A Path Toward Nuclear Thermal Spaceplanes

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Abstract

This study seeks to determine if it is both technically and regulatorily feasible to develop and commercially operate a reusable launch vehicle between ground and low earth orbit using a nuclear thermal propulsion engine exclusively. In other words, this report attempts to answer the question, “Is a launch vehicle utilizing exclusively nuclear thermal propulsion feasible, from technical and (in the US) regulatory perspectives?” To state the conclusion up front: probably yes, though more research and optimization is needed.

The regulatory analysis is necessary because operating an illegal launch vehicle has about the same chance of entering commercial operation as one failing to produce any thrust or lift. Additionally, regulations impose requirements the technical analysis must meet. Most notably, regulations and technical realities limit the practical options for nuclear fuel to High-Assay Low-Enriched Uranium (HALEU). Since legislation is more malleable than the laws of physics, a regulatory analysis like this is necessarily constrained to the regulations existing in a certain place and time. This report analyzes the situation in the country in which this work was conducted (the United States of America) at the time this report was completed (May 2025).

The study identifies two regulators with jurisdiction over a commercially operated nuclear thermal launch vehicle:

- The Federal Aviation Administration (FAA), within the Department of Transportation, has jurisdiction over the launch vehicle during commercial launch operations, from the moment operations for a given flight begin to the moment they end. Although certain details change, most notably the presence of regulatory concerns not significant for launch vehicles using chemical propulsion, the core regulatory structure does not. The spaceport this vehicle launches from will also need to be licensed by the FAA.
- At all other times, the Nuclear Regulatory Commission (NRC) has jurisdiction over the nuclear thermal engine. In particular, only NRC jurisdiction applies during ground tests, nuclear fuel handling, and maintenance. These operations must be accounted for during licensing, but the launch vehicle and the spaceport hangar it rests in are licensed in the same manner as any other non-power-generating reactor.

Our technical analysis demonstrates, despite the constraints of HALEU's reduced enrichment and the added mass of radiation shielding, which constraints do not exist for on-orbit nuclear thermal propulsion, achieving a thrust-to-weight ratio high enough to allow launch remains technically feasible. A significant difference from historical nuclear thermal propulsion engine designs is the insertion of a protective liner between the nuclear fuel and propellant, avoiding ablation of the nuclear fuel and thereby dramatically reducing radioactive content in the exhaust. While this approach maintains the core principles of the MITEE concept as published by the Department of Energy, this report identifies performance enhancements through targeted modifications, which

may be explored further in subsequent work. To establish practical benchmarks and create a roadmap toward implementation, this report also includes an analysis of a spaceplane flight profile utilizing this propulsion system, to confirm the performance thresholds required for successful operation.

Among the most promising pathways for future work are:
- Improving the engine's nuclear fuel geometry and hydrogen flow path to put more heat into the hydrogen per unit mass of engine,
- Redesigning to operate at higher temperatures, perhaps 3,600 Kelvin instead of the 3,000 Kelvin considered in this report, to achieve higher efficiency,
- Use of better materials than this study was allowed to cite, such as radiation shielding that provides lower kilograms per square meter shielded for the same dose reduction, and
- Testing to confirm simulated performance, which will be essential to obtain the licenses required to begin operations.

More development is needed to reduce this concept to practice, but this report disproves several of the common assumptions that would, were they true, make this approach impossible. The result lies within the bounds of what physics, and the laws of the United States of America as of May 2025, allow.

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Introduction

This study attempts to answer the question, "Is a launch vehicle utilizing exclusively nuclear thermal propulsion (NTP) feasible, from technical and (in the US) regulatory perspectives? Can it be done?"

To state the conclusion up front: probably yes, though more research is needed[1].

The regulations impart several constraints as challenging as the physical limits. If one were to construct and attempt to operate a launch vehicle using NTP without the necessary licenses, it would most likely be seized before any launch could occur[2], preventing said launch just as thoroughly as if the launch vehicle's thrust was less than its weight. Rocket engines need thrust greater than their weight to overcome gravity, a task made more difficult by regulations making it impractical to use the high grade of fissile fuel available to traditional NTP engines: this engine must use a concentration of less than 20% uranium-235, as opposed to the traditional case of over 90%. A NTP launch vehicle, as it will begin operation much closer to the public, also requires much more radiation shielding than a NTP engine only used in space, adding more mass the thrust must overcome. This report shows an engine design using this lower grade of fuel, which still has thrust significantly greater than its weight even after accounting for the radiation shielding, although further refinement to the design will be needed to make it practical[3].

On the regulatory side, both the Federal Aviation Administration (FAA) and Nuclear Regulatory Commission (NRC) - the agencies who would share licensing responsibilities for such a system - agree there is a path to licensing a nuclear thermal launch vehicle, assuming relevant safety and other criteria were met. However, the regulatory scheme which exists today does not explicitly address this type of launch vehicle, the application process is unproven, and there are other matters which could be addressed by revising the regulations or taking other actions. This project includes several aspects not currently in the regulations.

To limit this study's scope, it narrowly focuses on a launch vehicle using only nuclear thermal propulsion travelling to an approximate altitude of 400 km in Low Earth Orbit (LEO). This results in several details which do not apply to other NTP systems, such as operation of the system in atmosphere or the requirement to use only HALEU rather than any other nuclear fuel. Given the expected cost to manufacture such a launch vehicle, as well as the problems of disposing of spent nuclear fuel in space, this report assumes the system must be fully reusable in

[1] The refrain of almost every principal investigator at this stage of research.
[2] 51 U.S.C. § 50917.b.D.ii explicitly authorizes such action "when there is probable cause to believe the object…likely will be used in violation" ("Enforcement and penalty," 2023). There may be additional statutes authorizing government seizure.
[3] This has been likened to fusion researchers who managed to get more energy out of a fusion reactor than is put in: that there is a path to commercial viability (getting more energy out than was put in after accounting for inefficiencies in a commercial reactor) is clear, but they still have to traverse that path to reach their goal.

order to be practical - which generates constraints not applicable to most NTP systems, such as how to handle the regulatory requirements for a nuclear system operated in space to subsequently reenter Earth's atmosphere.

Some of the information relevant to making a full design is not approved for public disclosure. Where this report encounters this issue, it presents distillations or approximations based on publicly available data. A notable example is radiation shielding: there are better materials than were allowed to be cited here, as those who could provide better materials could not approve citing their materials' performance in this report. This report instead presents two examples for which there was publicly available data[4].

What Is A Nuclear Thermal Launch Vehicle?

Nuclear thermal propulsion uses a nuclear reactor to heat a propellant to temperatures much higher than traditional chemical rockets can achieve. This higher temperature translates to a higher exhaust velocity, which makes for a more efficient rocket: less propellant is needed to achieve a given mission, such as launch from Earth's surface to Low Earth Orbit.

Traditionally[5] NTP has only been attempted in orbit, with the reactor held safe and inoperable during launch. Letting the propellant flow directly over the nuclear fuel to maximize heat transfer, resulting in highly radioactive exhaust since the nuclear fuel would ablate into the propellant, was not seen as a safety concern as there would be no one nearby to irradiate. This is unacceptable for a launch vehicle, but adding a thin liner between the propellant and the nuclear fuel prevents this from happening[6]. Some of the fission products may diffuse into the liner, but

[4] In order to prove the minimum shielding is not orders of magnitude worse, such as massing several times the rest of the engine instead of the rest of the engine massing substantially more than the radiation shielding. If several tons of shielding would be needed when the engine massed much less than a ton, then given the approximate range of thrust-to-weight achievable using HALEU before accounting for shielding, the engine with shielding would be unable to produce more thrust than its weight and could not serve a launch mission.

[5] With limited exceptions such as NERVA. These exceptions happened long enough ago, with different enough technology bases, that the FAA and NRC do not consider them representative of this effort. Whether nuclear thermal propulsion itself has ever been operated in orbit is debated, such as by some proponents counting propulsion involving radioisotope thermoelectric generators, though that might be more properly classified as nuclear electric propulsion (using nuclear power to generate electricity that then generates propulsion, rather than using heat from nuclear processes directly in propulsion without first converting to electricity). In any case, NTP designs in the literature emphasize NTP for use only in or beyond Earth's orbit, but a nuclear thermal launch vehicle faces a different environment.

[6] There is still potential for radioactivity, from neutrons captured by the hydrogen propellant making deuterium and then tritium, but the proportion of this appears to be inconsequentially small.

for feasibility, the liner only needs to prevent the products[7] from diffusing all the way across the barrier during a single launch's operations[8].

Background

The concept of using nuclear power for rocketry dates back to at least 1944 (Corliss & Schwenk, 1971, p. 11-12), when scientists at Los Alamos first recognized the potential of nuclear energy's superior power density for propulsion applications. The most comprehensive investigation came through the Nuclear Engine for Rocket Vehicle Application (NERVA) program (1955-1973), a joint NASA-AEC effort successfully demonstrating the technology's viability. NERVA achieved peak specific impulse values of over 850 seconds (Finseth, 1991, p. C-2), nearly double of chemical rockets.

Although technically successful and achieving Technology Readiness Level 6 (Gerrish, 2014, p.1), the program was cancelled by President Nixon in 1973 amid broader cutbacks to space programs (Haslett, 1995, p. 2.1). Subsequent development efforts - including the Space Nuclear Thermal Propulsion Program (1987-1994), NASA's Nuclear Propulsion Office (1991-1993), Project Prometheus (2003-2005), and recent NASA/DOE collaborations - have advanced the technology incrementally but were limited in scope or terminated before flight demonstrations (see for instance Haslett, 1995, p. 2.5).

The core challenge for NTP development has not been technological feasibility but funding sustainability. NTP development has historically relied on mission pull rather than technology push (Haslett, 1995, p. 3.1). With relatively few missions beyond Earth orbit, funding for propulsion technologies optimized for deep space operations has been limited and inconsistent.

Launch from ground to Earth orbit presents a fundamentally different economic picture. The commercial launch sector has experienced remarkable growth, with over 2,800 objects launched to space in both 2023 and 2024 (as cited in Our World in Data, 2025, from the United Nations

[7] Technically, it only needs to stop enough that the dose to the public from any products escaping into the exhaust, combined with the dose to the public from direct gamma and neutron emissions, is within acceptable regulatory limits. In practice, dose to the public will be essentially entirely from direct gamma and neutron emissions at the launch site, and practically none from the exhaust. Diffusion through even a thin liner takes at least many hours at high temperature (or multiple human lifetimes at room temperature). Using the formula in Schwelger & White (1968), the diffusion constant D at 3000K (conservatively assuming maximums, given the uncertainties in Schwelger's measurements) is $(1.8 + 1.05) * 10^{-2} * \exp(-(93 - 6) / (1.987 * 10^{-3} * 3000))$, or $1.30 * 10^{-8}$ cm²/s. The distance penetrated is the square root of D times the time in seconds, which for 3,600 seconds (which significantly exceeds the total time at 3000K during a launch) comes to about 68.4 micrometers.

[8] This would entail replacing the liner between missions. For practicality, it would be best to be able to reuse the liner across several missions. Given the rate of diffusion in the above footnote, over 360,000 seconds (greater than the expected total time at 3000K during a year of 100 launches), the uranium would diffuse across just under 0.7 millimeters. Since the distance penetrated increases with the square root of the time, assuming an operational lifetime of no more than 4 years for the nuclear fuel, 1.4 millimeters of tungsten should suffice, so the tungsten could remain covering the uranium for this entire duration, to be replaced as part of replacing the nuclear fuel.

Office for Outer Space Affairs). Launch costs remain substantial as of May 2025 - SpaceX charges over US$1,000,000 to launch a single 180 kg payload to LEO (SpaceX, 2025)[9] - creating a significant market opportunity for more efficient alternatives. This robust market suggests, while development funding remains challenging, an operational NTP launch vehicle could potentially generate substantial ongoing revenue once developed. However, this use case is somewhat more technologically demanding than interplanetary operations.

The key technical advantages of NTP systems include higher specific impulse (approximately double chemical rockets), improved payload fractions, and operational versatility. These advantages could translate to significant cost reductions for Earth-to-orbit transportation if the technical and regulatory challenges of nuclear thermal launch systems can be successfully addressed. The following sections of this report examine these challenges and present an approach to addressing them.

Definitions and Key Terms

"Impossible"

As science fiction author Arthur C. Clarke put it, "When a distinguished but elderly scientist states that something is possible, he is almost certainly right. When he states that something is impossible, he is very probably wrong" (as cited in Ratcliffe, 2016).

Several times throughout this study, the authors of this report were told something involving the launch of nuclear material was completely impossible, only for follow up conversations to reveal the source meant "practically impossible" or "impossible using the methods the source was aware of". For example, it is "impossible" to obtain regulatory approval for nuclear thermal launch if, as has traditionally been done, there is no barrier between the propellant and the nuclear fuel, where the nuclear fuel ablates and results in a highly radioactive exhaust. This is not the same as "impossible by any means": once the problem is identified, a solution can be provided. In this example, a barrier can keep the nuclear fuel from ablating, resulting in far less radioactive exhaust.

A good example is one presentation at the 2025 NRC RIC which stated it was impossible to gain regulatory approval to launch nuclear systems after they had been tested on the ground, which greatly impacted reliability since they could not go critical until they were in space. What the speaker meant was, the existing launch failure rate (there was some debate, but the speaker was citing at least 1%) and the high degree of consequence (the speaker did not appear to allow time for the system to decay back to substantially pre-criticality conditions before launch, meaning there would be a high degree of fission products, which would have an increased environmental

[9] The estimated price for a 180 kg payload with minimum options was "$1.17M".

impact if they became fallout) exceeded any plausible maximum regulatory threshold for approval. Among the potential solutions is to develop a launch system with a much lower rate of failure. This study aims to enable the development of launch vehicles which eliminates several of the observed root causes of launch failure, mainly by greatly limiting the potential for an explosion to happen inside the launch vehicle, in the hopes this will reduce the rate of launch failure. If successful, this might make it practical to launch payloads containing nuclear reactors tested on the ground, greatly increasing safety when they are subsequently used in space.

Fuel

Nuclear engineers use “fuel” to refer to uranium or other fissile material. Rocket engineers use “fuel” to refer to the propellant.

As the term “propellant” exists, this document uses “fuel” to refer to nuclear fuel where the meaning might be ambiguous.

National Security Presidential Memorandum 20 (NSPM-20)

NSPM-20, issued on August 20, 2019 (Trump, 2019), is the ultimate source of many of the regulatory limits and realities unique to systems using nuclear material in space, in effect in the United States of America as of the publication date of this report. Other directives and legislation include Space Policy Directive 6 (SPD-6) and the ADVANCE Act.

Most of the applicable regulations outside of NSPM-20 are either things applying to nuclear systems in general, or to launch vehicles in general.

Code of Federal Regulations (CFR) and United States Code (U.S.C.)

The CFR and U.S.C. are the primary bodies of federal regulations and laws, respectively, in the United States. Certain items in the CFR this report cites are a result of NSPM-20, and the ADVANCE Act amended certain portions of the U.S.C., but it is the CFR and U.S.C. which define the law.

CubeSat and EELV Secondary Payload Adapter (ESPA)

CubeSat (Johnstone, 2022) and ESPA (Haskett et al., 1999) are two of the most widely used standards for small satellites. This engine described in this study requires a high number of launches per year to be feasible for use. As of May 2025, many commercial rocket designs define a maximum payload and then require operators to spend significant effort filling the manifest to justify each launch. It would be easier to fill the manifest, resulting in more launches, if the maximum payload was some already-in-use standard, so that one satellite meeting that standard would entirely fill the manifest and there is a pre-existing demand for launches of payloads meeting that standard. For this reason, the designs considered in this study define their maximum payloads as a single 12U CubeSat (24 kg) or a single secondary ESPA satellite (180 kg).

Low-Assay and High-Assay Low-Enriched Uranium (LALEU and HALEU)

For reasons discussed in a later section, NSPM-20 effectively requires this launch vehicle to use uranium enriched to no more than 20%[10] U-235. The term HALEU refers to uranium enriched to near 20%, but the literature does not provide a general term for uranium enriched much less than 20%, instead just describing all uranium up to 20% "low-enriched uranium". This report uses the term LALEU to describe low-enriched uranium, which is less enriched than HALEU. Specifically, this study uses "LALEU" to refer to uranium with 1% to 5% enrichment[11], "HALEU" for uranium with greater than 5% but no more than 20% enrichment, "HEU" (highly enriched uranium) for uranium with greater than 20% enrichment, and "natural uranium" for uranium with less than 1% enrichment[12].

