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Space Nuclear Propulsion for Space Applications: Multi-Mission Nuclear Thermal Propulsion Vehicle Design

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Executive Summary

Overview: Objectives, Challenges, and Results

This NASA Technical Memorandum (TM) is intended to capture design work that the Advanced Concepts Office (ACO) at NASA's Marshall Space Flight Center (MSFC) performed on behalf of the Space Nuclear Propulsion (SNP) Project during an SNP-driven design and mission analysis initiative in FY2025. Specifically, this TM focuses on several spacecraft design concepts intended to explore the feasibility of nuclear thermal propulsion (NTP) for use in a selection of notional, unmanned space missions. While references to the missions will be made in this TM, the primary focus will be on the spacecraft design concepts themselves, the ground rules and assumptions used for designing the spacecraft, a description of the various subsystems and their performance parameters, and the methodologies used to design them. The TM will also highlight areas of future work and additional considerations that may need to be taken into account when evaluating the feasibility of NTP space systems for other applications.

The report will discuss the results from ACO's analysis as they pertain to the different vehicle concepts that were evaluated. Each discipline section in this report will discuss the discipline-specific ground rules and assumptions, the methodology and design approaches employed, and present the results along with any relevant discussion, and highlight areas of future work where applicable.

Background

At the start of FY2025, the SNP project office sought to evaluate the feasibility and performance of nuclear-enabled spacecraft concepts in accomplishing a variety of unmanned space missions. The spacecraft concepts focused primarily on NTP and nuclear electric propulsion (NEP) and the selected missions included a wide range of unmanned concepts, ranging from deep space science missions to planetary defense scenarios. Three groups were tasked with evaluating these systems and missions: MSFC's ACO, the COMPASS team at NASA's Glenn Research Center (GRC), and the Analytical Mechanics Associates (AMA) contractor team.

To this end, ACO was tasked with evaluating the feasibility and performance of a notional NTP system in this context. Several mission concepts were selected as a means to evaluate the designs:

- A planetary defense scenario featuring a kinetic impactor approach
- A science kickstage mission tradespace evaluation – examining feasibility/performance for a range of outer planet and inner planet destinations, using an NTP vehicle as a kickstage
- A cislunar space flight demonstrator

Both ACO and SNP project offices sought to add not only the evaluated mission and nuclear-enabled vehicle concepts to existing mission/design portfolios, but also expand the portfolio to include alternative propellant architectures. As a result, the effort included an analysis of hydrogen, methane, and ammonia-based propulsion systems that needed to be integrated with the NTP reactor core.

Lastly, reference missions and vehicle profiles were used as a comparison point to quantify where NTP-enabled systems would either enhance existing reference mission concepts, or enable new mission concepts that were previously infeasible with heritage or conventional approaches.

Objectives

The objectives of this effort can be summarized as such:

- Evaluate feasibility of NTP for selected science/robotic missions
- Expand the NTP concept portfolio with NTP vehicles that include different propellant architectures
- Where applicable, compare NTP performance to conventional solutions

Key results and products from this effort included the following:

- Design data on multiple NTP vehicle concepts, including performance, mass, sizing and power data
- Performance data of the NTP vehicle designs for the multiple mission concepts evaluated in this effort
- Relevant datasheets, reports, and presentation packages as requested by SNP project office

Major Ground Rules and Assumptions (GR&A)

Despite the various vehicle concepts generated as a result of this effort, as well as their mission-specific variants, there are several underlying ground rules and assumptions that governed all of the vehicle designs, regardless of their configuration or mission application.

- ***Single-launch architecture –*** each vehicle concept is assumed to be deployed for each mission evaluated, on a single launch. Thus, there are no assumptions of element aggregation, rendezvous, or proximity operations involving multiple, separately-launched mission-critical elements. This has implications regarding the performance limits of an NTP vehicle for a given mission and provides important context for not only how the performance of an NTP vehicle compares to non-nuclear reference cases, but also scenarios that hit the limits of the performance of a single-launch NTP vehicle.
- ***"Plug and play" mission kit approach*** – certain subsystems, such as avionics/comms subsystems and CFM subsystems, are adjusted depending on the mission or the propellant architecture. As a result, the baseline NTP vehicle for a given propellant architecture is "scarred" for the maximum mass/power load case, with components removed according to the mission for which the vehicle's performance is being evaluated. Other supporting subsystems may be scaled accordingly based upon the resulting changes in mass/power requirements.
- ***Shared reactor concepts*** – while propellant architectures and changes in thrust class may affect reactor sizing, the underlying reactor systems were assumed to be of a common design configuration. The SNP project office was responsible for providing the reactor design as well as supporting data on adjustments from the baseline design to accommodate the various propellant architectures.
- ***Shared launch vehicle tradespace*** – to maintain some consistency on launch vehicle performance, the NTP stages were sized to fit in a selection of launch vehicles, regardless of mission profile. This meant that NTP spacecraft were not designed to fit a notional "custom-sized" launch vehicle and thus were constrained by the performance of the given launch vehicle, ranging from limits on supportable fairing volume, to limits on overall mission performance. Where applicable and where data was available, expendable launch configurations were assumed in order to maximize launch vehicle performance.

Approach

The designs that are documented in this report were developed at a pre-phase A concept level of fidelity. Thus, the results are generated using computational modeling, historical performance data, and physics-based extrapolations of performance data to address design areas where the technology readiness level (TRL) is low or lacking in observational/flight data.

Due to the level of overlap in vehicle design, many of the subsystems were similar in type, if not in scale, for each of the NTP configurations. Additional component scaling was accommodated based on the mission in which the vehicle configuration was being evaluated. For example, if a particular mission obviated the need for a long-duration, active cryogenic fluid management (CFM) system, then in addition to the removal of the CFM cryocoolers and other active components, the power systems – namely, the solar arrays – would be rescaled to account for the reduction in power needs. Thus, the primary differences in NTP vehicle configurations can be classified into two areas:

- Mass/size scaling: resulting from mission requirements, or launch vehicle volume/performance constraints
- Component changes/removals: resulting from mission requirements

As a result of the launch vehicle, propellant architecture, and mission configuration options, the tradespace for a given NTP vehicle could become considerable in size. Additionally, some level of rapid, low-fidelity modeling/performance prediction needed to be completed across the tradespace to ascertain the configurations that would be candidates for a more detailed analysis. As a result, the ACO approach to this effort was split into two periods:

1) A "Level 0" analysis period, where the tradespace was bounded and relevant configurations were given a low-fidelity evaluation
2) A more detailed analysis period, consistent with standard ACO practices for evaluating and designing vehicle concepts, where specific point designs from the Level 0 analysis were evaluated in greater fidelity.

The Level 0 analysis initially evaluated fifteen NTP configurational groups. This accounted for three propellant architectures (hydrogen, methane, ammonia) and five launch vehicles (SLS Block 2, SpaceX Starship, SpaceX Falcon Heavy, Blue Origin New Glenn, ULA Vulcan). Within each of these groups, variants were evaluated according to a conservative, "worst-case" configuration that accounted for the most penalizing mission configurations in terms of mass and power requirements. This "worst-case" baseline would then be scarred accordingly, with components removed/scaled down based on mission variations that were being evaluated. To accomplish this Level 0 analysis, subsystem discipline experts generated and used low-fidelity models based on historical NTP performance data, empirical models and physics-based equations where applicable.

While the Level 0 analysis is not the primary focus of this report, it was important to highlight its critical role in establishing the candidate configurations that would then be evaluated in greater detail.

Results, Recommendations and Next Steps

As a result of the study approach outlined in the previous section, three NTP configurations were highlighted and evaluated in greater detail:

1) Liquid hydrogen configuration launched on an SLS B2
2) Liquid methane configuration launched on a SpaceX Starship
3) Liquid ammonia configuration launched on a SpaceX Falcon Heavy

The following discipline sections will cover the results of the detailed analysis for these three configurations, accounting for overlap where applicable. However, it is worth noting important findings obtained in the course of this effort:

- While CFM systems still play an important role in dictating power requirements for the overall vehicle, avionics/comms components can also have a significant impact. This has implications for deep space missions where data bandwidths and long-distance communications have increasingly larger impacts on avionics performance requirements.
- Even though ammonia and methane propellant architectures have lower specific impulses compared to the hydrogen architecture, they have important benefits in comparison, including the reduced need for CFM systems (the ammonia configuration has no active CFM requirements and uses only heaters), and a much higher propellant density.
- Thrust levels were higher for ammonia and methane configurations compared to the hydrogen configurations, and their specific impulses were on par, if not better than, most current chemical propulsion systems. This suggests potential benefits for maneuvering in planetary gravity wells to minimize gravity losses, while still maintaining, if not improving, over the specific impulses provided by conventional chemical options.
- The biggest penalty that affects all configurations is the high dry mass (or, low thrust-to-weight ratio) of the system. While ammonia and methane configurations can potentially overcome the thrust-to-weight limitations by increasing the propellant flow rates and thus the engine thrust at higher levels compared to an equivalent sizing of a liquid hydrogen system, they face a tradeoff in a reduced specific impulse. Ultimately, the reactor, reactor shielding, and reactor structural support components and their masses are orders of magnitude greater than that of a comparable chemical propulsion thrust stage, and methods of reducing their mass need to be developed in order to improve their competitiveness in the single-launch mission architectures. It is worth noting, however, that much of the mass penalty being considered here is based on conservative estimates derived from flight demonstrator assessments. Next-generation reactor concepts being considered will likely drive down that dry mass further, helping reduce the penalty associated with the system.
- The single-launch architecture assumption has several benefits in terms of reducing launch and mission complexity, but at the cost of overall flight performance, due to the inability of the higher specific impulse of the NTP stage overcoming the high dry mass penalty (due to propellant volume constraints in a single launch). A multi-launch architecture has been explored recently by industry initiatives and could be considered here, as it could have significant improvements in terms of mission payload/transit time performance. The higher specific impulse of the NTP stage combined with a larger propellant load would overcome the penalties incurred by the higher dry mass. Such a system would also support multi-ton payloads to a variety of interplanetary destinations, including the outer planets.

Recommendations for follow-up work in this effort would involve focusing on evaluating the other configurations in the Level 0 tradespace in greater detail. Other launch vehicles and propellant/reactor architectures can also be evaluated in a similar Level 0 fashion, and used to expand the tradespace – this would include alternative hydrocarbon propellant architectures, and potentially other reactor configurations. Additionally, the mission analysis that was conducted for specific cases could be expanded to evaluate alternative trajectories, and further optimization can be recommended to focus on specific objectives such as minimizing transit time or maximizing delivered payload. Lastly, multi-launch architectures should also be evaluated in greater detail, especially for unmanned science missions that would seek to deliver much larger, multi-ton payloads across one or more destinations.

Discipline Results

Configuration

Overview

The initial objective for the configuration of the various NTP stages was to determine whether each propellant selection would be mass limited by the launch vehicle capability or be volume limited inside each vehicle's launch shroud. To perform this determination each case was sized to the maximum volume/mass for fitment. Whether the case was volume-limited or mass-limited was a function of the given mission and vehicle under consideration. As this report focuses on the vehicle design, the mission details will be reported in subsequent publications. For clarity, however, it can be pointed out that for the missions considered, the hydrogen SLS case and methane Starship cases were determined to be volume limited while the ammonia Falcon Heavy case was mass constrained. Due to thermal concerns, the NTP engine is not mounted to the tank dome as is normally done for an in-space stage. It was desired to have the thrust/mounting structure attached by trusses. These trusses connect to an upper cone, which holds the motor attachment struts. A cone was used because it was determined to have better structural rigidity than a truss design.

Key Trades

Launching the engine 'down' was evaluated and an associated propellant load determined. To maximize the propellant load, the single NTP engine was determined to be best launched in the 'engine up' position. This allowed for the engine to recess up into the shroud conical area while allowing the tank barrel length to be longer. This orientation is not standard but enabled greater propellant loads.

Ground Rules and Assumptions

Each propellant/launch vehicle case utilized the same common NTP engine placeholder design.

Key Analysis Details

Figure 1 shows how the LH2/SLS stages are configured in the launch stowed configuration. The solar arrays are located on the front truss structure/assembly and fold so that they rest on the forward structure and are supported by the radiator support plate. The view also shows the nozzle extension retracted, which gives more room in the launch vehicle shroud. The multilayer insulation (MLI) around the tank appears to have either a golden or silver outer layer depending on the rendering in question, which was done for artistic purposes. A silver outer layer is used to achieve a lower outer layer temperature and minimize heat leak into the tank. MLI also covers the forward and aft skirts.

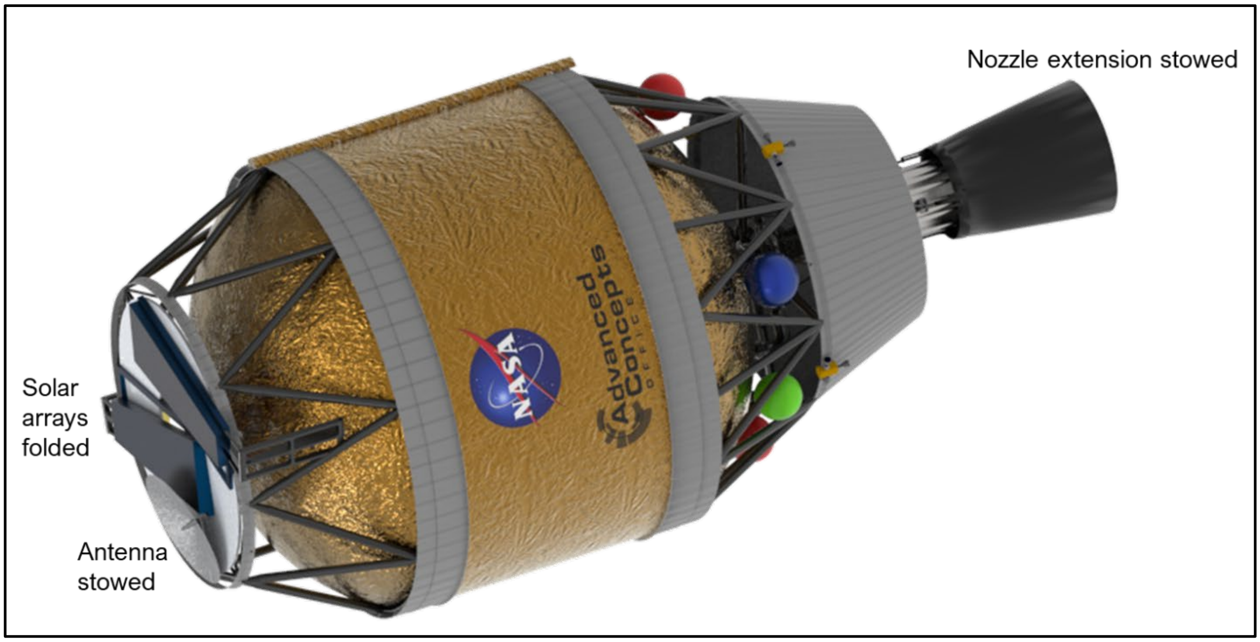

Figure 1. Stowed launch configuration

Figure 2 shows how the stages are configured in the deployed state. The deployed solar arrays are located on the front truss structure/assembly. The radiator and avionics are also located on this structure. The cryocoolers are mounted in this area as well. The aft truss structure is where the Reaction Control System (RCS) and Main Propellant System (MPS) pressurization tanks are located. The view also shows the nozzle extension deployed.

