Characterization of In-Space Propulsion Trade Space to Support Initial Human Mars Segment

Patrick R. Chai[a,1], **Michelle A. Rucker**[b], **Laura M. Burke**[c], **Stephen J. Edwards**[d], **Douglas J. Trent**[e]

[a] *NASA Langley Research Center, Hampton, VA, United States, patrick.r.chai@nasa.gov*
[b] *NASA Johnson Spaceflight Center, Houston, TX, United States, michelle.a.rucker@nasa.gov*
[c] *NASA Glenn Research Center, Cleveland, OH, United States, laura.m.burke@nasa.gov*
[d] *NASA Marshall Spaceflight Center, Huntsville, AL, United States, stephen.j.edwards@nasa.gov*
[e] *NASA Marshall Spaceflight Center, Huntsville, AL, United States, douglas.trent@nasa.gov*

The National Aeronautics and Space Administration's Exploration Systems Development Mission Directorate has been developing architecture concepts for human missions to Mars in alignment with the agency's Moon-to-Mars Strategy & Objectives. One of the key components of a human Mars mission is the in-space transportation system that delivers crew and cargo to Mars vicinity and returns the crew safely back to Earth. The Mars Architecture Team has been evaluating four in-space transportation options to satisfy this functional need: 1) Hybrid Nuclear Electric/Chemical Propulsion, 2) Nuclear Thermal Propulsion, 3) Hybrid Solar Electric/Chemical Propulsion, and 4) Chemical propulsion. To answer the question "which transportation option is best?" it is critical to understand the performance characteristics—and limitations—for each architecture. Each option could be considered "better" than the others, depending on the primary selection criteria and the order of the integrated decisions that are made. The two nuclear transportation options provide the highest performance ceiling but carry the penalty of the additional technology development timeline and cost. If applied to the highest energy need missions, the penalty may be justified to achieve the desired result. But for lower energy demand missions, the simplistic nature of the non-nuclear option may be a more optimized solution. Ultimately, the four transportation options may not inherently be mutually exclusive, but only with detailed analysis and evaluation can the transportation trade space be defined to help inform this decision.

Acronyms/Abbreviations

Exploration Systems Development Mission Directorate	ESDMD
Liquid Methane	LCH4
Liquid Hydrogen	LH2
Liquid Oxygen	LOx
Lunar Distance High Earth Orbit	LDHEO
Nuclear Electric Propulsion	NEP
Nuclear Thermal Propulsion	NTP
Solar Electric Propulsion	SEP

1. Introduction

The National Aeronautics and Space Administration's (NASA's) Exploration Systems Development Mission Directorate (ESDMD) has been developing architecture concepts for human missions to Mars in alignment with the agency's Moon-to-Mars Strategy & Objectives[1]. These Moon-to-Mars Objectives provide a comprehensive framework to ensure that human Mars architectures will meet—or can evolve to meet—more stakeholder needs. The focus of ESDMD's Strategy and Architecture Office has been to decompose the blueprint objectives into the actionable characteristics for both lunar and Mars missions and to develop functional needs to enable an integrated end-to-end mission architecture. After mapping objectives to the required functional capabilities, the architecture team will coordinate with technology and element concept developers and identify the key architecture decisions that must be made. Because decisions in one part of the architecture will ripple through other parts of the architecture, it is critically important that the agency understand the effect of each decision on the integrated architecture, including differences depending on the order in which the decisions are made.

One of the key components of a human Mars mission is the in-space transportation system that delivers crew and cargo to Mars vicinity and returns the crew safely back to Earth, as specified by multiple Moon-to-Mars Objectives and Recurring Tenets[1]. The Mars Architecture Team within ESDMD's Strategy and Architecture Office has been evaluating four in-space transportation options to satisfy this functional need: 1) Hybrid Nuclear Electric Propulsion (NEP)/ Chemical Propulsion, 2) Nuclear Thermal Propulsion (NTP), 3) Hybrid Solar Electric Propulsion (SEP)/ Chemical Propulsion, and 4) All-Chemical propulsion. To address the question "which transportation option is best?" the architecture team needs to understand the performance characteristics—and limitations—for each transportation option. Each option could be considered "better" than the others, depending on the primary selection criteria

