Design Reference Mission Development for Nuclear Thermal Propulsion Enabled Science Missions

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Nuclear thermal propulsion (NTP) presents a distinct alternative in the advanced propulsion landscape, providing "medium" efficiencies relative to the "high" efficiency of electric propulsion and the "low" efficiency of chemical propulsion. In exchange for this medium efficiency, NTP provides high acceleration, enabling much shorter operating times than the higher efficiency alternatives, often resulting in shorter trip times. This compromise has led to NTP's frequent consideration in human missions, where crew health and logistics benefit significantly from the schedule advantages provided by NTP. However, the logistics and timeliness of science return from science missions also potentially stand to benefit from reduced schedules, as many of the high energy missions exceed a decade in trip time. Presented here are the results of analysis looking at three missions: A mission to Neptune's moon, Triton, a solar polar orbiter, and a mission to interstellar space. Comparisons to results found in literature for conventional, and future propulsion technologies are presented.

I. Introduction

Science mission concepts targeting ambitious energy states rely on some combination of sophisticated trajectory design and/or advanced in-space propulsion technologies to achieve their objectives. Due to their extremely high efficiency, electric propulsion and solar sail technologies factor prominently in many science mission concept baselines. The primary challenges associated with these mission concepts lie in the long durations over which the propulsion systems must operate, and a sensitivity to distance from the Sun. This latter challenge is mitigated for electric propulsion concepts by leveraging nuclear power rather than solar power. However, this in turn introduces an additional major technology dependency, pushing initial operating capability dates further into the future.

Nuclear Thermal Propulsion (NTP) presents a distinct alternative in the advanced propulsion landscape, providing "medium" efficiencies relative to the "high" efficiency of electric propulsion and the "low" efficiency of chemical propulsion. In exchange for this medium efficiency, NTP provides high acceleration, enabling much shorter operating times than the higher efficiency alternatives, often resulting in shorter trip times. This combination has led to NTP's frequent consideration in human missions, where crew health and logistics benefit significantly from the schedule advantages provided by NTP [1-3]. However, the logistics and timeliness of science return from science missions also potentially stand to benefit from reduced schedules, as many of the high energy missions exceed a decade in trip time.

In February 2019, the US Congress directed NASA to develop a plan for performing a flight demonstration of NTP by 2024 [4]. As part of NASA's response to this direction, the Advanced Concepts Office (ACO) at the Marshall

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Space Flight Center (MSFC) was tasked with leading a study to develop an NTP Flight Demonstration (FD) concept that would be feasible in that timeframe. To understand the traceability of the FD concept developed to a first operational NTP mission application, an additional task was created to develop several candidate Design Reference Missions (DRMs). Due to considerations such as physical scaling, cost, and risk reduction for manned systems, a science mission application is viewed as a likely first use for NTP.

II. Science Missions

Three science missions were identified as potential Design Reference Missions (DRM's) for this study; a Triton orbiter, a solar polar orbiter, and an interstellar probe. The missions differ in terms of the mission delta-V budgets, science payloads and destination environments. These varying DRMs showcase the NTP system's potential capabilities and limitations.

A. Triton Orbiter

Neptune and its moon Triton are of scientific interest for several reasons. Modeling of Triton's highly inclined retrograde orbit around Neptune shows that it is likely a captured member of a Kuiper Belt binary system [5]. Despite having a temperature of only 37K, Triton has an atmosphere, organic compounds on the surface, and evidence of an active geology [6]. Radiogenic heating and tidal dissipation in a subsurface ocean are thought to generate the heat driving Triton's activity [7].

While a Triton orbiter mission would make observations of Neptune, the goal of this mission would be to place a science payload in orbit around Triton, which would likely include a lander as well. Triton is by far Neptune's largest moon with a dynamic atmosphere and volcanic activity. A number of types of ice have been detected on its surface including organic compounds. A lander has been proposed that is capable of utilizing the nitrogen in Triton's atmosphere and surface to repeatedly hop several kilometers and refuel [8].

The baseline mission for this study, Ref. [8], used a Solar powered Electric Propulsion (SEP) system to do flybys of Earth and Jupiter to intercept Neptune. Figure 1 from Ref. [8], illustrates the following trajectory. In order to insert around Neptune, the SEP system included an aeroshell to allow for a high energy aerocapture maneuver into an elliptical orbit with a 210 day period. The system would launch in late 2029 on a Delta IV Heavy. The aerocapture occurs at the closest point to Neptune, or periapsis, and uses atmospheric drag to reduce the vehicle's energy and enter orbit. At the farthest point from Neptune, or apoapsis, the vehicle will be at minimum velocity where it is most efficient to rotate the orbital plane inclination to coincide with Triton's. At the next periapsis approach, an aerobraking maneuver will lower the apoapsis to prepare for a powered Triton flyby using a solid stage. The powered flyby will further modify the Neptune orbit to intercept Triton again and enter its orbit at a 200km altitude using a liquid propulsion system to deliver the 1,115kg payload. A 792kg Neptune orbiter separates about the same time as the aeroshell to enter a 4,000 x 488,000 km orbit around Neptune.

