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Cryogenic Liquid Rocket and Nuclear Thermal Rocket: A Survey of the Current Technology and a Comparison for Future Mars Missions

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ABSTRACT

The innovation of reusable rockets holds the potential to revolutionize both Mars scientific exploration and commercial tourism. Thus, this paper aims to indicate the most promising propulsion system for the Mars mission. While SpaceX proposed the cryogenic liquid rocket Starship to achieve Mars colonization by 2050, the viable nuclear thermal propulsion has been outlined by rocketry experts for space travel. This paper surveys the current state-of-the-art of both popular propulsion systems, indicating their advantages and disadvantages. Methods to increase the delivery and cost efficiency are also offered, including on-orbit refueling for cryogenic liquid rockets and various nuclear core variants for nuclear thermal rockets. Subsequently, their comparative analysis is delivered in the discussion, focusing on their transportation efficiency and cost efficiency for future mass missions. With faster travel, greater payload, and reduced propellant consumption, nuclear thermal propulsion has shown to be a brighter prospect and potential than cryogenic liquid rocket. Nevertheless, the author believes a new class of bimodal nuclear thermal electric propulsion system is the most promising option, combining the merits and complementing drawbacks of cryogenic liquid propulsion, nuclear thermal propulsion, and nuclear electric propulsion.

Introduction

With the accelerating and increasing projects for deep space missions, new reusable rockets with higher delivery and cost efficiency need to be developed. These rockets will pave the way for future technological breakthroughs and new commercial domains. As space travel will be more accessible to the general public (Musk, 2017), the impressive profit of space tourism might accelerate the global economy with the innovation of reusable and more efficient rockets.

In December 2020, SpaceX developed prototypes of Starship, which consists of fully reusable Starship spacecraft and Super Heavy rocket system. This super heavy-lift launch vehicle is powered by cryogenic oxygen and methane (Palmer, 2021), which can be harvested on Mars (Musk, 2017). This record-breaking transport has a payload capacity of 100~150 tons and a thrust of 7,590 tons-force, twice as much as the thrust of the Saturn V rocket (Palmer, 2021), the most powerful super heavy-lift launch spacecraft of the National Aeronautics and Space Administration (NASA). The Starship is a conventional cryogenic liquid rocket (CLR). However, it will be the principal vehicle of SpaceX to achieve Mars colonization (Musk, 2017). Thus, the SpaceX company plans to launch the first crewed mission in 2026 (Palmer, 2021). Due to the heavy demand on the fuel caused by the payload and structural mass, the Starship will achieve an on-orbit refueling procedure before the long-duration flight to Mars, to improve both delivery and cost efficiency (Musk, 2017).

While CLR has been the main choice for space missions for decades, many professionals have demonstrated the prospect of nuclear thermal rocket (NTR) for Mars-manned missions (Palmer, 2021). The specific impulse (I_{sp}) of the NTR, which measures the period of one pound of rocket's propellant able to deliver one pound of thrust, is twice as much as that of the cryogenic liquid propulsion (CLP) (Xie et al., 2017). This means a halved traveling time compared to the Starship. The NTP system also represents a potential reduction in the propellant cost, the launch mass construction, and the exposure time for astronauts facing space radiation (National Academies of Sciences Engineering and Medicine, 2021), with fewer challenges for engineering, supplies, and logistics (Palmer, 2021). Thus, NASA has had a joint venture with the Defense Advanced Research Projects Agency (DARPA), to develop cutting-edge nuclear thermal propulsion (NTP) (Bardan, 2023). However, the development of NTP is not smooth. The early concept of NTP dates back to the 1960s-70s when NASA and the Atomic Energy Commission (AEC) cooperated on Nuclear Engine for Rocket Vehicle Application (NERVA) programs (National Academies of Sciences Engineering and Medicine, 2021). But the outcome of this new joint research with DARPA for NTP can only be achieved as soon as 2027 (Bardan, 2023).

This paper will review the current technology of the two rocket models with their propulsion approaches, i.e., CLR and NTR, integrating concrete cases. This will be followed by a discussion, comparing both rocket types regarding their delivery and cost efficiency for future Mars missions. While the delivery efficiency benefits Mars exploration with faster scientific research equipment transportation, the cost efficiency can potentially decrease the budget for abundant launching requirements in commercial circumstances. Another viable rocket type likely to be used for Mars exploration, the Bimodal Nuclear Thermal Electric Propulsion (BNTEP), will also be discussed (Clark, 2019).

The implications of this literature review aim to indicate the possible rocket propulsion system used for future Mars missions, for both Martian exploration and tourism. This paper will therefore contribute to the development of rocket propulsion evaluation and analysis, where further review should be conducted to update the subject of rocket propulsion systems.