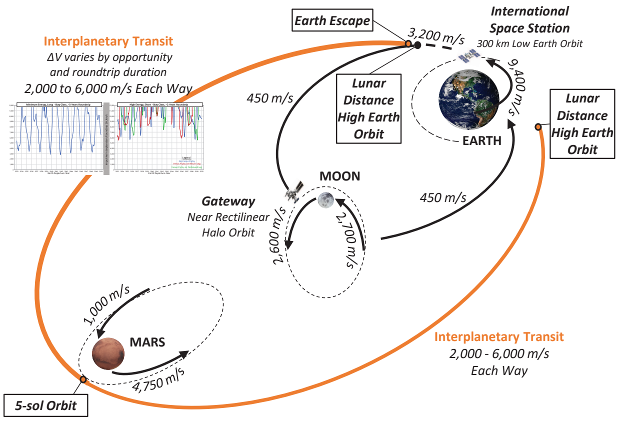

Figure 1: Earth-Moon and Earth-Mars Energy Map

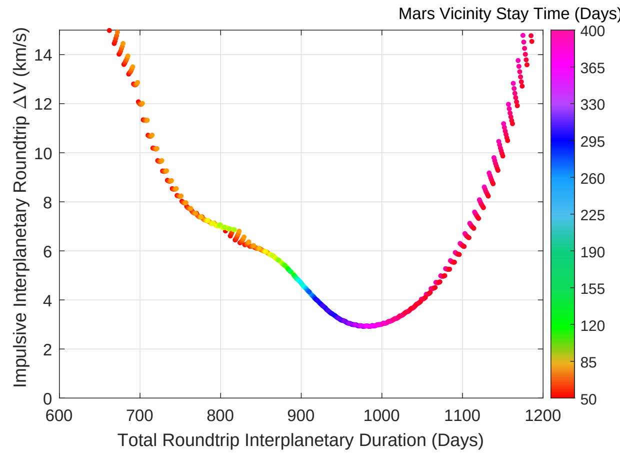

Figure 2: Earth-Mars Roundtrip Interplanetary Energy Needs Between LDHEO and Mars 5-Sol as Function of Duration and Mars Vicinity Stay Time for a Representative Mission Opportunity

and the order of the integrated decisions that are made.

Selection of a human Mars transportation system will be a complex decision shaped by numerous factors, such as mission objectives, exploration partner contributions and commitments, cost, schedules, and integrated risks. The four transportation architectures presented in this paper represent the range of options currently being analyzed. Specific implementation of the different transportation systems will depend on the reference mission of interest and a balance between the optimization of the system and the robustness to other mission parameters. This paper provides a high level overview of the in-space transportation trade space, with the goal of introducing additional context for the Mars in-space transportation decision in the future.

2. Interplanetary Transits between Earth and Mars

In Mars architecture discussions it is helpful to keep in mind that mission distances traveled will be at a scale far beyond the entirety of human space flight experience to date. A single roundtrip journey between Earth and Mars will put about 1.8 to 2 billion kilometers on a Mars transportation system's odometer, regardless of departure opportunity or trajectory. To put this distance in context, for the recent Artemis I mission between Earth and the Moon, the Orion spacecraft traveled only 2.2 million kilometers[2], or roughly 1/1000th the distance for a roundtrip Mars mission. The distance that the Moon circles Earth only varies by about 43 thousand kilometers over the course of its orbit, so it always takes about the same amount of energy to travel to the Moon and back no matter when we go. By contrast, the distance between Earth and Mars can vary by as much as 340 million kilometers as the two planets orbit the Sun. The closest Mars ever approaches Earth is 54.6 million kilometers; at their farthest, over 400 million kilometers of deep space separates the two planets.

In addition, lunar and Mars missions have unique challenges, and systems designed for one destination may not be directly applicable to the other as they have different energy and mission needs. Missions to Mars have higher energy need, require much longer system service life, and have more stringent departure window constraints. This higher energy need is evident in Figure 1, which shows an Earth-Moon-Mars energy map for missions to the surface of the Moon or Mars. The difference between missions to the Moon and Mars means that much of the operational experience and many of the paradigms, such as mission control, sparing/resupply strategy, crew rescue, or mission abort contingency planning do not apply, and a different approach than previously used on heritage programs will be required.

To that end, recent analysis was designed to explore the pros and cons of different transportation system options across a wider range of mission profiles than previously considered. The initial metric of interest for recent assessments was total roundtrip mission duration, due to the significant duration-related flow-down impacts to crew health and performance[3], technology investment, development timelines, and cost. Interplanetary transportation energy is a function of many variables, including the distance and relative velocity of the planets as well as the mission duration and orbital stay time. This relationship between the energy required as a function of total mission duration and Mars vicinity stay time is shown in Figure 2. For this particular analysis, Lunar Distance High Earth Orbit (LDHEO) is used as the starting point for Earth departure staging for Mars missions, and a 5-sol orbit is used for Mars arrival staging and departure.

Historically, Mars mission duration has been framed as a binary choice. On one end of the spectrum, to minimize the total energy required to achieve the roundtrip mission, mission planning has selected optimal planetary departure and arrival timing to maximize the benefit of the natural relative position and velocity between the planets. This results in what is typically known as minimum energy long-stay missions (or conjunction class), where both the Earth-to-Mars and Mars-to-Earth trajectories are minimum-energy

