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ANALYSIS OF ELECTRIC PROPULSION PROPELLANT TYPE FOR CREWED MARS MISSIONS

William Hurley[*], **Jacob Simmonds**[†], and **Richard Hofer**[‡]

An analysis of alternative Hall thruster propellants to xenon for a crewed Mars mission using nuclear electric/chemical propulsion is performed. Xenon, the preferred propellant when considering storage density and Hall thruster performance, is not readily available in the quantities needed for this mission, creating significant mission planning challenges. Therefore, this analysis considers the use of more available propellants and multiple storage strategies. This includes gaseous and cryogenic storage of krypton and argon and solid storage of zinc. The 1.8 MW Compass vehicle design and mission planning for a 2039 opposition class mission to Mars is utilized as a reference. A framework is developed to re-design the reference vehicle and recalculate the mission for each propellant type. The results indicate that cryogenic krypton or argon, and zinc may be feasible replacements to xenon for this mission. This result is largely driven by the comparable storage density/tank weight of these propellants to standard xenon. The low storage density of gaseous krypton and argon, coupled with launch vehicle mass/volume requirements, leads to the inability to meet mission requirements. The key limitations of this analysis and outstanding development challenges are discussed in the context of the EP system and each alternative propellant. The framework developed in this work is extendable to analyze any Hall thruster propellant for crewed Mars missions.

INTRODUCTION

NASA has outlined a strategic initiative for performing crewed Mars missions by 2040. The most favorable Mars architecture from a propulsive standpoint is a "conjunction" class mission. This mission class requires a ~ 500 day stay on the Martian surface to allow the orbits of Earth and Mars to re-align before returning home. Although this minimizes the necessary momentum change or "delta-V" required, the long mission duration (greater than 3 years), and extended stay on the Martian surface pose potentially large risks to the health of astronauts.[1] In an effort to minimize risks to the astronauts, NASA has pushed towards higher delta-V "opposition" class missions, which consist of a short 30 day Martian stay and an overall length of $\sim$ two years. Opposition class missions therefore necessarily place a greater stress on the propulsion system capability due to the larger delta-V requirements.

A few promising propulsion system options for these mission types include: chemical, nuclear thermal, and a hybrid nuclear electric propulsion (NEP)/chemical propulsion system. The NEP/chemical hybrid system offers unique flexibility and mass/power savings due to the multi-specific impulse system. As a result, a number of studies have looked at the feasibility of this propulsion architecture for crewed Mars missions. Most recently, this includes the NASA commissioned Mars Transportation Assessment Study (MTAS),[2] a National Academy of Sciences report,[3]

[*]Doctoral Candidate, Aerospace Engineering, University of Michigan

[†]Formerly with the Jet Propulsion Laboratory. Present address: Senior Propulsion Engineer, Vast, Long Beach CA

[‡]Principal Engineer and Supervisor, Electric Propulsion, Jet Propulsion Laboratory, California Institute of Technology

and many other detailed analysis[4–6] to name a few. Most of these studies consider using an array of high power (∼ 100 kW strings) xenon Hall thrusters for the electric propulsion system, although other options, such as lithium-propelled magnetoplasmadynamic thrusters, are also being studied by NASA. Hall thrusters are now the most widely flown type of electric propulsion system, with decades of flight heritage, but limitations in the maximum power thruster (100-200 kW) may prove to be too limiting when the total system power exceeds a megawatt. A high power array (∼ 60 kW total) of xenon Hall thrusters will also be used for the Power and Propulsion Element (PPE) as a part of NASA's Lunar Gateway.

Xenon has been the optimal propellant of choice for Hall thrusters for a number of reasons. First, xenon has a large ionization cross section and atomic mass, leading to high system efficiency and thrust to power ratios. Furthermore, xenon stores extremely dense (1768 kg/m^3) at moderate pressures, ∼ 1500 psi, which is essential for reducing the mass of propellant tanks. With that said, xenon is an extremely rare gas in the Earth's atmosphere, making it expensive and hard to acquire in large quantities. Indeed, the amount of xenon required for the Hall thrusters in human Mars missions, ∼ 250,000 kg,[7] far exceeds the 2023 world annual supply of 71,000 kg.[8] This gap in propellant supply could make xenon Hall thrusters unfeasible for these missions.

As a result of this gap in supply, there is a need to access the feasibility of using alternative propellants with higher natural abundance. This includes gases like krypton or argon, or solids like zinc. Krypton and argon are alternative noble gases that are now widely used for the Hall thrusters in the SpaceX Starlink constellation.[9] We choose to investigate zinc propellant for the Hall thruster over other possible metallic propellants (e.g., iodine or bismuth) because of its low toxicity and moderate sublimation temperature (400 C).[10] Furthermore, Starlight Engines is actively developing a zinc thruster for near-term flight opportunities.[11] We also consider cryogenically storing conventional gases like krypton and argon. Cryogenic liquids offer enhanced storage density over their gaseous counterparts but introduce challenges with long-term storage and fluid management. These challenges must also be solved for the chemical system, which assumes long term zero boil off storage technology of liquid oxygen/liquid methane (LOX/LCH4).[7]

For each alternative propellant, we analyze the key changes required to recalculate a NEP/chemical Mars mission. These changes include alternate tank sizes/storage pressures, heaters and coolers for solids/cryogens, and Hall thruster efficiency operating on each propellant. We make modifications to the Compass hybrid NEP/Chemical Vehicle 1.2,[7] utilized in the MTAS, and recalculate the baseline mission with vehicle modifications to assess the impact of each alternative propellant.

This paper is organized in the following way. First, we outline the challenges of xenon acquisition at the scale for Mars missions. Then, we describe our analysis approach to recalculate the baseline Compass design/mission for each propellant. Lastly, we present the key results, which includes an overview of key limitations of this analysis and outstanding challenges.

CHALLENGES WITH XENON ACQUISITION

Xenon is the optimal propellant of choice for Hall thrusters when considering flight heritage, device performance (efficiency and life), and storage density. Long lifetime magnetically shielded Hall thrusters have achieved efficiencies greater than 70% on xenon, which translates to higher thrust for a given power level and mass flow rate.[12] Xenon also stores as a dense super-critical fluid at moderate pressures (∼ 1100 psi), significantly reducing the tank mass over other noble gases (krypton, argon).[13] These advantages are in large part why many studies and baseline designs for a

NEP/Chem architecture assume the Hall thrusters operate on xenon propellant.

Since xenon is a trace gas in the Earth's atmosphere (0.086 parts per million), it is expensive and difficult to acquire in the quantities needed for Mars missions. Indeed, the world supply of xenon for 2023 was 71,000 kg[8] compared to more than 250,000 kg needed for a single crewed Mars mission.[7] If xenon production stays the same, mission planners would need to acquire 25% (18,000 kg) of the xenon world supply every single year from now (2025), until 2039 to have enough for the mission. This propellant estimate does not account for xenon needed for cargo missions or ground testing of the electric propulsion devices, which further exacerbates the challenges of acquiring of enough xenon for a crewed Mars mission.

Given these acquisition challenges, one option could be to try to increase the annual xenon supply. Xenon is collected from the atmosphere as a by-product of the cryogenic distillation process of atmospheric separation units (ASU). The first stage of these plants produce purified oxygen and nitrogen, which is the main commodity. Some ASU's have additional stages that can further purify the gas and extract trace gases like xenon and krypton. Given that it is not economical to build an ASU for just trace gas production, one option to increase xenon output would be to install the additional equipment to existing ASU's without this capability.

To estimate the effectiveness of this strategy, we can approximate the maximum possible xenon produced if every ASU in the United States had the additional stages needed to purify trace gases. In 2019, the total oxygen sequestered in the US was 11,717 M kg.[14] If we assume that xenon is extracted at the same rate as the oxygen, one kg of xenon is extracted for every 593,400 kg of oxygen. Therefore, the maximum possible US xenon production is 20,000 kg. We compare the amount of xenon needed for a crewed Mars mission to the world annual production and maximum US production in Figure 1.