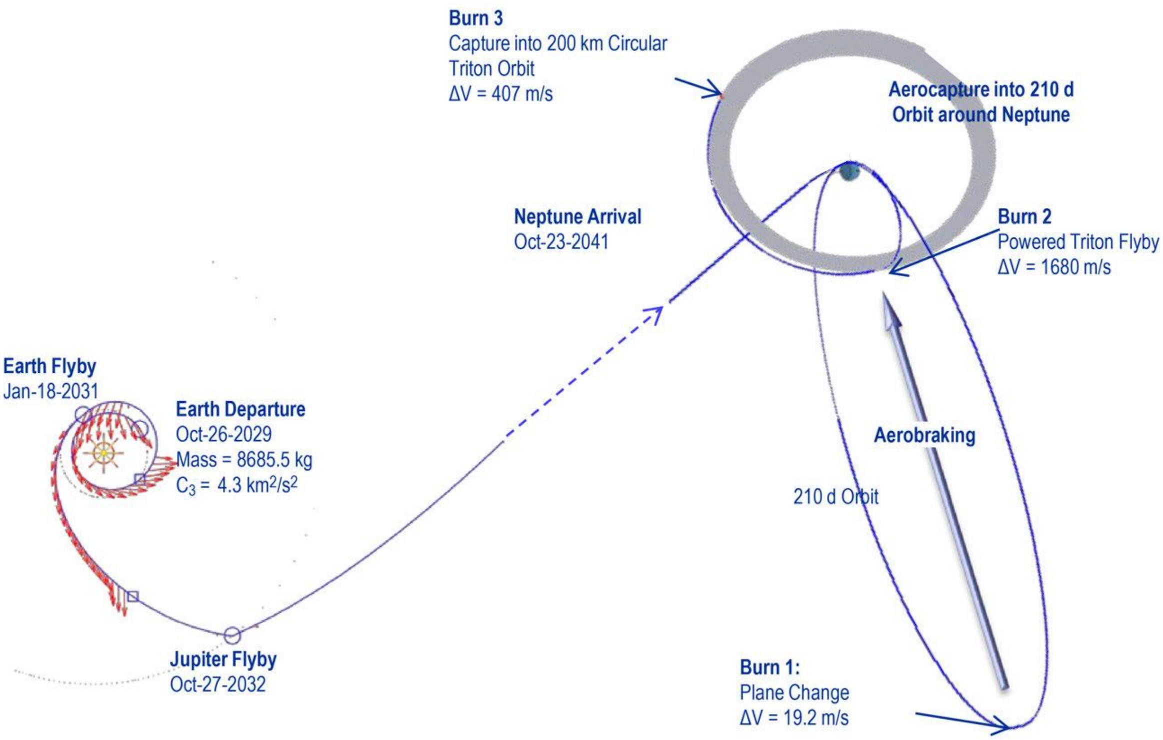

Figure 1: Triton Orbiter Baseline Trajectory [8]

For this study the NTP system will need to propel the vehicle to Neptune and capture into an orbit using the NTP stage or an optional storable chemical stage. Since Triton is in a highly inclined orbit, the location of the insertion burn is important to assure alignment with Triton's orbital plane to avoid high energy maneuvers to reorient the vehicle's orbital plane for Triton intercept. With negligible energy cost, the vehicle can alter the periapsis to Neptune, and the angle in the plane orthogonal to the incoming velocity vector, or B-plane angle. While the vehicle can change these two values to ensure the periapsis point is in the orbital plane of Triton for typical approach conditions, the orbital plane of the vehicle will likely be rotated relative to Triton's orbital plane requiring an inclination change maneuver. At the point of periapsis the vehicle will perform an insertion burn to reduce the vehicle's energy to capture into a Neptune orbit and not escape. To minimize the energy for both the insertion burn and the inclination change, the vehicle will insert into a highly elliptical orbit. At apoapsis the vehicle will burn to change its inclination and periapsis to intercept Triton. Finally, the vehicle will directly insert into an orbit around Triton at the periapsis of the Neptune orbit, taking advantage of Triton's orbital velocity. This study will try to maximize the payload mass delivered into this orbit.

B. Solar Polar Orbiter

The Solar Polar Orbiter mission will deliver a science payload to a polar, heliocentric circular orbit. From this orbit, the payload would provide a range of solar remote sensing and plasma environment data with an orbital period that facilitates coordinated observations with Earth based observatories. This mission would provide invaluable data on a variety of solar phenomena, such as the step-by-step evolution of solar structure/storm formation on the Sun's surface, as well as observations of coronal mass ejections and convection patterns in the polar regions [9].

The European Space Agency (ESA) has conducted studies that looked at the use of a solar sail to insert into a high inclination orbit. A solar sail does not use propellant for propulsion, but instead uses the transfer of momentum from solar photons as they are reflected off the solar sail. A reference study showed that a square solar sail with 153m sides could deliver approximately 45kg of science payload to an 83 degree inclination 0.48AU circular orbit in 6.8 years [9].

Inserting into this orbit directly is difficult for vehicles that use propellant, even with the use of gravity assist flybys of the inner planets, and requires the use of a Jupiter flyby to achieve a high inclination. This flyby will also reduce the energy of the vehicle to reach the required heliocentric distance. An additional maneuver is necessary to circularize the orbit. This study has the goal to maximize the payload mass delivered into this orbit.

C. Interstellar Probe

The interstellar probe mission involves a science payload deployed well beyond the outer planets of the solar system, into the range where the InterStellar Medium (ISM) begins to appear. The composition and density of the dust and gas in the ISM is not well known, and would provide insight into the formation of stars and planets. Direct measurements of the ISM magnetic fields and radiation would also be valuable. There is also uncertainty in the size and shape of the heliosphere around the Sun as well as the transitions from the local solar environment to the ISM [10].

There are current studies on ways to achieve high heliocentric excess velocities, which is the velocity in excess of the escape velocity. Achieving a high heliocentric escape velocity is difficult to do directly and requires much less energy if the escape maneuver occurs close to a large gravitational body where it has a high velocity. This is due to what is known as the Oberth effect, where the change in the vehicle's kinetic energy is proportional to not just the velocity change squared, but to the velocity change times the initial velocity. As a result, studies often consider the use of powered Jupiter or solar flybys to escape the solar system. The following studies assume the interstellar probe has a mass comparable to the New Horizons spacecraft. A study using an SLS Block 2, a Centaur derived third stage, a STAR 48BV fourth stage, and a STAR 48BV final stage was able to achieve an excess velocity of 8.6AU/year using a Jupiter flyby. Another used the same configuration, but with a CASTOR 30XL final stage and a solar shield to allow a solar flyby to achieve ~12.6AU/year [11]. Finally, a study using a solar thermal propulsion system to directly heat hydrogen using the Sun's energy and a final SEP system was able to achieve 19.1AU/year [10]. The ACO calculated the total time to 100AU for the first and third configurations to be 12.3 years and 14.5 years respectively. There was not enough information provided for the solar approach trajectory to calculate this time for the solar flyby mission using a solid stage.