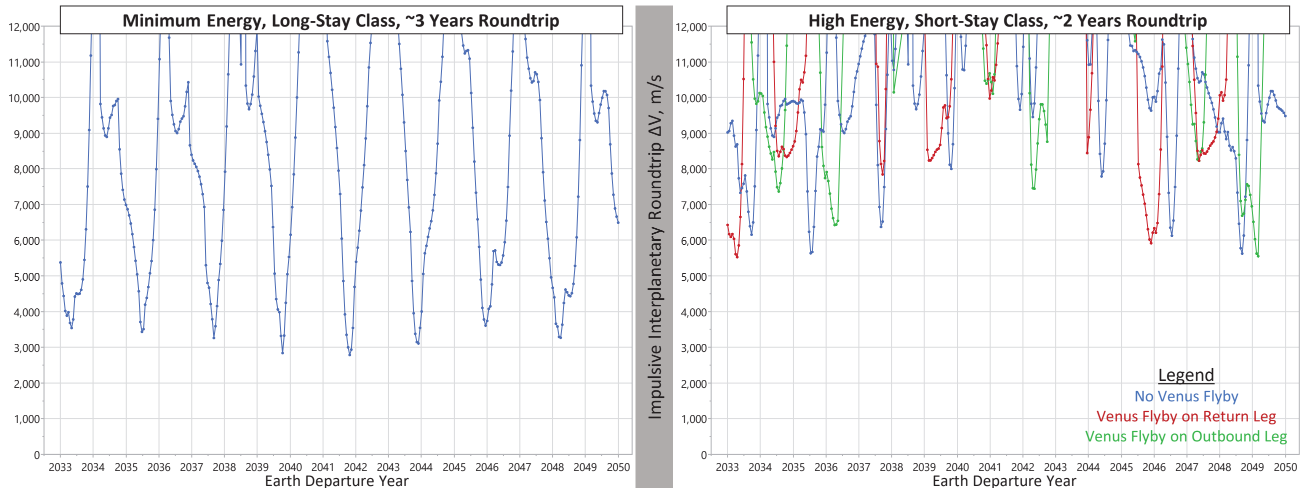

Figure 3: Earth-Mars Roundtrip Interplanetary Energy Needs Between LDHEO and Mars 5-Sol as Function of Duration and Mission Opportunity

in nature. This type of mission is generally ~3 years in total roundtrip duration and requires a Mars vicinity stay time on the order of 300 to 500 days while awaiting the next optimal planetary alignment for the return journey.

To achieve shorter duration roundtrip missions to Mars, higher energy trajectories must be utilized. Unlike the minimum-energy conjunction-class mission, where the Mars stay time is dictated by the waiting period for the optimal return trajectory, shorter roundtrip missions do not have built-in lower bound constraints for Mars stay time. This parameter becomes a key driving factor in interplanetary mission planning. Shorter mission duration also results in shorter stay time at Mars. Examples of these shorter roundtrip missions to Mars are typically known as short-stay (or opposition class) missions. These types of mission are generally ~2 years in total roundtrip duration, with relatively short Mars vicinity stay time (weeks to a month or two), and may potentially require additional planetary flyby maneuvers to help reduce the total energy cost.

These two classes of mission have traditionally been the focus of Mars mission design and planning, but it is important to note that roundtrip missions to Mars are not limited to these two options. The energy vs. time trade for a roundtrip mission to Mars is a continuum, as the mission duration is shortened, the energy required to achieve the roundtrip mission increases rapidly and becomes exponential as shown in Figure 2. This translates to an exponential increase in the vehicle mass required, including both propellant and propulsion system dry mass, to achieve the roundtrip journey. One can attempt to break the exponential nature of the energy need and "cheat" the rocket equation by breaking up how the energy is achieved. For example, pre-deploying the return propellant or collecting propellant from in situ resources are both possible ways to seemingly reduce the total propellant mass required for higher energy missions. However, these methods increase the overall operational complexity and risk of the integrated mission and may not reduce the total mass required to be launched.

Pre-deployment of propellant requires additional transportation systems to deliver the propellant to the destination and assets to help maintain propellant at remote locations for use. Propellant production in space or on another planetary body is a significant challenge and will require complex operations and numerous assets to be deployed for successful implementation. These methods do not necessarily solve the challenge of the higher energy missions but are often as touted as simple solutions to address the challenge while the implementation is anything but simple. Regardless, Mars mission design should not be a contest of "conjunction" vs. "opposition," but rather an integrated and thoughtful analysis of all parameters of interest.

In addition to the mission duration and stay time considerations, the energy required for the roundtrip transit from Earth to Mars and back is highly dependent on the timing. Because both planets are orbiting the Sun, both the distance and the relative velocity of the planets are constantly changing, cycling on a roughly 15- to 20-year cycle. It always takes about the same amount of energy to reach the Moon from Earth, but the amount of energy required to reach Mars varies considerably over this cycle. This variation leads to some mission opportunities requiring 20 to 60% more transportation energy to complete the interplanetary transits between Earth and Mars for the same mission duration than in other calendar years. Figure 3 shows this cyclical variation in energy during interplanetary roundtrip transits between Earth and Mars for both types of missions. Roundtrip transit time, Mars stay time, and departure dates are all important factors in determining the total energy required to achieve roundtrip missions. Analyzing the implications of each factor on all relevant systems will result in better understanding of the overall design trade space to support more informed decisions.

3. In-Space Transportation System Options

Earth-Mars in-space transportation systems serve to transport the crew and supporting systems to Mars and return crew to Earth. All Earth-Mars transportation architectures will consist of a propulsion and power backbone paired with one or more payload elements. A single transportation system design could be used for both crew and cargo deliveries, but to optimize for cost, development schedule, or other metrics of interest, variants may be mixed within a single Mars campaign. For example, a slower, less-expensive, non-nuclear transport for predeployed cargo could be combined with a faster, higher-powered nuclear system for crew transport. In the crew variant transportation system, the payload is a crew habitation system and all the utilization payloads, logistics, supplies, and spares for the in-space portion of the mission, including contingency operations. The cargo variant will deliver the supporting systems and/or elements to Martian orbit or the surface in preparation for the crewed mission.