In practice, low-enriched uranium available is either enriched up to 5%, or enriched around 19.5-19.75% levels, so there seems little need to discuss or create labels for levels in between. The restrictions applicable to HALEU would be the same for 10% or 15% enrichment, and performance would be worse.

Critical and K-effective

For those unfamiliar with nuclear terminology, "achieving criticality", "going critical", and the like in a nuclear context refer to a reactor becoming operational. These terms in this context do not refer to anything explosive.

"K-effective" is a measure of this criticality: for each neutron produced by nuclear fuel in a certain configuration, how many neutrons will be produced by a specific neutron? Neutrons escaping the system produce no further neutrons, while certain other neutrons may collide with another atom, splitting it and producing 2 neutrons. K-effective is the average given how often each case occurs. If k-effective is at least 1, then there is a sustained chain reaction, producing heat (so long as the nuclear fuel lasts; it takes much longer than a single flight of the launch vehicles described in this study to substantially use up nuclear fuel). If k-effective is exactly 1, the reactor is in a "critical" state, with power levels remaining steady, while if k-effective

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10 This study's definitions, and the common definition of HALEU, only consider the percentage of the isotope U-235 out of the total amount of U-235 and U-238. Any actual nuclear fuel includes further ingredients.
11 Typical commercial grade is 3-5%.
12 There is a range of values given for natural uranium enrichment. While 0.7% or 0.72% is typical, a literature search found values up to 2.2% being labeled "natural uranium", which apparently conflates "percentage by mass" (the meaning this report uses, and the meaning most sources found use) with "the portion of radiation of natural uranium that uranium-235 is responsible for". As commercial grade is generally at least 3%, a cutoff of 1% seems useful.

exceeds 1, it is in a "supercritical" state with power levels rising. In most cases, k-effective for a supercritical reactor slowly decreases as power and temperature rise, until k-effective reaches 1 and the reactor settles into a critical state. If k-effective is less than 1, then the chain reaction diminishes and the reactor is in a "subcritical" state. Critical and supercritical can be thought of as "on" states, with subcritical being the "off" state, though a reactor takes some time to fully start up or shut down. For a small reactor like the one described in this study, this can be several seconds.

Pitch-to-Nuclear-Fuel-Diameter (P/D) and Length-to-Nuclear-Fuel-Diameter (L/D)

Two ratios of significance in modeling the engine's geometry. These refer to different diameters:

- the central component of the engine is a hexagonal array of identical elements ("pin cells"), which contain cylindrical spaces with the nuclear fuel (with tungsten lining) and voids for the hydrogen propellant to flow through;
- "pitch" refers to the center-to-center distance between adjacent elements, while the "diameter" in P/D refers to the diameter of one of these cylindrical spaces; and
- "length" refers to the length of these cylindrical spaces, but the "diameter" in L/D is that of the overall collection of pin cells.

Single Stage To Orbit (SSTO)

During this study, the authors encountered a number of people who attempted to redefine this term from its plain meaning.

This study discusses a launch vehicle which starts on the ground, goes to orbit, and comes back to the ground (either the same place it launched from, or another suitably equipped landing point), having expelled only the payload (left in orbit) and propellant. All of the launch vehicle's hardware reaches orbit, and all of it comes back down.

As this vehicle uses only nuclear thermal propulsion to achieve thrust, the vehicle's engine must attain criticality on the ground. There is no option to wait until orbit to first achieve criticality. Alternative propulsion methods, detailed in the ConOps section, were not explored in depth as they would add complexity and risk[13] while reducing overall performance[14] in most cases.

Sievert and Rem

Two commonly used measurements of radiation doses. 1 sievert is 100 rem. 1 microseivert (μSv) is 0.1 millirem (mrem).

[13] Thus making the design less feasible from a regulatory perspective.
[14] Thus making the design less feasible from a technical perspective.

Banana-Equivalent Dose

The banana-equivalent dose, 0.1 μSv (which comes to 0.01 mrem), is a well known means of contextualizing radiation doses.

Many natural sources expose humans to measurable radiation levels. For example:

- Sleeping next to another person exposes one to 0.15 μSv overnight (Murphy, n.d.; Protopopescu, n.d.).[15]
- A single flight from London to New York exposes passengers to approximately 40 μSv due to cosmic radiation at high altitude (Protopopescu, n.d.).
- Eating a banana delivers approximately 0.1 μSv due to its natural potassium content (Protopopescu, n.d.).

Receiving 25 mrem[16] (the equivalent dose of consuming 2,500 bananas[17]) falls well within the range of radiation humans regularly encounter without observed adverse health effects.

A Note on Nuclear Misconceptions

Throughout this project, the authors of this study encountered fundamental misconceptions about nuclear technology that persist despite scientific evidence. Among the most common false assumptions about nuclear systems encountered were that:

1. Any radiation exposure, no matter how minimal, is harmful - or at least, too harmful to be allowable by law. This is contradicted by the regulations, which state that there is an acceptable level of public exposure to radiation greater than zero, as well as by examples such as the banana-equivalent dose. If the only acceptable amount of radiation was zero, coal power plants could not be allowed to operate[18]. As this report is about regulatory feasibility, this report leans toward the point of view expressed in the relevant regulations in the United States as of this report's publication, primarily NSPM-20.
2. Nuclear systems in general, whether for nuclear thermal propulsion or otherwise, cannot be engineered with sufficient safety features to render them safe for practical operation, despite numerous examples in naval propulsion, medical, power production, and other applications.

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[15] Combined source info with a banana-equivalent dose being 0.1 μSv. This does not include any dose your own body is giving yourself, just the extra dose someone else can give you.
[16] The lowest enumerated radiation threshold specified in NSPM-20 (Trump, 2019), and this study's designs' target to not exceed per launch.
[17] The authors of this study do not recommend eating 2,500 bananas in one meal, but it would not cause acute radiation poisoning.
[18] "The chances of experiencing adverse health effects from radiation are slim for both nuclear and coal-fired power plants—they're just somewhat higher for the coal ones" (Hvistendahl, 2007).

3. Nuclear thermal propulsion must follow historical design limitations, which renders it unsafe for operation within the atmosphere or unusable for the mission proposed in this report. As addressed in later sections of this report, this report's innovations specifically address safety concerns identified by previous nuclear thermal propulsion work, and optimize for use in launch vehicles.

These are not matters of opinion or risk tolerance, but factual inaccuracies that have been propagated through decades of incomplete or outdated information. Decisions based on these misconceptions, rather than current scientific understanding, hinder technological progress and national security objectives. Ironically, such decisions often even compromise public safety, despite often being made in the name of safety. The authors of this report invite all stakeholders to evaluate this report's systems based on their actual design features and safety analysis, rather than preconceptions that may have been formed from outdated or incomplete information.

Concept of Operations (ConOps)

In a NTP system, a nuclear fission reactor heats hydrogen propellant to extreme temperatures[19], which is then expanded through a nozzle to generate thrust. Unlike chemical propulsion that is limited by the energy released in chemical reactions, NTP performance is constrained primarily by materials' temperature limits and reactor design. The key performance advantage derives from hydrogen's low molecular weight, which combined with heating to these temperatures produces exhaust velocities approximately twice that of the best chemical engines. This translates to a specific impulse (Isp) in the range of 800-1000 seconds, compared to roughly 450 seconds for hydrogen-oxygen chemical rockets. This report's design aims for 3000K[20], for an exhaust velocity of about 10 km/s, resulting in a specific impulse of about 1000 seconds.

The baseline configuration under study derives from the MITEE design, a compact NTP engine intended for interplanetary missions (Powell et al., 1999b). Several modifications have been made to adapt to the mission requirements of a launch vehicle, including using control drums rather than a disposable control rod[21], using HALEU instead of HEU, and adding omnidirectional radiation shielding instead of a shadow shield. This report focuses on the engine itself. A launch vehicle using this engine would include auxiliary thrusters for attitude control; lightweight cryogenic hydrogen storage and pumps; a regeneratively cooled bell nozzle (likely an aerospike) with a spring-loaded airtight flap or other safety system to seal the engine when not in use (to keep atmospheric oxygen[22] from interacting with hydrogen inside the engine, or in case of splashdown, to keep water out of the engine); a secondary cooling system to remove decay heat from the engine after a burn; a heat shield to protect the vehicle during reentry; payload storage and dispenser; batteries sufficient to last from launch through landing; actuators to turn the control drums; and the usual array of sensors, computers, and telemetry equipment. A horizontal launch, horizontal landing launch vehicle, such as envisioned in this report, would

[19] Generally at least 2500K. The authors of this report considered using tristructural isotopic (TRISO) nuclear fuel, but found this requirement disqualified it for this application. TRISO would fail at the temperatures this report aims for. Operating at TRISO's peak safe temperature, below 2000K (Wells et al., 2021), would yield much lower performance, enough that this report's authors were not certain whether a TRISO-based engine could deliver a thrust-to-weight ratio of at least 1, without which it could not perform this mission. If it could, it would require far more nuclear fuel for the same payload to orbit, which would overcome any safety advantage that equal masses of TRISO might have over cermet.

To visualize this, imagine a large TRISO-based nuclear power plant, inefficient enough - due to the physics behind its design, in this analogy - that its output is the same as a small modular reactor that uses cermet at a higher temperature than TRISO can survive. Both options achieve the same result but the small modular reactor has less risk to the public, as well as less environmental impact, even though it does not use TRISO.

[20] A target chosen to represent what was initially believed to be the high end of achievable temperatures. Results suggest that even higher temperatures may be possible, resulting in better performance and propellant efficiency.

[21] There is concern about the ability of a control rod, aligned with the axis of thrust, to reliably operate while the vehicle is under thrust. More robust and independent control drums were favored instead. This report does not explore horizontally-inserted control rods, which could avoid issues from being aligned with the axis of thrust.

[22] Having been purged by a bit of cold hydrogen just before rotating the control drums and criticality achieved.

also possess wings and landing gear, primarily designed to accommodate landing, although the wings would also be of minor assistance during the ascent phase.

Launch operations are slightly modified from the usual sequence for chemical rocket launch vehicles. While elements of on-orbit nuclear thermal propulsion operations appear in this sequence, this is first and foremost a single-stage-to-orbit launch vehicle[23].

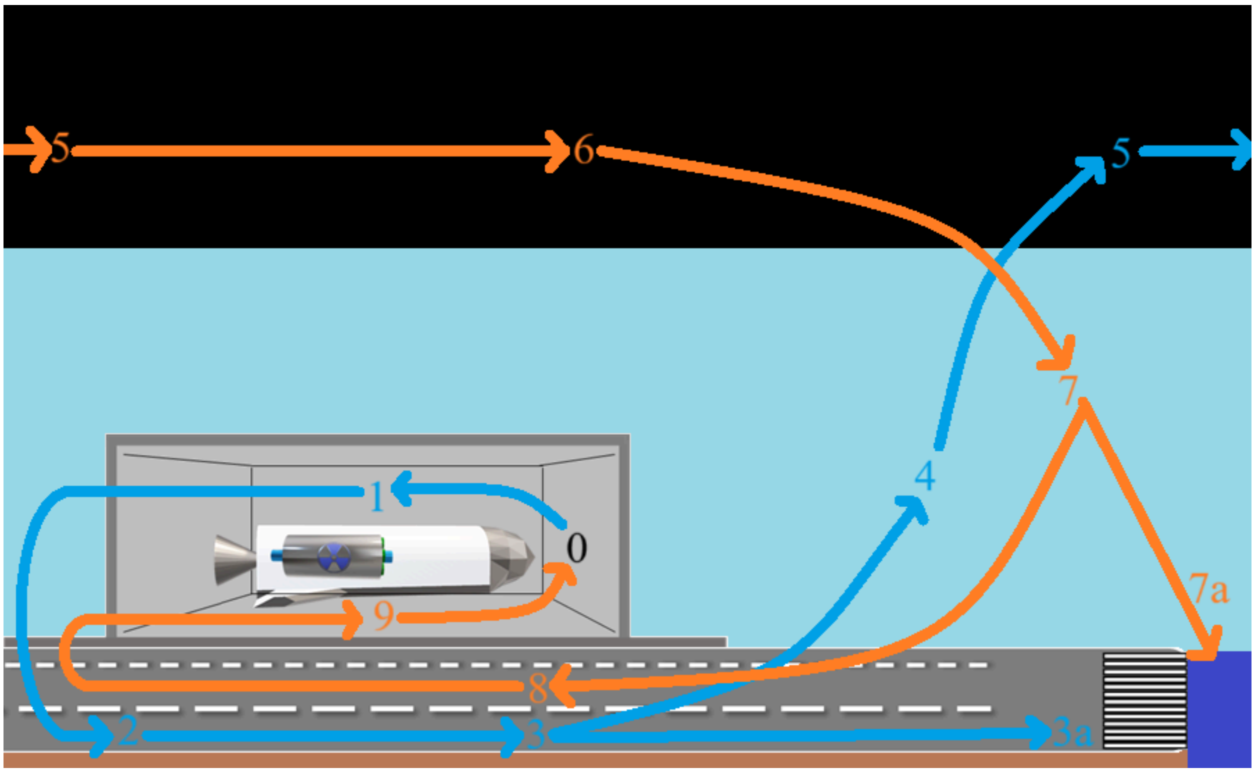

Figure 1. Diagram of concept of operations, showing each phase's location (from bottom to top) in the hangar, on the ground, in the atmosphere, or in orbit. Blue lines and numbers 1-5 are takeoff phases, orange lines and numbers 5-9 are landing phases, and the black number 0 is the outside-of-launch phase. Phase 5 includes multiple orbits not shown. Phases 3a and 7a are abort options in the event that phases 4 or 8, respectively, become not safe or not possible. This diagram is not to scale.

[23] Several alternative models were considered and found inferior. Using an existing aircraft to carry or tow the launch vehicle to altitude far from populated areas before achieving criticality would expose the aircraft's crew to high amounts of radiation, even ignoring increased risks and accident scenarios, as there do not appear to be commercially available uncrewed aircraft large enough to carry the smallest launch vehicle this report contemplates. Building a second stage to perform this function and then fly back to the launch point would need lower performance, but would incur substantially higher costs, may introduce more risk than it saves, and thus might not prove viable, especially for smaller prototypes. Using alternative propulsion for lower-atmosphere operations (as distinct from a second stage in that the entire vehicle proceeds to orbit), such as chemical rockets or MITEE-B-like (see General MITEE-type Reactor Geometry and Materials) electricity generation to power a propeller, would likewise appear to introduce more complexity and risk than it would remove, though it remains an option for future development.

0. Outside of Launch Operations: The reactor is maintained in a shielded hangar in a zero-power configuration (aside from any ground testing) under NRC jurisdiction until final launch countdown.
1. Pre-Flight Checklist: The start of this checklist marks the formal transition from NRC jurisdiction to FAA jurisdiction. Operators conduct pre-flight checks, load the payload[24], and fill the hydrogen tank[25].
2. Reactor Start-Up: The launch vehicle is brought to the launch point (launch pad for vertical launch, or runway for horizontal)[26]. Operators bring the reactor to initial criticality at minimal power.
3. Launch Commit: Operators increase criticality and power, begin propellant flow, build initial thrust, and release hold-down clamps (brakes, for horizontal launch).
    a. On-runway Abort Option: For this study's envisioned horizontal launch prototype, if a malfunction is detected at this time, the operators bring the reactor subcritical, apply the brakes, and scrub the launch without the launch vehicle leaving the ground. Operators keep propellant flowing, at a reduced rate that will not produce thrust the brakes can not overcome, until the reactor cools sufficiently.
4. Ascent Phase: Reactor power increases to full rated amount after the vehicle clears the launch facility, then throttles as needed to maintain optimal orbital insertion profile.

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[24] For some customers, the fact that the payload is loaded just before launch, rather than days or months in advance, would be the most substantial change. This enables several types of last-minute payload handling, such as having an agent show up on launch day with a classified payload in a sealed container, set up a temporary shroud around the launch vehicle, load the payload (closing the launch vehicle's payload port afterward), then lift the shroud: barring launch accident, no one else could get eyes on the payload from then until it is deployed in orbit. Other payloads that require special pre-launch handling, or that need to be launched soon after loading, would likewise be easier to accommodate.

