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Mars Transportation Assessment Study

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Mars Transportation Assessment Study

(MTAS)

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Study Sponsored by

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Foreword

The Mars Transportation Assessment Study (MTAS) was initiated in 2019 as an agency-level, independent assessment of Nuclear Thermal and Nuclear Electric transportation technologies required to put humans on Mars and return them safely to Earth. The study's objectives were to: 1) Establish the feasibility of transportation systems for a ~2-year human Mars mission in the 2030s with acceptable risk and minimal development, and 2) Provide the agency with necessary knowledge and insights to inform future architectures, concept of operations (ConOps), campaigns, and development of the required capabilities. MTAS provides a detailed technical analysis of these two propulsion technologies using a common set of ground rules that were developed based upon the anticipated human Mars mission architecture at that time.

This analysis is intended to be illustrative and will contribute to the broader discussion on how to send humans to Mars. Thus, NASA is releasing these study results to inform the ongoing assessment of Mars transportation propulsion options in the evolving Mars architecture development. Specifically, MTAS directly supports the Transportation and Habitation Goal identified in NASA's Moon to Mars Objectives[1] and is relevant to the following objectives:

- TH-5: *Develop a transportation system that crew can routinely operate from the Earth-Moon vicinity to Mars orbit and the Martian surface*.
- TH-6: *Develop a transportation system that can deliver large surface elements from Earth to the Martian surface*.
- TH-12: *Develop systems capable of returning large cargo mass from the Martian surface to the Earth, including the capabilities necessary to meet scientific sample return objectives.*

As NASA's ground rules for human Mars mission transportation continue to evolve with the mission architecture, this study's findings add valuable depth to the body of knowledge necessary for the transportation propulsion trades. The MTAS team performed analyses of each propulsion option without constraints on the concept of operations, beyond the common ground rules of trip time, crew operations, payloads, launch vehicles, rendezvous orbits, and infrastructure. This allowed each propulsion technology to take an optimal approach for the defined mission with technology choices made to balance required performance with development risk. As a result, the mission performance comparisons can be taken as relevant to any Mars transportation mission where each technology is similarly constrained. Furthermore, the technology comparisons and findings are independent of mission architecture, and valuable to the trade space evaluation. NASA encourages further discussions within the external technical community on the potentials of these two nuclear propulsion systems.

MTAS was supported by NASA's experts in propulsion, power, mission design, trajectory analysis, cost analysis, systems engineering, and programmatic development. The team also relied on the Department of Energy's nuclear engineering expertise.

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[1] https://www.nasa.gov/sites/default/files/atoms/files/m2m-objectives-exec-summary.pdf

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Executive Summary

Study Goal

The MTAS Team was tasked by NASA leadership to establish the feasibility of transportation system options for a ~2 year opposition class human Mars mission in the 2030s to reduce overall risks to astronauts. The initial focus of the team was on nuclear systems given the known challenges of chemical systems for opposition class trajectories. The team was to assess acceptable risk and minimal development for a viable mission, and provide the Agency with guidance for potential architectures, concept of operations (ConOps), campaign, and development of the required capabilities. Further assessment of the non-nuclear alternatives is recommended as follow-on work to ensure completeness of results; but is not currently anticipated to change the overall results.

Key Takeaways

(Corresponding sections in parentheses)

1. Both Nuclear Thermal Propulsion (NTP) and Nuclear Electric Propulsion with Chemical Combustion Propulsion (NEP/CCP) system concepts have the potential to fulfill the mission requirements for fast Mars transits in the 2030s, but with significantly different challenges (4.3)
2. A balanced portfolio of both NTP and NEP investment in technology development is required to retire risk and provide confidence for a down select decision (5.0)
   a. The current states of knowledge and technology are insufficient to commit to one without high risk (4.1.1, 4.2.1)
3. If the Mars Transportation Assessment Study (MTAS) technology projections are validated through technology development, NEP augmented with chemical propulsion would be the more advantageous option for human Mars transportation and has stronger synergistic benefits (5.0)
   a. One-third of the launches, with overall lower mission mass required vs NTP (4.3.4)
   b. Significantly lower development and overall/subsequent use cost vs NTP (4.3.5)
   c. Mission robustness offered by the agility of having both high and low thrust systems (4.3.3.4)
   d. Option to use the more affordable CCP stage alone for predeployed cargo deliveries (4.2)
   e. Greater performance potential by reducing system specific mass & increasing I_{sp} (4.3.2.2)
   f. Mitigates variance in mission performance across the Earth-Mars synodic cycle (4.3.3.4)
   g. Development of NEP has greater benefits to the range of NASA's exploration applications
      1. Fission power development for surface applications is synergistic with NEP (4.3.3.2)
      2. Small NEP enables high ΔV robotic exploration precursor and follow-on applications (4.3.3.2)
   h. NEP is an overall lower risk technology development with stronger synergy to terrestrial nuclear
      1. The use of Chemical Propulsion with NEP reduces the technology required (4.2.1)
      2. NEP can be tested in modular subsystems, with simulated interfaces, reducing facility requirements (4.2.1)
4. A DOD-NASA partnership is possible for either NTP or NEP. DOD/DARPA is interested in NTP due to its high thrust acceleration and rapid response capability. A concept for ~10 kW_e NEP based on the Kilopower reactor is also under consideration for development by the U.S. Space Force. (4.3.3.2)
5. The use of High-Enriched Uranium (HEU) leverages past nuclear fuel and reactor development for both NTP and NEP and results in the simplest, highest performance systems (4.3.2.3, 4.3.2.4)
   a. The use of High-Assay, Low-Enriched Uranium (HALEU) introduces significant system complexity and development risk.
6. For the faster human missions that nuclear propulsion enables, committed and significant investment in nuclear propulsion technology must begin now, for Mars missions in the 2030s. (4.1.1, 4.2.1)

1.0 SUMMARY

NASA is studying a strategic objective to put humans on Mars and return them safely to Earth by the mid to late 2030s. To investigate the feasibility of these objectives, NASA chartered the Mars Transportation Assessment Study (MTAS) to establish a human transportation solution based on technologies that would be achievable in the time frame of interest. Due to the current understanding of mission and crew risks associated with long duration interplanetary missions, the study was challenged with a two-year roundtrip, and 30-sol (Mars day) Mars surface stay as ground rules. Furthermore, a minimally viable overall architecture was given as a constraint to limit required developments and enhance programmatic affordability and achievability. Subsequently, MTAS was extended to refine what was initially established as a “proof of concept” design to address identified challenges, and balance overall risk between mission operations and technology development. This is the documentation of the findings of Phase 1 of this study, including two reference design points for two propulsion system alternatives that can achieve the Agency objectives.

The MTAS team conducted analyses to determine the feasibility of performing a two-year roundtrip human mission to Mars, and the cargo missions to support it. A two-year round trip Mars mission requires a higher energy “opposition-class” trajectory, with an associated short stay time at Mars. By comparison, these trajectories require more propulsive energy than “conjunction class” trajectories, which require approximately a 3-year roundtrip with longer stay times in the Martian system. Conjunction class trajectories take advantage of more favorable planetary alignments to minimize energy requirements. Chemical, solar-electric, nuclear-thermal, nuclear-electric, and hybrid combinations of these propulsion systems were initially considered for the assessment. The study team was tasked to evaluate technical development requirements to determine if the required technologies could be ready for a 2035 mission. Since the developed vehicle design would be expected to be useful for additional Mars-mission opportunities, requirements were assessed against very difficult planetary alignments in 2039 and 2042 to establish bounding performance requirements. Any vehicle capable of such a mission would be too large to be launched on a single rocket so the study team assumed elements would be assembled in space across multiple launches. Launch vehicle options considered include the Space Launch System (SLS), Heavy Commercial Launch Vehicles (H-CLVs), and Super Heavy Commercial Launch Vehicles (SH-CLVs). Because the Lunar Artemis Program will also need some of the limited SLS vehicles, constraints were placed on how many could be used for the Mars mission. However, no constraints were placed on the availability of the H-CLVs or SH-CLVs.

To support Agency strategic planning, and due to limited resources, the MTAS focused on the two most promising propulsion technologies: Nuclear Thermal Propulsion (NTP), and Nuclear Electric Propulsion (NEP). Because of propellant efficiency, as measured in specific impulse (I_{sp}), each of these propulsion technologies has significant advantages over Chemical Combustion Propulsion (CCP or “Chemical Propulsion”) and could potentially enable more robust and faster transits to Mars. The study innovated a hybrid NEP/CCP approach, similar to the hybrid SEP/CCP approach identified as a non-nuclear alternative for Mars conjunction class missions in the 1990s, whereby a balance of high thrust chemical propulsion and low thrust electric propulsion can perform fast missions with less propellant using less advanced technology. Additional studies are recommended to objectively assess both SEP/CCP and all-Chemical Propulsion against the MTAS NTP and NEP/CCP reference cases to provide the justification for the yet-to-be-selected Mars transportation propulsion option. Table 5.0-1 contains a compilation of recommended follow-on work.

MTAS Phase 1 concluded that both NTP and NEP/CCP propulsion system concepts have the potential to fulfill the mission requirements for fast Mars transits in the late 2030s, including enabling two-year roundtrip mission durations. But these options use different approaches with unique challenges. By technology comparison, NTP has technology challenges related to the high-temperature reactor operation and long duration liquid hydrogen storage and is more costly to develop and implement (according to the MTAS cost assessment). NEP requires a complex integration of five major subsystems: reactor, power conversion, heat rejection, power management

& distribution (PMAD), and EP thrusters. The NEP development benefits from previous efforts and synergies with current NASA electric propulsion (both space- and aircraft-based) and terrestrial and space nuclear energy developments. MTAS found that the augmentation of NEP with high thrust chemical propulsion reduces the required NEP power level by a factor of 3 to 4 to perform the objective mission, which should simplify ground testing and in-space deployment. NTP has a significant technology challenge to provide long-term, low-loss storage of liquid hydrogen (LH_2) as a cryogenic propellant, which is critical to maintain performance. NEP as proposed uses supercritical xenon propellant, which is much denser than hydrogen and easier to store but requires increased terrestrial production to meet the quantities needed for human Mars missions. Furthermore, the hybrid approach selected for NEP uses liquid oxygen (LOX) and liquid methane (LCH_4) as propellants, which simplifies cryogenic storage compared to LH_2 due to the higher storage temperature. By operational comparison: NTP has key utility as the highest I_{sp} high thrust capability of currently achievable technology, which benefits human Mars missions with rapid propulsive response to contingencies, and comparatively short capture into and departure from planetary gravity wells; NEP is a low thrust system with three-to-ten times higher I_{sp} resulting in greater performance potential for advanced missions, and with chemical propulsion adds the high thrust benefits of NTP. Previous studies suggest NEP is more versatile and extensible (Dudzinski, 1995; Dudzinski, 1992) than NTP when considering a range of applications for subscale precursor development, supporting science missions at 10-100 kW_e, and more advanced follow-on human missions (Moomaw, 2005; NASA, 2011; National Academies, 2022) however, additional studies are needed, and this has been captured on the list of follow-on work. NTP requires approximately three times more unique launches to support the campaign than NEP/CCP for the MTAS piloted mission objective, relying on Super Heavy Commercial Launch Vehicles largely because of the low density of hydrogen propellant.

Both NTP and NEP require an aggressive development program if we are to meet a 2030s launch date. Furthermore, the federal policy guidance in SPD-6 to emphasize High-Assay, Low-Enriched Uranium (HALEU) presents an additional schedule challenge because it is a departure from the High-Enriched Uranium (HEU) heritage fuel and reactor development, which established heritage feasibility for both systems. HEU has been the focus for space nuclear power and propulsion fuel since the dawn of the space age because it results in the simplest, highest performance systems. HALEU will require advanced technology for fuels and moderators, adding cost, schedule, complexity, and development risk. The only reasons to consider HALEU for space applications is to address proliferation concerns with the use of HEU, and security and licensing for the handling of HEU. However, robust public-private partnerships for space nuclear power and propulsion are possible with HEU systems utilizing existing licensed facilities at U.S. government labs and in industry.

NASA is currently investing in NTP components, with near-term plans for fuel characterization at the Department of Energy (DOE) Idaho National Laboratory (INL). NASA is also investing in the critical technologies for cryogenic fluid management of liquid hydrogen (as well as liquid oxygen and methane) to solve the propellant transfer and long-term storage challenges crucial for both NTP and cryogenic chemical propulsion systems. NTP is also receiving initial investment from DOD/DARPA due to its high thrust acceleration and rapid response capability with potential to enable more agile cislunar operations. If developed appropriately, the DOD systems could be adapted for NASA use, but coordinated development will be essential. Progress-to-date represents a small fraction of development needed for human Mars missions, and current HALEU NTP designs are a departure from the HEU design heritage established in Project Rover/NERVA in the 1960s.

NASA currently does not have a coordinated development activity for an NEP system but is investing in technologies for LOX-LCH_4 chemical propulsion, and is independently investing in several relevant subsystem technologies, such as electric propulsion and high voltage PMAD, among others. NEP may benefit from current synergistic investment in relevant technologies for terrestrial applications that can be leveraged by NASA, including megawatt-scale microreactors and power conversion, etc. NEP also has technology heritage from the Snapshot Program (Systems for Nuclear Auxiliary Power), the Space reactor Prototype (SP-100) program, and Project Prometheus, which was developing NEP for the Jupiter Icy Moons Orbiter mission. The heritage technology from these investments offers a promising starting point for Mars-relevant NEP development. Currently, a complete NEP concept based on a ~10kWe evolution of the Kilopower reactor is also under consideration for development by the U.S. Space Force (Helios-N). This development could be leveraged by NASA for science and Mars NEP precursors.

The MTAS Programmatic Assessment found moderate (~37%) cost advantage for NEP/CCP over NTP when considering both development and first Mars campaign use. The MTAS assessment found a ~16% lower Design, Development, Test, and Evaluation (DDT&E) cost for NEP. However, the costs for both have large uncertainties due to technology challenges and obtaining accurate DDT&E cost estimates will require further technology investments to benchmark performance projections. Since the end of Project Rover/NERVA, open-air testing at the Nevada Test Site is no longer acceptable. NTP development and qualification requires a full scale, integrated ground test, which requires a costly new facility to capture and process the, likely radioactive, propellant exhaust. This facility is a major development challenge with significant cost and schedule uncertainly. The development of NEP will also require some facility development, notably large, high-pumping capacity vacuum chambers for testing multiple large electric thrusters. That said, several nuclear facilities are in development for terrestrial microreactors that could be leveraged or adapted for the development of space reactors supporting NEP. Furthermore, the characteristics of NEP system development allows for separate subsystem ground testing of modular elements and simulated interfaces, which could be accomplished in separate facilities, easing facility development requirements.

Additional distinctions between NTP and NEP/CCP emerge when considering an ongoing Mars exploration campaign. A benefit of higher specific impulse electric propulsion is the mitigation of the energetic challenges that result from more distant planetary alignments between Earth and Mars reducing the variance in mission performance across the synodic cycle. An NEP reactor has greater lifetime potential than NTR for vehicle reusability. Furthermore, this study's implementation of NTP utilizes significant drop-staging to minimize round-trip propellant mass, which requires more hardware production and Earth launch to support a subsequent mission. A majority of NEP/CCP concept hardware is returned to Earth. Finally, NEP/CCP offers cost advantages over NTP for development and first Mars campaign use, and with reuse cost advantages, the NEP/CCP cost advantage could be multiplied for an ongoing Mars exploration campaign.

In summation, NTP offers high-thrust capability for rapid response to contingencies and gravity-well departures to support a human Mars mission with higher efficiency than chemical propulsion. NTP advantages result from the highest I_{sp} for currently achievable high-thrust propulsion. However, low-thrust NEP, when augmented with high-thrust LOX-LCH_4 chemical propulsion looks very promising as the more advantageous alternative. The NEP/CCP approach results in comparatively: fewer launches with fewer number and types of elements; lower launch costs, unit costs & campaign costs; and easier launch, storage, and management of propellants. Furthermore, NEP, when augmented with Chemical Propulsion offers both mission agility and architecture flexibility, whereby separate high and low thrust propulsion elements can be used where advantageous. In comparing the technology challenges, the study found that NTP has at least three high risk potential showstoppers: a reactor fuel achieving 900 sec I_{sp} while solving the hydrogen material and reactivity interactions; development of a full-scale engine test facility; and long-term, zero-leak hydrogen cryogenic fluid management (CFM). Whereas NEP, though requiring the integration of 6 subsystems, there are no subsystem technologies that are high risk to be show-stoppers[2]. Additionally, there are important NEP synergies and feed forward from terrestrial energy and lunar power and leveraging of existing electric propulsion (Gateway SEP Power and Propulsion Element), and PMAD (electrified aircraft). There are additional synergies with ongoing LOX-LCH_4

[2] Measured by lower average and peak Advancement Degree of Difficulty (AD2) provided by independent NESC study.

propulsion development. Many of the NEP challenges will be addressed by fission power for the lunar surface, and electric propulsion for Gateway. Other studies (Noca, 2001; Jones, 1984; Kokan, 2021; Pawlik, 1977; Cassady, 2007) have identified NEP in general has potential applications to robotic science, and missions beyond Mars, which represent opportunities to leverage NEP development for precursor and follow-on applications (Oleson, 2019). Similar benefits from NTP technology to other applications are more limited (Houts, 2012; Dudzinski, 1995). In consideration of the broad NASA needs for propulsion, and the broader national needs, both NTP and NEP possess an imperative for investment, as they each have noteworthy advantages in specific applications. However, if the Mars NEP/CCP performance metrics can be verified in technology development, NEP/CCP would be the more advantageous option for human and cargo transportation to Mars.

To support a 2030s Human Mars Mission, committed and significant investment in nuclear propulsion must begin now. NASA is currently investing in NTP to support synergy with other agencies and Congressional direction; however, the current state of understanding is insufficient to make a down-selection. The results of MTAS suggest a balanced portfolio of investment in both NTP and NEP is necessary in order to answer key feasibility and development questions where the risks are sufficiently understood and retired to focus investments for future Mars transportation needs. A coordinated technology research plan is suggested, with interim go/no-go decision points, and key figures of merit to inform a down-select decision. However, current NASA funding is insufficient to support a dual-track approach. NASA needs flexibility to rebalance investments between NTP and NEP to support an informed down-select decision as soon as practical.

The decision on propulsion for human transportation beyond the Moon could set the course of human exploration for decades to come. It will be a decision that defines a decade-long development program costing more than $10B. This decision must be made based on sound, objective analysis, and strong technical confidence, for which the current state of knowledge is insufficient. Thus, MTAS Phase 1 recommends more detailed analyses to guide investments along with active funding of strategically focused investments in both NTP and NEP to inform the propulsion decision for humanity's first interplanetary spaceship.

2.0 INTRODUCTION

The Mars Transportation Assessment Study was supported by most of NASA's best experts in propulsion, power, mission design, trajectory analysis, cost analysis and programmatic development to assess the challenges with human transportation to and from Mars and added the expertise of the Department of Energy in nuclear engineering to assess the nuclear aspects of the focus propulsion technologies. Together the team provided the Agency and the nation with an objective assessment of potential propulsion technologies to achieve the ground rule objectives and provided guidance for an appropriate transportation architecture, concept of operations, integrated Mars transportation campaign, and development approach for the required capabilities.

Concurrent to MTAS, the National Academies of Science (NAS) performed a similar study of nuclear propulsion options and received input from the MTAS team. Though the NAS study was very thorough, it concluded before the MTAS study completed iteration 2. Thus, the NAS did not receive a final report on the two propulsion systems, particularly the revised NEP/CCP concept. As a result, the NAS study is complimentary to MTAS, but does not reflect the most recent assessment of the technologies.

The objective of Phase 1 of the Mars Transportation Assessment Study was to provide sufficient guidance to establish a technically grounded investment plan for propulsion technologies necessary to accomplish the reference Agency concept of operations for a two-year class human Mars Mission. This ConOps is defined in the ground rules in Section 3.0. A two-year round trip Mars mission requires an opposition-class trajectory, which requires the propulsion systems to perform more ΔV than conjunction-class trajectories, which are optimized to minimize ∆V (or change in vehicle velocity – a measure of propulsive effort[3]). Mars opposition missions can achieve total trip times of two years or less with short stays at Mars, whereas conjunction missions are generally three years or more in total duration with required Mars stays of almost two years while waiting for optimum planetary alignment. However, higher ∆V missions, such as opposition class missions, put more emphasis on the performance of propulsion technologies (and highlights the distinctions between options).

The ∆V required for a Mars mission varies across mission opportunities defined around Earth-Mars closest approach, which occurs every ~26 months. Some opportunities are relatively easy such as 2033 and 2035, while other opportunities can be very challenging such as 2039 and 2042. Furthermore, the required ∆V varies much more significantly for opposition class trajectories, since more distant planetary alignment affects both outbound and inbound legs of challenging opportunities. Figure 2.0-1 shows the ∆V variance for both conjunction and opposition class missions from 2032 to 2050. The ∆V for the most difficult opposition class opportunities can be as much as twice the ∆V of the average conjunction class opportunity. The alignment of Venus with Earth and Mars is an important consideration for opposition class missions; most opposition opportunities offer a beneficial Venus Gravity Assist (VGA) on either the outbound or inbound leg, which can reduce the overall ∆V. Figure 2.0-1 depicts where a Venus gravity assist is opportune or not by color variance. While the study considered all the Mars mission opportunities in the mid to late 2030s and early 2040s, the 2039 opportunity was used as the sizing case for both the NTP and NEP options since it provides a reasonable bounding case for performance that can support all but the most difficult opportunities. This approach recognizes that any long-term endeavor, such as a human Mars exploration campaign, is likely to miss targeting the most advantageous opportunities, and will ultimately need to support most opportunities in the synodic cycle.

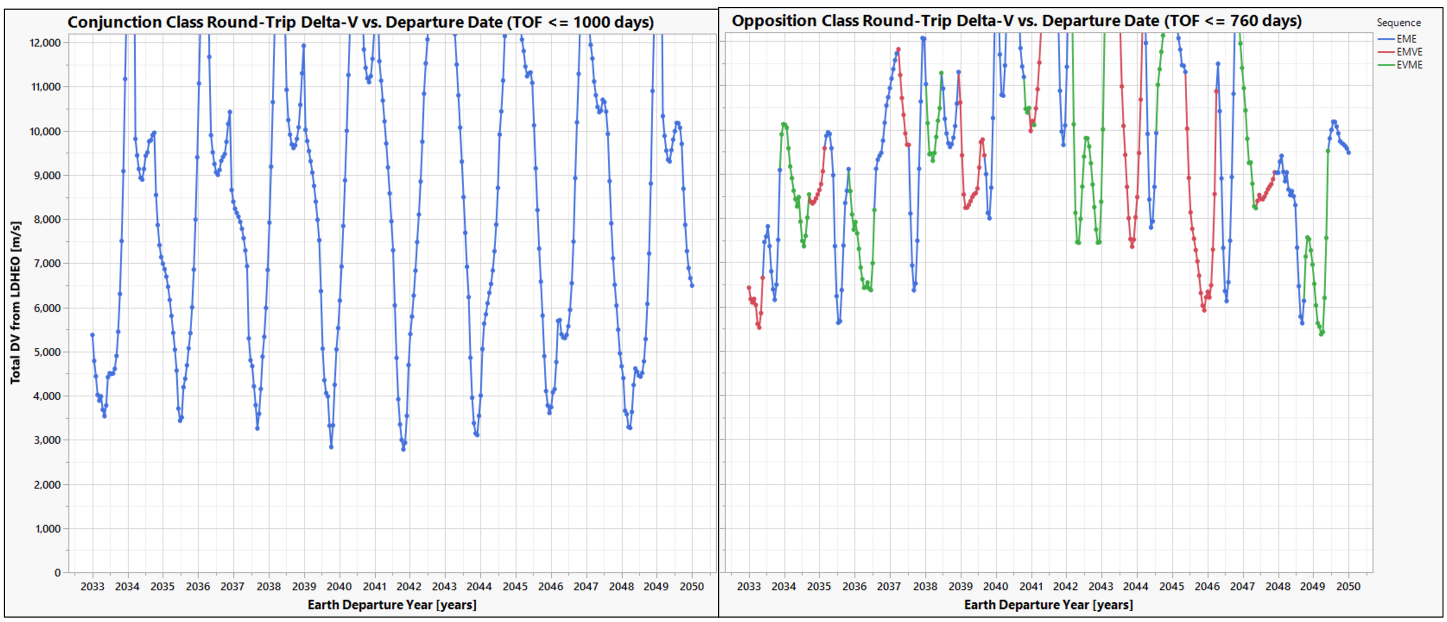


Figure 2.0-1. ∆V variance across Mars opportunities 2032-2050.

[3] Refer to appendix X for full definition.

The study was conducted in two iterations: Iteration 1 efforts are referred to as “1.1”, and Iteration 2 efforts are referred to as “1.2”.

Iteration 1: Proof of Concept

The objective of the first iteration was to establish a proof of concept that the required propulsion technology can realistically meet the agency human Mars transportation ConOps in the timeframe of interest, which was 2035. To accomplish this, the study team assessed the propulsion technology, elements, operations, infrastructure, and programmatics necessary to meet the ConOps ground rules, documenting the necessary assumptions. For each propulsion option, technologies for fuels, reactor designs, coolant schemes, power conversion, electric propulsion, and related subsystems were chosen based on expert assessments of the most mature, lowest risk options to meet performance goals in the mission timeframe. Thus, the MTAS reference cases provide a proof of concept based on a risk mitigation approach, and do not represent optimal performance from an exhaustive trade study. In many cases, alternative trades exist that should be evaluated as proposed in follow-on efforts from this study.

Subsequently, the study identified the areas where the technology, mission, campaign, and programmatics are high risk and/or insufficient in order to address these issues in a second iteration.