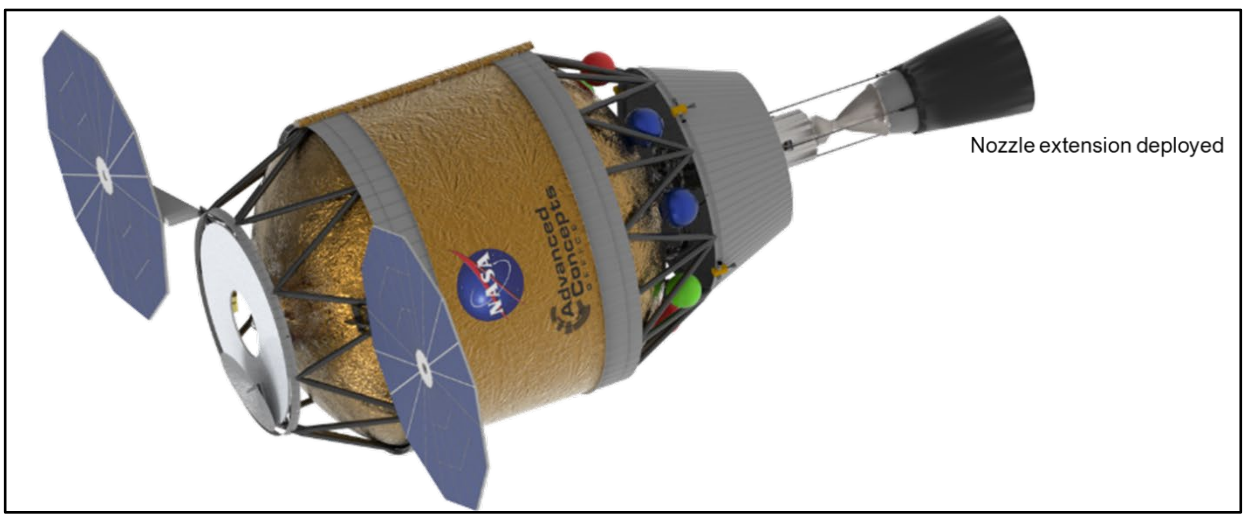

Figure 2. Stages configured in deployed state

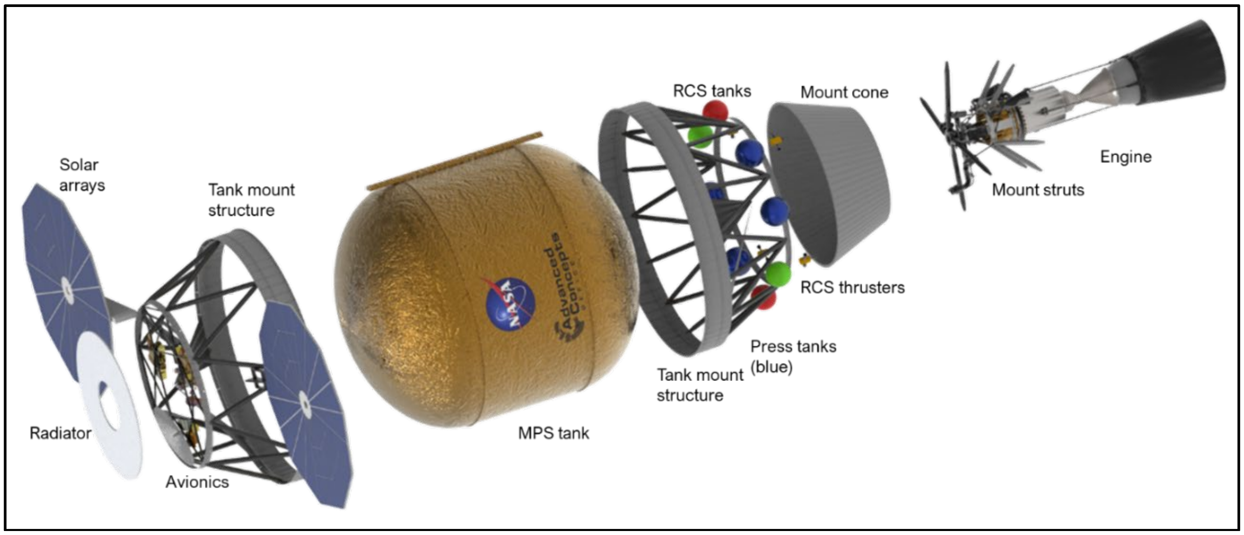

Figure 3. Exploded assembly view of the major components and assemblies

Figure 4 shows more detailed views of how the stages are configured in the forward and aft truss assemblies. The stowed solar arrays are located on the front truss structure/assembly. The radiator and avionics are also located on this structure. The cryocoolers are mounted in this area as well. The RCS system and MPS pressurization tanks are located on the aft truss structure assembly. The view also shows the nozzle extension deployed.

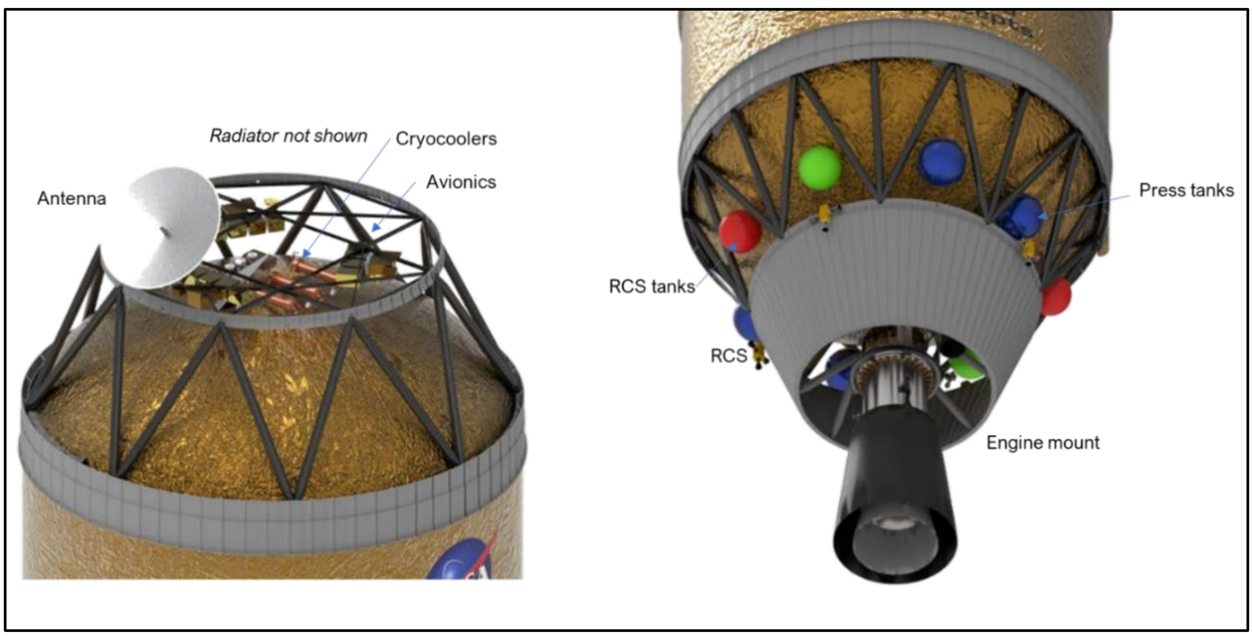

Figure 4. Detail view of stage configuration in forward and aft truss assemblies

Figure 5 shows the details of the NTP engine mounting design. This design approach was utilized for each design case. Due to thermal concerns the NTP motor is not mounted to the tank dome as is normally done for an in-space stage. It was desired to have the thrust/mounting structure attached by trusses. It would be required to redesign the motor mounting attachment.

A simple gimbal ring would be utilized which is common on many motor designs. These mounting trusses connect to an upper cone, which hold the motor attachment struts. A cone was utilized when it was determined to have better structural rigidity that utilizing a truss design.

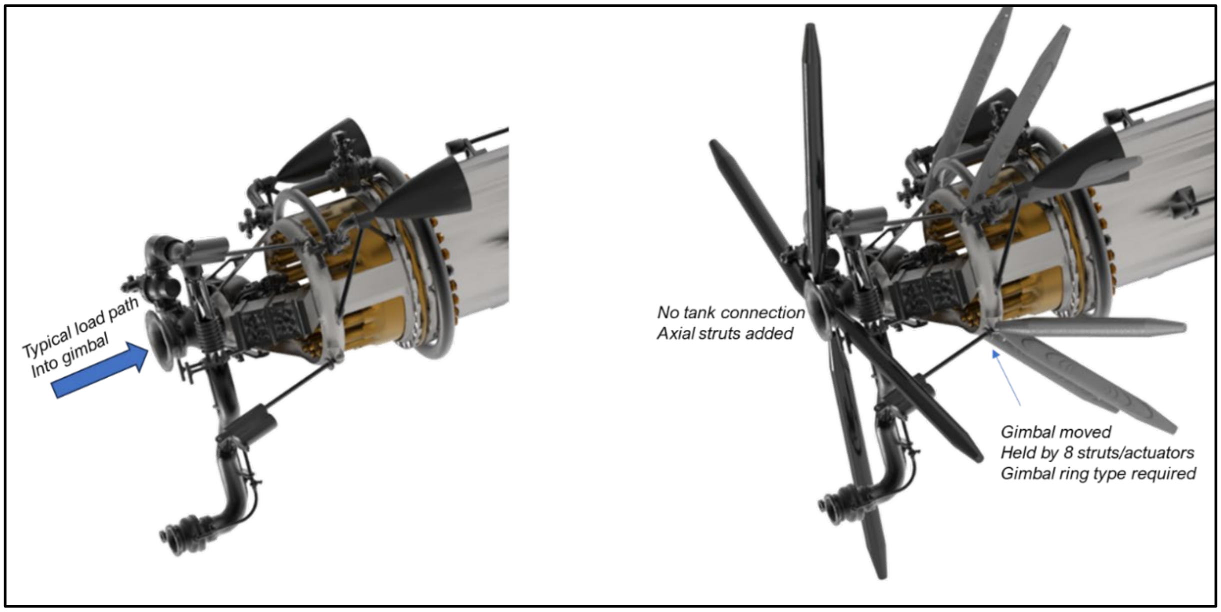

Figure 5. Details of NTP engine mounting design

Figure 6 shows the various stages' sizes for their respective launch vehicles. Each tank diameter is slightly smaller than the dynamic envelope of each payload shroud to allow for the truss system to attach to the tanks.

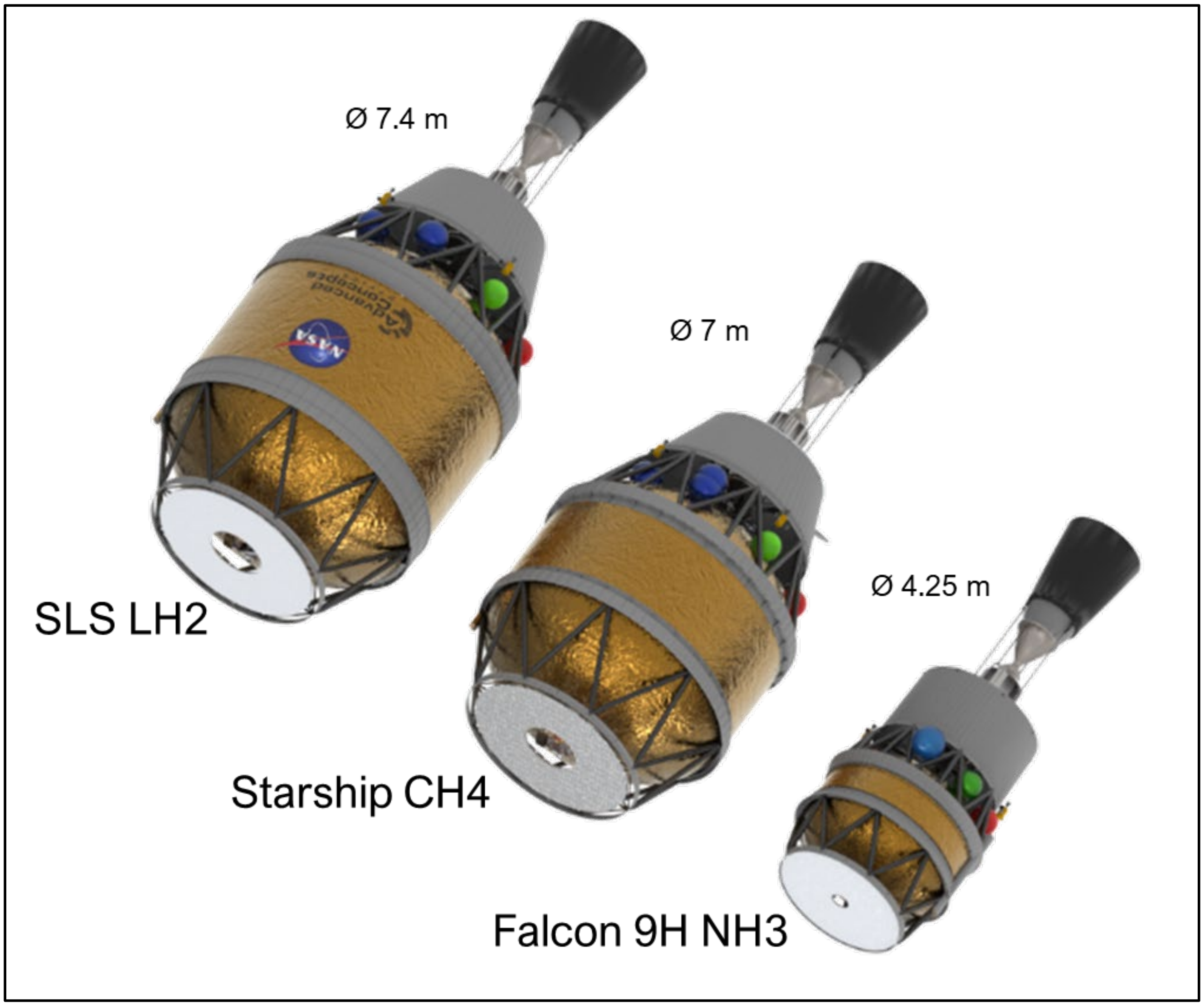

Figure 6. Three stage sizes for their respective launch vehicles

Figure 7 shows the three launch vehicles (to scale) with their propellant candidate and the fitment within their respective shroud. The SLS LH2 and the Falcon 9 Heavy cases ended up being shroud volume limited. The Starship Methane case was mass constrained. For each case a custom payload-attached fitting design would be required. This is due to the stage attachment interfaces being a different diameter and height position.

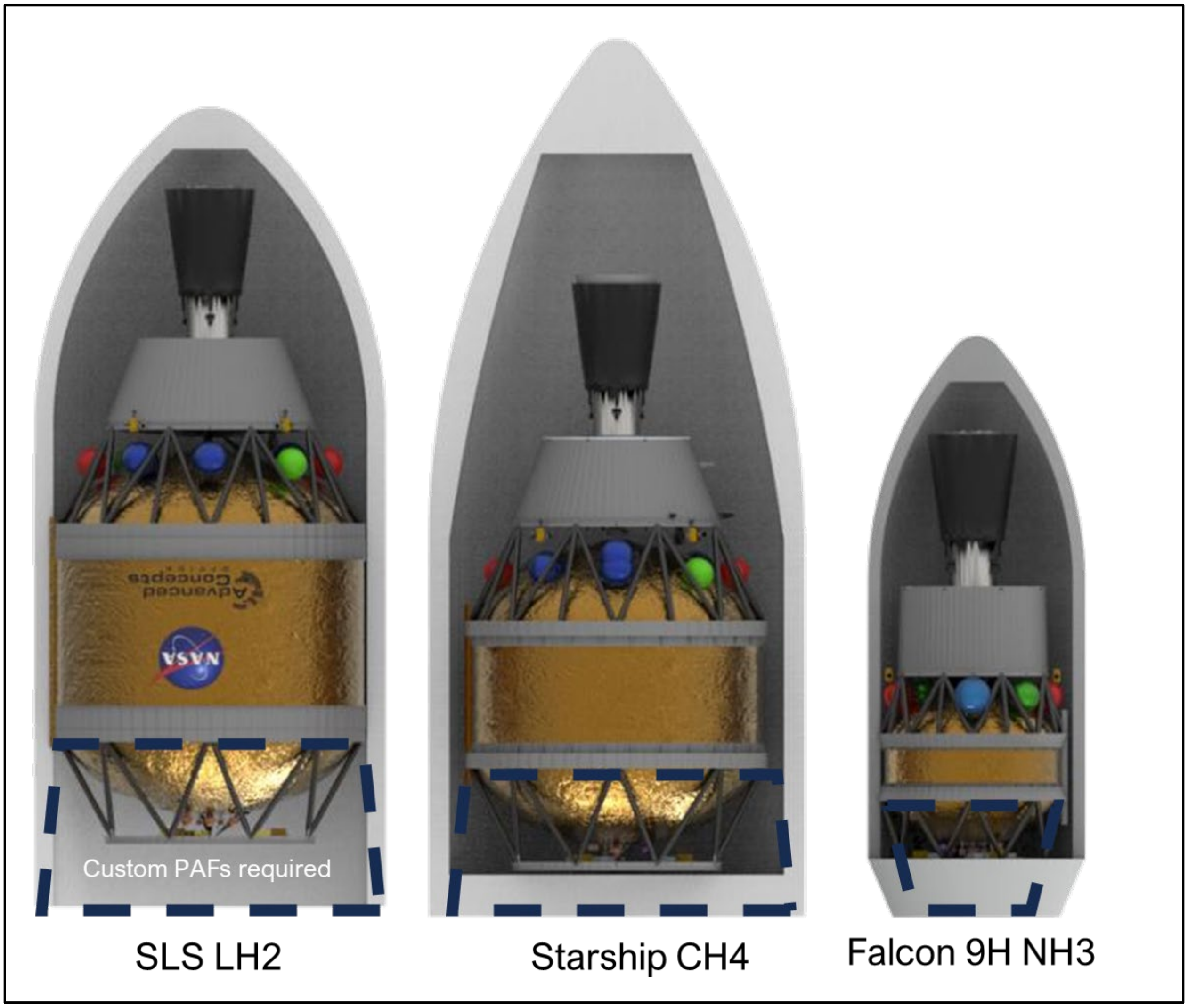

Figure 7. Three launch vehicles with their propellant candidate and fitment

Conclusions and Recommendations

Each stage's configuration approach is typical for in-space stages and does not pose any complicated or risky design features. The engine mounting truss approach would need more analysis to determine the best design solution/optimization. Other areas with additional recommended analysis include the engine mount gimbal system design, the payload attachment fitting design, and the avionics placement optimization.