Literature Review

Cryogenic Liquid Rocket (CLR)

The Working Principle and The Structure of CLR

The CLR is powered by chemical propellants. These gaseous propellants at room temperature are refrigerated in cryogenic liquid form (Raj & Jeyan, 2023) to make fuel tanks smaller and lighter. A CLR often requires both chemical fuel and oxidizer. Because fuel can only burn with an oxidizer, which is generally the oxygen in the air on Earth, but absent in the vacuum environment of space. As displayed in Figure 1, the fuel/oxidizer combination will be injected firstly with high pressure by the turbo pump into the combustion chamber. This is where they get vaporized, and burned afterward by the ignition to produce thrust. These accelerated hot gases will achieve high supersonic velocity in the nozzle (Chhaniyara, 2013).

However, excessive temperature implies the issue of thermal protection. High-temperature gases can exceed 3,500K with an elevated pressure of over 30 MPa, which would damage the rocket engine's thermal resistance (Raj & Jeyan, 2023). Regarding this problem, regenerative or film cooling systems are usually integrated into the CLR's engine. The regenerative cooling system reduces the engine's exterior surface temperature with newly-injected cryogenic propellant. Before entering the combustion chamber, the cryogenic flow would pass through channels covering the thrust chamber and nozzle section. This method serves as engine cooling for thermal protection and propellant preheating for heat loss reduction, increasing the exhaust velocity by ~1.5% (Raj & Jeyan, 2023). The film cooling corresponds to the injection of subcooled rich fuel through orifices over the injector perimeter and flows subsequently near the internal walls of the combustion chamber and the nozzle (Raj & Jeyan, 2023).

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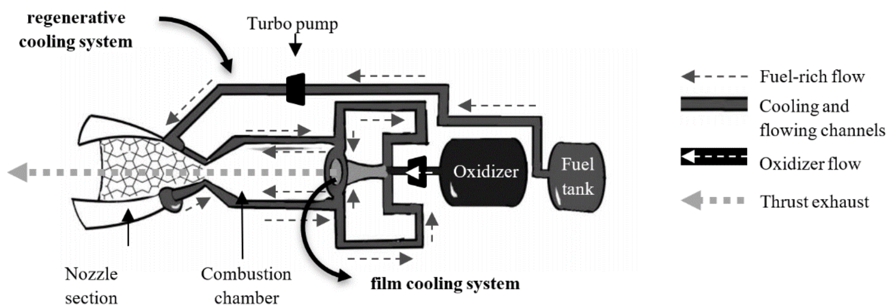

Figure 1. Schematic of the CLR engine's structure and propellant flow

Cryogenic Liquid Propellant Combinations

The efficiency of the CLR depends significantly on the propellant combination. While the popular oxidizer remains liquid oxygen (-183°C) (Chhaniyara, 2013), the performance of the rocket mostly depends on the fuel choice. Liquid oxygen (LO_2)/ Liquid hydrogen (LH_2) combination is the most utilised option. According to Raj & Jeyan (2023), "it is almost clear that LO_2-LH_2 when blend together results in producing peerless specific impulse effect". This mixture would have an incomparable I_{sp} of 455 m/s compared to that of the LO_2/Kerosene combination, representing merely 358 m/s, (Haidn, 2008). While the former combination's reaction output is water, more eco-friendly (Raj & Jeyan, 2023), the latter LO_2/Kerosene mixture troubles human health. However, LO_2/LH_2 choice demands a greater vehicle size than LO_2/Kerosene, with a mixture ratio of 4.83 and 2.77 respectively (Haidn, 2008).

Nevertheless, the best choice for the Mars mission seems to be LO_2/Liquid methane (LCH_4). "We started off initially thinking that hydrogen would make sense, but [...] the best way to optimise the cost-per-unit mass to Mars and back is to use [...] deep-cryo methalox" (Musk, 2017). LO_2/LCH_4 combination requires a medium fuel tank compared to both options aforementioned, with a mixture ratio of 3.45 (Haidn, 2008) and has an I_{sp} of 369 m/s (Raj & Jeyan, 2023). There are outstanding advantages of LO_2/LCH_4 choice, including high reusability level, cheaper propellant cost, propellant transfer convenience, and Mars propellant production. Both oxygen and methane can be produced on Mars with the Sabatier method, due to the presence of CO_2 in the atmosphere and the iced water in the soil (Musk, 2017). The in-situ resource utilization (ISRU) is crucial to send Starship back to Earth, which increases the reusability of the spacecraft, reduces cost and guarantees life support for astronauts in Mars missions (Musk, 2017). Compared to LCH_4, LH_2 is complicated to store due to its high boil-off rate during long travel. Hydrogen is liquified at a temperature of -253°C (Chhaniyara, 2013), near absolute zero of -273.15°C (Gainey, 2019).