This study will consider Jupiter and solar flybys with optional solid rocket motor kick stages. A solar flyby requires the addition of a solar shield to the vehicle that is sized depending on the periapsis radius. Whereas the other two missions have a goal to deliver as much payload as possible for this study, this mission has the goal to impart as high a heliocentric excess velocity as possible to a set payload.

III. Nuclear Thermal Propulsion

An NTP engine uses a fission reactor to heat a propellant to high temperatures and expands that propellant out of a nozzle like a conventional chemical rocket to produce thrust. The fission reactor is designed to use the propellant as a coolant with flow channels through the fissile material, and typically includes a regenerative cooling loop to partially heat the propellant to drive a turbopump. Control drums at the perimeter of the reactor can absorb or reflect neutrons to control the fission rate of the reactor and the resulting thermal power output. The thermal power of the reactor will drive the maximum thrust achievable by the NTP system. The specific impulse of the NTP system is proportional to the square root of the propellant temperature over its molecular weight. A reactor's maximum temperature will be limited by the materials used in the reactor and the ability of the reactor design to mitigate thermal stress. Several NTP systems were developed and tested on the ground in the 1960s, but none have flown in space.

The goal of this study is to assess the potential use of an NTP system; however, as this system is at a low Technology Readiness Level (TRL) the configuration and performance of an operational NTP system is unknown. For the NTP system itself, the reactor temperatures will be varied to determine the sensitivity of the system to performance, which could be used to make decisions on technology investment for fuel element materials and design. Two propellants, liquid hydrogen and ammonia, will be traded. For a hydrogen fueled system, an active Cryogenic Fluid Management (CFM) system will be required to cool the fuel if a particular mission profile requires an NTP burn outside of Earth departure.

Finally, at the vehicle level, the inclusion of a chemical second stage, use of solar or a Radioisotope Thermoelectric Generator (RTG), and launch vehicle will be traded. The high energy requirements of the DRMs frequently mean that the ratio of the vehicle dry mass to the payload is large enough to allocate some of the total Earth launched mass to a lower performing chemical stage. For missions to the outer solar system solar power becomes less mass efficient when compared to an RTG to power vehicle systems. Finally, launch vehicles used to put the vehicle into an Earth orbit have different mass and volume constraints which will limit the size of the NTP system, second stage and payload.

IV. Vehicle Sizing

Several different vehicle configurations were identified as potential vehicles of interest for flying each mission. All of the vehicle configurations were chosen and sized for each mission based on the trades of interest for this study. These different trades can be seen in Table 1 and are justified below:

- NTP Propellant (liquid hydrogen vs liquid ammonia) – Hydrogen (LH2) has a much better specific impulse than ammonia (NH3), but NH3 has a much better packing density and can use smaller tanks.
- Cryogenic Fluid Management (Active vs Passive) – No cryogenic fluid management (CFM) was considered for the NH3 configurations because NH3 does not have significant boil off like LH2. With LH2, passive cooling limits mission flexibility and means that the propellant needs to be used up early, but avoids the inert mass of an active cooling system. Active cooling adds more mass to the system, but provides more mission flexibility.
- Launch Vehicle (SLS Block 2 vs New Glenn) – Using SLS can fit larger tanks and more propellant, but using a commercial launcher can provide more mission flexibility and reduce cost.
- Number of Stages (Two Stages vs Single Stage) – Using two stages allows for more optimal staging to maximize ΔV, but using a single stage allows the vehicle to take advantage of high NTP performance for all burns.
- Performance Level (High vs. Medium vs. Low) – The three different performance levels, which are reflected by three different reactor temperatures (2586 K, 2216 K, and 1876 K), represent a tradeoff between NTP technology level and vehicle performance. The three temperatures correspond to specific impulses of 900 s, 825 s, and 750 s, respectively, for a hydrogen NTP system. For the ammonia NTP system the temperatures correspond to specific impulses of 440 s, 404 s, and 367 s, respectively.

Table 1: Vehicle configurations to be modeled for each mission

Configuration Numbers	NTP Propellant	CFM	Launch Vehicle	Number of Stages	Performance Level
1 – 3	Hydrogen	Active	SLS Block 2	Two Stage	High, Medium, or Low
4 – 6	Hydrogen	Active	New Glenn	Two Stage	High, Medium, or Low
7 – 9	Hydrogen	Active	SLS Block 2	Single Stage	High, Medium, or Low
10 – 12	Hydrogen	Active	New Glenn	Single Stage	High, Medium, or Low
13 – 15	Hydrogen	Passive	SLS Block 2	Two Stage	High, Medium, or Low
16 – 18	Hydrogen	Passive	New Glenn	Two Stage	High, Medium, or Low
19 – 21	Ammonia	None	SLS Block 2	Two Stage	High, Medium, or Low
22 – 24	Ammonia	None	New Glenn	Two Stage	High, Medium, or Low
24 – 27	Ammonia	None	SLS Block 2	Single Stage	High, Medium, or Low
28 - 30	Ammonia	None	New Glenn	Single Stage	High, Medium, or Low

To size this wide range of vehicle configurations, it was necessary to use a framework which can do multi-disciplinary design and allows for a flexible vehicle definition. This was accomplished by modeling all vehicles at the subsystem level using the Dynamic Rocket Equation Tool (DYREQT). DYREQT is a multi-disciplinary analysis and optimization (MDAO) tool in Python which allows for generalized synthesis and execution of space transportation architectures [12]. In DYREQT, the stages of a space vehicle can be defined as a unique sequence of objects referred to as "elements". Each stage element can be further broken down into multiple "subelement" objects which each represent a subsystem (structures, power, tanks, etc.) that make up the stage. When setting up a DYREQT model, one can define whatever subelement objects one wants to make up an element, as well as what simulation code one wants to use to size each subelement. Any vehicle elements which the user does not want to model can be treated as a lump mass. DYREQT can also be used to simulate the mission events for sizing the vehicle.