Avionics

Overview

The avionics system meets all Command and Data Handling (C&DH) requirements for the spacecraft, including Guidance Navigation Control (GNC), data acquisition, storage, and ground communications. It provides attitude and state knowledge for flight control and controls power, thermal, and propulsion systems. Various antenna systems are used for data downlinks and uplink commands.

Avionics for the 3-propulsion fuel type trade is very similar, with the exception being in the thermal system controller interfaces for the avionics systems, due to hydrogen and methane needing cryogenic cooling while the ammonia case does not. A common avionics core is used for the three different mission cases (planetary defense, science kick-stage, and NTP Demonstration mission), with specific equipment added for the different cases. For planetary defense, target acquisition and proximity navigation systems are needed. Also, near real-time downlink capability using a High Gain Antenna (HGA) is desired for target impact observation in determining a hit or not. For the NTP Demonstration case, extensive NTP instrumentation and monitoring is desired for verification of systems performance. For the science kick-stage with less demanding data rates and distances, a Medium Gain Antenna (MGA) system is used for communications instead of the more massive HGA systems.

The planetary defense configuration has the worst-case avionics power requirement of about 1,600 watts for the greater communications and navigation needs. The NTP Demonstration mission configuration has the worst-case avionics mass requirement of about 600 kg for the extensive instrumentation and sensors needed.

Key Trades

To size the communications systems of the various configurations, link-budget analysis was done trading different frequencies bands, antenna gains, and radio frequency (RF) power transmission to obtain optimal results. For planetary defense, a Ka-band 2-meter HGA is used along with a 150-watt TWTA to achieve the needed downlink from extreme distances up to 9 UA. Downlink for the NTP Demonstration mission and kick-stage missions is much easier because they're closer to Earth. For these missions, using X-band Phased Array Antennas (PAA) and patch antennas along with Solid State Power Amplifiers (SSPA) would be sufficient.

Table 1. Avionics Ground Rules and Assumptions

Value	Notes
Spacecraft C&DH system will perform avionic functions including flight control, data collection and formatting, thermal management control, and power switching control	
Use TRL 6 components minimum, TRL 9 desired where possible	Operational video will be provided, video streaming for Planetary Defense and NTP Demonstration mission
Use TRL 6 components minimum, TRL 9 desired where possible	
Provide Class A instrumentation for robust Health and Status telemetry	NTP Demonstration mission instrumentation to include reactor infrared spectrum video, radiation detection, and plume spectrometry video
On board memory as required to store all video and telemetry data until downlink	
Assume Upper Stage will provide separation function and range safety Kick-Stage to provide payload separation	

Assume a class A mission for planetary defense and kick-stage, class B for demonstration mission	Single-fault tolerant for all mission critical avionics systems
Avionics operating life: Planetary Defense 10 years, Kick-Stage and Demo 1 year	
Daily link frequency: 1 time per day to for H&S and navigation telemetry	Schedule dedicated DSN links for NTP burns and impact ops
Maneuvers and attitude control will be accomplished using an RCS system. No reaction wheels or CMG will be used.	
Attitude knowledge: 10 arcsecond x and y, 60 arcseconds z	
Pointing Accuracy: 10 arcminute/axis	
Maneuver rate: 4 deg/s (240 deg in 1 min)	
Must have significant control authority for vector control (Orion rate 4deg/sec)	
NTP will have dedicated Thrust Vector Control (TVC), RCS will be used to supplement TVC	
Tip-off rate de-spin 10 deg/sec maximum	

Avionics C&DH Design Approach and Results

The avionics components selected were based on TRL 6 or greater components where possible. Many avionics components selected have flight heritage. Because of the different NTP mission profiles, radiation environments, and 10-year flight for Planetary Defense, a TRL of 8 is assumed the best possible TRL. Some NTP and demo sensors have lower TRLs, because they have only been used in terrestrial applications.

Operational videos consist of NTP Demonstration mission monitoring, impact observation, and payload deployment. For the planetary defense mission, various video recording rates are to be used to accommodate the deep space distances and impact speeds, with increasing rate of a few fps (frames per second) from several km out to 1,000 fps or more for the last km before impact with the objective being determination of a target hit and hit accuracy. The planetary defense mission is also assuming the deployment of a smaller payload that would be deployed in the period preceding the interception, for the purposes of impact verification and remote observation. The final impact speed assumed was approximately 14 km/s, which is in the lower range of possible speeds for the various trades from mission analysis. At that speed, using 1,000 fps, the last frame should be within 14 meters of the asteroid. Higher impact speeds would require higher frame rates. The higher frame rates can be done using less resolution, which helps reduce the downlink volume required. The high-speed camera should be part of the target visual navigation system, along with Doppler Lidar and other sensors and processors as defined in the ALHAT and Argon rendezvous and proximity navigation systems. These cameras, sensors, and processors are included in the planetary defense GN&C section of the MEL.

For the NTP Demonstration mission, various camera types are used, including infrared and spectrometer cameras monitoring the status of the NTP engine before, during, and after burns. These cameras are part of a large instrumentation and monitoring system designed specifically for an NTP Demonstration mission that was performed by ACO in 2018. This demonstration instrumentation and monitoring system is included in the NTP demonstration MEL. Much faster video rates are possible for the demo and kick-stage missions being much closer to Earth, within cis-lunar distances.

Nominal data downlink is assumed at least once per day for all mission types. Dedicated data recorders are used to provide memory for at least two days of data to accommodate missed downlinks and contingency situations like safe modes. About 8 GB of memory is required for 1 day of NTP Demonstration mission data. For planetary defense at 2.5 AU distance, 2 GB of memory would take about 8 hours to downlink. It is estimated that 2 GB of memory should hold approximately 10 minutes of planetary defense approach video, with the last second only needing 150 MB (10:1 video compression is assumed). This last second of video would only take about 40 minutes to downlink, plus light time latency for 2.5 AU distance, which is about 20 minutes. will be shorter or longer depending on the asteroid distance at the time of impact.

The planetary defense mission is assumed a class-A mission, while the NTP Demonstration mission and science kick-stage missions could be class-B. But for both class A and B, single-fault tolerance is assumed for all mission critical avionics systems. For planetary defense with a mission travel time of up to 10 years or more, redundant components are put into a dormant mode to extend component life.

The following is a short summary of Avionics subsystems and major components:

- Attitude and Control System (ACS)
  - Sun sensors, star trackers, and IMU for state knowledge
- Command and Data System (CDS)
  - Flight computers – provides GN&C, data management, and vehicle systems control
  - Data acquisition units and solid-state data recorders
- Autonomous target acquisition and tracking for Planetary Defense
  - Long-range and short-range systems
  - High speed video camera, lighting
- Instrumentation
  - Pressure, temperature, strain sensors, etc.
  - Video monitoring cameras for health and status
  - Radiation tolerant cameras for NTP Demonstration
- Communication
  - Ka-band HGA system for Planetary Defense link up to 9AU distance
  - X-band HGA system for ground link from cis-lunar distances
  - S-band MGA system for ground link from LEO/GEO distances
- Vehicle Systems Control
  - NTP engine and TVC controllers, thermal and CFM controllers, separation controllers

Figure 8 shows the general location of where the different components are located on the vehicle. The MPS and other aft end avionics are protected from fission radiation by the NTP shadow shield. Some of the demonstration sensors and cameras are below the shadow shield in the high radiation environment. But these components were selected for this environment with heavy component shielding as required.[1] Table 2 shows the Avionics section of the combined MEL, which includes all components for all mission types. A list of basic mass for the specific mission types is included, where the non-applicable components are simply deleted from the combined mass to get the mission specific masses. Note that the NTP Demonstration mission has the greatest mass with its large instrumentation requirements. Included in the table is the avionics power required for the different mission types. These values were derived from a power profiling effort in which the maximum power demand was typically during the NTP engine burn periods. The exception is for the Planetary defense mission, in which maximum power is needed for navigation and comm during the final moments before impact. See the MEL summery section of the report for the full MEL showing individual mission specific summaries.

[1] OVERVIEW OF THE MAIN PROPULSION SYSTEM FOR A NUCLEAR THERMAL PROPULSION FLIGHT DEMONSTRATOR, Bean Q.A, Rodriguez M.A., Simpson S.P., accessed September 2025

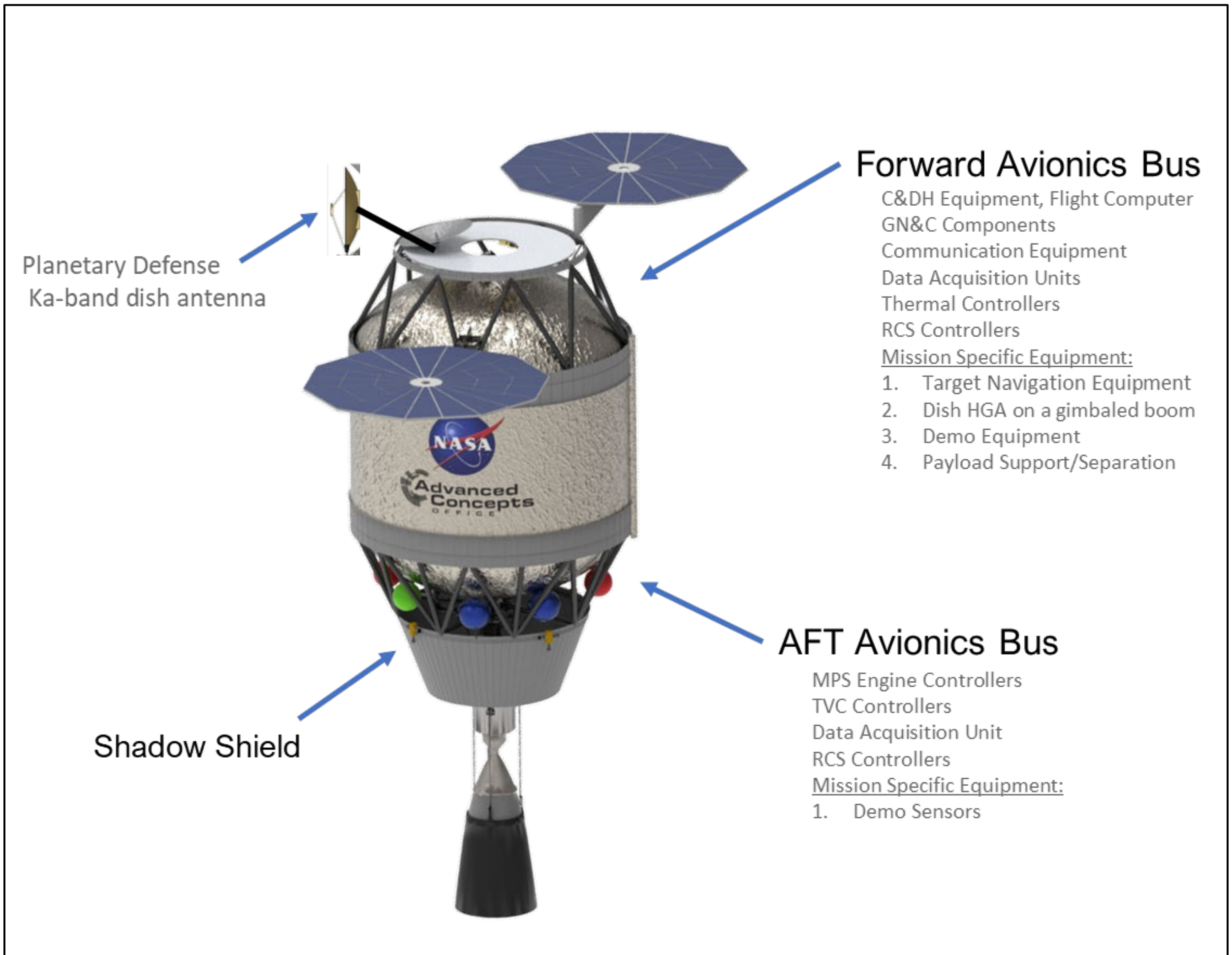

Figure 8. Avionics Element Layout

Communications Design Approach and Results

The communications strategy for the different configurations and missions is to use a common core X-band system and add components as needed to support the specific missions. This X-band system will link to NEN ground stations or the in-space TDRSS system when the spacecraft is inside GEO locations, and link to the DSN when beyond GEO distance in deep space. The X-band system is used for both telemetry and operation video of the NTP burns, NTP Demonstration mission, and Kick-Stage payload separation video.

Its assumed video is desired for real-time observation of NTP departure burns. For this, a schedule TDRSS link can be done if a NEN station is not available at that time. For the Planetary Defense impact determination data and observations video, a dedicated DSN link should be scheduled.

Table 2. Avionics MEL Summary

Mission Specific	Category Level	Master Equipment List (MEL)	Qty	Unit Mass (kg)	Basic Mass (kg)	Contin-gency (%)	Contin-gency (kg)	Predict-ed Mass (kg)
	1.0	Avionics			784.67	16.6%	130.61	915.28
	1.1	Command and Data Handling (C&DH)			461.12	19.5%	90.07	551.19
	1.1.1	Command and Data System	1	45.72	45.72	13.7%	6.26	51.98
	1.1.2	Instrumentation & Monitoring	1	55.00	55.00	16.0%	8.80	63.80
NTP Demo	1.1.3	Instrumentation & Monitoring - Demo	1	274.40	274.40	19.5%	53.51	327.91
	1.1.4	Avionics Cabling	1	86.00	86.00	25.0%	21.50	107.50
	1.2	GN&C			116.24	14.9%	17.32	133.56
	1.2.1	Attitude Control System	1	7.30	7.30	3.0%	0.22	7.52
Planetary Defense	1.2.2	Target Navigation System	1	108.94	108.94	15.7%	17.10	126.04
	1.3	Communications			119.91	9.5%	11.41	131.32
Planetary Defense	1.3.1	Ka-band HGA System	1	62.40	62.40	8.1%	5.05	67.45
	1.3.2	X-band MGA System	1	19.04	19.04	7.2%	1.37	20.41
Plan Def, NTP Demo	1.3.3	X-band Phased Array Antenna	1	13.60	13.60	15.0%	2.04	15.64
Kick-Stage	1.3.4	X-band Patch Antenna	1	1.20	1.20	3.0%	0.04	1.24
Kick-Stage, NTP Demo	1.3.5	S-band LGA System	1	13.67	13.67	3.0%	0.41	14.08
	1.3.6	Comm cabling	1	10.00	10.00	25.0%	2.50	12.50
	1.4	Vehicle systems, heaters, etc.			87.40	13.5%	11.80	99.20
	1.4.1	Vehicle Systems	1	1.00	1.00	3.0%	0.03	1.03
Planetary Defense	1.4.2	Ka Dish Antenna Release Mechanism	1	1.00	1.00	3.0%	0.03	1.03
Kick-Stage	1.4.3	Payload Separation Control System	1	10.00	10.00	13.0%	1.30	11.30
	1.4.4	Thermal Control Systems	1	43.40	43.40	13.0%	5.64	49.04
	1.4.5	Propulsion Control Systems	1	32.00	32.00	15.0%	4.80	36.80

To accommodate the near real-time impact video requirement of the Planetary Defense mission at great distances, up to 9 AU, a Ka-band High Gain Antenna (HGA) is used. Link budget analysis shows 500 kbps is achieved at 2.5 AU, and 50 kbps at 9 AU, using a 2m dish antenna and 150w RF power from a TWTA to a DSN 34m stations. The 500 kbps is good for near real-time impact determination and observation, about 1 hour downlink time for the critical last second of data before impact. Using the X-band PAA and 20w SSPA to a DSN 34m station at 2.5 AU, 300 bps downlink is achieved. This is sufficient for low power long transit time telemetry and navigation. The X-band system can also be used for contingency backup of the Ka system.

For the NTP Demonstration mission, 100 Mbps is achieved within Cis-Lunar distances using X-band PAA and 15w SSPA to a DSN 34m station. This rate is very good for real-time demo video and monitoring. For the Kick-Stage, 25 Mbps is achieved using X-band patch antennas and 20w SSPA to a NEN 11m station.

Conclusions and Recommendations

No new avionics technologies are required for these NTP missions. Most components are TRL 6 or greater, and many components have flight heritage. Some sensors and cameras for the NTP Demonstration mission are not space qualified. Some development and significant testing will be needed. Rendezvous, proximity, and impact navigation for Planetary Defense will need significant analysis and development. The Double Asteroid Redirection Test (DART) mission was a successful demonstration mission in 2022 and can be used for reference. It used the Didymos Reconnaissance and Asteroid Camera for Optical navigation (DRACO), which had a slow frame rate. For this study, it is thought a high frame rate camera is necessary. There is some avionics risk for the long duration and great distance for the Planetary Defense mission. But this risk is mitigated by providing redundancy and using dormant mode to prolong component life. Using a Ka-band dish HGA for Planetary Defense provides reasonable link time for the impact determination and observation. Camera requirements and analysis were rough approximations and significant detail analysis will be needed going forward.