For different mission durations, transportation systems can be optimized, from both a configuration and a performance perspective, for the specific requirements of that reference mission. But an optimized transportation implementation might come at the cost of compromising the extensibility and flexibility to other mission design parameters that may be of interest. To inform the total mission duration and Mars vicinity stay time decisions, which in turn will inform a host of other decisions (including transportation propulsion technology investments), architecture designers will need several pieces of information: an understanding of system-by-system performance sensitivity over the entire duration trade space and an integrated campaign assessment for the various possible implementations, including integrated risks to the human system. To that end, the Mars architecture concepts presented here are intended to populate a broad swath of the transportation trade space, allowing the architecture to be evaluated with different implementations of four different transportation systems in the context of different potential missions.

3.1. Hybrid Solar Electric / Chemical Propulsion

The Hybrid SEP/Chemical Propulsion concept was originally designed and optimized for minimum vehicle stack mass (and, hence, minimum Earth-launched mass/cost) during NASA's Evolvable Mars Campaign effort[4, 5]. The initial concept vehicle leverages the highly efficient propulsion system on a minimum energy type trajectory for both crew missions[6, 7] and cargo missions[8]. The initial main propulsion system, derived from technology planned for the Gateway program, consists of multiple Hall Effect Rocket with Magnetic Shielding (HERMeS)[9, 10] thrusters and bi-propellant thrusters as the integrated hybrid propulsion system. By combining chemical and electric propulsion into a single spacecraft and applying each where it is most effective, the hybrid architecture enables a series of Mars trajectories that are more fuel efficient than an all chemical propulsion architecture, without significant increases to trip time. This concept has seen many iteration and updates since its inception[11], including updates to the thruster design, updates to chemical engine design and propellant type, and other updates based on various analyses[12, 13, 14] to understand the performance of this concept. In the current analysis cycle, additional concept configuration analysis is underway to understand the potential for a higher power system to be designed to support shorter duration missions[15, 16].

The primary challenges associated with a propulsion system that deploys electric propulsion are the limitation associated with the power generation and conversion system and the nature of the electric thrusters themselves. For SEP systems, electrical power is generated by solar arrays and then used to drive the electrical thrusters. This results in significant limitation in two ways. First, as the spacecraft moves away from the Sun, the solar arrays generate less power, reducing the overall effectiveness of the SEP system. Typically, the solar arrays are oversized to account for this power deficiency, but this means carrying additional mass for these less optimal conditions. Second, the power generation by the solar panels degrades over time due to degradation of the solar cells. In the initial design of the SEP/Chemical hybrid propulsion system[6], the arrays were oversized to accommodate for end-of-life operations, as the intention of the design is to achieve multiple roundtrip missions to Mars.

As discussed in the initial publication[6], planning for these lower power operations can be tricky, and typically the arrays are sized based on the sub-optimal end-of-life operation. The additional challenge associated with this design comes from the variation of the roundtrip energy need based on mission opportunities (as seen in Figure 3). If the arrays are sized for end-of-life operation for a moderate energy need mission opportunity, then the system is unable to perform that same mission for a higher energy need opportunity. However, if the arrays are sized for the worse case scenario of end-of-life during the most challenging mission opportunity, the system may be able to perform the missions across opportunities, but it is doing so with a significantly oversized solar array that can be challenging to integrate into the spacecraft. The integrated trajectory and system design optimization is one of the key areas of analysis for the SEP-based mission architecture.

Another challenge associated with electric-propulsion-based systems is that while electrical thrusters are a highly efficient propulsion system when it comes to propellant usage, they generate very little thrust. In order to take advantage of the electric propulsion system's high efficiency, long mission durations are desired to aggregate the low thrust for very long periods of time to achieve the desired mission energy. The Cassini spacecraft's mission to Saturn is the best example of the efficient use of electric thrusters[17, 18]. For human missions, however, this application becomes challenging, as long duration spaceflight poses significant risk to the crew in many different forms[3]. Thus, the hybrid system is developed to combine a low thrust system with

Figure 4: Potential Variation in Hybrid SEP/Chemical Propulsion Concepts for Different Mission Durations

Figure 5: Potential Variation in Hybrid NEP/Chemical Propulsion Concepts for Different Mission Durations

a high thrust system, taking advantage of each propulsion system's strength. This worked very well for a fixed mission duration paradigm, where the energy required varies only moderately across mission opportunities.

The low thrust portion of the trajectory is relatively inflexible with fixed power, as the thruster power defines the propellant throughput and the array power defines how much the thruster can be utilized. If the system is underpowered or the energy required for the roundtrip mission increases significantly, the high-thrust chemical system must make up the difference with additional maneuvers at planetary departures and arrivals. This creates a sub-optimal design where the the advantage of the low thrust system is not being sufficiently leveraged. However, increasing the power level for the thrusters and the solar arrays can bring on additional challenges for the shorter missions, particularly with regards to spacecraft integration, where large solar array wings and their associated wiring must be assembled. It took the International Space Station decades to build up its solar array wings, with dozens of extravehicular activities to assemble and configure the array for use. Figure 4 shows conceptual renderings of potential SEP/Chemical hybrid propulsion systems and how the design could vary depending on the mission duration design needs.