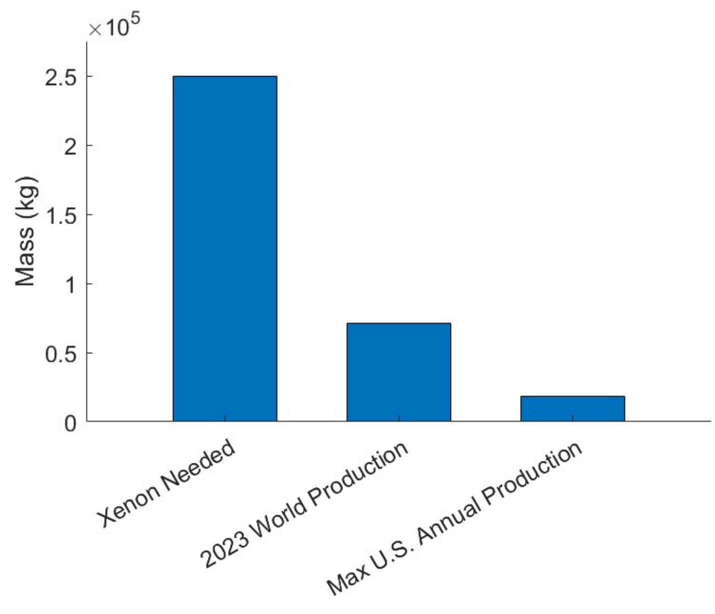

Figure 1: Comparison of the xenon required for crewed Mars missions to the 2023 word production and maximum possible US annual production. The maximum U.S. production is estimated based on the total oxygen output from atomic separation units.

If the entirety of this xenon could be used by NASA, given enough time, it may be possible to stockpile enough for a mission needing more than 200,000 kg. In reality, the programmatic challenges with securing this amount of xenon for 15 years or more make using this propellant difficult to imagine. Given these challenges, it is necessary to consider alternative electric propulsion propellant options with much higher natural abundance. In this analysis we consider three main alternatives: krypton, argon, and zinc. The world annual production for each of these gases and xenon is shown Table 1.

Propellant	World annual Supply (metric tons)
xenon	71[8]
krypton	453[8]
argon	>700,000*[15]
zinc	12,500,000[16]

Table 1: World annual production for candidate propellants. *US production in 1993.

ANALYSIS

In this section, we discuss the assumptions and analysis used to evaluate each alternative Hall thruster propellant option for crewed Mars missions. First, we summarize the vehicle and mission designed by ref. 7, which will serve as our starting point in the analysis. Then we discuss how the storage tanks are sized for each propellant, which emphasizes the importance of storage density. Lastly, we outline our optimization scheme and assumptions used to recalculate the baseline mission.

Reference Vehicle/Mission Overview

For this analysis, we leverage the vehicle design and mission planning by Oleson et. al.[7] for the Mars 2039 opportunity. Here, we provide an overview of their work, and outline the key vehicle changes when incorporating an alternative Hall thruster propellant.

In this design, the Compass team modeled key components of the NEP/Chemical vehicle. The reference vehicle is split into four different modules that are launched separately and assembled in orbit. These include: the NEP module, the chemical propulsion module, the EP propellant module (known as the xenon interstage module in Ref. 7), and the astronaut's habitat. The NEP module contains the nuclear reactor, radiators, shields, a single xenon propellant tank, and all components of the EP system. The reactor has a fixed output power of 1.8 MW that powers 18, 100 kW xenon Hall thrusters. The Hall thrusters produce 84 N of thrust at a specific impulse of 2600 s. The chemical propulsion module consists of the LOX/LCH4 chemical thrusters and chemical propellant tank. The EP propellant module has two composite overwrapped pressure vessels (COPV) to house the Hall thruster propellant. Multiple EP propellant modules are arrayed together to provide the necessary total propellant. The habitat houses the crew throughout the journey, and contains all necessary life support infrastructure. A detailed summary of each of the subsystems is outlined in Table 2.

The vehicle maneuvers for this mission design are broken up into two phases that are separated by when the astronauts enter the vehicle. A detailed concept of operations is shown in ref. 7 (pg. 24). First, the nuclear electric propulsion module, chemical module, and three EP propellant modules are

Module	Inert Mass (kg)	Assembly Location	Launch Vehicle
NEP Module	59,725*	500 km	SLS
Chemical Module	14,557*	500 km	SLS
EP propellant Module	5,774*,	500 km	SpaceX Starship
Habitat	36,371**	NRHO	SLS

Table 2: Overview of the modules used in the NEP/Chem vehicle design by Ref. 7.*Mass without main EP and chemical propellant tanks. **Mass without consumables and astronauts.

launched and assembled at 500 km. Then, the LOX/LH4 chemical system performs a series of burns to boost to a nuclear safe, 1100 km orbit. At 1100 km, a propellant tanker is launched on SpaceX's Starship to refuel the chemical system. Then, the reactor is turned on, and the EP system performs a 14 month burn to spiral the vehicle to a near rectilinear halo orbit (NRHO). Following the spiral, the propellant in two EP propellant modules is nearly depleted, so they are each jettisoned. Then, the habitat is launched with SLS, and docked to the vehicle. Next, the NEP system transfers the vehicle from NRHO to meet the crew at a lunar distance high earth orbit (LDHEO). The nominal four person crew in an Orion capsule is launched with SLS to LDHEO, and the Mars transfer part of the mission begins. At this point in the reference mission, the vehicle now consists of the NEP module with an EP tank, a single EP propellant module, the habitat with the crew, and a fully filled chemical propulsion module.

With the crew onboard, the chemical system performs a trans Mars injection burn, followed by a series of NEP burns to accelerate and decelerate to Mars. The chemical system is used to capture into Mars' orbit, and the entire vehicle docks with the Mars lander. The Mars lander is assumed to have been previously delivered during an un-crewed cargo mission that simultaneously serves as a risk-reduction demonstration of the NEP system. Following a 30-day Martian stay, the chemical system performs a final trans Earth injection burn before being jettisoned. Next, a series of NEP burns initiate a Venus gravity assist, and then capture back into Earth's orbit at LDHEO. This ends the mission.

This vehicle mass profile and mission have been optimized for xenon gas. Once we depart from the reference mission, either by changing the propellant type or operating at a different EP specific impulse, the required propellant mass changes. In turn, this alters the size and amount of propellant tanks needed. In the following section, we outline the sizing methodology used to re-design the EP tanks for each considered propellant.

Propellant Tank Sizing

Each Hall thruster propellant considered in this analysis stores differently, changing the design of the propellant tanks. Therefore, we must develop a tank sizing methodology to alter the reference mission. Here, we first discuss the storage density for each of the propellants, followed by our tank sizing approach. In this study, we consider gaseous storage of xenon, krypton, and argon, cryogenic storage of krypton and argon, and solid storage of zinc. Table 3 outlines the propellants considered and some of their key storage properties.

Table 3 highlights that, as expected, the cryogenic propellants store more densely than their gaseous counterparts. Zinc, a solid metal, stores $\sim 4\times$ denser than any other propellant consid-

Propellant	Storage Phase	Storage Temperature (K)	Storage Density (kg/m^3)
Xenon	gas	300	1768*
Krypton	gas	300	433*
Argon	gas	300	174*
Krypton	cryogenic liquid	120	2414**
Argon	cryogenic liquid	88	1394**
Zinc	solid	300	7140**

Table 3: Propellant options considered in this analysis and their storage properties. *Storage density at 1500 psi. **Storage density at 14.7 psi.

ered. For the gaseous propellants, xenon stores more densely than either krypton or argon at the same pressure, which leads to lighter/smaller tanks to hold a given mass. This advantage is in large part why xenon has been baselined for Hall thrusters in many Mars trade studies and previous commercial/science missions. While the cryogenic and solid propellants have approximately fixed density, the gaseous propellants vary significantly with pressure. As a result, there is some optimization involved to find the ideal storage pressure for each gas. This optimal pressure will not be universal, and is dependent on the total propellant needed and the size of the launch vehicle fairing. We detail the optimization process in ”Optimization and Assumptions”. Figure 2 shows the gas density as a function of pressure at 300 K for xenon, krypton, and argon.