Thermal

Overview

Thermal estimates for this study were performed for three different vehicles with three different propellants: hydrogen, methane, and ammonia. Analysis was done to estimate 1) the size, mass, and power of the thermal system, and 2) the pressurant mass required to pressurize the MPS tank before engine burns. The primary differences between these vehicles are: 1) the propellant storage temperatures (and therefore management accommodations), 2) the main propulsion tank sizes, and 3) the RCS tank sizes. The thermal subsystem also considered the most demanding vehicle use cases that were identified from the three possible missions.

Ground Rules and Assumptions

The asteroid interceptor vehicle was assumed to perform a main propulsion burn once at the start of the mission, so active cryogenic fuel management (CFM) technology was not required, but some mission trajectories go as far out as 9AU. Therefore, a 9AU case was considered to determine how this could affect the thermal management of avionics that distance from the Sun.

The technology demonstration mission requires the vehicle to go to cis-lunar space and then perform several NTP burns. This mission uses active CFM and pressurant to prepress the main propulsion tank before each demonstration burn. The team decided that the NTP Demonstration mission would not rely on the active CFM, therefore, an analysis was performed to estimate how much propellant would boil off if the cryocoolers failed. Cryocoolers are still included so they have the chance to become flight proven. The vehicle's distance from the Earth over time for a notional technology demonstration mission is shown in Figure 9.

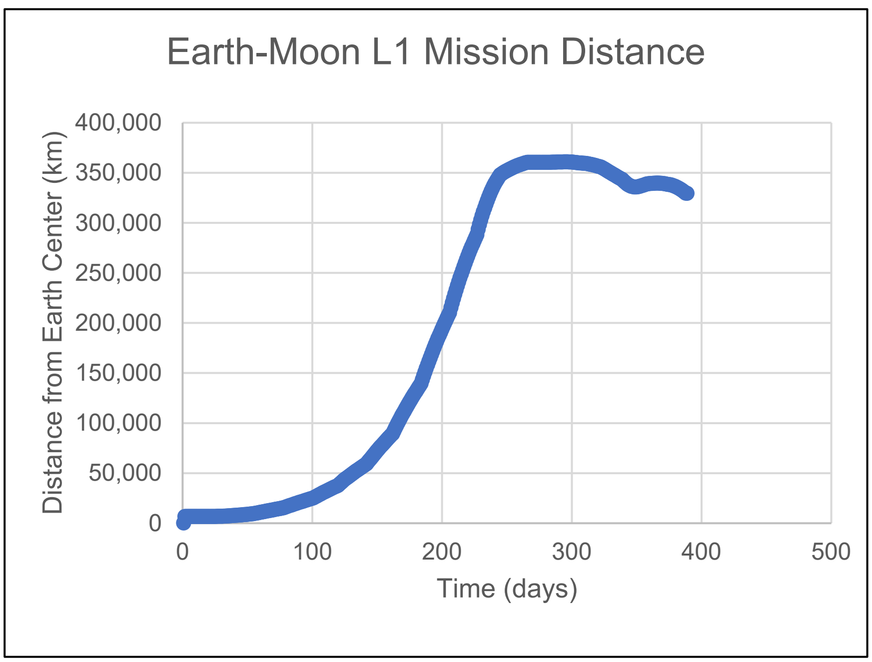

Figure 9: Spacecraft distance from the Earth

The science kickstage mission performed one burn at the start of the mission, and then the kick stage is dropped and does not have to survive from that point onward. Thermal management of the stage itself is not intermingled with thermal management of the payload, so fewer avionics support elements are required. Also, the science kickstage does not require cryocoolers or survival in deep space. An overview of the elements required by each mission and propellant is given in Table 3 and other GR&A are provided in Table 4.

Table 3: Thermal System Elements Matrix

Required Elements	Technology Demonstration			Science Kickstage			Asteroid Interceptor		
Propellant	H2	CH4	NH3	H2	CH4	NH3	H2	CH4	NH3
Radiator Panel	X	X	X				X	X	X
Radiator Heaters & Sensors	X	X	X				X	X	X
Radiator Louvers							X	X	X
Avionics MLI	X	X	X				?	?	?
RCS MLI	X	X	X				?	?	?
RCS Heaters & Sensors	X	X	X	X	X	X	X	X	X
MPS Tank MLI	X	X	X				?	?	?
MPS Tank Cryocooler(s)	X	X					?	?	
MPS Tank 2-Stage Cooling	X						?		
MPS Tank SOFI	X	X	X	X	X	X	X	X	X
Thermodynamic Vent System	X	X	X	X	X	X	X	X	X
MPS Tank RF Mass Gauge	X	X	X	X	X	X	X	X	X
MPS Tank Sensors	X	X	X	X	X	X	X	X	X
Capital X means "always needed." A question mark (?) means "sometimes needed depending on the mission trajectory."									

Table 4: Thermal Ground Rules and Assumptions

Property	Value		
Propellant	Liquid Hydrogen	Liquid Methane	Liquid Ammonia
Propellant Density	71.4 kg/m3	408 kg/m3	660 kg/m3
Max Storage Pressure	31 psi	31 psi	31 psi
Max Storage Temperature	23 K	121 K	256 K
Tank Max Fill	95%	95%	95%
Tank Size	300 m3	215 m3	39 m3
Residual Propellant	3% of useable	3% of useable	3% of useable
Autogeneous Propellant	1% of useable	1% of useable	1% of useable
Startup Pressurant	Helium	Helium	Helium
Helium Storage Pressure	4500 psi	4500 psi	4500 psi
Helium Storage Temperature	294 K	294 K	294 K
Soakback Heat from Engine	1.83 kW	1.83 kW	1.83 kW
Nuclear Radiation Heat	50 kW	50 kW	50 kW
SOFI thickness	1 in	1 in	1 in
Fluid Management	IMLI + 2-Stage Cooling + TVS	IMLI + Single Stage + TVS	tMLI + TVS

Cryocooler Performance Data	GRC 20K Tests (2022) MSFC 90K Tests (2023)	MSFC 90K Tests (2023)	N/A
Cryocooler ZBO Environment	Earth-Moon L1	Earth-Moon L1	N/A
ZBO Assumption	For prop inventory, assume cryocoolers fail but TVS does not	For prop inventory, assume cryocoolers fail but TVS does not	Liquid Ammonia

Propellant Cryogenic Fluid Management and Pressurization

Hydrogen, methane, and ammonia were the three propellants considered. The design considerations and intermediary analysis results are covered in the following sections.

Hydrogen Vehicle: Cryogenic Fluid Management

Hydrogen must be stored at very low temperatures and requires the greatest accommodation. The tank is cooled by flowing 20 K cryocooler refrigerant through tubes that are on the tank in a "tube on tank" configuration. An inch of SOFI surrounds the tank to prevent condensation and freezing of air onto the tank prior to launch. Multiple layers of integral MLI (IMLI) is between the SOFI and a 90K broad area cooling shield (BAC). The 90 K BAC is cooled with 90 K refrigerant to provide a low temperature boundary that intercepts heat for the <23 K tank. Additional layers of IMLIinsulates the 90 K BAC from the space environment. The outer layer of the MLI has a second surface silver coating which provides a high reflectivity in the solar spectrum and a high absorptivity in the infrared spectrum to give the MLI's outer layer the lowest temperature possible.

The temperature of the vehicle at Earth-Moon L1 is shown in Figure 10. The outermost layer of MLI averaged 123 K at Earth-Moon L1. 20.8 W of heat conducted into the <23 K tank through the tank's struts at Earth-Moon L1, and 17.6 W radiated into the tank through the tank's 40-layer IMLI. Another 18.4 W was added to the tank through insulation penetrations, such as wetted lines, non-wetted lines, sensors, valves, and tank ports. Estimates for the heat leak through these penetrations came from ground test data of a large hydrogen tank. In total, 56.8 W of cooling would be required from the 20 K cryocoolers while performing the technology demonstration mission at Earth-Moon L1. Therefore, three 20K cryocoolers are needed. The BAC received 39.1 W of heating, so only one 90 K cryocooler is necessary.

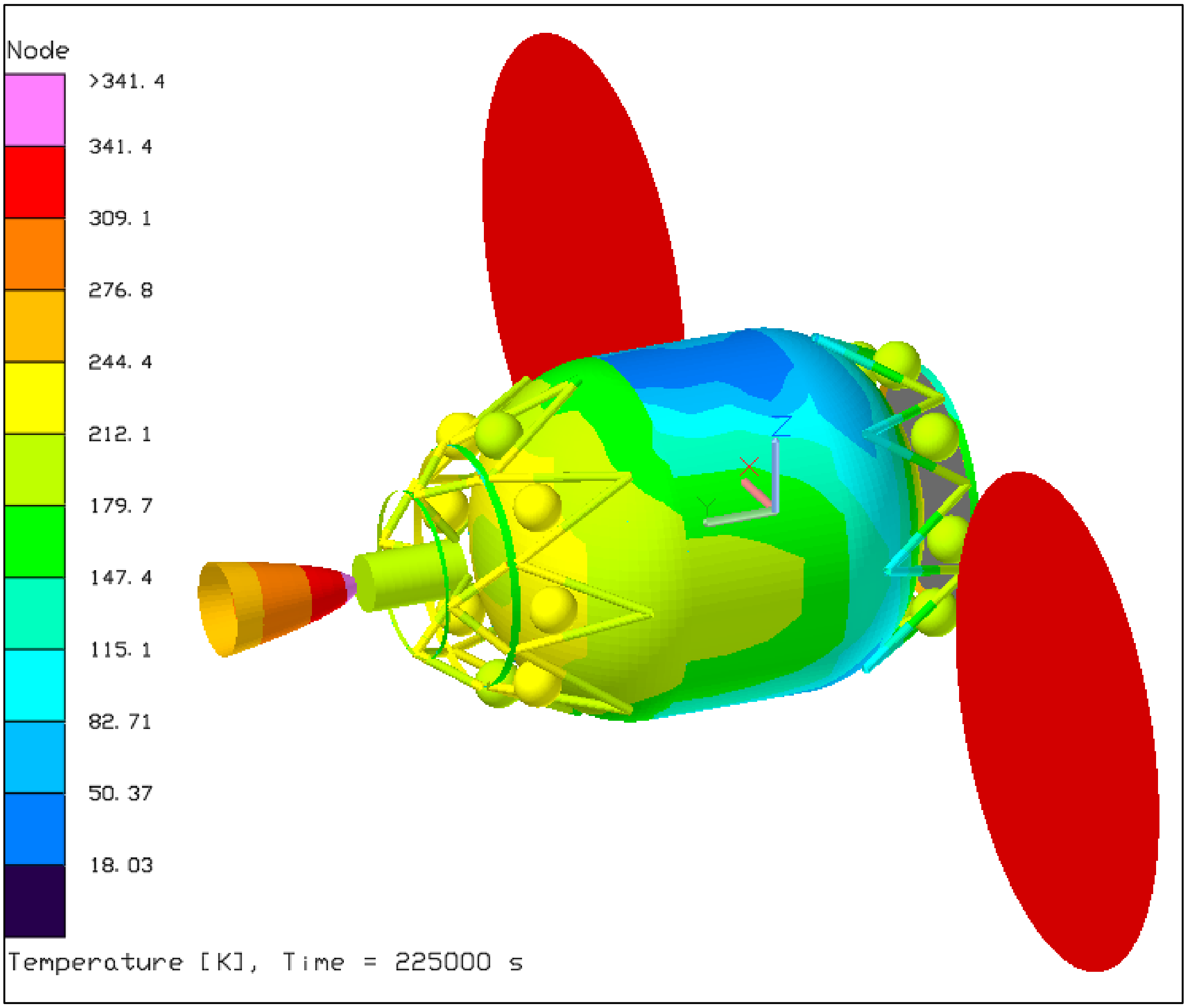

Figure 10: SLS Block 2 Hydrogen NTP Stage Near Earth-Moon L1 for the Technology Demonstration. Outer MLI layer temperatures are shown where present

A thermodynamic vent system (TVS) is also included in the tank to maximize cooling while venting propellant from the tank in case the cryocoolers fail and to accommodate unforeseen circumstances.

Tank Pressurization

During an engine burn, the tank is pressurized autogenously. The quantity of pressurant necessary to pressurize the tank before an engine burn was estimated using TankTherm (an ACO tank thermodynamics analysis tool). The helium pressurant tanks were assumed to store helium at 20 C and 34.5 MPa and the required pressurant was estimated such that the MPS tank increased in pressure by 2 psi.

Table 5: Hydrogen Tank Pressurization (fill levels and pressurant required)

Model	Case	Burn 1	Burn 2	Burn 3	Tank Size (m3)
TankTherm	LH2	100% (0.34 kg)	14% (5.45 kg)	8% (5.81 kg)	300

The total helium required to pre-pressurize the MPS tank was found to be 11.6 kg before margin. This information was relayed to the propulsion engineer and mission analyst who incorporated it into the total helium mass.

Methane Vehicle: Cryogenic Fluid Management

Methane's storage temperature was ground ruled to be less than 121 K, so it does not require as much insulation or the broad area cooling (BAC) shield. The tank is cooled with refrigerant from a 90K cryocooler, covered in SOFI, and insulated in-space with multiple layers of MLI.

In total, the CH4 tank is estimated to receive 40.5 W of heating at Sun-Earth L1 with 9 W coming through the 20-layer IMLI, 13 W conducting through the S-glass struts, and the remaining getting through tank penetrations such as propellant lines, vent lines, sensors, etc. According to test data, the maximum cooling capacity of the 90 K Creare cryocooler is 141.2 W when the compressor inlet temperature is 105 K. This cooling capacity will increase further for storage of CH4 up to 123 K. A single 90 K Creare cryocooler has more than enough lift capacity for the technology demonstration mission.

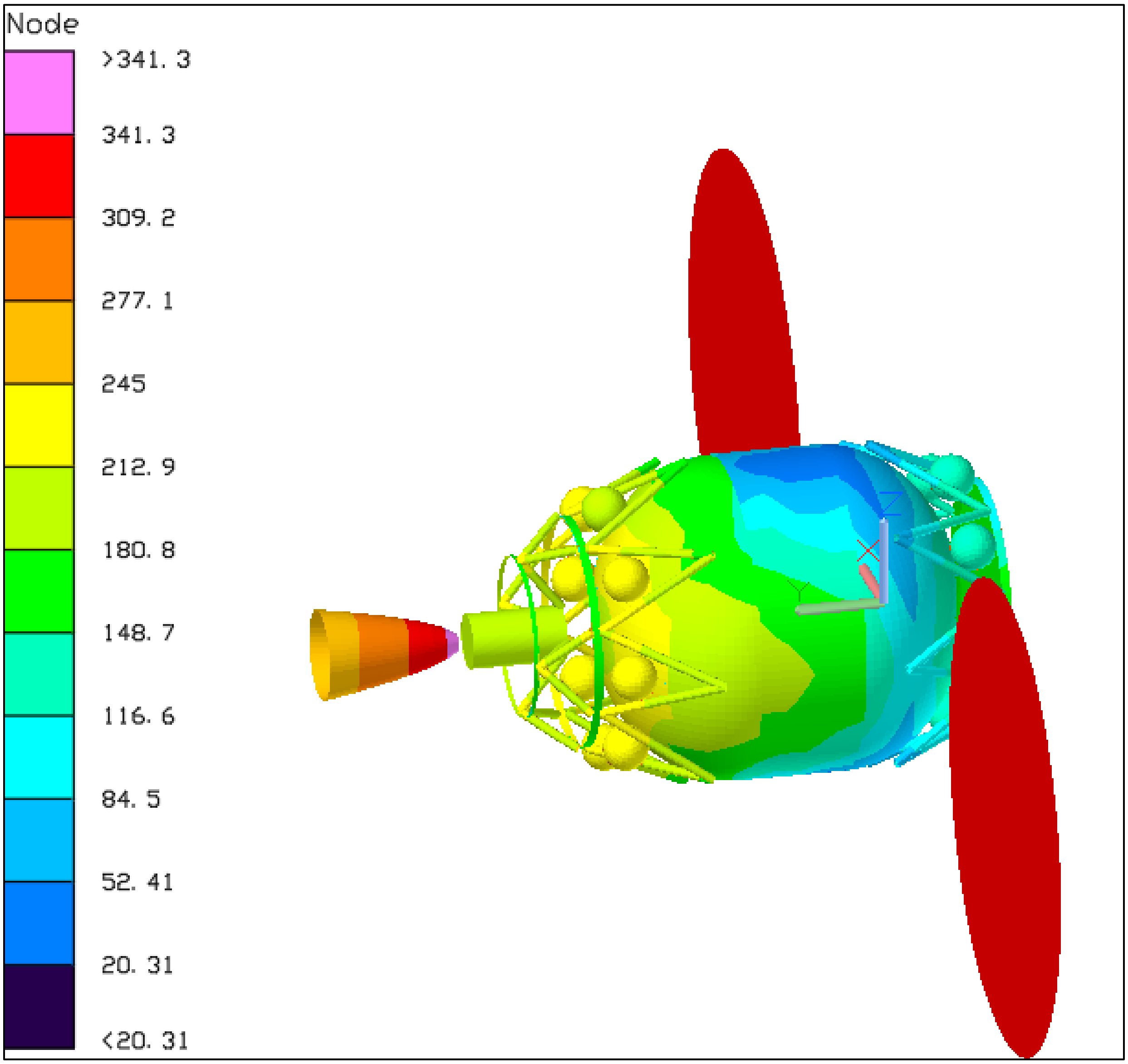

Figure 11: Starship Methane NTP Stage Near Earth-Moon L1 for the Technology Demonstration. Outer MLI layer temperatures are shown where present.

