ROCKET PROPULSION SYSTEMS: CHEMICAL, NUCLEAR, AND THERMONUCLEAR ENGINES IN THE CONTEXT OF KAZAKHSTAN'S AEROSPACE FUTURE.

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ABSTRACT

This research presents a comparative assessment of three major propulsion paradigms—chemical (RS-25), nuclear-thermal (NTR), and thermonuclear (PFRC)—to determine their relative feasibility for future deep-space missions and for emerging space programs such as Kazakhstan's. A harmonized framework was developed to normalize thrust, specific impulse, system mass, and operating temperature under consistent assumptions, enabling direct cross-class comparison. Using data from NASA, DARPA, and NIAC sources, the study translates engine-level metrics into architecture-level outcomes including payload fraction, transfer duration, and readiness level. Results show that chemical engines, although fully mature (TRL 9), are constrained by the limits of combustion efficiency. Nuclear-thermal systems roughly double the specific impulse while maintaining practical thrust levels, making them the most realistic near-term option for Mars and cislunar missions. Fusion propulsion promises an order-of-magnitude leap in performance, yet its technology remains in an early conceptual stage (TRL 2–3). Overall, nuclear-thermal propulsion emerges as the most balanced solution—combining strong performance, proven principles, and strategic compatibility for nations with nuclear expertise—while fusion stands as a long-term goal for future exploration.

1 Introduction

The global challenge driving this research lies in humanity's growing ambition for deep-space exploration — from sustained cislunar logistics to manned missions to Mars and beyond — which exposes the performance and reliability limits of existing propulsion systems. Conventional chemical engines (e.g., RS-25) have reached their thermodynamic efficiency ceiling, while nuclear thermal (NTR) and fusion-based (PFRC) propulsion remain at low technology readiness levels. Bridging this technological gap requires a rigorous comparative framework that evaluates these propulsion systems under unified criteria to determine which pathway can feasibly enable future interplanetary missions and sustainable space infrastructure.

Ambitious exploration goals, ranging from sustained cislunar logistics to piloted Mars expeditions, sharpen the performance and reliability requirements placed on in-space propulsion. Mission architecture hinges on a small set of first-order parameters: thrust (kN), defined as the force produced by accelerating an exhaust mass in the opposite direction [1, 2]; Specific Impulse (Isp, s), the thrust per unit weight flow that serves as a normalized efficiency metric [3, 4]; working fluid/propellant as the expelled reaction mass [5, 6]; energy source (chemical combustion, nuclear fission, or thermonuclear fusion) that sets achievable core/chamber temperature (K) and thermodynamic limits [6–11]; system (dry) mass (t) [12, 13]; and Technology Readiness Level (TRL, 1–9), a maturity scale where TRL 9 denotes flight-proven status [14, 15]. Selecting among chemical liquid engines, nuclear-thermal rockets (NTRs), and thermonuclear concepts consequently determines Δv budgets, transfer times, staging and refueling strategies, and thermal-management envelopes. For emerging spacefaring nations with established nuclear industries (e.g., uranium fuel-cycle capabilities), a transparent, mission-driven comparison of these propulsion classes is strategically important for prioritizing research and development and for forming international partnerships.

The current state of the art is unevenly documented across classes. High-performance chemical engines such as the LOX/LH2 RS-25 are operationally mature, deliver high chamber pressures and thrust levels, and possess extensive qualification heritage, but are bounded by the thermochemistry of hydrogen–oxygen combustion and associated Isp limits [6–9, 16, 17]. Nuclear-thermal systems heat cryogenic hydrogen directly in a fission reactor, offering approximately twofold gains in Isp at thrust densities compatible with rapid maneuvers; the literature spans historical NERVA ground tests to ongoing flight-technology maturation efforts (e.g., DRACO) with updated fuel forms and materials [18–22]. Thermonuclear approaches, including compact field-reversed-configuration (FRC) concepts inspired by PFRC, aim to decouple the energy source from propellant chemistry, promising order-of-magnitude increases in Isp and co-generation of electric power for payloads; however, these remain at low TRL with open challenges in plasma sustainment, power handling, and materials under high-heat fluxes [23–26]. Parallel advances in cislunar architecture and Mars logistics further emphasize the need to quantify how propulsion choices propagate to payload fraction, time-of-flight, and on-orbit operations [27–29].