[25] Propellant loading, and as much other preparation work as possible, happens in the hangar rather than at the launch location. Unlike a chemical rocket, a nuclear thermal launch vehicle is not made substantially more dangerous by having its propellant on board, as detailed in the next footnote. This makes the loading and preparation steps safer, to the point that they can be more safely done in the hangar. Once the launch vehicle is at the launch location, there should be a minimum of ground support equipment attached, preferably none. Another reason to choose horizontal takeoff for a prototype nuclear thermal launch vehicle is so there is no launch mount mechanical interface (which may degrade from exposure to multiple launches, and might cost more than a runway to maintain - see the note about asphalt in the Economic Analysis appendix - but larger launch vehicles with better radiation shielding may overcome this issue).

[26] This might be a few minutes or less than a minute before launch, but much less than an hour, with a total distance from the hangar of no more than a few dozen meters. Among other reasons for the short duration, the launch vehicle was loaded with cryogenic propellant in the hangar, which would boil off if launch preparation took several hours. It may be possible to get the launch vehicle to the launch location and then load it with liquid hydrogen, similar to chemical rocket fueling, but it would be safer to do this in the hangar under more controllable conditions - though equipping the hangar's propellant loading system with a hose long enough to reach the launch location, to enable rapid removal of propellant in an emergency, may be useful as a backup. The same ventilation control that checks for escaped radionuclides in the hangar during ground tests and maintenance, can provide venting and monitoring for excess hydrogen buildup, which enables better testing for propellant leakage. If there is a significant risk of substantial hydrogen leakage from the launch vehicle while it transitions from the hangar to the launch location (such as potential cracks in the insulation), or if the hangar's ventilation monitoring detects unexpected significant hydrogen leakage during fueling, the launch vehicle is not flightworthy and the launch should be scrubbed.

5. Orbital Operations: The reactor is brought subcritical and propellant flow ceases after the vehicle achieves target velocity, then the reactor shifts to decay heat removal mode[27]. At the appropriate time and place in orbit, the payload is ejected[28].
6. Deorbit Initiation: Once the engine has sufficiently cooled down and fission products have sufficiently decayed (a few hours after orbital insertion), the launch vehicle turns around and starts a new, short burn phase to decelerate, just enough to begin reentry (and using up most of the last of the propellant). This burn must be kept short for multiple reasons, including limiting the amount of fission products that are introduced[29]. It then turns again to achieve an optimal aerobraking profile, while shutting down the reactor for the final time during this flight. This maneuver is timed so the subsequent aerobraking will wind up with the vehicle near the desired landing area.
7. Reentry: The vehicle aerobrakes to decelerate to subsonic velocity. This phase is essentially ballistic, controlled by setting it up during the previous phase.
    a. Landing Abort Option: if for whatever reason the vehicle is not safe to or cannot land under active control, the ballistic trajectory of reentry carries it to a designated crash site away from any populated area[30]. This is completely passive.
8. Landing: If the vehicle lands horizontally, it glides to the landing runway. If it lands vertically, it maneuvers to its landing facility. Either way, after landing it returns to the hangar.
9. Post-Flight Checklist: Post-flight safety checks are conducted. This phase is primarily concerned with securing the launch vehicle, and does not include after-action reports or

[27] One option the authors of this report examined was to reuse the spaceplane's heat shield as essentially a "heat sponge" to transfer the reactor's heat to, once propellant is no longer removing this heat, since the heat shield is a significant mass that is not otherwise made use of outside of aerobraking. The heat shield would thus be designed to passively radiate absorbed heat, whether from the reactor or after aerobraking, and would need to dispose of most of the reactor's heat prior to aerobraking. This report focuses on the engine, though.

[28] This analysis assumes there is no other significant interaction between the launch vehicle and the payload. An explicit lack of such interaction is part of the CubeSat Design Specification (Johnstone, 2022) - as is a means for the payload to detect that it has been ejected - and might be specified for secondary ESPA class payloads. In general, this report assumes CubeSat-like operations in all payload-specific matters. More sophisticated operations, such as docking with a space station to transfer a payload, are physically possible (with additional regulations depending on the operation, such as getting permission from the station's owner) but beyond the scope of this analysis.

[29] The deorbit burn and subsequent atmospheric reentry happen after fission products from the earlier burns have decayed, to quote SPD-6, "to a level of radioactivity comparable to that of uranium-235" (Trump, 2020). Since reentry commences shortly after the deorbit burn, the level needs to still be comparable even with the deorbit burn's fission products. The phrasing of SPD-6 is somewhat vague on this point, and this is one of the points which likely will not be defined concretely until someone attempts to implement this. A case can be made that "of uranium-235" (Trump, 2020), with no further qualification and mentioning no other substance, means 100% uranium-235. As this report's design uses no more than 20% uranium-235, there is room in the system to have some degree of fission products while still being under, let alone comparable to, the level of radioactivity of 100% uranium-235.

[30] For instance, a launch vehicle operating out of a spaceport on the eastern coast of the continental United States might aim itself at the Atlantic Ocean. If all goes well, during the landing phase it corrects its course to land at the spaceport. If instead the launch vehicle breaks up during reentry, it lands in the ocean, with the engine's exhaust flap closed to prevent seawater from intruding into the engine before completing recovery. Failing that, analysis showed that the liner of 1.4 mm of tungsten, as suggested in a footnote in the Introduction section of this report, is enough to survive atmospheric reentry and keep the uranium intact until reaching sea level. Even if the engine core shattered upon hitting the ocean, it would quickly sink to the seafloor, resulting in a cleanup area less than 100 meters across.

other actions that are conducted between flights. Once flight operations formally end, jurisdiction reverts from FAA to NRC, where it remains until the next flight.

The design incorporates multiple safety systems. The most important one is the launch exclusion zone: driven more by radiation concerns than concerns over potential explosions, there is still a minimum keep-away distance for the public[31]. The control drums are independently controlled so that, if a few[32] seize up in a criticality-promoting position, the rest can shut down the reactor. Range safety includes remote control to safely shut down the reactor in contingency scenarios[33]. This report does not address whether the spaceplane would be fully autonomous once a safe distance from the launch site, or remain under remote control throughout the entire mission (with the assistance of ground stations around the world). Adequate cybersecurity is assumed[34].

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[31] For a SpaceX Falcon 1 launch, the public viewing distance is a few miles - some sources claim 5 or more, though according to Headout, Kennedy Space Center's LC-39 Observation Gallery is only 2.3 miles away (Headout Inc, 2025). By comparison, both of the alternatives examined in the MITEE-C Radiation Shielding Study section of this report achieve a public keep-away distance of less than 1 kilometer, though that analysis only considers radiation hazards.

In practice, noise may be the dominant consideration. A full analysis would require a full launch vehicle design, but a simple analysis can be presented here. The target exhaust velocity is 10,000 m/s, which is a bit over Mach 29. The FAA considers 65 dB to be the threshold of significant noise exposure (FAA, 2022). If the engine during takeoff (which will not be at full power, but may still have an exhaust velocity over Mach 1) created 135 dB at a 1 meter distance (potentially with the assistance of spaceport improvements, though this would only apply to the sound level on the runway), 65 dB would be reached about 3.16 km away. Keeping the exhaust velocity subsonic during takeoff (until several km away from the public) would reduce this need, at some cost to overall performance (specific impulse would greatly decrease during this phase, but takeoff is a small part of the total delta-v).

[32] The exact number depends on the design, but ideally a simple majority can bring k-effective below 1.

[33] In the event of a malfunction, off-nominal trajectory, or other event requiring premature engine shutdown after leaving the ground, the procedure is: shut off propellant flow, rotate the control drums to the criticality-suppressing position, and close the flap at the exit of the engine. Depending on the position, velocity, and condition of the launch vehicle when this determination is made, a horizontal landing launch vehicle may be able to glide and have some control over its landing site. If this happens shortly after takeoff, returning to the launch site will probably be preferred if feasible.

Given the potential for dispersal of radiological materials, an explosive flight termination system is not advised. Under every accident scenario considered where there might be an option to trigger such a system (which excludes situations such as total loss of control), blowing up the engine would likely induce or increase breaches in the engine cladding, relative to the same scenario if no such detonation occurred, resulting in worse consequences than keeping the engine intact as long as possible. For example, 7a involves the engine coming down fairly intact; even if the engine were to breach on impact, cleanup will be over a much smaller area than if a detonation was triggered while the launch vehicle was at an altitude of several kilometers. Furthermore, an analysis showed that the presence of such a system added risk in scenarios that would not, save for the capability being present, become accidents.

[34] Given the limited amount of commands during a given mission, it may be feasible to equip the spaceplane with a one time pad generated just before a flight, using it to authenticate all incoming commands. Outbound telemetry is less sensitive and does not need this protection - fortunately, as there will likely be substantially more outbound data (primarily telemetry) than inbound (primarily commands).

Readers familiar with traditional launch operations may notice the lack of several steps detailed in the above list. Many of these[35] occur as normal for any launch. Certain others[36] do not exist for NTP, or at least for this model of operations. Simplification of operations is a passive safety advantage.

In the event of an accident resulting in a crash within US territory or in international waters, the FAA and National Transportation Safety Board (NTSB) would have primary jurisdiction over the aviation and launch vehicle aspects of the incident, while the Nuclear Regulatory Commission (NRC) would have primary jurisdiction over the nuclear materials components of the response and would likely serve as the lead federal agency under the Federal Radiological Emergency Response Plan (FRERP). The Environmental Protection Agency would have secondary jurisdiction, such as for environmental assessment, establishing cleanup standards, and overseeing long-term environmental remediation and monitoring. Each agency would maintain authority within their domain under the FRERP's Unified Command structure.

To avoid complications with interactions with existing launch infrastructure[37], and because using existing launch infrastructure might not be possible[38], this report assumes launch will happen from a spaceport that is not yet licensed[39] as of May 2025, that will have been stood up specifically to support nuclear thermal launch vehicles. Landing would happen at this same facility, to avoid having to transport the vehicle from landing site to launch site between

[35] Such as range safety, weather, and other verifications grouped under "pre-flight checks". Filing notices to airmen and mariners would also occur for FAA-licensed launches, and is notably the main requirement preventing FAA-licensed launches from happening on a moment's notice, regardless of technical capability. There is an option for launches licensed under the Department of Defense to launch on demand: if a launch vehicle, ground crew, facilities, and payload were prepared in advance, a launch could begin within several minutes of a launch order being given. While this system would be physically capable of supporting such a use (needing only a change in license), a full analysis of that scenario is beyond the scope of this report.
[36] Most notably ignition (replaced with criticality) and staging-related steps (since this is a single stage vehicle).
[37] The maximum probable loss insurance requirement from potentially shutting down Kennedy Space Center, or most other spaceports that exist as of May 2025, makes launching a prototype nuclear thermal launch vehicle from there infeasible.
[38] At least for the prototype launch vehicle: see the note in the Conclusion about a hot cell.
[39] It might be inaccurate to say "does not exist", as it is possible this spaceport may be converted from an existing but little-used airport, where current traffic could be redirected to nearby airports with minimal impact. The North Carolina Department of Transportation (NCDOT) is, as of May 2025, conducting a high-level analysis of suitable existing airport facilities to support this project, if and when a nuclear thermal launch vehicle reaches high enough maturity to need such infrastructure. As detailed elsewhere in this report, dedicated infrastructure of this nature will likely not be called for until the time when a prototype engine reaches the point where it needs to be tested with nuclear fuel.

missions. This spaceport's runway[40] and infrastructure[41] would be designed accordingly, so buildup of neutron activations from supporting 100 launches per year does not cause significant environmental issues.

Radiation shielding is a critical concern, and a key example of regulations setting technical objectives, which do not necessarily apply to similar systems such as on-orbit-only NTP or combustion-powered launch vehicles. Meeting the FAA's limits[42] can be done by keeping maximum public exposure per flight to no more than 25 millirem[43]. Given noise concerns and resulting public keep-away distances at existing spaceports, this report assumes the public can be kept at least 1 kilometer away from the launch site, which means the radiation emissions objective can be met by keeping exposure at 1 kilometer for the duration of a launch to no more than 25 millirem. The mass of radiation shielding needed to accomplish this significantly decreases the engine's thrust-to-weight ratio, though not to a degree further optimization cannot overcome, even using only the radiation shielding materials that could be cited in this report.

The subsequent sections of this report detail the regulatory requirements this engine would operate under, then detail a technical approach to addressing the key engineering challenges in realizing this concept of operations.

[40] NCDOT is primarily considering conversion of airports with asphalt runways. While thermal management would be a concern (necessitating installation of cooling systems for the runway), long term neutron activations would be less of a problem than with concrete runways, though may still require rehabilitation of the runway every 5 years rather than every 10 to 15 as is typical for conventional asphalt runways. It is conceivable maintaining the runway might happen roughly every 4 years alongside replacing the nuclear fuel, especially if multiple launch vehicles use the same runway. See the Economic Analysis appendix for the resulting financial impacts.
[41] Structures significantly further away from the launch path than the runway - such as the hangar, control tower, and on-site public viewing booths - can, with proper material selection (such as favoring borated materials, and aluminum rather than steel for structural components), have much longer lifetimes than the runway. If an airport is converted into a spaceport, conversion may involve lining the runway-facing sides of structures with radiation-resistant materials and replacing the windows.
[42] The NRC's limits are met by having the spaceplane stay inside a hangar with 1-2 meter thick concrete walls while it is not under FAA jurisdiction. This provides adequate protection regardless of how much radiation shielding is on the spaceplane or whether said shielding fails during testing. If the launch vehicle's shielding fails while inside the hangar, the launch vehicle will conduct no further launches until the shielding is repaired, tested, and proven once more. The same attitude should be taken toward the rest of the spaceplane's systems: if it is not flightworthy then it is not flightworthy, no matter how important or urgent getting a certain payload to orbit is. The goal is to achieve approximately 100 flights per year per launch vehicle safely, not just to achieve 100 flights per year at any cost.
[43] Outside of accident scenarios, which would need a full launch vehicle design - which is beyond the scope of this report - to fully analyze, but the requirement can be summed up as so: accidents resulting in exposure to more than 25 millirem should be kept to a total chance of no more than 1/1,000,000.

Justification of HALEU

For the specific case of a high-cadence commercial launch vehicle using nuclear thermal propulsion, launched from the United States under the regulations in effect as of May 2025, HALEU is the only viable option. The rest of this section goes into detail but the main proximate reasons can be summarized briefly:

- NSPM-20 makes any nuclear fuel other than some form of LEU impractical for this launch vehicle.
- Of the LEUs, only HALEU can do the job adequately.

These reasons do not separate into purely regulatory and purely technical. The first part is a technical constraint for regulatory reasons, while the second part is a regulatory constraint for technical reasons.

Technical-Because-Regulatory: Only a LEU

NSPM-20 requires notification of the President for each launch of nuclear material other than low enriched uranium (Trump, 2019). Although this requirement was written in regard to nuclear material in the payload, the exact text does not differentiate: if any other form of nuclear material is anywhere on the launch vehicle, this notification is required. Therefore, the requirement also covers nuclear material in the engine. For the launch vehicle contemplated in this report, this would mean notifying the President 100 times per year. This was dismissed as infeasible: bureaucratic effort aside, it would be an invitation for the administration to modify or suspend authorization for every notice. By contrast, a launch vehicle using only low enriched uranium can get a launch license designed to facilitate regular, routine operations; with some administrative steps carried out for each launch, such as filing notices to airmen and mariners, these steps are much more conducive to routine operations with far less reason to impose last-minute holds.

Additional reasons abound why only low enriched uranium is viable for this purpose. For example, the estimated cost of an engine built with HEU is $15,000,000 per kilogram, while HALEU is only $25,000 per kilogram[44]. As seen in the Economic Analysis appendix, the former would not be economically viable given the expected program budget, while the latter fits within the budget. Security costs, and the increased likelihood of incidents from those wishing to steal the HEU, further make HEU economically infeasible for this purpose. A number of individuals the authors of this report spoke with over the course of this study dismissed the importance of economic feasibility, but if a commercial program - which, by definition, relies on commercial

[44] According to nuclear fuel manufacturers the authors of this report spoke with over the course of this study.

revenue to survive - is not economically viable, it is no more viable than a rocket engine that requires propellant to be continually teleported in[45].