For a given vehicle and mission architecture, DYREQT utilizes the Python package OpenMDAO to execute the code and handle all of the inputs and outputs between the different subelements, elements, and mission events. OpenMDAO handles the fixed-point iterations and finite differencing of the multi-disciplinary model [13, 14]. The optimization drivers in OpenMDAO also allow the user to define constraints and design variables such that DYREQT will produce a sized vehicle that meets all user constraints. One of DYREQT's biggest strengths is that it is very flexible in the vehicles and missions it can size and is very flexible in how the vehicle can be designed / optimized.

A. Vehicle Modeling Assumptions Common to All Missions

For this study, alterations were made to the standard library of DYREQT subsystem models and a few new subsystem models were developed in order to better reflect the NTP system. Each vehicle was either a two stage vehicle or a single stage vehicle. The single stage vehicles had the NTP stage integrated with the payload. The two stage vehicles had a separate NTP stage and a second stage which was integrated with the payload. In all vehicles the NTP stage was sized with engines, power, structures, tanks, thermal, and external radiation protection subsystems. An avionics subsystem was not sized on the NTP stage. It was assumed the payload would provide communications and sensors for the vehicle, and the NTP stage would use its RCS thrusters to actuate. For all missions, the following sizing assumptions were used for the following subsystems:

1. NTP Stage – Propulsion

The nuclear engine core, pressure vessel, nozzle, and shield were sized with level-zero physics equations from *Space Propulsion Analysis and Design* [15] and fluid properties from the National Institute of Standards and Technology Chemistry Webbook (NIST) [16]. The turbopump assembly for the engine was sized using a fit to NASA internal data.

2. NTP Stage – Power

It was assumed that the payload would provide all power to the avionics so the power subsystem was sized to provide enough power for the NTP stage thermal subsystem at peak load. Any solar arrays on the vehicle were sized from level-zero physics equations which take into account things such as degradation, distance from the Sun, and cable loss [17]. SMRTG data from JPL was used to size the number of RTGs for any vehicle using RTGs.

3. NTP Stage – Structures

The structures subsystem scales the structural mass of the stage based on the design envelope area of the vehicle and the type of structure being used [18].

4. NTP Stage – Tanks

The tanks are sized using level-zero physics calculations for determining volumes and wall thickness of pressure vessels.

5. NTP Stage – Thermal

The thermal subsystem was sized to the closest approach distance from the Sun. It was assumed that any vehicle with ammonia propellant would only use passive cooling. A vehicle with Hydrogen only used passive cooling if it was dropped immediately after its first burn (to avoid significant boil off), or else it used active and passive cooling. The passive cooling (multi-layer insulation, mass gauging, and liquid acquisition) is primarily scaled based on tank size and geometry. Any active cooling is sized to maintain zero boiloff propellant. The radiator is sized to reject the heat generated by the other subsystems in the stage.

6. NTP Stage – External Radiation Shield

The shield thickness contains a layer of Tungsten, a layer of Lithium Hydride, and a layer of Beryllium. The thickness of each shield layer was optimized to minimize the aerial density of the shield while meeting an attenuation target of 87.5% for both neutron and gamma flux from the engine.

At the full vehicle level, all vehicles were sized with a mass growth allowance of 20% and to a thrust to weight ratio of 0.35 since gravity losses were not being directly modeled when sizing.

B. Vehicle Modeling Assumptions Specific to Interstellar Mission

While all of the above modeling assumptions apply to vehicles sized for the interstellar mission, the interstellar vehicles also had some of their own unique assumptions based on the mission. Since the goal of the interstellar mission was to maximize excess velocity rather than payload mass, a size and mass had to be assumed for the payload. The payload element for the interstellar vehicles was assumed to be "New Horizons-like" with the same mass, height, and length as New Horizons. This allowed the NTP interstellar vehicle to be fairly compared to other interstellar vehicle concepts.

All of the interstellar, two stage NTP vehicles were designed to carry out a solar flyby. To handle the solar flyby, the second stage of the interstellar vehicles was modeled as a solid rocket motor with a solar shield and the payload. The solid rocket motor second stage was used for the solar flyby due to its favorable mass ratio and smaller solar shield size requirement, when compared to the NTP stage. In the single stage configurations only the ammonia, RTG power, single stage NTP vehicle was designed for a solar flyby. The hydrogen single stage vehicles were not used for a solar flyby, only a Jupiter flyby, because the very large solar shield that would be required to cool the vehicle. The additional subsystems in the interstellar second stage were sized with the following assumptions:

1. Second Stage – Solid Rocket Motor

The solid rocket motor (SRM) was only used for the two stage vehicle configurations. The propellant mass fraction and specific impulse of the motor were scaled using regressions to existing motors such as STAR 48BV and CASTOR 120. The SRM masses, dimensions, and specific impulses used in the scaling relationships were pulled from the Northrup Grumman catalog [19]. The second stage solid rocket motor was only meant to carry out the final solar flyby burn, all other burns were performed by the NTP stage before detaching (for the two stage configurations).

2. Second Stage – Solar Shield

The mass of the solar shield was scaled based on the closest distance to the Sun and the planar area facing the Sun of the payload plus the vehicle stage to which it was attached. The scaling equation was fit to data from a previous study which designed solar shields for similarly sized vehicles performing a solar flyby [11].

C. Vehicle Modeling Assumptions Specific to Triton and Solar Polar Missions

The assumptions listed in the common assumptions section apply to the Triton and solar polar designed vehicles, but these missions also had some of their own, unique vehicle modeling assumptions. For these missions, the goal was to maximize the payload mass so a "New Horizons-like" height and length were assumed for the payload while the optimizer was given control of the payload mass. The second stage for these vehicles did not need to be sized with a solar shield and was assumed to use liquid storables (mon / hydrazine) as the main propulsion system since there is no solar flyby for the triton and solar polar missions. The second stage was not modeled with any avionics or power and was assumed to use the avionics and power from the payload. Structures, tanks, and thermal for the second stage followed similar assumptions to the common NTP stage subsystem modeling assumptions. The primary, unique modeling that was done for the triton and solar polar vehicle second stages was for the propulsion system:

1. Second Stage – Propulsion

The second stage propulsion is designed assuming the stage only has RCS thrusters, but the axial RCS thrusters will be used for carrying out any maneuvers of the second stage.