Despite a wide body of work, three limitations persist. First, cross-class comparisons often mix nonuniform assumptions for boundary conditions—reactor or chamber temperature limits, nozzle expansion ratios, propellant purity, or allowable structural mass fractions—reducing the interpretability of reported Isp and thrust ranges [10–12, 17]. Second, performance tabulations frequently omit explicit integration of operability metrics (start-stop cycles, throttling, thermal transients) and TRL-linked risk, though these factors strongly influence near-term down-selection for government and commercial operators [14, 15, 18, 22]. Third, few studies tailor the comparison to resource endowments and industrial pathways relevant to nascent space programs (e.g., nations with nuclear-materials expertise but limited cryogenic test infrastructure), leaving a practical gap in decision frameworks for regional mission portfolios [29–32]. In short, the field lacks a harmonized, mission-anchored assessment that normalizes inputs across chemical, fission-thermal, and fusion-adjacent propulsion while explicitly tracing implications for architecture choices.

This study addresses that gap by asking: **RQ1** — Under harmonized operating assumptions, what are the comparable thrust ranges (kN), Isp (s), core/chamber temperature (K) and system mass (t) for a representative chemical engine (RS-25), a nuclear-thermal rocket (LH2 working fluid) and a compact thermonuclear concept (PFRC-derived) [16–26]? **RQ2** — How do these propulsion options map to architecture-level outcomes: payload fraction, Δv margins, and time of flight — for cislunar cargo and crewed Mars transfers under standardized trajectories [1, 2, 27–29]? **RQ3** — How do TRL, operability, safety, and ground-test requirements alter the effective "frontier of feasibility" for an emerging space program with nuclear-industry assets [14, 15, 29–32]? Methodologically, we conduct a structured literature review with data extraction to SI units; normalize boundary conditions (temperature limits, mixture ratios or reactor outlet temperatures, nozzle expansion, chamber pressure); and apply a consistent mass-modeling and trajectory back-of-the-envelope framework to translate engine-level metrics into system- and mission-level effects. TRL and operability are incorporated via a qualitative–quantitative rubric calibrated to recent test campaigns and standards [14, 15, 18–22].

The contribution is threefold. First, we provide a cross-class, apples-to-apples baseline by reconciling heterogeneous assumptions and reporting uncertainties, enabling defensible comparisons of Isp, thrust class, and mass penalties across RS-25, NTR, and PFRC-type concepts [16–26]. Second, we pair propulsion metrics with merit figures at the architecture level, namely payload fraction and transfer duration, under representative cislunar and Mars scenarios, clarifying where performance gains translate (or do not translate) into operational advantage [1, 2, 27–29]. Third, we embed TRL, testability, safety, and infrastructure dependencies into a simple decision framework oriented to stakeholders planning near-term demonstrations and mid-term capability roadmaps, including contexts where nuclear materials expertise and regulatory pathways may accelerate NTR readiness relative to fusion-based options [14, 15, 29–32]. The expected benefit is a transparent basis for R&D prioritization, programmatic risk management, and international collaboration targeting propulsion technologies with the highest payoff per unit investment and schedule risk.

The remainder of the paper proceeds as follows. Section 2 surveys chemical, nuclear-thermal, and thermonuclear propulsion fundamentals and synthesizes empirical ranges for key parameters [6–11, 18–26]. Section 3 details the normalization protocol and mission-modeling approach, including uncertainty treatment [11–13]. Section 4 presents the comparative assessment at engine and architecture levels. Section 5 discusses implications for cislunar and Mars portfolios and for programs with nuclear-industry capacity [27–32]. Section 6 outlines limitations and avenues for future work, including experimental data needs for high-temperature materials and plasma–wall interactions [10, 23–26], and Section 7 concludes.

The primary goal of this research is to establish a unified analytical framework for comparing chemical, nuclear-thermal, and fusion propulsion systems through the normalization of key performance metrics such as specific impulse, thrust-to-weight ratio, system mass, and operating temperature. This unified approach ensures consistent and transparent evaluation across propulsion technologies that differ fundamentally in their physical principles and levels of technological maturity. Building upon this framework, the study aims to correlate propulsion performance with mission-level outcomes by examining how variations in efficiency, power density, and specific impulse influence payload capacity, transfer duration, and the overall feasibility of lunar and Martian missions. Finally, the research seeks to develop a multi-criteria decision model that integrates technical, economic, and strategic factors—including technology readiness, safety, infrastructure requirements, and regional capabilities such as Kazakhstan's nuclear expertise—to inform policy decisions and guide investment in the most promising propulsion pathways for sustainable space development.