Tank Pressurization

A table showing the pressurant mass estimates is below.

Table 6: Methane Case Pressurization (fill levels and pressurant required)

Model	Case	Burn 1	Burn 2	Burn 3	Burn 4	Tank Size (m3)
TankTherm	CH4	100% (0.3 kg)	28% (4.33 kg)	20% (4.78 kg)	10% (5.33 kg)	215

The total helium required to pre-pressurize the MPS tank was found to be 14.7 kg before margin. This information was relayed to the propulsion engineer and mission analyst who incorporated it into the total helium mass.

Ammonia Vehicle: Propellant Storage

The case where ammonia is used as the propellant is by far the least demanding case for the thermal system. Ammonia can be stored up to 256 K (-17.2 C) and could be subcooled prior to launch down to 200 K (-73.2 C). By subcooling ammonia before launch and using second surface silver on the outer layer of MLI and on the tank struts, ammonia should never require cooling during any phase of the mission.

This eliminates the need for any cryocoolers which were the least reliable and most complex components in the H2 and CH4 thermal systems. The thermal system mass is considerably lower for ammonia because of 1) the elimination of cryocoolers and their ancillary components, 2) the reduced insulation, and 3) the smaller tank size (caused by ammonia's higher density).

Tank Pressurization

A table showing the pressurant mass estimates is below. TankSIM could not be used to estimate the pressurant mass, because TankSIM does not have an ammonia fluid object.

Table 7: Ammonia Case Pressurization (fill levels and pressurant required)

Model	Case	Burn 1	Burn 2	Burn 3	Burn 4	Tank Size (m3)
TankTherm	NH3	100% (0.045 kg)	39.4% (0.56 kg)	27.7% (0.66 kg)	13.9% (0.77 kg)	39

The total helium required to pre-pressurize the MPS tank was found to be 2.04 kg before margin. This information was relayed to the propulsion engineer and mission analyst who incorporated it into the total helium mass.

MELs, Conclusions and Recommendations

The MELs for the thermal systems for the three cases considered are below. All three of these MELs include louvers to eliminate heater power requirements if the system were to travel to 9 AU and used 40% mass growth allowance. The LH2 case, which had the largest volume tank and the most stringent propellant storage requirements, has predictably the heaviest thermal system at 2,973.87 kg. The predicted mass for the CH4 case was 1,366.41 kg and the NH3 was estimated to be 561.35 kg.

Table 8: SLS B2 Hydrogen Thermal System MEL

Mass Estimation List (MEL)	Predicted Mass (kg)
Thermal	2,973.87
Avionics Thermal	468.38
Radiator panels	222.56
Heaters & Thermostats	2.10
Avionics MLI	23.81
Louvers	219.91
RCS Thermal	62.65
Tank and Prop Line MLI	61.24
Tank and Prop Line Heaters & Wiring	1.41
CFM Thermal	2,442.84
Innermost Tank MLI	753.36
BAC Shield MLI	374.31
BAC Shield	153.60
Tank Cryocooler	443.52
BAC Shield Cryocooler	94.64
SOFI Insulation	421.81
TVS	180.60
RF Mass Gauge	9.80
Sensors	11.20

Table 9: Starship CH4 Thermal System MEL

Mass Estimation List (MEL)	Predicted Mass (kg)
Thermal	1,366.41
Avionics Thermal	468.38
Radiator panels	222.56
Heaters & Thermostats	2.10
Avionics MLI	23.81
Louvers	219.91
RCS Thermal	62.65
Tank and Prop Line MLI	61.24
Tank and Prop Line Heaters & Wiring	1.41
CFM Thermal	835.39
Innermost MLI	253.34
Tank Cryocooler	94.64
SOFI Insulation	285.80
TVS	180.60
RF Mass Gauge	9.80
Sensors	11.20

Table 10: Falcon Heavy NH3 Thermal System MEL

Thermal	561.35
Avionics Thermal	341.64
Radiator panels	222.56
Heaters & Thermostats	2.10
Avionics MLI	11.43
Louvers	105.55
RCS Thermal	62.65
Tank and Prop Line MLI	61.24
Tank and Prop Line Heaters & Wiring	1.41
MPS Thermal	157.06
Tank MLI	68.16
SOFI Insulation	77.69
Sensors	11.20

Propulsion

Overview

The objective for propulsion during this study was to design both main propulsion systems (MPS) and reaction control systems (RCS) for the three missions and three vehicle designs – for a total of nine mission/vehicle combinations. Each vehicle is tied to a different propellant type (shown below), which means the propulsion design can get complicated to keep track of.

Reactor Configuration

The reactor is tightly integrated into the NTP engine system, and as such will be briefly discussed here. All NTP vehicle configurations in this effort assumed the use of a particle-bed solid reactor core, the details of which were provided by SNP project office based on their FY2024-2025 baseline reactor configuration analyses.[2,3] Additional work to modify the reactor performance for the alternative propellant architectures was also coordinated with the SNP project office.

Main Propulsion System

Each launch vehicle design uses a different MPS propellant – though all three are nuclear thermal propulsion (NTP). Each propellant tank was sized based either on maximum volume or mass allowed by the launch vehicle. This was decided by the study lead in order to strategically assess each propellant type during this study cycle.

1. LH2 + SLS B2 (Max volume)
2. CH4 + Starship (Max mass)
3. NH3 + Falcon Heavy (Max volume)

The RCS was designed to use the same propellants (MMH/NTO) and use the same thrusters (25 lbf R-1E bipropellant thrusters) across all options. The propellant tank geometry was also sized for each mission and held the same for each vehicle configuration for simplicity. To save mass, the tanks were sized such that a single set of tanks could be removed from the smaller vehicles since the propellant tank geometry was being maintained across all of the different vehicle options. The ground rules and assumptions for the propulsion discipline are listed in the pages below.

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[2] "Engine Cycle Comparison for Alternative Propellant Nuclear Thermal Propulsion Engines." Paper presented at ASCEND 2023, Las Vegas, Nevada. ASCEND 2023, American Institute of Aeronautics and Astronautics, October 23, 2023. https://doi.org/10.2514/6.2023-4715.
[3] Nikitaeva, Daria, Corey Smith, and Matthew Duchek. "Methane-Based Nuclear Thermal Propulsion Engine Considerations." Paper presented at Nuclear and Emerging Technologies for Space (NETS 2025), Huntsville, AL. Nuclear and Emerging Technologies for Space (NETS 2025), ANS, May 2025. https://ntrs.nasa.gov/citations/20250004353.

Table 11. Main Propulsion System GRA for Liquid Hydrogen, Methane, and Ammonia Propellants (assume constant reactor power)

Liquid Hydrogen	
Propulsion System Type	Nuclear Thermal Propulsion (NTP)
Engines	25,000 lbf
Engine Configuration	Single NTP Engine
Propellant Fuel	Liquid Hydrogen
Propellant Storage Temperature	<23 K
Propellant Storage Pressure	30 psi
Residual Propellant	3% of maneuver propellant
Autogenous Propellant	1% of maneuver propellant
Reserve Propellant Budget	3% of maneuver propellant
Initial Tank Ullage	5%
Propellant Storage Densities	71.4 kg/m3
Pressurant	Autogenous LH2 (GHe for startup)
Pressurant GHe Storage Temperature	294 K
Pressurant GHe Storage Pressure	4,500 psia (310.3 bar)
Pressurant Storage Densities	Manually obtained from REFPROP
Methane	
Propulsion System Type	Nuclear Thermal Propulsion (NTP)
Engines	40,000 lbf
Engine Configuration	Single NTP Engine
Propellant Fuel	Liquid Methane
Propellant Storage Temperature	<121 K
Propellant Storage Pressure	30 psi
Residual Propellant	3% of maneuver propellant
Autogenous Propellant	1% of maneuver propellant
Reserve Propellant Budget	3% of maneuver propellant
Initial Tank Ullage	5%
Propellant Storage Densities	408 kg/m3
Pressurant	Autogenous LCH4 (GHe for startup)
Pressurant GHe Storage Temperature	294 K
Pressurant GHe Storage Pressure	4,500 psia (310.3 bar)
Pressurant Storage Densities	Manually obtained from REFPROP
Ammonia	
Propulsion System Type	Nuclear Thermal Propulsion (NTP)
Engines	45,000 lbf
Engine Configuration	Single NTP Engine
Propellant Fuel	Liquid Ammonia
Propellant Storage Temperature	<256 K

Propellant Storage Pressure	30 psi
Residual Propellant	3% of maneuver propellant
Autogenous Propellant	1% of maneuver propellant
Reserve Propellant Budget	3% of maneuver propellant
Initial Tank Ullage	5%
Propellant Storage Densities	660 kg/m3
Pressurant	Autogenous LNH3 (GHe for startup)
Pressurant GHe Storage Temperature	294 K
Pressurant GHe Storage Pressure	4,500 psia (310.3 bar)
Pressurant Storage Densities	Manually obtained from REFPROP

Table 12. Reaction Control System GR&A: Planetary Defense

Planetary Defense	
Propulsion System Type	Pressure-regulated, storable biprop system
Engines	R-1E (25 lbf, MMH/NTO)
Engine Configuration	Four pods, each containing three engines (aft-pointing and two laterals)
Propellant Fuel	MMH
Propellant Oxidizer	NTO
Mixture Ratio	1.65
Propellant Storage Temperature	294 K
Propellant Storage Pressure	400 psia
	27.6 bar
Residuals	3% of maneuver propellant
Reserve	15% of maneuver propellant
Tank Ullage	5%
Propellant Storage Densities	Manually obtained from NIST website'
Pressurant	GHe
Pressurant Reserve	25%
Pressurant Storage Temperature	294 K
Pressurant Storage Pressure	4,500 psia (310.3 bar)
Pressurant Storage Densities	Manually obtained from REFPROP

Table 13. Reaction Control System GR&A: Science Kickstage

Science Kickstage	
Propulsion System Type	Pressure-regulated, storable biprop system
Engines	R-1E (25 lbf, MMH/NTO)
Engine Configuration	Four pods, each containing three engines (aft-pointing and two laterals)
Propellant Fuel	MMH
Propellant Oxidizer	NTO
Mixture Ratio	1.65
Propellant Storage Temperature	294 K
Propellant Storage Pressure	400 psia
	27.6 bar
Residuals	3% of maneuver propellant
Reserve	15% of maneuver propellant
Tank Ullage	5%
Propellant Storage Densities	Manually obtained from NIST website'
Pressurant	GHe
Pressurant Reserve	25%
Pressurant Storage Temperature	294 K
Pressurant Storage Pressure	4,500 psia (310.3 bar)
Pressurant Storage Densities	Manually obtained from REFPROP

Table 14. Reaction Control System GR&A: Flight Demonstrator

Flight Demonstrator	
Propulsion System Type	Pressure-regulated, storable biprop system
Engines	R-1E (25 lbf, MMH/NTO)
Engine Configuration	Four pods, each containing three engines (aft-pointing and two laterals)
Propellant Fuel	MMH
Propellant Oxidizer	NTO
Mixture Ratio	1.65
Propellant Storage Temperature	294 K
Propellant Storage Pressure	400 psia
	27.6 bar
Residuals	3% of maneuver propellant
Reserve	15% of maneuver propellant
Tank Ullage	5%
Propellant Storage Densities	Manually obtained from NIST website'
Pressurant	GHe
Pressurant Reserve	25%
Pressurant Storage Temperature	294 K
Pressurant Storage Pressure	4,500 psia (310.3 bar)
Pressurant Storage Densities	Manually obtained from REFPROP

RCS Thruster Selection

The thruster selected for RCS was the R-1E 110 Newton bipropellant thruster. It uses MMH/NTO and is a flight-proven, reliable product from Aerojet Rocketdyne.

RCS Tank and Propellant Sizing

The propellant tanks were sized specifically to accommodate enough propellant to fit on the largest of the vehicle options (Starship). For the smaller two vehicles, a single pair of tanks were removed because of the lower propellant requirement and available volume. (*For example, the planetary defense mission uses 1 fuel and 1 oxidizer tank for the SLS and Falcon Heavy vehicles*.) All options use the same thruster configuration (R-1E, 25 lbf thrusters).

Table 15. Reaction Control System Propellant Sizing

Mission Type	Fuel	Ox	(kg)
Planetary Defense	260.4	429.6	Usable
	49.5	77.3	Residual*
	309.8	507.0	Total
	327.7	524.6	Tank + Prop
Flight Demo	464.2	765.8	Usable
	88.2	137.9	Residual*
	552.3	903.7	Total
	582.1	933.2	Tank + Prop
Science Kickstage	100.4	165.6	Usable
	19.1	29.8	Residual*
	119.4	195.4	Total
	128.7	204.7	Tank + Prop

The residual* row includes 15% margin (requested by mission analysis), 3% residuals, and 1% fuel bias.

Table 16. Reaction Control System Tank Sizing

Mission Type		Qty	Unit Mass	Basic Mass (kg)
Planetary Defense	Fuel Tanks	2	8.92	17.8
	Ox Tanks	2	8.85	17.7
Flight Demo	Fuel Tanks	2	14.89	29.8
	Ox Tanks	2	14.77	29.5
Science Kickstage	Fuel Tanks	3	3.09	9.3
	Ox Tanks	3	3.07	9.2

Pressurant Mass Estimation

The amount of helium needed to pressurize the propulsion systems was calculated for each mission and vehicle. MPS helium was estimated by the thermal analysts. It was assumed that the engine would be autogenously pressurized during the mission but would require helium to initialize start up. This means for the flight demo mission (which requires multiple startups), there would be a higher helium consumption by the MPS. The helium usage was calculated using conventional methods for the RCS – the amount of helium necessary to pressure-regulate the thrusters throughout the entire mission was calculated. For simplicity, both quantities of helium are combined into a single set of tanks, which runs both the MPS and RCS.

Table 17. Helium needed to start the Main Propulsion System engines (fill levels and pressurant required; note that these estimates only apply to the Flight Demo mission as it requires multiple burns)

Case	Burn 1	Burn 2	Burn 3	Burn 4	Tank Size (m3)	Used Pressurant (kg)
LH2_SLSB2	100% (0.34 kg)	14% (5.45 kg)	8% (5.81 kg)		300	11.6
CH4_Starship	100% (0.3 kg)	28% (4.33 kg)	20% (4.78 kg)	10% (5.33 kg)	215	14.74
NH3_Falcon	100% (0.045 kg)	39.4% (0.56 kg)	27.7% (0.66 kg)	13.9% (0.77 kg)	39	2.035

These estimates are only for the Flight Demo mission, as the other two missions only have one engine start. The numbers above use one of our thermal analyst's TankTherm model after getting cross-checked with the other analyst's TankSIM model to anchor the numbers. Both estimates offer a vast improvement over the Level 0 estimations, which used the ideal gas law and resulted in overestimation in the amount of helium needed to pressurize the MPS tank. Even better that their two different methods were in agreement and in-family.

Table 18. Total Helium Mass Needed for Main Propulsion System and Reaction Control System (per vehicle and per mission)

Helium Mass, Per Mission and Vehicle									
1	2	3	4	5	6	7	8	9	
LH2	CH4	NH3	LH2	CH4	NH3	LH2	CH4	NH3	
Planetary Def	Planetary Def	Planetary Def	Flight Demo	Flight Demo	Flight Demo	Science	Science	Science	
2.34	2.3	2.05	11.6	14.74	2.44	2.34	2.3	2.05	MPS GHe
7.2	7.2	7.2	12.8	12.8	12.8	2.8	2.8	2.8	RCS GHe
9.5	9.5	9.3	24.4	27.6	15.3	5.1	5.1	4.8	Total GHe Needed

Pressurant Tank Sizing

The following tanks were sized to contain the helium calculated in the previous section. There are two (2) helium tanks per vehicle. The helium tanks are spherical COPV tanks at a pressure of 4,500 psi and 70 degrees Fahrenheit. A single set of helium tanks pressurize both the MPS and RCS.