D. Surrogate Modeling

Many aspects of how the vehicles were sized depended on what the trajectory looked like. The thermal subsystem, for example, was sized to the trajectory perihelion and the tanks, for example, were sized to the propellant load needed to satisfy the mission delta-Vs. So to truly optimize each architecture, it was necessary to optimize the vehicle at the same time as the trajectory rather than optimizing them separately. This means that the vehicle sizing had to be integrated into the trajectory optimization. However, the vehicle sizing often took tens of seconds to run and the trajectory optimization may call the vehicle sizing hundreds of thousands of times, which would result in huge run times. To overcome this run time issue, surrogate modeling techniques were used to model the vehicle sizing environment.

The goal of the surrogate modeling was to create surrogates which could very quickly and accurately predict the sizing of the vehicles of interest across all possible trajectories that could be generated during the optimization. To do this, first a Latin Hypercube design of experiments (DoE) was created for each vehicle configuration and mission. Each DoE spanned over a range for each relevant trajectory and vehicle sizing variable. Variables such as closest solar approach for the first stage, closest solar approach for the second stage, propellant loading, etc. were used to build the DoEs. With the DoE variables as inputs, several thousand different vehicles across each DoE space were run and sized through the DYREQT environment.

Once the vehicles were sized and the responses were collected for both a training set and a testing set, the surrogate models could be built. Due to the nonlinear nature of the response space, neural network models were generally used to train the surrogate models for the vehicle sizing. Several surrogates were trained for each configuration to generate needed responses such as vehicle length, burnout mass, and RCS propellant loading as a function of the trajectory inputs. All of the surrogate models were built to have a max training and testing error of less than 1%, though most had a max error less than 0.5%. Not only were the surrogate models very accurate at predicting the vehicle sizing, but they were also significantly faster than the original environment within the trajectory space of interest. The surrogate models built for the two stage, high performance, liquid hydrogen vehicle configuration with active cooling and sized to a New Glenn shroud for the interstellar mission, for example, could evaluate over 100,000 vehicles in less than a tenth of a second. With these surrogate models, the integrated vehicle and trajectory optimization could be performed in a reasonable amount of time without giving up any significant amount of vehicle modeling fidelity. Surrogate models were created for all of the different vehicle configurations for each mission and were then integrated into the trajectory optimization environment.

V. Trajectory Analysis

The high energy requirements of the selected science missions are difficult to achieve with a direct transfer from Earth. By including one or more flybys, or gravity assists, of planetary bodies, the energy requirements can be reduced. A passive flyby will change the vehicle's direction and can either increase or decrease its heliocentric energy depending on the relative velocity, flyby distance, the body's gravitational parameter, and desired outgoing velocity vector. The change in vehicle trajectory needed to target a specific periapsis point is relatively small and neglected for this study. If the desired outgoing velocity vector cannot be achieved by altering the flyby parameters, the vehicle can do a propulsive maneuver to further alter the trajectory. When a change in energy is required, the maneuver should be performed at the flyby periapsis to maximize the benefit of the Oberth effect. If a change in direction is required to target the next encounter, it may be desirable to do a deep space maneuver at a lower velocity to minimize the delta v required. Deep space maneuvers are not considered in this study as they would add significant computational effort, and their exclusion is a conservative approximation.

For this study, it was required to look at a wide range of flyby sequences and vehicle types for each DRM. To evaluate a large number cases efficiently, a patched conic approximation was selected. A patched conic method reduces an n-body problem, where the gravitational effects of all planetary bodies on the vehicle are modeled, to a two body problem by only considering the effects of a body when the vehicle is in that body's sphere of influence. The sphere of influence is the region around a body where the effects of its gravity are dominant. So the trajectories between planets are modeled as heliocentric, only considering the Sun, and flybys only consider the flyby body. Interplanetary transfers are modeled as a Lambert problem, where the start and end coordinates of the transfer are set as well as the time of flight between those points. The user or optimizer picks times when the flybys will occur and those times are used with the NASA JPL SPICE toolkit to find the heliocentric coordinates and velocities of the bodies [20]. This study uses the Lambert solver in the pykep toolbox, an open source library made by the European Space Agency (ESA), which finds the starting and ending velocities the spacecraft should have for a minimum energy trajectory [21]. The Lambert solver is capable of finding solutions that complete more than one orbit, and the solver is capable of searching this space for the minimum total energy. These velocities are not influenced by the flyby bodies and when transformed into the bodies' coordinates the velocities are used as the velocity outside the bodies' sphere of influence, known as v-infinity.

Once the incoming and outgoing v-infinity for each flyby are known, Newton's method is used to iterate on the periapsis radius until the incoming and outgoing turning angles are equal to the angle between the velocity vectors. A maneuver is included at periapsis, if needed, to change the magnitude of the outgoing velocity vector. If the calculated periapsis distance is too low where there is a danger of impacting the body or its atmosphere, the periapsis is set to the minimum and the maneuver is modified to make up for the resulting decrease in turning angle.

In addition to planetary flybys, the interstellar DRM can include a solar flyby. It requires a non-negligible change in energy to target the center of the Sun instead of a close solar flyby so a point is generated at the desired flyby distance. The location of this point is dependent on the energy after the previous flyby and would need to be determined by iteration, but for the expected energies it is reasonable, and less computationally expensive, to approximate the point rotated by 180° from the previous point. This point is shifted slightly as the Lambert solver is unstable for exact 180° transfers. As opposed to the previously described flyby procedure, the final interstellar flyby uses what is known as B-Plane targeting to determine the outgoing velocity vector. The B-Plane is the plane orthogonal to the incoming v-infinity and passes through the center of the flyby body, and the B-Plane angle is between a reference, usually the ecliptic, and the point where the vehicle will pass through the B-Plane. The inputs for this type of flyby are the B-Plane angle, periapsis distance, and the incoming velocity vector. For this analysis, the angle and distance are provided by the user or optimizer, and the incoming velocity vector is an output of the Lambert solver. The outgoing velocity vector is then modified by a periapsis maneuver where the remaining vehicle propulsive energy is expended to maximize the heliocentric escape velocity. The final velocity direction can be calculated by finding the new turning angle and rotating the velocity vector in the trajectory plane.