As difficult as it would be to work around the regulatory and financial hurdles described above, the ultimate problem can be classified as a technical one. As this is envisioned for use by a commercial program[46], acquiring weapons-grade HEU would not be feasible[47]. By contrast, commercial use of HALEU has been authorized and is ongoing (U.S. DOE, 2025). If a certain nuclear fuel cannot be obtained, an engine and launch vehicle using that nuclear fuel cannot be constructed. A thing that cannot be built is, by definition, not technically feasible.

SPD-6 is sometimes cited as requiring HALEU, but it only requires low enriched uranium for development of and use by United States government agencies, and then only if the mission is not viable[48] with low enriched uranium (Trump, 2020). Commercial development and use, if not specifically funded by any government agency, would technically fall outside the specific language of SPD-6. Still, using HALEU opens up government funding for development, which would not be available for a launch vehicle using HEU or any non-uranium nuclear fuel.

More importantly, SPD-6 says, “DOE, in cooperation with NASA and DoD, and with private-sector partners, as appropriate, should identify feedstock and uranium that can be made available for planetary surface power and in-space propulsion demonstrations” (Trump, 2020). This can be read as requiring provision of HALEU for a prototype launch vehicle similar to the one described in this report. Having an identified pathway to obtain the required nuclear fuel makes the program more feasible, but accepting such nuclear fuel makes the program covered under SPD-6 if it was not already[49], and thus requiring HALEU if viable.

[45] As of May 2025, the authors of this report have yet to observe a reliable, repeatable demonstration of non-quantum mass teleportation that would be relevant to this use case.

[46] It might be theoretically possible for a government program to acquire HEU for this purpose, using Department of Defense licensing to bypass NRC and FAA regulations, making the question of regulatory feasibility moot. This sort of program would likely be funded on a cost-plus basis, and thus disregard economic viability. However, such a launch vehicle would likely only be able to serve a small number of government missions, greatly limiting its utility. This is one of the reasons this report considers an exclusively or primarily commercial use case instead. Further, such a government program is discouraged by SPD-6: “The use of highly enriched uranium (HEU) in SNPP systems should be limited to applications for which the mission would not be viable with other nuclear fuels or non-nuclear power sources” (Trump, 2020). As this report shows, HALEU is viable for this mission.

[47] One might generally assume this to be the case. We checked. It is. Also, many details about said investigation we are not allowed to cite in a public report such as this. Suffice it to say: as of May 2025, it is confirmed that acquiring HEU - in, and for use in, the United States - for a launch vehicle owned and operated by a non-government entity is difficult enough to be infeasible, and that this limitation will not be lifted in the foreseeable future, to as high a degree of certainty as any such regulatory matter can be predicted.

[48] “Viable” in SPD-6 can be read as “technically feasible” in the sense this report uses.

[49] Since this would make the DOE “involved in the development and use of” this system.

Regulatory-Because-Technical: Of LEUs, Only HALEU

Of the forms of low enriched uranium, only HALEU provides enough heat per unit mass to be viable. HALEU has a much lower power density than the highly enriched uranium used by many other designs, including MITEE, but as described in other sections, it can suffice with some optimization. It is apparent on brief examination other forms of low enriched uranium, given their substantially lower power density than HALEU[50], are likely not practically capable of performing this task, but this report provides further examination.

In practice, only three forms of low enriched uranium exist:

- HALEU, at 19.75% or 19.5% enrichment.
- LALEU, at 3-5% enrichment.
- Natural uranium[51], at about 0.7% enrichment.

As of May 2025, there do not appear to be available supplies of uranium under 20% enrichment at any other specific grade. While it would be technically possible to set up a fuel supply at 10% or 15% enrichment, this is infeasible from a regulatory point of view when HALEU exists, can do the job, and 10% or 15% would be under the same regulatory scrutiny as HALEU[52].

A MITEE-like configuration simulated using OpenMC showed the following k-effectives at the following enrichment levels, changing only the enrichment percentage between simulations:

- 19.75%: 1.00935 +/- 0.00101
- 19.5%: 1.00308 +/- 0.00094
- 5%: 0.70947 +/- 0.00072
- 3%: 0.62690 +/- 0.00090
- 0.7%: 0.41865 +/- 0.00074

Results using other configurations followed similar patterns. As can be seen, enrichment levels less than HALEU yielded k-effective substantially less than 1, and thus would not produce significant heat. It is possible to get k-effective above 1 using large enough amounts of

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[50] Although the enrichment percentage does not necessarily linearly map to performance, it provides a close enough overview for a cursory analysis. HEU, at around 90% enrichment, could power a NTP launch vehicle. HALEU, at just under 20% - around 4.5 times lower than HEU - can power a NTP launch vehicle with a lot of fine tuning. LALEU is 3-5%, about 4-6.5 times less than HALEU. Those without expertise in nuclear engineering or rocket science can take the page count of this report as evidence that getting HALEU to perform in this role is not a simple matter. LALEU would have much higher difficulty. This is not absolute proof of the technical infeasibility of a LALEU-powered launch vehicle, but as subsequent analysis shows, the actual relevant hurdle is regulatory.

[51] Arguably this does not count as low enriched uranium for NSPM-20's purposes, but even assuming it does, it is inadequate.

[52] In theory it might cost less to provide a lower enrichment, except for needing to set up a supply chain just to provide this lower enrichment grade. By contrast, a supply chain to provide HALEU is already being set up as of May 2025.

LALEU[53] or even natural uranium[54], but these are generally larger than the systems contemplated in this study.

Although this report does not rule out the possibility a nuclear thermal propulsion system using LALEU could be designed, the minimum scale necessary appears to be much larger than the minimum viable prototypes contemplated in this report, coming with far greater risk to the public[55] for no apparent benefit other than cost to replace the nuclear fuel[56], which would be outweighed by the larger development cost[57]. Thus: using LALEU might be feasible from a technical perspective, but for technical reasons it is not feasible from a regulatory perspective, at least before the safety and performance of a nuclear thermal launch vehicle has been proven using something other than LALEU.

[53] This is how most nuclear power plants work.
[54] Such as in CANDU reactors.
[55] As of May 2025, the authors of this paper are not aware of any relevant class of safety measure which is only possible, or reduces risk by a substantially greater amount, at scales much larger than the engine proposed in this study. Conversely, “How much harm is done to the public in the event of a rapid unscheduled disassembly?” is lower if there is less nuclear fuel present, assuming everything else is kept the same or, more realistically, if the amounts of other potentially hazardous material are likewise less.
[56] See the Economic Analysis appendix.
[57] If profit is the only motive, it would cost less to develop a smaller HALEU-powered launch vehicle, put it into operation, and use the lessons learned to reduce the cost to develop a larger LALEU-powered launch vehicle. Furthermore, over any time horizon likely to be imposed by typical funding sources, the even-lower development cost of a HALEU-powered launch vehicle, relative to a LALEU-powered launch vehicle of the same size, would make up for the higher cost of replacement HALEU, especially given HALEU’s likely decrease as production increases.

Regulatory Path to Commercial Operations

Nuclear thermal propulsion represents a transformative technology for space launch vehicles, offering substantially superior performance as measured by a significantly higher specific impulse (in the case of the engine in this study, 1000 seconds) compared to chemical propulsion systems. This performance advantage translates to greater payload capacity, reduced launch costs, and expanded mission capabilities. However, the deployment of nuclear systems in launch vehicles presents unique regulatory challenges at the intersection of nuclear safety and space launch operations.

This section examines the regulatory framework applicable to nuclear thermal propulsion launch vehicles in the United States, with specific focus on small launch systems given the payload sizes discussed in the Spacecraft section of this report. While no commercial entity has yet licensed an NTP launch vehicle, this analysis demonstrates existing regulatory structures, though not specifically designed for nuclear launch, provide a viable pathway to authorization and operation.

Space Nuclear and the U.S. Constitution

The Constitution of the United States separates the power of government between its three branches, the executive, legislative, and the judiciary. The basis of government authority in the US form of government is one of limited authority, the powers not explicitly granted to it are reserved to the States. The Constitution combines the principle of separation of powers with the concept of a limited government to establish a system of checks and balances on the power of government exercised by each of the three branches of government. The focus in this report is on the split of power between the executive and the legislative branches.

Legislative and Executive Constitutional Powers

The U.S. Constitution provides the foundation for regulatory authority over nuclear and space activities. Article I, Section 8 grants Congress the power to “provide for the common Defense and general Welfare of the United States” and to “regulate Commerce with foreign Nations, and among the several States.” These powers authorize Congress to legislate on matters of nuclear energy and space transportation.

The executive branch, under Article II, implements and enforces the laws enacted by Congress. For nuclear thermal propulsion, this implementation occurs primarily through the Nuclear Regulatory Commission (NRC) and the Federal Aviation Administration (FAA), with additional input from the Department of Energy (DOE), NASA, and other federal agencies.

The President of the United States has the most authority to govern where the Constitution has explicitly stated the President's power or the Congress has given the President specific powers[58]. There is a gray area where either is silent. For instance, the Constitution vests the President with full authority of government in matters of foreign relations, except the Senate must consent to any treaties which the President ratifies. Similarly, Congress gave the President authority to conduct nuclear energy research.

Legislative Powers and Delegation

It is well settled law[59] that Congress has authority under the Constitution to govern and regulate interstate commerce, as well as to provide for common defense and general welfare. Using this authority, Congress enacted several key statutes governing nuclear activities and space transportation:

- The Atomic Energy Act of 1954 (AEA), as amended in 42 U.S.C. § 2011 et seq., which provides the fundamental legal framework for civilian and military uses of nuclear materials in the United States ("Chapter 23...," 2023).
- The Commercial Space Launch Act (CSLA), as amended in 51 U.S.C. § 50901 et seq., which establishes the framework for licensing commercial space transportation activities ("Chapter 509...," 2023).
- The Energy Reorganization Act of 1974 - 42 U.S.C. § 5801 et seq., which created the Nuclear Regulatory Commission ("Chapter 73...," 2023).
- The National Aeronautics and Space Act - 51 U.S.C. § 20101 et seq., which established NASA and provides for civilian space activities ("Chapter 201...," 2023). Note that NASA is not a regulatory agency: while it has provided input into the development of regulations regarding space[60], it does not make the rules.
- The Accelerating Deployment of Versatile, Advanced Nuclear for Clean Energy Act of 2024 (ADVANCE Act), which modernizes nuclear regulation by streamlining licensing processes for advanced nuclear reactors, including provisions for testing and demonstration of new nuclear technologies.

Through these statutes, Congress has delegated specific regulatory authorities to executive agencies. For nuclear thermal propulsion launch vehicles, the most relevant delegations are to the NRC for nuclear systems and the FAA for launch activities.

[58] See concurring opinion of Jackson in *Youngstown Sheet & Tube Co. et al. v. Sawy*er, (1952).
[59] Such as in *United States v. Lopez* (1995).
[60] Compare NASA's NPR 8715.26, effective through 2027 (NASA, 2022) - to FAA Advisory Circular 450.45-1 (Nguyen, 2023) - which can be respectively seen as each agency's current, as of May 2025, leading guidance on the launch of spacecraft containing nuclear materials. A launch vehicle using NTP would fall into the category of "a spacecraft containing nuclear materials", in lieu of specific regulations about NTP launch vehicles per se.

Executive Powers and Delegation

Between the AEA, the CSLA, and others, Congress has given the executive branch authority to license and regulate nuclear activities and space activities. This includes activities which are both, such as a NTP launch vehicle, even if Congress has not explicitly addressed this particular combination.

The executive branch has acted on this authority. NSPM-20 authorizes the Secretary of Transportation to delegate authority to license launches of nuclear material, which authority has been delegated to the FAA. The FAA also has authority to license launch vehicles; the governing legislation makes no distinction for NTP launch vehicles, so the FAA's authority covers NTP launch vehicles too.

The Constitution gives the President other powers, but these do not apply to licensing and regulation of commercial systems. The executive branch's authority in this matter stems from the authorities Congress has granted.

As further evidence of Congressional consent, multiple sessions of Congress have taken place since August 20, 2019, when the President issued NSPM-20, and first authorized the Secretary of Transportation or designee to approve certain nuclear launches. Congress passed the ADVANCE Act in July of 2024, nearly 5 years later, so it had plenty of opportunity to revoke the authority given in NSPM-20. Clearly, Congress's failure to act loudly pronounces nuclear launch as described in NSPM-20 is an acceptable way to operate space nuclear systems. Put another way, Congress implicitly acknowledges the executive has authority to license space nuclear systems pursuant to NSPM-20. Finally, action taken to hasten adoption of small modular reactors arguably includes consideration for their application in space, as the ADVANCE Act was passed during a time of accelerating interest in space following three executive orders addressing and authorizing space nuclear systems as well as numerous scholarly and news media articles covering space nuclear systems over the past few decades.

Role of Treaties and International Agreements

The United States is party to several international treaties and agreements that influence the regulation of nuclear and space activities:

- The Treaty on Principles Governing the Activities of States in the Exploration and Use of Outer Space, including the Moon and Other Celestial Bodies (the "Outer Space Treaty").
- The Convention on International Liability for Damage Caused by Space Objects (the "Liability Convention").
- The Convention on Registration of Objects Launched into Outer Space (the "Registration Convention").

- The Treaty on the Non-Proliferation of Nuclear Weapons (the "Non-Proliferation Treaty").
- IAEA commitments with the force of ratified treaties, such as the Convention on Early Notification of a Nuclear Accident.

Treaties under the U.S. Constitution can either be self-executing or require legislative action to become binding federal law. Under Article II of the U.S. Constitution, treaties constitute "the supreme Law of the Land". The United States therefore has an obligation to ensure its domestic regulatory framework complies with these international commitments.

The first three treaties listed above incorporate into FAA launch license regulations and, save for the potential for radiological damage to objects in space (which is largely addressed by the same measures that prevent said damage to people on the ground), do not present significantly different procedures for NTP launch vehicles than for chemical launch vehicles. The non-proliferation treaty helped inform NSPM-20, insofar as HALEU is generally seen as not weapons-grade material. None of these outright forbid the construction, operation, or licensing of NTP launch vehicles, although they affect relevant details, such as the Convention on Early Notification of a Nuclear Accident's reporting requirements in the event of an accident.

Nuclear Legislation, Launch Legislation, and Their Combination

While there exists a significant body of legislation about possession and use of nuclear materials, and a significant body of legislation about activities and objects in space, Congress has passed no law which combines these issues as of May 2025.

Nuclear Legislation

The cornerstone of nuclear regulation in the United States is the AEA, as amended. The AEA established a comprehensive regulatory framework for civilian and military uses of nuclear materials, with a focus on ensuring public health and safety while promoting the beneficial development of nuclear energy.

The AEA requires licenses for the possession, use, and transfer of nuclear materials, as well as for the operation of nuclear facilities. If a NTP engine were operated on the ground, the facility it is operated at would qualify as a "nuclear facility" for this purpose, and thus need licensing. Likewise, transportation of the nuclear fuel - or of the engine once it contained nuclear fuel, or of the launch vehicle as a whole (other than while under FAA jurisdiction) once it had an engine containing nuclear fuel - would be subject to regulations from the NRC ("Part 71...," 2021) and the Department of Transportation ("Transportation: Subchapter C...," 2025).

The AEA directs the NRC to establish standards for radiation protection, nuclear security, and nuclear safety. These would apply to the design, construction, and operation on the ground of a NTP system, though they would not necessarily apply to its operation during flight, at which time FAA jurisdiction would apply. The FAA does not have jurisdiction over ground tests of a launch vehicle, nuclear or otherwise, so ground testing of a NTP launch vehicle would happen under the NRC's standards. Likewise, nuclear fuel handling and maintenance between flights would be covered under NRC rules.

The Energy Reorganization Act of 1974 created the Nuclear Regulatory Commission as an independent agency to license and regulate civilian uses of nuclear materials and facilities.