Table 19. Helium Tank Details (Mass and Volume)

LH2/Planetary Defense		CH4/Planetary Defense		NH3/Planetary Defense	
GHe Tank Mass (kg)	21	GHe Tank Mass (kg)	21	GHe Tank Mass (kg)	20
GHe Tank Diam (m)	0.61	GHe Tank Diam (m)	0.61	GHe Tank Diam (m)	0.60
GHe Tank Length (m)	Sphere	GHe Tank Length (m)	Sphere	GHe Tank Length (m)	Sphere
GHe Tank Vol (m3)	0.12	GHe Tank Vol (m3)	0.12	GHe Tank Vol (m3)	0.11
LH2/Flight Demo		**CH4/Flight Demo**		**NH3/Flight Demo**	
GHe Tank Mass (kg)	50	GHe Tank Mass (kg)	56	GHe Tank Mass (kg)	32
GHe Tank Diam (m)	0.83	GHe Tank Diam (m)	0.86	GHe Tank Diam (m)	0.71
GHe Tank Length (m)	Sphere	GHe Tank Length (m)	Sphere	GHe Tank Length (m)	Sphere
GHe Tank Vol (m3)	0.30	GHe Tank Vol (m3)	0.34	GHe Tank Vol (m3)	0.19
LH2/Science		**CH4/Science**		**NH3/Science**	
GHe Tank Mass (kg)	12	GHe Tank Mass (kg)	11	GHe Tank Mass (kg)	11
GHe Tank Diam (m)	0.49	GHe Tank Diam (m)	0.49	GHe Tank Diam (m)	0.48
GHe Tank Length (m)	Sphere	GHe Tank Length (m)	Sphere	GHe Tank Length (m)	Sphere
GHe Tank Vol (m3)	0.06	GHe Tank Vol (m3)	0.06	GHe Tank Vol (m3)	0.06

MPS Propellant Volume and Mass

The MPS tank volumes were driven by the vehicle geometry as governed by these constraints:
- LH2 + SLS B2 (Max volume)
- CH4 + Starship (Max mass)
- NH3 + Falcon Heavy (Max volume)

The MPS is assumed to be autogenously pressurized (except for engine startup, which needs GHe). MPS tank masses were sized and analyzed by the structural analyst (detailed information can be found in the structures section).

Table 20. MPS Tank Geometries and Propellant Masses

	Tank Volume (m3)	Propellant Volume (m3)	Usable Propellant Mass (kg)
LH2/SLS B2	300	285	19,566
CH4/Starship	215	204	21,987
NH3/Falcon Heavy	39	37	23,513

Propulsion MELs

Table 21. Liquid Hydrogen Mass Estimation List

Mass Estimation List (MEL)	Planetary Defense	NTP Demo	NTP Kickstage
LH2	Predicted Mass (kg)	Predicted Mass (kg)	Predicted Mass (kg)
Propulsion	5,781.46	5,781.46	5,781.46
Main Propulsion System	5,504.15	5,504.15	5,504.15
NTP Engine	4,648.66	4,648.66	4,648.66
Reactor Core	3,673.20	3,673.20	3,673.20
Engine System	975.46	975.46	975.46
External Shield	720.00	720.00	720.00
LH2 Feed System	121.47	121.47	121.47
Pressurant System	14.02	14.02	14.02
Auxiliary Propulsion System	277.32	277.32	277.32
R1E, 25 lbf thruster	26.40	26.40	26.40
Propellant System	75.86	75.86	75.86
Pressurant System	175.06	175.06	175.06
Non-Prop Fluids	927.14	927.14	927.14
MPS	787.00	787.00	787.00
Autogenous H2	195.66	195.66	195.66
Reserve LH2	0.00	0.00	0.00
Residual LH2	586.98	586.98	586.98
Initial GH2 Pressurant	4.36	4.36	4.36
RCS	140.14	140.14	140.14
Residual	28.40	28.40	28.40
Reserve	85.20	85.20	85.20
Fuel Bias	2.14	2.14	2.14
GHe Pressurant	24.40	24.40	24.40
Useable Propellant	20,134.00	20,134.00	20,134.00
MPS	19,566.00	19,566.00	19,566.00
LH2	19,566.00	19,566.00	19,566.00
RCS	568.00	568.00	568.00
Oxidizer (N2O4)	353.66	353.66	353.66
Fuel (MMH)	214.34	214.34	214.34

Table 22. Methane Mass Estimation List

Mass Estimation List (MEL)	Planetary Defense	NTP Demo	NTP Kickstage
CH4	Predicted Mass (kg)	Predicted Mass (kg)	Predicted Mass (kg)
Propulsion	6,044.15	6,044.15	6,044.15
Main Propulsion System	5,705.75	5,705.75	5,705.75
NTP Engine	4,850.26	4,850.26	4,850.26
External Shield	720.00	720.00	720.00
LH2 Feed System	121.47	121.47	121.47
Pressurant System	14.02	14.02	14.02
Auxiliary Propulsion System	338.40	338.40	338.40
R1E, 25 lbf thruster	26.40	26.40	26.40
Propellant System	120.14	120.14	120.14
Pressurant System	191.86	191.86	191.86
Non-Prop Fluids	3,486.13	3,486.13	3,486.13
MPS	3,209.52	3,209.52	3,209.52
Autogenous CH4	801.29	801.29	801.29
Reserve CH4	0.00	0.00	0.00
Residual CH4	2,403.87	2,403.87	2,403.87
Initial CH4 Pressurant	4.36	4.36	4.36
RCS	276.61	276.61	276.61
Residual (3% of useable)	61.10	61.10	61.10
Oxidizer (N2O4)	38.04	38.04	38.04
Fuel (MMH)	23.06	23.06	23.06
Reserve (2% of useable)	183.30	183.30	183.30
Oxidizer (N2O4)	114.13	114.13	114.13
Fuel (MMH)	69.17	69.17	69.17
Fuel Bias (1% of useable fuel)	4.61	4.61	4.61
GHe Pressurant	27.60	27.60	27.60
Useable Propellant	81,351.00	81,351.00	81,351.00
MPS	80,129.00	80,129.00	80,129.00
CH4	80,129.00	80,129.00	80,129.00
RCS	1,222.00	1,222.00	1,222.00
Oxidizer (N2O4)	760.87	760.87	760.87
Fuel (MMH)	461.13	461.13	461.13

Table 23. Ammonia Mass Estimation List

Mass Estimation List (MEL)	Planetary Defense	NTP Demo	NTP Kickstage
NH3	Predicted Mass (kg)	Predicted Mass (kg)	Predicted Mass (kg)
Propulsion	5,922.95	5,922.95	5,922.95
Main Propulsion System	5,705.75	5,705.75	5,705.75
NTP Engine	4,850.26	4,850.26	4,850.26
External Shield	720.00	720.00	720.00
NH3 Feed System	121.47	121.47	121.47
Pressurant System	14.02	14.02	14.02
Auxiliary Propulsion System	217.20	217.20	217.20
R1E, 25 lbf thruster	26.40	26.40	26.40
Propellant System	66.14	66.14	66.14
Pressurant System	124.66	124.66	124.66
Non-Prop Fluids	1,048.62	1,048.62	1,048.62
MPS	944.88	944.88	944.88
RCS	103.74	103.74	103.74
Residual (3% of useable)	21.70	21.70	21.70
Reserve (2% of useable)	65.10	65.10	65.10
Fuel Bias (1% of useable fuel)	1.64	1.64	1.64
GHe Pressurant	15.30	15.30	15.30
Useable Propellant	23,947.00	23,947.00	23,947.00
MPS	23,513.00	23,513.00	23,513.00
NH3	23,513.00	23,513.00	23,513.00
RCS	434.00	434.00	434.00
Oxidizer (N2O4)	270.23	270.23	270.23
Fuel (MMH)	163.77	163.77	163.77

Conclusions and Recommendations

The objective for propulsion during this study was to design both main propulsion systems (MPS) and reaction control systems (RCS) for the three missions and three vehicle designs – for a total of nine options. Each vehicle corresponded to a different propellant type, which meant the propulsion design got complicated fast.

Each launch vehicle design uses a different MPS propellant – though all three are nuclear thermal propulsion (NTP). Each propellant tank was sized based either on maximum volume or mass allowed by the launch vehicle. This was decided by the study lead to strategically assess each propellant type during this study cycle.

LH2 + SLS B2 (Max volume)
CH4 + Starship (Max mass)
NH3 + Falcon Heavy (Max volume)

The RCS was designed to use the same propellants (MMH/NTO) and use the same thrusters (25 lbf R-1E bipropellant thrusters) across all options. The propellant tank geometry was also sized for each mission and held the same for each vehicle configuration for simplicity. To save mass, the tanks were sized such that a single set of tanks could be removed from the smaller vehicles since the propellant tank geometry was being maintained across all the different vehicle options.

Identification of Concept Maturity and Recommendations

NTP technology is cutting edge, but not as proven as conventional liquid propulsion systems. The use of CH4 and NH3 in NTP applications represents an even greater departure from proven methods than LH2-based systems. This pioneering nature resulted in multiple unexpected obstacles throughout our study:

Potential for carbon coking and methane pyrolysis (as discussed in the paper "Methane-based Nuclear Thermal Propulsion Engine Considerations" by Daria Nikitaeva, Corey D. Smith, and Matthew Duchek)[4]
Lack of detailed engine designs to reference (for example, this came into play when trying to find the desired inlet pressure of the fuel pumps)
Rough "back of napkin" estimates given for engine thrust upscaling due to differences in propellant density rather than proven examples

In cases where methane or ammonia options demonstrate compelling advantages, additional research focusing specifically on carbon coking and methane pyrolysis phenomena is recommended. Significant resources should also be allocated to advance NTP engine modeling for these propellants. Meanwhile, if liquid hydrogen remains a competitive option, it carries the benefit of leveraging extensive existing research and substantial previous investment in its development.

[4] **Nikitaeva, D., Smith, C. D., & Duchek, M. (2025, May 4–8).** Methane-based nuclear thermal propulsion engine considerations. *In Proceedings of the Nuclear and Emerging Technologies for Space (NETS 2025)* American Nuclear Society. Presented in Huntsville, AL. https://ntrs.nasa.gov/citations/20250004353

Power

Overview

The objective is to design and size a power system for each of 3 cases

- LH2 propellant launched on SLS
- CH4 propellant launched on a SpaceX Starship
- NH3 propellant launched on a SpaceX Falcon Heavy

Each of these cases uses a prescribed NTP engine and is designed to maximize the volume or mass of the launch vehicle whichever is most constraining. Each of these cases must close 3 Design Reference Missions as described in Mission Analysis above:

- Planetary Defense (impacting a NEO to divert its course)
- CFM Demo
- Science Kick Stage (impart an impulse to a separate science vehicle)

Each power system case is thus designed to meet the most demanding requirements of each of the DRMs. This means that the designs may not be optimal for any of the missions, although it will close.

Ground Rules and Assumptions

The following table lists the major Ground Rules and Assumptions informing and constraining the power system design.

Table 24. Power Ground Rules and Assumptions

GR/Assumption	Item	Value	Rationale
Ground Rule	Max Acceleration	2.5g	SNP Ground Rule
Assumption	Array Type	UltraFlex	Based on Max Acc
Assumption	PV Cell Type	SolAero IMMα	Selected for radiation Tolerance
Assumption	Battery Packing Factor	1.3	Typical for small Li-Ion batteries
Assumption	Solar Irradiance AM0	1367 W/m^2	Typical ACO value
Assumption	Operational Array Conversion Efficiency	27%	Conservative Value
Assumption	Array yearly degradation	3%	Typical ACO value

The max acceleration (2.5 g) for the Solar Array structure is required to support the maximum thrust of the NTP Engine. 2.5g is the maximum acceleration tested for an UltraFlex array.

Power Requirements

The power requirements for each case are listed in the tables below (by mission phase):

Table 25. Power Requirements for each case

SLS LH2

Source	Init	NTP	Coast	Dormant	TCM	Impact
Avionics	741	1170	741	331	1050	1644
Thermal	616	129	6979 *	0	165	129
MPS	156	156	0	0	0	0
RCS	278	210	0	0	278	0
Totals	1791	1665	7720	331	1493	1773
Power Growth (30%)	537.3	499.5	2316	99.3	447.9	531.9
Predicted Power	**2328.3**	**2164.5**	**10036**	**430.3**	**1940.9**	**2304.9**

*Includes 6377 w/ cryo

Starship CH4

Source	Init	NTP	Coast	Dormant	TCM	Impact
Avionics	741	1170	741	331	1050	1644
Thermal	616	129	1148 *	0	165	129
MPS	156	156	0	0	0	0
RCS	278	210	0	0	278	0
Totals	1791	1665	1889	331	1493	1773
Power Growth (30%)	537.3	499.5	566.7	99.3	447.9	531.9
Predicted Power	**2328.3**	**2164.5**	**2455.7**	**430.3**	**1940.9**	**2304.9**

*Includes 546 w/ cryo

Falcon Heavy NH3

Source	Init	NTP	Coast	Dormant	TCM	Impact
Avionics	741	1170	741	331	1050	1644
Thermal	616	129	602 *	0	165	129
MPS	156	156	0	0	0	0
RCS	232	210	0	0	232	0
Totals	1745	1665	1343	331	1447	1773
Power Growth (30%)	523.5	499.5	402.9	99.3	434.1	531.9
Predicted Power	**2268.5**	**2164.5**	**1745.9**	**430.3**	**1881.1**	**2304.9**

*Includes no cryo power

The major difference in power requirements for the cases is the cryo-cooler power.

Sizing

Each of the major power subsystems are sized at fidelity Level 1 using either physics-based tools or representative off-the-shelf components.

RadioIsotope ThermoElectric Generator

The Planetary Defense Mission is unique in that its trajectory takes it 8.9 AU away from the Sun – too far to use solar arrays for its power. For this mission, an RTG-based power system is added to power the spacecraft when it is more than 1.1 AU from the Sun. The vehicle is scarred for the addition of two general-purpose heat sources/radioisotope thermoelectric generators (GPHS RTGs) – each providing 216W – 300W of power for the dormant mission phase. Each of the (2) GPHS RTGs has a mass of 57kg.

Table 26. Predicted Masses for each Launch Vehicle

NTP Fuel	Array Diameter	Predicted Mass
LH2 - SLS	6.5m	252.5 kg
CH4 - Starship	3.8m	87.5 kg
NH3 – Falcon	3.4m	70.0 kg

Batteries

The batteries are sized by the "Init" mission phase, and are used to power the craft while it is de-Tumbled and the arrays are deployed. This phase of the mission requires 2165W of power for 1 hour for all 3 cases. 4 Li Ion batteries are required, 9.1kg each.

Solar Arrays

For the Cryo cases (LH2 & CH4), the "Coast" mission phase (when the cryocoolers are running) sizes the Solar Arrays. For the passive, NH3 case, the NTP propulsive mission phase is the sizing case.

Power Electronics

The Power Electronics is sized from a special VPX 3U (230mm x 160m) board set.

The sizing is performed by determining the number of each type of board (Array Regulation, Battery Management, Power Switching, etc), adding up the board masses, and then using a mechanistic board enclosure sizing tool to estimate the mass of the enclosure. Because the Planetary Defense mission is highly critical, the power electronics mass includes complete redundant units. The CFM Demo and Science Kick Stage missions require only one, resulting in half the mass

Mass Summary (Power)

Table 27 details the summary of predicted masses for all three cases.

Table 27. Total Predicted Mass of Power System Elements

	Total Predicted Mass			Notes
Item	LH2	CH4	NH3	
RTG Kit	125.4	125.4	125.4	Planetary Defense Only
Solar Arrays	252.5	87.5	70.0	
Power Electronics	84.8	55.0	46.5	No redundancy for CFM, Science
Batteries	36.4	36.4	36.4	
Cabling	115.8	60.9	54.0	
Total System	614.9	365.2	332.3	

Conclusions and Recommendations

The power system design is a conventional, low risk design. Although the GPHS RTG uses Pu238 (in very limited supply), a planetary emergency would motivate an immediate increase in production. The actual production process (from Np 237 found in spent nuclear fuel) is well understood and easily scalable. The design is not optimal – the power system is sized to worst-case (planetary defense) mission requirements, and is thus somewhat over-sized for the CFM Demo A.

Structures

Overview

The SNP mission study is comprised of a total of 54 stage designs which consider various launch vehicles, propellant types, mission profiles, and thermal control options. Table 28 below provides the trade space considered for the study.

Structural sizing/optimization was performed on all primary structure for each of three selected point designs to ensure that all structural components meet or exceed strength and stability requirements per NASA guidelines. A complete set of subsystem mass properties has been provided in the study MEL with some line items containing side notes to highlight items that are mission profile specific (i.e. Planetary Defense, Science Kick-Stage, Tech Demo).

Table 28. Trade Spaces Considered

Propellant	LH2	CH4	NH3
Launch Vehicle	SLS Block II	Starship	Falcon Heavy
Mission Profile	Planetary Defense	Technology Demo	Science Kickstage
Thermal Control	Cryocooler (Active)	Passive	

Due to study time constraints, primary structures were only sized for the three following point designs:

- SLS/LH2/Cryocooler (Planetary Defense) – NTP propulsion stage which uses the SLS launch vehicle. In-space propellant is assumed to be liquid hydrogen.

- Starship/CH4/Cryocooler (Planetary Defense) – NTP propulsion stage which uses the Starship launch vehicle. In-space propellant is assumed to be methane.

- Falcon Heavy/NH3 (Planetary Defense) – NTP propulsion stage which uses the Falcon Heavy launch vehicle. In-space propellant is assumed to be liquid ammonia.

All primary structures assume typical 2,000 series aluminum construction with exception of struts that interface with the MPS tank where S-Glass struts are used for passive thermal isolation. Grid stiffened panels are used for tank barrels and for the equipment shelf where higher bending stiffness is needed.