On-Orbit Refueling Concept of CLR

Besides the appropriate propellant choice, on-orbit refueling can also maximize the delivery efficiency of CLR with increased payload transportation ability. It refers to the process when a spacecraft is refueled in LEO. Without this process, all propellants required for Martian travel have to be brought within a single rocket. In this scenario, the rocket needs to have a tremendous gross mass with extra propellant quantity to push the spacecraft out of the Earth's attraction. Therefore, the cost rises, with higher complexity and risk of launch failure. With on-orbit refueling, for the same payload, a smaller spacecraft without propellant is launched into LEO by a launch vehicle (Cirillo et al., 2010). The launch vehicle only requires a little propellant to achieve this short trajectory, where the fuel consumption stands at the highest point to detach from Earth's gravity (9.8 m/s^2). The spacecraft will be subsequently refueled by several propellant-carrying vehicles with autonomous rendezvous and docking (ARD) (Cirillo et al., 2010).

However, difficulties might be encountered during refueling flight launch and cryogenic liquid propellant transfer. The more spacecraft are launched for refueling, the higher the complexity and risks for the overall mission are estimated. The potential ARD failure for each of these rockets might interrupt the whole project (Cirillo et al., 2010). Furthermore, the evaporation of cryogenic fuel and collisions with micrometeorites and orbital debris would occur in LEO (Cirillo et al., 2010). One solution proposed was re-flying refueling tankers or launching backup rockets in case of ARD failure. While these methods can reduce the mission failure rate, the spacecraft is still exposed to large danger during the LEO parking (Cirillo et al., 2010).

Moreover, challenges would appear during propellant transfer. The zero-gravity and high radiation environment demands technologies including liquid mass gauging, propellant acquisition, and propellant storage for zero boil-off and no-vent filling (Clark, 2021). As outlined by Clark (2021), the measurement of liquid level in space can be achieved with radio frequency mass gauging (RFMG). The RFMG has also demonstrated zero boil-off of LCH_4 with a wick-and-heater technique for autogenous pressurization in Robotic Refueling Mission 3 (Clark, 2021). The propellant acquisition and vapor-liquid separation can be solved with low acceleration settling, generating an artificial gravity, which has been verified previously on the Saturn V rocket (Kutter et al., 2006). To reduce the evaporation of cryogenic fuel, NASA proposed both active and passive cooling techniques (Ma et al., 2016). The active cooling system refers to the thermodynamic vent system in the fuel tank, which controls the internal pressure with fluid mixing, cold recovery, and on-orbit venting (Ma et al., 2016). The passive method is the multilayer insulation application, which has proved that thermal boundaries between 77-300K can be reduced to $1\mathrm{W/m^2}$ (Jiang et al., 2023). While Clark (2021) outlined the “chill and fill” method for no-vent filling, when the receiver tank’s wall is cooled with a small proportion of the cryogenic flow itself and subsequently vented before injecting all propellants, this process might increase the fuel consumption and transfer cost.

Nuclear Thermal Rocket (NTR)

The Working Principle and the Structure of NTR

The thrust production of the NTR is illustrated in Figure 2. LH_2 is firstly pumped with high pressure from twin turbopump assemblies. This liquid flow will travel to the nozzle, pressure vessel, neutron reflector, and control drums, before entering the turbine (Borowski et al., 2009). This procedure serves as preheating and system cooling, raising the thermal resistance and thrust efficiency of NTP. LH_2 will subsequently pass through the reactor core where the nuclear reaction occurs, to be superheated into gas form (Borowski et al., 2009). The gas will be forced afterward through the nozzle to produce thrust (National Academies of Sciences Engineering and Medicine, 2021). To regulate the nuclear reaction, twin turbopump assemblies will manage the LH_2 flow. Control drums in the reflector region will also monitor the neutron population and reactor power level (Borowski et al., 2009).

The NTR does not require an oxidizer, as LH_2 only has a function of thrust production rather than energy production (Xie et al., 2017). While the CLP’s energy comes from the combustion, that of the NTP originates from the nuclear reaction core. The LH_2 with a low molecular weight has been a popular and efficient propellant for NTR and can achieve an I_{sp} of nearly 1,000 seconds (Xie et al., 2017). Furthermore, advanced configurations of NTR with liquid or gaseous core variants can improve the I_{sp} up to 1200 seconds (Raju, 2022). Due to the reduced rocket fuel needed, NTP has a bright future for Mars missions.

Another characteristic of the NTR is the radiation emitted. Elevated radiation dose generated during the nuclear reaction can harm the crew’s physical health and the rocket’s hardware, affecting the mechanical resistance (Gabrielli & Herdrich, 2015). Thus, appropriate shielding in the pressure vessel and increased distance between the crew cabin and the reaction core are required (Borowski et al., 2009). Hydrogen-based materials can effectively reduce space radiation, which is composed of protons and heavy ions (Moore, 2010).

Therefore, LH_2 in the cooling channels offers a protection layer, although the coating is needed to avoid hydrogen erosion (Gabrielli & Herdrich, 2015).

