problem, in our case the global transfer time and ΔV . The structure of the problem can be summarized as following:

$$\begin{aligned} \text{Minimize : } & F(X, y) = f_1(X, y), \dots, f_n(X, y) \\ \text{Subject to : } & g_i(X, y) \leq 0 \\ & X_{lb} \leq X \leq X_{ub} \\ & y_{lb} \leq y \leq y_{ub} \end{aligned} \quad (1)$$

with $g_i(X,y)$ being the constraints of the problem, (X_{lb}, X_{ub}) and (y_{lb}, y_{ub}) are the lower and upper bounds for X and y respectively.

In the case of MGA trajectory optimization, X will store the sequence of visited planets, while y will store the continuous variables like the visiting epoch, flybys parameters, and DSMs. $g_i(X,y)$ will represent mission specific objectives other than time of flight and ΔV [10].

The model used to simulate the trajectories is the MGA-DSM model [14], under which a single DSM can occur between two consecutive gravity assists. The transfer between planets and DSMs is modeled under the two-body dynamics, using Lambert arcs.

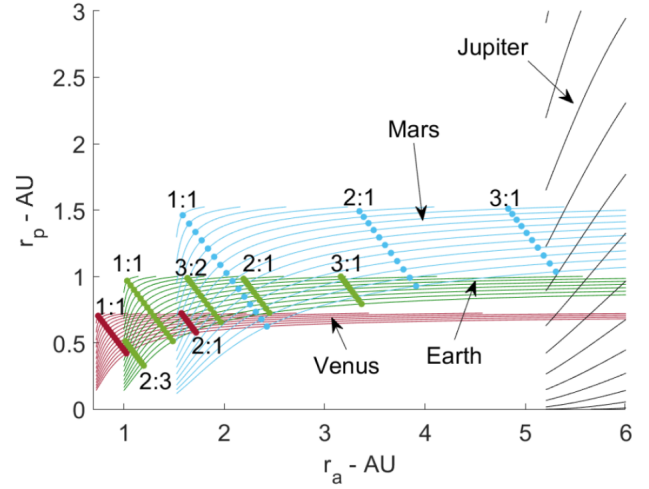


Figure 1: Tisserand graph for orbits intersecting Venus, Earth, Mars and Jupiter [10].

Generation of MGA sequences via Tisserand graph exploration

The first step of designing MGA trajectories is to assess the feasibility of given sequences using Tisserand graphs as a low fidelity model.

Tisserand graphs are tools built using energetic considerations to connect orbits perturbed by a planetary flyby. For a flyby of a given planet, Tisserand's parameter defined in Eq. (2) remain constant. By varying the semi-major axis a , eccentricity e and inclination i of the orbit of a spacecraft while keeping T_p constant, it is possible to construct families of orbits that can be linked via a planetary flyby [15].

$$T_p = \frac{a_p}{a} + 2\sqrt{\frac{a(1-e^2)}{a_p}} \cos(i) \quad (2)$$

Figure 1 shows an example of a Tisserand graph for multiple infinity velocities V_∞ at Venus, Earth, Mars and Jupiter. Each contour represent flybys about their planets, with the corresponding V_∞ . An intersection between two contours means that an orbit connecting the two planets exist, making a transfer possible. By connecting multiple intersecting contours, it is then possible to design an MGA sequence between two planets. On the other hand, Tisserand graphs provide no information on the phasing required for a sequence since it is based on energetic considerations.

Therefore, the exploration of a Tisserand graph can be summarized as simply evaluating all the planetary sequences and excess velocities that enable the spacecraft to reach the desired orbit by following intersecting contours [10]. This is achieved by using a breadth first search strategy [16], following the implementation described by Afssa et al. [17]. Although the number of possible sequences is basically endless, the search space can be limited by the fact that any object in the Solar system can be reach using four to five planetary flybys. This makes the number of combinations to explore manageable by complete enumeration strategies like breadth first. Finally, the application of Tisserand graphs is not limited to interplanetary transfer as in this paper, but they can also be used for designing moon tours at Jupiter or Saturn for example.

Trajectory determination

The second step in the design of MGA transfers is exploring the possibilities in term of planet phasing and DSM ΔV for sequences identified via Tisserand graph exploration.

The trajectory determination is in two steps. First, trajectories are simulated using the defect model. In this model, it is assumed that a DSM is performed right after a planetary flyby. It enables to simulate the leg between two planets as a single Lambert arc. In that case, the cost of the leg is the amplitude ΔV of the defect, corresponding to the difference of relative velocity between before and after the flyby. Defects are computed as:

$$\Delta V = \left| \vec{V}_{\infty}^+ - \vec{V}_{\infty}^- \right| \quad (3)$$

in the case where $\delta \leq \delta_{max}$, or

$$\Delta V = \sqrt{\left| \vec{V}_{\infty}^+ \right|^2 + \left| \vec{V}_{\infty}^- \right|^2 - 2 \left| \vec{V}_{\infty}^+ \right| \left| \vec{V}_{\infty}^- \right| \cos(\delta_{max} - \delta)} \quad (4)$$

otherwise. In these equations, $\vec{V}_{\infty}^-$ and $\vec{V}_{\infty}^+$ are the velocities of the spacecraft relative to the planet before and after the flyby, δ is the deflection angle between $\vec{V}_{\infty}^-$ and $\vec{V}_{\infty}^+$ measured counterclockwise, and δ_{max} is the maximum deflection angle that can be achieved for a given $\vec{V}_{\infty}^-$. It is computed as follows:

