Development of a Kick Stage and Trajectory Optimisation for Large Interplanetary Scientific Missions

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With SpaceX's Starship heralding an era of high-capacity launch vehicles, large-scale scientific missions like the proposed Arcanum mission to Neptune are now feasible. However, traditional methods for initially estimating fuel masses, like the impulsive burn approximation, are wholly inadequate and cannot be used to create realistic initial design concepts. This paper provides an in-depth analysis using the General Mission Analysis Tool (GMAT) to assess the Oberth effect and various Earth Departure Stage (EDS) designs, exploring propulsion technologies from hypergolic to cryogenic. We consider factors like fuel boil-off and the implications of using Starship as a launch vehicle for delivery into parking orbits which can then be used for deep space trajectories. Our findings, including a performance comparison of different EDS designs and heuristic tips for optimising large-mass ejections, are valuable for mission planners. The study also demonstrates GMAT's utility in combination with MATLAB in simulating realistic mission profiles. These results not only advance the Arcanum mission but also contribute broadly to deep space mission strategies.

Keywords: Astrodynamics, Starship, Hypergolic, Cryogenic

1 INTRODUCTION

1.1 Background

The advent of high-capacity launch vehicles, such as SpaceX's Starship [1], marks a new era in space exploration, promising to deliver ambitious missions to far-reaching destinations in the Solar System. This advancement, while offering unparalleled opportunities, also introduces complex challenges in trajectory planning and propulsion technology for such missions, where maximum payload masses are an order of magnitude higher than those previously seen [2]. The Arcanum mission, a concept developed by Conex Research, serves as a state-of-the-art use case to demonstrate these challenges and opportunities. Targeting the Neptunian system, the mission is designed to advance our understanding of distant celestial bodies, leveraging the increased payload capabilities of modern launch vehicles.

In this paper, we build upon previous studies on the Arcanum mission [3, 4, 5], and focus on the development of an efficient Earth Departure Stage (EDS) for the ejection of the mission from Earth orbit and onto a trajectory to Neptune. The mission's significance lies not only in its wide ranging scientific objectives, possible given this shift to larger capacity launch vehicles, but also in its demonstration of the potential of new propulsion units and astrodynamics, something which is useful in the planning of any large-scale deep-space missions in the future.

The challenge of launching substantial payloads beyond Earth's orbit is not entirely new, and has been addressed in several historic deep space missions. Notably, the Mars Science Laboratory mission, which delivered the Curiosity rover to Mars, had a launch mass of about 3,893 kg, including the rover, aeroshell, and fuel [6]. Another example is the Cassini-Huygens mission to Saturn, with a launch mass of approximately 5,712 kg, including the Huygens probe, which landed on Saturn's moon Titan [7]. What is new, however, is the order of magnitude difference in launch masses which will be possible with Starship - up to 100 tonnes [1] – and that, given the booster is reusable, their delivery will more than likely be to Earth orbit. It should be noted that this number, while supported by some real-world ground-based testing and significant funding [22, 23], is preliminary. Assuming 100 tonnes to low-Earth-orbit (LEO) to be possible, and with Starship refuelling 100 tonnes to geostationary transfer orbit (GTO) to be possible, and realised within the next 5-10 years, it provides a stark contrast to the previously mentioned missions, which were delivered onto a deep space, or near-deep space Earth-escape trajectory by their launch vehicle [6, 7]. The scale of the Arcanum mission, larger than those missions seen before at over 20 tonnes [3, 4, 5], amplifies these challenges, necessitating advancements in propulsion technology and trajectory optimisation to accommodate the significantly larger payload mass.

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1.2 Challenges with Large-Mass Ejections

The designing of the Arcanum mission, particularly the more recent and detailed assessment of the trajectory of the spacecraft, has revealed the impact of handling large masses on the traditional space mission design pipeline. For first-order sizing, designs were rooted in the conventional rocket equation and impulsive burn approximations, standard practice for mass sizing and fuel estimation [8]. However, in the case of Arcanum and indeed in the case of any large mission, the discrepancies in fuel mass estimations became glaringly apparent when finite burn modelling was applied due to the Oberth effect reducing burn efficiency when performed at lower speeds and in this case, away from the orbit periapsis [9]. The initial EDS design was capable of a single propulsive manoeuvre at Earth to set Arcanum on a deep-space trajectory, and one additional relight during a flyby of Earth to perform a gravity-assisted manoeuvre. Drop tanks, to contain the fuel for the first of these burns and then to be immediately ejected, were included to minimise the structural mass during the gravity assist. However, when we applied finite burn modelling to this design, we found it very difficult to achieve the required change in velocity, ΔV, with hypergolic technology, even when a complex series of propulsive manoeuvres were introduced. Therefore, while this technology is favoured for simplicity, reliability, and inert nature in long-term storage, in handling the demands of such a large-scale mission it is necessary to explore other propulsion technologies.

This study, therefore, explores alternative propulsion strategies, pivoting towards cryogenic propulsion due to its higher specific impulse and efficiency. Cryogenic systems, though complex and sensitive to fuel boil-off offer the high thrust and efficiency necessary for large-mass Earth escape manoeuvres. The boil-off specifically is seen to drastically impact the optimal mission profile, and therefore the number of burns which are used, outweighing the losses due to the Oberth effect and necessitating immediate engine firing for the optimal fuel usage. We also briefly discuss our initial assessment of electric propulsion options, known for their high efficiency but low thrust, something which makes them completely infeasible for large mass ejection given the duration of such a manoeuvre. Nuclear thermal propulsion is also briefly discussed, offering high thrust and efficiency. However, the associated technological and regulatory hurdles place it beyond our mission's scope. A performance guide for the various propulsion technologies is presented in Table 1.

TABLE 1: Performance characteristics for different propulsion technologies [8]

Propulsion type	Specific Impulse (s)	Thrust (N)
Hydrolox	450	$1\text{-}10^6$
Hypergolic	300-340	$1\text{-}10^6$
Hall Effect	1,500-2,500	$10^{-6}\text{-}0.02$
Lithergol	280-300	$10\text{-}10^8$

1.3 Methodological Advancements and Implications

Tackling the complexities of large-mass ejections required a novel methodological approach. In this paper, we introduce an integrated approach that harnesses the capabilities of MATLAB and GMAT for enhanced first-order estimations and detailed finite burn modelling. This integration allows us to simulate and optimise trajectories with greater accuracy, taking into account the nuances of different propulsion technologies and their operational constraints, while also capturing the complex behaviours of finite burns. By establishing a framework that integrates advanced computational tools and real-world mission constraints, this study not only lays the groundwork for the Arcanum mission's journey to Neptune but also paves the way for the future of interplanetary exploration.