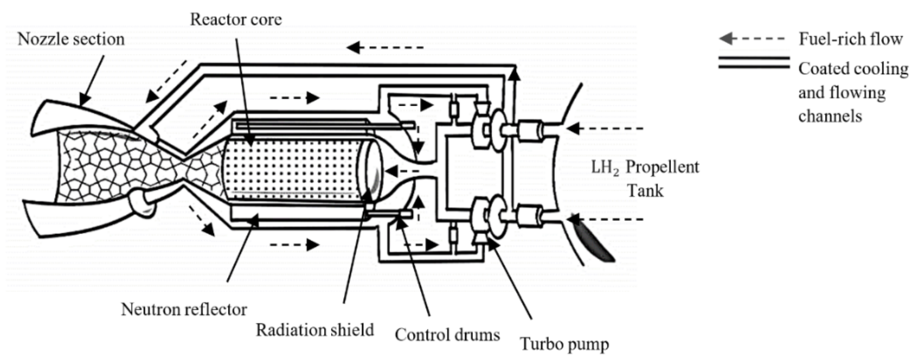

Figure 2. Schematic of NTR engine's structure and propellant flow

The Nuclear Reactions in the NTR

Nuclear reactions for NTP application can be classified into three main categories, namely fission reaction, fusion reaction, and radioisotope thermal radiation. Nuclear fission thermal propulsion has been the most studied since the late 1940s (Gabrielli & Herdrich, 2015). It refers to the process when a heavier atom, usually uranium-235, is split into 2 or more lighter isotopes after bombardment with a neutron. This reaction will release thermal energy, but also harmful beta or gamma radiation. Additional neutrons produced will create further collisions with heavy atoms in the neighborhood (Gabrielli & Herdrich, 2015). A self-sustaining reaction chain is therefore produced and must be controlled to prevent the exponential reaction as in atomic bombs (Raju, 2022).

Meanwhile, the nuclear fusion reaction has the same investigation and experiment period as the fission reaction for NTP application. Yet the fusion is not applied in commercial reactors (Gabrielli & Herdrich, 2015). In contrast to fission, fusion reaction is the process when two lighter atoms are joined into a single atom, releasing thermal energy, radiation and further neutrons. While fusion reaction requires an extreme temperature of 100,000,000 K, an identified quantum effect can decrease the condition to 1,000,000 K (Gabrielli & Herdrich, 2015). But reduplicating the Sun core's temperature (1,500,000 K) remains difficult. The deuterium/tritium (DT) fuel combination for nuclear fusion is the most accessible with current state-of-the-art. While the radioactive tritium reaction requires heavy shielding compared to the deuterium/helium-3 fusion reaction, the latter option will demand 10 times higher temperature to fuse, which is unachievable by the current technique (Pitts, 2019).

Then, radioisotope-heated thermal propulsion has been proposed as well. Thermal energy is released when a radioactive and unstable element mutates to become a stable isotope, known as radioactive decay (Gabrielli & Herdrich, 2015). The radiation and energy produced will subsequently heat the propellant. This propulsion was previously applied in an NTR during the 1960s with the Radioisotope Propulsion Technology Program, also called the POODLE project, which used polonium-210 as the nuclear fuel. (Gabrielli & Herdrich, 2015)

Considering that the nuclear fission application is the most accessible and mature with the current state-of-art, the following sections will focus predominantly on the nuclear fission thermal rocket.

The Different Fission Core Reactors of NTR

Different core reactors of NTR correspond to the various states of fission fuel. Solid core reactor is considered as the most mature system, due to its previous application on NERVA-derived engines (Gabrielli & Herdrich, 2015). As shown in Figure 3 the "graphite matrix" fuel element has a hexagonal cross-section and contains 19 axial coolant channels coated with zirconium carbide to prevent hydrogen erosion. With highly enriched U-235, each fuel element can deliver ~1 MW of thermal power (Borowski et al., 2009). Therefore, the I_{sp} of NTR has reached ~900-910 seconds and a thrust temperature of 2,700K during the NERVA program (Borowski et al., 2009). Solid fuel elements can provide fixed channels for LH_2 heating, which is easy to monitor (Xie et al., 2017). As the solid nuclear core reactor is limited by its melting point, LH_2 heating is restrained to 3,000K, confining therefore the I_{sp} of the NTR (Xie et al., 2017). Current studies focus on high-assay low enriched uranium-235 (Raju, 2022), for easier explosion risk management.