Active thermal control is assumed for the LH2 and CH4 point designs. This is achieved using Cryocoolers to prevent MPS fuel boiloff. Note that the NH3 point design relies solely on passive thermal control using SOFI and MLI where required.

Non-structural mass is included as areal mass density on all tank barrels to represent MMOD and passive thermal protection (SOFI, MLI, etc).

Ground Rules and Assumptions

- All primary structure must meet or exceed strength and stability requirements as stated in NASA STD5001B assuming a proto-flight analysis approach
    - FSu=1.4 and FSy=1.25 for Metallic Structures
    - FSu=1.4 for composite structures
- Truss joints and fittings are assumed to have a mass equivalent to 50% of the truss member mass
- Secondary Structure mass is assumed to have a mass equivalent to 5% of the primary structure mass
- An MPS tank ullage pressure of 30psi is assumed to exist for all earth to orbit load cases
- Applied inertial loads for all earth to orbit cases assumes a worst on worst axial / lateral load combination obtained from each launch vehicles payload users guide.
- Mass growth allowance is set in accordance with the AIAA MGA standards. Note that MGA selected is related to the concept maturity and level of analysis performed

Model Description

Structural sizing/optimization was performed for the three point designs shown below in Figure 12. Each point design consists of an MPS tank with forward and aft RCS bus assemblies. An NTP motor is fitted to the aft end of the propulsion stage using a thrust cone. An adapter fitted to the forward end of the inverted propulsion stage is used to interface the NTP stage to each respective launch vehicle. Mounting the propulsion stage in an inverted orientation allows for more efficient packaging inside of the launch vehicle shroud.

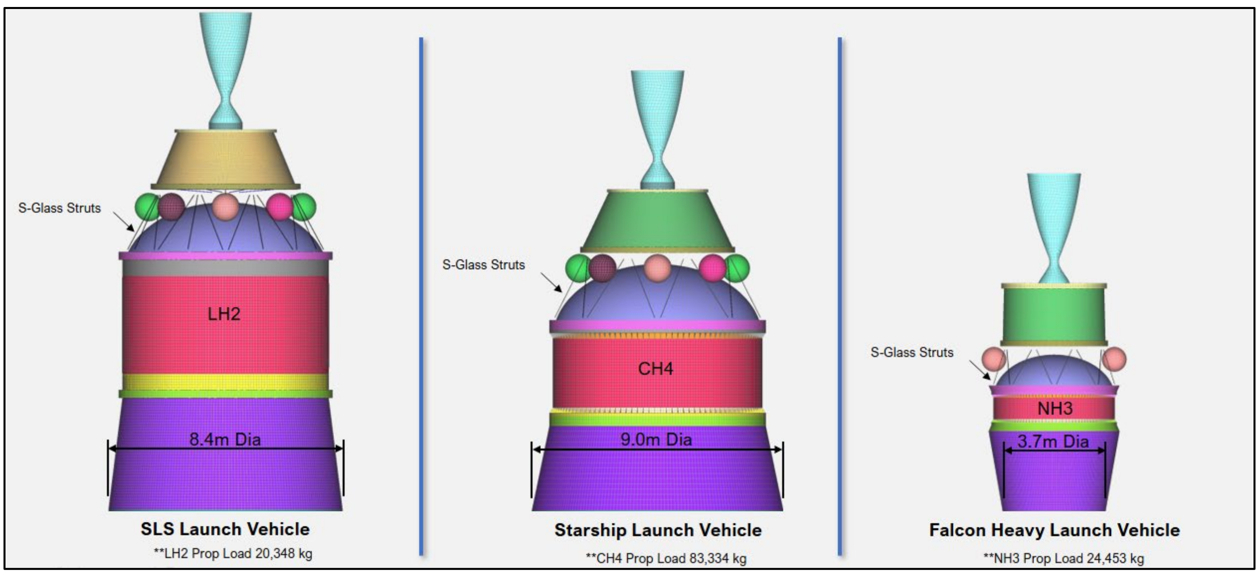

Figure 12. NTP DAC Point Designs

Figure 13 depicts an exploded view of the SLS/LH2 propulsion stage without the launch adapter. Avionics, Power, and Thermal subsystem components are mounted underneath a grid stiffened plate at the forward end of the stage. Solar arrays and radiators exist on the other side of the grid stiffened plate and are represented in each model as point masses connected to structure using Nastran Multi Point Constraint definitions. RCS fuel, oxidizer, and press bottles are mounted at each end of the MPS tank using Nastran Multi Point Constraint definitions.

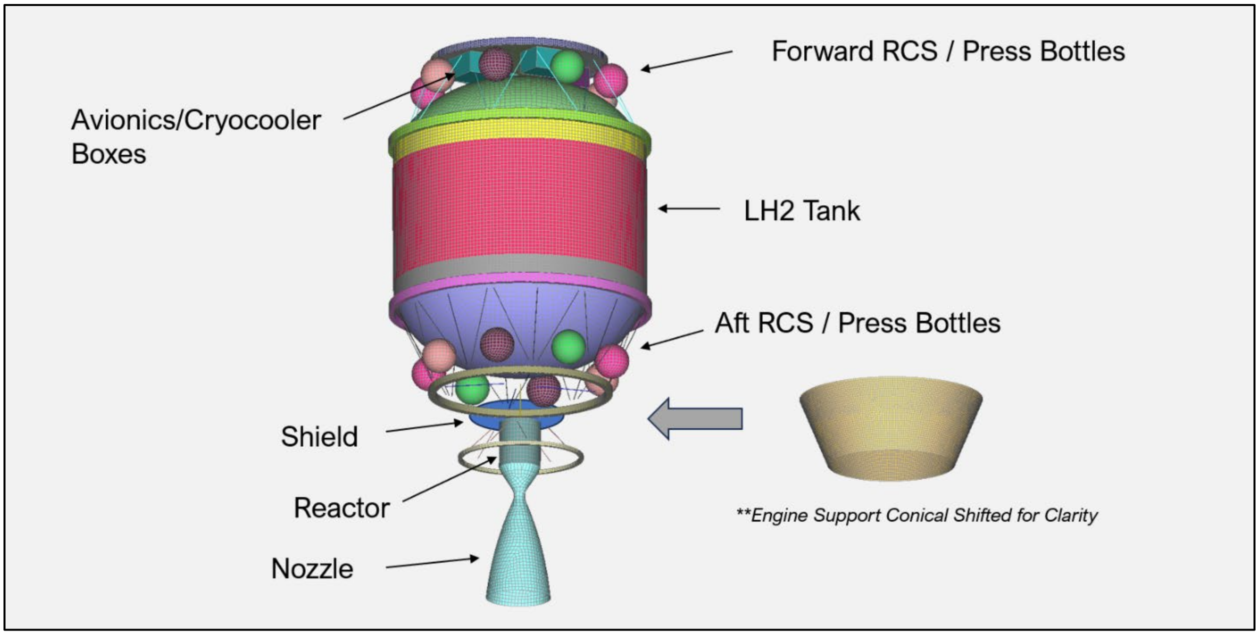

Figure 13. SLS/LH2 NTP Propulsion Stage (Typical)

Loads and Constraints

Each point design was analyzed using worst on worst axial/lateral inertial loads obtained from each variant's respective payload users guide. Figure 14 depicts the applied loads and constraints for each of the three-point designs evaluated.

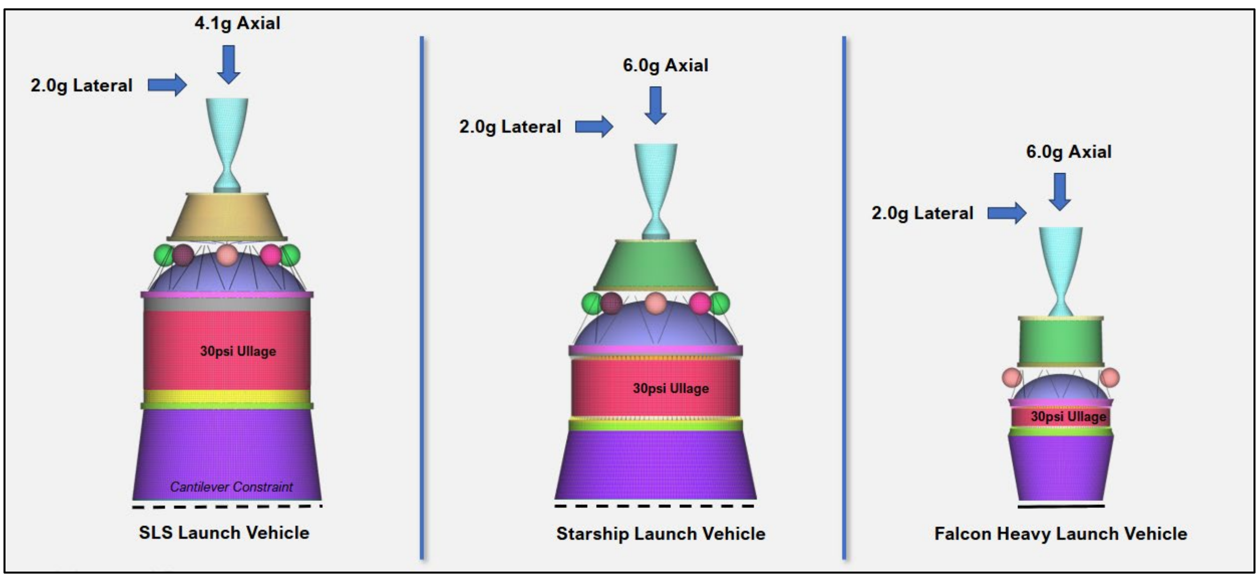

Figure 14. Applied Loads and Constraints

Conclusions and Recommendations

- All structures meet or exceed strength and stability requirements as stated in NASA STD5001B assuming a Proto-Flight analysis approach.

- Sizing for in-space propulsion events is enveloped by the earth to orbit load cases. This is due to the high mass of the NTP reactor and high G loads experienced during earth launch/ascent.

- Much forward work remains to detail the NTP motor support system. The current design represents a very notional set of support struts which will need to be carefully evaluated as the design matures.

- All joints and fittings are currently represented using Nastran Multi Point Constraint definitions. Future design work is needed to provide detail designs for all joints and fittings.

Mass Summaries

Table 29. LH2_SLSB2 Concept Mass Summary

SNP Concept - LH2 / SLS B2

		Planetary Defense						NTP Demo						NTP Kickstage					
	Mass Estimation List (MEL)	Qty	Unit Mass (kg)	Basic Mass (kg)	MGA (%)	MGA (kg)	Predicted Mass (kg)	Qty	Unit Mass (kg)	Basic Mass (kg)	MGA (%)	MGA (kg)	Predicted Mass (kg)	Qty	Unit Mass (kg)	Basic Mass (kg)	MGA (%)	MGA (kg)	Predicted Mass (kg)
1.0	Structures			4,918.61	18.3%	898.35	5,816.96			4,918.61	18.3%	898.35	5,816.96			4,918.61	18.3%	898.35	5,816.96
1.1	Forward RCS BUS			867.07		173.41	1,040.48			867.07		173.41	1,040.48			867.07		173.41	1,040.48
1.2	LH2 Tank			*2,590.84*		*518.17*	3,109.01			*2,590.84*		*518.17*	3,109.01			*2,590.84*		*518.17*	3,109.01
1.3	Engine Thrust Structure			615.00		123.00	738.00			615.00		123.00	738.00			615.00		123.00	738.00
1.4	Truss Joints and Fittings	1	81.04	81.04	20.0%	16.21	97.25	1	81.04	81.04	20.0%	16.21	97.25	1	81.04	81.04	20.0%	16.21	97.25
1.5	Secondary Structures	1	195.54	195.54	20.0%	39.11	234.65	1	195.54	195.54	20.0%	39.11	234.65	1	195.54	195.54	20.0%	39.11	234.65
1.6	MMOD	1	569.12	569.12	5.0%	28.46	597.57	1	569.12	569.12	5.0%	28.46	597.57	1	569.12	569.12	5.0%	28.46	597.57
2.0	Propulsion			4,823.92	19.8%	957.54	5,781.46			4,823.92	19.8%	957.54	5,781.46			4,823.92	19.8%	957.54	5,781.46
2.1	Main Propulsion System			4,587.64	20.0%	916.50	5,504.15			4,587.64	20.0%	916.50	5,504.15			4,587.64	20.0%	916.50	5,504.15
2.2	Auxiliary Propulsion System			236.28	17.4%	41.04	277.32			236.28	17.4%	41.04	277.32			236.28	17.4%	41.04	277.32
3.0	Power			490.12	25.4%	124.73	614.85			376.12	30.1%	113.33	489.45			376.12	30.1%	113.33	489.45
3.1	GPHS RTG	2	57.00	114.00	10.0%	11.40	125.40	This item not used for this mission						This item not used for this mission					
3.2	Solar Array Wing (W/O Boom)	2	92.00	184.00	25.0%	46.00	230.00	2	92.00	184.00	25.0%	46.00	230.00	2	92.00	184.00	25.0%	46.00	230.00
3.3	Solar Array Drive Actuator	2	9.00	18.00	25.0%	4.50	22.50	2	9.00	18.00	25.0%	4.50	22.50	2	9.00	18.00	25.0%	4.50	22.50
3.4	Power Electronics	2	33.90	67.80	25.0%	16.95	84.75	2	33.90	67.80	25.0%	16.95	84.75	2	33.90	67.80	25.0%	16.95	84.75
3.5	Secondary Battery	4	7.28	29.12	25.0%	7.28	36.40	4	7.28	29.12	25.0%	7.28	36.40	4	7.28	29.12	25.0%	7.28	36.40
3.6	Solar Array Harness	2	15.60	31.20	50.0%	15.60	46.80	2	15.60	31.20	50.0%	15.60	46.80	2	15.60	31.20	50.0%	15.60	46.80
3.7	Spacecraft Harness	1	46.00	46.00	50.0%	23.00	69.00	1	46.00	46.00	50.0%	23.00	69.00	1	46.00	46.00	50.0%	23.00	69.00
4.0	Avionics			485.40	15.5%	75.35	560.75			601.13	17.8%	107.08	708.21			324.33	16.3%	52.87	377.20
4.1	Command and Data Handling (C&DH)			186.72	19.6%	36.56	223.28			461.12	19.5%	90.07	551.19			186.72	19.6%	36.56	223.28
4.2	GN&C			116.24	14.9%	17.32	133.56			7.30	3.0%	0.22	7.52			7.30	3.0%	0.22	7.52
4.3	Communications			105.04	10.4%	10.97	116.01			56.31	11.2%	6.32	62.63			43.91	9.8%	4.32	48.23
4.4	Vehicle systems, heaters, etc.			77.40	13.6%	10.50	87.90			76.40	13.7%	10.47	86.87			86.40	13.6%	11.77	98.17
5.0	Thermal			2,124.19	40.0%	849.68	2,973.87			2,124.19	40.0%	849.68	2,973.87			2,124.19	40.0%	849.68	2,973.87
5.1	Avionics Thermal			334.56	40.0%	133.82	468.38			334.56	40.0%	133.82	468.38			334.56	40.0%	133.82	468.38
5.2	RCS Thermal			44.75	40.0%	17.90	62.65			44.75	40.0%	17.90	62.65			44.75	40.0%	17.90	62.65
5.3	CFM Thermal			1,744.89	40.0%	697.95	2,442.84			1,744.89	40.0%	697.95	2,442.84			1,744.89	40.0%	697.95	2,442.84
	Dry Mass			**12,842.24**	**22.6%**	**2,905.66**	**15,747.90**			**12,843.97**	**22.8%**	**2,925.99**	**15,769.96**			**12,567.17**	**22.9%**	**2,871.77**	**15,438.95**
	System Level Mass Margin (15%)				15.0%	1,926.34	1,926.34				15.0%	1,926.60	1,926.60				15.0%	1,885.08	1,885.08
	Total Predicted System Dry Mass						**17,674.23**						**17,696.55**						**17,324.02**
6.0	Non-Prop Fluids			927.14			927.14			927.14			927.14			927.14			927.14
6.1	MPS			787.00			787.00			787.00			787.00			787.00			787.00
6.2	RCS			140.14			140.14			140.14			140.14			140.14			140.14
7.0	Useable Propellant			20,134.00			20,134.00			20,134.00			20,134.00			20,134.00			20,134.00
7.1	MPS			19,566.00			19,566.00			19,566.00			19,566.00			19,566.00			19,566.00
7.2	RCS			568.00			568.00			568.00			568.00			568.00			568.00
	Total Stage Gross Mass			**33,903.39**	**14.3%**	**4,831.99**	**38,735.38**			**33,905.12**	**14.3%**	**4,852.58**	**38,757.70**			**33,628.32**	**14.1%**	**4,756.85**	**38,385.16**
	Payload Adapter	1	1,378.68	1,378.68	20.0%	275.74	1,654.42	1	1,378.68	1,378.68	20.0%	275.74	1,654.42	1	1,378.68	1,378.68	20.0%	275.74	1,654.42
	Total Launch Mass						**40,389.80**						**40,412.12**						**40,039.59**