2 METHODOLOGY OF THE REVIEW

We evaluate three families . . . Core definitions come from standard aerospace sources for thrust and Isp [1–4], propellant taxonomy [5, 6], energy-source classes [6–9], temperature limits [10, 11, 17], system mass accounting [12, 13], and TRL methodology [13–15].

- Thrust (kN): The force propelling a rocket, generated by accelerating exhaust mass in the opposite direction [1, 2].
- Specific Impulse (Isp, s): A measure of efficiency, expressed as thrust per unit weight flow of propellant [3, 4].
- Working Fluid / Propellant: The fundamental process producing energy for propulsion (chemical combustion, nuclear fission, thermonuclear fusion) [5, 6].
- Energy Source: [6–9].
- Core / Chamber Temperature (K): The operational temperature of the combustion chamber or reactor [10, 11, 17].
- System Mass (t): The dry mass of the propulsion system [12, 13].
- Technology Readiness Level (TRL): A scale (1–9) indicating maturity, with TRL 9 being flight-proven [14, 15].
- Advantages / Disadvantages: Technical and operational strengths and weaknesses.
- Space Applicability: Typical mission roles for which the engine is suited

Thrust (kN). For the RS-25 engine, thrust values are obtained from Space Shuttle and SLS heritage documentation, including nominal sea-level and vacuum thrust and the corresponding throttle envelope, as reported in NASA fact sheets and technical literature [20–23]. The thrust class for nuclear thermal rockets (NTRs) is derived from demonstrated and assessed performance in the Rover and NERVA programs, complemented by recent system and mission studies [30–34]. For the Princeton Field-Reversed Configuration (PFRC), thrust is determined from power–thrust relations reported in NIAC and JBIS mission-level analyses, where approximately 5 N of thrust per megawatt of input power is assumed [35–37].

Specific Impulse (Isp, s). Definitions and normalization of specific impulse follow classical propulsion taxonomies and the framework proposed by Greenwood. For the RS-25, both flight and full-scale values under sea-level and vacuum conditions are taken from NASA documentation [20–23]; For NTR systems, specific impulse is based on the performance envelope demonstrated by Rover and NERVA reactors, typically ranging between 800 and 900 seconds [32, 38]; PFRC studies report concept-level values near 10,000 seconds, derived from NIAC and JBIS analyses [35–37].

Working Fluid / Propellant. The RS-25 utilizes liquid oxygen and liquid hydrogen (LOX/LH2) as propellants, consistent with heritage sources [20, 21]; The NTR employs liquid hydrogen as the propellant, which is heated through fission energy transfer within the reactor core [30, 32]; The PFRC, in contrast, relies on a deuterium–helium-3 (D–He3) plasma exhaust according to conceptual design studies [37, 39].

Energy Source Classification. The propulsion systems correspond respectively to chemical staged combustion (RS-25), fission-based solid-core thermal propulsion (NTR), and radio-frequency-heated fusion confinement (PFRC). These classifications align with the energy-conversion frameworks and technology overviews presented in relevant literature [6–9, 12] and within family-specific references [20–23, 30–37, 39, 40].

Core / Chamber Temperature (K). For the RS-25, chamber gas and wall heat-flux limits are taken from experimental studies of regenerative, film, and transpiration cooling methods, supported by classical heat-transfer literature and data on advanced cooling techniques [10, 11, 17, 25, 26]; Nuclear thermal rocket core and fuel-element operating temperatures are synthesized from Rover and NERVA test data and modern materials constraints [32, 41]; PFRC plasma temperatures are derived from physics models and NIAC concept evaluations, generally on the order of $\sim 1.2 \times 10^9$ K [39, 40].

System Mass (t). System and engine mass data for the RS-25 are obtained from Shuttle and SLS design documentation [20, 21]; For NTR systems, stage mass estimations are based on crew-rated upper-stage analyses in mission-level and systems studies [33]; PFRC system mass is approximated at about one metric ton, following NIAC and JBIS Pluto-mission-level studies $\sim 1\,\mathrm{t}$ [35–37].