3.2. Hybrid Nuclear Electric / Chemical Propulsion

NEP systems have been studied and proposed in the past for use for planetary exploration missions[19] and cargo delivery for Mars missions[20], but the significant power need to facilitate a timely planetary departure and arrival maneuver meant that applying this technology for crewed missions remained a major challenge. However, similar to the SEP concepts, the augmentation of the NEP system with a high thrust chemical propulsion system opened new possibilities for crewed missions. Compared to using solar arrays to power the electric thrusters, nuclear systems have the benefit of constant power provided regardless of solar distance, but they do come with the cost of a significantly larger system mass and higher system complexity.

NASA Glenn Research Center's COMPASS concurrent engineering team developed a concept for a Nuclear Electric / Chemical Hybrid Propulsion[21] system for human Mars missions. The vehicle concept builds upon the design experience from previous solar and nuclear electric propulsion system studies. The result is a vehicle concept that has separate nuclear electric propulsion and chemical propulsion elements that are integrated together with the deep space habitation[22] system to support the roundtrip mission to Mars. In the current analysis cycle, additional concept configuration analysis is underway to understand the scaling effect of a lower power NEP system to support longer mission durations, and to identify vehicle break points for integrated NEP system design.

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. While NEP systems enjoy the benefit of constant power with relatively minimal power loss as a function of operating duration (as compared to SEP array degradation), a similar spacecraft integration complexity challenge exists due to the radiators. 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. However, as power demand increases for the higher energy shorter duration missions, the challenge of packaging and deploying larger and more complex radiators sufficient to reject the waste heat of higher power NEP system can become significant.

An additional challenge with regards to any nuclear-based system is the requirement for additional radiation shielding for crew protection. Any crewed mission that traverses outside the Earth's magnetosphere will require some form of radiation protection from both galactic cosmic rays and solar particle events. These are major challenges for the design of the crew habitation systems. A nuclear-energy-based propulsion system will add to that challenge, as additional protection may be needed to protect the crew from the nuclear reaction that is powering their own spacecraft. This can result in a spacecraft design with long structural booms to increase the distance between the crew and the nuclear reactors, or additional heavy shielding that will increase the mass, complexity, and cost of the transportation system.

Similar to the SEP/Chemical Propulsion system, the NEP/Chemical's power need will be driven by the mission duration and the roundtrip energy need. While the NEP has constant power, higher and higher power thrusters are needed to provide enough thrust to achieve higher energy, shorter duration missions. This higher demand in power compounds both the radiation protection for the crew and the radiator sizing and complexity challenges. For the lower energy missions, the mass penalty of the nuclear reactor power generation system (as compared to solar arrays) may not result in a more mass optimal solution. There are also some significant mass and scaling challenge with lower power NEPs, as some components of the NEP systems cannot be shrunk down to accommodate lower power operation. Figure 5 shows conceptual renderings for potential NEP/Chemical Hybrid propulsion systems and the potential configuration and scaling that could result from designing the system for different energy need.

3.3. Nuclear Thermal Propulsion

A NTP system provides thrust by heating propellant that is passed through a nuclear reactor, with the propel-

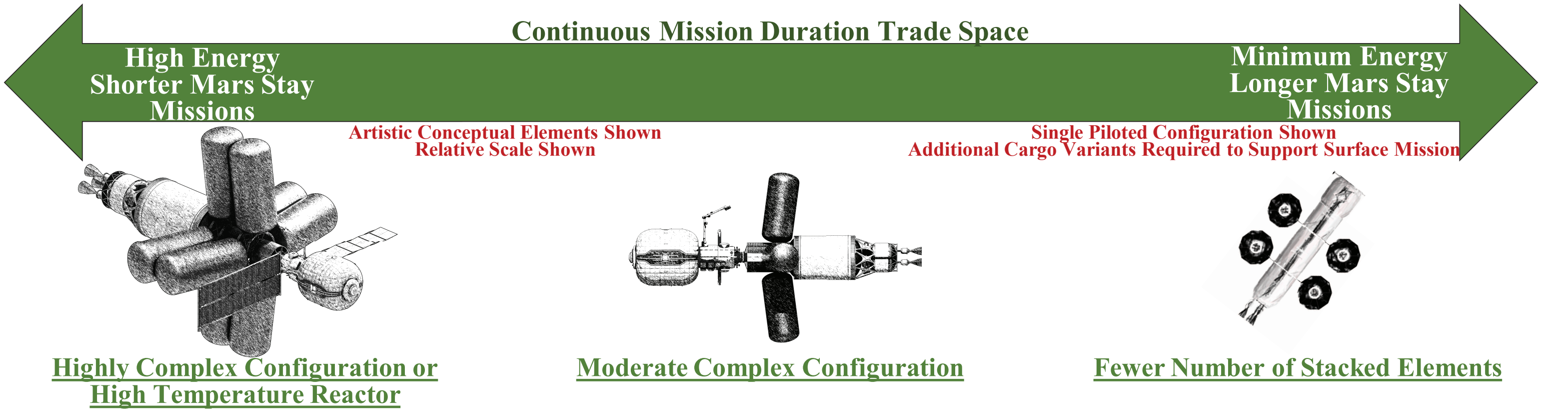

Figure 6: Potential Variation in NTP Concepts for Different Mission Durations

Figure 7: Potential Variation in Chemical Propulsion Concepts for Different Mission Durations