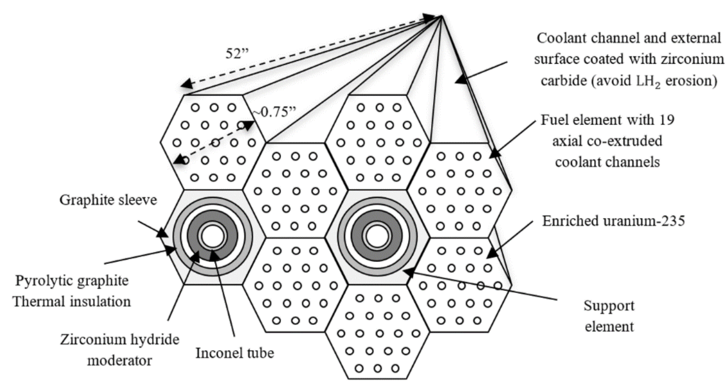

Figure 3. Composite fuel element and tie tube of Rover/NERVA

The liquid core reactor is composed of molten uranium fuel. It can heat LH_2 up to 5,000K to offer an Isp of 900-1800 seconds (Raju, 2022) but is limited by the boiling point of the core (Xie et al., 2017). Centrifugal Nuclear Thermal Rocket (CNTR) (Allen et al., 2020) uses high-speed centrifugal rotation of fuel elements to hold and manage the molten uranium fuel (Raju, 2022). However, due to the direct contact of LH_2 with the fission fuel, the nuclear fuel loss and the saturation of the propellant with heavy particles can decrease the exhaust velocity (Gabrielli & Herdrich, 2015). Furthermore, raising the thermal resistance of materials to 5,000K is another key issue to tackle.

Finally, a gaseous core can generate a high Isp of 2,500-7,000 seconds (Raju, 2022). However, as illustrated by Gabrielli & Herdrich (2015), "Due to the expected high temperatures of 40,000K beyond any material resistance, contact with the thruster solids is highly inadvisable". It is suggested to add carbon particles in LH_2 to raise the thermal radiation absorbance, but achieving a critical pressure to keep the fission plasma is another challenge. (Gabrielli & Herdrich, 2015). Yet, it is solved by Los Alamos National Laboratory with a toroidal fissile plasma in a spherical geometry (Gabrielli & Herdrich, 2015). To avoid nuclear fuel loss and reduced exhaust velocity with heavy atoms saturation, a closed-cycled Nuclear Light Bulb Reactor is proposed. While the Quartz wall will be protected from the extreme temperature of 8,000K with a flow of gaseous Neon, the exhaust velocity is limited to ~18 km/s, which is comparable to liquid core systems (Gabrielli & Herdrich, 2015).

Discussion

Comparison Between CLR and NTR for Mars Missions

The following section will compare the CLP used on the fully reusable Starship and the nuclear fission thermal rocket. Considering that the NTR needs to be firstly delivered to LEO by a first stage of CLP, due to an insufficient thrust to detach from the Earth's gravity, the discussion and the comparison would begin from the LEO.

Delivery Efficiency Comparison

Firstly, the NTR has a higher delivery efficiency than CLR. Specifically, NTP is not exposed to high risks compared to CLR in LEO, because the former does not require on-orbit refueling. For CLR, it is projected to consume 587 tons of propellant for each Starship to deliver 115 tons of payload to Mars from LEO. Thus, 5 tanker flights are necessary for on-orbit refueling (Zubrin, 2019). The more flights are required, the more risks to the overall mission are expected to fail in case of ARD problems. Furthermore, staying in LEO too long will have other potential risks, including micrometeorites and orbital debris collisions, increasing exposure to solar and galactic radiation, and rising time for crews to stay in a low-gravity environment, which is expected to have adverse consequences on human health. The launching window for Mars traveling represents a 26-month interval (Bradley, 2018). Therefore, Musk (2017) expected over 1,000 spacecraft waiting in LEO for the Mars colonial fleet departure. This will skyrocket the risks aforementioned by 1,000 times, with further complexities and difficulties in managing the overall mission architecture: in case of collision between Starships in LEO, a domino effect can be produced. Thus, NTR is safer.

Although the NTR has a longer mission span than CLR, this time can be easily reduced with effective launching schedule management. Borowski et al. (2009) have proposed the Mars Design Reference Architecture (DRA) 5.0 for the mission. It is worth noting that the mission period is expanded to two extra years, as the Cargo and Habitat Landers need to be launched 2 years before the crewed missions with NTP Mars Transfer Vehicle (MTV), to install a nuclear surface power system and in-situ resource utilization plant to produce propellant required for Mars Ascent Vehicle (MAV). Nonetheless, each mission can be projected in advance, by launching crewed MTV of the current mission and Cargo and Habitat Landers of the next mission for each Mars launching window. Consequently, apart from the first launch, the mission periods of NTR and CLR are the same.