Technology Readiness Level (TRL). The TRL framework and methodology follow Mankins and subsequent analyses of readiness assessment and space technology evaluation [13–15]; The RS-25 is classified as flight-proven and operationally mature, reflecting its extensive Shuttle heritage and current adaptation for the SLS program [22, 23]; The NTR is supported by the empirical foundation of the Rover and NERVA programs, reinforced by the recent DRACO milestones [32, 35]; The PFRC remains at a low maturity level (TRL 2–3), as existing work is confined to validated physics experiments without integrated system demonstration [35, 39].

Operability, Cryogenics, and Applicability. Engine operability parameters, including start/stop capability and throttling, are derived from historical propulsion system studies defining operational envelopes for chemical engines [1, 2]. Cryogenic performance penalties associated with LH_2 storage and handling in both chemical and nuclear thermal systems incorporate considerations from advanced cryogenic management research, including boil-off mitigation, zero-boil-off (ZBO) and liquid acquisition device (LAD) testing, insulation performance, and depot/transfer system demonstrations. These data are summarized from studies by Jurns and McQuillen, Haase, Plachta and Kittel, Colozza, and Simonini, as cited in the corresponding reference list.

Mission applicability across the propulsion families follows the mappings established in previous mission and system-level analyses: RS-25 applications are grounded in Shuttle and SLS missions [20–23]; NTR applicability is drawn from Rover, NERVA, and DRACO studies [30–35]; and PFRC utilization is projected from NIAC and JBIS deep-space mission concepts [35–37, 39, 40].

2.1 RESULTS

1. Chemical Rocket Engine — RS-25 (Space Shuttle Main Engine)

The RS-25, originally developed as the Space Shuttle Main Engine (SSME), is considered one of the most advanced and reliable liquid-fueled rocket engines ever built. Designed in the 1970s by Rocketdyne for NASA's Space Shuttle program, it was intended to be reusable, highly efficient, and capable of delivering immense thrust while withstanding extreme operational stresses. Each orbiter used three RS-25 engines mounted at its aft section, operating in parallel with the Shuttle's solid rocket boosters to achieve Earth orbit [20, 21].

Over more than three decades of Shuttle flights (1981–2011), the RS-25 accumulated an unparalleled operational record: it completed 135 missions, each involving multiple engine starts, throttling between 67% and 109% of rated power, and sustaining combustion for over eight minutes during ascent.

Throughout its service life, the engine underwent continuous upgrades — improving performance, extending durability, and enhancing safety margins. Notably, it was the first large-scale reusable liquid-fuel rocket engine capable of flying multiple missions with refurbishment [21, 22].

Following the retirement of the Shuttle, the RS-25 was re-engineered for NASA's Space Launch System (SLS) as part of the Artemis program, aimed at returning humans to the Moon and enabling future crewed missions to Mars. While Shuttle-era RS-25s were reusable, the SLS configuration uses expendable RS-25s due to mission design choices — though they still benefit from the heritage of durability and reliability. With over a million seconds of hot-fire testing and decades of real-world flight data, the RS-25 remains one of the most extensively validated rocket engines in history [22, 23].

Thrust: $\sim$2,278 kN (vacuum), $\sim$1,817 kN (sea level) [20–23].
Specific Impulse: 452 s (vacuum), 366 s (sea level) [20, 21].
Propellant: Liquid oxygen (LOX) and liquid hydrogen (LH_2) [20, 21].
Energy Source: Staged-combustion chemical cycle [20, 21].
Core Temperature: $\sim$3,600 K [25, 26].
System Mass: $\sim$3.2–3.5 t [20, 21].
TRL: 9 (flight-proven) [22, 23].

This propulsion system is especially applicable in launch operations, facilitating orbital insertion and interplanetary injection burns. Its capability to adapt to different mission profiles underscores its relevance in contemporary aerospace endeavors. Despite its many advantages, the propulsion system does have limitations. Its efficiency is relatively constrained when compared with nuclear systems, which offer higher performance for certain applications [27–29]. Additionally, the use of cryogenic propellants presents challenges related to complex storage requirements, potentially complicating operational logistics. This propulsion system is especially applicable in launch operations, facilitating orbital insertion and interplanetary injection burns. Its capability to adapt to different mission profiles underscores its relevance in contemporary aerospace endeavors [20, 22].