lant exiting the reactor at high temperature and expanding through a nozzle to generate thrust. Several NTP concepts that are designed for potential lunar and Mars application have been published recently[23, 24, 25]. In many instances, vehicle designs consist of a core stage that houses the main reactor and the nozzles to produce thrust, inline tanks that dock to the core stage to provide propellant capacity, and other elements to assist in the integration across the stages and provide the power and propellant feed systems between the propellant tanks and the main engines. Figure 6 shows conceptual renderings of NTP systems in the context of the continuous mission duration trade space.

For NTP systems, the main factor of performance is the engine's specific impulse (propellant efficiency), which is primarily determined by the achievable fuel temperature, which drives the temperature of the propellant. NTP reactors operate at the extremes of material capabilities and engineering design. This is necessary to achieve the highest reactor temperature to maximally add energy to the propellant and accelerate the propellant to the highest exhaust velocity, which translates to highest specific impulse. Low molecular weight propellant also provides higher efficiency for the NTP system. A second factor of performance is the thrust-to-weight ratio of the propulsion system – which affects vehicle acceleration and achievable mission energy. The use of the NTP system can provide unparalleled performance for in-space transportation. The combination of the high thrust capability with high propellant efficiency makes it the ideal propulsion system for performing quick and powerful maneuvers in space.

While the use of hydrogen as the propellant provides the NTP system with the highest propellent efficiency, it creates numerous technical challenges for the integrated system. 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 and propellant flow control. Because hydrogen is not only the propellant, but also the reactor coolant, any imbalance in hydrogen flow could lead to hot spots in the engine and fuel, which can cause significant thermal balancing issues. In addition, high temperature hydrogen corrosion of fuel elements and coatings in the core is a major consideration for materials development. Corrosion rate increases with fuel temperature and can significantly limit the engine lifetime. Finally, while the low molecular weight of the hydrogen may be a benefit for the efficiency of the engine, it comes at a cost of requiring extremely large propellant storage volume, pushing the boundaries of element design and resulting in significant challenges to launch vehicle integration. Related to the storage volume of the hydrogen is the extreme storage temperature required. Typically, hydrogen is stored in a liquid state (LH2) to maximize the storage capabilities of designed elements; however, this creates a serious challenge to manage the cryogenic propellant at hydrogen's boiling temperature to keep losses due to boil-off at a minimum.

For the system and vehicle integration perspective, the volume requirement for hydrogen fuel means that most elements will be volume constrained by the launch vehicle. For high energy missions, large quantities of hydrogen are required to achieve the energy need. While both SEP and NEP systems can increase their power to increase energy delivery, NTP systems are currently limited by material thermal limits. So more elements are required to be assembled into a single spacecraft to provide the vehicle with the performance required for higher energy missions. This creates significant integration complexity, as more structural docking interface and propellant management components are required to be assembled in space. These additional elements also result in a negative feedback loop with the rocket equation, where more element dry mass means more propellant is needed to perform the same maneuver, resulting in more propellant demand and more element dry mass. Pre-deployment of propellant and/or staging of expended propellant tank elements can provide relief to this challenge, but it adds operational complexity to the integrated architecture.

3.4. *Chemical Propulsion*

The general concept of utilizing chemical propulsion systems for Mars missions dates to before the Apollo Program. While the NTP system utilizes the nuclear reaction to heat the propellant to generate thrust, a chemical propulsion system relies on the chemical combustion between the fuel and oxidizer, typically liquid oxygen (LOx). This means carrying two types of fluids and the associated propellant management systems to facilitate the combustion. To maximize the performance of the system, hydrogen fuel is preferred for the same reasons that led to its preference for the NTP system, but it also results in the same challenges that the NTP system faces. The utilization of liquid methane (LCH4) as the fuel for in-space propulsion sacrifices performance to mitigate the impact of these challenges. The reduction in performance will ultimately depend on engine design and specific implementation.

Regardless of the chosen propellant, the engine's performance is limited by the combustion reaction and thus will never reach the level of the NTP system. However, chemical propulsion system's simplistic nature (and the decades of operational experience) makes it an attractive option. The lower performance of the engine does mean that a large number of elements will be required to be aggregated and assembled in space for even the minimum energy missions. This creates significant operational complexity when the elements are constrained by the launch vehicle's capability.

For roundtrip Mars missions, utilizing chemical propulsion systems will be a challenged without pre-deployment of propellant at the destination and/or in situ propellant production and refueling. The operational complexity of these additional elements makes the integrated architecture unattractive from an overall risk and implementation perspective. However, considering the emerging commercial super heavy-lift capabilities, chemical propulsion systems must be reevaluated across a range of concepts that leverage reusable launch vehicle capabilities to reduce the cost of launching the large quantities of transportation elements needed for missions of various duration. Figure 7 shows high level concepts of various chemical propulsion systems in the context of the continuous mission duration trade space.