The solar polar DRM uses B-Plane targeting for the final planetary flyby which will vary the resulting inclination of the vehicle. A Jupiter flyby is sufficient to produce a polar orbit, but additional heliocentric maneuvers are required to circularize the orbit. Explicit coplanar impulsive relations are used to model these heliocentric maneuvers. After the Jupiter flyby the heliocentric orbit will have an apoapsis at Jupiter's distance. Additional flybys at Earth or Venus can be used to lower the apoapsis, but relative velocity for these flybys get larger as inclination increases, which makes the maximum turning angle small. The small turning angle means that each flyby can only change the heliocentric velocity by a small percentage, requiring many flybys to lower the apoapsis to the desired radius. Furthermore, each flyby must produce an orbit that has a period that is an integer ratio of the flyby body in order to flyby that body again, which may require an additional propulsive maneuver.

For the Triton DRM, the incoming velocity vector to Neptune is found by the Lambert solver for the input intercept time. The position and velocity of Triton is found using the SPICE library, which are used to determine the angle between the v-infinity and Triton's orbital plane. For the v-infinity and minimum Neptune periapsis distance the turning angle to periapsis can be found, and decreased as needed by targeting a larger periapsis. If the turning angle is small enough relative to the angle between Triton's orbital plane and v-infinity, there exists a B-Plane angle such that the periapsis point, and thus the node line between periapsis and apoapsis of the captured orbit, will be in Triton's orbital plane. The vehicle will do a propulsive maneuver at periapsis to capture around Neptune and target a long period orbit to minimize the velocity at apogee. At apogee another maneuver will change the orbit inclination to be co-planar with Triton and also change the perigee to intercept. It is assumed that small changes to the previous orbital periods have been done to assure Triton will be at the perigee point at the same time as the spacecraft so that a final maneuver can decrease the vehicle's energy to capture into an orbit around Triton. The above calculations use explicit impulsive orbital maneuver equations.

VI. Optimization

The ESA Pagmo/Pygmo library provides a unified interface to a large selection of optimization algorithms that can be executed in parallel. This study used this library to explore different optimization algorithms, minimize development time, and run large parallel studies. The ability to search a large design space is necessary as there are a large number of input variables that preclude an attempt at manual optimization. Each DRM trajectory analysis requires a sequence of flyby bodies, input times for each flyby, and final flyby parameters for the interstellar case. Each vehicle sizing function type is run separately and requires the energy provided by the launch vehicle, propellant loading, and reactor temperature in addition to the values provided by the trajectory to size the vehicle.

For this study the vehicle and trajectory analysis functions were integrated together such that the optimizer can use them to produce the necessary outputs to find optimal combinations. For the Triton and solar polar DRMs, the optimizer attempts to maximize the delivered payload. The interstellar mission assumes a New Horizons class payload and the optimizer searches for the maximum heliocentric escape velocity. The optimal value is subject to constraints to ensure the resulting vehicle is a valid point. These constraints are the launch vehicle maximum delivered mass to the input energy, the launch vehicle fairing volume, and the delta-v of the NTP vehicle stage(s). The launch vehicle constraints ensure the vehicle can lifted to the desired orbit, and the delta-v of the vehicle stage must be greater than or equal to the delta-v requirement found by the trajectory analysis. Finally, for a passively cooled vehicle with a solid second stage there is an additional constraint as any intermediate flybys, if any, must have a negligible delta-v requirement.

Through trial and error, the Extended Ant Colony Optimization algorithm was found to consistently find optimal points with the shortest computation time of the single objective constrained algorithms available. Optimizations were run in parallel with individual computational threads exchanging their best solutions in between evolution steps with neighboring threads in a ring topology. This optimization step will attempt to find the optimal global region, but often does not find the optimal point so a local optimization algorithm, Sequential Least-Squares Quadratic Programming (SLSQP), is used to refine the best solutions. The resulting points are stored and the next vehicle configuration and/or flyby sequence is analyzed. For a given flyby sequence, the program can skip a number of subsequent points that are known to be lower performing if the current point could not find a solution that met the problem constraints. For example, if the optimization could not find a point for a high NTP temperature configuration then it should not find a point for a lower NTP temperature configuration.

VII. Results

A. Triton Orbiter

The Advanced Concepts Office (ACO) at NASA-Marshall Space Flight Center studied NTP-powered missions to Triton, considering many different planetary flyby, launch vehicle, and propellant combinations. The objective was to find the maximum payload mass delivered to the same 200km circular orbit around Triton with a baseline-matching 12 year trip time that departed in the same timeframe, but did not employ aerocapture/aerobraking. Payload to Triton with and without delivery of the Neptune orbiter described in Ref. 10 was calculated. A two stage vehicle was modeled with a single engine NTP first stage. Both liquid hydrogen and ammonia were considered. The second stage was a storable chemical propellant stage with an assumed I_{sp} of 320s and an RTG electrical power system. Propellant load and the split between the stages was determined by the optimizer and limited by the launch vehicle payload volume of an SLS Block 2 (SLS 2B) with an 8.4m long shroud or a New Glenn (NG).