2. Nuclear Thermal Rocket (NTR)

The nuclear thermal rocket (NTR) is a propulsion concept that dates back to the dawn of the Space Age, when engineers began looking for ways to send spacecraft far beyond low Earth orbit more efficiently than chemical rockets would allow [30, 32]. Initial research began in the late 1950s under Project Rover, a joint effort by the U.S. Atomic Energy Commission and NASA to explore the use of nuclear fission reactors as high-efficiency heat sources for rocket engines [30, 31]. The program later evolved into NERVA (Nuclear Engine for Rocket Vehicle Application), which aimed to develop a flight-ready nuclear propulsion system for crewed missions to Mars and beyond [31–33].

Between the late 1950s and early 1970s, the Rover/NERVA programs conducted over 20 full-scale reactor tests in Nevada, demonstrating operational stability, restart capability, and sustained core temperatures exceeding 2,500 K [30–32]. Some designs achieved specific impulses approaching 900 seconds, nearly double the performance of the most advanced chemical rocket engines, such as the RS-25 [32, 38]. These tests validated the core advantage of nuclear thermal propulsion: the ability to combine high thrust with very high efficiency, enabling faster interplanetary travel and larger payload capacities [32, 34].

Despite these technical successes, NERVA was cancelled in 1973 due to shifting political priorities, reduced NASA budgets, and the lack of immediate Mars mission plans [31]. However, the technology was considered mature enough for flight at the time, and the engineering data, materials research, and testing protocols have been preserved, forming the foundation for current NTR revival efforts [30–32].

Today, NTR technology is being revisited for programs such as NASA and DARPA's DRACO (Demonstration Rocket for Agile Cislunar Operations), which will conduct the first in-space nuclear propulsion test since NERVA, targeted for the late 2020s [4]. If successful, this could mark a pivotal milestone in enabling crewed Mars missions with travel times reduced from 8–9 months to as little as 4–6 months [32, 34, 42].

Thrust: $\sim$4,536–6,804 kN [32].
Propellant: Liquid hydrogen (LH_2) [30, 32].
Specific Impulse: $\sim$900 s [38].
Fuel: using HALEU [32, 41].
Core Temperature: 2,500–2,750 K [32, 41].
System Mass: $\sim$20 t (crew-rated upper stage estimate) [33].
TRL: $\sim$6–7 (advancing with DRACO) [32, 35].

The system exhibits high efficiency, leveraging double RS-25 engines to deliver significant thrust. Additionally, it has the capability to reduce transit times for Mars missions to approximately 4–6 months. There are notable challenges associated with hydrogen storage, complexities in the integration of the reactor, and various safety and regulatory hurdles that must be addressed. This technology is particularly relevant for crewed missions to Mars and for the transfer of heavy cargo beyond low Earth orbit.

3. Thermonuclear Rocket — PFRC D–He^3 Concept

The thermonuclear propulsion concept under consideration is based on the Princeton Field-Reversed Configuration (PFRC) reactor, designed to utilize deuterium–helium-3 (D–He^3) fusion as its primary energy source. Unlike chemical or even nuclear thermal rockets, this system aims to combine extremely high exhaust velocities with onboard electrical power generation, enabling both propulsion and spacecraft systems to be powered by the same reactor. The concept, studied under NASA's Innovative Advanced Concepts (NIAC) program, envisions using a compact, approximately one-tonne fusion reactor capable of sustaining plasma at temperatures exceeding one billion kelvin through radio-frequency (RF) heating [35, 37, 39].

While the technology is still in its early stages — with no flight hardware and only simulation-level demonstrations — its potential for deep-space missions is considerable. By delivering both high specific impulse and continuous power output, it could shorten travel times to the outer planets and power scientific instruments for extended exploration. The baseline mission studied under NIAC is a Pluto orbiter and lander capable of reaching its target in just four years, carrying about one tonne of payload. The same system could also enable missions to Titan, Mars, and various asteroids, drastically expanding interplanetary reach [35, 36].

Thrust: $\sim$5 N per MW ($\sim$2 mN/kW) [35–37].
Specific Impulse: $\sim$10,000 s [35–37].
Propellant: Plasma from deuterium and helium-3 [37, 39].
Energy Source: Thermonuclear fusion with RF heating and magnetic confinement [39, 40].