4. Summary

As mission duration decreases, the total energy required to perform a roundtrip Mars mission increases exponentially, significantly increasing Earth-launched mass. For both high thrust propulsion systems (NTP and chemical propulsion), the total mass required is at the mercy of the exponential nature of the rocket equation. For the hybrid systems (SEP and NEP), for any given power level, there is a limit to how much energy the low thrust system can produce. Thus, to enable higher energy missions, either the chemical part of the hybrid system must be more heavily utilized or the power level must be increased. Mass curves for the low thrust systems are, therefore, also exponential in nature as mission duration is shortened.

Each of the transportation systems has its own strengths and weaknesses, especially in the context of the continuous mission duration trade space. One of the challenges associated with the transportation system option decision is the desire to down-select a single transportation system without fully appreciating the integrated trade space and impact of that decision. The two nuclear transportation options provide compelling benefits for the higher energy missions due to their higher performance but carry the penalty of the development timeline and cost, and these drawbacks can be significant if the desired missions are the longer duration, lower energy missions. Similarly, forcing the higher energy missions on the lower performance, but less complex, non-nuclear options means that the inherent weakness is amplified to result in substantially more challenging mission concepts. Having the knowledge and understanding of the integrated transportation trade space is paramount to selecting the appropriate transportation system for the appropriate desired mission.

Of course, these four transportation options are not inherently mutually exclusive options. The chemical propulsion system is already a key component of both the NEP and SEP hybrid systems, the NEP and SEP systems share commonality in the electric thruster, power generation, and power distribution components, and the NTP and chemical propulsion system have synergy in the propellant storage, management, and distribution subsystems. Just like the NEP and SEP hybrid concepts where dissimilar systems are brought together to complement each other, different transportation systems can be combined to achieve more than the sum of the components in the context of the different energy demand for a roundtrip Mars mission. It is the intention and goal of ESDMD's Strategy and Architecture Office to continue to explore and evaluate various transportation options to meet the needs of the Moon-to-Mars Objective.

References

[1] NASA's Moon To Mars Strategy and Objectives Development, NP-2023-03-3115-HQ (Mar. 2023).
URL https://go.nasa.gov/3zzSNhp

[2] Artemis I - About The Mission (2022).
URL https://www.nasa.gov/specials/artemis-i/

[3] Mission Duration, techreport NASA-STD-3001 Technical Brief. OCHMO-TB-007., National Aeronautics and Space Administration. Office of the Chief Health and Medical Officer. (Mar. 2023).
URL https://www.nasa.gov/sites/default/files/atoms/files/ochmo-tb-007_mission_duration.pdf

[4] D. A. Craig, N. B. Herrmann, P. A. Troutman, The Evolvable Mars Campaign - Study Status, in: 2015 IEEE Aerospace Conference, IEEE, 2015, 2015-8.0101. doi:10.1109/aero.2015.7118925.

[5] K. Goodliff, P. A. Troutman, D. A. Craig, N. B. Herrmann, Evolvable Mars Campaign 2016 - A Campaign Perspective, in: AIAA SPACE 2016, American Institute of Aeronautics and Astronautics, 2016. doi:10.2514/6.2016-5456.

[6] P. R. Chai, R. G. Merrill, M. Qu, Mars Hybrid Propulsion System Trajectory Analysis, Part I: Crew Missions, in: AIAA SPACE 2015 Conference and Exposition, American Institute of Aeronautics and Astronautics, 2015, AIAA 2015-4443. doi:10.2514/6.2015-4443.

[7] L. M. Burke, M. C. Martini, S. R. Oleson, A High Power Solar Electric Propulsion - Chemical Mission for Human Exploration of Mars, in: 50th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, American Institute of Aeronautics and Astronautics, 2014. doi:10.2514/6.2014-3719.

[8] P. R. Chai, R. G. Merrill, M. Qu, Mars Hybrid Propulsion System Trajectory Analysis, Part II: Cargo Missions, in: AIAA SPACE 2015 Conference and Exposition, American Institute of Aeronautics and Astronautics, 2015, AIAA 2015-4444. doi:10.2514/6.2015-4444.

[9] R. Hofer, A. Gallimore, High-Specific Impulse Hall Thrusters, Part 1: Influence of Current Density and Magnetic Field, Journal of Propulsion and Power 22 (4) (2006) 721–731. doi:10.2514/1.15952.

[10] R. Hofer, A. Gallimore, High-Specific Impulse Hall Thrusters, Part 2: Efficiency Analysis, Journal of Propulsion and Power 22 (4) (2006) 732–740. doi:10.2514/1.15954.

[11] M. L. McGuire, S. R. Oleson, L. Burke, S. McCarty, J. M. Newman, M. Martini, D. Smith, NASA GRC Compass Team conceptual point design and trades of a hybrid Solar Electric Propulsion (SEP)/Chemical Propulsion Human Mars Deep Space Transport (DST) Vehicle, in: 2018 AIAA SPACE and Astronautics Forum and Exposition, American Institute of Aeronautics and Astronautics, 2018. doi:10.2514/6.2018-5141.