1.4 Trajectory Overview

The trajectory Arcanum will follow to Neptune was determined by taking into account the transfer time and burn efficiency. Utilising the sophisticated astrodynamics tool, pykep, developed by the European Space Agency (ESA) [10], we have optimised the trajectory to minimise the fuel required to reach Neptune in 15 years. To do this, given the large spacecraft mass and the inability of any current or near-future launch vehicle to provide the energy required for a direct deep-space trajectory injection, a series of gravitational assists are required.

The launch time of the mission in 2032 allows the spacecraft to perform a gravitational assist at Jupiter, similar to the Voyager and Cassini missions [7]. After considering several possible iterations using pykep, an EVEEJN trajectory was found to be that which most closely matched the requirements of speed and fuel use.

Given this, the Arcanum spacecraft will start from a geostationary transfer orbit and then, using the EDS, perform a manoeuvre of less than 4 km/s that directs it to Venus. At Venus's perigee, a short burn is performed that directs the spacecraft back to Earth where it will perform two gravity assists over a period of two years and head towards Jupiter. The final gravity assist at Jupiter will allow Arcanum to reach Neptune by November 2045.

In total, Arcanum will take 15 years and 4 gravitational assists to get to Neptune. A complete summary of this trajectory is shown in Table 2.

TABLE 2: Summary of the trajectory of the Arcanum mission from Earth to Neptune

Course Corrections and Manoeuvres	Date	ΔV [m/s]
Earth Departure Manoeuvre	31 October, 2030	3,723
Venus Course Correction	27 June, 2031	0.5033
Earth Course Correction 1	24 August, 2031	0.2146
Earth Course Correction 2	11 April, 2033	2,006
Jupiter Course Correction	23 June, 2034	0.7332
Neptune Orbital Insertion	30 October, 2045	2,763
Total Flight Time	**15.007 years**	**8,403**

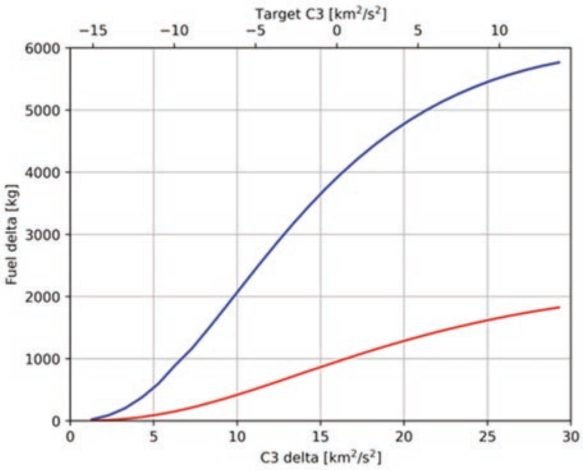

Fig.1 The difference in fuel use (fuel delta) as calculated by finite-numerical methods rather than impulsively when a burn injects given characteristic velocity C3 to a trajectory (C3 delta). This is analysis is performed for spacecraft of 10,000 kg (blue) and 20,000 kg (red) total mass.

1.5 Preliminary Concepts and Assumptions

Rocket Equation and Impulsive Burn The rocket equation, also known as Tsiolkovsky's equation, is the fundamental basis for understanding the kinematics of rocket flight [11]. The equation represents the change in velocity ΔV as a function of the initial and final mass of the rocket m_i and m_f respectively and the effective exhaust velocity v_e, and is:

$$\Delta V = v_e \, ln \frac{m_i}{m_f} \tag{1}$$

While the rocket equation provides a useful initial approximation, in which it is commonly used to calculate the performance of an impulsive burn, the need for a more complex solution becomes increasingly apparent for missions as their masses – and therefore required burn times – and complexity increases.

Oberth Effect The Oberth effect is a phenomenon in astronautics where a spacecraft can gain more kinetic energy from a manoeuvre when it is executed at higher speeds, typically at its closest to the centre of a celestial body it is orbiting [9]. Mathematically, the Oberth effect can be understood as the kinetic energy KE being proportional to the square of the velocity v:

$$KE = \frac{1}{2} \, m \, v^2 \tag{2}$$

For the Arcanum mission, optimising the Earth departure manoeuvre to exploit the Oberth effect is critical for minimising fuel consumption and thus maximising the science payload capacity.

Fig. 1 shows the increased fuel use that can be expected for the injection of given characteristic velocity C3 values, for missions at the same order of magnitude mass values as Arcanum.

1.6 Earth Departure Stage Designs

The EDS has two main purposes, among others to propel Arcanum out of Earth's gravity well onto its deep space trajectory. Two different concepts have been investigated in detail for this.

Hypergolic More than two years separate the initial Earth departure burn and the burn performed during Earth flyby that sets Arcanum for the outer Solar System. Evidently, at least the second task needs to be performed by storable propellant. We focused on a hydrazine-nitrogen tetroxide propellant because technology and experience with this fuel combination is readily available, reducing cost and improving reliability. Fuel density is high; allowing smaller tanks to be installed, which not only reduces weight, but also handling during integration. Its hypergolic nature simplifies ignition and restartability, even after many years in deep space.

For the same reasons, this combination was used to facilitate Earth departure. In addition to the advantages already mentioned, it allows the complete EDS to be fuelled prior to integration with the launch vehicle and it does not produce fumes or vapours of any kind inside the launch bay. Toxicity is a downside, but considering multi-decade experience in handling hydrazine this is a manageable problem. As aforementioned, drop tanks are used to store the fuel used in the first manoeuvre and are the ejected immediately after expenditure. Fig.2 shows the general arrangement of the drop tanks around the column-like centre stage, though further refinement of

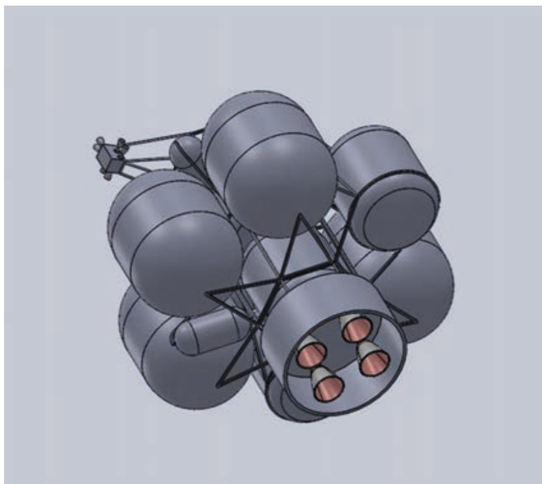

Fig.2 Schematic depiction of the hypergolic EDS configuration including drop tanks, three quarter aft view.