At the completion of Iteration 1, a defined process point in the design of several alternatives, an internal non-advocate critical review was performed to assess the designs and critique the process. This “Red Team” representing a balance of relevant technical experts and nonadvocates invited from NASA (not part of the MTAS team), DOE, and the private sector, reviewed and provided a valuable independent assessment to ensure the credibility of the study results. Iteration 2 proceeded to address the challenges identified by the Red team, which included significant development risk for the technologies required in 2035, and significant cost, schedule, and operational risk for the number of launches required to support each option.

Iteration 2: Risk reduction and concept refinement

The objective of the second iteration was to identify alternative technologies, mission approaches, campaign approaches, and architectures to address the risks and insufficiencies from the proof of concept. After revising the 1.1 mission, design, and performance, the study team reassessed the technology necessary to support the revised ConOps ground rules, again documenting the necessary assumptions. In some cases, where risk was assessed as high, or performance was insufficient, technology subsystem trades performed resulting in alternative technologies for the 1.2 reference cases. DOE representatives were added to the team to support an expert assessment of the nuclear subsystems. Finally, the team reassessed the programmatics to ensure that an executable investment plan with acceptable risk was feasible.

The goal of Iteration 2 was to establish one or more reference design points with a balance of overall risk between the mission operations and technology development. These reference design points can be used to trade alternatives for technology, operations concepts, and mission architectures as a standard of measure to guide investment and implementation decisions. Phase 2 of the study is recommended to formally perform these trades as the Agency moves forward with formulating a human Mars mission and investing in the necessary transportation technology, including defining strategic investments looking toward advanced capabilities.

Due to constraints on study resources, and timing of the Agency need for a proof of concept, MTAS focused on two promising propulsion technologies: Nuclear Thermal Propulsion (NTP), which uses a fission reaction to heat a propellant, which is then accelerated using a rocket nozzle to produce thrust in a Nuclear Thermal Rocket (NTR) engine; and Nuclear Electric Propulsion (NEP), which converts the heat generated by a fission reaction to electricity, which is then used to power a propellant acceleration device, or “thruster”, to produce thrust. Example system schematics of an NTP and NEP system are shown in Figures 2.0-2 and 2.0-3, respectively. MTAS selected these options because of the propellant efficiency advantage, measured in higher specific impulse, that NTP (~900 sec) and NEP (~2,000-10,000 sec) would have over Conventional Chemical Propulsion

(~330 - 460 sec). Greater propellant efficiency reduces the propellant required to perform changes in vehicle velocity necessary to transit from Earth orbit to Mars orbit and back. Without an efficient propulsion system, the propellant mass for a Mars mission can become prohibitive, especially for fast Mars transits. Higher propellant efficiency also enables more robust missions, as the propulsion system can afford propellant margin to respond to contingencies.

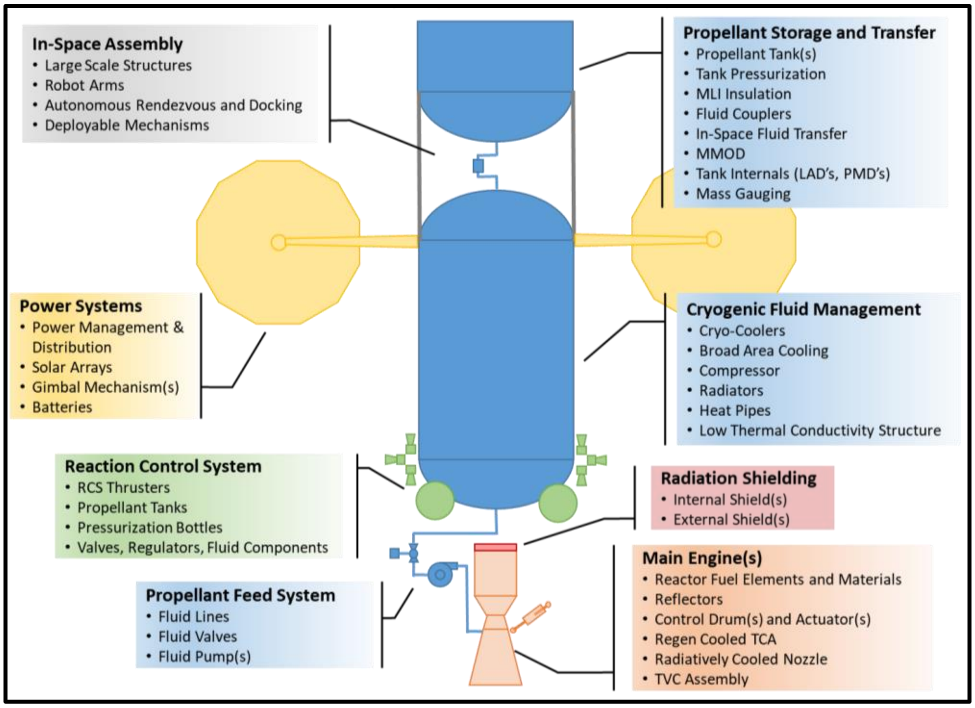


Figure 2.0-2. Example NTP system schematic from the NESC assessment of NEP/NTP.

One of the challenges for the use of any electric propulsion technology is the low thrust acceleration, limited by power, which results in extended time to propel into and out of planetary gravity wells (regions where gravitational forces are much higher than deep space). To provide faster transits with NEP, the study identified a hybrid NEP/CCP approach, an innovation based on the hybrid SEP/CCP approach identified as a non-nuclear alternative for Mars transportation in the 1990s (Uppal, 2020; Davis 2011; Mani, 2020) whereby high thrust acceleration CCP is used for gravity-well maneuvers.

MTAS was initially chartered to complete a comprehensive assessment of credible propulsion options for human Mars transportation. It is recommended that a future phase of study assess SEP/CCP against the NTP and NEP/CCP reference cases. Furthermore, innovative implementations of Chemical Propulsion in alternative architectures should be assessed. These additional assessments should be completed objectively to establish the justification for the selected Mars transportation propulsion option, especially if it is nuclear.

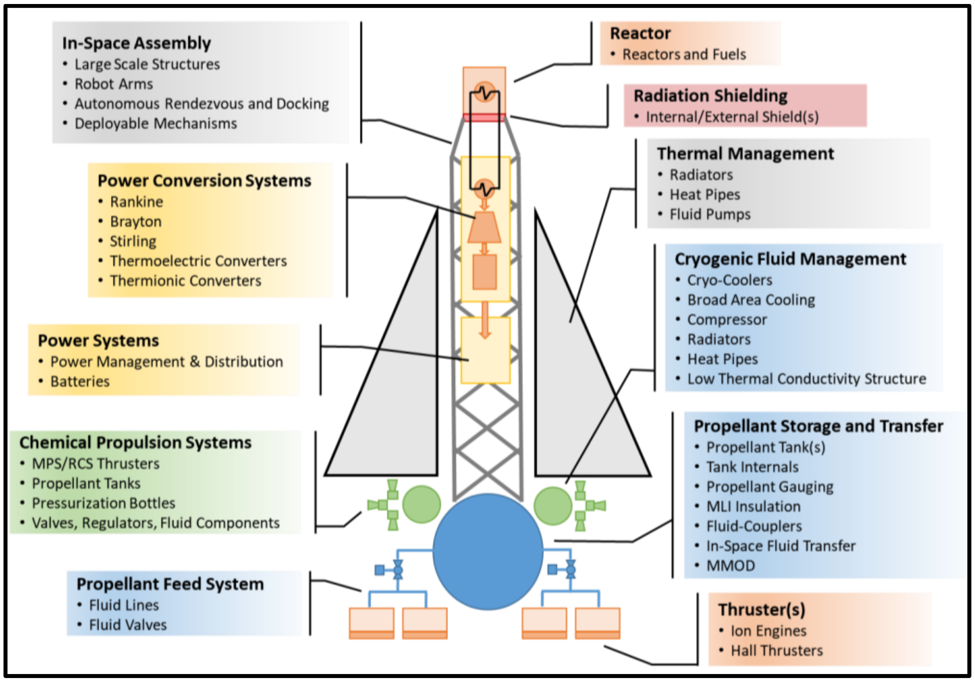


Figure 2.0-3. Example NEP system schematic from the NESC assessment of NEP/NTP.

Two engineering teams were simultaneously employed to perform mission and systems analysis for MTAS: the Advanced Concepts Office (ACO) at NASA Marshall Space Flight Center (MSFC), and the Compass concurrent engineering team at NASA Glenn Research Center (GRC). Because of both the diversity of technologies and engineering specialties involved, and the comprehensive nature of the assessment, a concurrent engineering approach was advantageous. Thus, a diversity of perspectives was gathered to provide a thorough and objective assessment, as well as promote greater team understanding of the issues and challenges to produce more creative solutions. The two teams studied both propulsion concepts in Iteration 1 in order to provide independent validation of results. Because the resulting mission analyses and concept vehicles built around each propulsion system were similar, the two teams specialized for Iteration 2 - the ACO team performed two iterations of the NTP vehicle design, and the Compass team performed two iterations of the NEP/CCP vehicle design. After the first iteration, both teams received feedback from the independent Red Team review. Nuclear reactor subject matter experts from the DOE joined the teams in Iteration 2 to provide expertise in the design and testing of the reactors and reactor components as the teams refined their technology assessments. Furthermore, a Campaign Assessment Team assessed logistics and operational integration of Mars transportation into an assumed ongoing Artemis campaign of lunar exploration, and an internal Programmatic Assessment Team reviewed both iterations to assess issues of cost, schedule, risk, infrastructure, and implementation. The Programmatic Team defined figures of merit for comparison broadly as: Affordability, Safety and Mission Assurance, and Program Robustness. A DOE expert also participated on the Programmatic Assessment team to provide estimates of fuel fabrication cost and schedule, testing requirements and their cost, and the need for new test facilities, what they would cost, and how long they would require to become available. Each sub team presented their progress at weekly meetings for review by the entire team to facilitate transparency and coordination across the entire MTAS team. MTAS was coordinated with Agency activities through the NASA Federated Board. The study process is depicted in the flowchart in Figure 2.0-4.

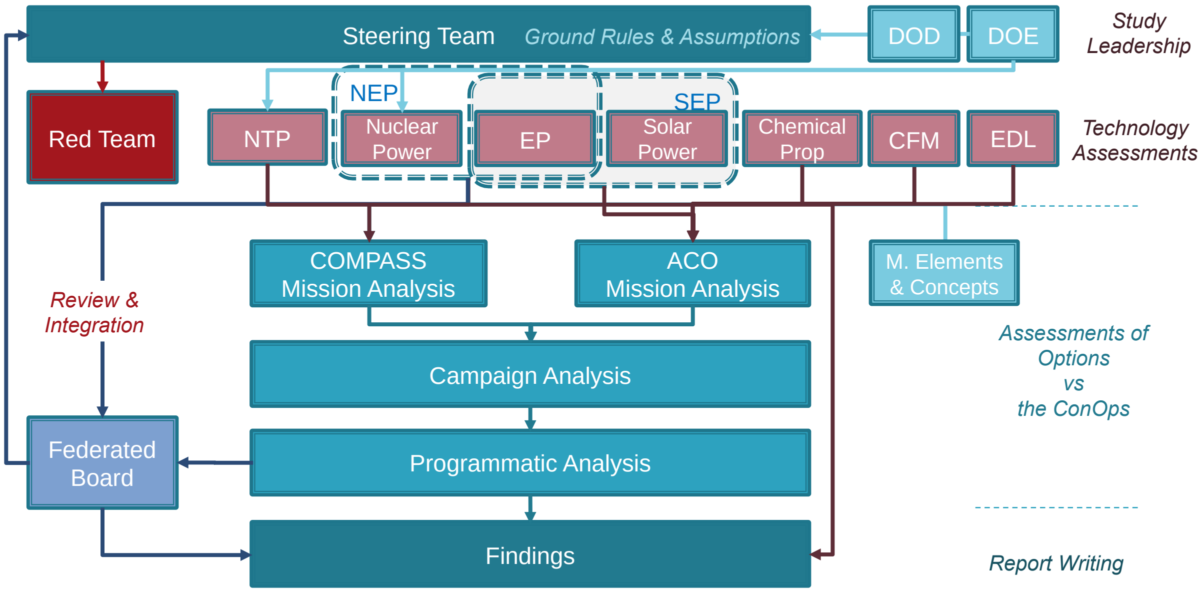


Figure 2.0-4. MTAS Process Flow.

This document contains the summary details of the 18-month Mars Transportation Assessment Study. A more detailed documentation of the study and results for the reference cases, including proposed technology plans, will be published as a NASA Technical Memorandum.

3.0 GROUND RULES AND ASSUMPTIONS

NASA is studying a strategic objective to put humans on Mars and return them safely to Earth in the mid-to-late 2030s. Furthermore, NASA is considering a minimally viable overall architecture to limit required developments and enhance programmatic affordability and achievability. This is, in part, achieved by limiting the stay time on the surface of Mars, and thereby reducing the surface infrastructure required to support initial Mars landings. Short stays do not require robust habitation, ISRU, or associated large surface power supplies. It is further achieved with smaller crew-size and shorter mission time, reducing logistic masses and costs.

It is envisioned that initial short-stay missions would grow humanity's understanding of living and working on Mars, build infrastructure, and prove it out in order to support later, long-stay missions.

The ground rule architecture for an initial Mars mission campaign consists of a crew mission and the pre-deployment of assets to support the crew mission and crew operations. An "all-up" mission architecture, whereby all required mission assets travel with the crew, was not considered for Phase 1 of this study due to the increased requirements on the propulsion systems. The crew missions require a piloted Deep Space Transport (DST) consisting of the propulsion system, Deep-Space Habitation (DSH) for the crew, and associated spacecraft systems, propellant, and logistics supplies. In support of the crew mission, additional resources must be pre-deployed to Mars' orbit and surface. The predeployed systems include the descent system to take the crew to the surface of Mars, the ascent system to return the crew from the surface back to Mars orbit, and the surface assets needed for the surface mission. The study reference predeploy campaign involves delivering three 25,000 kg payloads to the surface of Mars. For the transportation system, this translates to delivering three 65,000 kg Mars descent systems to Mars orbit. The lander delivery cadence as specified by the ground rule is that two of the three landers must arrive at least 18 months prior to crew arrival, and one lander must be delivered ~4 years prior to crew arrival. The first lander is intended to be the first, full-scale EDL demonstration, and must be fully successful before committing additional assets and the exploration crew to Mars. This ground rule for the second and third landers is set to allow the payload enough time to complete autonomous operations in preparation for crew arrival[4], which retires risk for the mission before the crew departs Earth.

The Deep-Space Habitat is predeployed to a Near-Rectilinear Halo Orbit (NRHO) about the Moon, where it will be co-orbital with the Lunar Gateway. There it will be outfitted, checked out, and perform a shakedown cruise as an independent, free-flying vehicle. The DSH is returned to NRHO at the return to Earth after Mars, where it will be refurbished and potentially reused. Because of the ground rule, a propulsive rerendezvous with Earth is the reference Concept of Operations (ConOps). Also, due to perceived added risk to the crew, a Mars aerocapture or direct Mars entry was not considered. However, more propulsive advantageous mission approaches are considered for follow-on work.

The key ground rules for the vehicles and mission architectures levied on MTAS Phase 1 are as follows.

- 2-year roundtrip, opposition class mission in 2030s
- 30 day stay for 2 crew on surface, 2 remain in orbit
- Leverage lunar exploration infrastructure where appropriate and advantageous
- Optimize aggregation point for the Deep-Space Transports (DST) for crew and cargo
- The same Deep-Space Habitation (DSH) is assumed for both concepts and will be predeployed in NRHO about the Moon and returned to NRHO at the end of mission.
- Mars supporting SLS launch timing must consider ongoing Artemis SLS support
- 8.6 m diameter, long SLS fairing (7.5 m by 19.8 m dynamic envelope)
- Three 65,000 kg Landers predeployed to Mars prior to crew departure

Why two years?

The MTAS ground rules are based on the strategic objective to land humans on Mars and return them safely using lowest risk approach for mission success. This includes considering the risk of in-space vs surface time for the exploration crew. There remains debate among human mission planners whether there is greater risk to the crew in space, where humanity has decades of experience living and working, or on the Mars surface, where there is gravity, atmosphere, and some radiation protection, but also is increased infrastructure dependency and unknown hazards. The complexity of human operations on Mars, given the number of additional technologies required to keep astronauts alive and productive, with no rescue or resupply opportunities, may make the surface riskier than space for initial missions – until experience is gained and infrastructure is proven. MTAS studied the first mission for humans on Mars or its vicinity. It was considered that an initial use of shorter opposition class round-trip trajectories, with associated short stays on Mars, may help mitigate these unforeseen risks. Lessons learned and experience gained from this initial mission builds toward long-stay, conjunction class missions.

Furthermore, NASA has studied long-stay missions as the baseline approach in the Mars Design Reference Missions (DRMs) since the Space Exploration Initiative in the early 1990s and has numerous studies of Conjunction Class missions to understand the challenges. MTAS supported the need to understand the risk and benefits of Opposition Class missions.

Secondly, the global data set to date for humans in zero-g is limited to 6-12 months with most individual experiences only approaching ~6 months. Interplanetary space environment risks increase with time, such as equipment malfunctions, weightlessness, vacuum, radiation from Galactic Cosmic Rays (GCR) and solar particle

[4] The first lander was also discussed as an Entry Descent and Landing (EDL) demonstration, but this needs to be critically examined.

events (SPE), and space debris. Many of these risks continue on Mars, where additional crew risks include physical injury, weather, and failure of surface systems, etc. Furthermore, mission challenges increase with time, such as logistics of consumables, and the psychological effects of isolation, constrained living space, and being far away.

Thirdly, crew radiation from both natural sources, such as GCR and SPE, as well as the propulsion system are experienced throughout the mission, though at a somewhat lesser rate on Mars. Radiation is identified as a “Red Risk” by NASA as the highest-priority astronaut health challenge. Radiation damages DNA and can lead to mutations that can trigger cancers. It causes cardiovascular health problems such as heart damage, the narrowing of arteries and blood vessels (Belzile-Dugas, 2021). Radiation also causes neurological problems that can lead to cognitive impairment during the mission (Michaud, 2019; Turnquist, 2020; Smart, 2017).

By reducing trip time with mission success and crew survival as the focus, the risks to the crew and mission are proportionally reduced. For these reasons, two years was established by the Agency as a target round-trip mission duration, and MTAS was challenged to assess the viability of the propulsion technology to achieve this goal. Significant reductions in round trip time below two years may be possible with advanced NTP and NEP technology, but these technologies are considered beyond viability for the 2030s.

The key assumptions and trades are as follows:

1. 2035 is the primary human mission opportunity, and the development target for programmatic assessment. 2039 will be analyzed to envelope the performance required for a ~2-year Mars opposition class mission across the range of synodic opportunities.
    a. Roundtrip mission time is counted as time from Earth orbit departure (TMI) to Earth orbit capture (EOC) and is afforded 760 days or less.
    b. A Venus gravity assist will be included in the mission when advantageous to add mission value.
    c. Earth aggregation orbit for cargo and crew transportation systems will be optimized separately for NEP and NTP.
2. The reference case will be a design point with minimum technology to accomplish the Agency objectives as defined in the GR&A. The nuclear propulsion unit for the first Mars mission will not be reused, but the potential for future units to be reused will be considered.
    a. The study will trade off the reference case to identify technology to improve performance, mitigate risk, and/or reduce cost.
3. The reference case will use SLS, requiring no more than two cargo launches in a single year. If two are required, the following year is limited to one SLS cargo.
    a. Commercial Launch Vehicles, including Super Heavy CLVs, will be used to fill in where the SLS cannot meet the required launch cadence.
4. Primary reactor trade study variables include fuel type, uranium enrichment, reactor type (fast or moderated), and moderator for both NTP and NEP. For NEP, additional trades include coolant type, and reactor outlet temperature.
5. Crew dose from the onboard nuclear systems will be limited to no greater than the mission GCR dose.
6. First lander supporting the crew mission will successfully land on Mars before the second lander is launched to provide a proof of the Entry, Descent, and Landing (EDL) system for this size-class of payloads.
7. Remaining landers supporting the crew mission must be in Mars orbit 18 months prior to crew departure.

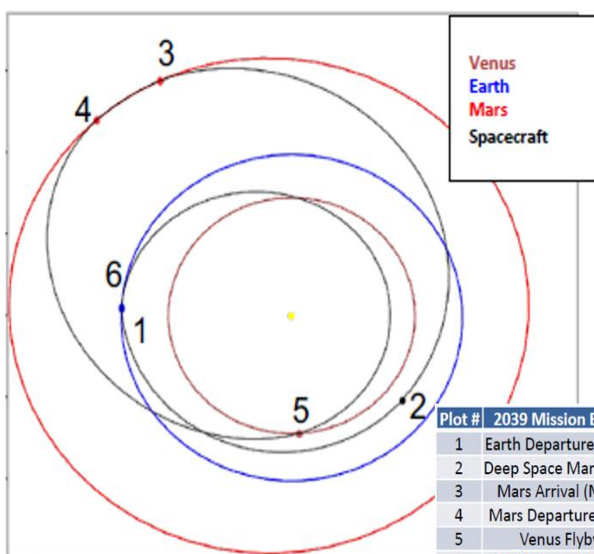
8. Orion is used for crew rendezvous in LDHEO prior to TMI, and to return the crew at EOM.

4.0 MARS TRANSPORTATION ASSESSMENT

Phase 1 of MTAS established two reference design points with a balance of overall risk between mission operations and technology development. The following sections summarize these reference cases in terms of mission design, vehicle concept, performance, and technology requirements. Also summarized is the assessment of each reference Mars mission in the context of an ongoing Artemis lunar campaign. Finally, a high-level technology development plan is provided. Greater detail on the MTAS assessments, the reference cases, and the technology plans is provided in focused MTAS reports, published as NASA TMs.

4.1 Nuclear Thermal Propulsion Reference Case

The trajectory for the NTP reference case is shown in Figure 4.1-1. All velocity changes are performed by Nuclear Thermal Rocket engines. This mission is 690 days for the 2039 opportunity as a bounding case for performance. This trajectory is a conservative case, assuming an NTR engine-out prior to Mars departure; a 670–680-day mission is possible with no engine out. The planetary alignments in 2039 provide a Venus gravity assist on the in-bound leg which enables a round trip ~30 days shorter than two years for this reference case. A longer mission in 2039 would eliminate the VGA, and save propellant mass.



Plot #	2039 Mission Event	Date
1	Earth Departure (TMI)	4/26/2039
2	Deep Space Maneuver	7/27/2039
3	Mars Arrival (MOI)	2/16/2040
4	Mars Departure (TEI)	3/26/2040
5	Venus Flyby	10/6/2040
6	Earth Arrival (EOI)	3/16/2041

Figure 4.1-1. NTP Schedule and Trajectories for the 2039 Reference Mission.

The landers supporting the crew mission would be delivered to Mars by two, separate NTP systems. This approach would provide direct and early experience with the NTP system in deep space prior to the piloted mission in 2039, but it would also require earlier development of NTP. Alternately, to offset the development schedule risk associated with this earlier need date is the fact that the first cargo mission can easily be performed with a subscale nuclear thermal rocket in the 10-15 klb_f thrust class, perfectly in line with an envisioned subscale demonstrator. For the 2039 reference mission, the single lander would be launched in 2035. In 2037, the remaining landers would be launched also using an NTP propulsion system.

Figure 4.1-2 shows the Concept of Operations (ConOps) for the NTP cargo predeployment campaign. The first cargo transfer vehicle would be launched to a Medium Elliptical Earth Orbit (MEEO) for aggregation. The first cargo transfer vehicle consists of a single NTR "Stub Core" Stage and three Super Heavy Class Commercial

Launch Hydrogen Inline Stages (CL-HIS). The MEEO of 1,200 km x 7,000 km altitude is determined to maximize the mass of LH_2 propellant delivered in each drop tank that fills the launch vehicle shroud volume of an SH-CLV. The first lander would be launched to MEEO where it is integrated with the NTP cargo transfer vehicle, then raised to Lunar-Distant High Earth Orbit (LDHEO) of 400 km x 400,000 km as the departure orbit for Mars.

Once the first lander is successfully on Mars, the second lander would be launched in 2036 to MEEO and mated with a chemical propulsion stage, which will place the second lander in NRHO to wait for the second NTP Mars cargo transfer vehicle. In 2037, the final lander and the elements of the NTP cargo transfer vehicle would be launched to MEEO. The second NTP Mars cargo transfer vehicle consists of a second, single NTR "Stub Core" Stage, four CL-HIS, and a Multi-Dock Truss Element (MDTE) to dock two of the CL-HIS and the two landers. After assembly, the NTP stage would transfer the stack to NRHO to add the other lander, then transfer to the LDHEO departure orbit for Mars. After arrival, the MAV would descend to the Mars surface and be verified operational before the piloted vehicle departs Earth orbit in 2039.