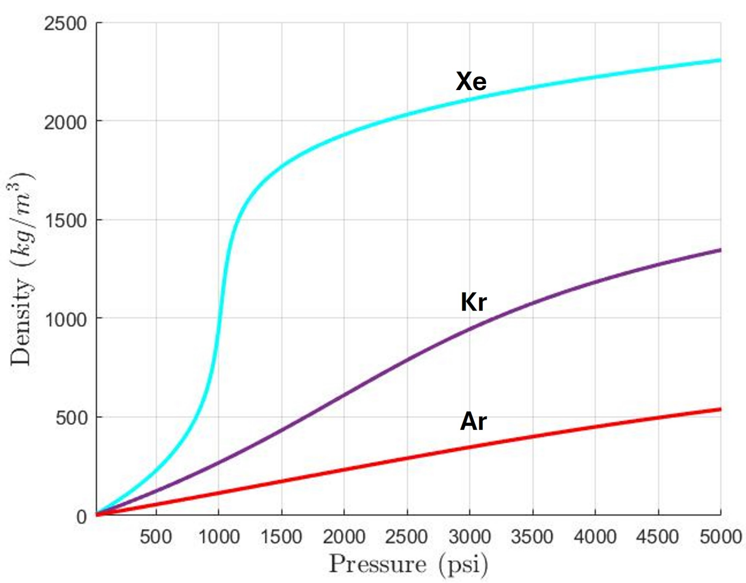

Figure 2: Xenon, krypton, and argon density at 300 K as a function of pressure. All curves are extracted from NIST.[13]

All three elements undergo a change from a pure gas state to a supercritical gas along the curves shown in figure 2. When supercritical, the density of the liquid and gas phases are the same, and are therefore indistinguishable. Xenon exhibits a sharper increase in density with pressure than krypton or argon because 300 K is much closer to its critical temperature, 290 K, than krypton (209 K), or argon (151 K). We utilize the curves in figure 2 to aid in the tank sizing and optimization process.

The propellant storage strategy, and thus tank design, is heavily dependent on the way in which the fuel is stored. Therefore, we follow three different tank design strategies for each phase considered: pressurized gas, cryogenic liquid, or solid.

Pressurized Gas: Pressurized gases cause significant stress in the tank walls, and must be carefully designed to avoid burst. To start, for an arbitrary mass m_{prop}, we first calculate the volume V_{tank} needed as

$$V_{tank} = m_{prop}/\rho_{prop} \tag{1}$$

where ρ_{prop} is the propellant density. The value of ρ_{prop} is uniquely determined by the storage pressure and temperature for each propellant. With the storage pressure P, and volume, we calculate the stress σ on a thin walled spherical tank as

$$\sigma = \frac{P r_{tank}}{2 t_{walls}} \tag{2}$$

where r_{tank} is the tank radius, and t_{walls} is the wall thickness. Equation 2 demonstrates that for a tank of radius r, the stress increases with higher gas pressures and thinner walls. The maximum stress the tank walls can handle is the material dependent yield stress σ_y. Therefore, assuming a safety factor SF, we utilize equations 1-2 to solve for the thickness of walls needed to hold a mass m_{prop} at a given pressure as

$$t_{walls} = \frac{SF * P}{2\sigma_y}\left(\frac{3}{4\pi}\frac{m_{prop}}{\rho_{prop}}\right)^{1/3}. \tag{3}$$

When it is key to minimize the tank mass, as is the case in many space applications, a composite overwrapped pressure vessel (COPV) is typically used. There are many different types of COPV's, but we will only consider the most common one which is typically referred to as a "type 3." Type 3 COPV's include a metallic liner, typically aluminum or titanium, and a composite fiber overwrap. The liner can be designed to carry some of the load ($\sim$ 10-30%), but its primary purpose is to keep the gas from escaping through the stronger, lighter, over wrapped fibers. An example COPV is shown in figure 3.

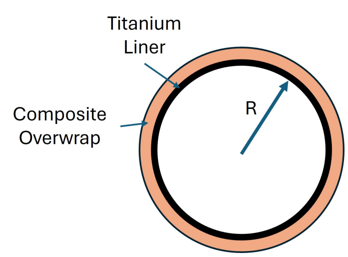

Figure 3: Example composite overwrapped pressure vessel.

To size the COPV tank, we make the following assumptions:

- **Materials:** The liner material is titanium, and the overwrap material is T1000 carbon fiber. The relevant material properties for titanium and the composite are highlighted in table 4.

- **Liner load:** At the maximum expected operating pressure, the titanium carries 20% of the load.[17]

- **Liner mechanical response:** Up to the yield strength, the titanium liner is linear-elastic. After yield, the liner is perfectly plastic.[18]

- **Safety Factor:** The safety factor (SF) is 1.5 $\times$ the maximum expected operating pressure (MEOP).

- **Structural support:** The tank needs some structural support to secure it in the EP propellant module. No specific information on the mass of the "mounting flange" is provided from the reference design (ref. 7), so we estimate it here. First, we assume most of the support structure mass is from a rectangular titanium beam around the tank center. This feature can be seen on the xenon tanks in ref. 7 in Figure 3-26. Assuming the beam cross section is 5 cm $\times$ 15 cm, and the mean radius is 2.3 m, the resulting beam mass is $\sim$ 300 kg. We add this structural weight to the total mass of each pressurized tank. We note here that the tanks in our analysis are always smaller/less massive than the reference tanks (see "Optimization and Assumptions" for further explanation), so this structural mass may be conservative for some designs.

- **Growth and margin:** The mass growth allowance (MGA) on the propellant tanks is 15 %. The MGA is an additional factor to account for uncertainty in the mass of vehicle components. Furthermore, an additional 15% is applied for margin. Both of these values are consistent with the assumptions made when sizing the xenon tanks by the reference design.[7]

Material	Yield Strength (MPa)	Density (kg/m^3)
Titanium	780	4540
Carbon Fiber Composite	2200	1730

Table 4: Material properties for the COPV liner and overwrap.

With these assumptions, we can utilize equation 3 to calculate the titanium liner thickness to carry 20% of the load (pressure), and the composite thickness to carry the remaining. A key figure of merit for any propellant storage system is the tankage fraction, defined as the ratio between the tank and propellant mass m_{tank}/m_{prop}. The tank mass is calculated directly from the radius and thickness of the titanium liner and composite. In figure 4, we plot the tankage fraction for xenon, krypton and argon at 300 K as a function of storage pressure. Note that we neglect the growth and margin factors, as well as the 300 kg structural support in figure 4 to show the best case tank fraction.

Figure 4 shows that the tank fraction is non-linear, and there is an optimal storage pressure for each gas. The tank fraction is proportional to the storage density, with xenon exhibiting superior performance.

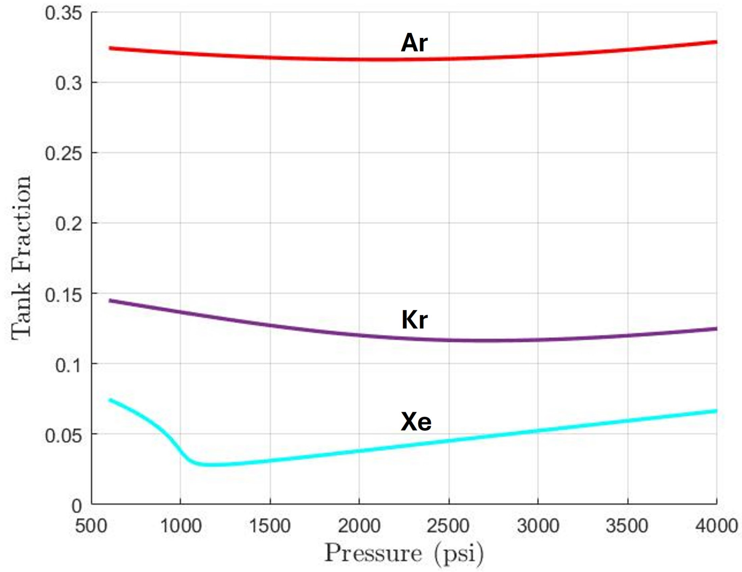

Figure 4: Best case tank fraction for pressurized xenon, krypton, and argon at 300 K.