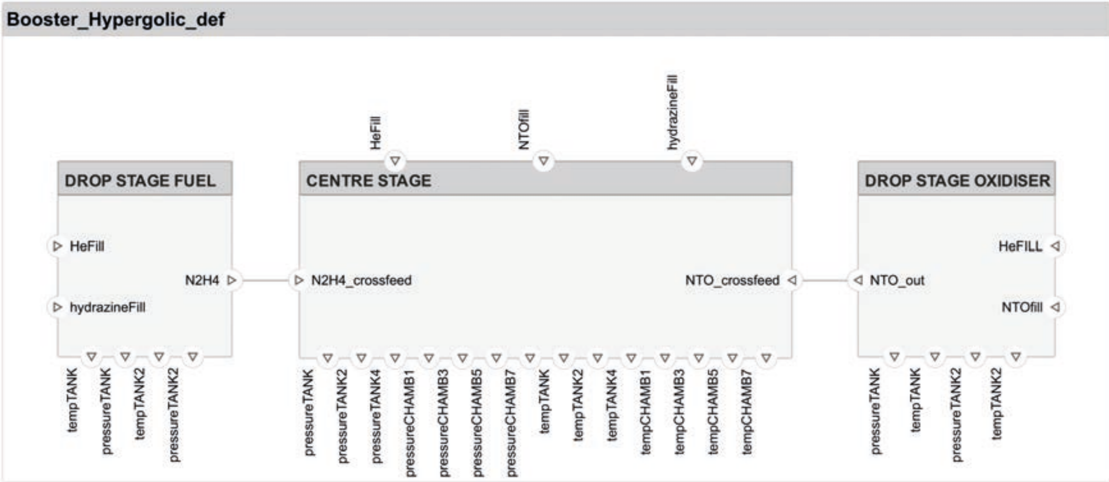

Fig.3 Block diagram of configuration 1 of the Earth Departure Stage.

tank shapes, structure and layout is still required.

Fig. 3 shows the EDS from a structural view. *CENTRE STAGE* contains the engines, attitude control thrusters, interfacing to both the orbital spacecraft and the launch vehicle and propellant for the second Earth flyby. *DROP STAGE FUEL* and *DROP STAGE OXIDISER* contain only the propellant for the Earth departure manoeuvre.

The composition of the centre stage is shown in Fig. 4. Fuel and oxidiser are stored in separate assemblies, called *Fuel_Assembly_CS* and *Oxidiser_Assembly_CS*, respectively. Both are driven by high-pressure helium that is stored in *Pressuriser_Assembly_CS* towards *Engine_Assembly_1* and *Engine_Assembly_2*. The four reaction control system thruster quads *RCS_Assembly_+y* through *-z* are supplied with propellant from them as well.

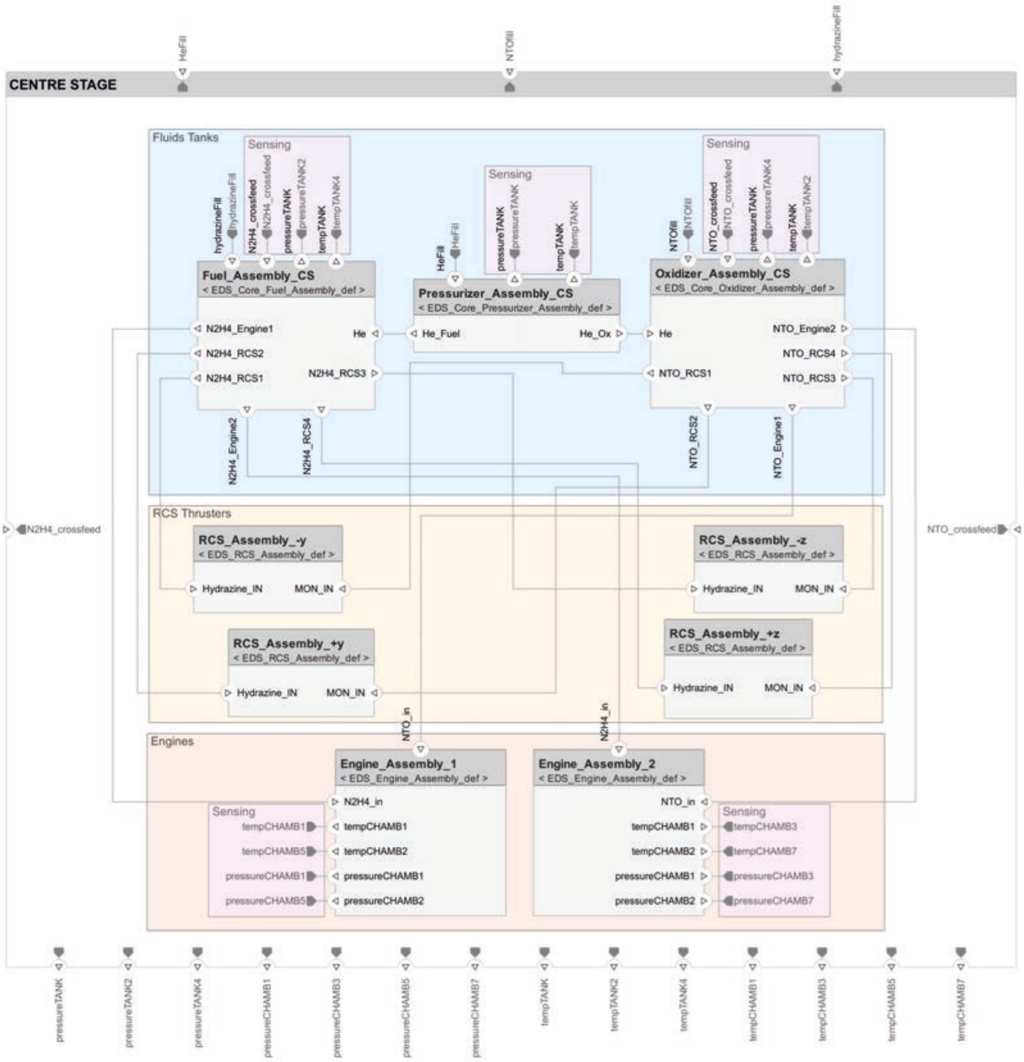

Fig.4 Block diagram of the centre stage of configuration 1.

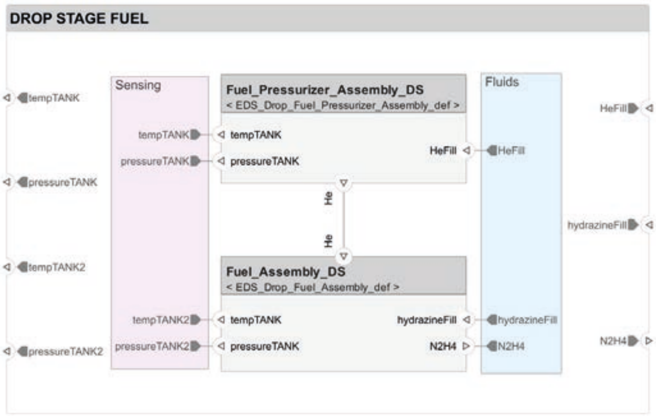

Fig.5 Block diagram of the drop tank subsystem containing the fuel.