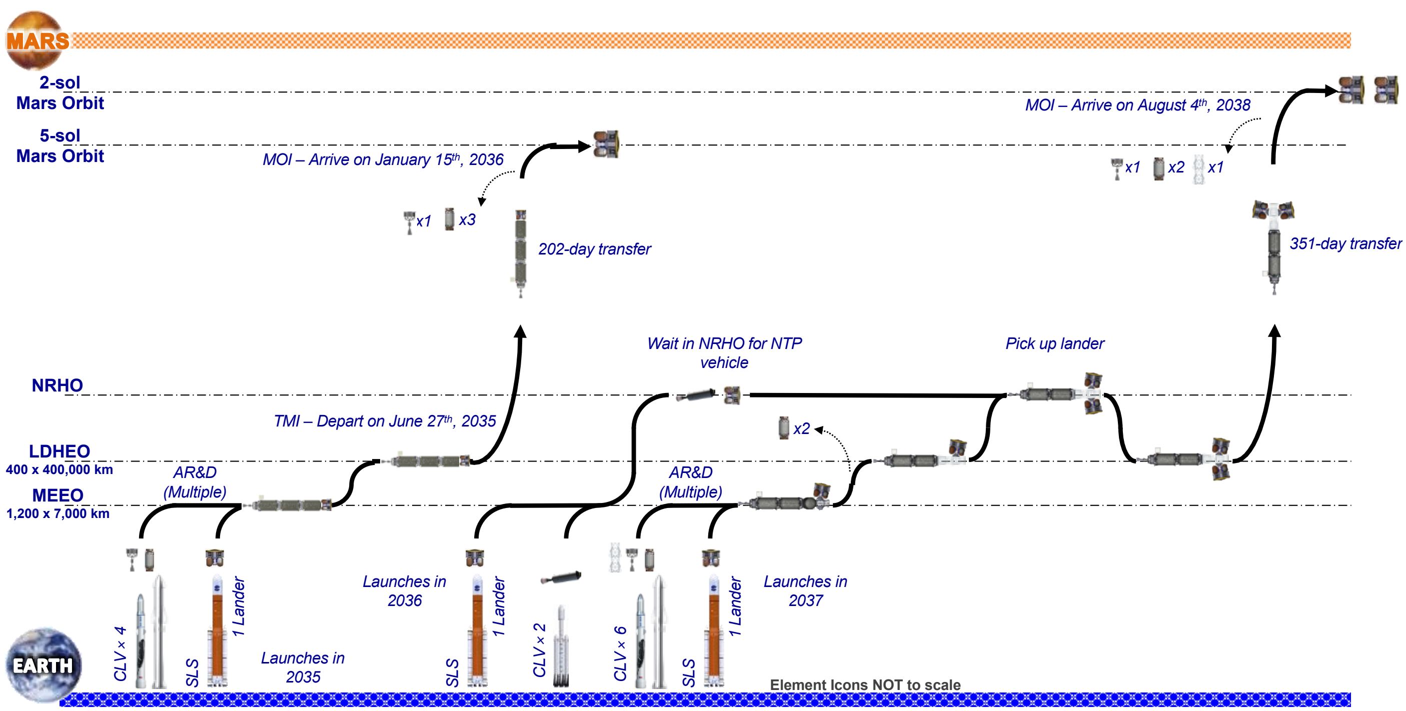


Figure 4.1-2. Mars 2039 NTP Cargo Lander ConOps.

Figure 4.1-3 shows the ConOps for the piloted DST. Its components would be launched to the MEEO for aggregation. The NTP DST consists of:

- an SLS-launched core stage with two-NTR engines;
- one SLS-launched Hydrogen Inline Stage (SLS-HIS);
- two side-mounted "Stub Core" stages, each with two side-mounted CL-HIS;
- two MDTE, docking fourteen CL-HIS, and a Power and Propulsion Module (PPM).

Once assembled, the DST would propel itself to LDHEO where it would jettison the two Stub Core Stages and their associated four CL-HIS, along with four of the CL-HIS docked to the MDTE. These jettisoned stages could later be reused or disposed of in a safe orbit. The Deep-Space Habitation requires a chemical propulsion transfer stage to descend from NRHO to LDHEO to rendezvous with the DST. An Orion capsule brings the crew to LDHEO and rendezvous with the DST. After adding the DSH and crew, the DST departs for Mars from LDHEO. By the time it arrives at Mars, it will discard the remaining ten CL-HIS along the way. It will enter a two-sol orbit and two crew members will transfer to the crew lander that will take them to the surface of Mars. After 30 sols of surface operations, the surface crew will use the MAV to return to the DST. The DST will then return to Earth with a Venus Gravity Assist (VGA). NTP would be used to capture the DST back at Earth into LDHEO, where an Orion capsule would rendezvous to bring the crew back to the Earth's surface. The DST would transfer to NRHO via NTP for disposal or potential future use.

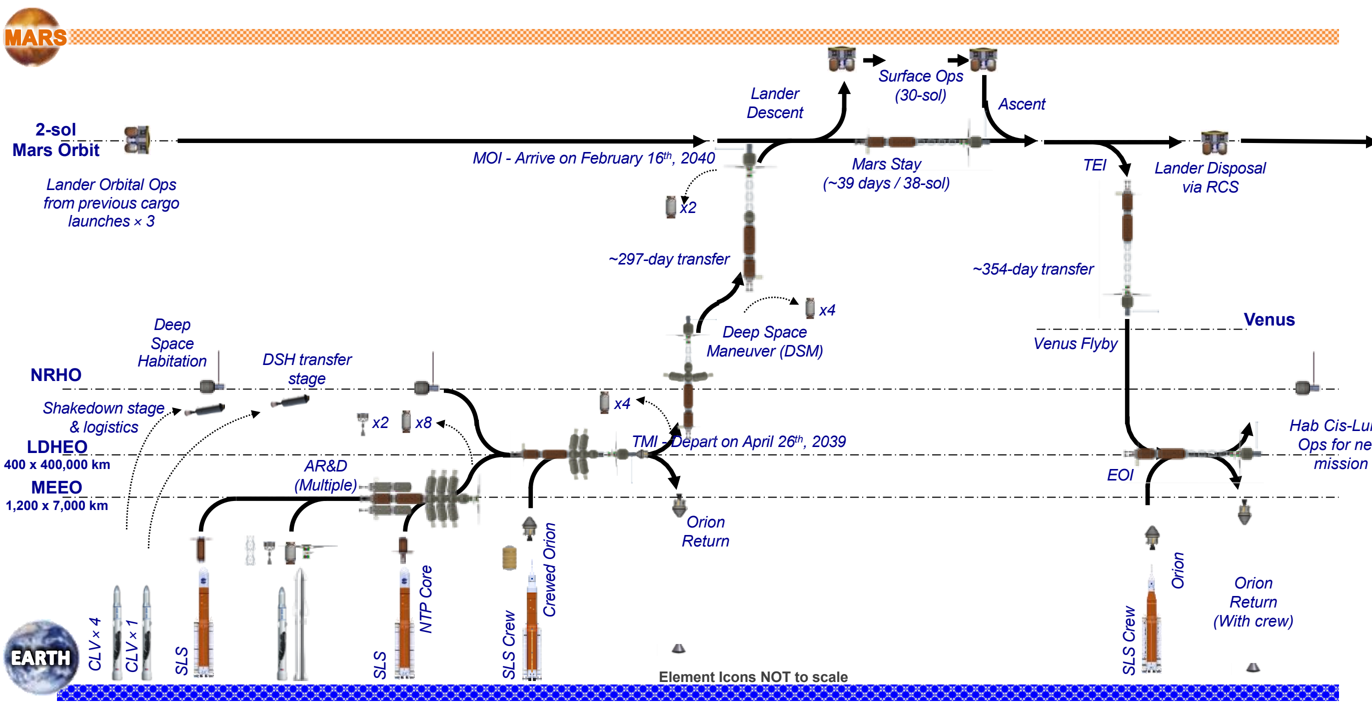
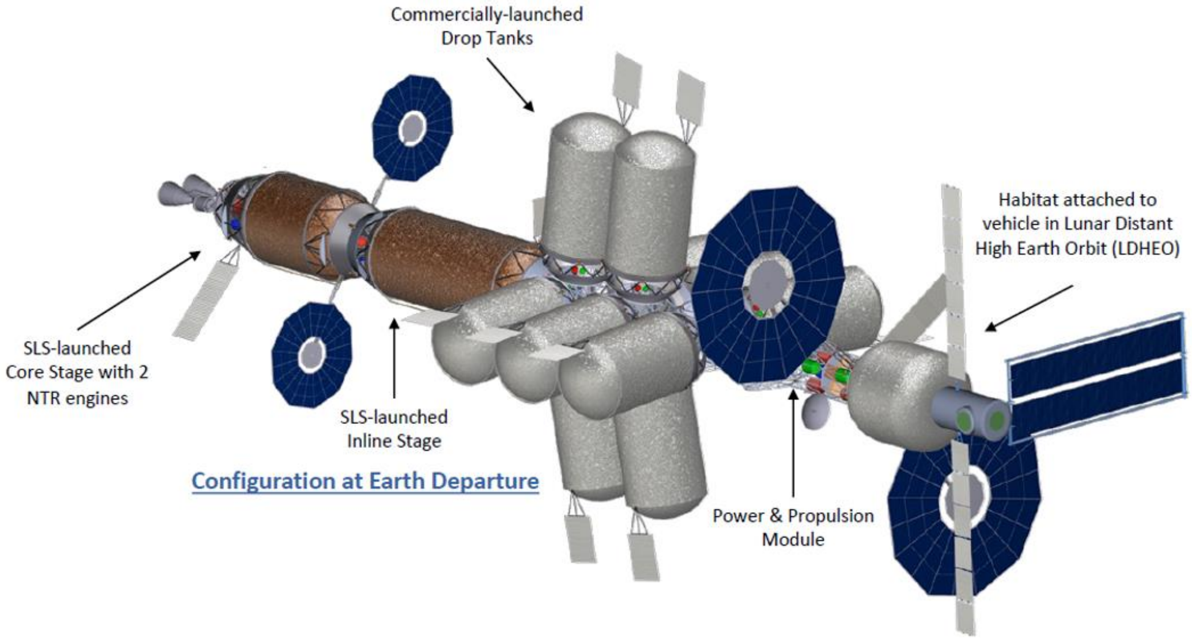


Figure 4.1-3. Mars 2039 NTP Piloted Mission ConOps.

The NTP configuration that would depart Earth orbit for Mars with a crew in the Transit Habitat is shown in center isometric view in Figure 4.1-4. The initial MEEO configuration, as well as the Earth departure and Earth return configuration are shown in the lower left of the figure. As discussed above, two side-mounted Stub Core Stages, and eight of the SH-CLV launched hydrogen inline stages will be jettisoned after the vehicle transfers from MEEO to LDHEO. The NTP option requires a Power and Propulsion Module (PPM) to provide the primary power, communications, command, and avionics for the full integrated DST vehicle. The DSH concept on the right-side of the image will rendezvous with the vehicle in LDHEO (see campaign discussion below). It will be connected to the PPM and then to the Multi-Dock Truss Element (MDTE) to which the additional CL-HIS (silver in the center of the figure) are mounted. Each Commercial-Launch Hydrogen Inline Stage is launched with a spacecraft BUS for power and orbital maneuvering to reach the aggregation point and has a radiator to reject heat from the cryocoolers that maintain the propellant at cryogenic conditions. The MDTE mounts to the SLS-launched hydrogen inline stage and the SLS-launched, two-NTR core stage element. Solar panels on the PPM and NTR core stage provide 60 kW_e of electrical power for the habitat, avionics, communications, reaction control system (RCS), and cryogenic coolers. The individual components are described in greater detail in a NASA TM.



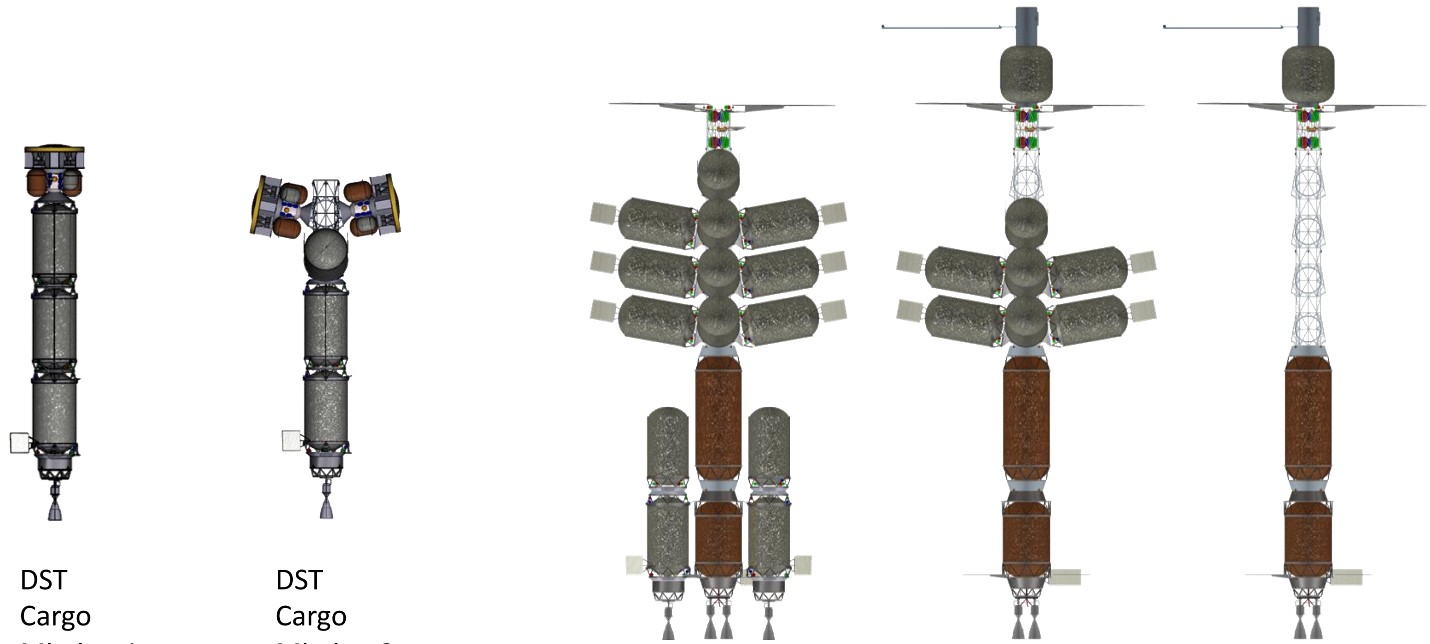


Figure 4.1-4. Mars 2039 NTP Piloted DST (Earth Departure Configuration).

Figure 4.1-4 also depicts how the NTP Stub-Core Stages and Commercial-Launched Hydrogen Inline Stages can be configured to propel the two cargo predeployment missions. The first cargo mission delivers one lander using a single subcore stage and three CL-HIS. In the second cargo mission delivering two landers requires an additional CL-HIS and a Multi-Dock Truss Element.

Based on MTAS vehicle sizing studies for the 2039 reference case, four NTR engines are needed, each producing 25,000 lbf of thrust at a specific impulse of at least 870 sec for the mission to be completed with a realistic amount of propellant. Reducing I_{sp} below 870 sec would rapidly increase the amount of propellant required and increase vehicle mass. The only propellant able to achieve this I_{sp} is hydrogen, which is discussed later in this document. For margin, the study selected a design goal of 900 sec average specific impulse, which includes engine start-up and cool-down inefficiencies. Each engine also needs to be able to operate for as long as 4 total cumulative hours to be able to continue the mission if one engine were to fail before Mars departure. The maximum number of engine start-ups (thermal cycles the reactor fuel must withstand) for the mission is about eight.

To achieve 900 sec I_{sp}, the hydrogen must be heated to an exit temperature of 2700K (higher with losses). Peak temperatures in the fuel can be a few hundred degrees above the propellant exit temperature from the reactor. To achieve these temperatures, the reactor fuel elements must be made of materials that can hold together at the high temperatures and withstand corrosive exposure to hot hydrogen. The reactor must have the endurance to provide the I_{sp} and thrust over the thrust lifetime to accomplish the mission requirements with reliability margin. Based on four hours of operation, CerMet (ceramic and metal) types of fuel elements are projected to reach I_{sp} close to 870 seconds and CerCer (all ceramic) can reach an I_{sp} of 900 seconds with margin. However, CerCer fuels have never been subject to NTP conditions, and their performance is planned to be proven under NASA's NTP development efforts. Two promising advanced carbide fuels are also predicted to be robust under these conditions and offer the potential for even higher I_{sp}. Fuel maturity and development challenges, as well as associated reactor design challenges, are being considered in the selection of a fuel that can meet these performance requirements by the late 2030s.

4.1.1 NTP Technology

The NTP development activities can be defined in three phases: (1) technology maturation, subscale component development, and non-nuclear testing; (2) full-scale development, integration, and testing; and (3) fabrication and assembly of the flight Nuclear Thermal Rocket (NTR) engines and associated stage systems. The MTAS NTPS development roadmap is depicted in Figure 4.1-5. Phase 1 includes design, fabrication, and testing of fuel elements. Phase 2 includes design, fabrication, and testing of the reactor and engine. Once a fuel and fuel element design are selected, fabricating the fuel will take an estimated ten years for the six NTR flight engines (one for each of two cargo vehicles, four for the piloted DST), plus a development qualification engine. Reactor development and qualification will also drive the overall schedule. Including the design, construction, training, Nuclear Quality Assurance (NQA-1) certification, and testing, the process is estimated to take nine years. For the development testing, extensive facility modifications will be necessary and new facilities will be necessary for integrated full-scale testing. The development roadmap culminates with the fabrication, assembly, and operational testing of the mission vehicles. Based upon this schedule for NTP development, it is unlikely NTP can support a Mars campaign until the late 2030s, assuming a fuel selection by 2025.

The first focus of an NTP development program will be the development of the fuel elements, other reactor components, non-nuclear engine components, instrumentation and control, and system engineering and integration. Fuel and fuel-element development will start subscale in Phase 1, proceeding to full scale design and testing in Phase 2. The best fuel and fuel design will be determined through subscale nuclear and non-nuclear testing. Full-scale design and development of the tanks, avionics, stage power, guidance navigation and control, and stage structures are allocated more than ten years and will be worked in parallel during Phase 1.

The most capital-intensive and time-consuming element of either vehicle concept is the development and qualification of the new reactor. This includes manufacturing large quantities of the selected fuel, designing and building specialized test facilities, and building and qualifying a unique reactor design, which are all long lead-time activities for Phase 2.

Since the end of Nuclear Thermal Rocket testing in the Nevada desert during Project Rover/NERVA, open air testing is no longer acceptable, and the development of an NTP test facility has been a major challenge to cost and schedule. The fuel, reactor core components, reflectors, Instrumentation & Control (I&C), and shielding, as well as the non-nuclear rocket engine components will all need to be developed and tested as an integrated system for flight qualification. This will require a new facility to capture and process the rocket exhaust, which may contain radioactive elements. Demonstration of viable NTP ground test exhaust capture and filtering to support full-scale facility development is a second focus of NTP technology maturation in Phase 1 to retire development risk for Phase 2.

A third focus for NTP Phase 1 technology maturation is the demonstration of Cryogenic Fluid Management (CFM) technologies necessary for long-duration hydrogen propellant storage and transfer. Storing large quantities of cryogenic propellants with negligible losses has always been a challenge. Storing liquid hydrogen for years compounds the challenge. Hydrogen is a small molecule and can readily leak through very small gaps in valve seats and seals at other interfaces. Further, to prevent boil-off losses, liquid hydrogen must be maintained at extremely cold temperature (20 K or -423 °F). Zero boil-off (ZBO) of liquid hydrogen and very low hydrogen leakage during the two-year mission has been assumed. Achieving this will require the development of no/low leakage valves and couplers, advanced insulation, and high-capacity, high-efficiency cryocoolers. Finally, the NTP concept leverages integration of multiple liquid hydrogen tanks to satisfy the full propellant requirement. Managing a cryogenic propellant, like liquid hydrogen, during microgravity coasts greater than a few hours and transferring that propellant from one tank to another in space has never been demonstrated; these represent additional technology maturation challenges. The Artemis missions, and other commercial space transportation developments share the need to solve the challenge of long-term cryogenic fluid management and transfer for hydrogen as well as other cryogens. NASA and industry are currently investing in technologies to solve these challenges, but many of the technologies remain low maturity.

Because of NASA contract obligations related to NTP development at the time of this writing, a more detailed technology plan is not provided here.

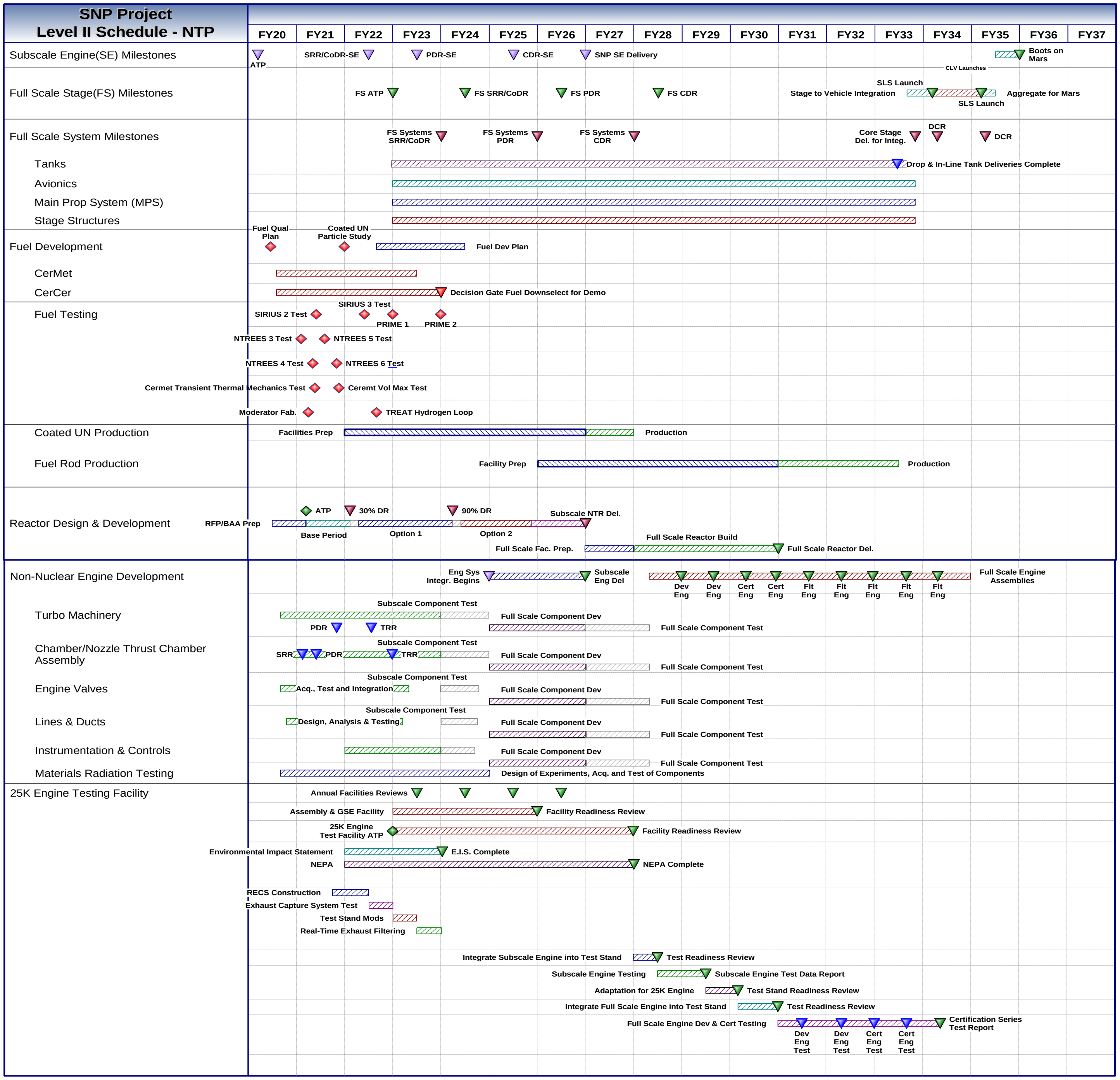


Figure 4.1-5. NTP Development Plan executing to a 2034 launch date.

4.1.2 NTP Campaign

For the Nuclear Thermal Propulsion option, the mission architecture has six unique transportation system elements (in bold, below), and requires a total of 32 launch elements to support the piloted mission, including 26 that are unique to the NTP campaign. These elements are:

- Space Launch System launched **Nuclear Thermal Propulsion Core Stage** (1)

- Space Launch System launched **Hydrogen Inline Stage** (1)
- Heavy CLV launched **Nuclear Thermal Propulsion Stub Core Stage** (2)
- Heavy CLV launched **Multi-Dock Truss Element** (2)
- Heavy CLV launched **Power and Propulsion Module** (1)
- Super Heavy CLV launched **Hydrogen Inline Stage** (18)
- Heavy CLV launched Chemical Propulsion Stage for DSH transfer (1)

Common piloted launch elements:

- Space Launch System Orion launches for crew delivery and return (2)
- Heavy CLV launched Deep Space Habitation (2)
- Heavy CLV launched Chemical Propulsion Stage for DSH shakedown cruise (1)
- Heavy CLV launched logistics for Deep Space Habitation (1)

As an example of the variation between different opportunities in the Earth-Mars synodic cycle, a 2035 mission would require fewer than half the commercial launched hydrogen inline stages needed for the 2039 mission.

The Deep Space Habitation (DSH) is launched early in the campaign to support Artemis and Gateway activities. The Mars mission campaign includes two H-CLV launches to deliver and outfit the DSH in NRHO where it will be checked-out at Gateway. Then a third H-CLV launch provides a chemical propulsion stage to perform a DSH shakedown cruise. Finally, an H-CLV delivers logistics to the DSH prior to Mars departure.

The Nuclear Thermal Propulsion Core stage is the primary propulsion system for the integrated vehicle. It is launched on a Block 2 variant of the SLS. The Nuclear Thermal Propulsion Stub Cores Stages are smaller elements that can be launched on heavy class commercial launch vehicles. The architecture requires two of these Stub Core Stages for the crew mission. There are two types of inline stages to hold the hydrogen propellant needed for the mission: one large inline stage launched on a Block 2 variant of the SLS and eighteen smaller inline propellant stages that are sized to launch on super heavy class commercial launch vehicles. Two, multi-dock truss elements are required to support the propulsion system and the hydrogen inline stages that make up the integrated stack. These are launched on heavy class commercial launch vehicles. The PPM element is also launched on a heavy class commercial launch vehicle. Finally, as with the NEP/Chemical hybrid option (discussed in the next section), the DSH is prepositioned in cislunar NRHO for checkouts. However, because the NTP vehicle does not travel to cislunar space for aggregation, an additional Heavy-CLV launched chemical propulsion stage is needed to retrieve the DSH for rendezvous in LDHEO with the transportation system to support the piloted mission to Mars.