$$\delta_{max} = 2 \arcsin \left(\left(1 + \frac{r_{p,min} \left| \vec{V}_{\infty}^- \right|^2}{\mu_{pl}} \right)^{-1} \right) \quad (5)$$

where $r_{p,min}$ is the minimum periapsis that can be achieved for the flyby, and μ_{pl} is the gravitational parameter of the planet. Using the defect model allows each leg to be dependant on the previous leg only through the vector $\vec{V}_{\infty}^-$. This property is exploited through the application of the decision variables on grids of departing dates and transfer times between two planets. It allows to transcribe the problem into a discrete optimization problem where the goal is to find the visiting epochs for each planet that minimize functions like the global ΔV and transfer time.

Once the full path is computed, defects are changed into DSM to further optimize the solution. On the leg preceding a defect, a DSM is inserted, and its position is solved through an optimization problem. The aim of this new optimization problem is to determine the epoch of the DSM that minimize its ΔV while ensuring that the spacecraft still encounters the targeted planet at the correct time with the appropriate $\vec{V}_{\infty}^+$ to remove the defect (hence having $\Delta V = 0$ in Eq. (3)) [10].

3. SATELLITE-AIDED CAPTURE VIA LAMBERT ARC SOLVING

Once the desired interplanetary transfer is determined, the initial conditions at the sphere of influence of the planet are known. It is then possible to identify the possible satellite-aided capture sequences meeting these conditions.

Models

In order to simplify the modeling of such a complex problem, multiple assumptions are made. Firstly, all planetary systems used for satellite-aided capture are assumed to be circular

and co-planar. This assumption is made possible by the low inclinations and eccentricities of the moons that can be used for satellite-aided capture. The choice of these moons is based on their mass, as it must be high enough to change significantly the trajectory of a spacecraft. The moon that will be used in the pipeline are the Galilean moons for Jupiter (Callisto, Ganymede, Europa and Io); Titan, Rhea, Dione, Tethys and Enceladus for Saturn; Oberon, Titania, Umbriel, Ariel and Miranda for Uranus; and Triton for Neptune. In the case of Triton, its orbit is retrograde with an inclination of 157° , which is far from an equatorial orbit. On the other hand, since Triton is the only moon usable in satellite-aided capture, it will not be an issue for the co-planar model.

Concerning the trajectories, they are simulated using the linked-conics approximation, in which flybys are approximated as discrete events only changing the orientation of the velocity vector at infinity of the secondary body, while reducing its sphere of influence to a point. This approximation is valid in our case, as the semi-major axis of the orbit about the primary body is typically multiple order of magnitude greater than the radius of the sphere of influence of the secondary body. Hence, in this model, the time and position relative to the primary body remain the same during a flyby. This model has one main advantage over the patched-conics approximation: since the sphere of influence of the body is approximated to a point, the spacecraft only has to match his position with that of the body. Then, when simulating the trajectory inside the sphere of influence, the entering point can be set as wanted.

By using both the circular co-planar and the linked-conics models, the orbit of the spacecraft can be contained in the same plane as the moons' orbits. This enables the inclination and longitude of the ascending node to be kept at 0° , meaning that only the argument of periapsis needs to be varied in order for the spacecraft's orbit to encounter the moon at the chosen date.

Single flyby capture

Simulating a single flyby satellite-aided capture is the simplest case possible, as no phasing consideration has to be taken into account in the linked-conics approximation.

In this model, it is assumed that the incoming orbit can be rotated as wanted, in order to meet the targeted moon at the flyby date. From the incoming velocity V_{∞} , the gravitational parameter of the planet μ and the radius of periapsis r_p , the semi major axis a , eccentricity e and true anomaly θ can be computed using the classical equations of hyperbolic orbits (Eq. (6)).

$$\begin{aligned} a &= -\frac{\mu}{V_{\infty}^2} \\ e &= 1 - \frac{r_p}{a} \\ \theta &= -\arccos\left(-\frac{1}{e}\right) \end{aligned} \quad (6)$$

As mentioned in the previous subsection, the inclination and the longitude of the ascending node are kept as 0° , while the argument of periapsis is varied such as the spacecraft encounters the moon at the flyby date using Eq. (7).

$$\omega_{Sc} + \theta_{Sc}(r_M) = \theta_M(t_{fb}) \quad (7)$$

with ω_{Sc} the argument of periapsis of the spacecraft, $\theta_{Sc}(r_M)$ the true anomaly of the spacecraft when crossing the moon's orbit, and $\theta_M(t_{fb})$ the true anomaly of the moon at the flyby date. As the moon's orbit is under the circular co-planar model, its longitude of the ascending node and argument of periapsis are also kept as 0° , explaining why these terms are

not involved in Eq. (7).