Cryogenic Liquids: Evidenced by figure 4, the more widely available propellants argon and krypton have poor storage densities compared to xenon, leading to heavier propellant tanks. To circumvent this disadvantage, one option is to store these propellants as denser cryogenic liquids. Cryogenics are widely used in many industries, including chemical rocket propulsion. Indeed, launch vehicles like the Saturn V, Falcon 9, and Starship all utilize cryogenic gases such as oxygen (LOX) and methane (LCH4) to increase propellant storage density. Furthermore, the baseline chemical system for the Compass NEP/Chemical architecture, which we utilize in this analysis, assumes cryogenic LOX/LCH4 storage and thrusters.[7] With that being said, the Compass study and NASA acknowledged that significant technological maturation is needed to realize long term cryogenic storage and fluid management in space.

Even though this technology remains at a low technology readiness level (TRL), long duration, zero boil off cryogenic storage and fluid management is necessary for the chemical system in a NEP/chemical Mars architecture. Therefore, in this analysis, we assume these technological developments are also available to deliver cryogenic krypton or argon to the EP system. One caveat that could invalidate the assumption is that the EP propellant distribution system will have to operate for much longer burns and distribute orders of magnitude lower mass flow rates. While these differences may impose additional technological challenges, for the purposes of this analysis, we leverage the same assumptions made by the Compass vehicle to size the cryogenic tanks.

In the Compass design, the cryogenic LOX/LCH4 tank is actively cooled by space rated cryocoolers. The tank is wrapped in multi-layer insulation (MLI) to prevent heat leaks, and has slosh baffles, mixers, and a thermal vent system to manage the liquid. Similar to the pressurized xenon tanks, both a 15% growth and margin factor is added to the basic mass of the tank. After accounting for all these components and factors, the chemical tankage fraction is $m_{tank,chem}/m_{prop,chem} = 0.046$. To size cryogenic tanks for the EP propellant in our analysis, we assume the same tankage fraction as the chemical system. This assumption is likely valid because cryogenic argon/krypton store at similar temperatures and the same pressure as LOX/LCH4. One difference is that the EP tanks are sized to store $\sim$ 40,000 kg of propellant, while the chemical system houses greater than 200,000 kg. It is possible this size difference could lead to a change in the tankage fraction, but we ignore this effect for the purposes of this analysis. With the assumption of constant tankage fraction, we can directly calculate the tank weight from the EP propellant mass $m_{prop,EP}$. To summarize, the key assumptions are:

- **Tank weight:** We assume that the tankage fraction $m_{tank,chem}/m_{prop,chem}$ will be the same as for the LOX/LCH4 tanks designed in ref. 7: tank fraction = 0.046. This fraction includes the mass of coolers, flow management equipment, and tank structural support. Furthermore, this fraction includes the 15% growth and margin utilized in the Compass reference design.[7]
- **Tank shape:** The EP cryogenic tanks are assumed spherical, which is in contrast to the pill shaped reference design. This shape is chosen to be consistent with the pressurized gases.
- **Growth and Margin:** The tankage fraction includes the 15% growth and margin factors.

Solids: Solid propellants like zinc have unique benefits and challenges compared to standard gaseous propellants. Namely, zinc at standard temperature and pressure stores $\sim 4\times$ denser than xenon at 1500 psi. The high density, coupled with the ambient pressure storage, makes zinc an attractive propellant to reduce the overall vehicle mass.

Many efforts have investigated using other solids like bismuth for its high thrust to power ratios and efficiency,[19,20] but the necessary temperatures (greater than 1000 C) to melt and then sublimate this propellant make it hard to realize in a flight system. In contrast to bismuth, zinc readily sublimates at more manageable, moderate temperatures $\sim$ 400 C.[10] The propellant lines must be kept above this temperature from the propellant tanks until thruster delivery to avoid condensation/deposition. To avoid ion induced spacecraft erosion, the Hall thrusters in the reference design are placed on booms far from the propellant tanks. Therefore, this long propellant line length, which all needs to be at temperature, may pose a significant challenge to deliver a consistent flow rate of zinc to the thrusters. In the baseline design, the NEP module has $\sim$ 300 kg of components to manage and provide propellant from the xenon tanks to the thrusters. We make a conservative estimate that $3\times$ the mass of the standard xenon feed system is required to deliver zinc to the thrusters. We add this 900 kg of mass to the NEP module. Furthermore, for each tank, we estimate that an additional 150 kg of infrastructure will be needed to sublimate the zinc. This number is more than triple the mass associated with each tank's feed system in the reference design (41 kg). To summarize, the main sizing assumptions are:

- **Tank shape/sizing:** Many solid propellant delivery systems use a wire-fed scheme that negates the use of tanks. For this analysis, we assume the solid propellant will be sublimated directly from spherical tanks. We also assume that the tank walls are made of titanium, and are sized to 50 psi using equation 3. An additional 150 kg of equipment per tank is assumed needed to sublimate the zinc.
- **Propellant management mass:** We assume that an additional 900 kg will be needed to manage and deliver the heated zinc gas from the tanks to each Hall thruster. This mass

is applied to the NEP module and is 3× the mass of the flow management system of the reference vehicle.[7]

- **Structural support:** Similar to the pressurized gases, we assume 300 kg of structural support is needed to hold the tanks in the EP module.
- **Growth and Margin:** Following the standard practice of the Compass reference design,[7] we multiply the mass of all components of the tank/flow distribution system by a factor of 1.3. This allows for a 15% mass growth allowance and a 15% margin.

Optimization and Assumptions

To access the impact of utilizing more available EP propellants than xenon, we must re-optimize the reference vehicle for each case. In the Compass vehicle, xenon Hall thrusters operating at a specific impulse of 2600 s were baselined. In this analysis, we vary the Hall thruster specific impulse for each propellant type to analyze the impact on the vehicle and mission. As a result, the mass of the vehicle is variable, and we must recalculate the baseline mission to determine the amount of propellant needed. Before we outline our process to recalculate the mission, we establish key ground rules and assumptions:

- **Vehicle Sizing:** We assume that the mass assumptions made by ref. 7 for the components of the reference vehicle, outside of the EP propellant tanks and chemical propulsion tanks, remain constant. This means that the overall power, 1.8 MW is fixed. We analyze this constraint, and the limitations it poses, further in the discussion.
- **Propellant Margin:** In the reference mission, the specific impulse of the EP system is decremented by 6 % (from 2600 s to 2444 s) for all maneuvers before Mars for propellant margin. This strategy assumes the propellant margin for each maneuver is consumed during said maneuver. For the Mars return part of the mission, the 6% propellant margin is stored as non-used mass in the propellant tanks. To be consistent with ref. 7, we assume the same 6 % reduction in I_{sp} for these maneuvers in our analysis and incorporate the assumed unused propellant mass into the ”Inert Mass” shown in table 2. The reference mission follows the same strategy for the chemical system, where the I_{sp} is lowered from 360 to 351 for all maneuvers.
- **Specific impulse:** We vary the Hall thruster specific impulse for each propellant from 1500-4000 s to evaluate a wide range. In practice, the Hall thruster specific impulse, a measure of effective exhaust velocity, is controlled with the discharge voltage. With that being said, long lifetime magnetically shielded Hall thruster technology has to date been demonstrated over 200 - 800 V. Operation outside this range is possible (e.g., unshielded xenon thrusters have been operated up to 1700 V), but requires additional investment. Still, we use this voltage range as a practical bound for the I_{sp} of each propellant based on the existing technology base. For the purposes of this analysis we assess the full I_{sp} range (1500 -4000 s) for each propellant, and then evaluate the required voltage in post to determine near term feasibility.
- **Launching propellant modules:** All EP tanks are launched in pairs of two in a propellant module. Similar to the Compass study, we assume the launch vehicle for the propellant modules to be the SpaceX Starship, which has a maximum cargo weight of $\sim$ 100,000 kg.

- **EP tank Size** The EP propellant tanks must be small enough to fit in the launch vehicle fairing. Therefore, these tanks must have less volume than those in the Compass reference vehicle: 31.9 m^3.[7] We calculated this volume from the maximum stored propellant and the xenon density at 1100 psi storage pressure.