The launch campaign for the NTP reference case is depicted in Figures 4.1-2 and 4.1-3. For the campaign build up, the Nuclear Thermal Propulsion option optimized its aggregation orbit to a single medium elliptical Earth orbit of 1,200 km x 7,000 km altitude, which is considered safe for nuclear operation. All transportation system elements are aggregated in this orbit except for the Deep Space Habitation, and unlike the NEP/Chemical Hybrid option, all elements in the NTP option are launched fully fueled to the aggregation orbit. This orbit was chosen to maximize the orbital energy provided by the SH-CLV while at the same time maximizing the mass of LH_2 propellant delivered in each hydrogen inline stage that fills the launch vehicle shroud volume. The optimization of the aggregation orbit for NTP crew and cargo vehicles was the major campaign improvement between Iteration 1 and Iteration 2. Iteration 1 considered NRHO as the aggregation point for both options, significantly reducing the payload delivery capability of all launch vehicle options, while placing the Mars propulsion systems in a much higher energy orbit, reducing their necessary ΔV. By optimizing the aggregation orbit vs NRHO, the launch campaign unique to NTP for the piloted mission was reduced from 72 launches to 26

The NTP predeploy campaign is supported by initial uses of transportation system elements for the piloted mission, and requires a total of 15 launch elements, including 12 that are unique to NTP. These elements are:

- Heavy CLV launched Nuclear Thermal Propulsion Stub Core Stage (2)
- Heavy CLV launched Multi-Dock Truss Element (1)
- Super Heavy CLV launched Hydrogen Inline Stage (7)
- Heavy CLV launched Chemical Propulsion Stage (2)

Common cargo launch elements:

- Space Launch System launched Mars Landers (3)

For the NTP launch campaign, the landers will be delivered to Mars by using the smaller Stub Core and series of hydrogen inline stages, all launching on commercial launch vehicles. The first lander, delivered on the 2035 Mars conjunction mission opportunity, utilizes a single Stub Core Stage launched on a Heavy-CLV and three Super-Heavy-CLV-Launched hydrogen inline stages (see Figure 4.1-6). For this mission, the transportation system will rendezvous with the lander at the MEEO drop off point of the lander by the SLS before departing for Mars. The second and third landers, delivered on the 2037 Mars conjunction mission opportunity, utilizes a single Stub Core Stage, four CL-HIS, and a single MDTE launched by an H-CLV. To help reduce the launch cadence requirement of the SLS, the second lander is boosted to cislunar space for aggregation using commercial launch vehicles and the third lander is picked up by the NTP cargo transfer vehicle in MEEO. After the NTP cargo transfer vehicle rendezvous with the third lander, the system transits to cislunar space to retrieve the awaiting second lander before departing for Mars.

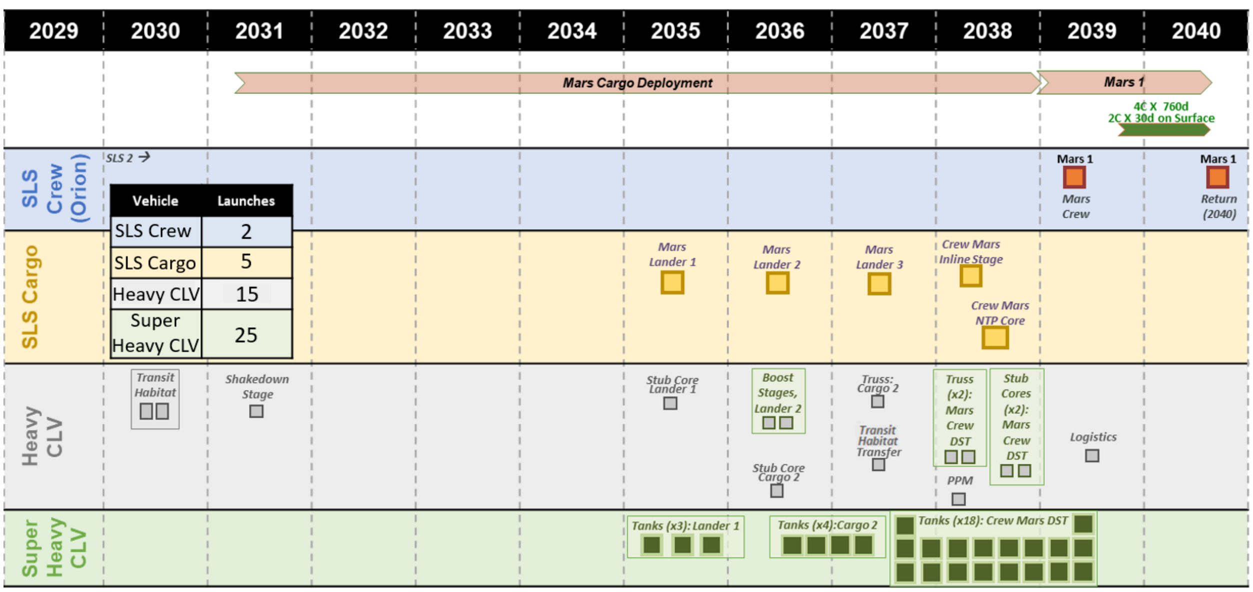


Figure 4.1-6. NTP Cargo Vehicles and Piloted Vehicle Launch Campaign.

Overall, to support the complete crew mission to the Mars surface, the Nuclear Thermal Propulsion option requires a total of forty-seven launch vehicles for the full campaign, including 38 that are unique to NTP. These launches are mostly clustered between 2035 and 2041 (including the crew return), with significant concentration in 2037 and 2038. A total of seven Space Launch Systems are required: two crew variant and five Block 2 cargo variant. Fifteen heavy class commercial launch vehicles deliver chemical propulsion stages and supporting elements where propellant volume is not a consideration, like the Stub Core Stages, the Power and Propulsion Module, and the MDTE for the cargo and crew transfer vehicles. Finally, twenty-five super heavy class commercial launch vehicles are needed to deliver twenty-five inline stages full of hydrogen.

4.2 Nuclear Electric Propulsion Reference Case

The 2039 design reference mission interplanetary trajectory for the hybrid NEP – Chemical Propulsion concept is shown in Figure 4.2-1. The trajectory is optimized to employ the high thrust and low thrust systems where they are most advantageous. High thrust chemical propulsion is employed to propel the vehicle away from Earth/Moon at Trans Mars Injection (TMI), then again at Mars Orbit Insertion (MOI) and Mars Departure/Trans Earth Injection (TEI) after which point it is jettisoned. The low-thrust NEP system would continuously accelerate the vehicle more than halfway to Mars, and then decelerate it for the remainder of the trip. The return trip would be accomplished after firing the chemical stage and using the NEP thrusters to escape Mars followed by an extended coast for a large fraction of the return trip. After a Venus Gravity Assist (VGA), the NEP system will decelerate the vehicle so it can be captured into a high elliptical orbit around Earth. The total time away from Earth would be 760 days (25 months).

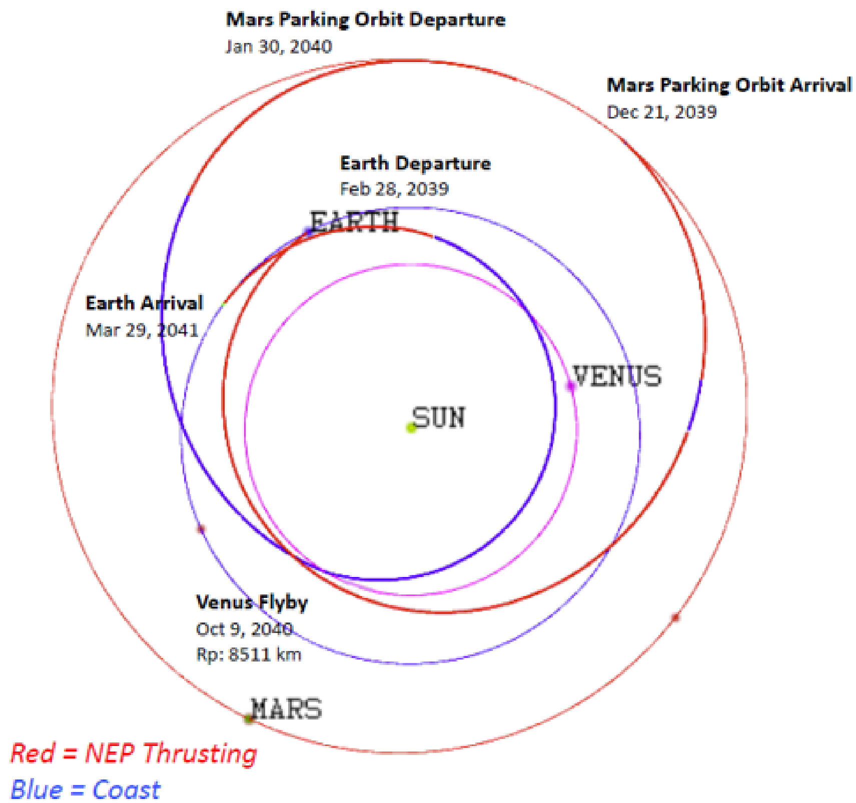


Figure 4.2-1. NEP/Chem Schedule and Trajectories for the 2039 Design Mission.

Figure 4.2-2 shows the ConOps for the cargo predeployment campaign utilizing the NEP/CCP elements. The first cargo transfer vehicle is a unit of the Chemical Propulsion Module (CPM) also used for the piloted DST. It is launched partially filled to NRHO, based on the capability of a Block 2 SLS, and LOX and LCH_4 propellants are brought to NRHO to fill the LOX and LCH_4 propellent tanks. The first lander would be launched at MEEO, with a maximum energy determined by the capability of an SLS and assembled and fueled there. The lander is raised to NRHO by chemical propulsion upper stages for integration to the CCP transfer vehicle and from there departs for Mars.

After the first lander is successfully on Mars, the second and third landers are similarly shown being assembled and fueled in MEEO before being raised to NRHO. This happens simultaneously with the assembly of the first unit of the NEP Propulsion Module (NEPM) with a Xenon Interstage Module (XIM) in 500 km circular LEO, to be used for the second cargo transfer vehicle. The NEP cargo transfer vehicle is raised to a 1100 km circular nuclear safe orbit by a chemical propulsion upper stage, where the NEP system is commissioned and used to raise the vehicle to NRHO. Once the NEP cargo transfer vehicle and the two landers are mated, they depart for Mars from NRHO. The Mars Ascent Vehicle (MAV) descends to the Mars surface and is verified operational before the piloted vehicle departs Earth orbit.

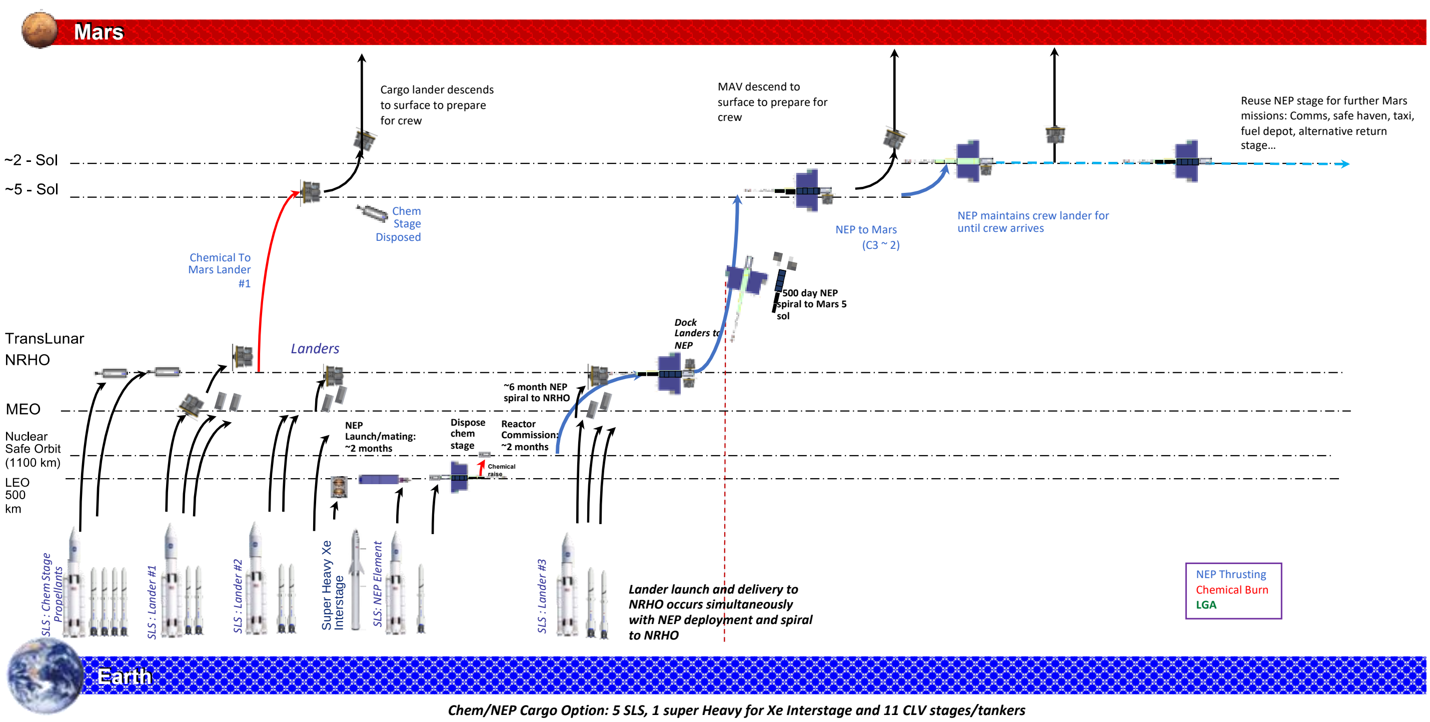


Figure 4.2-2. Mars 2039 NEP/Chem Cargo Lander ConOps.

Figure 4.2-3 shows the ConOps for the NEP/CCP piloted Deep Space Transport mission. Its components are a second unit of the NEP Propulsion Module, a second unit of the Chemical Propulsion Module (CPM), and three Xenon Interstage Modules containing additional propellant. These components are delivered to a 500 km circular LEO to utilize the maximum payload capability of Block 2 SLS and Super-Heavy CLVs. Once assembled, the Chemical Propulsion Module propels the DST more rapidly through the low Earth orbits where it will need to avoid a relatively high concentration of space debris to an 1100 km nuclear safe orbit. Once the NEP system is commissioned, and the Chemical Propulsion Module propellants are refilled, the NEP Propulsion Module propels the DST through a spiral trajectory to NRHO. The DST will rendezvous with the Deep Space Habitation and jettison two depleted XIMs in NRHO. Using NEP, the Deep-Space Transport transfers to Lunar-Distant High Earth Orbit of 400 km x 400,000 km altitude, where Orion will rendezvous with the DST to bring the crew. The DST departs LDHEO for Mars via a chemical propulsion burn. NEP is used in interplanetary space to accelerate the trajectory, and then chemical propulsion is used to capture the DST at Mars before a series of NEP burns brings the DST to a two-sol staging orbit. In a two-sol orbit, two crew members transfer to the lander that takes them to the surface of Mars. After 30 sols of surface operations, the surface crew uses the MAV to return to the DST. Following the final chemical stage burn, the chemical stage is jettisoned, and the DST returns to Earth using NEP and via a Venus gravity assist. Finally, a Lunar Gravity Assist (LGA) maneuver captures the DST with the crew and DSH back to LDHEO. A second Orion capsule rendezvous with the DST in LDHEO to bring the crew home. The DST is parked in NRHO using NEP for disposal or potential future use.

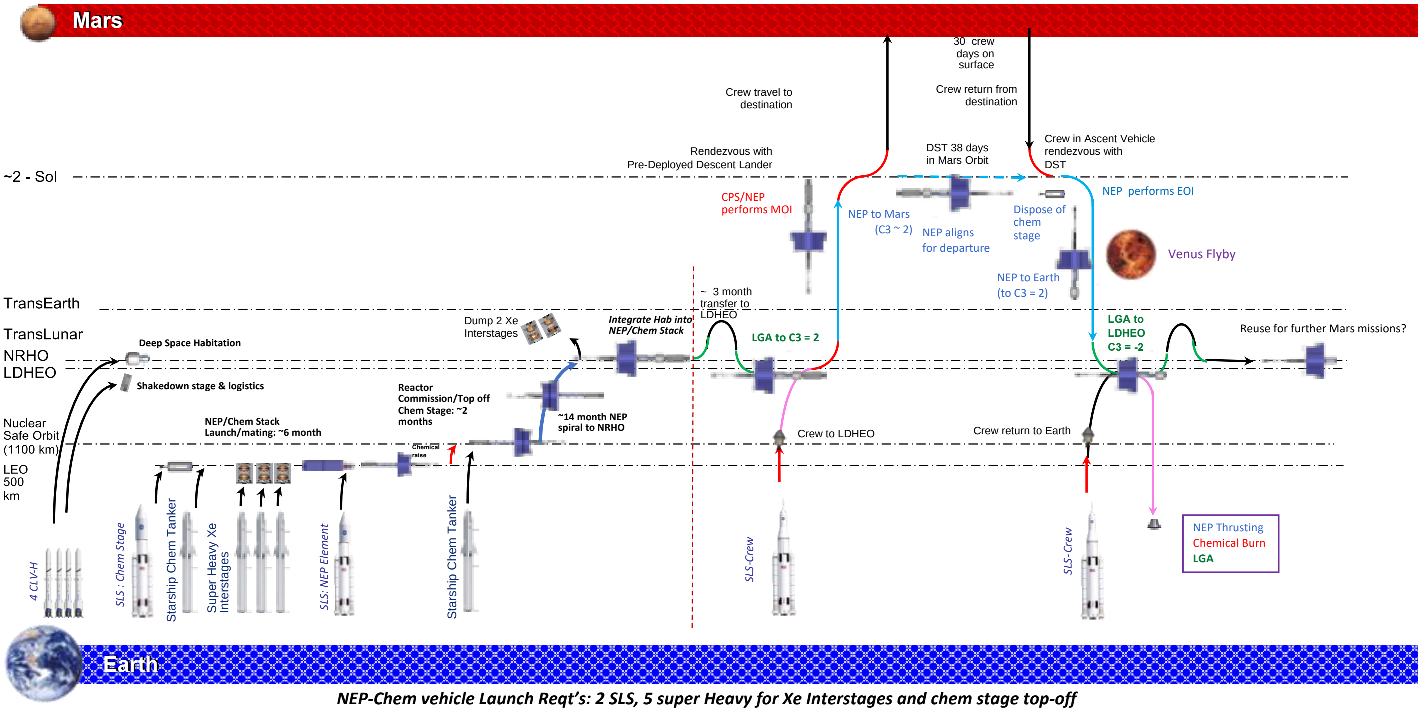


Figure 4.2-3. Mars 2039 NEP/Chem Piloted Mission ConOps.

The NEP/CCP reference concept is shown in isometric view in Figure 4.2-4 in the Earth departure configuration. This reference design was the result of a COMPASS conceptual level design study, which was not exhaustive, and should not be considered optimal. More advanced configurations were identified and merit further study. Planform views of the aggregation orbit configuration, with additional Xenon Interstage Modules, and the earth return configuration are shown below it. The NEP Propulsion Module is on the left. A 50 m telescoping boom holds the nuclear reactor far from the radiator, electronics, and Deep Space Habitation. A radiation shield at the base of the reactor creates a radiation-free cone that envelopes all those vehicle components. The electric thrusters are positioned to propel the vehicle in the direction of the Deep-Space Habitat and Chemical Propulsion Module. The radiators are depicted blue, and the electronics and Power Management and Distribution (PMAD) are within the box at the base of the radiators. The XIM is between the radiators and the DSH, containing several xenon propellant tanks. The companion LOX-CH_4 Chemical Propulsion Module (CPM) is at the right end of the figure. The NEP system was sized to maximize the power system that could be launched in a single SLS 8.6 m diameter, long SLS fairing. The CPM was also sized to volumetrically fit in a second, single SLS 8.6 m diameter, long SLS fairing.

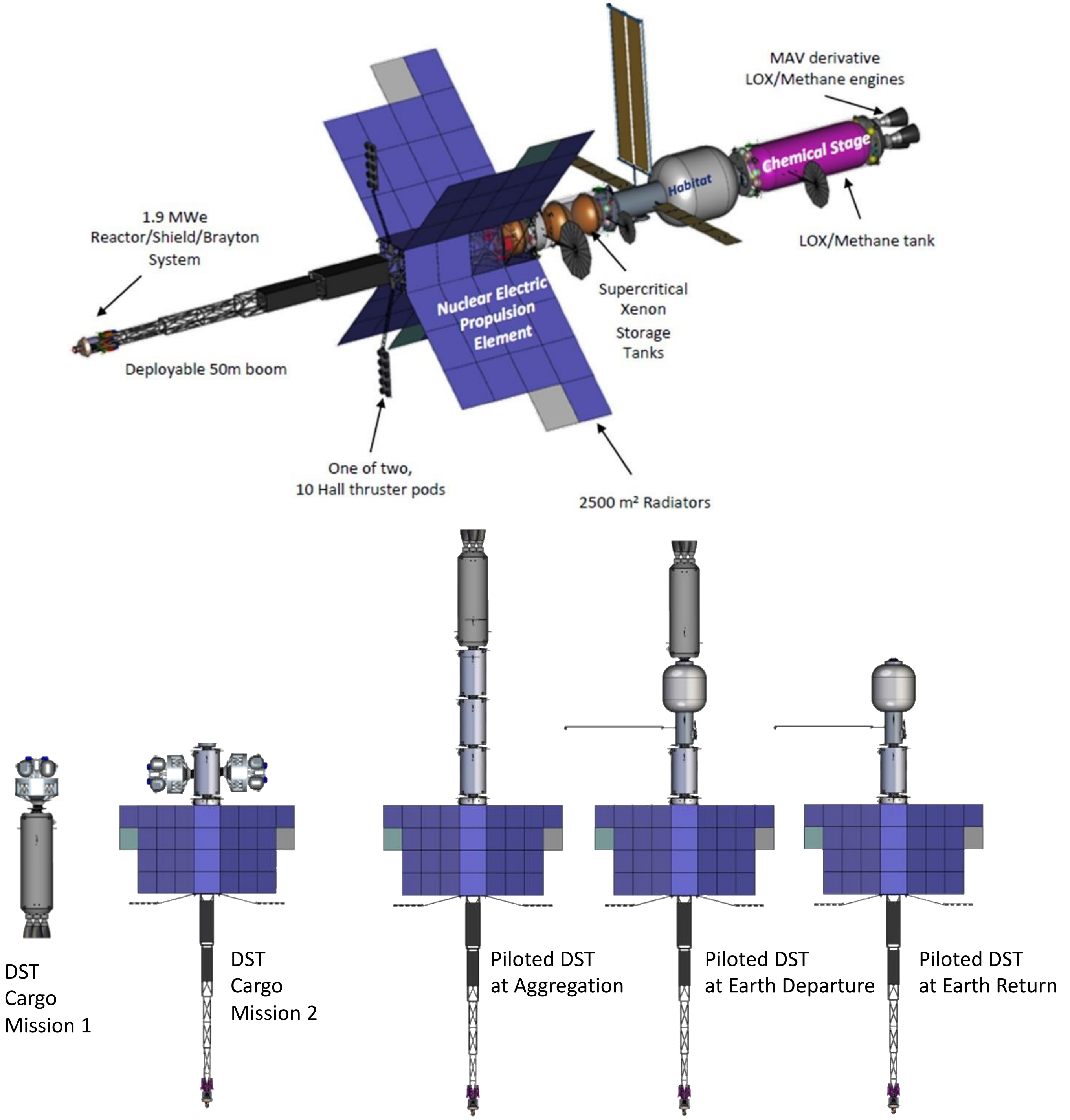


Figure 4.2-4. NEP/CCP Reference Vehicle Concept.

Figure 4.2-4 also depicts how the NEP module and the Chemical Propulsion Module are used separately to propel the two cargo predeployment missions. The NEP module can propel all three landers to Mars on subsequent missions, when the first lander is not required to predeploy before the remaining two as a capability verification.

MTAS assumed Hall-effect thrusters for the NEP system, based on the development of high-efficiency, 12.5 kW_e Advanced Electric Propulsion System (AEPS) thrusters for the Gateway SEP Power and Propulsion Element. Additionally, higher power Hall thruster technology has been demonstrated by the NASA 457M, and a separate, nested channel design, which have been tested up to 100 kW_e. Based on these technologies, a specific impulse (I_{sp}) of ~2600 seconds was assumed for estimating the propellant required for the NEP mission. This I_{sp} is near-optimum for the reference Mars trajectory. Other thruster technologies are possible in this size and I_{sp} class and should be considered in subsystem trades as identified in follow-on work at the end of this document. The I_{sp} of the LOX-CH_4 chemical propulsion system was assumed to be ~350 sec.

A major consideration in the design of a nuclear electric propulsion system is the size of the power conversion radiators, which define packaging and deployment requirements as well as the need for in-space assembly of large structures. Radiator size is determined by the size and efficiency of the thermal-to-electric power conversion, and the designed heat rejection temperature for the operating environments encountered. The radiators are a critical element of the power conversion cycle, and they are required for mission success. To minimize the risk of coolant leaks, the current best practice is to assemble, fill and check the radiators for leaks before launch, and avoid disassembly/reassembly in space. Thus, it is advantageous to package the radiators in a single launch vehicle with the reactor and power conversion, with a reliable and structurally sound means to deploy them on orbit. A concept was developed to fold a fully assembled radiator as large as 2500 m^2 to fit within the volume of an SLS 8.6 m diameter, long cargo fairing, based on technology developed under Project Prometheus. A more advanced radiator folding concept was identified for about 4000 m^2 in the same volume. However, to minimize the packaging complexity, number of folds, and flexible fluid joints, 2500 m^2 was assumed as a constraint to size the MTAS NEP power system. As in-space assembly capabilities improve, more advanced systems requiring larger radiators can be considered.