Since the initial orbital parameters are known at the sphere of influence of the planet, they can be propagated until the encounter with the moon. Once at the moon, the flyby is performed using the equations described in the appendix. For maximum efficiency, the flyby should be performed as low as possible and the angle of the flyby plane should be $\frac{\pi}{2}$ in order to reduce the orbital energy of the spacecraft and stay in the initial plane of the planetocentric trajectory. As the trajectory is described under the linked-conics model, there is no restriction to the minimum altitude of the flyby, apart from constraints coming from the precision on the orbit or guidance requirements. In order to reflect real cases, the flyby can be simulated at a higher altitude than the minimum. Once the flyby is performed and post-flyby orbital elements are known, the orbit can be propagated until its periapsis and the capture burn computed using Eq. (8).

$$\Delta V = |V_T - V_p| \quad (8)$$

with V_T the velocity at periapsis of the targeted orbit, and V_p the velocity at periapsis of the post-flyby trajectory. Depending on the parameter used for defining the final orbit in addition to the radius of periapsis, V_T can take multiple forms derived from the Vis-viva equation (Eq. (9)) and the equations of the specific energy (Eq. (10)) and orbital period (Eq. (11)) [18].

$$V = \sqrt{\mu \left(\frac{2}{r} - \frac{1}{a} \right)} \quad (9)$$

$$\varepsilon = -\frac{\mu}{2a} \quad (10)$$

$$T = 2\pi \sqrt{\frac{a^3}{\mu}} \quad (11)$$

Double flyby capture

Multiple flyby satellite-aided capture encounters phasing issues when it comes to designing the trajectory. Each branch between two flybys must be connected to the previous and following ones by feasible flybys, meaning that they must meet the constraints set by the environment on their parameters. This is achieved here by Lambert arc solving, as the time period typically surveyed while using this pipeline is relatively short (less than a month).

The trajectory determination for double flyby capture starts the same way as for single flyby capture, with the initial orbit being rotated around the equatorial plane to encounter the first moon of the sequence when crossing its orbit using Eq. (6). Once the first moon is reached, Lambert's problem is solved between its position and the position of the second moon for multiple transfer times. Each transfer time will give different flyby parameters, and since it is not possible to know in advance the exact transfer time needed, multiple values have to be tested with enough precision. Lambert's problem is solved following Izzo's methodology for zero revolutions [19], as the first flyby is very unlikely to make the spacecraft leave its hyperbolic trajectory to go on an elliptic orbit. As a flyby doesn't change the amplitude of the velocity at infinity V_∞ , only the transfer times that keep the difference of velocity before and after a flyby below a given precision are saved (Eq. (12)). A high precision on the velocity difference means that a greater precision on the transfer times tested will be needed, increasing the computation time as more values need to be tested. Typically, a value of 1 m/s will give satisfactory results.

$$|V_\infty^+ - V_\infty^-| < \delta V_{min} \quad (12)$$

The second moon encounter must also occur before the periapsis of the planetocentric trajectory, meaning that the true anomaly of the spacecraft at the encounter should be negative. Then, for all the saved Lambert arcs, the parameters of the flyby are computed by inverting the equation in the appendix. A new filter is applied, where only flybys that reduce the orbital energy (Eq. (13)) and have an acceptable altitude are kept. As for one flyby, the lower bound is constrained by the precision that can be achieved on the flyby altitude while the upper bound is constrained by the size of the moon's sphere of influence.

$$\gamma = \frac{\pi}{2} \quad (13)$$

Finally, from all the possible flybys that are left, the chosen solution is the one with the lowest altitude, ensuring the maximum efficiency in term of ΔV . Once the appropriate trajectory after the first flyby and the corresponding flyby parameters are determined, the orbital parameters at the second moon can be identified. From that point, the second flyby is simulated similarly to a one flyby sequence, meaning that the flyby is performed as low as possible, the trajectory is propagated until its periapsis and the capture computed using Eq. (8) through (11).

Two-flyby sequences being available for a short period of time, it is possible that the selected interplanetary transfer has an arrival date just a few days away from a double flyby capture. In that case, the previous methodology can be reproduced for multiple arrival dates around the initial value, in case a more favourable sequence is available. If it is the case, the interplanetary trajectory will have to be updated to encounter the first moon at the new date. These types of considerations are not relevant in this pipeline for one flyby sequences, as the incoming trajectory is assumed to rotate as wanted to meet the first moon.

Algorithm 1 summarize the steps used to determine satellite-aided capture trajectories.

Algorithm 1 Satellite-aided capture algorithm

Require: $t_0, V_\infty, \mu T r_p$ and sequence
Determine initial orbital parameters (a, e, θ)
 $\omega_{Sc} \leftarrow \theta_M(t_{fb}) - \theta_{Sc}(r_M)$
Propagate to seq(1)
if length(seq) = 2 **then**
Solve Lambert's problem for multiple transfer times
 $\delta V \leftarrow |V_\infty^+ - V_\infty^-|$
if $\delta V < \delta V_{min}$ **then**
Determine the flyby parameters (r_p, γ)
if $\gamma = -\frac{\pi}{2}$ or $\theta_{Sc}(2) > 0$ **then**
Don't save this solution
else
Save the flyby parameters
end if
end if
if No solution found **then**
Switch to one flyby
end if
 $r_p(1) \leftarrow \min(r_p)$
Propagate to seq(2)
Perform flyby at desired altitude
Propagate to periapsis
 $\Delta V \leftarrow |V_T - V_p|$
else if length(seq) = 1 **then**
Perform flyby at desired altitude
Propagate to periapsis
 $\Delta V \leftarrow |V_T - V_p|$
end if

Table 1: Optimization scenario for the EVEEJ sequence with a launch in 1997.

