* **Introduction**

The mission’s extreme distance from the Sun leads to a very high fuel requirement. In order to achieve a low mass design, the impulses imparted on the spacecraft must be minimised as much as reasonably possible; this can be done with an optimised trajectory design. The current approach is to focus on minimising the fuel mass by reducing the Δv required to carry out the mission.

Three manoeuvres are performed after the spacecraft reaches its parking orbit around Earth: an initial departure burn, the largest of the mission, which will place the spacecraft on its trajectory to Neptune. If multiple gravity assists are used, this burn will be of a lower Δv compared to a direct transfer. Either way, the large impulse required necessitates a high-thrust, high-efficiency engine, so hydrolox propellants will be used. The next burn will slow the spacecraft at Neptune enough to capture into a highly eccentric orbit. Again, this requires a reasonable amount of Δv, but due to the high transfer time, boiloff is a concern and hence hypergolic propellant will be used. Finally, the spacecraft can gradually insert itself into the desired final orbit; since this can be done slowly, chemical propulsion will be used.

This report summarises the methods used to find such trajectories – both direct and using gravity assists – as well as showing the validation of these compared to established tools. Mission constraints will also be applied to design the final trajectory, and then this is used to decide on launch windows and acceptable Δv budgets for the rest of the design team to use. Finally, the effect of perturbations on the spacecraft will be analysed to increase the fidelity of the initial patched-conic trajectory.

* **Initial trajectory design**

The trajectory design is split into the following main sections: selection of a series of gravity assists with the aid of a Tisserand plot (or no gravity assists at all) which allow the spacecraft to reach Neptune following the mission constraints; and optimisation of the path between these planets using a simple patched-conic trajectory. Finally, the fidelity of this model can be improved by considering the perturbing accelerations due to n-body effects and solar radiation experienced by the spacecraft.

We chose to use Tisserand plots and a two-body patched-conic approach for the low-fidelity solver. The Tisserand plots were examined by other group members with theoretical optimum results, which are, of course, timing dependent, given in Table 1:

[table]

The timing of these trajectories, along with a simpler direct transfer, can be analysed using a Lambert solver, programmed in MATLAB. The premise of this method is to use vis-viva matching to chain together multiple heliocentric orbits in such a way that their intersections at the planetary flybys not only coincide with the planet’s position at that epoch, but also that the inbound and outbound C3 values are equal – a condition imposed by the laws of orbital mechanics, providing that there is no additional Δv imparted. It should also be noted that the turn angle should be checked to make sure that the spacecraft’s hyperbolic flyby trajectory passes the planet at a safe distance.

Burns during the flybys were considered and analysed using other tools, with their data presented in other reports. However, this was outside of the scope of this MATLAB implementation, especially as the concept of hibernation to save costs on the long transfer to Neptune requires a minimal number of manoeuvres. Some manoeuvres are a necessity such as the trajectory corrections.

A trajectory designed this way depends on the initial launch date (and hence planetary ephemeris) and a further parameter to determine the first Lambert transfer, which can be the initial C3 or the first leg time. In this case, we use the initial C3 as it is particularly important data in the optimisation process. Following this process for varying launch dates and finding the minimum required Δv for each case gives the following optimal trajectories corresponding to those from the Tisserand plots in Table 2, and are plotted in Figure 1:

[table]

[figure]

These data are also supported by tools such as KSP-TOT and PyKep, all of which are compared in Table 3:

[table]

This validates the Lambert solver implemented in MATLAB.

* Consideration of maximum transfer time – direct approach

A direct transfer can also be analysed in the same way; however, there are no flybys to consider. Traditionally, the advantage of such trajectories is in the reduction of flight time, which for this mission is very important due to the ambitious development timeline. This does come at the cost of increased Δv, which can be calculated in much the same way. For this method, since there are less variables, the data can be conveniently presented on a porkchop plot which shows the Δv required for the three manoeuvres as they vary with the launch date and transfer duration. This is shown graphically in Figure 2 and helps to decide Δv requirements to give a suitable launch window. This window repeats every 367.5 days, equal to one synodic period of Earth and Neptune. Using this we chose to design the system to have a 9.5 km/s departure burn and 3.5 km/s capture burn, giving a launch window approximately 2 weeks long.

* Perturbations???