The MTAS NEP reference concept assumed the use of a liquid-metal cooled reactor with Brayton power conversion, utilizing Supercritical-CO_2 as a working fluid. The reactor concept reference is a HALEU TRISO fuel in a moderated core, based on the DOE Transformational Challenge Reactor (TCR) in development, or alternately an HEU UN pin fuel in a fast-neutron spectrum core based on the heritage SP-100 design. Other coolant, power conversion, and working fluid options can be considered in this mission time frame and are identified as subsystem trades for follow-on work. Assuming these fuel and power conversion technologies and the desire to minimize the use of refractory metals due to manufacturability and testing challenges, the reactor outlet temperature was designed to be 1200K permitting a power conversion hot-side temperature of 1150K with nickel-based superalloy construction. These design choices and constraints define the nuclear electric power system size to be about 2 MW_e. With conservatism, the mission was designed with 1.8 MW_e provided to the propulsion system, with an additional 0.1 MW_e needed for the habitat and vehicle systems. With a chemical propulsion system providing the high thrust needed near high-gravity bodies such as the Earth, Moon, and Mars, 1.8 MW_e propulsion power was found to be near-optimum for maximizing overall mission performance.

In comparison, an all-NEP Deep-Space Transport for the same 2-year piloted mission and ConOps would require 5+ MW_e for propulsion power. While such power levels and propulsion systems are feasible, such a system introduces additional challenges on technology development and system packaging. Most significantly, the power conversion radiators would be 2.5 times larger, requiring multiple launches due to volume packaging constraints, and have associated fluid connections that would have to be mated, sealed and verified as reliable on orbit. Such in-space assembly and check-out operations are considered feasible in the 2030s, but the study considered it lower risk to avoid them. The higher propulsion power is beyond the limits of current Hall thruster technology as well as existing test facilities, driving the need for more advanced thruster concepts and larger test facilities unlikely to support a 2030s mission. Thus, the hybrid NEP/CCP propulsion concept innovated under MTAS enables the consideration of a 2030s NEP mission with achievable technology.

4.2.1 NEP Technology

A preliminary MTAS NEP development plan is presented in Figure 4.2-5. It consists of three major phases: (1) technology maturation, (2) engineering development (DDT&E), and (3) flight system development. The first step is the development of detailed government reference designs of the NEP reactor power system and electric propulsion system to inform hardware procurement efforts with industry. While the MTAS concept studies provide a good starting point, they are not of sufficient fidelity to establish design requirements for industry RFPs. A Failure Modes and Effects Analysis (FMEA) would be a key element of the system engineering process that starts under the government reference design and is continued by the industry subsystem suppliers. The results would inform subsystem testing requirements and higher-level vehicle decisions being made by the system integrator. In lieu of the FMEA, the team referenced prior projects and used engineering judgement to determine testing requirements for this study.

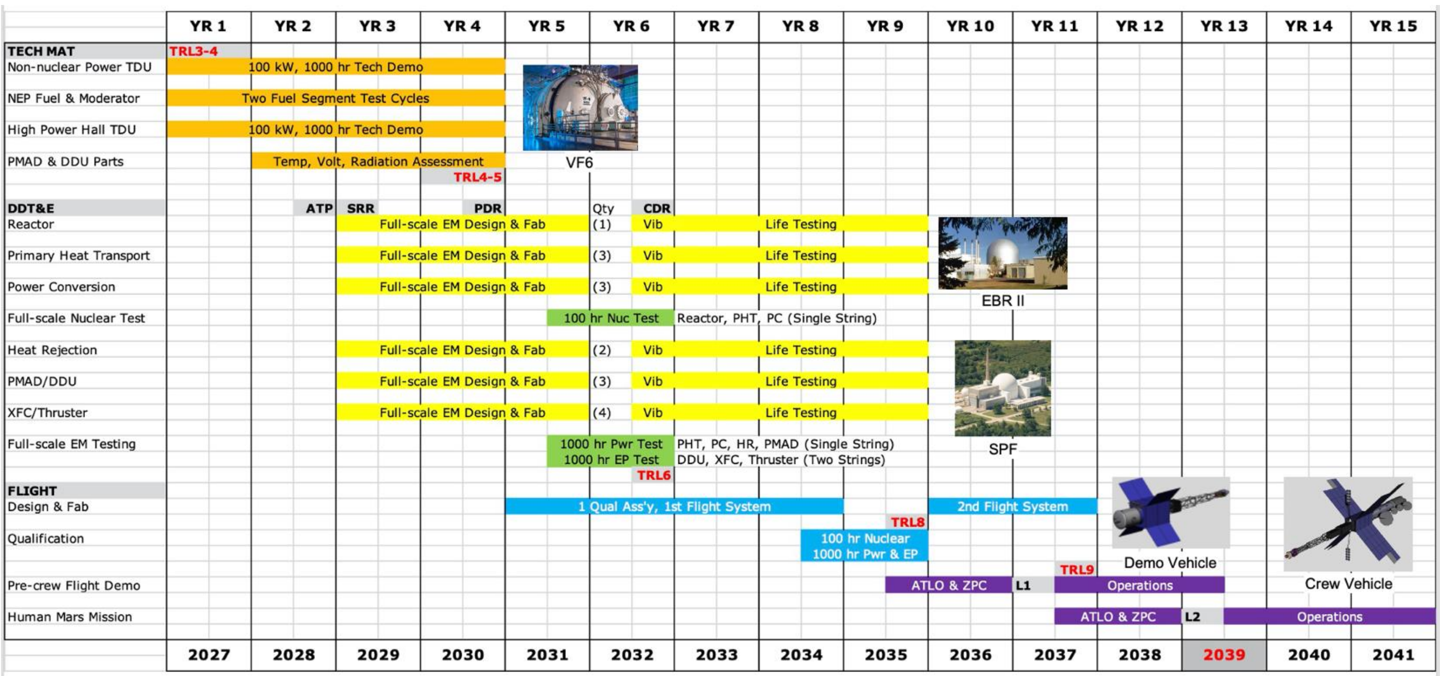


Figure 4.2-5. Preliminary NEP Power Development Plan.

The technology maturation phase would include three main elements: power and EP Technology Demonstration Units (TDUs), reactor fuel characterization, and PMAD/DDU parts assessment. The power and EP TDUs will focus on representative-scale breadboard subsystems that could be combined for integrated system tests to advance the TRL. A non-nuclear power system TDU, shown in Figure 4.2-6, would perform a 1000-hour performance test at GRC Vacuum Facility #6 (VF6), integrating:

- a 500 kW_t reactor simulator,
- 1200 K lithium primary loop,
- 100 kW_e Brayton unit,
- 500K NaK heat rejection loop,
- and two 40 m^2 radiator panels

The Power TDU would include an external 1000 Vac PMAD assembly coupled to simulated 650 Vdc Hall thruster loads. Similarly, full-scale 100 kWe Hall thruster, 650 Vdc DDU and XFC breadboard test articles will be procured and tested at GRC Vacuum Facility #5 (VF5). In parallel, DOE would lead efforts to characterize and qualify the NEP reactor fuel and moderator through several test iterations. The DOE fuel development effort is envisioned to be like the Advanced Gas Reactor TRISO fuel qualification performed by INL, Oak Ridge National Laboratory (ORNL), and BWX Technologies with tasks for production, irradiation, post-irradiation examination, and modeling. The final element of the technology maturation phase is the electronic parts assessment for the PMAD and DDU. This activity will focus on a combination of radiation and thermal testing to evaluate high voltage electronics with an emphasis on wide band gap devices such as SiC or GaN. The technology maturation phase will conclude with the Mission PDR in year 4 at which time most of the power and EP technologies will have reached TRL 5. ROM cost estimates by the MTAS NEP team estimate the technology maturation effort to be about $350M over four years.

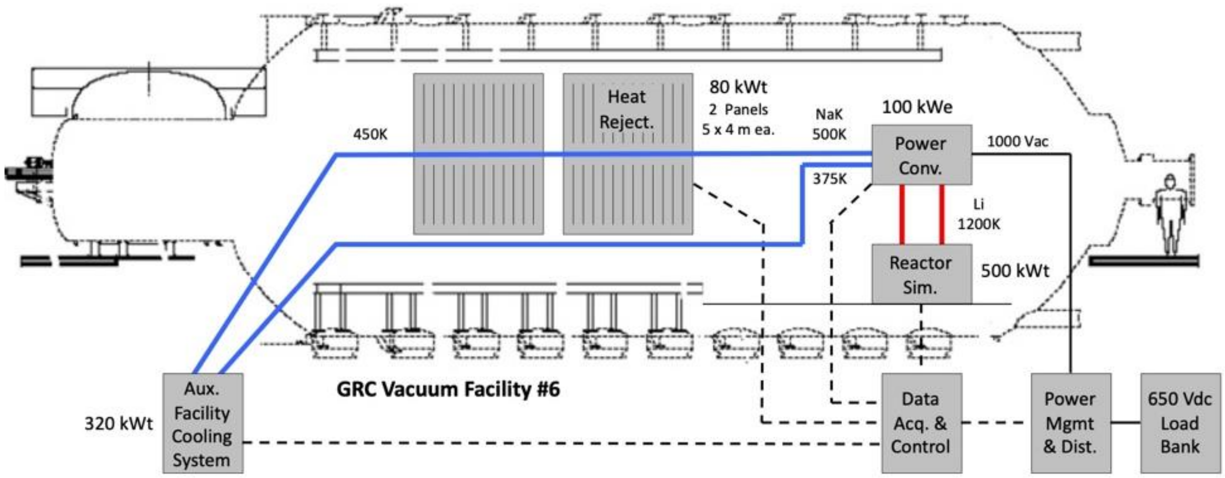


Figure 4.2-6. Non-nuclear Power TDU.

The engineering development phase of the NEP power system development plan starts with the mission SRR in year 3. The design of full-scale engineering model subsystems will occur in parallel and be informed by the TDU testing. The plan is to build multiple full-scale Engineering Models (EMs) that would be used to support various test objectives. A complete full-scale power string will be assembled with a high-fidelity reactor simulator, single Brayton unit, heat rejection loop, and PMAD power channel for an integrated 1000-hour non-nuclear system test, possibly at the Armstrong Test Facility (ATF), formerly known as Plum Brook Space Power Facility (SPF). A second set of EM units will be subjected to launch vibration environments and placed on long-duration tests to evaluate performance relative to the mission design life, currently estimated at 27,100 hours for the power system and 22,860 hours for the EP system. Life verification will be accomplished through a combination of testing and analysis, similar to the approach being used for the AEPS thrusters planned for Gateway. A third Brayton unit will be supplied to the DOE reactor team for use in a full-scale 100-hour reactor prototype test to demonstrate nuclear performance. Candidate nuclear facilities to accommodate the reactor prototype test include the INL Experimental Breeder Reactor II (EBR-II) facility or the Nevada National Security Site, U1A facility. Similar testing will be performed on full-scale EM thrusters, DDUs, and XFCs culminating in a thruster performance verification test at GRC VF6. The three separate integrated tests of full-scale EM test articles (non-nuclear power string, reactor prototype, and EP string) combined with the subsystem launch vibration tests would provide critical data to support Mission CDR and establish TRL6 for the key NEP elements by year 6.

The presumed use of GRC and ATF vacuum facilities for the non-nuclear power and EP testing is based on the need to provide a large volume and high pumping speed. There are other vacuum facilities at NASA (e.g., JSC Chamber A), Department of Defense (e.g., Arnold Engineering Development Complex), industry, and academia, but they are smaller and lack the pumping speed needed for high power EP thrusters. Current facilities prohibit simultaneous testing of multiple 100 kW thrusters, but there is no precedent to do so. Tests performed of multiple thrusters at lower power levels indicate negligible thruster interactions when appropriate separation distance is maintained. An upgraded VF6 could be utilized to test full-scale EP EMs. Analysis shows that pumping speeds of 1600-1800 kL/s xenon could be achieved corresponding to a background pressure of 1.5E-5 torr for a 2600s I_{sp}, 60% efficiency 100 kW thruster. Alternatively, ATF could be used, but vacuum facilities there would require upgrades that are more extensive than VF6.

One of the primary MTAS ground rules was to deliver crew to Mars by 2039, which drove the technology decisions (and thus the facilities) selected for the mission concept. If NASA postpones the Mars missions to the late 2040s or 2050s, lower TRL technology and construction of new facilities could be pursued.

The flight system development phase will include design, fabrication, qualification, and acceptance testing of flight components. A full assembly of subsystem qualification hardware elements will be built and tested. The first complete set of flight components would be used for an NEP demonstration mission, possibly an all-NEP Mars cargo delivery mission that would precede the crew mission. That flight mission would validate the overall NEP propulsion stage performance in space before a duplicate version would be used on the crew vehicle. The cargo mission is fundamental to demonstrating in-space performance, life and reliability of the nuclear power and propulsion element before it is used to transport crew. The NEP demo vehicle would be assembled, and checkout testing of the power and EP elements would be performed using a reactor simulator. After checkout, the flight reactor would be integrated, and a zero-power critical (ZPC) test would be performed to validate reactor neutronic performance before the demo vehicle launch in year 11. The second set of flight components will be fabricated and tested in parallel with the demo mission and readied for launch with the crew vehicle in year 13. Assuming a 2039 launch of the crew vehicle and working backward using the 12-year development life-cycle results in a required project start in 2027. While the proposed NEP development schedule in Figure 4.2-5 is achievable, it is very aggressive and success-oriented with little margin for setbacks. If 2035 remains the target for the first human Mars mission, the required project start would be in 2023.

4.2.1.1 Chemical Propulsion Technology

Technology maturation is also required for the CCP, demonstrating of Cryogenic Fluid Management (CFM) technologies necessary for long-duration liquid oxygen and liquid methane cryogenic propellant storage and transfer. Storing large quantities of cryogenic propellants with negligible losses has always been a challenge. While storing LOX and LCH_4 is easier than LH_2, this capability and the CFM technologies necessary for multi-year missions have never been demonstrated in space. Oxygen and methane can readily leak through small gaps in valve seats and seals at other interfaces. Further, to prevent boil-off losses, the propellants must be maintained at extremely cold temperature (90 to 100 K or -297 to -280 °F)." Zero boil-off (ZBO) of LOX and LCH_4 and very low leakage during the two-year mission has been assumed. Achieving this will require the development of no/low leakage valves and couplers, advanced insulation, and high-capacity, high-efficiency cryocoolers. Finally, the NEP/CCP concept leverages refueling of the cryogenic propellant tanks in orbit. Managing the cryogenic propellant during microgravity coasts greater than a few hours and transferring propellant from one tank to another in space has never been demonstrated and represents additional technology maturation challenges. As with NTP, these technology challenges are shared with the Artemis missions and other commercial space transportation developments, but the risks for developing this low maturity technology remain.

4.2.2 NEP Campaign

For the NEP/CCP Hybrid propulsion option, the mission architecture has predominantly four unique transportation system elements (in bold, below) to support the piloted mission, and requires 13 total launch elements to support the roundtrip piloted mission, including 7 that are unique to the NEP/CCP campaign. These elements are:

- Space Launch System launched **Nuclear Electric Propulsion Module** (1)
- Space Launch System launched **Chemical Propulsion Module** (1)
- Super-Heavy CLV launched **Xenon Interstage Modules** (3)
- Super-Heavy CLV launched LOX-LCH_4 **propellant and transfer capability** (2)

Common piloted launch elements:

- Space Launch System Orion launches for crew delivery and return (2)
- Heavy CLV launched Deep Space Habitation (2)
- Heavy CLV launched Chemical Propulsion Stage for DSH shakedown cruise (1)
- Heavy CLV launched logistics for Deep Space Habitation (1)

The Deep Space Habitation is potentially part of the initial Artemis & Gateway program and is launched in the early 2030s to support various lunar activities and Mars analogue missions. The Mars mission campaign includes two H-CLV launches to deliver and outfit the DSH in NRHO where it will be checked-out at Gateway. Then a third H-CLV launch provides a chemical propulsion stage to perform a DSH shakedown cruise. Finally, an H-CLV delivers logistics to the DSH prior to Mars departure.

The Nuclear Electric Propulsion Module and the Chemical Propulsion Module are launched on a Block 2 variant of the Space Launch Systems, while the Xenon Interstage Modules and the LOX-CH_4 propellant refueling vehicles are launched on super heavy classed commercial launch vehicles. The remaining elements are launched starting in the mid-2030s to support the piloted mission to Mars in 2039.

The launch campaigns for NEP/CCP predeployment and piloted missions are depicted in Figures 4.2-2 and 4.2-3. Both figures show the launch of elements to a 500 km circular Low Earth Orbit (LEO) aggregation orbit. This orbit was chosen for the NEP/CCP option to maximize the launch vehicle payload delivery, where shroud volume for propellant is not a consideration. The five hardware elements and two refueling flights must be launched and aggregated in a relatively short period of time, as the thermal and orbital debris environment in low Earth orbit can pose challenges to the systems. After the elements have been aggregated, the integrated stack performs orbit raising maneuvers using the Chemical Propulsion Module to a higher Earth orbit of 1,100 km altitude, which is considered nuclear safe, with consideration for orbital debris and the Van Allen radiation belts. From here, the nuclear reactor is deployed, and the stack performs orbit raising maneuvers using the electric propulsion system over an 18-month period to reach the final aggregation orbit at NRHO in cislunar space, where the propulsion system rendezvous with the Deep Space Habitation in preparation for the piloted mission. Finally, approximately six months prior to crew departure for Mars, the integrated deep space transport performs a low energy transfer from its aggregation orbit in cislunar space to a lunar-distance high Earth orbit of 400 km x 400,000 km altitude to rendezvous with the crew and prepare for the piloted mission to Mars.

The optimization of the aggregation orbit for NEP crew and cargo vehicles was the major campaign improvement between Iteration 1 and Iteration 2. Iteration 1 considered NRHO as the aggregation point for both options, significantly reducing the payload delivery capability of all launch vehicle options, while placing the Mars propulsion systems in a much higher energy orbit, reducing their necessary ΔV. By optimizing the aggregation orbit vs NRHO, the overall launch campaign for NEP/CCP was reduced from 41 launches to 30.

To delay the first need date of the nuclear system, the predeployment campaign concept of operations for the NEP/CCP option utilizes a combination of the two primary propulsion elements to deliver the three lander systems. The first lander is delivered using the Chemical Propulsion Module identical to the piloted mission during the 2035 Mars conjunction mission opportunity. Landers 2 and 3 are delivered to Mars using a Nuclear Electric Propulsion Module and a single Xenon Interstage Module during the 2037 Mars conjunction mission opportunity. To support this cargo campaign, a total of 17 elements are required, including 14 that are unique to NEP/CCP:

- Space Launch System launched Nuclear Electric Propulsion Module (1)
- Super-Heavy CLV launched Xenon Interstage Modules (1)
- Space Launch System launched Chemical Propulsion Module (1)
- Heavy CLV launched Chemical Propulsion Boost Stage (7)
- Heavy CLV launched LOX-LCH_4 propellant (4)

Common cargo launch elements:

- Space Launch System launched Mars Landers (3)

The ~65,000 kg Mars Lander systems are launched on Block 2 variants of the Space Launch System. They are meant to aggregate in cislunar space, at NRHO, but because the SLS launch capability is limited to ~45,000 kg to translunar injection condition, they are delivered to an elliptical, Medium Earth Orbit. Additional chemical "boost stages" are required to deliver the landers to the NRHO aggregation orbit. For the first lander Mars transfer, the Chemical Propulsion Module, launching on a Block 2 variant of the SLS to NRHO, is only launched partially fueled due to the payload limitation of the SLS. Thus, additional H-CLV LOX-LCH_4 propellant flights are required to supply the CPM enough propellant to deliver the lander to Mars.

The NEP Module and XIM are delivered to a 500 km circular LEO by a Block 2 SLS and SH-CLV, respectively. If placed in a 500km orbit, astronauts could help with assembly and checkout, which has proven very useful for space station assembly and checkout. To reduce the time needed to deploy the NEP module, a chemical propulsion boost stage is delivered by an H-CLV to rendezvous with the NEP cargo transfer vehicle and is employed to take the vehicle to the 1100 km circular, nuclear safe orbit. From there the NEP Module is employed to spiral to a rendezvous with Landers 2 and 3. In all, seven H-CLV launched, chemical propulsion boost stages are required to deliver all three lander systems to cislunar space for aggregation, two per lander and one for the NEP module.

Overall, to support the complete crew mission to the Mars surface, the Nuclear Electric/Chemical Hybrid Propulsion system option requires a total of 30 launch vehicles for the campaign, 21 of which are unique to NEP/CCP. As an example of how the high I_{sp} of NEP reduces the variance between launch opportunities, the same number of launches would be required for a 2035 piloted mission. For 2039, these launches are spread out between 2030 and 2041 (including the crew return) and are shown in Figure 4.2-7. A total of nine Space Launch Systems are required; two will be the crew variant and seven will be the block 2 cargo variant. Fifteen heavy class commercial launch vehicles serve mainly as support vehicles, delivering propellant and boosting stages to their final aggregation orbits. Finally, six super-heavy class commercial launch vehicles are required to provide the propellant requirements in the Xenon Interstage Modules, and the LOX-LCH_4 propellant refueling operation for the piloted vehicle during its initial low Earth orbit aggregation.

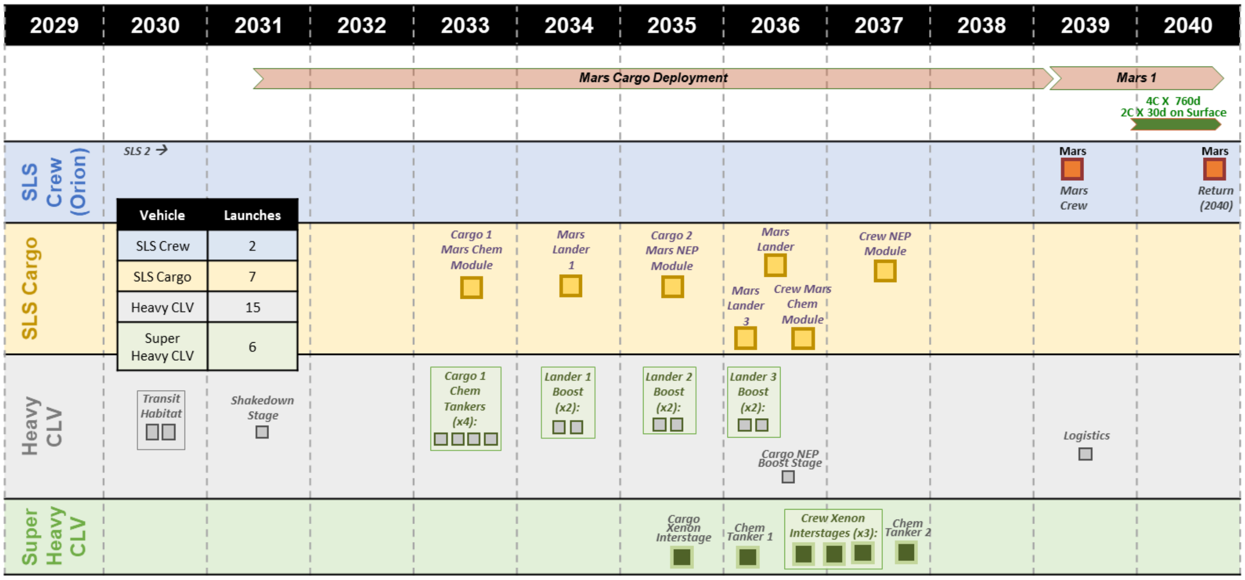


Figure. 4.2-7. NEP/Chem Cargo Vehicles and Piloted Vehicle Launch Campaign.

4.3 Comparison between Propulsion Alternatives

MTAS Phase 1 concluded that both NTP and NEP/CCP propulsion system concepts have the potential to fulfill the mission requirements for fast Mars transits in the late 2030s, including enabling two-year roundtrip mission durations. But these options use different approaches with unique challenges. While a 2035 mission is possible, it is programmatically high risk, and would require substantial investment within two years of the date of this study. The following section provides a discussion of the distinctions between the propulsion approaches.

4.3.1 Performance

Figure 4.3-1 provides a summary of the second iteration (1.2) NTP and NEP/CCP performance assessment and vehicles designed for the objective two-year piloted mission to Mars in the late 2030s. These are considered reference cases resulting from trades for technology, systems size, mission performance, and operational approach and define what performance is possible for the mission ground rules and ConOps, but do not define optimality and are open to further trades. In both cases, the cargo predeployment campaign was performed with the propulsion elements defined by the piloted mission architecture. The NTP reference mission design is for a round-trip time away from Earth orbit of 690 days, assuming an engine-out before Mars departure. The trip time below two years is enabled by a Venus gravity assist on the return trajectory in 2039. A 760-day round-trip mission comparable to the NEP/CCP reference case would save propellant mass, but the VGA would no longer be beneficial in 2039. The mission requires four, 25,000 lb_f NTR engines with ~900 sec I_{sp}, each with ~4-hour full-thrust life. The four NTR engines are needed to provide sufficient thrust to leave the aggregation orbit with acceptable gravity losses. By comparison, the NEP reference mission design is for a round-trip time away from Earth orbit of 760 days to minimize the total number of launches and accommodates partial power loss on the Earth return leg. A shorter NEP/CCP mission would require a larger LOX-CH_4 stage. The NEP/CCP mission requires: one, 1.9 MW_e reactor with twenty, 100 kW_e Hall thrusters operating at 2600 sec I_{sp}; and three, 25,000 lb_f LOX-CH_4 engines for high thrust. The difference in round-trip mission time for the two reference cases is an example of how each propulsion option optimizes differently around the two-year mission objective; in this case the timing of a Venus gravity assist for the 2039 impulsive trajectory provides an advantage. Both the NTP and NEP systems are capable of mission times significantly less than two years at the expense of increased propellant mass, and/or larger propulsion systems, and/or more advanced technology. However, the two-year round-trip objective is a reasonable target, resulting in performance requirements for Nuclear Thermal Propulsion and Nuclear Electric Propulsion (when augmented with Chemical Propulsion) that are achievable by the late 2030s.