Additional nuclear legislation relevant to NTP includes:

- Price-Anderson Nuclear Industries Indemnity Act - 42 U.S.C. § 2210: this legislation establishes a system of financial protection and indemnification for nuclear incidents ("Indemnification and...," 2023). This only applies while the launch vehicle is under NRC jurisdiction in the hangar, not to FAA-licensed operations.
- Nuclear Waste Policy Act - 42 U.S.C. § 10101 et seq.: this act governs the disposal of spent nuclear fuel and high-level radioactive waste ("Chapter 108...," 2023), with implications for the end-of-life management of NTP reactors and fuel.
- ADVANCE Act: this legislation streamlines the licensing process for advanced nuclear reactors (ADVANCE Act of 2023, 2024). The NTP engine discussed in this report arguably qualifies under this legislation's definitions. To cite a couple of the advancements, which explicitly count as an "advanced nuclear reactor", the engine contemplated in this study has lower waste yields compared to commercial nuclear plants in operation in 2019, and in theory has an "ability to integrate into electric and nonelectric applications": while the nonelectric application is its primary use, one could dismount the engine from a launch vehicle and attach a turbine instead[61].

Launch Legislation

The CSLA, as amended and codified at 51 U.S.C. § 50901 et seq., establishes the framework for commercial space transportation in the United States ("Chapter 509...," 2023). The CSLA designates the Secretary of Transportation as the authority for licensing commercial launches and reentries. This authority has been delegated to the FAA's Office of Commercial Space Transportation.

The FAA has implemented the CSLA's licensing, safety, financial, and other requirements through regulations via 14 CFR § 450 Launch and Reentry License Requirements, which

[61] The MITEE-B architecture mentioned later in this report was designed to generate thrust and electricity, though it would require significant modification to this report's design, so the potential to adopt this architecture likely does not fulfill this requirement.

establishes a performance-based licensing regime for commercial space transportation ("Part 450...," 2024). The primary consideration is safety to the public[62].

Integration of Nuclear and Launch Regulatory Frameworks

While no specific legislation addresses NTP launches directly, the existing frameworks can be integrated to provide a viable regulatory pathway. A NTP launch vehicle would require both an NRC license for the nuclear reactor component - under 10 CFR § 50/52 for the reactor ("Part 50...," 2025; "Part 52...," 2024) and 10 CFR § 70 for the HALEU fuel ("Part 70...," 2023) - and an FAA license for the launch and reentry activities under 14 CFR § 450 ("Part 450...," 2024). A key factor for consideration is that jurisdiction switches: upon the start of flight activities, the FAA gains exclusive jurisdiction, which remains until the end of flight activities, which in this case means either a safe landing and return to hangar, or in the event of an accident scenario, at some point after all components of the launch vehicle stop moving (other than motion externally induced by wind, currents, cleanup efforts, et al).

Under NRC regulations, a NTP reactor would be classified as a non-power reactor (since it does not generate electricity) and possibly[63] as a small modular reactor (being under the 1,000 MW thermal threshold), allowing it to utilize existing licensing pathways. As of this report's publication, it is an open question whether the fact that most or all of a NTP launch vehicle's income would accrue while under FAA jurisdiction, would cause such income to not be considered when deciding whether it should get a Class 103 (commercial and industrial) or 104 (research and development) license, assuming it was licensed under 10 CFR § 50 rather than 10 CFR § 52, given regulatory precedent for licensing non-power reactors under 50 and 50's greater flexibility for design changes as prototype development proceeds ("Part 50...," 2025; "Part 52...," 2024).

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[62] This has been described as, "On a statistical average, how many people will one launch kill?" This phrasing acknowledges that perfect safety is impossible to achieve without explicitly forbidding the activity (and possibly not even then, especially when factoring in opportunity costs), so a launch license application must contain statistical analyses showing that the sum total of accident scenarios - multiplying the probability of accident times the expected impact of that accident - comes to no more than some figure. The exact answer has varied between this and prior versions of the regulations but has consistently been very low, such as 3/100,000: if one hundred thousand identical launches were conducted, on average only three people in total would die as a direct result of all of those launches combined. Simulations, ground tests, and suborbital flight tests will be necessary to prove this, and will likely constitute the bulk of the effort needed to achieve permission from the FAA to conduct orbital launches once the engine prototype is ready for such tests.

[63] Among the details unlikely to be definitively ruled on until after a license application is filed, is whether a small modular reactor can only be a power reactor, or if a non-power reactor can be licensed as a small modular reactor. Even if the former interpretation holds, much of the review and other licensing pathway actions for small modular reactors would still apply, making licensing the engine more feasible if it stays under 1,000 MW thermal. This report thus takes it as an objective to keep the engine below this limit.

Under FAA regulations, a NTP launch vehicle would be a launch vehicle. If it passed a licensing review, it would be licensed for operation, by definition. This review would pay more attention to certain parts:

- Unlike traditional NTP, the thrust must be essentially non-radioactive. As noted in the Introduction section, the design this report contemplates in this review anticipates and accomplishes this objective. Extensive simulation does not appear to be warranted[64], but measurement to confirm this will likely be part of the step to achieve Technology Readiness Level 6 described in the Conclusion section.
- Accident analysis would factor in, and likely be dominated by, scenarios involving damage to the engine core, primarily broken into scenarios where the cladding fails and nuclear fuel gets into the exhaust, and where the spaceplane fails and the core is sent uncontrollably into the ocean or onto the ground at significant velocity. Keeping the chances of such scenarios as low as reasonably possible, while performing well enough the launch vehicle can achieve orbit, must be a primary objective of the launch vehicle's design. Testing and simulation to prove the low odds of such scenarios will be a significant part of the effort to develop a working launch vehicle.
- AC 450.45-1 7.2.1 lists three tiers[65] of radiation exposure levels to the public, with corresponding odds of exceeding them, which if satisfied, will demonstrate operation of the launch vehicle "is consistent with public health and safety" (Nguyen, 2023). Keeping planned exposure to the public from any one flight to no more than 25 millirem[66], with a

[64] See the footnote using Schwelger's equation in the Introduction section.
[65] The odds of 0.25-5 rem to any member of the public is no more than 1/100, the odds of 5-25 rem is no more than 1/10,000, and the odds of exceeding 25 rem is no more than 1/100,000.
[66] Some have stated concern that the linear no-threshold (LNT) framework, as used by the NRC, might override NSPM-20's thresholds, but this seems to be ruled out. LNT claims that exposing 100,000 people to 0.025 rem each, below the threshold of measurable health impacts to an individual, is technically the same as exposing 1 person to 2,500 rem, a lethal dose. This equivalency is disputed by emerging scientific evidence, and this specific example is more extreme than commonly used in practice, but those aspects need not be detailed here. What matters is that NSPM-20 explicitly establishes that the 25 millirem limit applies "to any member of the public" (Trump, 2019), avoiding language such as "cumulative to all members of the public" that would invoke population-based standards.

NSPM-20's recent promulgation (2019) substantially postdates the adoption of LNT, demonstrating that it represents a modern regulatory framework, in this case specifically tailored for space nuclear applications. The memorandum's detailed radiation provisions omit any reference to collective dose limitations despite its coverage of other radiation protection concerns, strongly suggesting an intentional policy departure from LNT principles. This aligns with precedent in nuclear regulation where specialized applications received domain-specific frameworks acknowledging their unique characteristics and risk profiles. Presidential directives like NSPM-20 hold substantial weight in directing federal agency implementation, and LNT adoption originated primarily from just such an implementation. In combination with other executive orders and Congressional legislation urging reform of the nuclear regulatory process, this suggests that LNT should not be used with regard to space nuclear systems.

Further, NSPM-20 explicitly calls for a "risk informed" process (Trump, 2019). The core thesis of LNT is to assume risk levels in all situations whether or not there is direct evidence of harm in that particular situation, which is fundamentally at odds with a risk-informed approach that emphasizes evidence-based assessment. While NSPM-20 is silent on NRC licensing while the spaceplane is in the hangar (with lower emissions to the public) between flights, its emphasis on individual dose limits and risk-informed approaches strongly suggests that it directs the FAA to not apply LNT to its portion of regulatory analysis for space nuclear systems, even in consultation with other agencies.

probability of no more than 1/100,000 of an accident that would exceed this threshold, will satisfy all three. As seen in the MITEE-C Radiation Shielding Study section of this report, no more than 25 millirem of nominal exposure can be achieved without an extraordinary amount of radiation shielding. Calculating the accident probability would require a full launch vehicle design and so is beyond the scope of this report, but is a requirement that should be kept in mind for such a design.

- Insurance requirements apply as for any launch vehicle, as described under the CSLA[67]. The Price-Anderson Act provides some mechanisms to insure the smallest of reactors, though this only applies to the period while the launch vehicle is in the hangar under NRC jurisdiction. There is a firm (American Nuclear Insurers) that insures nuclear facilities in the United States as of May 2025, in case no launch vehicle insurer proves willing to write insurance for a NTP launch vehicle.
- A good case would need to be made that this launch license serves the public interest. This report outlines benefits to the public elsewhere, including but not limited to lower-cost launches, likely identifying more as part of any specific application.
- There would need to be a plan to deal with any public pushback. This begins with engaging the community around any proposed launch site. As of May 2025, a number of spaceport proposals within and outside of the United States have foundered when some outside agency (in the U.S., most often a statewide or national entity) designated some location to be a spaceport, did not solicit feedback from (in many cases, never bothered to inform) the local community (which sometimes did not find out until the entity attempted to begin construction), and ran into “unexpected” (but predictable) local opposition. While the exact concerns and relevant sets of information may differ somewhat for a NTP launch as opposed to a chemical launch, the basic principle is identical.

In addition, the export of nuclear or space technologies is subject to controls under the Export Administration Regulations and the International Traffic in Arms Regulations. This would complicate efforts to involve non-U.S. partners[68], should any propose involvement.

[67] Another reason to start development with a smaller prototype is that the maximum probable loss (MPL), which needs to be insured per the CSLA, is lower for a smaller launch vehicle. Under the process described in the ConOps section of this report, and assuming malfunction turns or other causes of overflight of populated areas are sufficiently unlikely, the MPL seems likely to be primarily the cost of cleanup, plus the cost of replacing the launch vehicle, should phase 7a or a similar flight termination occur. As noted in the ConOps section, the cleanup area would probably be less than 100 meters across. A launch license application would be well served to include a detailed cleanup plan with validated cost estimates, as the FAA would incorporate this information directly into their MPL determination, resulting in significantly lower insurance requirements.

[68] Other than customers with payloads. Among the reasons the designs in the Spacecraft section favor the CubeSat and ESPA standards are that these standards can be freely discussed with foreign parties. Customers can design payloads to these specifications and be assured that they will fit on the launch vehicle, without needing to know any potentially restricted data about the launch vehicle.

Current and Historical Space Nuclear Regulation

The United States has launched numerous nuclear power systems into space, primarily radioisotope thermoelectric generators (RTGs) for deep space missions. However, nuclear thermal propulsion[69] has never been deployed operationally in space, despite significant development work during the NERVA (Nuclear Engine for Rocket Vehicle Application) program in the 1960s and early 1970s. Historically, space nuclear systems have been regulated through an interagency review process rather than through formal licensing:

- The Interagency Nuclear Safety Review Panel (INSRP), established in the 1960s, conducted independent safety evaluations of space nuclear systems, culminating in a Safety Evaluation Report (SER) that informed the launch approval process.
- For significant space nuclear missions, the final launch decision rested with the President or their designee, based on the INSRP's safety evaluation and other considerations.
- For missions conducted by NASA or the Department of Defense, additional agency-specific safety and review processes were applied.

While these historical processes provided a framework for government missions, they were not designed for commercial operations and did not offer the predictability and transparency needed for private investment in space nuclear systems.

The modern regulatory framework for space nuclear systems has evolved significantly, particularly with the issuance of National Security Presidential Memorandum-20 (NSPM-20) in August 2019 and Space Policy Directive-6 (SPD-6) in December 2020. NSPM-20 established a revised process for authorizing the launch of spacecraft containing space nuclear systems (Trump, 2019). Of primary relevance is its three-tier system:

- Tier I requires the nuclear system never be able to become critical (Trump, 2019)[70]. As criticality is required to achieve thrust for a NTP system, a functioning NTP system cannot be Tier I.
- Tier III requires notifying the President for each launch (Trump, 2019). For a commercial launch vehicle intended to be launched roughly 100 times per year, this is infeasible.
- Therefore, this system must fit itself into Tier II. The requirements to stay out of Tier III are the odds of exceeding 25 rem to any member of the public be kept under 1/1,000,000, and the only nuclear material on board is low-enriched uranium (Trump, 2019). The former can combine with the above FAA requirements to set a combined goal of, “no

[69] Not including RTGs and such alternatives.
[70] Tiers I and II are also distinguished based on whether the radioactive source exceeds 100,000 times the A2 value listed in Table 2 of the International Atomic Energy Agency’s Specific Safety Requirements No. SSR-6 (Rev. 1), Regulations for the Safe Transport of Radioactive Material, 2018 Edition. However, as the required potential for criticality puts this system outside of Tier I anyway, this analysis is irrelevant.

more than 1/1,000,000 chance of exceeding 25 millirem to any member of the public"[71]. For more analysis on the latter, see the Justification of HALEU section of this report.

SPD-6 further developed the national strategy for space nuclear power and propulsion, emphasizing the development of both radioisotope power systems and nuclear fission systems, including nuclear thermal propulsion (Trump, 2020).

NSPM-20 also established an Interagency Nuclear Safety Review Board (INSRB), to facilitate intergovernmental review of space nuclear activity (Trump, 2019). It is likely both the NRC and FAA would consult the INSRB in the course of reviewing any NTP-related license application. The INSRB had not previously paid attention to the possibility of a commercially owned and operated NTP launch vehicle, and has updated its playbook to begin to address this scenario as a direct result of this study.

Future of Commercial Space Nuclear Regulation

The commercial viability of nuclear thermal propulsion launch vehicles depends significantly on the efficiency and predictability of the regulatory process. While the existing framework provides a pathway to licensing, several improvements could enhance its suitability for commercial operations:

- The dual licensing requirement (NRC for the nuclear system, FAA for the launch) creates potential for delays and inconsistencies. A more integrated approach, perhaps through a joint review process, could improve efficiency. It works for developing a first-of-a-kind prototype - as noted elsewhere in this report, NRC licensing will be needed substantially before the project is ready for FAA licensing - but once the technology is more mature, joint licensing should be considered.
- Establishing clear and predictable timelines for regulatory reviews - perhaps 60 days for initial review and 30 days for final review, with a 72-hour response time for routine inquiries - would allow companies to plan effectively and reduce regulatory uncertainty.
- Currently, staff working on space nuclear regulatory matters typically have other primary responsibilities. Establishing dedicated staff positions at the NRC and FAA for space nuclear systems would build expertise and improve review efficiency.
- Expanding NSPM-20's risk-informed authorization process to explicitly include NRC licensing would help align the regulatory burden with the actual risk profile of NTP systems.

[71] This is per flight. 10 CFR § 20.1301 sets an annual public exposure limit of 100 millirem (with certain exceptions up to 500 millirem) and an hourly limit of 0.02 millirem ("Dose limits...," 2017), but that appears to only apply to operations under NRC license, and thus to operations outside of flight. To achieve this amount, operators keep the engine inside a hangar with thick concrete walls between flights, as discussed elsewhere in this report.

NSPM-20 provides a foundation for launching space nuclear systems, but its implementation could be enhanced to better support commercial NTP:

- While NSPM-20 establishes a process for government missions, its application to commercial launches requires clarification. Extending its risk-informed framework explicitly to commercial licenses would provide greater certainty.
- As a presidential memorandum, NSPM-20 could be rescinded or significantly altered by a future administration. Codifying its key provisions in legislation would provide longer-term stability.
- Currently, the jurisdiction switch between the FAA and NRC is informal. Developing a formal interagency agreement between the NRC and FAA to codify this switch, and in general regarding reviews of NTP launch vehicles, would clarify responsibilities and prevent duplicative efforts.

The current regulatory framework, while not optimal, is sufficient to support the development and launch of a minimum viable NTP prototype. The NRC has authority to license non-power reactors under 10 CFR § 50/52 ("Part 50...," 2025; "Part 52...," 2024), and the FAA has authority to license launches under 14 CFR § 450 ("Part 450...," 2024). These existing authorities can accommodate NTP launch vehicles without new legislation. There is even an existing environmental impact statement (Haslett, 1995, p. 4.2 - 4.7) which can largely be duplicated and edited, curtailing the effort needed to comply with the National Environmental Policy act.