Design variable	Values and bounds
Sequence	EVEEJ (known)
Resonances	2:1 on E-E branch (known)
Characteristic energy	$C_3 < 17 \text{ km}^2/\text{s}^2$
Launch window	Year 1989
Total DSM	$\Delta V_{DSM} < 100 \text{ m/s}$
Number of revolutions on one leg	$N_{rev} < 1$

4. RESULTS

General results and applications of this pipeline are presented here. Test cases replicating a real mission and a novel scenario are also included in this section.

The methodology used for a single-flyby satellite-aided capture allows to always perform a flyby as low as possible, hence the most efficient, as the linked-conics approximation allows to choose the entering point of the sphere of influence of the moon. This may not be representative of real case scenarios, so the pipeline can be modified to just have a flyby at the desired altitude. For two flybys, the altitude of the first flyby is constrained by the position of the second moon. It means that, if the mass of the first moon is greater than the mass of the second, it is sometimes more interesting to perform a single, low flyby than force two flybys with the first one being at an high altitude hence less effective. It is for instance the case of Saturn, where the mass of Titan is multiple order of magnitude greater than the mass of the second greatest moon Rhea. For example, for the same final orbit², a single flyby of Titan at 25 km will reduce the ΔV by 254.3 m/s, when a flyby of Titan at 500 km followed by a flyby of Rhea at 25 km will only reduce the ΔV by 236.6 m/s.

Multi-objective optimization of Earth-Jupiter transfer and satellite-aided capture: the case of Galileo

The implementation of the pipeline is tested against a well-documented case, the Galileo mission. This mission established multiple firsts in the history of spaceflight, being the first spacecraft to orbit one of the giant planets of the Solar system, and performing the first satellite-aided capture in the process.

Interplanetary transfer—Galileo used a Earth-Venus-Earth-Jupiter sequence (EVEEJ) for its transfer from the Earth to Jupiter, with a launch in October 1989 and a 2:1 resonance on the Earth-Earth branch. The maximum C_3 that could be achieved by the Inertial Upper Stage (IUS) used by the probe was $17 \text{ km}^2/\text{s}^2$, with about 100 m/s of ΔV available for DSMs [20]. These constraints on the optimization scenario are summarized in Table 1.

The real Galileo mission included two asteroid flybys that are not simulated here. As a consequence, the optimization process will not return the trajectory of the actual mission, but solutions that are more efficient in term of ΔV . In order to perform the comparison, the trajectory will be chosen to be as close as possible to the actual mission. For this reason, the chosen departure date is chosen to be the same, on the 18th of October 1989.

²Periapsis of $5 R_S$ and orbital period of 200 days.

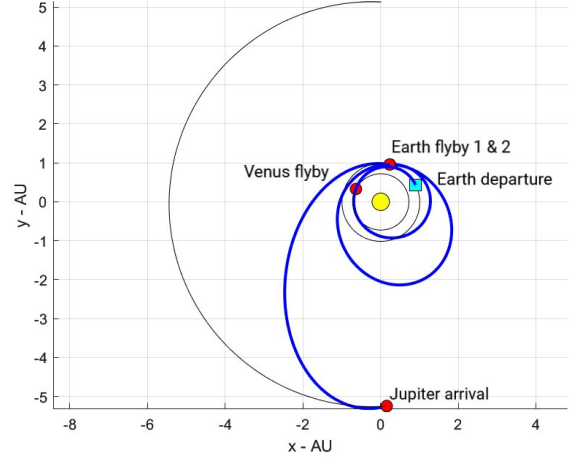


Figure 2: Transfer to Jupiter for a launch on the 18 Oct. 1989.

Table 2: Comparison between the simulated trajectory and the actual mission

Event	Simulation	Galileo mission
Earth departure	18 Oct. 1989	18 Oct. 1989
C_3 (km^2/s^2)	15.6	-
Venus flyby	9 Feb. 1990	10 Feb. 1990
DSM 1 (m/s)	58	35
Earth flyby	9 Dec. 1990	8 Dec. 1990
DSM 2 (m/s)	-	21
Earth flyby	9 Dec. 1992	8 Dec. 1992
DSM 3 (m/s)	26	38
Jupiter arrival	22 Feb. 1996	7 Dec. 1995
V_∞ (km/s)	5.76	6.26 ⁽¹⁾
Total DSM (m/s)	84	94

⁽¹⁾ computed from capture burn

The trajectory for the chosen solution is shown in Figure 2. For this solution, the transfer time is 6.35 years. The launch is on the 18th of October 1989 with a C_3 of $15.6 \text{ km}^2/\text{s}^2$, meeting the requirement of not exceeding $17 \text{ km}^2/\text{s}^2$ set by the IUS. The transfer to Venus is 114 days long, with the flyby occurring on the 9th of February 1990 with an incoming velocity at Venus of 6.359 km/s. During this transfer, multiple trajectory correction maneuvers were needed, but they cannot be simulated in a deterministic way. After the Venus flyby, a DSM of 58 m/s is planned in order to target the Earth, while the real Galileo mission only required 35 m/s for this DSM [21]. The transfer to the Earth is 303 days long, with the first Earth flyby occurring on the 9th of December 1990. During this flyby, the velocity at infinity is 8.975 km/s while it was only 3.950 km/s at Earth departure. This flyby puts Galileo on a 2:1 resonant orbit with the Earth, which is encountered two years later on the 9th of December 1992. On this resonant orbit, the real mission performed a flyby of the asteroid Gaspra. This caused an additional DSM of 21 m/s to target the Earth following this flyby. The second Earth flyby gives enough energy to the spacecraft to encounter Jupiter. This is achieved after a 1170-day transfer and a 26 m/s DSM,

on the 22nd of February 1996, with a velocity at infinity of 5.76 km/s. The real Galileo mission also performed a flyby of the asteroid Ida, causing the Jupiter encounter to occur earlier but with a higher velocity at infinity. For this transfer, the constraint on the ΔV for the DSM is respected, with a total cost of 84 m/s, and the number of revolutions per leg is kept below one. The differences between the trajectory simulated here and the actual mission can be explained by the asteroid flybys that are performed during the transfer, especially Ida. By performing the Ida flyby, an encounter is forced, leading to a solution ignored by the Pareto front because more expensive. Table 2 summarizes the chosen transfer and compares it with the actual mission.