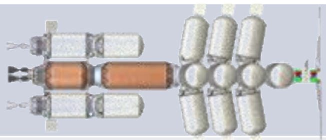
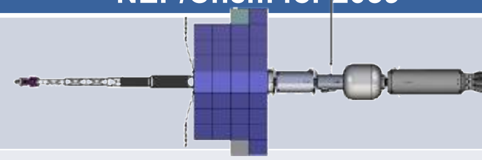
	NTP for 2039	NEP/Chem for 2039
Vehicles in Low Earth Aggregation Orbit **(Habitat added later)**		
Primary Technologies	➢ **Nuclear Thermal Rocket (NTR)**: 900s I_{sp} / 25klb$_f$ ➢ **Zero Boil-off LH_2 storage & transfer** ➢ High capacity 90K & 20K cryocoolers ➢ **Cryogenic Feed System**: ➢ Low-leak 8" and 3" cryogenic valves ➢ Interstage 7" cryo couplers	➢ **Nuclear Power**: 1.9 MW_e HEU or HALEU Reactor, SCO_2 Brayton Convertors, Deployable modular radiators ➢ **Electric Propulsion**:100 kW_e Xenon Hall thrusters @ 2600s, 13t throughput ➢ Large High pressure Xe storage Tanks ➢ **Zero Boil-off LOX/LCH_4 storage & transfer**: high capacity 100K cryocooler ➢ **Chemical Propulsion**: LOX/LCH_4 derived from Mars lander ascent engine
2039 Crew Mission	➢ 690 round trip, 30 Sols on surface	➢ 760 round trip, 30 Sols on surface
Number of reactors & engines	➢ **Four 25 klb$_f$ NTR**: 4 hour life, 2700K, 530 MW_{th}, HALEU reactor/engines	➢ **One reactor**: 2 year life, 1200K, 10 MWth/1.9 MW_e HEU or LEU ➢ **Twenty Xe Hall thrusters**: 100 kW_e ➢ **Three 25 klb$_f$ LOX/LCH_4 engines**
Unique Elements	➢ **NTR Core Stage** (1) *SLS* ➢ **Hydrogen Inline Stage** (1) *SLS* ➢ **NTR Stub Core Stage** (2) *H-CLV* ➢ **Multi-Dock Truss Element** (2) *H-CLV* ➢ **Power and Propulsion Module** (1) *H-CLV* ➢ **Hydrogen Drop Stage** (18) *SH-CLV* ➢ *Additional launch for DSH transfer stage (1) H-CLV*	➢ **Nuclear Electric Propulsion Module** (1) *SLS* ➢ **Chemical Propulsion Module** (1) *SLS* ➢ **Xenon Interstage Modules** (3) *SH-CLV* ➢ **LOX-LCH_4 propellant & transfer capability** (2) *SH-CLV*
Required Launch Fleet	➢ 2 SLS, 18 SH-CLV, 5 H-CLV ➢ ***934 MT in 1200 x 7000 km Aggregation Orbit*** ➢ ***~850 MT of LH_2 propellant***	➢ 2 SLS, 5 SH-CLV (alternately, H-CLV could be employed) ➢ ***678 MT in 500 km Circular Aggregation Orbit*** ➢ ***~256 MT Xe EP propellant, ~ 252 MT LOX-LCH_4 propellant***

Figure 4.3-1. Top Level Summary of NTP and NEP/Chem Approaches to Mars 2039.

MTAS performed a comparative analysis of these two reference cases between alternative propulsion technologies, considering distinguishing characteristics, development, operations, programmatics, and performance metrics. Details on these assessments are summarized in the following subsections.

4.3.2 Development

4.3.2.1 Technology-NTP:

Nuclear Thermal Rocket (NTR) engines had achieved a high degree of maturity from significant past investment in both the U.S. and Russia. From 1955-1973 Project Rover focused on NTR technology, while in 1961 the companion Nuclear Engine for Rocket Vehicle Applications (NERVA) contract (awarded to the Aerojet-Westinghouse team) focused on the development of a flight engine. The combined Rover/NERVA efforts developed several engine designs using HEU in UO_2, UC graphite composite (UC-ZrC)C, and carbide (U, Zr)C fuel forms, in channeled prismatic fuel elements, to high technical maturity; with a thorough test program solving many of the technical issues. The NERVA cores operated in the epithermal neutron spectrum, with graphite as a moderator. Project Rover tested smaller cores (25 klb$_f$ vs 75 klb$_f$) using a Zirconium Hydride (ZrH) moderator in the cooled structural tie tubes to achieve criticality. By the end of Rover/NERVA, the XE' engine was a successful test of a flight configuration and had been certified by the Space Nuclear Propulsion Office to meet the requirements for a human mission to Mars as defined at the time. The XE' achieved greater than 2400K average propellant exhaust temperature, sufficient for about 840 sec I_{sp}, and demonstrated successful start-up and shut-down transients. However, NASA's Mars mission plans never materialized, and in 1973 Project Rover and the NERVA contract were cancelled. From 1987 to 1994, the U.S. Air Force was developing an ultra-high performance NTR concept under the Space Nuclear Thermal Propulsion (SNTP) Program. The SNTP NTR design was to utilize HEU UC_2 fuel particles in a cooled block moderated core to achieve a very high thrust-to-weight ratio (T/W) of ~30. Comparatively, in the 1990s, Russia developed HEU (U,Zr,Ta)C fuel in a bundled twisted ribbon form to be operated in a cooled moderator block. These later programs achieved some design, modeling, and component testing, but did not complete reactors for testing. Much of the expertise developed during these programs has retired and is at risk of being lost.

For Nuclear Thermal Propulsion, a Nuclear Thermal Rocket engine must be integrated to a stage, which provides propellant to the system, along with spacecraft systems for guidance, navigation, and control. Briefly during Rover/NERVA, from 1961-1963, the Reactor In-Flight Test (RIFT) project focused on developing the additional technologies for systems necessary for the operation of an NTP stage. This project was cancelled prior to much development in these areas.

For NTP the main factor of performance is specific impulse, which is primarily determined by the achievable fuel temperature, which drives the temperature of the propellant. Low molecular weight propellant is key. A second factor of performance is thrust-to-weight ratio of the propulsion system – which affects vehicle acceleration and achievable single mission ΔV. Additionally, reactor full-power lifetime also affects the achievable ΔV over the mission life of the system. For the NTP reference case, the required I_{sp} is 900 sec, while the assessed thrust-to-weight ratio for the engines is 3.6, and the assumed full-thrust life of the reactor is 4 hours. The NTP technology plan is to meet these performance numbers, while performance significantly beyond this has yet to be demonstrated.

NASA is currently investing in Nuclear Thermal Propulsion system component technologies. Nuclear Thermal Rocket (NTR) engine subsystems, such as: engine gimbals; regeneratively cooled exhaust nozzles; and cryogenic hydrogen turbopumps, valves, and flexible fluid lines, benefit from strong synergies with the SSME and ongoing liquid chemical combustion engine development. Chemical rocket components will need to be developed for operation in a high radiation field for application to an NTR engine. NASA has near-term plans for characterization of novel fuel concepts at DOE Idaho National Lab (INL). NTP is also receiving some initial investment from DOD/DARPA due to the potential for high thrust acceleration and rapid response capability, and this provides agility in the Earth and Moon gravitational environments. Some of these technologies could be adapted to NASA use for Mars transportation. Both current NTP efforts are focused on HALEU reactor designs, which are a departure from the HEU design heritage established in the 1960s – 1990s. Progress-to-date represents a small fraction of development needed for flight; thus, significant development risk remains.

4.3.2.2 Technology-NEP:

NEP was demonstrated on-orbit in 1965 when the SNAP-10A reactor powered an early cesium ion thruster, though there was no follow-on mission application. SNAP-10A used U-Zr-H fuel and NaK coolant with thermoelectric power conversion to produce ~500 W_e. Russia has flown over 30 nuclear reactors in space, the most recent design being the Topaz I reactor, which first flew in 1987 with an experimental Hall thruster. They also developed the Topaz II reactor, which was tested and exhibited but never flown. The Topaz reactors use HEU UO_2 fuel with ZrH moderation, and in-core thermionic power conversion. They were short-lived reactors. The strongest U.S. design heritage for fission power in space was developed from the 1983-1994 SP-100 program, which was a DOD/DOE/NASA tri-agency program developing a 100 kW_e space nuclear reactor power system. NEP was one of several NASA mission applications envisioned. SP-100 developed HEU UN pin fuel and liquid lithium cooling technologies with strong feed forward to current NEP requirements. SP-100 achieved significant design development and component testing but did not complete a reactor system for testing.

More recently, the 2002-2005 Project Prometheus advanced NEP development for planetary science missions. Project Prometheus was developing 100-200 kW_e NEP for the Jupiter Icy Moons Orbiter (JIMO) mission in 2002-

2005 for flight in 2016. Most of the early JIMO government studies considered SP-100 derivative concepts with HEU UN fuel pins, lithium cooling, and HeXe Brayton conversion. At the conclusion of the project after the Naval Reactor Prime Contracting Team conducted a reactor down-select study, the Prometheus Baseline-1 design assumed a gas-cooled HEU UO_2 pin fueled core and direct HeXe Brayton cycle power conversion system. JIMO was cancelled before any significant component testing was accomplished. There have also been several relevant efforts to study and/or develop NEP technologies including the 1990s Space Exploration Initiative, the 2014 Mars Design Reference Architecture 5.0 (Addendum 2), and the Constellation-era Fission Surface Power project, which developed and tested a 12 kW_e Stirling TDU test in GRC VF6.

In 2018, the STMD Kilopower project successfully demonstrated the inherent control of an HEU U-Mo solid fuel core using negative temperature reactivity feedback. The Kilopower design transferred heat to Stirling convertors using heat pipes in contact with the core perimeter and was designed to produce ~1 kW_e. The ground test demonstrated load-following and stable thermal control over a number of power system operations and contingencies. NASA has explored derivatives of the Kilopower concept for a 10 kW_e surface power reactor, and combined with low-power electric thrusters, for NEP to support outer planet robotic science missions.

For Nuclear Electric Propulsion, a space nuclear reactor is combined with thermal-to-electric power conversion, thermal heat rejection, and power management & distribution to deliver regulated electric power to the electric propulsion subsystem. The EP subsystem includes power processing equipment to convert and distribute the input power among the EP loads and a propellant feed system for the storage, management, and supply of propellant. Thus, NEP requires complex integration of all these distinct elements. Past development of the subsystem technologies for NEP has been at a smaller scale than required for the 1.9 MW_e MTAS NEP reference case. For instance, NASA GRC developed a 25 kW_e Brayton converter with HeXe working fluid, heat exchangers, and heat rejection in the 1980s for the Space Station Freedom solar dynamic power module and demonstrated that technology during the 1990s 2 kW_e Solar Dynamic Ground Test Demonstration. Also, electric propulsion, both Hall-Effect Thrusters and Gridded-Ion Thrusters, have been used in space since the 1990s at the ~1 to ~10 kW_e scale. During Project Prometheus, the 2 kW_e Brayton unit from SD GTD was connected to a 2 kWe NSTAR Ion thruster to demonstrate power delivery and stable electric control under thruster transient operations.

A major system challenge in developing an NEP power system is the size, mass, packaging, and deployment of large space radiators to reject the waste heat generated by a thermal-to-electric power conversion. Important advancement of large, high-temperature composite radiator technology occurred during the Fission Surface Power project that supported NASA's Constellation Program after Project Prometheus. Also as important is the radiator system developed and deployed on the International Space Station. Although the ISS External Active Thermal Control System is operated at a lower heat rejection temperature and uses different materials and fluids than required for NEP, the total size (approximately 1200 m^2) is comparable to the size required for the MTAS NEP reference design – demonstrating that deploying and operating radiators of this size is feasible. Important engineering remains in the development of flexible fluid lines and radiator coolant manifolds that are suited to the NEP heat rejection temperatures.

The main factor of performance for Nuclear Electric Propulsion is the mass-to-power ratio of the reactor power system and electric propulsion. This is often measured in specific mass (kg/kW_e) or specific power (Watts/kg) for smaller systems. This performance factor translates in effect to the thrust acceleration potential of the NEP system, where lower specific mass results in higher acceleration potential, and thereby faster missions or larger payloads. For the NEP/CCP reference case the specific mass of the reactor power system alone was about 12 kg/kW_e, and the EP subsystem (thrusters, DDUs, and XFCs) adds about 2.5 kg/kWe. The overall mass of the dry propulsion system (including the power system, structure, stage subsystems, thrusters, and propellant tankage mass) resulted in a specific mass of ~40 kg/kW_e.

NASA currently does not have a coordinated development activity for an NEP system but is independently investing in several relevant subsystem technologies. To begin with, the Artemis Gateway Power and Propulsion

Element project is planning a 50 kW_e Solar Electric Propulsion system with multiple 12 and 6 kWe Hall thrusters, and this technology can be leveraged for larger scale devices to suit Nuclear Electric Propulsion. A 50 kW_e class Hall thruster, the NASA 457M, was fabricated and tested up to 100 kW_e at GRC as a building block for 300 kWe SEP missions. Under the NextSTEP High Power EP project, HEOMD awarded contracts to EP developers to design 100 kW_e thrusters and test them for 100 continuous hours. The necessary NEP power management and distribution technology could significantly benefit from current ARMD PMAD development for MW_e electric aircraft.

Importantly, there is current synergistic investment in space fission power relevant technologies for terrestrial applications, including microreactors and power conversion at the scale required for Mars NEP. Relevant terrestrial microreactor technology is being funded by DOE under the Advanced Reactor Demonstration Program and by DOD under the Pele project. DOE, DOD, NASA, and industry have also joined forces to establish production capabilities for TRISO fuel that could be used for NEP reactors. Further, DOE has been investing in Brayton power conversion in the MW_e class using supercritical CO_2 as a working fluid under the Solar Energy Technologies Office. Further study and development are required to adapt this technology for space. The NEP concept's compact nuclear power generation capability has strong interest from terrestrial energy and can strongly benefit from synergistic component development.

Finally, in consideration of the companion conventional chemical propulsion technologies required in the hybrid NEP/CCP approach, many programs, both government and commercial, are developing LOX/LCH_4 propulsion systems for both launch vehicles and in-space propulsion. Leveraging opportunities exist for the development of engines and stage technologies, as well as the important cryogenic fluid management of LOX and liquid methane.

4.3.2.3 Reactor – NTP:

NTP reactors operate on the extremes of material capabilities and engineering design. This is necessary to achieve the highest reactor temperature to maximally add energy to, and accelerate propellant to high exhaust velocity, which translates to high specific impulse (propellant efficiency). Past NTR reactors designed for high I_{sp} operate close to the melting point of the fuel (within ~100-200 K). Since hydrogen is not only the propellant, but the reactor coolant, any significant imbalance in H_2 flow could lead to hot spots and melted fuel. Furthermore, NTP reactors have been the highest power density nuclear reactor cores ever designed in order to provide thrust at minimal mass, which translates to high thrust-to-weight ratio. However, high power density translates to quicker reactor response to transients, and higher risk to melting the core.

The SNTP NTR design used HEU UC_2 fuel particles, with high surface-to-volume for high heat transfer, in a moderated core to minimize size and mass, to achieve a very high thrust-to-weight of ~30. However, most past Nuclear Thermal Rocket (NTR) engine designs achieved thrust-to-weight of ~3; over 10-times lower thrust-to-weight than conventional chemical rocket engines, which negates some of the advantages of higher I_{sp}.

Additionally, the use of hydrogen as a propellant introduces engineering complexities to the design and operation of a fission reactor and requires rapid and precise reactor control. This is due to the neutronic interaction of hydrogen atoms in the balance of a critical nuclear reaction (hydrogen slows neutrons when they collide, thermalizing them and increasing their fission potential with uranium-235 – more hydrogen leads to more heat, but also more cooling). This interaction requires precise control to balance hydrogen flow and criticality control during reactor startup, when the core is cold and dense, hydrogen flow is introduced; and during transition to full thrust, as hot hydrogen becomes less dense, while the hot core requires more hydrogen cooling to keep from melting. Similar control complexity occurs during the shut-down transient when hydrogen flow is still required for cooling the reactor while not sustaining the nuclear reaction.

Finally, high temperature hydrogen corrosion of fuel elements and coatings in the core is a significant consideration for materials development and corrosion rate increases with fuel temperature. This issue typically limits engine lifetime to a few hours of thrust operation. Because of this, higher I_{sp} NTR designs are driven to shorter-life, and longer-lived NTR designs are driven to lower I_{sp}. Longer operation is also possible with fuel choice, and with hydrogen resistant material coatings, but the technology for full-thrust life greater than a few hours at high I_{sp} has yet to be demonstrated. Thus, I_{sp}, thrust-to-weight ratio, and full thrust lifetime are important considerations driving NTR reactor technology to extremes.

The consideration of high-assay, low-enriched uranium (HALEU - uranium enriched to 5-20% ^{235}U, and typically 19.75% for space applications) for space nuclear power and propulsion is very recent, resulting from the policy push to use such fuel where possible to promote nonproliferation of weapons-grade uranium. Uranium-235 is the only naturally occurring isotope in any practical abundance that is fissile with thermal neutrons. However, the neutrons generated by each ^{235}U fission are born fast and must be slowed down (thermalized) to efficiently sustain the chain reaction with HALEU. If the neutrons are not thermalized, a HALEU core (even at 19.75% enrichment) must be large and heavy to contain enough ^{235}U for a critical mass. Typical terrestrial nuclear reactors used for power are thermal neutron spectrum cores using LEU (some like the Canadian CANDU reactor even use natural uranium at 0.7% ^{235}U), but size and mass are not key design limitations. For space applications, size, mass, complexity, lifetime, and reliability are all major design considerations.

The reference NASA NTP fuel development plan focuses on achieving the necessary performance from HALEU. Current concepts are evolved from heritage 1960s HEU NERVA and 1990s HEU SNTP fuel designs. The use of HALEU requires a new fuel, and a thermal spectrum core design to keep the fuel mass and core size comparable to HEU designs. A thermal spectrum core requires moderator material (usually a high temperature hydrogen compound, or less effectively, graphite to slow fast fission neutrons to the thermal spectrum) and additionally requires cooling to maintain the moderator integrity over the operational life of the core. Hydrogen compound moderator materials, such as ZrH and Yttrium Hydride (YH), must be maintained at temperatures much cooler (~1000 K or less) than the high temperature fuel (~2700 K) for the hours of thrust operation. This introduces design complexity & challenges with integration and cooling of moderator materials. Furthermore, high power density requires this cooling to be precise and responsive. These materials are also low TRL and require significant material development. The NERVA cores used graphite in the fuel elements to moderate the neutrons to an epithermal spectrum, and Project Rover experimented with ZrH material in cooled structural tie-tubes. The MTAS reference NTR design, assuming an HALEU core, requires full moderation to a thermal neutron spectrum, and thus uses a block moderator design with coolant channels in high temperature areas similar to the 1990s Russian design. ZrH moderator is under development at Los Alamos National Lab and remains low TRL.

The heritage knowledge base for space nuclear power and propulsion, both NTP and NEP, is fast-spectrum reactors using HEU (uranium enriched to greater than 20% ^{235}U and typically around 93%), which has been developed since the late 1950s. This includes for NTP the fuel and reactor development under Rover/NERVA and SNTP, and for NEP includes significant development of UN fuel under SP-100. While much of this legacy technology will require efforts to redevelop, the successes of the past represent feasibility proofs, while the promises of new technology have yet to be proven.

HEU reactors typically operate in the fast neutron spectrum, rather than the thermal neutron spectrum, to maintain simplicity and reliability, and facilitate long life by avoiding the use of moderator materials. Fast spectrum neutrons can sustain a fission reaction, albeit with less efficiency than thermal neutrons, and this inefficiency can be afforded if ^{235}U is abundant, as in HEU. A thermal spectrum, HEU reactor has the potential to be the lowest mass solution for space nuclear power and propulsion, but it comes at the expense of design complexity and likely lifetime. The engineering of lightweight, compact, long-life, reliable reactors utilizing moderators, especially hydrogenated moderator materials, will require significantly more schedule and cost for technology development. Until the challenges inherent in this engineering and development are solved, the implementation of moderated, HALEU reactors represents significant risk for a 2030s nuclear propulsion system.

4.3.2.4 Reactor-NEP:

NEP reactors operate at lower temperatures than NTP reactors, but for longer duration. Overall, the NEP reactor is projected to be less difficult to develop given the lower operating temperatures and steady-state operation, which is more analogous to terrestrial reactors. Higher temperature reactors are desirable to increase the efficiency of power conversion systems, as well as increase the heat rejection temperature to minimize radiator size. The operating temperature of space power reactors are typically limited by the material limits of the fuel, pressure vessel, and power conversion hot-side (e.g., the turbine for Brayton or Rankine systems). These limits are projected to be ~1200K for the use of super-alloy power conversion, ~1500K for refractory alloys, and ~1800K for ceramic materials.

NEP cores typically are designed to operate behind a conic frustrum “shadow” shield to mitigate gamma and neutron emissions from the operating reactor and protect spacecraft systems and crew. Shields are dense and heavy, typically larger and heavier than the reactor itself, and are sized by the shape of the reactor. Because of this, compact cores are desirable, which has implications on reactor heat transport (which drives the volume of coolant channels in the core), as well as the choice between HEU and HALEU (HALEU requires more volume for both fuel and moderators). The use of HEU in a fast neutron spectrum core design results in the simplest, most compact, lightest space power reactor, especially for smaller systems up to a few MW_e.

MTAS found that designing for much lower fuel density HALEU system requires employing hydride moderators for a thermal spectrum core. This holds similar design and technology development challenges with HALEU NTP and introduces design complexity and development risk. A moderated space power reactor core must maintain hydrogen compound moderator materials, such as ZrH or YH, at less than ~1000K for years of operation while the fuel in the core is typically hotter (~1200K for the NEP reference design). Time at high temperature determines how long a hydrogen compound moderator material will maintain effectiveness, as hydrogen dissociates and escapes over time (an effect that increases with temperature); thus, the use of hydrogen compound moderators will limit reactor lifetime. Reactor temperature, core sizing factors, and full power lifetime are important considerations driving NEP reactor technology and favoring the use of HEU. The only reasons to consider HALEU for space power applications are to alleviate policy concerns with the use of HEU and to address issues with security and licensing for the handling of HEU. However, robust public-private partnerships for the development of HEU space nuclear power and propulsion are possible, leveraging existing facilities, organizations, and entities licensed for handling HEU with adequate security. Furthermore, new facilities, organizations, and entities will have to be licensed to handle HALEU (as discussed in 4.3.2.3) and could be developed to the higher standards of HEU if the business case warrants it. Until technology challenges with HALEU fuel, and moderated space reactor cores are solved, such a space fission power approach remains a high cost, schedule, and technical risk, while the performance required for human Mars propulsion may drive a return to HEU solutions.

The reference NEP plan builds on the SP-100 heritage for liquid metal reactor cooling, and on anticipated development of HALEU nuclear fuels and moderators under the NASA Fission Surface Power project. The reference NEP approach departs from the SP-100 fuel heritage but considers the leveraging of advanced terrestrial nuclear development of HALEU fuels and moderators. NEP has higher heritage with HEU, and alternately could employ HEU UN fuel from SP-100 to reduce development risk, with a performance advantage due to a smaller, lighter core and shield. In all MTAS NEP cases, a 1200K reactor operating temperature was assumed, based on the material limitations of super-alloy metals for the Brayton converter, but still requiring refractory metals in the reactor and primary heat transport.

4.3.2.5 Propellant-NTP:

Nuclear Thermal Rockets may employ a few potential propellants, but the main benefit of NTP comes from high specific impulse. High I_{sp} is derived from accelerating propellant molecules to high velocity, and the kinetic energy equation, $E_k={}^1/_2mv^2$, tells us that lower mass molecules achieve higher velocity for a given energy extracted from the reactor. H_2 is the lowest mass molecule, and results in the I_{sp} potential of 900 sec or higher. The next lowest mass molecules that could be considered as a practical propellants, methane or ammonia, would have similar or slightly higher I_{sp} to LOX-LH_2 CCP. With the lower thrust-to-weight of NTP there would be no propulsive advantage.

A critical technology for NTP performance is the storage, transfer, management, and containment of hydrogen as a propellant. To minimize volume of propellants for launch, to minimize mass for containment during in-space operation, and to minimize leakage over the course of a mission, hydrogen must be maintained in liquid form at 20K cryogenic temperatures. Storing large quantities of cryogenic propellants with negligible losses has always been a challenge. Storing liquid hydrogen for years required for a human Mars mission compounds the challenge. Molecular or atomic hydrogen can leak through very small valve gaps, and even diffuse between the atoms in the metal lattice meant to contain the hydrogen. Any loss of hydrogen over the course of a mission is, in effect, a loss of propellant efficiency. Zero boil-off (ZBO) of liquid hydrogen, and very low hydrogen leakage during the 5+ year campaign is required. Achieving this will require the development of no/low leakage valves, disconnects, advanced insulation, and high-capacity, high-efficiency 20K cryocoolers. These technologies represent significant development risk. Though CFM has been used throughout the history of human spaceflight, the requirements have never been as stringent, and the criticality has never been as high as on an NTP mission of this duration.

NASA is currently investing in the critical technologies for cryogenic fluid management in order to solve the long-term hydrogen, oxygen, methane, and other fluid storage challenges for propellants, ISRU products, and other exploration systems. Cryogenic hydrogen requires 20K cryocoolers, and advanced thermal insulation systems, which is a significantly greater challenge than higher temperature cryogens (the only more difficult cryogen is liquid helium). This technology is being pursued for multiple applications and has benefits for the terrestrial utilization of hydrogen.