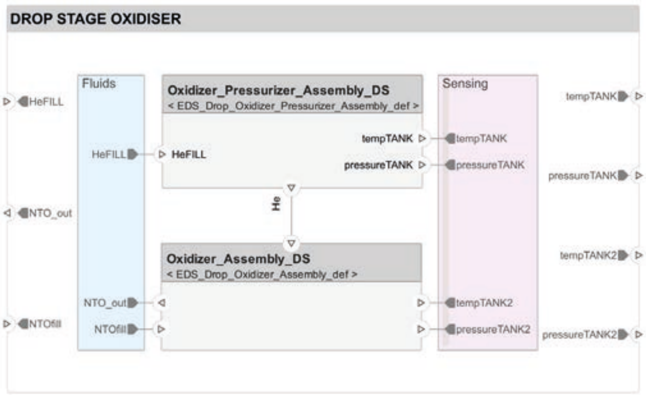

Fig.6 Block diagram of the drop tank subsystem containing the oxidiser.

Both drop tanks, schematically shown in Fig. 5 and Fig. 6 contain, besides their respective propellant components in *Fuel_Assembly_DS* and *Oxidiser_Assembly_DS*, high pressure helium in separate assemblies, specifically for the Earth departure burn.

Both engine assemblies are identical and, as shown in Fig. 7 built around two R-4D HiPat bipropellant thrusters. The engines, which are in a fixed position without any gimbling, are arranged in the shape of a square. Two engines belonging to the same assembly are placed diagonally to each other so that in case of a failure of one of the two engine assemblies the malfunctioning one can be turned off without affecting the centre of thrust.

Cryogenic As concerns were raised about Hypergolic solutions not providing the necessary thrust and specific impulse to achieve the desired trajectory for the first burn, a second booster design, using cryogenic propulsion was developed. Due to its wide availability and heritage in spaceflight, liquid Hydrogen was selected as the fuel of choice with six parts of oxidizer (oxygen) per part of fuel (liquid hydrogen), by mass. Two primary engines were then considered for the cryogenic phase, with their principle values that were used in the detailed trajectory analysis listed in Table 3.

Upon simulating the first burn using cryogenic propulsion, it was established that a single Vinci engine could provide the necessary thrust to depart from Earth orbit. The Vinci engine is currently in the final stages of development and is to be deployed within the upper stage of the Ariane 6 [13]. It is not yet, however, planned to include a self-start capability, strictly meaning that some further development of this baseline version would be required for its use in the proposed capacity. A system architecture was subsequently developed around the Vinci engine as the primary propulsion. As the Vinci engine itself incorporates more of the fuel and oxidiser processing procedures compared to the R-4D-15 HiPAT thrusters used for the hypergolic proposal, the architecture is significantly more simplified by considering the engine as a black box with the relevant interfaces. The use of cryogenic propulsion also mitigated the need for drop tanks as were required for the hypergolic configuration shown in Fig. 4, simplifying the system further.

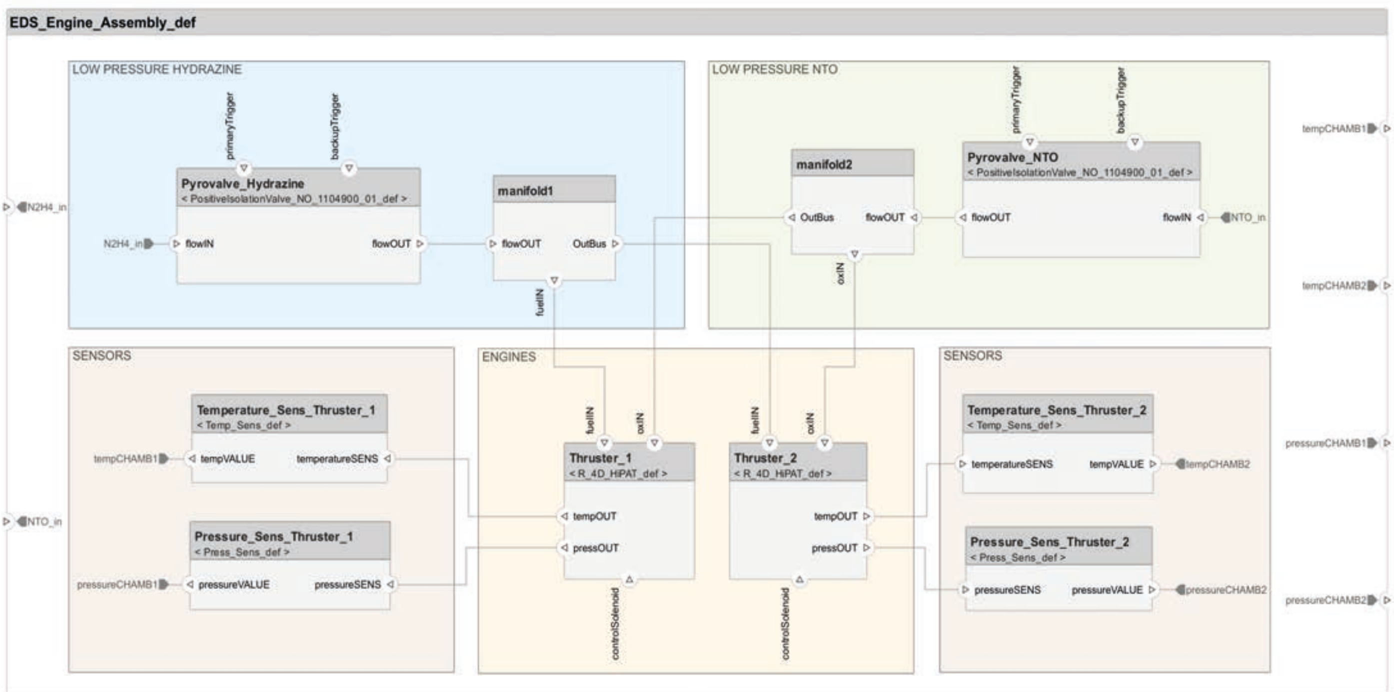

Fig.7 Block diagram of the engine assemblies

TABLE 3: Performance characteristics of considered cryogenic propulsion engines

Engine	Status	Dry Mass (kg)	Thrust (kN)	I_{sp} (s)
RL10C-2-1	Active	301	109	465.5
Vinci	First launch in 2024	550	180	457

The system architecture is composed of the same functional blocks as the hypergolic system, though the engine blocks as mentioned prior are more simplified. The oxidiser and fuel assemblies however had to consider the fuel boil-off that would occur between the time the tanks are filled to when the ejection burn is performed. This boil-off was conservatively estimated to be up to 5 percent.