Several improvements would enhance long-term commercial viability and investor confidence:

- Legislation specifically addressing commercial nuclear thermal propulsion would provide greater clarity and stability. This could be modeled on the ADVANCE Act's provisions for advanced reactors.
- Developing performance-based standards specifically for NTP systems would provide clear objectives while allowing flexibility in achievement.
- As other countries develop their own approaches to regulating space nuclear systems, efforts to harmonize standards and processes would facilitate international operations.
- A coordinated strategy for public engagement on space nuclear systems, involving industry, government, and non-governmental organizations, would help address potential concerns and build support.

Concluding Thoughts

The United States has an opportunity to lead the development and deployment of nuclear thermal propulsion technology for space launch. Failure to establish a clear, efficient regulatory pathway could cede this leadership to other nations, with implications for national security, economic competitiveness, and scientific advancement that could last for decades.

The substantial performance advantages of NTP create the potential for transformative capabilities in space access. These include single-stage-to-orbit vehicles, rapid transit to the Moon and Mars, and economical launch of small satellites to various orbits. This study focuses on the first and last of those, with the understanding that development of NTP launch vehicles will inevitably assist the development of NTP engines optimized for lunar and martian missions.

A key challenge in the current regulatory framework is its dependence on executive actions, which can change with administrations. NSPM-20 and SPD-6, while providing important direction, are subject to modification or rescission by future presidents. This regulatory uncertainty can deter investment and complicate long-term planning. Codifying the key elements of space nuclear regulation in legislation would provide greater stability and confidence. This legislation could:

- Establish clear lines of authority between the NRC and FAA for NTP launch vehicles,
- Set maximum timelines for regulatory reviews,
- Direct the development of NTP-specific safety standards,
- Authorize dedicated funding for regulatory staff and expertise, and
- Create a streamlined process for prototype and demonstration missions.

The development of nuclear thermal propulsion launch vehicles requires significant investment over extended periods. Investors and companies need confidence the regulatory pathway will remain stable and predictable. By establishing a clear, efficient, and durable regulatory framework, the United States can create the conditions for successful commercialization of this transformative technology.

The potential market for NTP launch vehicles is essentially the entire launch market, by providing both lower-cost and more responsive launch vehicles. This better service can also enable new uses for space, expanding the launch market. See the Economic Analysis appendix for more details. However, realizing this potential requires a regulatory environment, which balances safety and innovation.

In conclusion, nuclear thermal propulsion launch vehicles represent a significant opportunity for the United States to advance its space capabilities and maintain leadership in an evolving global space economy. With targeted improvements to enhance clarity, efficiency, and stability, the regulatory framework can support the successful development and deployment of NTP technology for space launch. Until then, the current regulatory framework provides a viable pathway to licensing these vehicles.

MITEE-HALEU

Background

The MIniature ReacTor EnginE, also known as MITEE, builds on the legacy of reactors developed for nuclear thermal propulsion (NTP) systems. Such systems leverage nuclear process/ fission-generated thermal energy for high-thrust propulsion. The diagram below shows that such an application is just one of the space-based nuclear energy use cases.

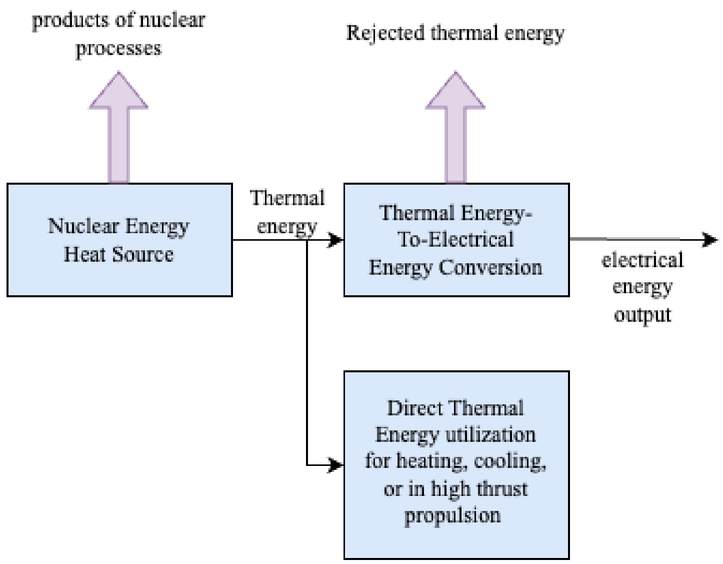


Figure 2. Generic Space Nuclear Power Systems (Buden, 1993)

The development of nuclear high-thrust propulsion systems in the United States began with Project Rover in 1955 and, with its success, followed Project NERVA before the program was canceled in 1973 due to a lack of a defined mission (Powell et al., 1999a). In 1986, with the onset of the Strategic Defense Initiative, more commonly known as President Ronald Reagan's Star Wars program, there was a renewed interest in nuclear thermal propulsion systems with a conceptual study done on a particle bed reactor. The particle bed reactor achieved a smaller size and similar power (~1000 MW) relative to the ROVER/NERVA program reactors. This was achieved using a 7LiH moderator compared to the graphite moderators that the ROVER/NERVA program reactors used, which was a relatively poor moderator that required the reactor to be at least ~1m in diameter for criticality to be achieved.

The MITEE concept, conceived around 1998 (Powell et al., 1998), aimed to continue developing the prerogative of increasingly lightweight and compact nuclear thermal propulsion system designs. In addition to utilizing the 7LiH moderator, cermet W/UO2 sheets developed during the GE 710 NTP program (Powell et al., 2003) were incorporated to allow for an annular fuel configuration. This design approach achieves a lighter mass and retains similar heat transfer properties compared to the PBR (Powell et al., 2003).

The components enabling the production of thrust, which classify MITEE (Micro Nuclear Thermal Propulsion Engine) as a NTP system, illustrated in Figure 3. The 710 cermet fuel is fabricated as metal matrix sheets with recessed zones designed to facilitate gas flow when rolled into a multi-layered annular configuration, forming the fuel region. This annular fuel is housed within a beryllium pressure tube. Hydrogen, used as the propellant, is introduced at 20 K through the cold frit-free volume between the pressure tube and the annular fuel. As the hydrogen flows through the recessed fuel regions, it is heated to approximately 2700-3000 K within the central hot gas exit channel. The heated hydrogen expands and exits the pressure tube, generating thrust. A lattice arrangement of these pressure tubes and fuel elements constitutes the reactor core, collectively producing a nominal thrust of approximately 14,000 N (Powell et al., 2006).

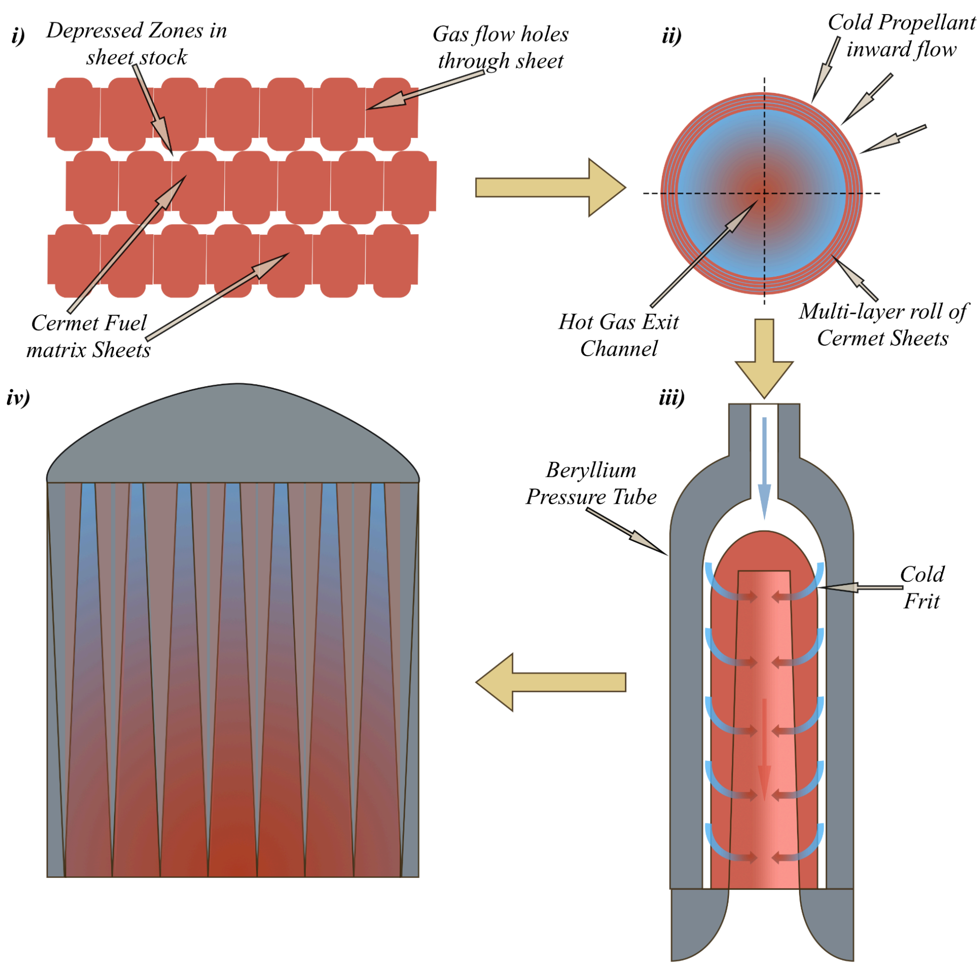


Figure 3. Constitutive elements of the MITEE reactor/NTP engine (Powell et al., 2006)

General MITEE-type Reactor Geometry and Materials

The MITEE reactor features a central-axial flow channel (inner nozzle) and two concentric cermet-type fuel rings composed of WUO_2 and $MoUO_2$. These fuel rings contain small-diameter perforations, constituting approximately 25% of the volume, facilitating hydrogen propellant flow from the cold to the hot frit. This inward flow mechanism directs the cold propellant through the perforations, allowing it to be heated by the fuel before exiting through the nozzle. To prevent direct contact between the hot hydrogen propellant and the UO_2 fuel, the cermet fuel is vapor-coated with a tungsten layer of a few mils (where 1 mil is 0.001 cm). This study selected a liner/cladding thickness of approximately 0.01 cm[72].

Two reactor variants were developed: MITEE and MITEE-B. MITEE serves as the baseline NTP system. In contrast, MITEE-B incorporates a closed-cycle helium loop for bimodal operation, enabling thrust generation and auxiliary power production via Brayton, Stirling, or steam cycles. The key distinction between MITEE and MITEE-B is the latter's inclusion of eight additional beryllium tubes bonded to the cold frit, which facilitate thermal energy transfer through a closed-cycle helium coolant to an external energy conversion system.

The moderator utilized in MITEE is lithium-7 hydride, configured within a honeycomb structure. The pressure tube and honeycomb framework that houses the fuel assembly are constructed from beryllium.

The OpenMC plotted horizontal cross-sections of the MITEE-B and MITEE pin-cell and pressure tube, highlighting the specified design features, are shown below:

[72] Exceeding the thickness needed to prevent uranium diffusion during a launch: see the footnote in the Introduction section that uses Schwelger's equation.

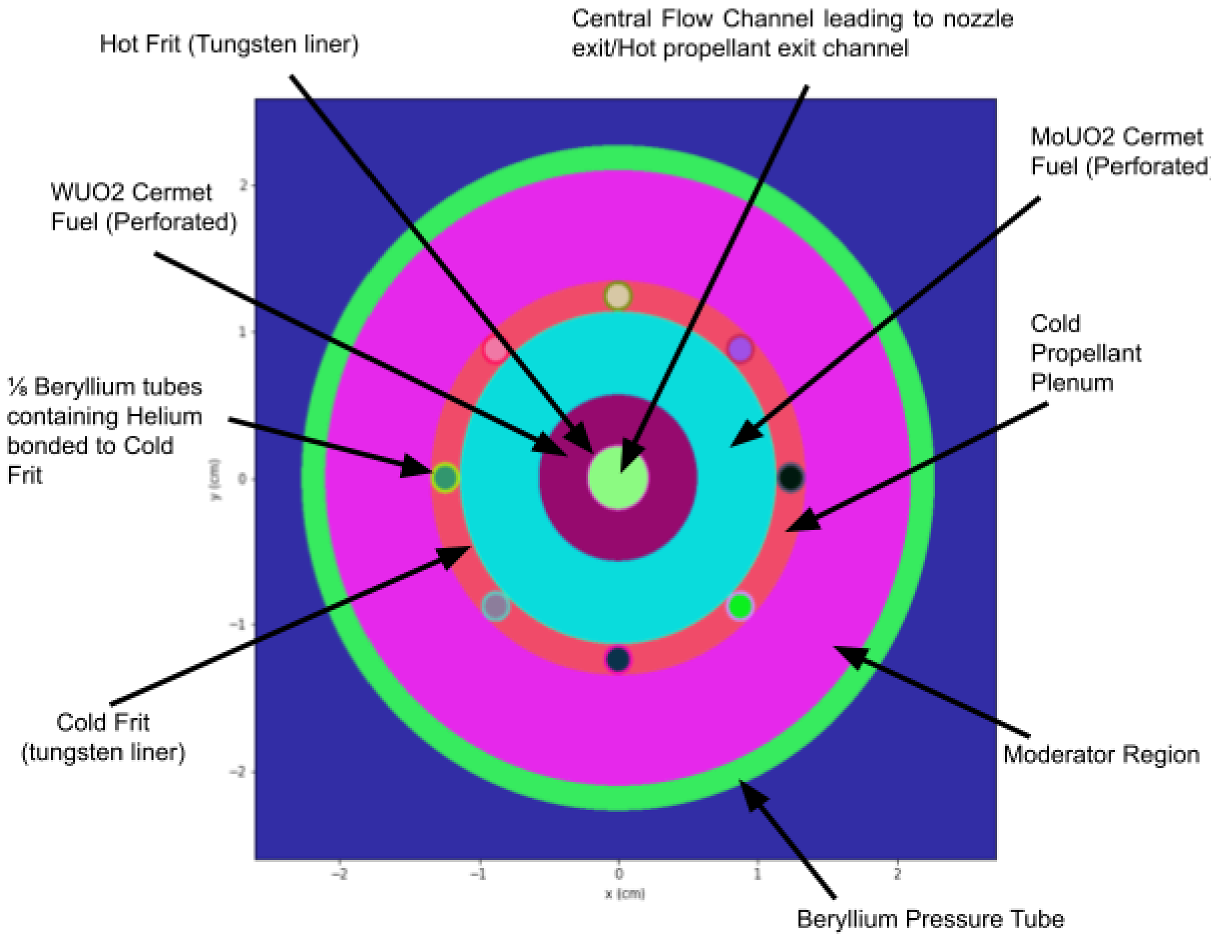


Figure 4. Horizontal cross section of MITEE-B

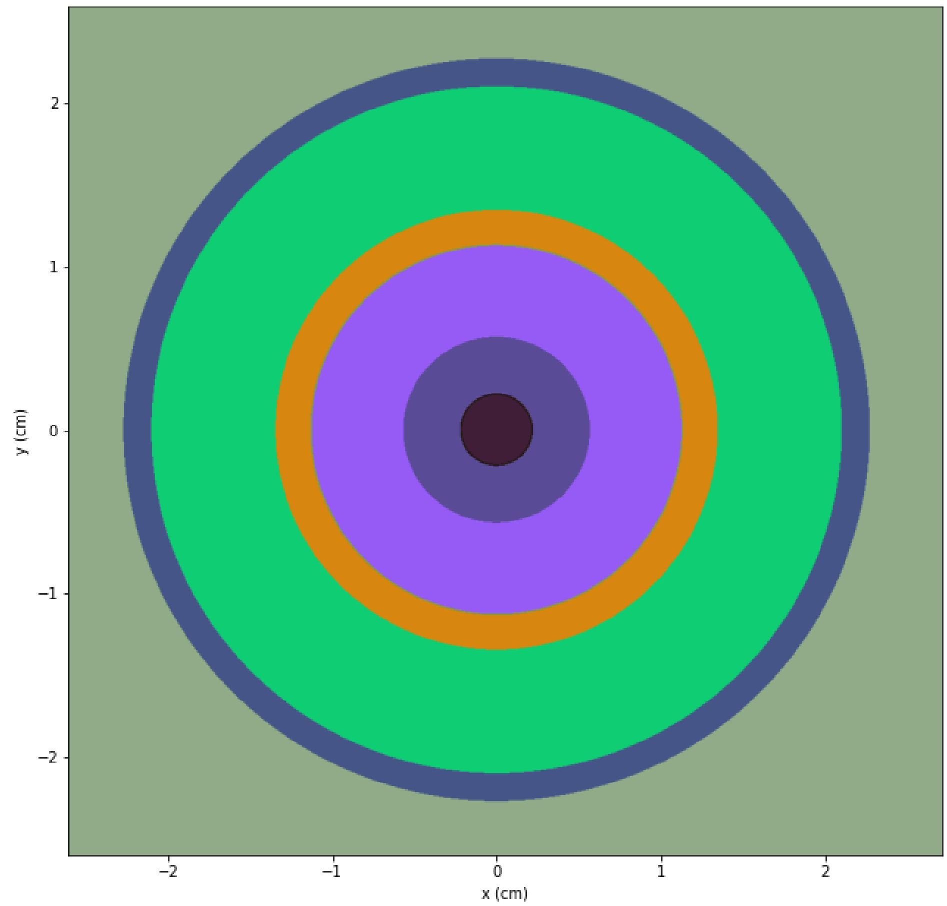


Figure 5. Horizontal cross section of baseline MITEE (without helium tubes)

The MITEE reactor core design features two loading patterns, distinguished by the number of pressure tubes, pin cells, and fuel elements arranged within the hexagonal lattice. In this study, a 37-fuel-element configuration was selected over the alternative 61-fuel-element design. The core is surrounded by radial reflector elements, which consist of beryllium pressure tubes filled with lithium hydride. For the 37-fuel-element configuration, the total number of pressure tubes, including the reflector elements, is 61. The horizontal and vertical cross-sections of the MITEE-B core design are shown below.