[12] P. Chai, R. G. Merrill, M. Qu, P. D. Kessler, R. T. Joyce, Sensitivity Analysis of Hybrid Propulsion Transportation System for Human Mars Expeditions, AIAA SPACE 2017 Conference and Exposition, Orlando, FLdoi:10.2514/6.2017-5283.

[13] P. Chai, R. G. Merrill, M. Qu, H. Shen, Integrated Optimization of Mars Hybrid Solar-Electric/Chemical Propulsion Trajectories, in: 2018 AIAA SPACE and Astronautics Forum and Exposition, American Institute of Aeronautics and Astronautics, 2018, AIAA 2018-5346. doi:10.2514/6.2018-5346.

[14] P. R. Chai, R. G. Merrill, K. G. Pfrang, M. Qu, Hybrid Transportation System Integrated Trajectory Design and Optimization for Mars Landing Site Accessibility, in: AIAA Propulsion and Energy 2019 Forum, American Institute of Aeronautics and Astronautics, 2019, AIAA 2019-3961. doi:10.2514/6.2019-3961.

[15] S. Oleson, L. Burke, E. Turnball, B. Kamhawi, B. Faller, W. Johnson, N. Lantz, B. Dosa, C. Schmid, Z. Zoloty, D. Smith, T. Packard, A. Colozza, J. Gyekenyesi, J. Fittje, A 1 MW Solar Electric and Chemical Propulsion Vehicle for Piloted Mars Opposition Class Mission, in: AIAA ASCEND 2023, American Institute of Aeronautics and Astronautics, Las Vegas, NV, 2023.

[16] L. M. Burke, S. R. Oleson, Z. C. Zoloty, D. A. Smith, Combined 1-MW Solar Electric and Chemical Propulsion for Crewed Mars Missions, in: AIAA ASCEND 2023, American Institute of Aeronautics and Astronautics, Las Vegas, NV, 2023.

[17] D. Gray, Y. Hahn, Maneuver Analysis of the Cassini Mission, in: Guidance, Navigation, and Control Conference, American Institute of Aeronautics and Astronautics, 1995. doi:10.2514/6.1995-3275.

[18] R. T. Mitchell, The Cassini Mission at Saturn, in: 57th International Astronautical Congress, American Institute of Aeronautics and Astronautics, 2006. doi:10.2514/6.iac-06-a3.2.01.

[19] S. Oleson, Electric Propulsion Technology Development for the Jupiter Icy Moon Orbiter Project, in: 40th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, American Institute of Aeronautics and Astronautics, 2004. doi:10.2514/6.2004-3449.

[20] M. LaPointe, S. Oleson, E. Pencil, C. Mercer, S. D. Stefano, J. Gilland, L. Mason, MW-Class Electric Propulsion System Designs for Mars Cargo Transport, AIAA SPACE 2011 Conference and Exposition, Long Beach, Californiadoi:10.2514/6.2011-7253.

[21] S. R. Oleson, L. Burke, L. Dudzinski, J. Fittje, L. S. Mason, T. Packard, P. Schmitz, J. Gyekenyesi, B. Faller, A Combined Nuclear Electric and Chemical Propulsion Vehicle Concept for Piloted Mars Opposition Class Missions, in: ASCEND 2020, American Institute of Aeronautics and Astronautics, 2020. doi:10.2514/6.2020-4055.

[22] T. Polsgrove, M. A. Simon, J. Waggoner, D. V. Smitherman, R. L. Howard, T. K. Percy, Transit Habitat Design for Mars Exploration, in: 2018 AIAA SPACE and Astronautics Forum and Exposition, American Institute of Aeronautics and Astronautics, 2018, AIAA 2018-5143. doi:10.2514/6.2018-5143.

[23] C. R. Joyner, M. Eades, D. Hanks, T. Jennings, T. S. Kokan, D. J. Levack, C. Reynolds, NTP Design Derivatives and Enhancements for Lunar and Mars Missions, in: AIAA Propulsion and Energy 2019 Forum, American Institute of Aeronautics and Astronautics, 2019, AIAA 2019-4453. doi:10.2514/6.2019-4453.

[24] C. B. Reynolds, J. F. Horton, C. R. Joyner, T. Kokan, D. J. Levack, Applications of Nuclear Thermal Propulsion to Lunar Architectures, in: AIAA Propulsion and Energy 2019 Forum, American Institute of Aeronautics and Astronautics, 2019, AIAA 2019-4032. doi:10.2514/6.2019-4032.

[25] M. G. Houts, D. P. Mitchell, T. Kim, W. J. Emrich, R. R. Hickman, H. P. Gerrish, G. Doughty, A. Belvin, S. Clement, S. K. Borowski, J. H. Scott, K. P. Power, The NASA Advanced Exploration Systems Nuclear Thermal Propulsion Project, in: 51st AIAA/SAE/ASEE Joint Propulsion Conference, American Institute of Aeronautics and Astronautics, 2015. doi:10.2514/6.2015-3772.