This study considers only EDS designs – both hypergolic and cryogenic – which are possible with commercial off-the-shelf (COTS) components, meaning both stages could be further optimised. While gains to the hypergolic stage performance would ordinarily be comparable to those in the cryogenic configuration during an optimisation process, one area which disproportionately disadvantages the cryogenic stage is the boil-off, which as aforementioned is a large contributor to the performance of the stage. By taking a conservative estimate for boil-off, we ensure that if the cryogenic stage appears preferential to the hypergolic option in this study, it would certainly out-perform the hypergolic option once boil-off optimisations have been applied. Light consideration of such optimisation is briefly discussed here, but detailed analysis is reserved for future work. The boil-off rate could be reduced by, for example, using subcooled tank technologies like those studied by the United Launch Alliance (ULA) in the design of the Advanced Common Evolved Stage (ACES) [19, 20, 21]. Here, the gaseous hydrogen and oxygen from boil-off are passed through ULA's Integrated Vehicle Fluids (IVF) module. This is used for pressurising and vapour cooling the propellant tanks, generating electrical power via combustion in a light-weight auxiliary power unit and providing reaction control using GH2/GO2 thrusters. This eliminates the need for helium pressurising gas, batteries and hydrazine respectively while increasing the performance and endurance of the stage for the same mass of propellants [19]. ULA's integrated approach is estimated to yield a negligible boil-off rate for oxygen and approximately 4.4e-5 kg/s for hydrogen [19]. This reduced boil-off rate is desirable as is shown later in this article. However, as aforementioned, our COTS limitation means we precluded these advantages for now. The boil-off optimisations implemented in the ACES should be evaluated for their suitability to the Arcanum mission in a future analysis.

The attitude control system (ACS) for the Earth departure phase will be powered separately from the cryogenic system that performs the first burn. In its current design iteration, the attitude control thrusters would be fuelled by the hypergolic stage that would be used for the second large burn at the Earth flyby two years after launch. The ACS propulsion systems can be seen in Fig. 8 and can be seen to have no physical connec-

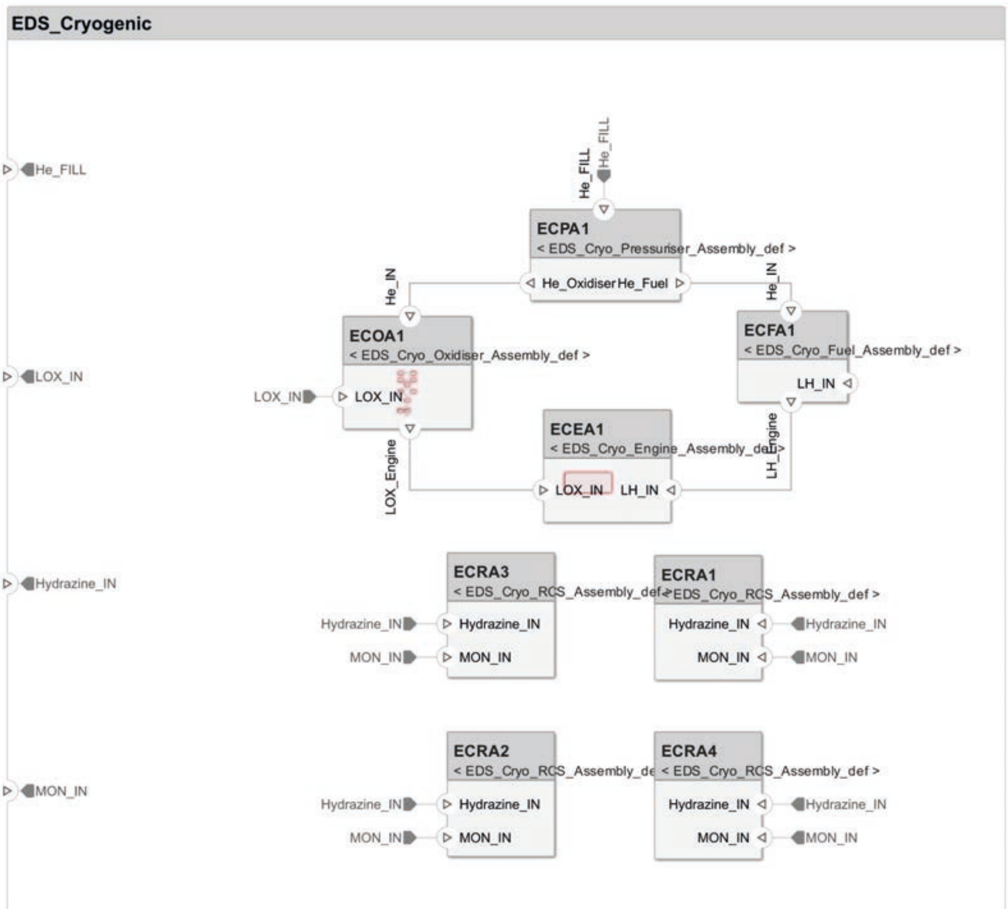

Fig.8 **Block Diagram of the top level cryogenic booster design.**

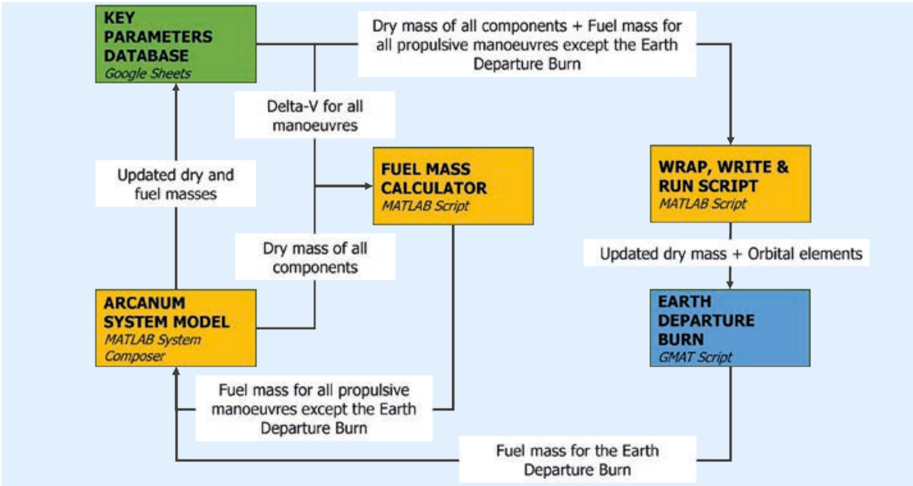

Fig.9 A process diagram showing the simulation workflow to assess the performance of stage designs.

tion to the rest of the system, though they are included as they would be used during the Earth departure Stage.