Table 2: Neptune/Triton NTP-powered mission study results for multiple launch vehicle and propellant options

Propellant/ Launch Vehicle	1st Stage ΔV [km/s]	Capture Stg ΔV [km/s]	1st Stage Prop [kg]	1st Stage Burn Out Mass [kg]	Capture Stg Prop [kg]	Capture Stg Burn Out Mass [kg]	**Aero-capture / brake**	**Payload to Triton Orbit [kg]**	**Triton Payload÷ Baseline**	**Triton Payload w/o Orbiter [kg]**
LH_2 / SLS 2B	5.14	1.68	28,090	17,090	7,650	2,180	**No**	**8,000**	**7.17x**	**8,620**
NH_3 / SLS 2B	5.03	2.28	40,620	5,730	6,560	1,950	**No**	**3,530**	**3.17x**	**4,190**
LH_2 / NG	5.63	2.53	16,950	11,330	4,220	1,410	**No**	**1,350**	**1.21x**	**1,980**
Baseline/ Delta 4 Heavy	7.01	2.11	2,610 Xe	1,600	1,233	NA	**Yes**	**1,115**	**1.0x**	**NA**

While there were hundreds of solutions for this mission, the three shown in Table 2 arrive in a Neptune orbit in ~12 years, launch in the early 2030s, and provide a payload mass that exceeds the option in Ref. [10]. The three solutions all had one powered flyby of Jupiter using the NTP 1st stage, which was jettisoned prior to arrival at Neptune. The capture ΔV includes insertion around Neptune into a transfer orbit, an inclination change at apoapsis, and insertion into Triton orbit at the aligned transfer orbit periapsis. The NTP delivered payload could enable more capable or multiple landers and/or orbiters. While these solutions maximize payload, the optimizer could also reduce total mission time while holding the payload constant if shorter mission times are desirable.

Other solutions showed higher delivered payloads of up to 10,000kg, but at the expense of significantly higher mission times of 18-20 years. These solutions included the addition of an Earth flyby, a Saturn flyby prior to Jupiter, or a Mars flyby. The most favorable solutions were still Earth, Jupiter, Neptune sequences, but with longer flight times which typically lowered the energy required for capture into an orbit around Neptune.

B. Solar Polar Orbiter

The high energy requirement to change the orbital inclination around the Sun precludes the use of chemical propulsion to directly perform the maneuver without utilizing flyby maneuvers. Furthermore, the gravitational potential of the inner planets is insufficient to produce the needed inclination change without using many flybys. Of the many flyby combinations analyzed, only a flyby of Jupiter was able to partially meet mission requirements. The Earth to Jupiter missions were able to produce a high inclination as well as lower the periapsis to the required .48AU or below, however the circularization of the orbit would need to be done propulsively. This circularization burn requires a velocity change of over 15km/s, which is not practical for an NTP system. Flybys can't be used to circularize a polar orbit as the velocity of the planets are nearly perpendicular to a polar orbit which means that any energy change to the orbit will be nearly zero. For this study, the only mission profile found that could produce a circular polar orbit with energy requirements practical for an NTP system is to use a Jupiter flyby to target a high inclination, but not polar, orbit targeting a Venus flyby. A single Venus flyby can't circularize the orbit or result in a polar orbit. Instead, the Venus flyby can produce a small change in apoapsis, an increase in inclination, and target an orbital period of some integer multiple of Venus's orbital period. This process was carried out for different multiples of Venus's orbital period for each flyby until the orbit was polar, and with a periapsis at .48AU. When the flyby could not produce an orbit with the correct period, a propulsive maneuver was used during the flyby to raise or lower the opposing node as needed. The best solutions found for different flight times are shown in Figure 2. The energy requirements do not include a 4150 m/s velocity change to circularize the orbit after the final Venus flyby. While these energy changes are achievable with an NTP system, the flight times make such a mission unattractive compared to the solar sail baseline mission profile of 6.8 years.

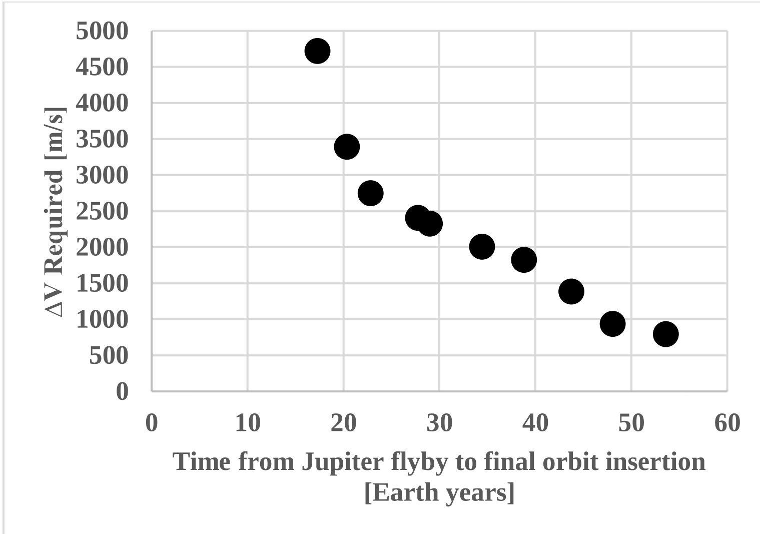

Figure 2: Mission Time vs. Energy Requirement

With the multiple Venus flyby mission showing unacceptably long mission times, an additional mission profile was examined that would look at the characteristics of a range of polar elliptical orbits. This analysis would utilize a Jupiter flyby to reach a polar inclination and lower the periapsis to a range of values below .48AU. These orbits would result in higher heat loads on the spacecraft, but lower energy requirements. Additionally, the observation time spent over the Sun's poles would be vary so the time spent over a 60° sweep at the poles was calculated and compared to the baseline .48AU circular orbit. Figure 3 shows the relationship between observation time, energy requirement, and heat load to periapsis.