The use of liquid hydrogen as a propellant has significant implications on vehicle design, launch of propellant, and orbital assembly due to its low volumetric density, and requirements for cryogenic fluid management from the moment liquid hydrogen is tanked on the launch pad. The low volumetric density (~70 kg/m^3 compared to water at 1,000 kg/m^3) typically limits the amount of hydrogen propellant that can be launched on any rocket by payload volume, rather than launch mass capability. Thus, NTP systems are driven to utilize large payload shrouds, and require significantly more launches to provide the necessary propellant for a human Mars mission. Each propellant launch element requires a large propellant tank maximized to the payload shroud that includes either cryogenic fluid management, with its associated power system, or massive tank insulation to mitigate hydrogen boiloff until the propellant is integrated with orbital assets. Each propellant element in the MTAS NTP reference case includes a fully capable transfer vehicle to deliver the propellant element to the aggregation orbit where the NTP Mars Transfer Vehicle is being assembled. This transfer vehicle also provides power for active CFM. Both the number of launches and the number of vehicle elements adds considerably to the development and operational cost of an NTP human Mars mission.

4.3.2.6 Propellant-NEP:

Electric propulsion using Hall Effect Thrusters (HET) or Gridded Ion Thrusters typically favor propellants that have a low first ionization energy, to minimize the energy needed to create an ion for acceleration, and a high atomic weight, to maximize the momentum generated by the acceleration (with low thrust acceleration propulsion, thrust is more important than specific impulse for fast missions). Noble gases have these advantages, as do cesium and iodine. Xenon, krypton, and argon have been considered for Mars class electric propulsion, and among these xenon is the best for relatively low I_{sp} needed for fast missions. Xenon has additional advantages since it can be stored under pressure (~1,000 to 2,000 psi or ~7,000 to 14,000 kPa) in a supercritical state without cryogenic fluid management. Under these conditions, supercritical Xe is about as dense as a brick (comparatively, cryogenic hydrogen is about a dense as Styrofoam, cryogenic methane is about as dense as oak wood, and cryogenic oxygen is slightly denser than water). Pressure vessels for these conditions are typically flown in space for He pressurants, but the size needed for Xe propellant will require a much larger scale tank.

The main challenge with Xe is acquiring the quantities needed to propel each human Mars mission. Xe is overall abundant, but it is a small fraction of the constituents of Earth's atmosphere. Air is liquified in many locations throughout the earth, where purified oxygen is particularly needed for steelmaking. Air is liquified in a series of increasingly cold condensation steps, and the noble gasses argon, krypton, and xenon are the last to condense. Many liquification plants do not have the equipment to separate these three gasses at the end of the condensation line. A human Mars mission will require several years of the world's supply of xenon at current production rates. Solutions to this challenge include stockpiling xenon over several years and increasing the infrastructure to purify xenon at existing air liquification plants throughout the world. This represents an opportunity for all nations to contribute to space exploration.

The hybrid NEP approach uses liquid oxygen (LOX) and methane (LCH_4) as propellants for the companion chemical combustion propulsion. LOX and LCH_4 both have cryogenic storage temperatures near ~90 to 100K, which is comparatively easier for cryogenic fluid management relative to hydrogen. NASA is also investing in the critical technologies for cryogenic fluid management of liquid oxygen and methane to solve the long-term propellant storage challenges. These include no/low leakage valves, disconnects, advanced insulation, and high-capacity, high-efficiency 90K cryocoolers.

4.3.2.7 Testing and Development Facilities-NTP:

Because of the complex operation of a Nuclear Thermal Rocket engine, where core reactivity is tightly coupled with hydrogen propellant flow, NTP development requires a full scale, integrated ground test to prove the system and qualify it for spaceflight. This includes start-up and shut-down transients, and because of the potential for reactivity and material changes over the thrust life of the reactor, should include the full mission profile of start-ups, shutdowns, and propellant throughput. Several integrated full system nuclear qualification tests, including post-irradiation examination of suspect fuel elements, would be necessary to provide confidence that the final design would achieve 900 sec I_{sp}. Subsequently, a full system demonstration would be necessary to certify the overall system for human spaceflight.

During Rover/NERVA such testing was performed in open air at the Nevada Test Site (now the Nevada National Security Site), where the full-scale engine exhaust was neither captured nor filtered. While the fission product inventory for an NTR test is very small due to the short run times of the engines, such testing cannot be performed today without capturing and/or filtering all rocket exhaust to eliminate any risk of releasing radioactive fission products into the environment. This requires a new test facility. The Rover/NERVA program successfully tested the "Nuclear Furnace" NF-1, which demonstrated one concept for an effluent treatment exhaust system with a 44 MW_t reactor. Such a facility will be costly, and require significant time for construction, including the necessary approvals under the National Environmental Policy Act. The development of this facility adds significant cost and schedule risk for NTP. Since the end of Project Rover/NERVA, the design and development of this facility has been, and remains, a major hurdle for the development of NTP.

An alternative approach to avoid the development of this facility is an in-space development test. Such an approach is success oriented, and still has major risk. The challenge is troubleshooting a test failure, anomaly, or nonconformance without physical post-test inspection and the opportunity for disassembly. Without adequate instrumentation, a test failure, anomaly, or nonconformance may only be discovered through physical post-test inspection. As development of NTP moves forward, NASA will need to work closely with DOE to determine what fraction of NTP testing can adequately be accomplished in space.

4.3.2.8 Testing and Development Facilities-NEP:

The NEP system development allows for separate subsystem ground testing of modular elements and simulated interfaces, which could be accomplished in separate facilities, easing both facility requirements and development and testing processes. Several nuclear facilities are in development for terrestrial small modular reactors that could be leveraged or adapted for testing space reactors supporting NEP. The MTAS study did consider the characteristics and need for new test facilities for NEP. The development of NEP will require some facility development, notably large, high-pumping capacity vacuum chambers for testing multiple, large electric thrusters.

4.3.3 Operational Distinctions

4.3.3.1 Operational Distinctions: Thrust

The amount of thrust acceleration any propulsion system provides in relation to the acceleration of the local gravity field determines the efficiency of providing change in velocity, as well as the time required. High thrust acceleration allows for quick changes in orbital velocity, in turn resulting in agility and quick response, as well as rapid departures from, and arrivals into, planetary orbits from interplanetary space. Lower thrust acceleration results in slow orbital changes, which introduces inefficiencies from thrusting against gravity, termed “gravity losses.” However, with time, high propellant efficiency (high I_{sp}) propulsion overcomes gravity losses and can result in an overall more efficient performance. The best of all propulsion has both high thrust acceleration and high I_{sp}.

Thrust-NTP: Nuclear Thermal Propulsion represents the highest I_{sp}, high thrust propulsion technology currently achievable. By comparison, NTP offers ~900 seconds while Chemical Combustion Propulsion typically offers ~330 – 460 seconds. The effect of propellant efficiency is to significantly reduce the propellant necessary to perform the large changes in vehicle velocity necessary to perform the transit from Earth orbit to Mars orbit and back. The faster the Mars mission, and the more distant the Earth-Mars planetary alignment during the year of opportunity, the larger the required changes in velocity.

Thrust-NEP: The advantage of electric propulsion is that much higher propellant exhaust velocity can be achieved through either electrostatic, electromagnetic, or electrothermal acceleration of ionized propellant. Specific impulse of ~1,000 – 10,000 seconds is achievable and is a matter of how the thruster is designed, the voltages applied to acceleration, and the propellant choice.

One of the challenges for the use of any electric propulsion technology is the low thrust acceleration, limited by power, which results in significant time to propel into and out of planetary gravity wells (regions where gravitational forces are much higher than deep space). Low thrust applied in a strong gravity field results in long, spiral shaped trajectories as thrust acceleration is applied over time. However, low thrust acceleration applied during interplanetary transit can be very efficient as the gravity field is much weaker, and there is significant time to apply acceleration. This results in potentially very fast interplanetary transit, but significant additional time to transit from low Earth orbits and into low orbits around planetary destinations. NEP mission times can be reduced with higher thrust acceleration, achieved with either advanced, lower specific mass technology, or higher power systems with lower specific mass by economy of scale.

Hybrid mission approaches, utilizing both high thrust and low thrust where they are most advantageous, can improve overall mission performance if the mass of the combined propulsion systems is not prohibitive. Typically, high thrust is used in gravity well maneuvers, such as at Earth and Mars, to reduce or eliminate the long spiral trajectories, while low thrust is used for acceleration in interplanetary space to reduce the size of the gravity well maneuvers. The optimization of the propulsive maneuvers depends on the I_{sp} of each propulsion type and the demands of the mission objective. The study found that the augmentation of NEP with high thrust LOX-LCH_4 chemical propulsion significantly reduces the size of the electric propulsion system required, and with it the technologies required for the objective 2-year Mars mission.

An additional benefit of high specific impulse, combined with the efficient application of low thrust acceleration in interplanetary space, is the mitigation of the energetic challenges that result from more distant planetary alignments between Earth and Mars during more difficult mission opportunities (such as 2039 and 2042 when Mars is more distant from Earth at closest approach due to its elliptic orbit). In effect, higher I_{sp} electric propulsion reduces the variance in mission performance between best and worst Earth-Mars planetary alignments. This effect was observed in the MTAS analyses between the mission performance for 2035 and 2039.

4.3.3.2 Operational Distinctions: Extensibility

Both NTP and NEP require significant technology development. The development roadmap for either should consider appropriate and valuable steppingstones that may involve precursor systems that benefit other missions or applications. Furthermore, both NTP and NEP can evolve to advanced capabilities that support advanced human missions to Mars and human and robotic exploration beyond. The extensibility of either propulsion to precursor and follow-on applications is a distinction on the value of the technology investment. Future conjunction class missions, that might follow initial opposition class missions, may benefit from these technologies that could provide either: lower launch mass, faster transits, higher payload delivery, or some combination of the three. While extensibility was considered under MTAS, no specific mission studies were conducted to evaluate the merits of the various technology options for alternative applications. The following conclusions are drawn from past studies with appropriate references.

Extensibility-NTP: Nuclear Thermal Propulsion has been studied over the years for several applications where both higher I_{sp} and high thrust acceleration is required. Project Rover initially conceived of an NTR as an upper stage engine for ICBMs during the Cold War, and the Air Force SNTP program considered the same. NASA's NERVA effort was initially focused on an NTP upper stage for the Saturn or Nova rocket to support the Apollo human missions to the Moon. Only after NASA settled on the Apollo Lunar Orbit Rendezvous architecture and the sufficiency of the Saturn V rocket did NERVA focus on human Mars mission application. The upper-stage application of NTP is no longer considered necessary or appropriate.

DOD/DARPA is considering NTP due to its high thrust acceleration and rapid response capability, and this provides agility in the Earth and Moon gravitational environments. These systems could be adapted to NASA for more efficient Earth-Moon transportation supporting a robust, ongoing lunar exploration campaign.

Lower thrust-class NTP has been studied for smaller robotic science missions (Dudzinski, 1995; Ballard, 2019) where a higher ΔV Earth orbit departure is valued, but the benefits of such an application are highly dependent on the I_{sp} and stage dry mass (Gabrielli, 2015; Rawlins, 2022). Because of NTR lower T/W and the use of large tanks and CFM for hydrogen storage, an NTP stage is often not advantageous over Conventional Chemical Propulsion stages at smaller scales. The utility of small NTP for distant destinations could be enhanced by bimodal operation of the reactor for both thrust and power – whereby the NTR engine could supply power to maintain CFM for hydrogen during the long transit, to the spacecraft, and potentially even to electric propulsion. Such a bimodal system would enable NTP to provide ΔV at the destination of the mission as well. This is an advanced capability that was not considered for a mid-2030s human Mars mission development.

The extensibility of NTP to advanced applications, requiring higher ΔV, will be defined by the potential for advancements in specific impulse, stage dry mass (largely determined by NTR thrust-to-weight and hydrogen CFM system mass), NTR full-thrust lifetime, and the long-term storage of cryogenic hydrogen. The material limitations of solid core nuclear fuel will limit NTR I_{sp} to about 1050 seconds. More advanced concepts for liquid cores and extreme concepts for gas cores may reach 1,300 seconds and 3,000 seconds, respectively, but these systems are merely conceptual at the time of this writing.

Extensibility-NEP: The use of nuclear power for electric propulsion was conceived of in the 1950s and depicted as a propulsion system for a human Mars mission in the 1957 Walt Disney film "Mars and Beyond", where Wernher von Braun and Ernst Stuhlinger were consultants. NASA has considered NEP for the Jupiter Icy Moons Orbiter, and NEP has been studied for missions such as a Neptune-Triton orbiter, and an Interstellar Probe, where extreme ΔV is needed at distances from the Sun where high photovoltaic power is impractical (Jones, 1984; Yam, 2012; Noca, 2003; Nock, 2012; Satter, 2005; Oleson, 2004; Oleson 2019). The practical mission ΔV for NEP is limited by the mass-to-power ratio (specific mass), and specific impulse. The I_{sp} potential of electric propulsion (10,000+ sec) is enabling for even more advanced applications as the specific mass of space nuclear power is lowered by advances in technology.

The development of NEP technology for Mars has potential to both leverage other development and be leveraged to benefit other missions as a precursor application, such as subscale systems for robotic science. Mars NEP systems could evolve from systems already being developed for near-term NASA missions including the 10 kW_e Fission Surface Power Technology Demonstration Mission and the 50 kW_e Gateway Power and Propulsion Element. A potential intermediate development step between FSP and Mars NEP is to adapt the 10 kW_e FSP system for use with off-the-shelf EP thrusters to perform a robotic orbiter mission to Uranus, Neptune, or Pluto. A complete NEP concept based on a ~10kW_e evolution of the Kilopower reactor is also under consideration for development by the U.S. Space Force (Helios-N) and a partnership could possibly be leveraged by NASA for these missions. NEP technology has potential feed forward benefits to future NASA science and exploration applications at various scales, such as a heat source (e.g., ice world melt probes at ~50 kW_t) a large-scale surface power system (e.g., commercial ISRU propellant production at ~1 MW_e) as well as high ΔV, large payload propulsion (e.g., Europa subsurface mission at ~200 kW_e) (Oleson 2019; Oleson 2022; Oleson 2022). MTAS did not study these applications but identified them from past studies (Gowen, 2011).

4.3.3.3 Operational Distinctions: Reusability

The development of NTP or NEP to support a human Mars mission will require significant investment in time, effort, and funding, and may also require the development of unique facilities. Once developed, each propulsion system will be more valuable if it is reusable. However, the additional requirements for reusability and extended lifetime will themselves increase the investment necessary for the first system. This value proposition must be explored and fully assessed in consideration of a reusability requirement. MTAS Phase 1 considered but did not fully assess the implications of reusability. Such an important assessment is recommended forward work.

Reusability-NTP: There are several factors limiting NTP lifetime that will affect reusability for Mars missions. High temperature hydrogen corrosion of fuel elements and coatings in NTR cores is a significant consideration for materials development and fuel temperature (which drives I_{sp}) and sets a limit on engine operational lifetime to a few hours of thrust operation. Extending the lifetime of NTR cores, given a specific material type, requires operating at lower hydrogen temperatures. Thus, longer life, reusable NTRs are possible with reduced specific impulse[5]. An additional approach to enabling NTR reusability is to reduce the total engine operation time in a mission by increasing the number of engines on the vehicle at the expense of higher engine mass. Increasing from two engines to three would reduce burn times by a third, but increase the propellant required by the additional engine mass.

Use of hydrogen compounds as moderator materials also place limitations on lifetime based on hydrogen dissociation over time, which is accelerated at higher temperatures. Moderated NTR cores can address this issue by designing to maintain hydrogen compound moderator materials at lower operating temperatures to extend lifetime at the expense of performance.

Finally, the ConOps for the reference NTP case utilizes drop-staging of empty propellant tanks to minimize round-trip propellant mass (eighteen propellant tanks and two stub core stages are dropped for the 2039 mission). This significantly improves performance but requires the resupply of that hardware to reuse the core DST vehicle. The NTP reference case also requires 70% more propellant mass than NEP/CCP for the same mission. Thus, a reusable NTP system would require additional hardware production as well as greater propellant mass resupply to support subsequent missions.

[5] The current con-ops presented here does not reuse NTRs

Reusability-NEP: The Mars Nuclear Electric Propulsion system has significant potential for reusability, since space power reactors can be designed for long operating life. The factors for NEP reusability include reactor lifetime, power conversion wear and failure mechanisms, and electric propulsion thruster throughput capability. Space nuclear power reactors are typically limited by fuel burnup and issues with fission product build-up in the core over time. Such reactors can be designed for long life by adding U235 fuel to the core – resulting in larger, more massive cores and the corresponding shielding. This has some effect on performance, but at MW_e scales the effect is small. NEP reactors may also experience operational life challenges from lithium corrosion and refractory alloy embrittlement, which must be addressed by materials development. Helium gas generation over time must be captured and controlled in the system design. More significantly, the incorporation of hydride moderator materials will limit lifetime based on hydrogen dissociation over time. As with NTP, fast neutron-spectrum HEU reactors are inherently simpler with greater potential for long-life.

Brayton power conversion for space applications is engineered to eliminate sources of wear by employing gas or magnetic bearings. Other potential areas of concern include material creep in the high temperature turbomachinery and the hot-side plenums. Radiators and other fluid systems will also be subjected to micro-meteoroids and orbital debris during the mission and may develop leaks and holes from impacts. The fault tolerance of these systems will need to be addressed through both shielding and redundancy. Utilizing heat pipes with high-conductivity fins in the NEP radiators helps to reduce the vulnerable area and minimize the impacts of unit failures. The reliability of electric circuits and components operating at high voltage and high temperature in a radiation environment will also need to be addressed through design and testing, which will impact schedule and cost. Finally, most electric thrusters have a limitation on propellant throughput due to erosion of thruster components by interaction with the accelerated propellant. The thruster plume also may erode surfaces of the power and propulsion system, and mitigation of this concern may drive the vehicle configuration. A straightforward approach to address EP thruster life is to replace them between missions, and such requirements can be designed into the system. Alternatively, an NEP system can carry spare thruster sets, as the thruster masses are a comparatively small percentage of the overall propulsion system.

4.3.3.4 Operational Distinctions: Robustness

Robustness involves the propulsion systems agility to respond to contingencies and prevent loss of mission, and more importantly loss of crew. While robustness can be designed into a propulsion system, some elements of robustness are natural characteristics of each propulsion system and may present distinctions between them. The issue of mission robustness involves complex mission modelling and analysis, and was not assessed in MTAS, but is recommended for additional analysis in a future study. This assessment will be required to support actual mission design for a human mission to Mars. For this study, both NTP and NEP system designs and mission analyses accounted for engine-out and partial thrust and power contingencies.

Robustness-NTP: One benefit of increased propellant efficiency is that contingency propellant can be carried throughout the mission for less penalty, thereby affording mission robustness. The higher I_{sp} of NTP over CCP allows for this. Other mission operations also can provide for mission robustness, such as predeploying contingency propellant or replacement propulsion stages at Mars using the cargo mission capacity afforded by NTP. High thrust acceleration, such as with NTP, also provides for rapid response to contingencies.

Additionally, The multiple NTR engines within a Nuclear Thermal Propulsion system provides for engine-out capability. However, the resulting assessment of mission reliability with multiple reactor engines must consider all failure modes, including possible catastrophic reactor failure, which could affect multiple engines and even destroy the vehicle. The NTP reference case includes four NTR engines to provide the necessary thrust at Earth departure, when the vehicle is most massive, while reducing both burn time and gravity losses. Two of these engines are jettisoned on parallel stack stages after a deep space maneuver. The reference performance assumes one-engine out operation on the two-engine core stage returning from Mars.

Robustness-NEP: NEP offers even higher propellant efficiency than NTP, and therefore, better affords the mission lien of carrying contingency propellant. Responding to contingencies in interplanetary space with low thrust acceleration will require more time, but that time is available during interplanetary transit. However, while the time for low thrust acceleration response to contingencies in planetary orbit may be an issue, the hybrid NEP/CCP approach provides a high thrust capability for rapid response if necessary. The availability of both high I_{sp} low thrust capability and high thrust capability allows for increased mission flexibility in response to contingencies. The MTAS NEP/CCP reference case assumes the LOX-CH_4 stage is present and available until Mars departure. The MTAS NEP/CCP reference case also assumes contingency propellant is carried throughout the mission, though half of the contingency propellant is left at Mars to increase performance on the return leg (this ConOps should be reconsidered for robustness in the next phase of analysis).

The reference NEP system was designed with one reactor and four, independent Brayton power conversion strings, and can operate at partial power with fewer thrusters if one or more power conversion strings fail. Each Brayton string includes a dedicated heat rejection segment (radiator) and PMAD channel that supplies power to five Hall thrusters. Consideration was given to including spare power strings for the reference design, but that approach was rejected because the benefits were negated by the added complexity. Again, the resulting mission reliability with multiple strings must consider all failure modes, including possible catastrophic reactor failure, which could affect all strings or eliminate the prime power conversion and even destroy the vehicle.

MTAS performed a cursory analysis demonstrating that the mission could be completed on partial power thrust, albeit with a longer mission time. A more thorough analysis is required. Past studies (Barnett, 1991; Jansen, 2016; Casani, 2020) have documented analyses that show NEP has a robust partial power abort potential (Dudzinski, 1992) throughout the mission due to high I_{sp}. These studies also identified that NEP offers wide departure windows from Earth and Mars, adding to the mission robustness.

Finally, electric propulsion provides an additional benefit to the robustness of the overall human Mars campaign by reducing the performance variance across Mars transit opportunities. This effect was observed in the MTAS analyses between the mission performance for 2035 and 2042, where the same size power system, and same specific impulse EP system completes both opportunities with similar performance. This allows for more confidence in mission planning and mitigates schedule risk for development.

4.3.4 Campaign

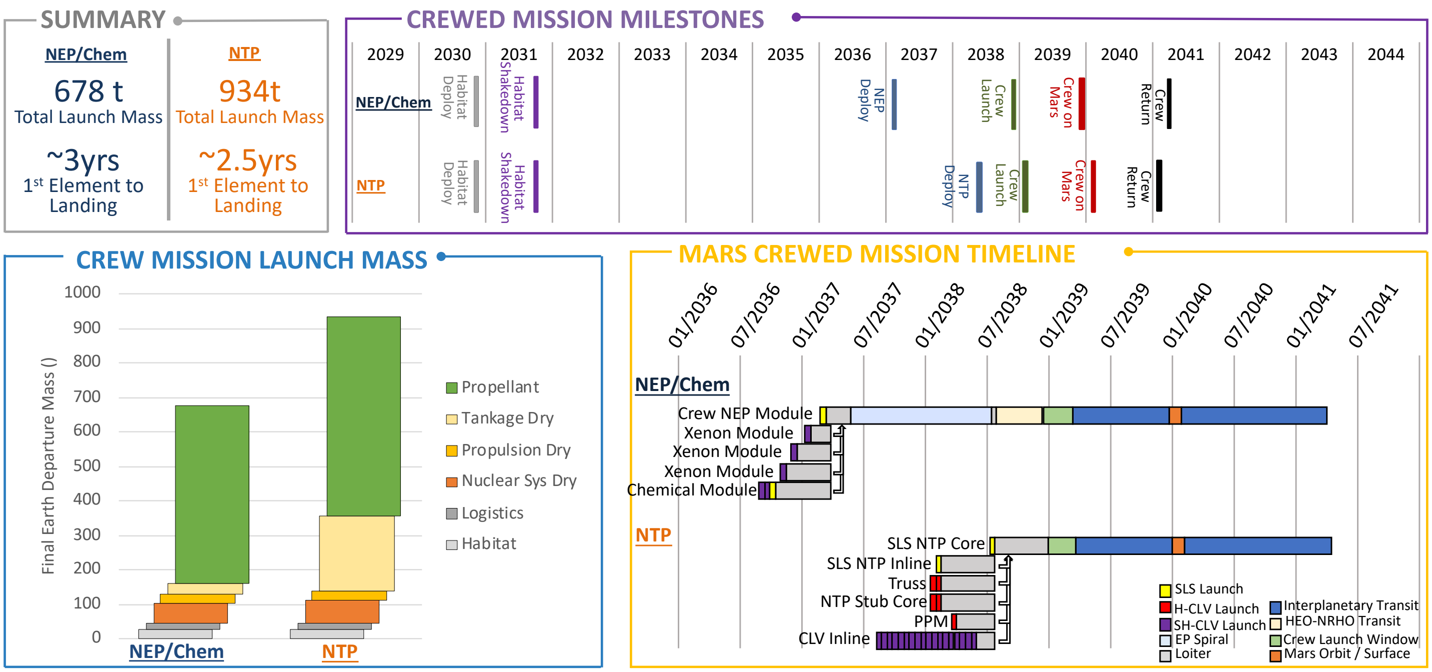


Figure 4.3-2. Comparison of NTP and NEP/CCP Crew Mission Campaigns.

The MTAS campaign assessment found the launch, assembly, and mission characteristics of the two propulsion systems are similar in many aspects. Where they differ is summarized in Figure 4.3-2 for the first crew mission, and in Figure 4.3-3 for the supporting cargo missions. The mass of the vehicles as they would leave Earth orbit differ by 37%. The propellant density of the two propulsion systems drives this difference, as the dry mass of the propulsion systems is about the same. As discussed earlier, an NTP system optimizes with a propellant with low atomic mass to increase I_{sp}, whereas an electric propulsion system optimizes with a propellant with high atomic mass to maximize thrust. As a result, the mass of the propellant required only differs by ~10% (~60,000 kg) but the mass of the tanks required to store those propellants differ by a factor of 7 (more than 200,000 kg). The mass of hydrogen propellant and tanks, as well as the volume, drive NTP's need for 18 SH-CLVs to support the crew mission alone. Because the CFM of hydrogen is so challenging, the desire is that they be delivered over a short period. By comparison, the denser NEP and LOX-LCH_4 propellants are delivered in 5 SH-CLVs for the same mission and could be launched on a greater number of smaller CLVs without concern for launch volume.