Additionally, NTP has a viable crew transportation ability, despite an inefficient crew delivery proposed in DRA 5.0. While each Starship can contain over 100 crews in the pressurized section and can be expanded to 200 people to achieve a self-sustaining Martian civilization early (Musk, 2017), in the DRA 5.0 concept, six crews will be sent separately with three Ares V launches (CLR) to meet the NTP Mars Transfer Vehicle (MTV) pre-launch into LEO (Borowski et al, 2009). Therefore, the NTP crewed transportation is particularly inefficient compared to CLR. Nonetheless, ignoring the ancient concept of DRA 5.0, if the reusable Starship launching system with higher crew transportation capability can replace the non-reusable Ares V, and ARD with enlarged MTV pre-launched into LEO, the transportation ability will be similar for both CLR and NTR.

Finally, it is crystal clear that the delivery time from LEO of NTR is less than CLR, therefore NTR has a better delivery efficiency. This is due to an I_{sp} of NTR (~1,000 seconds) twice higher than that of the CLR (369 seconds). With an initial mass of 300 tons for both, while CLR would require 181 days to arrive on Mars, the travel from LEO to Trans-Mars Injection (TMI) will only take 160.54 days with NTR (Clark, 2019). To be more specific for the trajectory, the CLR will achieve a delta-V of 3.706 km/s, while burning during 2.2 minutes in LEO. After 181 days, the CLR engines will be activated for 1.3 minutes to decelerate by 3.0525 km/s once arrive at Mars periapsis (Clark, 2019). Afterward, a delta-V of 0.4 km/s from TMI to Mars landing will be required (Zubrin, 2019). For NTP, once arrives in LEO, the NTR engines will first be activated for 48 minutes with an exhaust velocity of 3.840 km/s to detach from LEO and the entire Earth's gravity toward Mars. After reaching Mars periapsis, a delta-V of 4.605 km/s would be obtained within 37 minutes of activation (Clark, 2019). Consequently, it is noticeable that NTR requires further time to achieve a specific velocity compared to CLR's combustion. However, the NTP can be activated for a longer time due to less propellant consumption. Although the initial mass is 300 tons for CLR and NTR, as the NTP does not require an oxidizer, the payload mass ratio is higher than CLR, which is more efficient regarding the delivery. Furthermore, with potential liquid and gaseous core variants in the future, providing an extreme I_{sp} of 7,000 seconds, this traveling time difference will be increased if high thermal resistant materials can be found.

Cost Efficiency Comparison

Afterwards, NTR has a greater cost efficiency than CLR. Although the SpaceX-typed CLR Mars mission scenario can effectively reduce the launch cost with full reusability, refueling in-orbit, and propellant production on Mars, the launching cost will still be higher than the NTR's mission architecture. Specifically, both CLR and NTR are reusable rockets, therefore the manufacturing and maintenance costs won't have a great difference. Nevertheless, on-orbit refueling will increase significantly the overall mission cost. As each launch will cost 200,000$ for the launch site (Musk, 2017), the 5 tanker launches required for each mission (Westphal & Maiwald, 2022) will have an additional 1,000,000$ of cost. These extra tankers would also increase the total cost of building extra launching systems, although Super Heavy and Tanker Starship have a life expectancy of 100 flights due to short distance travel (Westphal & Maiwald, 2022). Thus, the cost per mission of CLR will ultimately surge.

The greater CLR's mission cost can be also explained by a higher propellant consumption, although LH_2 of NTP would cost more than LCH_4 of CLP and has a high boiling rate. This is because those 5 extra tanker versions of Starship would demand more propellants to activate the booster. Furthermore, the supplementary propellant is consumed during propellant acquisition for on-orbit refueling to create artificial gravity, precisely 100 pounds of propellant per hour (Kutter et al., 2006), and during no-vent filling with the "chill and fill" method outlined by Clark (2021). Additionally, compared to CLR, the NTR requires less fuel, due to a high I_{sp} of ~900-910 seconds (Borowski et al., 2009), which indicates a halved propellant mass needed to bring. Furthermore, the NTR has a low LH_2 flow rate of 73.38 kg/s compared to that of CLR (36.67 kg/s) (Hanes, 2016), rising consequently the payload mass and reducing the cost per ton for Mars delivery.

Furthermore, the greater cost of CLR is caused by the extra oxidizer required. For the combustion of CLR, more propellant is indispensable to be produced on Mars and to be brought for the round trip. After arriving on Mars, both CLR and NTR need to be refueled with ISRU. With the presence of 5 million km^2 of ice and 25 trillion metric tons of CO_2 on the red planet, LO_2 and LCH_4 can be produced with Sabatier and electrolysis methods: $2H_2O + CO_2 \rightarrow 2O_2 + CH_4$ (Musk, 2017). For NTR, only the electrolysis process is required to generate LH_2, which reduces the complexity of ISRU. Moreover, Mars's mission will use photovoltaics to produce the necessary power. Thus, more energy, time, and panels are required for CLR than NTR for propellant liquefaction and storage. These solar panels will occupy extra payload mass, with lower space for crews or mission equipment, raising therefore the cost of delivery.