1.7 Future design work and other propulsion technologies

This study focused on only those technologies considered most feasible, these being chemical, bipropellant propulsion. Other methods are available however. These methods include nuclear-thermal, electric, solar sails and solid chemical rockets.

Nuclear Thermal This was rejected because it is a difficult and highly restricted technology, well outside the reach of any non-governmental organisation.

Solar Sails Solar sails, although rated with a technology readiness level (TRL) of at least 7 [13] were rejected because for even standard missions a flight to Neptune would result in a high arrival velocity of approximately 20 km/s [14]. This is an order of magnitude larger than the velocity change for the current high thrust trajectory, resulting in a prohibitively high propellant mass for the mission.

Electric Electric propulsion was also rejected because initial estimations showed that chemical propulsion would fulfil every requirement and would be more reliable and simpler. With the results outlined in this paper, especially highlighting the insufficient thrust of commercially available bipropellant thrusters, we will reevaluating electric propulsion in our future work.

Solid Chemical During our investigation into potential propulsion technologies, solid propellant was initially considered, but quickly rejected. Although solid rocket motors provide high thrust and are very reliable, their specific impulse is usually low compared to other commercially available technologies like liquid or electric propulsion (e.g. Table 1). Another disadvantage is that the propellant mass of solid rocket motors are fixed; this is in contrast to liquid propulsion where the tanks - and therefore the amount of fuel - can be chosen independently from the engines they feed.

Literature research did hint to advanced solid rocket motors with lithium in the context of the Mars Expedition proposal by North American Rockwell. This engine could produce a specific impulse up to 325 s [15], but there does not seem to be any experimental data supporting this performance. Other experiments did, however, report an increase in performance of around 10% [16]. Nonetheless the uncertain nature of this technology, combined with a specific impulse that is still poor compared with liquid propellant propulsion, led us to reject solid rocket engines as a potential EDS motor.

2 METHODOLOGY

The performance of the different configurations of the Earth Departure Stage was assessed using the iterative simulation method detailed in Fig. 9. This process aims to minimise the fuel mass required for the Earth departure manoeuvre, the number of burns and the duration of each burn. These parameters were optimised with a common constraint of delivering the required characteristic energy to escape Earth's sphere of influence. The major advantage of this Model-Based System Engineering (MBSE) approach is that it minimises the time between iterations. A core element of this simulation workflow is MATLAB due to its wide range of libraries, such as the system composer, and ease of integration upstream and downstream of the simulation when pulling and pushing data respectively. Each of the blocks in Fig. 9 has the following functions:

- **Key Parameter Database:** A database to store the value of mission variables, such as delta-V, spacecraft dry mass and fuel mass, that can be called and rewritten by simulation code.
- **Arcanum System Model:** A visual model of the Arcanum mission architecture organised by subsystem and containing performance data for the components that are used by each system.
- **Fuel Mass Calculator:** A script that calculates the fuel mass required to provide the required delta-V for each propulsive manoeuvre using the Tsiolkovsky Rocketry Equation.
- **Wrap, Write & Run Script**: A script that calculates the total mass of the system, writes the value to a GMAT script and runs the GMAT script to calculate the Earth Departure Burn fuel mass.
- **Earth Departure Burn:** An orbital simulation that minimises the fuel mass required for the Earth Departure Burn to produce the required delta-V using a non-linear optimisation algorithm.

2.1 General Mission Analysis Tool (GMAT)

GMAT is an open-source space mission analysis tool developed by NASA and private industry [17]. The tool contains powerful features for the initial design of space missions. These include optimising orbital trajectories, calculating the duration of communications blackouts and simulating the power generation and usage from solar panels to name a few. These capabilities have led to GMAT being used in the development of notable missions such as NASA's OSIRIS-REX and the Artemis program.

The orbital simulation capabilities of GMAT were necessary to determine the efficacy of each stage design. Trajectories can be optimised to minimise certain parameters such as fuel mass. In the case of the EDS design, minimising fuel mass used in the Earth departure burn would leave sufficient fuel for all subsequent manoeuvres. The trajectory and location of propulsive manoeuvres were optimised by an iterative process with GMAT. However, as the model matured, other contributions to the total fuel mass required were included such as the fuel required for attitude control.

2.2 Propagator

The default parameters for the propagator were selected due to their proven reliability for near-Earth trajectories with a duration on the order of one month. This was applicable as the spacecraft's initial position was set as an equatorial geostationary transfer orbit. After the departure burn, the spacecraft would be in a hyperbolic orbit. Due to the high starting altitude of the spacecraft, the Jacchia-Roberts atmospheric drag model, which is only effective below altitudes of 2,500 km [18], was omitted from the simulation.

However, specific adjustments were made to enhance the accuracy of the simulations. Notably, the inclusion of solar radiation pressure in the model was deemed essential. Solar radiation pressure is known for its substantial impact on spacecraft trajectories, especially those with large surface areas relative to their mass. The Arcanum system architecture does not feature any components with a significantly large area. Therefore, it may have been sufficient to omit solar radiation pressure from the simulation, but its negligible effect was included for accuracy.

The Sun and Luna (Moon) were the only celestial bodies included as point masses due to their significant gravitational influence on a spacecraft's trajectory. This approach avoids the complexity and computational load of incorporating less influential bodies. Furthermore, this approach is particularly suitable for efficiently simulating trajectories of spacecraft in Earth orbit or cis-lunar space.

2.3 Mission Sequence - Pump Up

The mission sequence in GMAT controls the order of operations that must take place to simulate a chosen orbit whilst tracking key parameters, such as fuel mass, and displaying the path travelled by the spacecraft. The mission sequence as shown in Fig. 10 was used for analysing the hypergolic EDS designs as it was found that the burn efficiency increased when using multiple short burns. It begins with setting the initial fuel level of the EDS. The fuel mass has been pulled from the key parameter database and written to the fuel tank of the EDS via MATLAB detailed in Fig. 9. GMAT must then write this into mission sequence which is a clear example of the data interface between MATLAB and GMAT.

An optimisation process, named *optimisePumpUp*, is then initiated with the Yukon solver. This is a non-linear optimisation algorithm that tries to achieve a desired value when operating with multiple constraints. The sequence includes constraints and objectives for the optimisation, such as targeting a characteristic energy (C3) and minimising fuel mass.

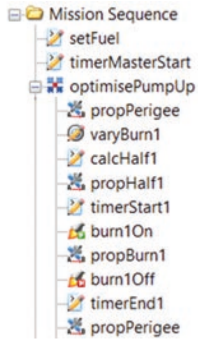

Fig.10 A single loop of the pump-up mission sequence in NASA's GMAT.