- **EP Tank Weight:** The EP tanks, when filled with propellant, must be light enough to launch. Therefore, the total mass of the EP propellant module must be less than that in the reference vehicle: 96,035 kg. Assuming the module housing is the same mass as the reference vehicle, the maximum combined mass of the EP tank and propellant is then 44,210 kg.

- **EP storage pressure** When sizing the pressurized gas tanks, we evaluate the vehicle at each storage pressure from 500-5000 psi.

- **Chemical tank sizing:** The chemical propellant tank must be optimized to have a tankage fraction of $m_{tank,chem}/m_{chem} = 0.046$. This is the same fraction assumed by the Compass vehicle.[7] This accounts for the mass of the coolers and insulation for the cryogenic LOX/LCH4, and the fluid management system. We note here that the insulation mass actually scales with tank surface area (not the propellant mass), which makes the tankage fraction shrink with larger tanks and grow with smaller ones. We neglect this potential change in tankage fraction for this analysis.

- **Maximum mass of chemical propellant:** In the reference mission, the maximum stored chemical propellant (usable) in the vehicle is $\sim$ 206,000 kg. This upper-bound is set by the LOX/LCH4 density and chemical tank volume. The tank volume is optimally sized by ref. 7 to fit into an SLS fairing. Therefore, we must abide by the 206,000 kg limit to avoid using multiple chemical tanks/re-designing the mission. We note that this limit is lower than the total amount of chemical propellant used ($\sim$ 250,000 kg) because of on-orbit re-fueling.

- **Hall thruster efficiency:** We make the strong assumption that the Hall thruster efficiency does not change as we vary the discharge voltage to hit a target specific impulse. This simplifying assumption is employed to avoid injecting an additional non-linear variable into the analysis. In reality, we see that the Hall thruster efficiency on xenon typically increases with discharge voltage/specific impulse.[21] We discuss the impacts of this assumption in section: "Limitations of this Analysis". We assume that the efficiency does change, however, between various gas types. A table of assumed Hall thruster efficiencies is shown in Table 5.

Propellant	Hall Thruster Efficiency
Xenon	60%[7]
Krypton	55%[22,23]
Argon	50%[9]
Zinc	50%[24]

Table 5: Assumed Hall thruster efficiency when operating on each gas.

- **Constant ΔV's:** We make the strong assumption that the ΔV of each maneuver in the mission of ref. 7, is constant. This avoids using trajectory optimization tools, like those developed in ref. 25, which was beyond the scope of this initial analysis. In reality, the ΔV can vary depending on the time it takes to complete each maneuver, which will inherently be variable due to the changing vehicle mass and specific impulse. With that being said, if the time for each maneuver remains similar to the baseline mission, the ΔV's are approximately the same. As a metric for this, we analyze the time needed to complete the various EP burns in transit from the moon to Mars with the astronauts on board. In the reference mission, the transit time to Mars is $\sim$ 270 days, with $\sim$ 240 days of EP burn time. Given that the EP system is burning for the majority of the transit, we take the total transit time of $\sim$ 270 day to be the burn time limit. Beyond this limit, the ΔV's of the reference mission may not be valid.

Ultimately, we need to determine how much propellant, both EP and chemical, will be required to complete the mission maneuvers defined in the Compass report (pages 21-22).[7] To calculate the propellant required for the maneuver "x", we utilize the rocket equation, defined as

$$\frac{m_{f,x}}{m_{0,x}} = exp(\frac{\Delta V_x}{I_{sp,x} g}), \tag{4}$$

where $m_{0,x}/m_{f,x}$ are the spacecraft wet mass before and after the maneuver, $I_{sp,x}$ is the specific impulse of either the chemical or EP engines, and g is the gravitational constant. The required propellant for the maneuver is calculated as

$$m_{p,x} = m_{0,x} - m_{f,x}. \tag{5}$$

We repeat this calculation for each step of the mission to estimate the total amount of EP $m_{EP,prop}$ and chemical $m_{chem,prop}$ propellant needed. This process is complicated by the fact that we can not directly estimate the initial vehicle dry mass. This is because the mass of the EP and chemical tanks are dependent on the total propellant used. As a result, we developed a scheme to ensure that the tanks are optimally sized. Once optimized, the chemical tank mass fraction is 0.046, and the EP tanks are sized to hold the propellant while minimizing the vehicle mass.

To start our solution process, we break the mission up into the two segments which are roughly separated by the arrival of the vehicle at NRHO (after the 14-month EP spiral). For a given propellant and Hall thruster specific impulse, we first find the required vehicle wet mass to complete the second part of the mission: from the moon to Earth return. To do this, we repeatedly solve the mission maneuvers in an optimizer to find the minimum vehicle mass while properly sizing the chemical tank. Since the chemical propellant is assumed to be stored in a single tank at a fixed tankage fraction, the vehicle mass is minimized through proper EP tank sizing. For all the propellants considered, this includes determining the number of required tanks and their size. Additionally, for the gaseous xenon, krypton, and argon, we must determine the ideal storage pressure. To illustrate this minimization, in figure 5 we plot the minimum vehicle wet mass near the moon (NRHO) for xenon, krypton, and argon as a function of storage pressure when the Hall thruster specific impulse is 3000 s.

As shown in figure 5, the ideal storage pressure is different for each gas, and is not necessarily at the minimum tank fraction (figure 2). The sharp steps in some of the curves result from a change in the required number of EP propellant modules. The benefits of enhanced storage density are evident in figure 5, with xenon outperforming krypton or argon. For the non-pressurized gases considered, there is a single value for the minimum wet mass at NRHO at each Hall thruster specific impulse.

Once the vehicle is sized at NRHO, we move to part one of the mission. Part one starts in LEO, and transfers the uncrewed vehicle to NRHO. The maneuvers include the component launches,

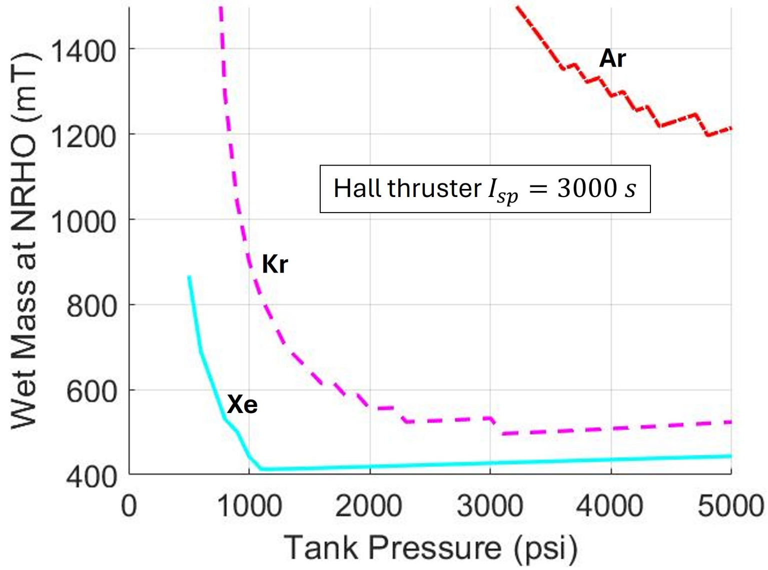

Figure 5: Vehicle wet mess at NRHO as a function of storage pressure for xenon, krypton and argon propellants. The Hall thruster specific impulse is fixed to I_{sp} = 3000 s.

vehicle assembly, a chemical burn to a nuclear safe orbit, refueling of the chemical tank, and a long EP spiral to NRHO. According to ref. 7, the ΔV for this specific EP spiral is independent of vehicle mass, specific impulse, or thrust. This is contrary to the Mars transfer part of the mission described previously, where we assume the transfer time could impact the ΔV. We iteratively solve the maneuvers to simultaneously determine the amount of EP and chemical propellant required, and number of EP tanks needed to hold that propellant. Similar to the Mars transfer part of the mission, additional EP propellant modules are added (each with two tanks) as needed. We utilize the same EP tank design as the optimized vehicle at NRHO for the LEO to NRHO transfer. Furthermore, the chemical tank size/weight remains the same as that in the Mars transfer optimization. The chemical tank is refilled during the LEO to NRHO transfer, so the maximum stored chemical propellant – which determines the tank size – is at NRHO.