Satellite-aided capture at Jupiter—Following the interplanetary transfer of this test, Galileo reaches Jupiter on the 22nd of February 1996 with a velocity at infinity of 5.76 km/s. The planned satellite-aided capture sequence was a single flyby of Io at an altitude of 1000 km, which was in the end performed at 897 km. The targeted orbit had a period of 230 days and a perijove of $4 R_J$, meaning that the capture will be performed $3 R_J$ above the surface. As this test aims to assess the ability of the pipeline to replicate real case scenarios, the flyby will be performed at the altitude of 897 km, the same as the real mission.

As only one flyby is performed for the capture of Galileo, there is no need to solve Lambert’s problem. Therefore, the computation time is almost instantaneous, being less than a second. The capture trajectory is shown in Figure 3. With the parameters determined from the interplanetary transfer, a Io flyby at 897 km lowers the ΔV for insertion by 161 m/s, from 741 m/s to 580 m/s. If the flyby was performed at 1000 km like it was planned, the performances would have been lower, with a gain of 155 m/s causing an insertion burn of 586 m/s. The difference between the 664 m/s of the real mission and the trajectory simulated here can mainly be explained by the lower V_∞ of the test. The circular co-planar model will also impact by a few meters per second the insertion ΔV , as it may change the distance of the flyby from the planet.

Multi-objective optimization of Earth-Uranus transfer and satellite-aided capture: Designing a novel scenario

After ensuring the proper functioning of the pipeline with a comparison against a real mission, a test with a new scenario is performed. The aim of this test is to assess the ability of the pipeline to design a mission without initial knowledge of the solution. For this purpose, a trajectory to Uranus will be designed, targeting a two-flyby satellite-aided capture. Missions to Uranus have gained an interest in the scientific community following the 2022 issue of the Planetary Science Decadal Survey, calling for a flagship mission to Uranus that could be launched as early as 2029 [22] [23].

Interplanetary transfer—One of the possible sequences for a launch towards Uranus is Earth-Venus-Earth-Earth-Jupiter-Uranus (EVEEJU), with a 2:1 resonance on the Earth-Earth branch [23]. The opportunities for this sequence will be surveyed over the year 2029, with a C_3 below $20 \text{ km}^2/\text{s}^2$, a maximum total ΔV for DSMs of 500 m/s and a maximum of two revolutions per leg. The constraints on the optimization scenario are summarized in Table 3.

The Pareto front for this scenario is given in Figure 4, with transfer times ranging from 10.8 to 13.3 years. The ΔV here is computed as the sum of the velocities at infinity of the departure and arrival planets, and of the DSMs performed. For this test, the chosen solution will be the one with the

Table 3: Optimization scenario for the EVEEJU sequence with a launch in 2029.

Design variable	Values and bounds
Sequence	EVEEJU (known)
Resonances	2:1 on E-E branch (known)
Characteristic energy	$C_3 < 20 \text{ km}^2/\text{s}^2$
Launch window	Year 2029
Total DSM	$\Delta V_{DSM} < 500 \text{ m/s}$
Number of revolutions on one leg	$N_{rev} < 2$

lowest ΔV according to this definition, as important interplanetary mission like the one in this scenario usually have more constraints on the propellant mass than the transfer time. The transfer will also be chosen to target a two-flyby satellite-aided capture, using flybys of Oberon and Titania. As these two moons are the most massive ones at Uranus, this sequence will provide the maximum ΔV gains. The opportunities will be surveyed for 30 days before the arrival date of the longest transfer, on the 30th of January 2043. If no solution using these two flybys can be found, a single Titania flyby will be performed on this date.

After surveying the Oberon-Titania opportunities for one month before the 30th of January 2043, these moons are properly aligned on the 18th of January. Hence, following the Pareto front, the transfer arriving at this date will be the one ensuring the minimum ΔV while targeting a Oberon-Titania satellite-aided capture. The chosen trajectory is displayed in Figure 5. For this trajectory, the launch occurs on the 22nd of October 2029, with a C_3 of $13.000 \text{ km}^2/\text{s}^2$. The transfer to Venus is 106 days long, being reached on the 5th of February 2030 with a velocity at infinity of 5.377 km/s. The flyby puts the spacecraft on a trajectory back to the Earth, with a DSM of 250 m/s being performed on that branch. The first Earth flyby is performed on the 22nd of June 2032 after two revolutions around the Sun. This flyby puts the spacecraft on a 2:1 resonant orbit with the Earth, with the second flyby performed on the same date in 2034. Following this flyby, a DSM of 47 m/s is performed en route to Jupiter. The Jupiter flyby is performed 644 days later on 27th of March 2036, giving the spacecraft enough energy to reach Uranus. Uranus is reached on the 30th of January 2043 after a final DSM of 20 m/s. At Uranus, the velocity at infinity is 9.036 km/s, and the total time of flight is 13.2 years. The transfer is summarized in Table 4.