Additionally, it involves varying the end time of a burn and a backward propagation time. The backward propagation time is then inverted to a negative value, and the spacecraft is propagated backwards for this duration. This allows for the duration of the burn to operate for an equal amount of time when approaching and departing the perigee. This was found to be most efficient for minimising fuel mass which was calculated once the optimised burn had been completed. This section of the mission sequence was repeated a set number of times which had to be adjusted manually.

2.4 Mission Sequence – Ejection

After the *optimisePumpUp*, the spacecraft is in a highly eccentric GTO orbit. Following this, the spacecraft is then propagated to the periapsis of its orbit to initiate an optimisation function named *optimiseEject*. The mission sequence shown in Fig. 11 is the only sequence required for analysis of the cryogenic EDS design. This is due to an assumption that Arcanum equipped with a cryogenic EDS would be placed in the same highly eccentric orbit that results from the *optimisePumpUp* function.

As with the previous optimisation loop, the Yukon solver minimised the fuel mass whilst targeting a C3 of 70 km^2/s^2. This differs slightly from the *optimisePumpUp* function where the C3 targeted in each loop is equal to 70 km^2/s^2 divided by the number of burns. Another similarity with the previous optimisation function is that the burn commences from the periapsis and then is backpropagated so that half the burn occurs before and after the perigee. Finally, the optimisation process ends, and the spacecraft is propagated for an additional 120,000 seconds. This visually displays that the spacecraft has achieved a hyperbolic orbit.

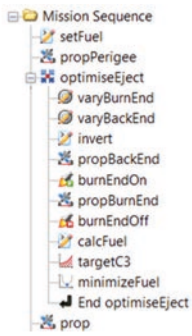

Fig.11 The mission sequence for optimising the ejection from Earth's sphere of influence in NASA's GMAT.

2.4 Starship Constraints

SpaceX's Starship represents a paradigm shift in launch vehicle capabilities, offering high payload capacity and potentially lower launch costs. However, planning to use Starship for the Arcanum mission entails several constraints and considerations.

Battery Life and Power Systems: The longevity and reliability of Starship's power systems, particularly its batteries, are yet to be tested in long-duration space missions. The uncertainty regarding battery life and performance under extended space conditions mean the duration for which Starship can remain in space is not known. It was therefore decided that the use of Starship should be no longer than one month.

Launch Vehicle Performance and Payload Integration: Given Starship is designed to carry large payloads, it is essential to examine how the vehicle's performance metrics, such as lift capability, align with the requirements of the Arcanum mission. Additionally, the process of payload integration into Starship's payload bay needs careful planning to ensure compatibility. While preliminary numbers are available – 100 tonnes to low-Earth orbit and also to GTO when additionally considering on-orbit refuelling [1] – these cannot be considered certain until successful orbital flights.

Orbital Refuelling: One of the innovative concepts proposed for Starship is orbital refuelling, which could significantly extend the mission reach. It was decided not to rely on this as a mechanic as it still requires significant development and qualification. Without refuelling, Starship is planned to deliver up to 21 tonnes to GTO in a single launch [1], covering the EDS configurations considered in this study.

2.5 MATLAB Simulink for Auxiliary Calculations

The EDS stage starts its journey from a highly elliptical GTO orbit and relies on its propulsion system to attain a hyperbolic trajectory towards Venus. This thrusting phase will, in the cryogenic configuration, last for a duration of 305 seconds. During this phase, we faced a critical decision: whether to perform the burn along the velocity vector or adhere to a constant direction in space.

To resolve this dilemma, we conducted extensive simulations using GMAT to assess the disparity in fuel consumption between these two scenarios. After careful analysis, we found the fuel requirement to be lower by 11 kg if we burn along the velocity vector. Intrigued by the challenge of ascertaining the fuel requirements for maintaining the attitude necessary to keep the main thruster oriented opposite to the velocity vector, and to determine its impact on the overall mass budget, we embarked on a small-scale study.

Attitude control thrusters need to be very reliable, be capable of performing multiple restarts, and also have a low response time. Given these stringent requirements, non-hypergolic propellants were deemed unsuitable. Consequently, we confronted the choice between cold gas thrusters and hypergolic propellants. Our current stage design incorporates hypergolic thrusters intended for attitude correction during subsequent gravity-assist burns, making it advantageous to employ these thrusters for the EDS stage as well.

Our initial calculations for attitude control fuel consumption necessitated an estimation of the moment of inertia, for which the stage was considered as a cylinder. These parameters, along with our prescribed trajectory, were processed with the help of the aerospace blockset within Simulink. It is worth noting that our analysis entailed several assumptions. First, we treated the moment of inertia as a constant throughout the burn, which represented a conservative estimate for fuel consumption. This simplification obviated the need to account for the complexities arising from a variable moment of inertia.

Second, we computed the attitude control fuel requirement for the entire duration of the burn, both before and after the perigee, under the assumption that the main engine would remain inactive during this period — although this does not align with reality. This simplified our calculations, and it is worth noting that it will lead to a higher estimate of fuel than the case when the main engine is burning because the angular change required will be higher.

Third, we assumed that the attitude control thrusters would maintain a constant thrust, avoiding the unnecessary complexity of throttling. This choice also contributed to a higher estimated fuel requirement compared to what would be obtained with variable thrust. Notably, the majority of rockets and spacecraft employ attitude control thrusters with constant thrust profiles.

In light of these assumptions, our Simulink setup and control scheme estimated that approximately 10 kg of fuel would be required to maintain attitude. This fuel would be required for both burning along velocity vector or burning along a constant direction in space.

3 RESULTS AND DISCUSSION

3.1 Performance of Single Burn Cryogenic Configuration

The simulation results for the single burn cryogenic configuration reveal critical insights into the efficiency and feasibility of this approach. Two key performance metrics — specific impulse (I_{sp}) and thrust — were varied to assess their impact on fuel consumption, a crucial factor in this design process. A single burn case was taken given that boil-off makes any other staging, such as orbital 'pump ups' as mentioned previously, infeasible.

Fuel Usage versus Specific Impulse and Thrust: The colourmap plot, shown in Fig. 12, illustrates the relationship between fuel usage, specific impulse, and thrust for this cryogenic case,

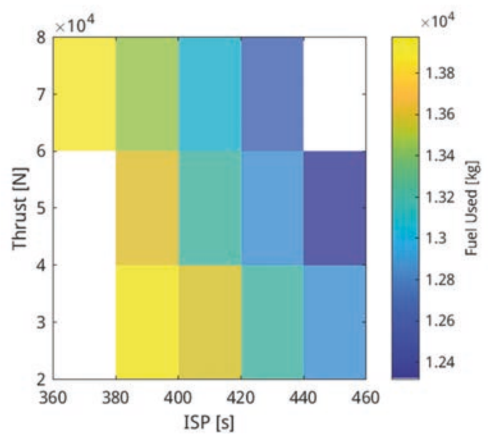

Fig.12 Colourmap showing fuel usage as a function of specific impulse and thrust for the single burn cryogenic case targeting a C3 of 70 km²/s², where the ejection was possible.