Comparing the cargo delivery approaches; because the plan with the NTP campaign is to send the first lander using NTP, it can deliver the first lander, a campaign pacing item, to Mars on a 2.5-year mission. The NEP/Chem plans to use the chemical propulsion system to deliver the first lander on a slower, 3-year transit time. This means the lander development plan for the NTP option can be 6-12 months longer. However, under this scenario, the first NTR engine and NTP system must be deployed about a year earlier, in 2035 to support a 2039 piloted mission, than the NEP option, which deploys the first NEP module in 2036. Thus, NTP would require a shorter development time than NEP/CCP.

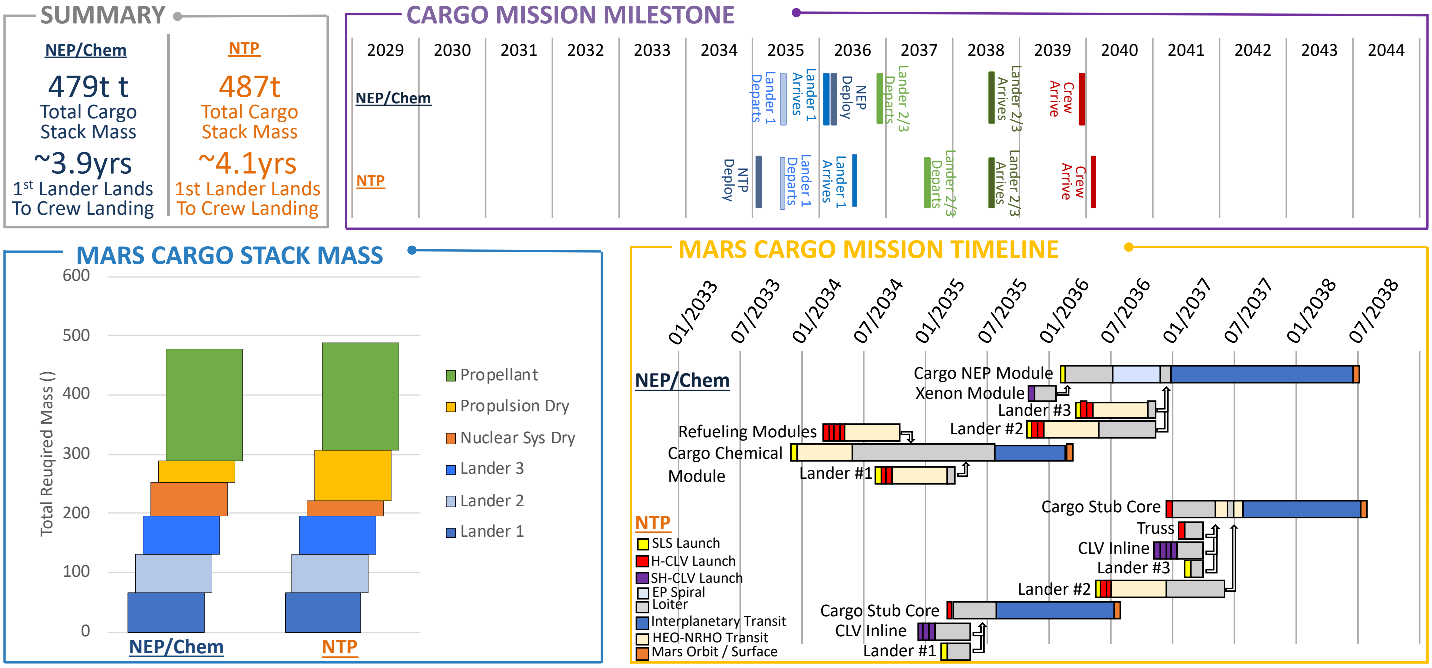


Figure 4.3-3. Comparison of NTP and NEP/CCP Cargo Mission Campaigns.

Both NTP and NEP/CCP crew mission approaches require two SLS launches for the propulsion systems. Additionally, the NTP system requires five Heavy-CLV launches for truss structures, power, and spacecraft systems, and additional side mounted Stub Core Stages. Overall, the NTP crew mission requires 26 unique launches, while an NEP/CCP crew mission requires 7 unique launches. The NTP cargo delivery approach requires 12 unique launches, while the NEP/CCP approach requires 14 unique launches to support a single Mars mission. However, again because of the larger launch volume of hydrogen propellant, the NTP option requires Super-Heavy CLVs, with larger payload volume. The number and type of launch vehicles required to support the NTP and NEP/CCP Mars campaign is shown in Figure 4.3-4. The overall higher number of launches required to support the NTP campaign, as well as the higher number of larger, more costly launch vehicles is reflected in the higher NTP mission cost in the comparison, below.

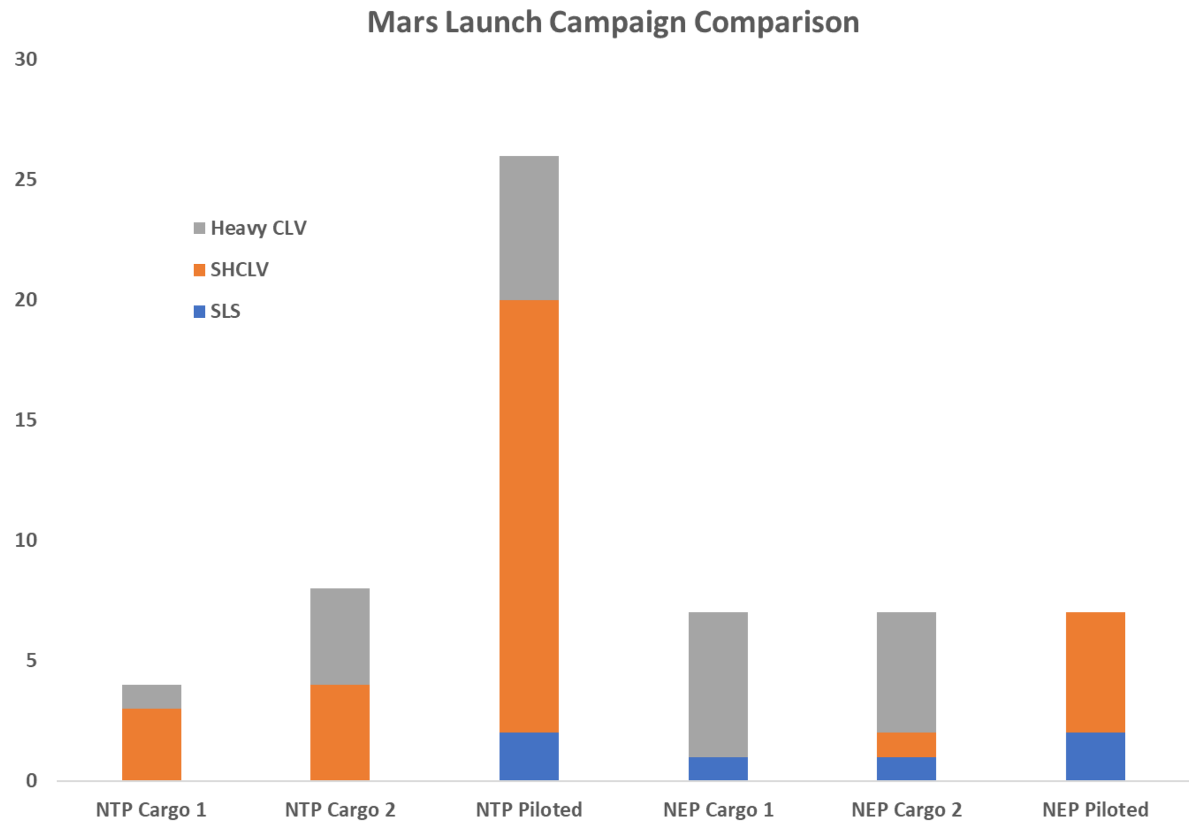


Figure 4.3-4. Comparison of Launch Count for NTP and NEP/CCP Mars Campaigns.

From an integrated campaign launch cadence perspective, the aggregation of the elements in lower Earth orbit significantly reduces the number of launches required to support both the crew and the cargo missions. However, the elements cannot loiter in the lower Earth orbit for long duration due to thermal and Micrometeoroid and Orbital Debris (MMOD) considerations (like ISS). The aggregation orbit must account for the thermal heat from the Sun and Earth reflection, and possible MMOD impact. To minimize the MMOD risk, both NTP and NEP/CCP reference cases selected an altitude with minimum MMOD, and the reference tank designs account for the worst-case total heat loads. With this in mind, the aggregation strategy increases the burden on the overall launch cadence of the elements during the aggregation phase with a significant number of launches required in a relatively short period of time. Significant improvements to the launch rate of commercial launch vehicles are currently being implemented by several commercial vendors, but this risk to the overall mission should be monitored.

Note that, due simply to the number of launches required, the probability of successful completion of either launch campaign is quite low (0-20%), with NTP probability perforce being lower. However, if as few as four spare launch vehicles and associated lower value elements are budgeted, this probability rises above 50% for both campaigns.

4.3.5 Cost

As a cost comparison between these alternatives has never been performed to this level of detail, the MTAS Programmatic Assessment Team performed a detail cost assessment of the NTP and NEP/CCP 1.2 reference cases based upon the best available historical data, bottoms-up and heuristic approaches, and collective expert-judgments. These assessments contain significant uncertainties in absolute cost figures but are valuable for relative comparisons between the alternatives.

The MTAS Programmatic Assessment found a significant (~37%) cost advantage for NEP/CCP over NTP when considering both development and first Mars campaign use. The major reason for the cost difference results from the larger number of unique NTP elements and launches. Figure 4.3-5 shows an overall cost comparison of the summation of costed elements and activities supporting a first Mars mission, and Figure 4.3-6 compares a breakdown of these cost considerations. The development (DDT&E) cost estimates for NEP and NTP each have large uncertainties due to technology challenges, and the differences identified to date are not significant compared with those uncertainties. Further technology maturation is required to answer key technology questions that will better bound the DDT&E costs, and significant differences between the options may be revealed once the technologies are more mature. DDT&E cost estimates for NTP were also affected by choices in the NTP vehicle design made to minimize the number of launches at the expense of requiring the development of additional, unique propellant elements. This approach should be revisited in future analyses to assess the potential to reduce NTP cost. Forward work to assess subsequent mission costs is also recommended, particularly to assess new build/launch of elements that are jettisoned during first mission use, and to assess retained element life limit extension costs for subsequent missions.

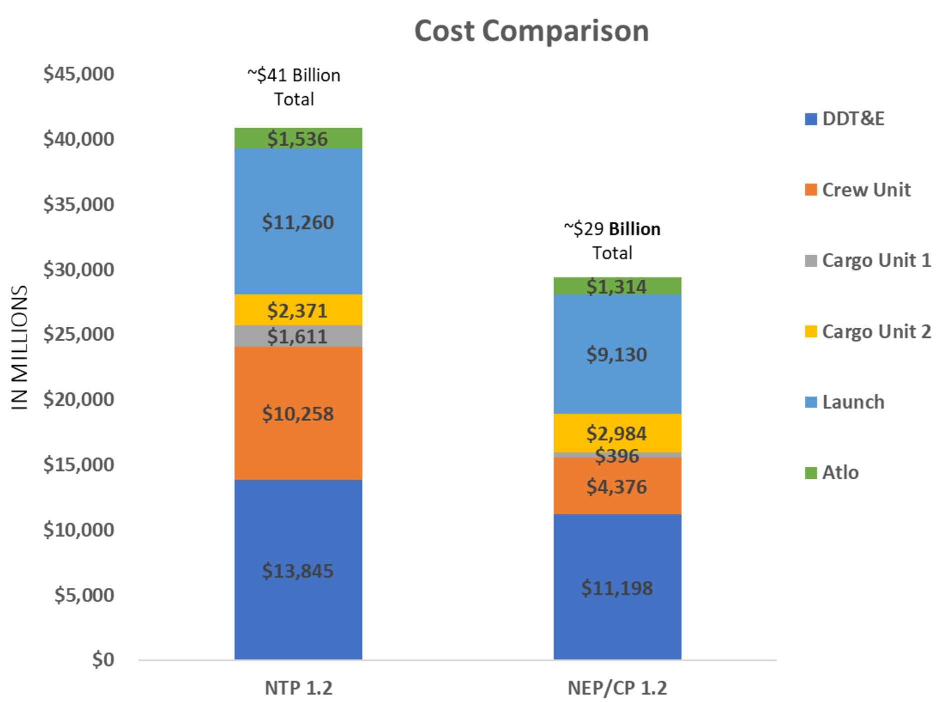


Figure 4.3-5. Cost Comparison of NTP and NEP/Chem Approaches to Mars 2039 in Millions.

Another area of cost uncertainty in DDT&E is the need for, and cost of, new facility development. It has long been considered, since the end of the Rover/NERVA program, that any new NTR engine development requires a new test facility to capture and/or filter the propellant exhaust. An estimate of these costs is reflected in the NTP DDT&E. A new facility for an integrated NEP system test is also included in the NEP DDT&E for comparison; however, several nuclear facilities are in development for terrestrial small modular reactors that could be leveraged or adapted for the development of space reactors supporting NEP with significant cost saving potential. Furthermore, the modular characteristics of NEP systems allows for separate subsystem ground testing of modular elements with simulated interfaces for qualification, which could be accomplished in separate facilities, easing the requirements for new facility development. Thus, the NEP DDT&E estimate is considered potentially conservative. That stated, the development of NEP will also require some facility development, notably large, high-pumping capacity vacuum chambers for testing multiple, large electric thrusters, which is included in the NEP DDT&E.

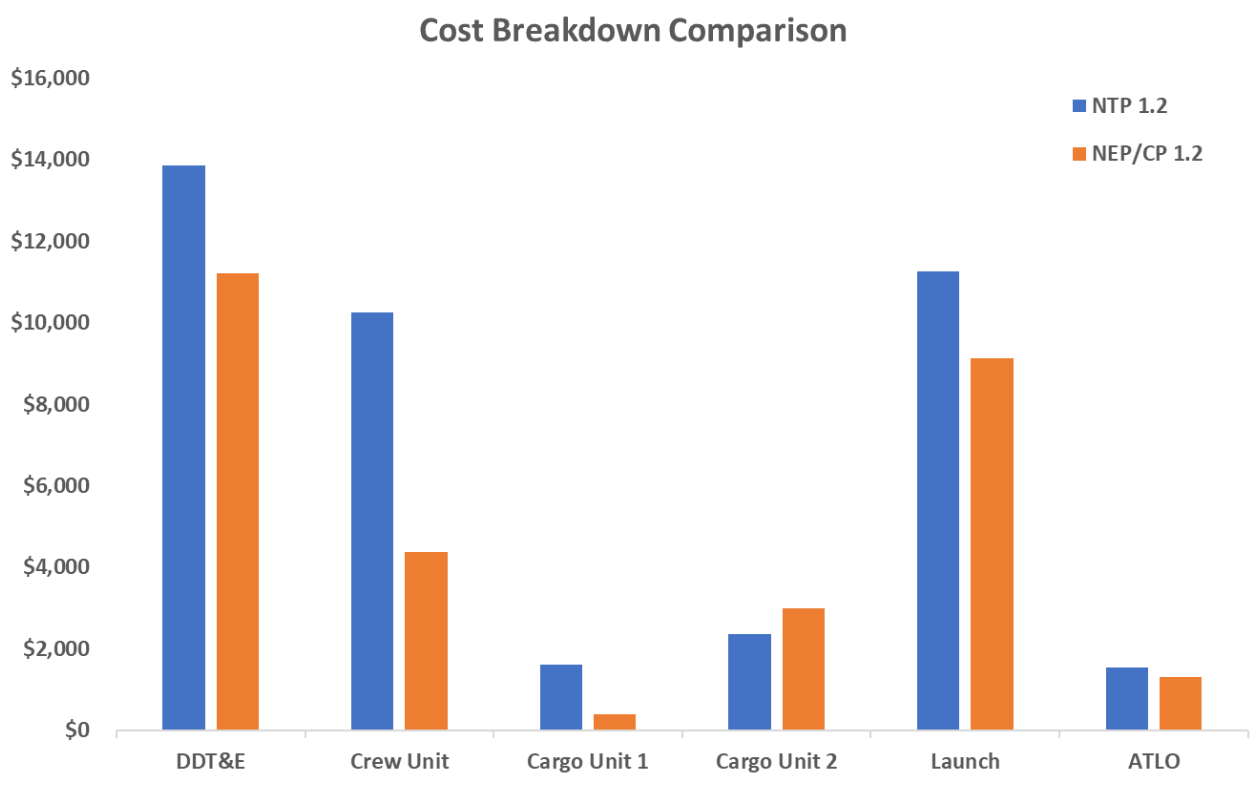


Figure 4.3-6. Cost Breakdown of NTP and NEP/Chem Approaches to Mars 2039 in Millions.

A tangible advantage of the high I_{sp} and high propellant density of the NEP/CCP approach is the significant reduction in launches and launch costs due to the comparatively low propellant mass required, as well as significantly smaller launch volume for the propellants. This advantage results in not only ~20% lower launch costs, but over two times lower hardware cost for the NEP/CCP vehicle, which has significantly fewer elements. This advantage is compounded when considering the comparison of cargo delivery vehicle costs for the campaign.

4.3.6 Other Considerations

The effect of system generated radiation on the crew and overall system must also be considered and may offer a distinction between NTP and NEP. MTAS assessed the shielding that would provide equivalent system doses to the crew to provide comparable performance. However, to be complete, this assessment must consider the crew habitation and its configuration and integration with the propulsion system, which is not fully know at the time of this study. An integrated assessment with the Transit Habitat research team and with the Office of the Chief Health and Medical Officer within NASA is still necessary. Furthermore, NASA's Office of the Chief Health and Medical Officer must develop crew standards for radiation exposure that consider both onboard radiation and natural sources in space, such as Galactic Cosmic Radiation (GCR) and Solar Particle Events (SPE). Therein lies a significant mission – technology trade between the propulsion to drive even faster missions to reduce crew exposure to natural sources, and the radiation those systems generate. This more detailed and comprehensive assessment has been identified as work for future study.

A further distinction between NTP and NEP may also be revealed upon completion of a Loss-of-crew/Loss-of-mission (LOC/LOM) assessment for each of the options. For example, while the presence of multiple fission cores in the NTP concept (vs. the single core in the NEP option) may appear to provide protection against a failure mode leading to a SCRAM of a reactor core (i.e., "engine out" capability), this factor also presents a notably increased exposure to a failure mode resulting in a reactor explosion. An assessment of which option would offer lower LOC/LOM risk is not trivial and is beyond the scope of this study.

5.0 CONCLUSIONS

This study is an initial step toward a complete understanding of the Mars transportation trade space and options. The objective of the short-stay approach is to minimize the overall mission risk, including risk to the exploration crew, as well as the programmatic risk of developing the numerous technologies needed to support a long surface stay for the exploration crew on Mars. The study found that the high ΔV required for fast Mars missions with short stay times drives the need for nuclear propulsion technology and highlights the performance differences between promising alternatives. Numerous past studies have shown that non-nuclear options require extremely aggressive technologies and concepts of operations to close a fast Mars mission. An apples-to-apples all-chemical ConOps is not likely to be viable. However, an objective assessment of non-nuclear options must be completed, and is recommended in the table of follow-on work, below.

In summation, both NTP and NEP/CCP propulsion system concepts have the potential to fulfill the mission requirements for fast Mars transits in the late 2030s and meet the Agency's objectives, including enabling ~2-year roundtrip mission durations. But these options use different approaches with unique challenges. NTP offers more efficient high thrust capability for rapid response to contingencies and gravity well departures to support a human Mars mission as well as DOD applications. NTP advantages result from the highest I_{sp} for currently achievable high acceleration propulsion. However, low thrust NEP, when augmented with high thrust LOX-LCH_4 conventional chemical propulsion looks very promising as the more advantageous alternative. The NEP/CCP approach results in comparatively: fewer launches with fewer number and types of elements; lower launch costs, unit costs & campaign costs; and easier launch, storage, and management of propellants. An additional benefit of higher specific impulse with NEP is the mitigation of the energetic challenges that result from more distant planetary alignments between Earth and Mars reducing the variance in mission performance across the synodic cycle. Furthermore, NEP/CCP has the potential for easier development due to the possibility of separate subsystem ground testing possible with modular elements and interfaces that can be accurately simulated. Additionally, there are stronger NEP synergies and feed forward from terrestrial energy and lunar power, and strong leveraging of existing development for electric propulsion (Gateway SEP Power and Propulsion Element), PMAD (MW_e all-electric aircraft). There are additional synergies with ongoing LOX-LCH_4 propulsion development. NEP technology challenges for most of the subsystems will be addressed by fission power development for the lunar surface, and by SEP for Gateway. None of these challenges are at high risk to become showstoppers[6]. In contrast, NTP has at least three technology development challenges that are at high risk to become show stoppers: a fuel and core design to achieve lifetime at 900 sec I_{sp}, test facilities for full scale integrated testing, and long term hydrogen CFM. Finally, NEP offers stronger benefits for precursor and follow-on potential applications, such as robotic science and missions beyond-Mars. In consideration of the broad NASA needs for propulsion, and even broader national needs, both NTP and NEP possess an imperative for investment, as they each have significant advantages in specific applications. However, if the apparent Mars NEP/CCP performance metrics can be verified in technology development, NEP/CCP would be the more advantageous option for human and cargo transportation to Mars. Furthermore, NEP/CCP offers significant cost advantages over NTP for development and first Mars campaign use, and these could be multiplied by greater potential for reuse.

MTAS programmatic assessment found both concepts could complete Operational Readiness Review (ORR) by ~2035 assuming no schedule delay, but only if significant investment in nuclear propulsion begins immediately. Furthermore, the federal policy guidance in SPD-6 that HEU should be limited to applications for which the mission would not be viable with other nuclear fuels. The consideration of HALEU presents an additional schedule challenge because it is a departure from the HEU heritage fuel and reactor development for both systems. HALEU will require advanced technology for fuels and moderators, adding cost, schedule, complexity, and development risk. The only reason to consider HALEU for space applications is to address policy concerns with the use of HEU.

Both NTP and NEP require an aggressive development program to meet a 2030s launch date. A strategically focused investment plan in both NTP and NEP is recommended to address key feasibility and development questions and retire risk before selecting a propulsion system for human Mars transportation. The technology development plan recommended in this document matures both NTP and NEP to TRL 5 with an integrated component test in a relevant environment. For NTP, this requires three focus areas: first, developing and testing full scale fuel elements to validate ability to produce an I_{sp} of 900 seconds or higher for greater than 4 hours; second, demonstrating viable NTP ground test exhaust capture and filtering to support full-scale facility development necessary for flight validation; and third, demonstration of CFM technologies necessary for long-duration hydrogen propellant storage and transfer. For NEP this is defined as a 1,000-hour thermal vacuum test campaign with a non-nuclear representative scale power subsystem Technology Demonstration Unit (TDU), parallel fuel and moderator irradiation testing at the required temperature, fluence and burnup, and a separate vacuum chamber test of an EP subsystem TDU. LOX-CH_4 chemical propulsion will require demonstration of CFM technologies necessary for long-duration liquid oxygen and liquid methane propellant storage and transfer. The down selection of a propulsion system would happen after both NTP and NEP subsequently complete a preliminary design review. NASA is currently investing in NTP to support synergy with other agencies and Congressional direction; however, as a result of the favorable NEP/CCP conclusions provided by the MTAS assessments, a balanced portfolio of investment in both NTP and NEP is necessary until the PDR milestones are met, and sufficient risk is retired to confidently support a decision.

The propulsion choice for human transportation beyond the Moon could set the course of human exploration for the U.S. and humanity for decades to come. It will be a decision that defines a decade-long development worth more than $10B. This decision must be made based on sound, objective analysis, and strong technical confidence, for which the current state of knowledge is insufficient. Thus, the MTAS team recommends further,

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[6] Measured by lower average and peak Advancement Degree of Difficulty (AD2) as assessed by the NESC independent study.

more detailed studies to guide investment, and funding for a strategically focused technology maturation in both NTP and NEP to support the best propulsion decision for humanity's first interplanetary spaceship.

RECOMMENDED FOLLOW-ON WORK

- *Complete initial study scope .*
    - *Complete a non-nuclear reference case and conjunction reference case for comparison*
    - *Thorough assessment of issues touched on in Phase 1: mission robustness, human rating, reusability, extensibility*
    - *Complete remaining HEU vs HALEU independent assessment in order to provide an objective, comprehensive assessment of the relative merits and challenges. Include an assessment of fast/epi-thermal NTP*
    - *Assess key technology trades, including subsystem technologies, to inform design down-selection toward focused technology investments, and quantify performance enhancing potential for advanced technology investments*
- *Provide an analytical foundation for investment and support Agency partnerships for long-term success*
    - *Engage DOD to participate in synergistic requirements development*
    - *Continue engagement with DOE and DOE Labs to build understanding of NASA missions and requirements*
    - *Look at other demand side missions for NASA, other Agencies, emerging commercial interests, and international partnerships*
- *Answer key questions emerging at the end of MTAS Phase 1*
    - *Assess mission sensitivity to Key Performance Parameters.*
    - *Assess the value to the Agency of precursor missions using subscale systems on the development path. Consider a precursor mission to demonstrate both propulsion system and full-scale Entry, Descent, and Landing capability*
    - *Assess potentially advantageous mission architecture trades, such as predeployment of propellant to Mars, crew direct entry at Mars, and crew direct entry at Earth return with subsequent rerendezvous of the Deep-Space Transport habitat and propulsion elements*
    - *Deeper evaluation of the facilities needed for technology, development, I&T, and flight*
    - *Address 1.2 shortcomings - Cargo delivery scenarios need refinement*

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