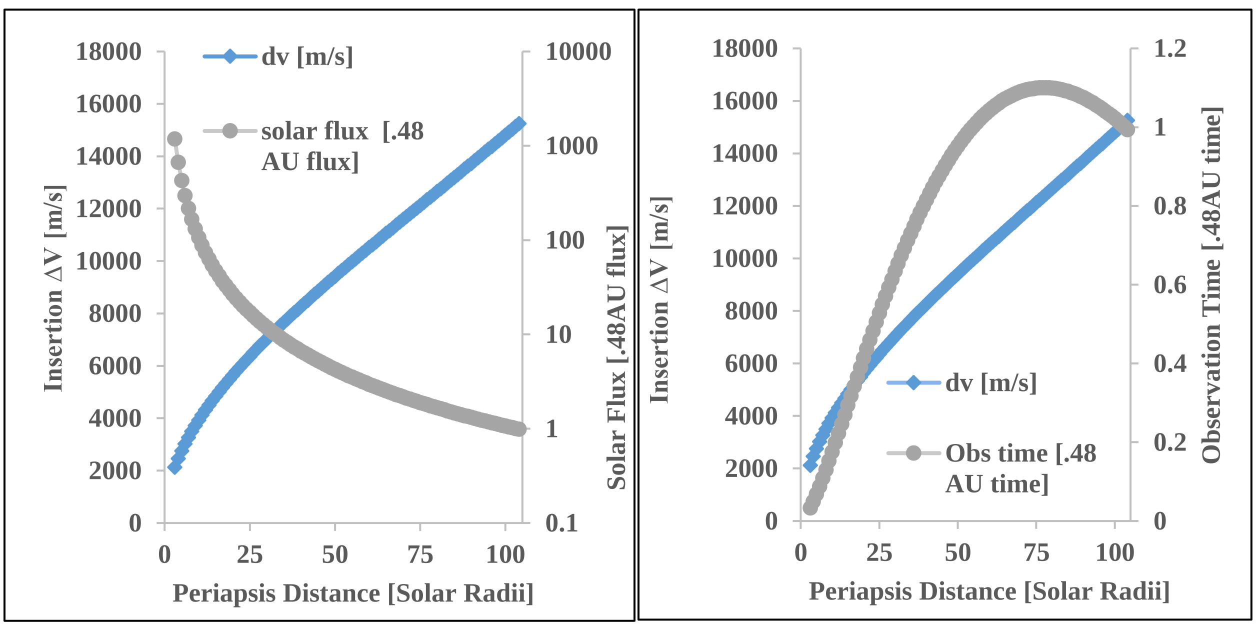

Figure 3: Energy, Heat Load (left), and Observation Time (right) vs Periapsis Distance for Solar Polar Orbiter

An NTP system would be capable of achieving the lower energy orbits, but these dramatically increase the heat loads on the spacecraft and result in significantly lower observation times over the poles. Since the above options presented in this section have significant downsides compared to the baseline mission, further analysis utilizing integrated vehicle sizing were not conducted.

C. Interstellar Probe

A MSFC ACO Interstellar probe study considered an NTP-powered spacecraft using Jupiter and solar flybys with optional solid propellant kick stages. A solar flyby requires the addition of a solar shield to the vehicle that is scaled depending on the periapsis radius, while the final stage is sized based on the work in Ref. [11]. Table 3 shows a summary of selected mission results for different mission and NTP options. The NTP 1st stage is used for propulsive maneuvers prior to the final flyby. If no propulsion is required between the Earth departure and final flyby maneuvers, a passive cryogenic fluid management (CFM) system is used to reduce 1st stage burn out mass. Should mission ground rules still require an active CFM system for a single Earth departure burn, the reduction in escape velocity was found to be approximately 6% less than the passive CFM escape velocity.

Table 3: Interstellar Probe NTP-powered mission selected results for multiple launch vehicle and propellant options

Vehicle	Flyby Sequence	Escape V [AU/y]	Years to 100AU	Departure Date	Escape Man. Date	CFM Req.	Case Notes
LH_2 / SLS 2B	E,V,V,E,J,Sun	**14.8**	**16.4**	9/2/2038	5/27/2048	N	**Max Escape V**
LH_2 / SLS 2B	E,J,Sun	**13.9**	**10.1**	11/27/2039	11/20/2042	N	**Soonest to ISM**
NH3 / SLS 2B	E,M,E,J,Sun	**14.1**	**15.5**	2/16/2039	7/30/2047	N	NH_3 Max V
LH_2 / NG	E,V,V,E,Sun	**13.3**	**16.3**	2/4/2031	1/4/2040	Y	Comm. Max V
LH_2 / SLS 2B	E,Jupiter	**9.1**	**11.7**	3/3/2031	2/2/2032	N	No Solar Shield
Chem. Baseline	E,Jupiter	**8.6**	**est. 12.3**	9/2/2038	5/27/2048	N	No Solar Shield
Chem. Baseline	E,J,Sun	**12.6**	**---**	~7/2034	Unknown	N	
STP Baseline	E,J,Sun	**19.1**	**est. 14.5**	2/16/2036	5/25/2045	Y	**Max non-NTP V**

While the second option given in Table 3 (LH_2, SLS launch, solar flyby) did not produce the highest escape velocity, it is likely the most desirable as it shows the lowest mission duration to 100AU for a slightly-reduced final escape velocity. Depending on mission objectives, this may be a desirable trade. The results show that an NTP-powered solar flyby mission to the ISM can reach 100AU faster and have a higher final escape velocity relative to comparable chemical propulsion missions in the literature. While a solar thermal propulsion system delivers the maximum escape velocity for an interstellar mission, such a system has limited utility for missions within the solar system. Additionally, the solar thermal propulsion system takes longer to reach the ISM than the second NTP option and would not overtake an NTP system launched at the same time for over 40 years at 600 AU from the Sun.

The third option shows that an ammonia fueled NTP system is potentially competitive despite the lower specific impulse. This is due to the lower dry mass of the ammonia system and removing the volume constraints as a concern relative to the hydrogen fueled system. The fourth option is a commercially launched, hydrogen fueled NTP system. It does not present the highest escape velocity or lowest mission time but could be the best solution if the cost or availability of an SLS is a primary concern. Finally, while the fifth option does not show the best mission time or escape velocity, the exclusion of a solar flyby could be advantageous when the development cost and programmatic risk to include a solar shield is considered.

VIII. Conclusion

The ACO's analysis shows that an NTP system can support larger payload masses to Neptune and Triton than an SEP system, and this without the need to for aerocapture. The NTP system cannot approach the performance that a solar sail can provide for a solar polar mission. The proximity to the Sun and the extreme energy requirements make the propellantless operation of a solar sail highly advantageous. An NTP system can provide significant benefits in escape velocity and/or mission time to the ISM compared to conventional propulsion systems. The ACO analysis even shows total mission time benefits over a potential solar thermal propulsion system for the launch dates analyzed. The benefits shown for the Triton and ISM missions indicate that NTP could also show advantages to other outer planets, and their moons.

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