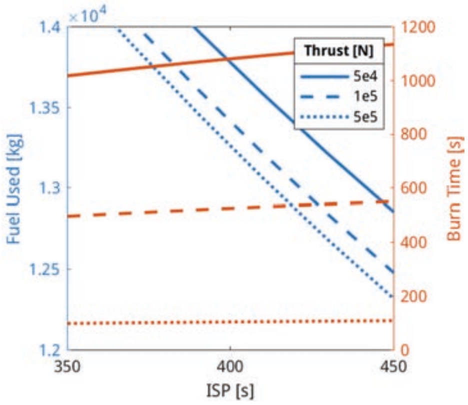

Fig.13 Plot showing the fuel usage for fixed thrust and specific impulse values, targeting a C3 of 70 km^2/s^2.

when targeting the C3 of 70 km^2/s^2 required for the trajectory to Neptune. This plot highlights that while the clearly expected trend - that the specific impulse and thrust increases, the fuel requirement decreases - the difference in fuel consumption is not as significant as in the hypergolic case. This configuration is, therefore, much more robust to changes in design.

The line plot, shown in Fig. 13, further illustrates this relationship, showing the fuel usage resulting from fixed specific impulse and thrust values. Additionally shown are the burn times for these ejection burns.

Analysis and Implications: The results from these simulations are pivotal in guiding the propulsion system design for the Arcanum mission. A wide parameter space of cryogenic propulsion options are capable of reaching the required velocity to reach the desired Neptune trajectory, and so can be safely considered as a feasible EDS propulsion option.

While this analysis is tailored to the Arcanum mission, the insights gleaned have wider applications to any missions with a mass on the order of the Arcanum mission

3.2 Performance of Two-Orbit Pump-Up Hypergolic Configuration

In addition to the single burn cryogenic configuration, our study also examined a hypergolic configuration employing a two-orbit 'pump-up' strategy. This approach involves performing burns at the periapsis of the first two orbits, capitalising on the Oberth effect, and then executing the ejection manoeuvre in the third orbit. The results of this configuration provide valuable insights into the efficiency and dynamics of this more complex manoeuvre. As aforementioned, the required C3 of 70 km^2/s^2 is not possible with this configuration, and so a parameter space targeting the achievable 43 km^2/s^2 was investigated.

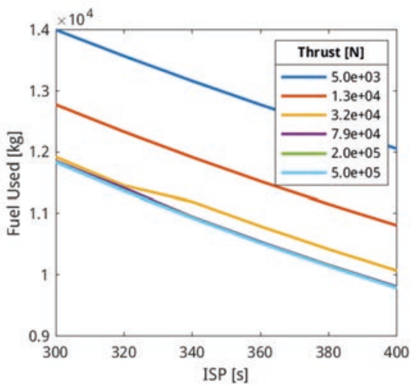

Fig.14 Plot illustrating fuel usage against specific impulse for various thrust levels in the two-orbit pump-up hypergolic configuration targeting 43 km^2/s^2.

Fuel Usage in Relation to Specific Impulse and Thrust: Fig. 14 presents a view of the fuel usage in relation to specific impulse (I_{sp}) and varying thrust levels.

A notable trend observed in the plot is that the benefits in fuel usage diminish significantly beyond a thrust of 3×10^4 N. This indicates that while increasing the thrust does contribute to fuel efficiency, there is an optimal range beyond which the improvements are marginal. Additionally, as expected, the fuel usage decreases with an increase in specific impulse, underscoring the importance of high-efficiency propulsion systems in such complex manoeuvres.

Analysis and Implications of Hypergolic Configuration: The hypergolic configuration results underscore the efficacy of a staged propulsion approach, especially when utilising the Oberth effect in the first two orbits. The analysis reveals that careful calibration of thrust and specific impulse is crucial in optimising fuel usage for such intricate manoeuvres. The diminishing returns observed at higher thrust levels provide critical guidance for propulsion system design, indicating an optimal thrust range for such missions.

4 CONCLUSIONS

This study of the Earth-ejection manoeuvre for the Arcanum mission to Neptune, which acts as a case study for the capabilities of advanced launch vehicles like SpaceX's Starship, has resulted in significant insights into the challenges and solutions for large-scale interplanetary missions. Our in-depth analysis, utilising the General Mission Analysis Tool (GMAT) integrated with MATLAB, has led us to pivot from initial hypergolic Earth Departure Stage (EDS) designs to more appropriate cryogenic propulsion systems. This shift was driven by the need for higher thrust and efficiency to manage the substantial mass of the Arcanum spacecraft and to overcome limitations encountered with hypergolic systems, particularly when considering the Oberth effect.

Through numerical simulations, we have demonstrated the robustness of cryogenic propulsion in maintaining fuel efficiency across a wide range of specific impulse and thrust values. This adaptability makes cryogenic systems a preferred choice for such missions, ensuring a balance between fuel efficiency and necessary thrust for Earth departure manoeuvres. The study also explored hypergolic configurations using a two-orbit 'pump-up' strategy, revealing the importance of thrust calibration in optimising fuel usage for complex trajectories. However, the higher efficiency and simplicity of single-burn cryogenic configurations emerged as more favourable for the Arcanum mission.

Furthermore, our methodological advancements have broader implications for future deep space missions. The integration of GMAT with MATLAB for trajectory optimisation and the development of a versatile EDS adaptable to various mission profiles set a new standard in space mission design. It allows for easily reachable first order mass estimates which are accurate when the rocket equation becomes inaccurate.

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APPENDIX MATLAB WORKFLOW

MATLAB System Composer was chosen as a MBSE tool not least because of the integration of git as a source control system. The structure of the Arcanum mission and its various components were modelled in System Composer and MATLAB itself executed scripts for calculation. The git remote repository was implemented in GitHub.

We followed the common practice of maintaining one master branch, where commits are not permitted. Any changes were done inside individual branches that each team member created for a specific task. The naming convention of these branches included information on the kind of task, the affected file/element and the creator of the branch. We re-merged branches, regardless of their state of completion, in regular intervals during dedicated meetings in form of pull requests. This ensured every team member can work with the latest version of the repository.

Fig. 15 shows as an example the branches during a specific "sprint" of the system development. The lilac-coloured graph in the centre represents the master branch, from which during the first meeting two individual working branches have been created. During the second meeting assigned team members introduced all changes to the rest of the team and, after approval, the branches were merged back into main. Following that a new cycle could begin.

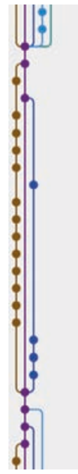

Fig.15 Exemplary commit graph of the system model in MATLAB Systems Composer.

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