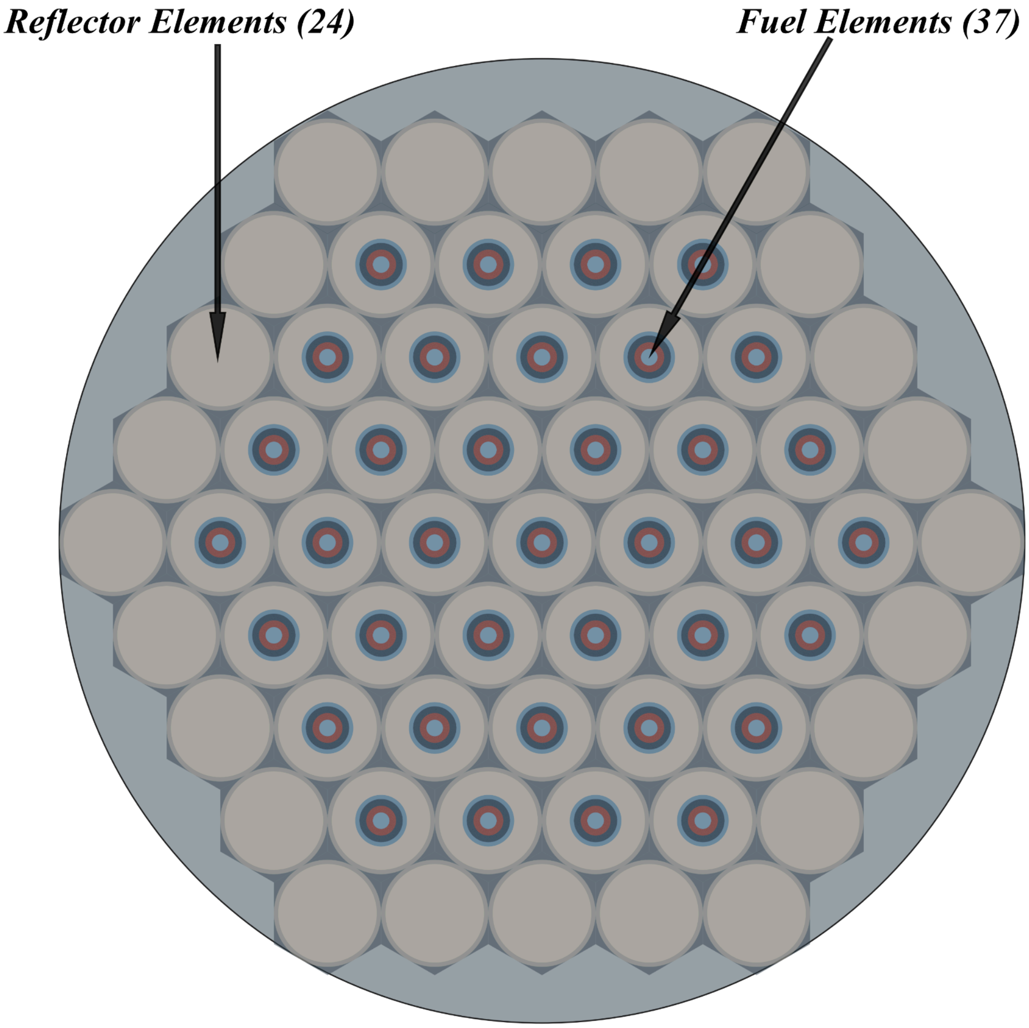


Figure 6. MITEE-B reactor horizontal cross-section

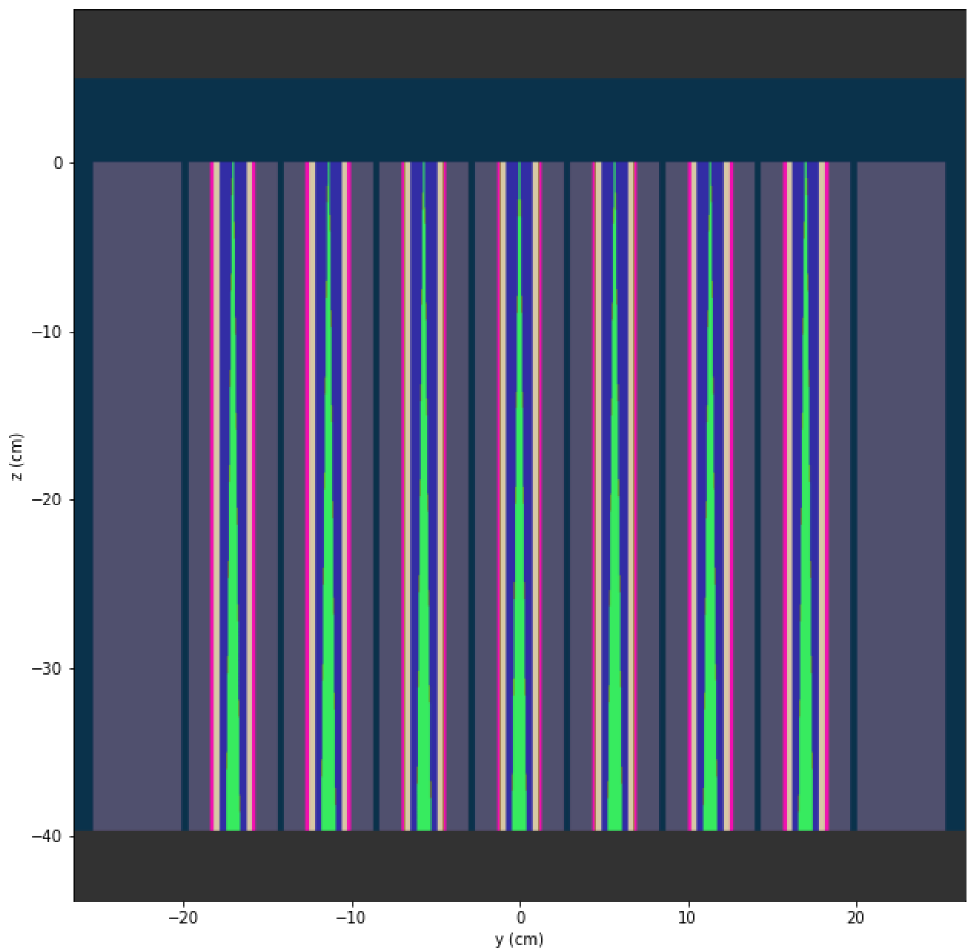


Figure 7. MITEE-B reactor vertical cross-section (note that there is a 5 cm upper axial reflector located on top of the core made out of beryllium)

MITEE Geometry Study

To calibrate and obtain confidence in MITEE variant OpenMC results, the literature-based MITEE models with 37 fuel elements were studied across P/D ratios of 1.5, 2.0, 2.5, and 3.0. The basis for these studies was normalized across a fuel outer diameter (containing the nozzle, WUO2, and MoUO2 sections) of 2.27 cm.

In this current study, the densities of the fuel were obtained from the literature (Powell et al., 2002) instead of using the sum function, providing WUO2 with a density of 8.85776 g/cc and MoUO2 with a density of 6.1468 g/cc. The results provided in Table 1 were accomplished with the Evaluated Nuclear Data File (ENDF/B-VIII.o) cross section library integration in OpenMC. Figures 8-10 visualize the library derived specific reaction cross section variation against incident neutron energies with respect to the materials of interest (U-235, U-238, beryllium, tungsten-184, hydrogen, lithium-7, and oxygen). Last but not least, the OpenMC simulation settings were also defined to 100 simulation batches, 10 inactive simulation batches, and 10000 particles per generation.

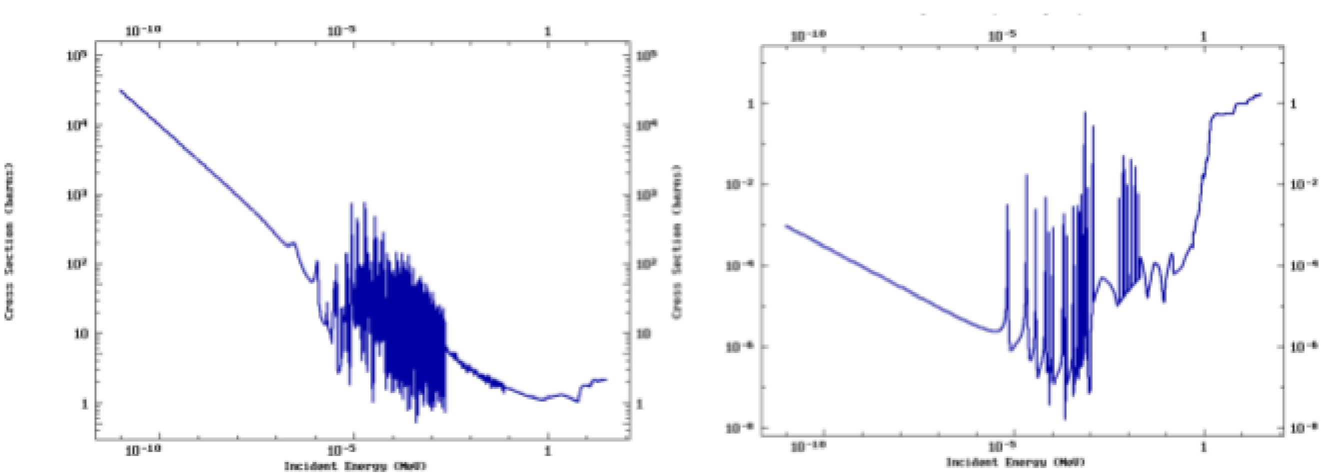


Figure 8. Fission cross sections $\sigma(n, fission)$ against incident neutron energy for U-235 (left) and U-238 (right) [Taken from ENDF/B-VIII.o]

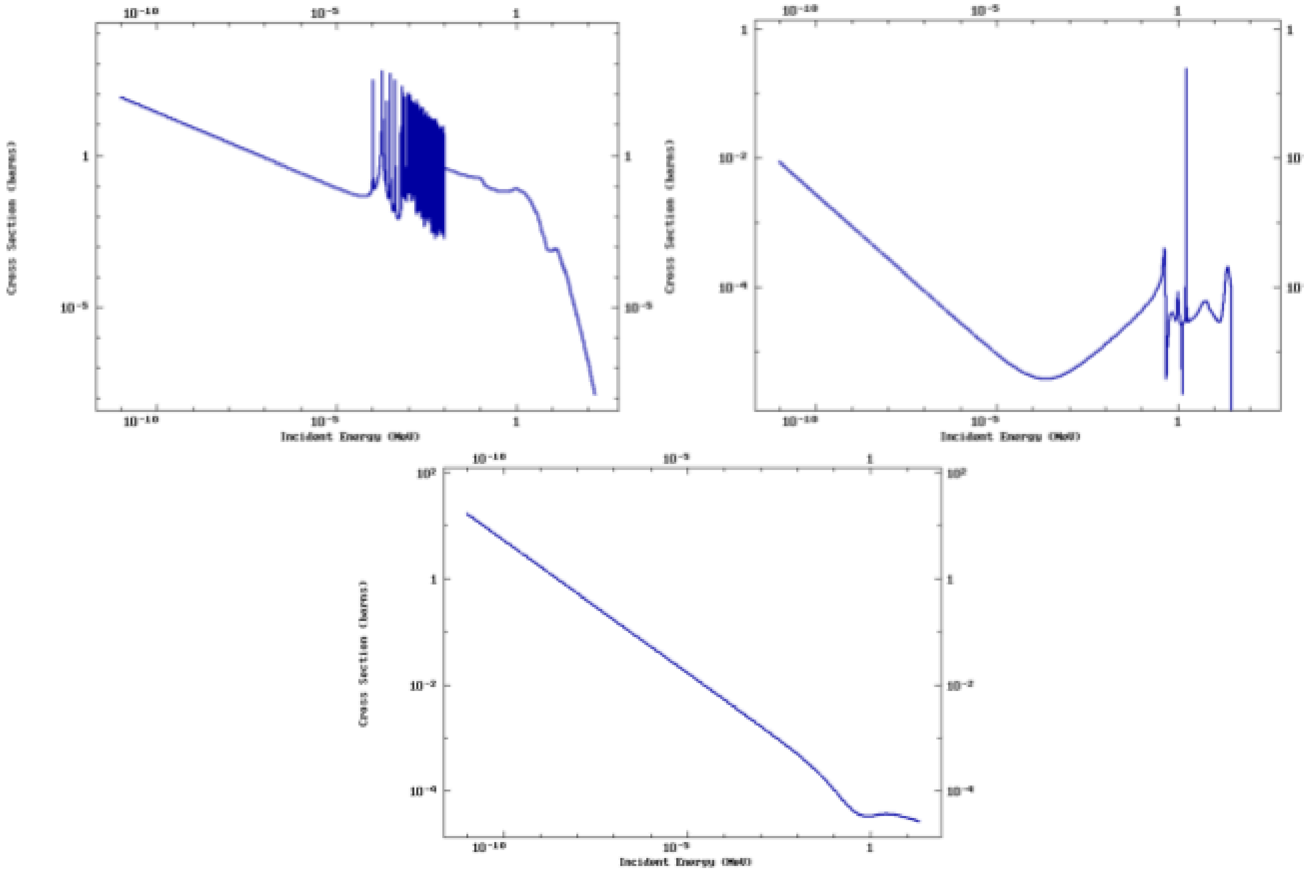


Figure 9. Capture cross sections $\sigma(n, capture)$ against incident neutron energy for tungsten-184 (top left), oxygen-16 (top right), and hydrogen (bottom) [Taken from ENDF/B-VIII.o]