RESULTS AND DISCUSSION

Here, we present key results of our analysis. To generate these results we re-sized the reference vehicle designed by ref. 7 for each Hall thruster propellant type and specific impulse. The vehicle was re-sized subject to the assumptions outlined in ”Optimization and Assumptions” and the tank sizing outlined in ”Propellant Tank Sizing”. First, we compare our analysis for xenon propellant to the reference vehicle for validation. Next, we highlight the impact of alternative EP propellants on the crewed Mars reference mission. Lastly, we discuss the limitations of our analysis and some key outstanding challenges for realizing the benefits of alternative EP propellants.

Comparison to the Reference Design

Before we assess the impact of alternative Hall thruster propellants for crewed Mars missions, we first compare the output of our vehicle sizing scheme for xenon to the baseline design of ref. 7.

In table 6 we compare key metrics of the optimized xenon vehicle at I_{sp}= 2600 s to the reference vehicle.

Parameter	Reference Vehicle (ref. 7)	This Analysis	Δ
EP I_{sp}	2600	2600	-
Total EP tanks	7	7	-
Earth Return Mass (kg)	108,974	107,861	1,113
Xenon Tank Weight* (kg)	2,378	2,007	371
$m_{EP,prop}$ (kg)	256,363	250,290	6,073
$m_{chem,prop}$ (kg)	251,853	244,050	7,803
EP Δt_{burn} Moon $\rightarrow$ Mars (days)	$\sim$ 240	236	4

Table 6: Comparison of the reference vehicle to our analysis for xenon propellant with the Hall thrusters at I_{sp} = 2600 s.*Tank weight includes growth and margin factors for clarity. In the ref. 7, the margin weight only appears at the systems level.

As evidenced by table 6, our analysis yielded a vehicle that is close to the reference, but slightly less massive. We see there is a 1,113 kg difference (1%) between the reference vehicle and our analysis upon Earth return. The difference in mass is a direct result of the lighter xenon tanks sized in our analysis. Indeed, since there are three xenon tanks upon Earth return, the total xenon tank mass difference is exactly equal to 3$\times$371 = 1,113 kg. The difference in tank mass results in a $\sim$2% difference in the total EP ($m_{EP,prop}$) and chemical propellant ($m_{chem,prop}$) used.

There are a couple reasons why our analysis yielded lighter tanks. First, in our simple analysis we assumed ideal spherical tanks, which reduces wall stress and thus tank mass compared to the "door-knob" shaped design in the reference vehicle. Second, we found that a xenon storage pressure of $\sim$ 1200 psi was more optimal (reduced weight) at 2600 s specific impulse than the 1100 psi used in the reference vehicle. Lastly, and potentially most important, we may have underestimated the mass of the structure used to secure the tank in place. We could have used the structural mass to "tune" the tanks to the correct mass, but since our light tanks are due to a multitude of coupled reasons, this would likely give un-physically high values. All that being said, the purpose of this analysis is to compare results as we change the EP propellant. Therefore, even though the tank sizing is slightly different, our analysis methodology yields a vehicle close enough to the reference case for our purposes.

Results for Alternative EP propellants

In this section, we present the results for each EP propellant considered in this work. These include the pressurized gases xenon, krypton, and argon, the cryogenic liquids krypton and argon, and the solid zinc. We use our vehicle sizing for xenon gas as a baseline to compare each alternative propellant to.

One of the key assumptions underpinning this analysis is that the ΔV for each EP maneuver in the mission remains constant. This assumption is only valid if the EP burn time for the maneuvers between the moon and Mars is less than the total Mars transit time of $\sim$ 270 days. We assess this in figure 6, by plotting the moon to Mars EP burn time as a function of specific impulse for each propellant.

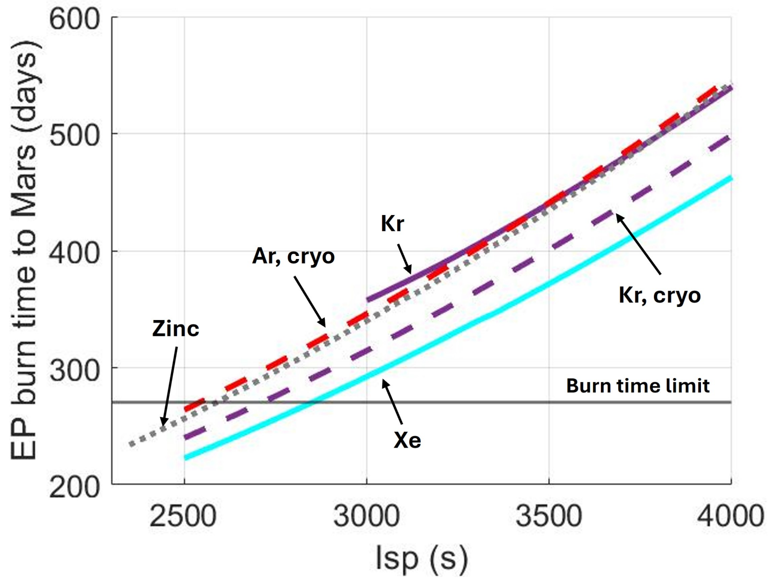

Figure 6: EP burn time to complete the maneuvers between the moon and Mars for various Hall thruster propellants and specific impulses. In the reference mission, the total transit time to Mars is 270 days, with ~ 240 days of burn time.

For each propellant, figure 6 shows that the burn time monotonically increases with specific impulse. Since we are at a fixed power and efficiency, the thrust (burn time) and specific impulse are inversely proportional. Furthermore, we see that the curve ”cuts off” for each propellant at a different lower bound specific impulse. This cutoff is due the vehicle design exceeding the limit of stored chemical propellant of $\sim 206,000$ kg. The specific impulse cutoff for zinc is 2350 s, for cryogenic argon/krypton and xenon it is 2500 s, and for krypton it is 3000 s. This requirement ensures that the single bulkhead LOX/LCH4 tank can fit into an SLS block two fairing. We note that over this specific impulse range, pressurized argon never meets the chemical propellant requirement due to its poor storage density, and therefore does not appear in figure 6. The reason this limit appears as an I_{sp} lower-bound is a direct result of the rocket equation (equation 4). Lower Hall thruster specific impulses require more propellant to complete a maneuver. In turn, the now heavier vehicle requires more chemical propellant for each burn. The specific impulse at which the limit occurs is directly related to the tankage fraction for each propellant. Gases that store dense (zinc), have less tank weight and can operate at a lower specific impulse without exceeding the chemical mass limit than poor storing gases like pressurized krypton or argon. The result is that there is a narrow range (if any) of specific impulse's for each propellant where the EP Mars burn time is less than the assumed upper bound of 270 days and the chemical propellant mass limit is observed. Based on figure 6, only pressurized xenon, cryogenic krypton and argon, and the solid zinc store dense enough to meet these parameters. We also notice that even though zinc stores more densely than xenon, the lower Hall thruster efficiency (table 5), leads to longer burn times.

Interestingly, figure 6 shows that the range of feasible specific impulse is very close to the baseline vehicle design $I_{sp} \sim 2600s$. We can understand this result by plotting, in figure 7, the number of total EP propellant tanks required for pressurized xenon, krypton, and argon. We note that the tank curves for the cryogenic propellants and zinc are similar to xenon and omitted for clarity.