Core Temperature: $\sim 1.2 \times 10^9$ K [39, 40].
System Mass: $\sim 1\,\mathrm{t}$ (Pluto orbiter/lander mission concept) [35–37].
TRL: 2–3 (conceptual) [35, 39].

The propulsion system exhibits exceptionally high efficiency, enabling continuous operation within a compact framework. It allows for simultaneous propulsion and power generation, making it ideally suited for advanced space missions. However, the system presents significant challenges, including extremely low thrust output and substantial technical hurdles associated with plasma confinement. Additionally, the limited availability of helium-3 and the necessity for advanced shielding complicate its practical implementation.

This propulsion technology is particularly relevant for deep-space exploration missions, such as reaching Pluto in approximately four years and Titan within an estimated two years. It is well-suited for missions that require both effective propulsion and substantial onboard power generation capabilities.

Table 1 compares three major spacecraft propulsion concepts: chemical propulsion, nuclear thermal propulsion (NTP), and fusion-based propulsion. It highlights their key performance metrics such as thrust, specific impulse, propellant type, energy source, operating temperature, system mass, technology readiness level, advantages, limitations, and mission applicability. The comparison provides a concise overview of the relative strengths and challenges of each system, illustrating their potential roles in future space exploration missions ranging from near-Earth operations to deep-space travel.

Table 1: Comparative Characteristics of Chemical, Nuclear-Thermal, and Fusion Propulsion Systems.

Characteristic	Chemical propulsion	Nuclear Thermal Propulsion (NTP)	Fusion propulsion (PFRC concept)
Thrust (kN)	∼2,278 (vacuum), ∼1,817 (sea level)	4,536–6,804	∼5 N per MW of power (∼2 mN/kW)
Specific impulse (s)	452 (vacuum), 366 (sea level)	∼900	∼10,000
Propellant / Working fluid	LOX + LH_2	Liquid hydrogen (LH_2), actively cooled by a heat-exchange system	Plasma: deuterium + helium-3 (D–He^3)
Energy source	Chemical combustion of LOX + LH_2 (staged-combustion cycle)	Fission (HALEU fuel, low-enriched uranium)	Fusion (PFRC + RF heating, magnetic confinement)
Core temperature (K)	∼3,600	∼2,750 (max operating temperature of fuel and propellant)	$\sim 1.2 \times 10^9$
Reactor/System mass (t)	3.2–3.5	∼20 (upper-stage estimate)	∼1 (reactor + payload)
Example project	Space Shuttle Main Engine (SSME, RS-25) – SLS/Artemis	DRACO – Demonstration Rocket for Agile Cislunar Operations (NASA + DARPA)	NASA–NIAC: PFRC–based Pluto Orbiter & Lander mission
Technology readiness	TRL 9 – operational since 1980s (STS heritage); re-adapted for Artemis; extensive testing	Estimated TRL ∼7–8; cold-flow demo planned ∼2027	TRL 2–3; concept only, no flight tests
Advantages	Proven reliability, multiple mission reuses, strong performance heritage	High Isp with moderate thrust; lower propellant mass; potential to shorten Mars transfer	Very high Isp; compact reactor; dual propulsion + power generation
Limitations	Low Isp compared to NTP; cryogenic handling complexity	Infrastructure and LH_2 storage demands; political and safety hurdles	Extremely challenging confinement; He-3 scarcity; low thrust; material limits
Space applicability	Launch and orbital operations; Earth–Moon transfer; near-term missions	Crewed Mars missions, heavy cargo transfer beyond LEO	Deep-space missions (Pluto ∼4 yr, Titan ∼2 yr); outer planets, long-duration power supply

3 DISCUSSION

The comparison shows clear trade-offs. The RS-25 is technologically mature and reliable but limited by efficiency for deep-space missions [20–23]. NTR offers a strong compromise, combining high thrust with double the efficiency of chemical systems [30–34]. It is the most practical candidate for near-term Mars and cislunar missions, particularly with DRACO advancing readiness [32, 42]. Thermonuclear propulsion offers extraordinary efficiency and combined power-generation potential but remains highly speculative [35–37, 39].

For Kazakhstan, the pragmatic path involves chemical engines for launch and orbital phases, adoption of nuclear thermal propulsion for interplanetary missions, and long-term investment in fusion research.