Consequently, both CLR and NTR are viable spacecraft for Mars missions. While CLR is already a mature concept for space tourism mission architecture, the NTR can surpass CLR in the future due to a higher potential delivery and cost efficiency.

Bimodal Nuclear Thermal Electric Propulsion (BNTEP)

Despite the bright prospect of the NTP, the efficiency of this propulsion system can be further improved by integrating nuclear electric propulsion (NEP) for Mars missions: New Class of Bimodal NTP/NEP with a Wave

Rotor Topping Cycle (Gosse, 2023). The NEP uses the nuclear fission reaction to generate electric power (Palmer, 2021). This energy will be subsequently used to accelerate the discharge of electrons and transform neutral gases into ions with ionizing collisions (Loeb et al., 2015). NEP has an extremely high I_{sp} of >1,000 seconds but a relatively low thrust and flaws regarding mass-to-power ratios. Moreover, the conversion rate from thermal to electrical energy is merely 30-40% under ideal conditions (Gosse, 2023). Despite the limits of pure NEP, while having complementary traits of NTP, the bimodal NTP/NEP will have outstanding efficiency. The bimodal approach will raise the I_{sp} to 1800-4000 seconds (Gosse, 2023). Considering that both NTP and NEP have insufficient thrust to detach from the Earth's surface, a CLP is still required. Thus, this section will offer a better propulsion alternative, namely BNTEP, taking advantage of the benefits of CLP, NTP, and NEP, with maximized delivery and cost efficiency for future Mars exploration and tourism.

The Merits of BNTEP

The NEP's particularities can solve the NTP's crucial drawbacks, as the electricity generated by the NEP is very beneficial to the latter's propulsion system. Although NTP is fuel-efficient compared to CLR, LH_2 has a high evaporation rate due to its low condensation point, which is near absolute zero. This would potentially raise the wasted propellant mass during Martian travel and, subsequently, the launch cost. However, when LH_2 is supplemented with a cryocooler powered by NEP-generated electricity, this issue can be completely solved, which has been integrated into the concept of Borowski since 2003. Thus, the BNTEP can effectively avoid propellant waste.

Moreover, NTP's advantages can support NEP as well. Although NEP can generate a high I_{sp} for a longer time and ideal propulsion for interplanetary travel, the thrust offered will be insufficient to detach from the LEO and enter the Martian orbit. Consequently, these phases need to be achieved by the NTP, which can generate a high thrust for a short period to support escaping the LEO. The different propulsion systems utilized for different phases would ultimately reduce fuel consumption and result in a minimal initial mass in LEO. As outlined by Borowski (2003), for the same payload of approximately 100 tons, the CLP is estimated to have 1.5 times more initial mass than BNTEP, where the mass difference varies mostly in the TMI stage. These complementary characteristics can therefore ultimately elevate the cost-efficiency of the journey.

Furthermore, the bimodal NTP/NEP will have a higher delivery efficiency than the pure NTP. NEP has a tremendous potential of greater I_{sp} with rising thermal-electric conversion efficiency, which would break the I_{sp} limit of NTP because of the core temperature and the engine's thermal resistance restraints. The Shanghai Space Station Institute affirmed that the electric power of NEP can exceed 5MW or even 500MW (Chen, 2021). As mentioned by Clark (2019), for a payload of 300 tons, the BNTEP demands only 147.88 days to arrive in TMI from LEO, compared to 160.54 days with a pure NTP-powered NTR. Specifically, the BNTEP will fire the NTP engines in LEO to achieve a delta-V of 3.891 km/s within 48 minutes. Thereafter, the NEP system is activated for 142.32 days to gain a delta-V of 1.905 km/s, after which the NTP is re-burned to enter the Martian orbit with a deceleration of 4.13 km/s for 32 minutes. The ability of NEP to "specify the final transverse and radial velocity of the spacecraft" (Clark, 2019) and the benefit from the Oberth Effect working in the NEP spacecraft's trajectory favor (Clark, 2019) produce higher delivery efficiency for BNTEP. This indicates the viability of bimodal NTP/NEP for Mars travel or even beyond, due to this faster hybrid propulsion system and reduced propellant limitation. The reduced time of the journey also represents a reduced exposure of crews to the hazardous space environment, either for scientific exploration or commercial tourism.

The Limits of BNTEP

However, the complex BNTEP's overall mission architecture represents a higher failure rate and greater cost, but just marginally higher than that of the NTR. As the thrust of both NTP and NEP could not support escaping Earth's influence, the CLP is still needed to achieve LEO in BNTEP's mission architecture. Due to over three types of propulsion systems utilized, various engines should be constructed with different requirements. The cost and challenge of manufacturing and maintenance will ultimately surge. However, the total mission architecture of BNTEP would merely require 3 launches, to assemble the "Bimodal" NTP/NEP core stage, the "In-Line" propellant tank, and the crewed "TransHab" module (Borowski, 2003). There is only one extra launch compared to the NTR's mission architecture and a half compared to the 6 launches for the CLP-based Starship. Furthermore, all three stages of propulsion systems are reusable, which will effectively reduce the budget required to build a great number of rockets to meet increasing demands for space tourism applications.