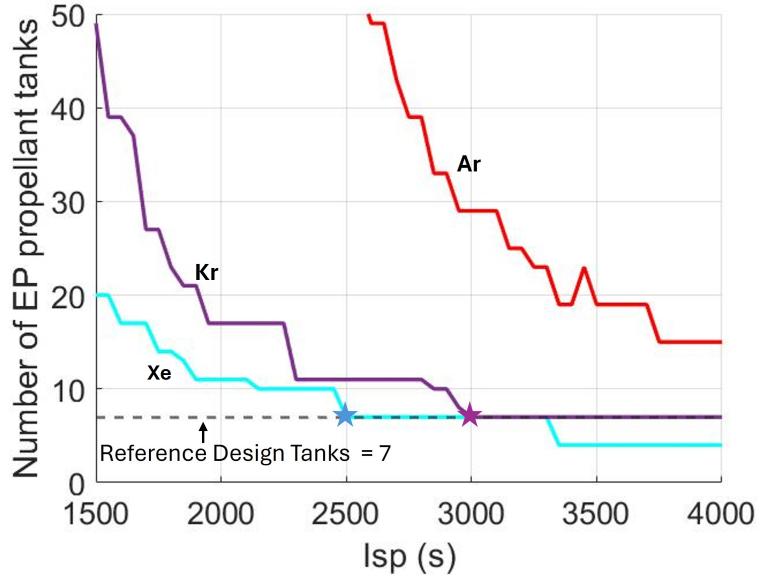

Figure 7: Total number of EP propellant tanks needed for pressurized xenon, krypton, and argon. The considered cryogens argon/krypton, and solid zinc have similar curves to xenon and are omitted for clarity.

Figure 7 shows the expected result that the total number of required EP tanks decreases with specific impulse. At I_{sp} = 1500 s, xenon requires 20 tanks, krypton requires 49, and the mission does not close for argon. As the specific impulse increases to 4000 s, the required tanks for xenon is 5, krypton is 7, and argon is 15. In all cases, the denser storing propellant requires less (or the same), number of tanks at a given specific impulse. The star on the xenon and krypton curves in figure 7 represent the lower bound in I_{sp} where the required chemical propellant is less than the baseline ($\sim$ 205,000 kg). When the chemical propellant is more than this value, the chemical tank would not fit in the launch vehicle fairing. Notably, we see that this point also corresponds to the I_{sp} where the number of tanks equals that of the reference vehicle (7 tanks). Therefore, if our analysis requires more tanks than the reference, the vehicle is heavier and the required chemical propellant is beyond the limit. That is why pressurized argon, which always requires greater than 7 tanks over this specific impulse range, never meets the chemical propellant limit requirement. In short, the transit time and chemical propellant constraints, combined with the propellant tank weight/volume limits, forces the the ”feasible” vehicle designs for each propellant to be very similar (same I_{sp} and number of tanks) as the the reference design. The fact that our analysis is heavily constrained by the reference design is one of the key limitations of this analysis and will be discussed further in section: ”Limitations of this Analysis”.

Another crucial consideration for ”feasibility”, that motivated this analysis, is each propellant’s availability. In figure 8, we plot the fraction of the world annual supply needed for each propellant to complete the mission as a function of specific impulse. Similar to figure 6, these curves show cases where the maximum chemical propellant limit is met, but not necessarily the transit time limit.

Figure 8 shows the anticipated trend that the percent of the world supply needed for each propellant decreases with specific impulse. The required fraction of the world supply is largest for xenon followed by krypton/cryogenic krypton, cryogenic argon and zinc. Pressurized argon does not ap-

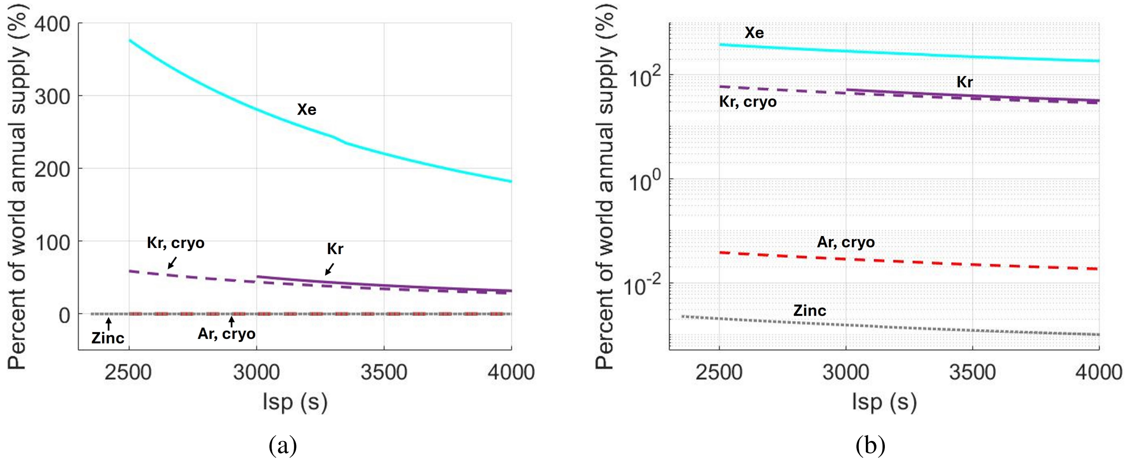

Figure 8: Percent of the world annual supply needed for each propellant to complete the mission. A log scale is used in (b) to illuminate the zinc and cryogenic argon fractions.

pear in this chart (similar to figure 6) because of its poor storage density which leads to chemical propellant requirements beyond the limit for this specific impulse range.

Figure 8 highlights that the total amount of xenon needed for this mission is prohibitive. At the 2500-2600 s specific impulse range, the amount of xenon is greater than $3\times$ the current world annual supply. Even if the xenon Hall thrusters are operated at an I_{sp} greater than 4000 s, which would take a significant increase in voltage to more than 1000 V, the amount required is $\sim 2\times$ the world supply. Again, we note that this number does not account for propellant needed in development, wear testing, or subsequent missions. Overall, the quantities of xenon needed pose significant mission planning risks, and is not a sustainable way to establish a human Martian presence.

Looking at the propellants other than xenon in figure 8, cryogenic argon and zinc require a minuscule fraction of their total supply (less than 0.05%). Cryogenic krypton requires a more manageable fraction than xenon, at $\sim$55% of its world supply, but this amount could be challenging to procure even with a multi-year acquisition strategy. When accounting for subsequent missions and thruster development testing, the amount of krypton required may quickly accumulate. Similar to xenon, these risks must be accounted for when assessing krypton as a propellant option.

Some may argue that even those these alternative propellants are available, the difficulties with using cryogenic or solid propellants presents an even larger risk to a mid 2030's NEP/chemical crewed mission with Hall thrusters than the xenon shortage. While this may be true for zinc due its current lack of flight heritage, it is not necessarily the case for the cryogens. This is because many crewed Mars design plans require zero boil off cryogenic storage of the chemical propellant. Therefore, it is reasonable to assume that the developments that will be utilized to store and manage the LOX/LCH4 could and should be utilized for the EP system. We expand further on the key outstanding challenges related to the EP system further in section: "Key Outstanding Challenges".

We summarize the main results from this section in table 7. For each propellant's feasible specific impulse range, we tabulate the required Hall thruster discharge voltage V_d, and fraction of the world supply needed to complete the mission. The feasible I_{sp} range is determined as the range where the EP burn time and chemical propellant requirements are met. This is shown in figure 6 as curves below the "burn time limit".

Propellant	Feasible I_{sp} range (s)	Required HET V_d*	% of World supply required
Xenon	2500-2850	555-720	376-304
Krypton	None	-	-
Argon	None	-	-
Krypton, cryo	2500-2700	420-490	59-52
Argon, cryo	2500	240	.04%
Zinc	2350-2550	350-415	<.01%

Table 7: Summary of the main results after sizing the vehicle for each propellant.