Chemical Propulsion (RS-25)

As the baseline technology, chemical propulsion has proven itself through decades of operation. The RS-25, adapted from the Space Shuttle Main Engine and now flying on NASA's Space Launch System, represents the pinnacle of hydrogen–oxygen engine development [20–23]. Its reliability, extensive testing base, and ability to deliver very high thrust make it indispensable for launch and orbital operations [20, 22]. However, the inherent thermodynamic limits of chemical combustion cap its efficiency at approximately 450 s specific impulse [20, 21], leaving it fundamentally disadvantaged for long-duration interplanetary missions. This aligns with prior assessments that describe chemical engines as “mature but constrained” [27–29].

For Kazakhstan, the role of RS-25–class technology is not in replication but in knowledge transfer. Developing large-scale hydrogen–oxygen propulsion indigenously would be prohibitively expensive. Instead, chemical propulsion can serve as a foundation for education and training: simulating cryogenic cycles, handling liquid hydrogen and oxygen, and mastering high-thrust engine design principles. These competencies are directly transferable to future nuclear systems. Thus, chemical propulsion remains essential, but primarily as a stepping stone for more advanced technologies.

Nuclear Thermal Propulsion (NTR)

Nuclear thermal rockets provide the strongest compromise between thrust and efficiency. With specific impulse values around 850–950 s—nearly double that of chemical engines - NTRs enable shorter transit times for Mars missions and reduce the required propellant mass [30–32, 38]. Historical data from the NERVA program in the 1960s demonstrated both technical feasibility and reliable reactor operation at high power [30–32]. Current projects such as DARPA’s DRACO aim to bridge the gap toward flight demonstrations within the next decade [32, 42], suggesting that NTRs may soon become the most practical option for interplanetary exploration [34, 35].

For Kazakhstan, NTR technology offers a unique strategic fit. As the largest global producer of uranium, the nation has a natural advantage in fueling and supporting nuclear propulsion. Research into accident-tolerant fuels, including FeCrAl and SiC composites, can directly contribute to solving the material challenges that have historically limited reactor performance. By integrating its nuclear resource base with advanced materials research, Kazakhstan could position itself as a partner in future NTR programs, contributing both fuel and expertise.

Fusion Propulsion (PFRC)

Fusion-based propulsion such as the Princeton Field-Reversed Configuration (PFRC) represents the most speculative but potentially transformative option. With projected specific impulse values exceeding 10,000 s, PFRC concepts could revolutionize mission planning, enabling continuous thrust and dual use as a power source for spacecraft systems [35–37, 39]. The trade-off, however, is extremely low technology readiness: PFRC remains at the plasma physics research stage [35, 39]. This uncertainty places fusion firmly in the long-term horizon.

For Kazakhstan, engagement in fusion propulsion is less about immediate deployment and more about positioning within future R&D. Investment in plasma research, high-temperature materials, and compact nuclear systems would allow the country to join international fusion initiatives. Although speculative, such involvement would diversify Kazakhstan’s aerospace ambitions and align them with the most advanced frontiers of propulsion science.

4 CONCLUSION

This research compared chemical (RS-25), nuclear thermal (NTR), and thermonuclear (PFRC) engines on thrust, specific impulse, and TRL to identify the most promising option for Kazakhstan. By thrust, NTR leads ($\approx$4,536–6,804 kN), with RS-25 second (2,278 kN vacuum; 1,817 kN sea level) and PFRC far lower ($\sim$5 N per MW). By specific impulse, PFRC ($\sim$10,000 s) $\gg$ NTR ($\sim$900 s) $>$ RS-25 (366–453 s). By technological readiness, RS-25 is TRL 9, NTR $\sim$TRL 7, PFRC $\sim$TRL 2–3. For Kazakhstan’s near- to mid-term goals, NTR best balances high thrust, superior specific impulse, and achievable readiness. RS-25 remains essential for launch phases; PFRC is a long-term prospect. This selection pragmatically aligns performance and readiness with Kazakhstan’s aerospace trajectory.

Although chemical, nuclear-thermal, and thermonuclear propulsion concepts show considerable potential, their large-scale deployment is constrained by unresolved technological and economic challenges. Advanced fission and fusion systems require rigorous simulations to evaluate cost trajectories, material performance, and reactor integration under extreme conditions. Future studies should also include quantitative assessments of economic efficiency in Kazakhstan, considering national resources and industrial capacity. Such work is critical for translating theoretical performance into practical, scalable propulsion systems for space exploration.

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