Satellite-aided capture at Uranus—The spacecraft in this scenario reaches Uranus on the 18th of January 2043 with a velocity at infinity of 9.036 km/s, allowing a Oberon-Titania satellite-aided capture to be performed. The capture will be targeted at $1.1 R_U$, and the final orbit will have a period of 200 days. In a moon tour proposal as part of a mission concept for a Uranus orbiter by Simon et al. [23], the minimum flyby performed has an altitude of 25 km. Hence, this is the minimum that will be used for determining possible sequences.

Performing the one-month survey took approximately half an hour, while refining the solution on the 18th with more precision took a few minutes. The majority of the computation time here comes from solving the Lambert arcs, making necessary a trade-off between precision and computation

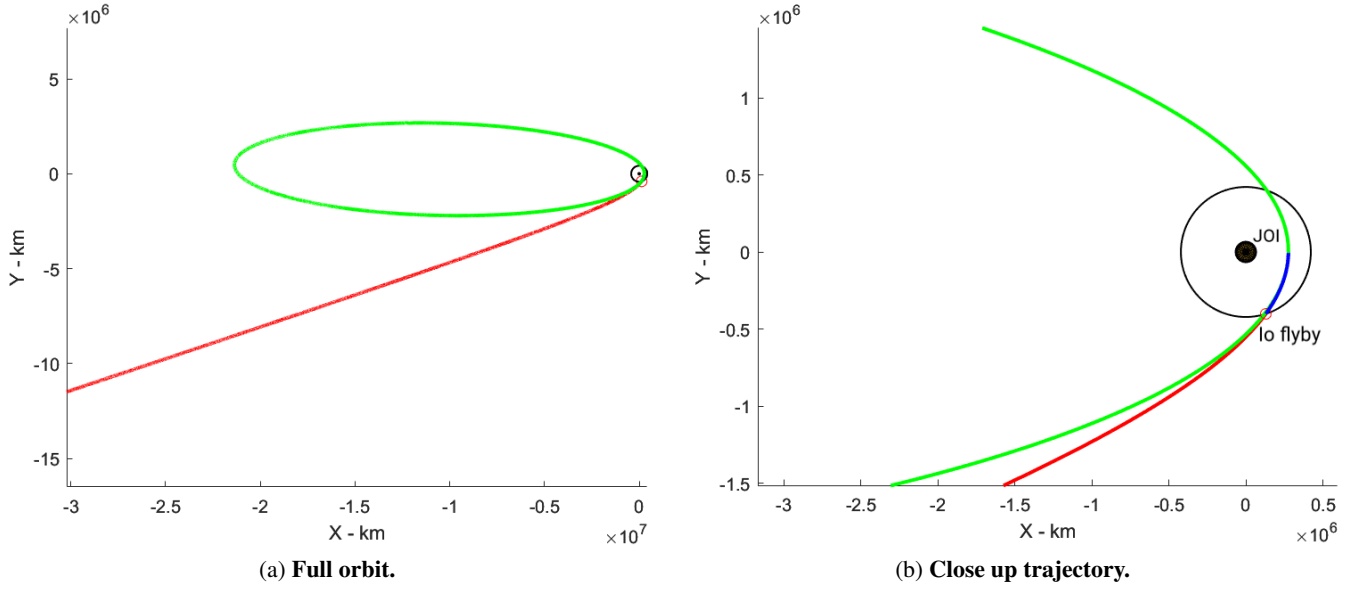


Figure 3: Galileo trajectory at Jupiter.

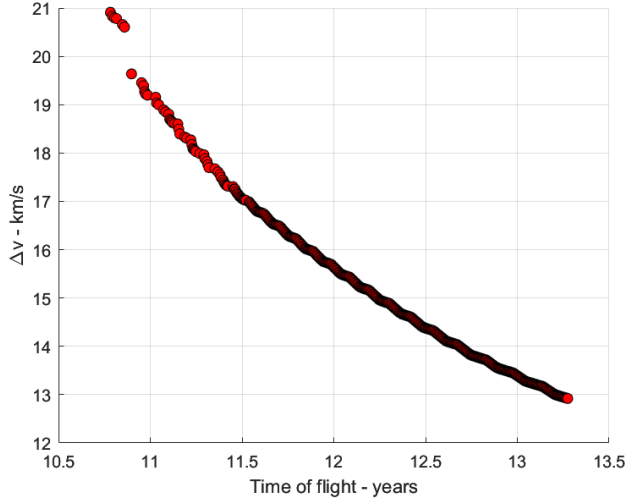


Figure 4: Pareto front for the EVECJU sequence in 2029.

time. For surveying great periods of time, a lower precision is necessary to reduce the computation time, but it is limited by the physical parameters of the moons, as a precision too low can cause some solutions to be lost. The satellite-aided capture trajectory is shown in Figure 6. The Oberon flyby is first performed at an altitude of 115 km, 4 hours before performing the Titania flyby. While the altitude of the first flyby is constrained to encounter the second moon, the second flyby can be performed as low as possible to ensure the maximum efficiency. Hence, the Titania flyby will be performed at 25 km. The capture burn is performed 10 hours later and requires 1.942 km/s of ΔV , with 16 m/s being saved from these flybys. Combined with the cost of the DSMs, it is a total of 2.259 km/s of ΔV that are required for the orbiter. These low values of ΔV saved, compared to what can be achieved at Jupiter and Saturn, can partly be explained by the low gravity of Uranus' moons compared to the ones of Jupiter, Saturn and even Neptune [24] [25] [26]. The higher incoming velocity at Uranus is also a cause of the lower