Table 30. CH4/Starship Concept Mass Summaries

SNP Concept - CH4 / Starship		Planetary Defense						NTP Demo						NTP Kickstage					
	Mass Estimation List (MEL)	Qty	Unit Mass (kg)	Basic Mass (kg)	MGA (%)	MGA (kg)	Predicted Mass (kg)	Qty	Unit Mass (kg)	Basic Mass (kg)	MGA (%)	MGA (kg)	Predicted Mass (kg)	Qty	Unit Mass (kg)	Basic Mass (kg)	MGA (%)	MGA (kg)	Predicted Mass (kg)
1.0	Structures			5,098.95	19.0%	966.73	6,065.68			5,098.95	19.0%	966.73	6,065.68			5,098.95	19.0%	966.73	6,065.68
1.1	Forward RCS BUS			714.01		142.80	856.81			714.01		142.80	856.81			714.01		142.80	856.81
1.2	LH2 Tank			*2,994.36*		*598.87*	*3,593.24*			*2,994.36*		*598.87*	*3,593.24*			*2,994.36*		*598.87*	*3,593.24*
1.3	Engine Thrust Structure			765.07		153.01	918.08			765.07		153.01	918.08			765.07		153.01	918.08
1.4	Truss Joints and Fittings	1	79.76	79.76	20.0%	15.95	95.72	1	79.76	79.76	20.0%	15.95	95.72	1	79.76	79.76	20.0%	15.95	95.72
1.5	Secondary Structures	1	192.01	192.01	20.0%	38.40	230.41	1	192.01	192.01	20.0%	38.40	230.41	1	192.01	192.01	20.0%	38.40	230.41
1.6	MMOD	1	353.74	353.74	5.0%	17.69	371.43	1	353.74	353.74	5.0%	17.69	371.43	1	353.74	353.74	5.0%	17.69	371.43
2.0	Propulsion			5,041.81	19.9%	1,002.34	6,044.15			5,041.81	19.9%	1,002.34	6,044.15			5,041.81	19.9%	1,002.34	6,044.15
2.1	Main Propulsion System			4,755.64	20.0%	950.10	5,705.75			4,755.64	20.0%	950.10	5,705.75			4,755.64	20.0%	950.10	5,705.75
2.2	Auxiliary Propulsion System			286.16	18.3%	52.24	338.40			286.16	18.3%	52.24	338.40			286.16	18.3%	52.24	338.40
3.0	Power			297.72	22.7%	67.48	365.20			183.72	30.5%	56.08	239.80			183.72	30.5%	56.08	239.80
3.1	GPHS RTG	2	57.00	114.00	10.0%	11.40	125.40	This item not used for this mission						This item not used for this mission					
3.2	Solar Array Wing (W/O Boom)	2	32.00	64.00	25.0%	16.00	80.00	2	32.00	64.00	25.0%	16.00	80.00	2	32.00	64.00	25.0%	16.00	80.00
3.3	Solar Array Drive Actuator	2	3.00	6.00	25.0%	1.50	7.50	2	3.00	6.00	25.0%	1.50	7.50	2	3.00	6.00	25.0%	1.50	7.50
3.4	Power Electronics	2	22.00	44.00	25.0%	11.00	55.00	2	22.00	44.00	25.0%	11.00	55.00	2	22.00	44.00	25.0%	11.00	55.00
3.5	Secondary Battery	4	7.28	29.12	25.0%	7.28	36.40	4	7.28	29.12	25.0%	7.28	36.40	4	7.28	29.12	25.0%	7.28	36.40
3.6	Solar Array Harness	2	9.80	19.60	50.0%	9.80	29.40	2	9.80	19.60	50.0%	9.80	29.40	2	9.80	19.60	50.0%	9.80	29.40
3.7	Spacecraft Harness	1	21.00	21.00	50.0%	10.50	31.50	1	21.00	21.00	50.0%	10.50	31.50	1	21.00	21.00	50.0%	10.50	31.50
4.0	Avionics			485.40	15.5%	75.35	560.75			601.13	17.8%	107.08	708.21			324.33	16.3%	52.87	377.20
4.1	Command and Data Handling (C&DH)			186.72	19.6%	36.56	223.28			461.12	19.5%	90.07	551.19			186.72	19.6%	36.56	223.28
4.2	GN&C			116.24	14.9%	17.32	133.56			7.30	3.0%	0.22	7.52			7.30	3.0%	0.22	7.52
4.3	Communications			105.04	10.4%	10.97	116.01			56.31	11.2%	6.32	62.63			43.91	9.8%	4.32	48.23
4.4	Vehicle systems, heaters, etc.			77.40	13.6%	10.50	87.90			76.40	13.7%	10.47	86.87			86.40	13.6%	11.77	98.17
5.0	Thermal			976.01	40.0%	390.40	1,366.41			976.01	40.0%	390.40	1,366.41			976.01	40.0%	390.40	1,366.41
5.1	Avionics Thermal			334.56	40.0%	133.82	468.38			334.56	40.0%	133.82	468.38			334.56	40.0%	133.82	468.38
5.2	RCS Thermal			44.75	40.0%	17.90	62.65			44.75	40.0%	17.90	62.65			44.75	40.0%	17.90	62.65
5.3	CFM Thermal			596.70	40.0%	238.68	835.39			596.70	40.0%	238.68	835.39			596.70	40.0%	238.68	835.39
	Dry Mass			**11,899.89**	**21.0%**	**2,502.31**	**14,402.19**			**11,901.62**	**21.2%**	**2,522.64**	**14,424.25**			**11,624.82**	**21.2%**	**2,468.43**	**14,093.24**
	System Level Mass Margin (15%)				15.0%	1,784.98	1,784.98				15.0%	1,785.24	1,785.24				15.0%	1,743.72	1,743.72
	Total Predicted System Dry Mass						**16,187.18**						**16,209.50**						**15,836.96**
6.0	Non-Prop Fluids			3,486.13			3,486.13			3,486.13			3,486.13			3,486.13			3,486.13
6.1	MPS			3,209.52			3,209.52			3,209.52			3,209.52			3,209.52			3,209.52
6.2	RCS			276.61			276.61			276.61			276.61			276.61			276.61
7.0	Useable Propellant			81,351.00			81,351.00			81,351.00			81,351.00			81,351.00			81,351.00
7.1	MPS			80,129.00			80,129.00			80,129.00			80,129.00			80,129.00			80,129.00
7.2	RCS			1222.00			1,222.00			1222.00			1,222.00			1222.00			1,222.00
	Total Stage Gross Mass			**96,737.02**	**4.4%**	**4,287.29**	**101,024.31**			**96,738.75**	**4.5%**	**4,307.88**	**101,046.63**			**96,461.95**	**4.4%**	**4,212.15**	**100,674.09**
	Payload Adapter	1	1,260.77	1,260.77	0.0%	0.00	1,260.77	1	1,260.77	1,260.77	0.0%	0.00	1,260.77	1	1,260.77	1,260.77	0.0%	0.00	1,260.77
	Total Launch Mass						**102,285.08**						**102,307.40**						**101,934.87**

Table 31. NH3/Falcon Heavy Concept

SNP Concept - NH3 / Falcon Heavy		Planetary Defense						NTP Demo						NTP Kickstage					
	Mass Estimation List (MEL)	Qty	Unit Mass (kg)	Basic Mass (kg)	MGA (%)	MGA (kg)	Predicted Mass (kg)	Qty	Unit Mass (kg)	Basic Mass (kg)	MGA (%)	MGA (kg)	Predicted Mass (kg)	Qty	Unit Mass (kg)	Basic Mass (kg)	MGA (%)	MGA (kg)	Predicted Mass (kg)
1.0	Structures			1,491.86	19.3%	288.00	1,779.86			1,491.86	19.3%	288.00	1,779.86			1,491.86	19.3%	288.00	1,779.86
1.1	Forward RCS BUS			297.11	20.0%	59.42	356.53			297.11	20.0%	59.42	356.53			297.11	20.0%	59.42	356.53
1.2	LH2 Tank			*739.32*	*20.0%*	*147.86*	*887.18*			*739.32*	*20.0%*	*147.86*	*887.18*			*739.32*	*20.0%*	*147.86*	*887.18*
1.3	Engine Thrust Structure			294.39	20.0%	58.88	353.27			294.39	20.0%	58.88	353.27			294.39	20.0%	58.88	353.27
1.4	Truss Joints and Fittings	1	35.78	35.78	20.0%	7.16	42.93	1	35.78	35.78	20.0%	7.16	42.93	1	35.78	35.78	20.0%	7.16	42.93
1.5	Secondary Structures	1	56.10	56.10	20.0%	11.22	67.32	1	56.10	56.10	20.0%	11.22	67.32	1	56.10	56.10	20.0%	11.22	67.32
1.6	MMOD	1	69.17	69.17	5.0%	3.46	72.63	1	69.17	69.17	5.0%	3.46	72.63	1	69.17	69.17	5.0%	3.46	72.63
2.0	Propulsion			4,942.61	19.8%	980.34	5,922.95			4,942.61	19.8%	980.34	5,922.95			4,942.61	19.8%	980.34	5,922.95
2.1	Main Propulsion System			4,755.64	20.0%	950.10	5,705.75			4,755.64	20.0%	950.10	5,705.75			4,755.64	20.0%	950.10	5,705.75
2.2	Auxiliary Propulsion System			186.96	16.2%	30.24	217.20			186.96	16.2%	30.24	217.20			186.96	16.2%	30.24	217.20
3.0	Power			272.32	22.0%	59.98	332.30			158.32	30.7%	48.58	206.90			158.32	30.7%	48.58	206.90
3.1	GPHS RTG	2	57.00	114.00	10.0%	11.40	125.40	This item not used for this mission						This item not used for this mission					
3.2	Solar Array Wing (W/O Boom)	2	25.00	50.00	25.0%	12.50	62.50	2	25.00	50.00	25.0%	12.50	62.50	2	25.00	50.00	25.0%	12.50	62.50
3.3	Solar Array Drive Actuator	2	3.00	6.00	25.0%	1.50	7.50	2	3.00	6.00	25.0%	1.50	7.50	2	3.00	6.00	25.0%	1.50	7.50
3.4	Power Electronics	2	18.60	37.20	25.0%	9.30	46.50	2	18.60	37.20	25.0%	9.30	46.50	2	18.60	37.20	25.0%	9.30	46.50
3.5	Secondary Battery	4	7.28	29.12	25.0%	7.28	36.40	4	7.28	29.12	25.0%	7.28	36.40	4	7.28	29.12	25.0%	7.28	36.40
3.6	Solar Array Harness	2	9.20	18.40	50.0%	9.20	27.60	2	9.20	18.40	50.0%	9.20	27.60	2	9.20	18.40	50.0%	9.20	27.60
3.7	Spacecraft Harness	1	17.60	17.60	50.0%	8.80	26.40	1	17.60	17.60	50.0%	8.80	26.40	1	17.60	17.60	50.0%	8.80	26.40
4.0	Avionics			469.20	15.5%	72.92	542.12			584.93	17.9%	104.65	689.58			308.13	16.4%	50.44	358.57
4.1	Command and Data Handling (C&DH)			186.72	19.6%	36.56	223.28			461.12	19.5%	90.07	551.19			186.72	19.6%	36.56	223.28
4.2	GN&C			116.24	14.9%	17.32	133.56			7.30	3.0%	0.22	7.52			7.30	3.0%	0.22	7.52
4.3	Communications			105.04	10.4%	10.97	116.01			56.31	11.2%	6.32	62.63			43.91	9.8%	4.32	48.23
4.4	Vehicle systems, heaters, etc.			61.20	13.2%	8.07	69.27			60.20	13.4%	8.04	68.24			70.20	13.3%	9.34	79.54
5.0	Thermal			400.96	40.0%	160.39	561.35			400.96	40.0%	160.39	561.35			400.96	40.0%	160.39	561.35
5.1	Avionics Thermal			244.03	40.0%	97.61	341.64			244.03	40.0%	97.61	341.64			244.03	40.0%	97.61	341.64
5.2	RCS Thermal			44.75	40.0%	17.90	62.65			44.75	40.0%	17.90	62.65			44.75	40.0%	17.90	62.65
5.3	MPS Thermal			112.18	40.0%	44.87	157.06			112.18	40.0%	44.87	157.06			112.18	40.0%	44.87	157.06
	Dry Mass			**7,576.95**	**20.6%**	**1,561.62**	**9,138.58**			**7,578.68**	**20.9%**	**1,581.95**	**9,160.64**			**7,301.88**	**20.9%**	**1,527.74**	**8,829.62**
	System Level Mass Margin (15%)				15.0%	1,136.54	1,136.54				15.0%	1,136.80	1,136.80				15.0%	1,095.28	1,095.28
	Total Predicted System Dry Mass						**10,275.12**						**10,297.44**						**9,924.91**
6.0	Non-Prop Fluids			1,048.62			1,048.62			1,048.62			1,048.62			1,048.62			1,048.62
6.1	MPS			944.88			944.88			944.88			944.88			944.88			944.88
6.2	RCS			103.74			103.74			103.74			103.74			103.74			103.74
7.0	Useable Propellant			23,947.00			23,947.00			23,947.00			23,947.00			23,947.00			23,947.00
7.1	MPS			23,513.00			23,513.00			23,513.00			23,513.00			23,513.00			23,513.00
7.2	RCS			434.00			434.00			434.00			434.00			434.00			434.00
	Total Stage Gross Mass			**32,572.57**	**8.3%**	**2,698.17**	**35,270.74**			**32,574.30**	**8.3%**	**2,718.76**	**35,293.06**			**32,297.50**	**8.1%**	**2,623.02**	**34,920.52**
	Payload Adapter	1	619.05	619.05	0.0%	0.00	619.05	1	619.05	619.05	0.0%	0.00	619.05	1	619.05	619.05	0.0%	0.00	619.05
	Total Launch Mass						**35,889.78**						**35,912.10**						**35,539.57**

Summary and Conclusions

During this study, three NTP configurations were highlighted and evaluated in greater detail:

1) Liquid hydrogen configuration launched on an SLS B2
2) Liquid methane configuration launched on a SpaceX Starship
3) Liquid ammonia configuration launched on a SpaceX Falcon Heavy

The previous discipline sections covered the results of the detailed analysis for these three configurations, accounting for overlap where applicable. However, it is worth noting important findings obtained in the course of this effort:

- While CFM systems still play an important role in dictating power requirements for the overall vehicle, avionics/comms components can also have a significant impact. This has implications for deep space missions where data bandwidths and long-distance communications have increasingly larger impacts on avionics performance requirements.
- Even though ammonia and methane propellant architectures have lower specific impulses compared to the hydrogen architecture, they have important benefits in comparison, including the reduced need for CFM systems (the ammonia configuration has no active CFM requirements and uses only heaters), and a much higher propellant density.
- Thrust levels were higher for ammonia and methane configurations compared to the hydrogen configurations, and their specific impulses were on par, if not better than, most current chemical propulsion systems. This suggests potential benefits for maneuvering in planetary gravity wells to minimize gravity losses, while still maintaining, if not improving, over the specific impulses provided by conventional chemical options.
- The biggest penalty that affects all configurations is the high dry mass (or, low thrust-to-weight ratio) of the system. While ammonia and methane configurations can potentially overcome the thrust-to-weight limitations by increasing the propellant flow rates and thus the engine thrust at higher levels compared to an equivalent sizing of a liquid hydrogen system, they face a tradeoff in a reduced specific impulse. Ultimately, the reactor, reactor shielding, and reactor structural support components and their masses are orders of magnitude greater than that of a comparable chemical propulsion thrust stage, and methods of reducing their mass need to be developed in order to improve their competitiveness in the single-launch mission architectures. It is worth noting, however, that much of the mass penalty being considered here is based on conservative estimates derived from flight demonstrator assessments. Next-generation reactor concepts being considered will likely drive down that dry mass further, helping reduce the penalty associated with the system.
- The single-launch architecture assumption has several benefits in terms of reducing launch and mission complexity, but at the cost of overall flight performance, due to the inability of the higher specific impulse of the NTP stage overcoming the high dry mass penalty (due to propellant volume constraints in a single launch). A multi-launch architecture has been explored recently by industry initiatives and could be considered here, as it could have significant improvements in terms of mission payload/transit time performance. The higher specific impulse of the NTP stage combined with a larger propellant load would overcome the penalties incurred by the higher dry mass. Such a system would also support multi-ton payloads to a variety of interplanetary destinations, including the outer planets.

Recommendations for follow-up work in this effort would involve focusing on evaluating the other configurations in the Level 0 tradespace in greater detail. Other launch vehicles and propellant/reactor architectures can also be evaluated in a similar Level 0 fashion and used to expand the tradespace – this would include alternative hydrocarbon propellant architectures, and potentially other reactor configurations. Additionally, the mission analysis that was conducted for specific cases could be expanded to evaluate alternative trajectories, and further optimization can be recommended to focus on specific objectives such as minimizing transit time or maximizing delivered payload. Lastly, multi-launch architectures should also be evaluated in greater detail, especially for unmanned science missions that would seek to deliver much larger, multi-ton payloads across one or more destinations.