Moreover, the NEP stage is powered by expensive noble gases, especially Xenon and Krypton, which are particularly rare in the Earth's atmosphere. However, the elevated propellant purchasing cost will be balanced with the BNTEP's minimal fuel consumption. Since the NEP has a high I_{sp} and these noble gases are inert, this propulsion has a relatively low fuel consumption and low boil-off rate. As Mars's atmosphere is composed of 96% of CO_2, <2% of Argon, <2% of nitrogen, and <1% of others (Musk, 2017), the presence of Xenon and Krypton is consequently rare, with considerable difficulties for ISRU on Mars. Nevertheless, the rocket can bring sufficient noble gases for the round trip, which would not occupy a lot of space due to the minimal noble gas tank required. In addition, the period for ISRU will be largely reduced, as less propellant is required to produce, therefore raising the mission efficiency. With further methods to produce noble gases found in the future, the propellant could meet all the large demands for commercial space use.

Additionally, the generated ions from the NEP have engine degradation issues, leading to a reduced lifetime of the spacecraft and potential harm to astronauts in space (Palmer, 2021). However, this ion drive technology has been successfully applied to the Tianhe core module in China's Tiangong Space Station (Chen, 2021). The Chinese Academy of Sciences outlined that Chinese scientists used a magnetic field to cover the engine's interior wall. This method will effectively eliminate the damaging particles. A special ceramic material is also developed to resist excessive heat and radiation (Chen, 2021). Thus, BNTEP is a viable alternative propulsion system for Mars missions.

Conclusion

In conclusion, this paper has reviewed two types of rockets with different propulsion systems, namely the CLR and the NTR.

The survey begins with an overview of the characteristics and limitations of both rockets for Mars exploration and tourism. This is supplemented with possible solutions to resolve the limitations and methods that can maximize the delivery and cost efficiency of CLR and NTR before proceeding with a comparison with these aspects in the discussion. The CLR's efficiency can be improved with better propellant combinations and on-orbit refueling, although some challenges might be encountered in LEO and during the transfer phase. Yet, other solutions with current concepts and technologies are still offered with a precise analysis of advantages and disadvantages. For NTR, solutions are offered for radiation shielding. Different types of nuclear core concepts in NTP result in different delivery efficiencies, although only a solid nuclear fission core can be achieved by the current technology.

In the discussion, both rocket types were compared regarding their delivery ability and their monetary costs. Both rockets have sophisticated mission architecture, but the NTP requires fewer launches from Earth, reduced time staying in low-Earth orbit, and faster transfer to Mars. This also indicates more safety for crewed missions and higher delivery efficiency. However, SpaceX's concept Starship is more mature than the DRA 5.0 for NTR in terms of space commercial use and travel, due to high crew transportation efficiency and reduced difficulties in achieving it with modern technologies. Nonetheless, the NTR consumes less propellant for the overall mission architecture, has a decreased cost for launching and extra rockets to build, and reduces the fuel required to be produced on Mars. These delivery and cost efficiencies prove its potential for future commercial utilization. Thus, the NTR is a viable alternative in the future. However, the best option for Mars missions has proven to be the BNTEP. It has greater delivery efficiency by allocating different propulsion systems to suitable phases and higher cost efficiency with reduced propellant consumption. By taking advantage of all the benefits of the three propulsion systems, i.e., CLR, NTR, and NEP, BNETP is undoubtedly a promising choice for Mars exploration and tourism.

Regarding the dissertation's limitation, the author is not a research-oriented scholar, therefore, the discussion relies significantly on its personal interpretation, which potentially introduces a lack of professionalism. Moreover, most of the propulsion systems reviewed in this paper are still in the investigation phase or theoretical concept form. Even the most mature CLP-based Starship failed its First Integrated Flight Test on April 20, 2023 (Malik & Wall, 2023). Although the Starship reached the height record in its Second Integrated Flight Test on November 18, 2023, it still exploded before reaching LEO. Thus, this paper is limited by real-life cases achieved in the present day, the experimentation and research for means of transport to Mars is still an awe-inspiring realm to be explored in the future.

There are well-thought-out suggestions for further work and an awareness of any wider implications. The engines and mission architecture of reusable rockets are incredibly complex, leading to high maintenance and construction costs. Consequently, further virtual simulations are recommended to mitigate expenses. Second, the excessive pursuit of avant-garde technologies explains the repeated failures in the Starship's Integrated Flight Tests. When researching the fully reusable Starship, it is recommended to first build a partially reusable Starship. Shifting the focus from the pursuit of advanced technology to that of higher reliability and lower cost can effectively increase the successful launch rate with a stable trend of improvement.

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