Table 4: Transfer to Uranus summary

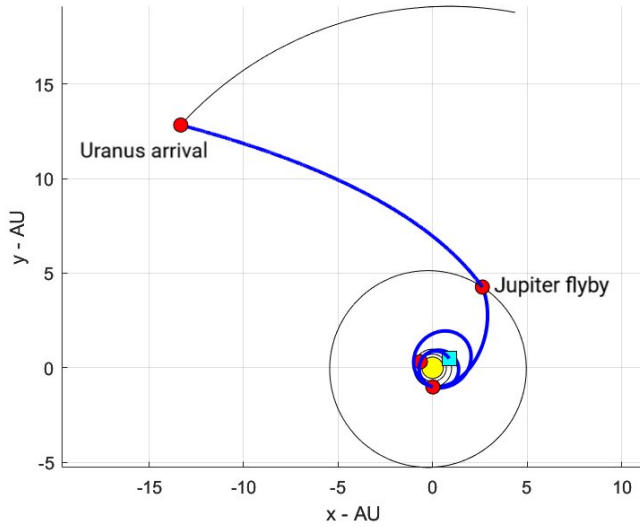
Event	Simulation
Earth departure	22 Oct. 2029
C_3 (km^2/s^2)	13.000
Venus flyby	5 Feb. 2030
DSM 1 (m/s)	250
Earth flyby	22 Jun. 2032
Earth flyby	22 Jun. 2034
DSM 2 (m/s)	47
Jupiter flyby	27 Mar. 2036
DSM 3 (m/s)	20
Uranus arrival	18 Jan. 2043
V_∞ (km/s)	9.036
Total DSM (m/s)	317

efficiency, as is means that the moon flybys will be performed at a higher velocity, reducing the maximum deflection angle, thus the maximum variation of orbital energy.

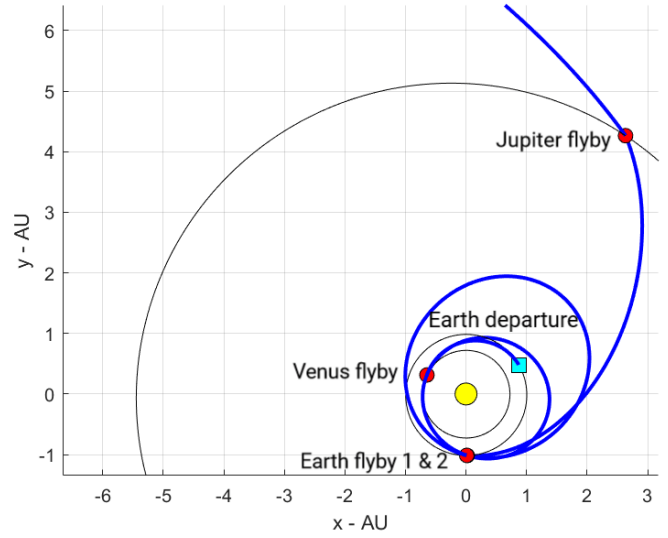
5. CONCLUSIONS

This paper presented a global approach to the design of complex interplanetary missions. The pipeline that was developed allow to perform a preliminary design of a mission that includes an MGA-DSM transfer and satellite-aided capture of any planet of the outer Solar system. This is achieved by using a multi-objective dynamic programming to solve the interplanetary transfer, and Lambert arcs for satellite-aided capture.

Multi-objective dynamic programming allow to efficiently survey the search space, guarantying the optimum Pareto front completing the transfer objectives. When short periods of time are surveyed, solving Lambert's problem is an efficient way of finding solutions for two-flyby satellite-aided capture. Properly choosing the lower and upper bounds for

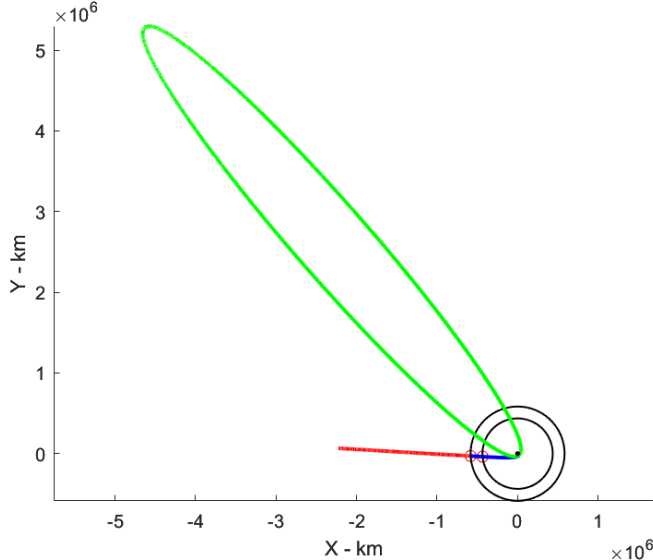


(a) Full trajectory.