We make a few strong assumptions to estimate the required Hall thruster discharge voltage at each specific impulse. First, we make the same assumption as our prior analysis that the global Hall thruster efficiency is independent of specific impulse/V_d. Next, we assume that the difference in efficiency between xenon and other propellants is solely due to varying propellant ionization fraction. This has been commonly noted as the largest efficiency driver between xenon alternative propellants.[26] Lastly, we assume that the fraction of the beam current carried by each ion species (singly charged/double charged) is the same for the different propellants. With these assumptions, we use the xenon reference voltage $V_{d,Xe} = 600$ V, specific impulse $I_{sp,Xe} = 2600$ s, and efficiency η_{Xe} = 60% to estimate the $V_d - I_{sp}$ relationship for each propellant "i" as

$$V_{d,i} = V_{d,Xe}\left(\frac{I_{sp,i}\eta_{Xe}}{I_{sp,Xe}\eta_i}\right)^2 \frac{m_i}{m_{Xe}}, \tag{6}$$

where m is the atomic mass of a single molecule. Using this relationship, we calculate the Hall effect thruster voltage for the specific impulse ranges in table 7. Notably, we see that even though the ideal specific impulse range for this mission is similar between propellants, the resulting Hall thruster voltage varies significantly. The discharge voltage for xenon is $\sim$ 600 V, for cryogenic krypton is $\sim$ 450 V, for cryogenic argon is 240 V, and for zinc is $\sim$ 375 V. The main driver for these voltage estimates is the atomic mass of each propellant. Lighter gases are accelerated to faster velocities, leading to higher I_{sp} at a given discharge voltage. We see that the required voltages for each propellant do fall within the typical magnetically shielded Hall thruster voltage range of 200 - 800 V. With that being said, for the same input power, a lower discharge voltage necessarily results in a higher required discharge current. High currents have been shown to improve Hall thruster performance on alternative gases,[22] but it also could decrease lifetime of the cathode emitter and key components of the thruster such as the pole covers.

In summary, we have demonstrated, according to our assumptions, that cryogenic krypton/argon, and zinc meet the stringent requirements for this mission. In contrast, the poor storage density of pressurized krypton and argon leads to heavy tanks and the inability to meet the transit time/chemical propellant requirements. Krypton, argon, and zinc are also produced in the quantities needed for this mission, which strengthens their case over xenon. With that being said, our analysis is not without key limitations, which we discuss in section: "Limitations of this Analysis".

Limitations of this Analysis

Here, we discuss the key limitations of this analysis, and the potential impact of these on the results. The primary limitation of our analysis is constraining ourselves to the reference vehicle design and mission planning. As seen in section: ”Results for Alternative Propellants”, the the optimal vehicle design for each propellant is very close to the reference. Two main assumptions that led to this similarity were a constant power of 1.8 MW and fixed mission $\Delta V's$. A constant power is assumed so that we could utilize the vehicle mass assumptions outlined in the Compass reference design.[7] As shown in refs.27,28, the optimum specific impulse for the EP system in hybrid architectures is proportional to the power level. Therefore, it is not surprising that our analysis centered on the same specific impulse as the reference vehicle: $\sim$ 2600 s. Since the amount of propellant used is also directly tied to the specific impulse, higher power levels are more favorable to poorer storing gases like krypton and argon. Therefore, while these gases appear to be an unfeasible substitute for xenon in the Compass design,[7] they may be acceptable (at least krypton) at higher power levels.

Furthermore, since we did not incorporate a trajectory optimizer such as outlined in ref. 25, it was necessary to utilize the same $\Delta V's$ as the reference mission. To ensure the validity of this assumption, we were forced to place stringent requirements on the total EP burn time to complete maneuvers. Since the burn time is proportional to the specific impulse, this requirement was another key factor that led to a similar I_{sp} as the reference design. The end result is that our analysis deemed any propellant whose tank fraction is similar to or less than xenon, ”feasible”, and any higher ”unfeasible”. In reality, if we could optimize the power level/$\Delta V's$ for each propellant at every specific impulse a poor storing gas like krypton may be more practical. In spite of these limitations, the key conclusion of our analysis remains: Mars missions with Hall thrusters operating on cryogenic argon and krypton as well as solids like zinc are similar to xenon reference designs but may be more practical due to propellant availability.

Another limitation of this work is that we assume the Hall thruster efficiency is constant with specific impulse. For Hall thrusters, the specific impulse is typical controlled by changing the discharge voltage between the anode and cathode. Most Hall thrusters operate between 200 V and 800 V, and the system efficiency – a measure of how well the input power is converted to thrust – typically increases over this range.[21] Furthermore, the discharge voltage that corresponds to each specific impulse is a function of the propellant molecular weight. As a result, functional range of specific impulses for each propellant is more narrow than assumed in this analysis.

Key Outstanding Challenges

Here we focus on a few of the outstanding physics and engineering challenges for utilizing Hall thrusters and various alternative propellants for a crewed Mars missions.

In this analysis, we demonstrated that cryogenically storing krypton and argon propellant for the Hall thruster may be a competitive option for Mars missions. One major outstanding challenge to realizing these benefits is long term zero-boil of cryogenic storage. This technological development is also needed to store chemical propellants like LOX and LCH4. Some key development efforts to date in this area are outlined in ref. 29. Highlights include tests on the ISS of cryogenic storage and fuel transfer,[30] space rated cryo-coolers,[31] and experimentally validated modeling efforts.[32] Another challenge with using krypton and argon is that they condense at lower temperatures than xenon, making them harder to pump in vacuum chambers. Since the thermal load on the cryo-pumps increases with mass flow rate/beam power, this may exacerbate the difficulties associated with high power Hall thruster testing.

Similar to the cryogenic propellants, there are significant development challenges needed to convert and deliver room temperature solid propellants to the EP system at the scale of a Mars vehicle. Zinc readily sublimates above $\sim$ 400 C, but all propellant lines to the thruster likely will need heaters and insulation to maintain this temperature. Otherwise, the zinc propellant could condensate or deposit in the lines, which may cause catastrophic clogs. Maintaining a temperature of $\sim$ 400 C for the EP burn duration, $\sim$ two years, may also place significant stress on the heaters and propellant line materials. Furthermore, even components inside the thruster may need heaters, like the propellant distribution system (which is typically also the anode), to evaporate/sublimate zinc after thruster shut offs. While these challenges are likely not insurmountable, they will require a significant development effort to implement on the scale of a crewed Mars vehicle.

Lastly, we cannot assume that the long lifetime (greater than 20 kh) demonstrated by magnetically shielded xenon Hall thrusters[33] will directly translate to each alternative propellant. This is because the erosion rate, induced by ion bombardment, will likely change between propellant types and operating conditions.[33] Therefore, representative wear tests must be performed on each propellant for validation.

CONCLUSION

The amount of xenon required for the Hall thrusters in hybrid nuclear electric/chemical propulsion systems for crewed Mars missions (greater than 200,000 kg) is significantly greater than the world annual supply (71,000 kg). This disparity poses significant xenon acquisition challenges for a single mission, never-mind a sustainable Martian presence. Therefore, in this work, we analyze the feasibility of using the more available Hall thruster propellants krypton, argon, and zinc for these missions. Motivated by the liquid oxygen/liquid methane chemical propellant, we also consider storing krypton and argon as dense cryogenic liquids. To analyze these propellants, we develop a framework for re-designing the 1.8 MW Compass hybrid nuclear electric/chemical vehicle and then recalculating the 2039 opposition class mission. Key vehicle changes from the xenon reference case include tank size/design, alternate propellant delivery systems, and Hall thruster efficiency. When recomputing the mission, we strategically size the propellant tanks to meet the volume and mass requirements for the assumed launch vehicles. Furthermore, we place strict requirements on the EP burn times to ensure that the ΔV assumptions of the reference mission remain valid. With these assumptions, we recalculate the baseline mission for each propellant over a range of Hall thruster specific impulses. The results indicate that propellants that store densely - zinc and cryogenic krypton/argon - may be feasible replacements for xenon in this mission. Furthermore, unlike xenon, these propellants are available in the quantities required for these missions. The low storage density of gaseous krypton and argon leads to heavy tanks, and the inability to meet the mission requirements under our assumptions. We find that the stringent assumptions of our analysis drive the optimal specific impulse for each propellant to be similar to the reference design: 2600 s. The similarity to the reference vehicle is not unexpected, given that we leverage the design/mission, but is one limitation of this work that certainly does not preclude higher specific impulses closing for more generalized solutions. Indeed, it is worth re-assessing each of the propellants analyzed in this work (except perhaps pressurized argon) when removed from the constraints of the Compass vehicle design. Lastly, we outline some remaining key challenges for the EP system to realize the benefits of cryogens or solid propellant. These challenges include zero boil off cryogenic storage, and long lifetime heaters to evaporate the solid propellant. In spite of these challenges, this analysis demonstrates that a nuclear electric/chemical system using more available alternative propellants to xenon for the Hall thrusters is a promising way to help deliver humans to Mars.

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