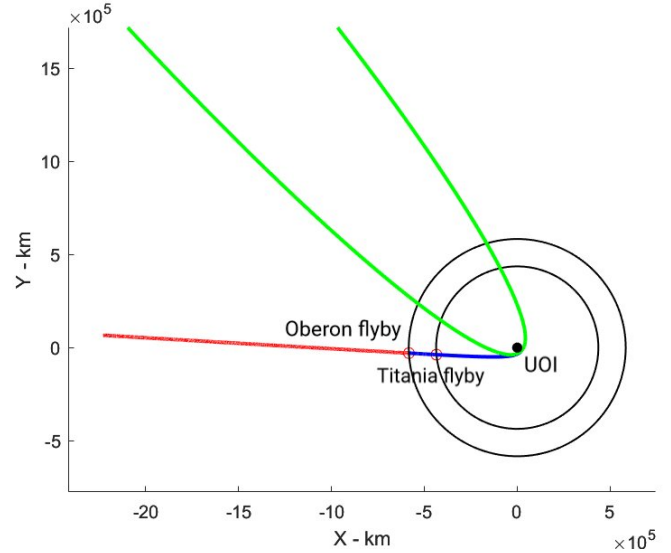


(b) Trajectory to Jupiter.

Figure 5: Transfer to Uranus for a launch on the 22/10/2029.



(a) Full orbit.



(b) Close up trajectory.

Figure 6: Orbital insertion at Uranus.

the transfer time between the two moons allows to reduce the computation time to a minimum while keeping enough precision to ensure no solution is missed. On the other end, the computation time is also proportional to the length of the surveyed period, meaning that for surveying long periods of time of one month and more, other methods like using phase angles would probably be more efficient. Another limitation appear for moons with a low gravitational field, mainly the ones of Uranus. Because the maximum deflection angle for these moons is lower, a greater precision on the transfer times is required when solving Lambert's problem. This means that more values must be tested, greatly increasing the computation time.

The capacity of the presented pipeline to recreate real missions showcase its ability to identify optimums without prior knowledge of the solution, for both interplanetary transfers

and satellite-aided capture. Additional tests show that the pipeline is also capable of creating new scenarios for planets that weren't visited by an orbiter before, namely Uranus and Neptune. These scenarios are more complex than missions to Jupiter and Saturn, mainly because of the lower gravity of their moons and the greater distance from the Earth, putting more constraints on the interplanetary sequence.

The present pipeline can also be updated to make use of a more precise model instead of the circular co-planar model when computing the satellite-aided capture, thanks to the use of Lambert arcs to design the transfer trajectories between moons. In order to effectively change the model, the definition of the initial trajectory inside the sphere of influence of the planet should be reworked.

APPENDIX: FLYBY SIMULATION

When following an hyperbolic trajectory through the sphere of influence of a celestial body, the initial velocity at infinity $\vec{v}_\infty^-$, given by Eq. (14), will be rotated by an angle δ to become $\vec{v}_\infty^+$ following Eq. (15). In these equations, $\vec{v}^-$ and $\vec{v}^+$ are the velocity vectors of the spacecraft before and after the flyby in the reference frame of the main body, and $\vec{v}_{pl}$ is the velocity vector of the secondary body.

$$\vec{V}_\infty^- = \vec{V}^- - \vec{V}_{pl} \quad (14)$$

$$\vec{V}_\infty^+ = \vec{V}^+ - \vec{V}_{pl} \quad (15)$$

The deflection angle δ is given by

$$\sin\left(\frac{\delta}{2}\right) = \left(1 + \frac{r_p \times V_\infty^2}{\mu}\right)^{-1} \quad (16)$$

with r_p the closest approach to the body, v_∞ the norm of the incoming velocity and μ the gravitational parameter of the body.

As part of the linked-conics approximation, the entering point of the trajectory in the sphere of influence of the secondary body is not defined. Therefore, there is a degree of freedom that is left undefined, the angle γ of the flyby plane. γ is defined as such

$$\vec{V}_\infty^+ = V_\infty \left(\cos(\delta) \vec{b}_1 + \sin(\delta) \sin(\gamma) \vec{b}_2 + \sin(\delta) \cos(\gamma) \vec{b}_3 \right) \quad (17)$$

with

$$\begin{aligned} \vec{b}_1 &= \frac{\vec{V}_\infty^-}{V_\infty} \\ \vec{b}_2 &= \frac{\vec{b}_1 \times \vec{r}_{pl}}{|\vec{b}_1 \times \vec{r}_{pl}|} \\ \vec{b}_3 &= \vec{b}_1 \times \vec{b}_2 \end{aligned} \quad (18)$$

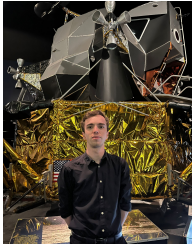
$\vec{r}_{pl}$ being the position vector of the body where the flyby is performed.

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BIOGRAPHY



Hugo Garny completed his engineering degree from ISAE-ENSMA (France), and his M.Sc in Astronautics and Space Engineering from Cranfield University (UK) in 2023. His areas of interest include astrodynamics and trajectory design. Following his M.Sc, he worked as a research assistant at Cranfield University for the Hydrotterra project, focusing on orbit simulation and control of geostationary orbits. He is now working as an R&D engineer at Eutelsat Group.



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Design of a pipeline for satellite-aided capture at the giant planets of the solar system

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