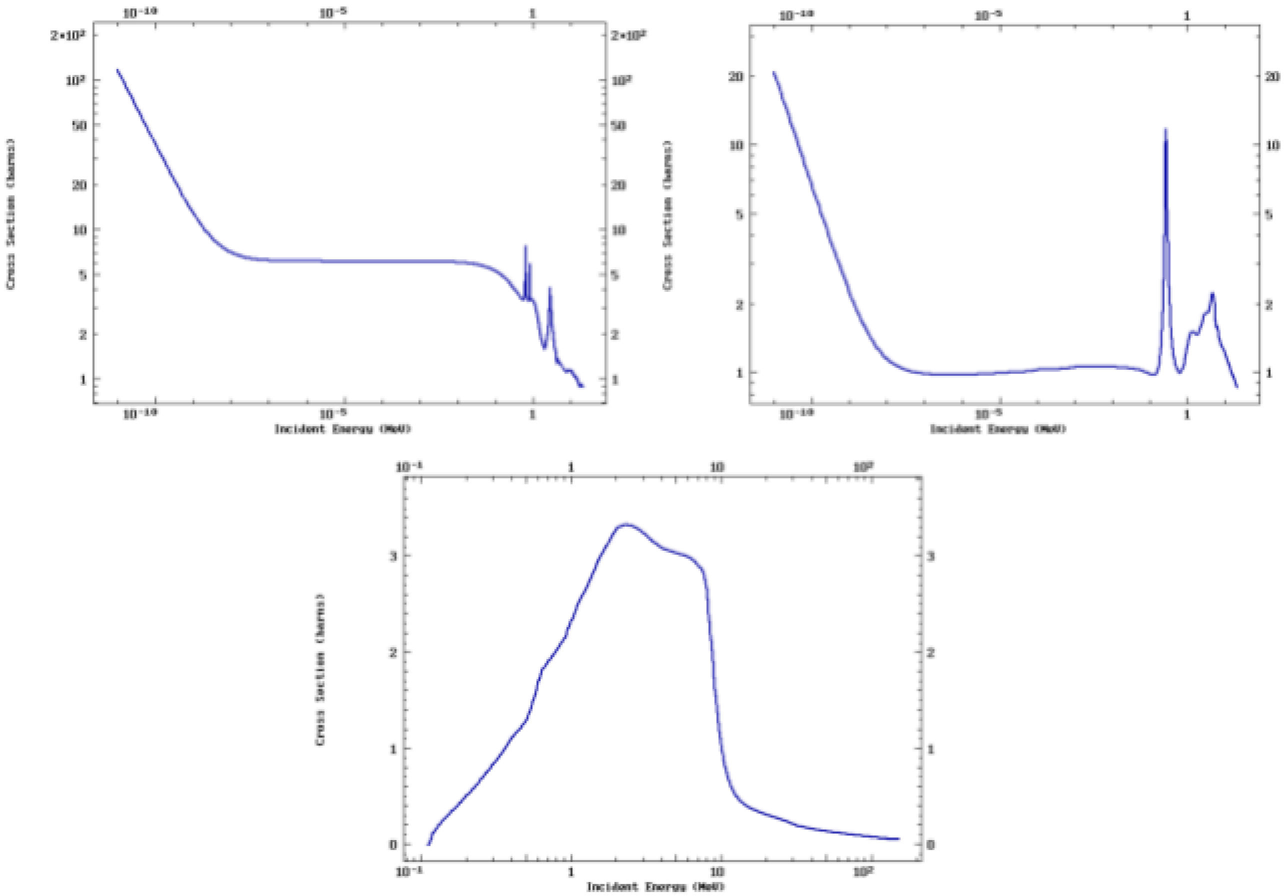


Figure 10. Beryllium-9 (top left) and lithium-7 (top right) elastic scattering cross sections against incident neutron energy. Tungsten-184 inelastic scattering cross section against incident neutron energy (bottom) [Taken from ENDF/B-VIII.o]

Table 1. OpenMC results of baseline MITEE model compared to literature assuming a core diameter to height ratio of 1

P/D Ratio	Pitch [cm]	Nuclear Fuel Diameter [cm]	Core Outer Diameter/ Height [cm]	Reactor Outer Diameter [cm]	Keff (HEU, OpenMC)	Keff (HALEU, OpenMC)	Keff (HEU, literature[73])
1.5	3.405	2.27	23.835	30.645	0.88951 +/- 0.00143	0.65000 +/- 0.00119	0.89
2.0	4.54	2.27	31.78	40.86	1.28907 +/- 0.00111	0.99943 +/- 0.00117	1.07

[73] Note that values obtained from literature have slight inconsistencies (refer to Study Limitations), mainly concerning published values.

2.5	5.675	2.27	39.725	51.075	1.28629 +/- 0.00112	0.97388 +/- 0.00123	1.12
3.0	6.81	2.27	47.67	61.29	1.15988 +/- 0.00162	0.85554 +/- 0.00103	1.075

Looking at Table 1, the OpenMC Keff value for P/D = 1.5 agrees with the literature. However, this correlation diverges as P/D increases, which might be related to some literature-based discrepancies and assumptions noted in the Study Limitations section of this report.

MITEE-B Geometry Study

The MITEE-B's published dimensions are significantly different from the baseline MITEE, most notably in the radius of the outermost fuel element (MoUO2): 1.0069 cm for MITEE-B compared to 1.135 cm for the baseline.

Table 2. Literature based MITEE-B dimensions for P/D of 2.5 and L/D of 1.0 (Powell et al., 2002)

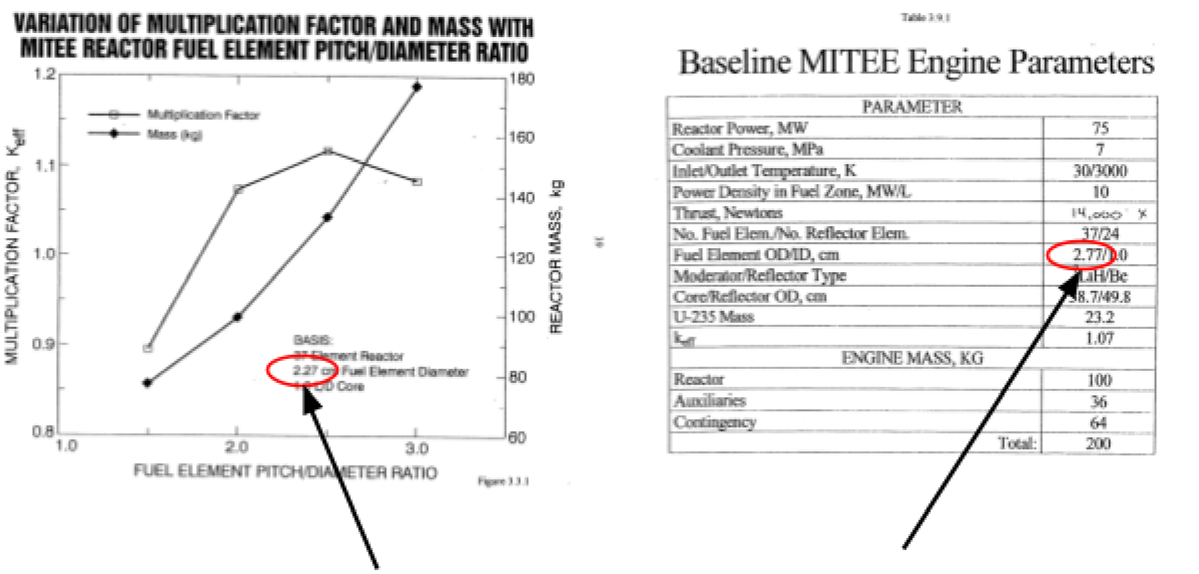
Geometric regions	**Dimensions [cm]**
Nozzle Radius	0.3873
WUO2 Inner Radius	0.6971
MoUO2 Inner Radius/Diameter	1.0069/2.0138
Cold Propellant Region	1.345
Lithium Hydride Moderator	2.3
Pressure Tube Region	2.5
Pitch	5.0345
Liner Thickness	0.01
Pitch to Element Diameter Ratio	2.5
Core Height-to-Diameter Ratio	1.0
Core Outer Diameter and Height	35.2415
Reactor Outer Diameter	45.3105
Beryllium Tube (Helium Containing) OD	0.195
Beryllium Tube (Helium Containing) ID	0.117
Upper Axial Beryllium Reflector Thickness	5

According to the literature, the optimal P/D for a 61 fuel element MITEE-B is 2.5 (HEU), and the Keff found for this P/D is 1.09. A 37-fuel assembly configuration was not studied in the literature. According to OpenMC simulations for the 37 fuel assembly design:

- The MITEE-B (37 fuel element) Keff for HALEU (20% U-235 and 80% U-238) is 1.02494 +/- 0.00111, for a 41436.7 particles/second reaction rate.
- The MITEE-B (37 fuel element) Keff for HEU (90% U-235 and 10% U-238) is 1.33453 +/- 0.00124, for a 35499.7 particles/second reaction rate.

Study Limitations

Several discrepancies were found within the literature on MITEE. One of these lies within the published values used within the geometry cards for MCNP calculations. An example of such discrepancies that might have affected the calculated results found in the baseline MITEE example in Table 1 is shown in Figure 11. In this report, the 2.27 cm value was used for fuel element outer diameters.



PARAMETER	
Reactor Power, MW	75
Coolant Pressure, MPa	7
Inlet/Outlet Temperature, K	30/3000
Power Density in Fuel Zone, MW/L	10
Thrust, Newtons	14,000
No. Fuel Elem./No. Reflector Elem.	37/24
Fuel Element OD/ID, cm	2.77/[illegible]
Moderator/Reflector Type	LiH/Be
Core/Reflector OD, cm	[illegible]8.7/49.8
U-235 Mass	23.2
k_{eff}	1.07
ENGINE MASS, KG	
Reactor	100
Auxiliaries	36
Contingency	64
Total:	200

Figure 11. Differences in published MITEE values (Powell et al., 1998; Powell et al., 1999a)

The current OpenMC study assumes that all materials are at standard room temperature of 293.6 Kelvin (20 degrees Celsius). The original MITEE study, however, models hydrogen entering the cold frit at 20 K and exiting the central plenum at 3000 K, with the radial temperature distribution, along with the fuel-region gradient, shown in Figure 9. The ENDF/B-VIII.0 library used in OpenMC only supplies cross-section data at 250, 293.6, 600, 900, 1200, and 2500 K, limiting the current analyses from achieving either stepwise or smooth temperature gradients utilized in the original MITEE papers. This study is, as of May 2025, exploring the incorporation of hydrogen and cermet fuel cross-sections at such temperature gradients, which brings up the discussion of future work.

Future Work

Future work should supplant the isothermal assumption with temperature profiles that exemplify both the fuel and hydrogen temperature gradients found in Figure 12. This also goes for the MITEE-C section where neutronics in NTP conditions remain. In this case, emergent physical and metallurgical properties of the fabricated cermet fuel at these temperatures will need to be investigated to validate the original MITEE design's, mainly of the fuel region, consideration towards structural integrity, thus informing the ongoing MITEE-C design efforts.

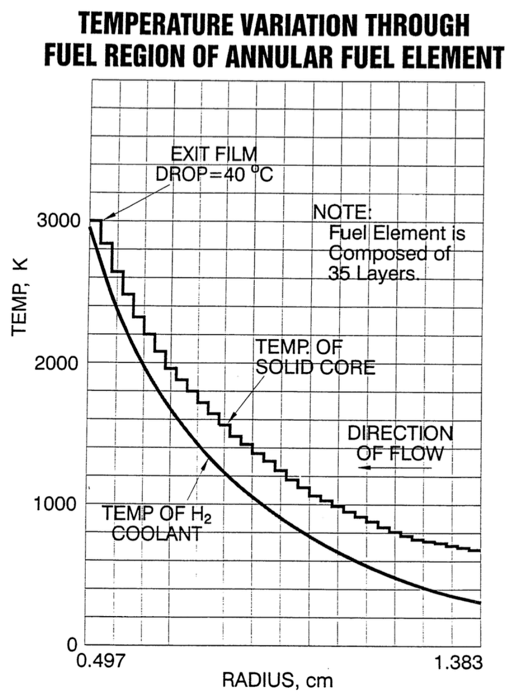


Figure 12. Temperature profile of fuel region and hydrogen propellant through WUO2 and MoUO2 regions (Powell et al., 1999a)

Table 3 provides an initial glance on cermet fuel integrity, with regards to the materials at an upper-range temperature of interest found in this report. Table 4 is a reformatted list of stable and unstable mass loss categorizations citing high-temperature hydrogen exposure from historical WUO2 cermet fuel studies, compiled from data mostly accomplished in the 1960s (Stewart, n.d.).

Table 3. Melting Points and Surface Vaporization rates of Cermet Fuel Materials of interest in Vacuum (Stewart, n.d., Table 1)

Material	Melting Point (K)	Surface Vaporization Rate at 2800 K (mil/hr)
Tungsten, W	3680	<0.01
Molybdenum, Mo	2890	>>10
Uranium Dioxide, UO2	3075	6×10^3

Table 4. Stability and Mass Loss of Tested UO2 and WUO2 Cermet samples from historical testing (Stewart, n.d., Table 2)

Feature/Condition	1	2	3	4	5	6	7	8	9	10	11	12	13	14
UO2 Only	✓						✓							
W-UO2		✓	✓	✓	✓	✓		✓	✓	✓	✓	✓	✓	✓
Partial Clad (Not Edges)								✓			✓			
Full Clad									✓	✓		✓	✓	✓
Coated Fuel Particles			✓									✓		
Stabilizers (Various)						✓	✓	✓	✓		✓	✓	✓	✓
Temperatures														
2000 °C / 2273 K														
2300 °C / 2573 K														
2350 °C / 2623 K														
2500 °C / 2773 K														
2600 °C / 2873 K														
2650 °C / 2923 K														
2700 °C / 2973 K														
2800 °C / 3073 K														
Cycles Tested								25	>25				<30	<10
Fuel Samples Tested		29 + 14	46	19	2		25+	~30	~20	6	2	1	2	2

Table 4 Legend

	Unstable: Cracks or Forms Powder		Stable: Mass Loss > 5%		Stable: Mass Loss < 5%

The validity of future OpenMC and Monte Carlo simulation results also relies upon the continued study of the stoichiometry of the fuel mix to be used. The studies on cermet fuel, most notably the GE 710 and ANL 200/2000 programs (Stewart & Schnitzler, 2015), mention binders and stabilizers crucial to the integrity of the cermet fuel. This should be further studied and, in terms of stoichiometric impact, should be incorporated into the OpenMC model. The atomic fractions (HALEU) for the WUO2 and MoUO2 regions are shown in the tables below (Tables 5 and 6) and represent the most rudimentary representations of the MITEE fuel mix, including hydrogen as the propellant flows through the perforated areas.

Table 5. HALEU Atomic Fractions (MoUO2)

Elements/nuclides	**Atomic fraction**
U-235	0.0135207277
U-238	0.05408291106
Oxygen	0.1352093058
Molybdenum	0.796303205
Hydrogen	0.0008838504038

Table 6. HALEU Atomic Fractions (WUO2)

Elements/nuclides	**Atomic fraction**
U-235	0.02963528677
U-238	0.1185411471
Oxygen	0.296357313
Tungsten	0.5547051059
Hydrogen	0.0007611472124

The values above assume a 20% U-235 enrichment value and a weight fraction of 0.54 to 0.46 UO2/tungsten fuel mix and 0.417 to 0.583 UO2/molybdenum fuel mix, as determined by literature (Powell et al., 1999a). Regarding the hydrogen propellant amount, the hole volume fraction in the fuel was also taken from the literature as 0.25 (Powell et al., 1997).

In addition to examining temperature dependencies and fuel stoichiometry in future OpenMC studies, optimizing the core design warrants further investigation. More information on this is in the Conclusion section.

MITEE-C

MITEE-C is a derivation of MITEE and MITEE-B incorporating a similar fuel assembly design with a HALEU configuration and an additional ring of control drums lining the periphery of the core. The fuel mix was also simplified to incorporate only the tungsten UO2 cermet instead of both tungsten and molybdenum UO2 cermets. This decision was made to fully leverage the WUO2's superior thermal properties over the molybdenum UO2 cermet, trading off with a larger parasitic neutron absorption cross section as shown in the table below (Gaffin et al., 2025):

Table 7. Summary of NTP fuel matrix material class melting point and cross sections (Gaffin et al., 2025)

Material Class	Fuel Matrix	Melting Point (K)	$\sigma_S(b)$	$\sigma_A(b)$
Graphite	Graphite	4100	5.55	0.0035
Cermet	Molybdenum	2896	5.67	2.48
Cermet	Tungsten	3695	2.97	18.3
Carbide	ZrC	3910	11.99	0.19
Carbide	NbC	3760	11.80	1.15
Carbide	HfC	4100	13.15	104.0
Carbide	TaC	4250	11.55	20.6

$\sigma_S(b)$ is the bound scattering coherent cross section, and $\sigma_A(b)$ is the thermal neutron absorption cross section as compiled by Sears (1992). The cross-section values in the table are detailed concerning compounds or elements in their natural abundance. Thus, to offset the negative side effects of natural tungsten's larger parasitic neutron absorption cross section, tungsten (W) enriched to the stable W-184 isotope with a significantly smaller thermal neutron absorption cross section as shown in Table 8 of 1.7 barns is chosen. As a point of validation, this is consistent with the chosen tungsten isotope utilized in the MITEE and MITEE-B designs.

Table 8. Summary of relevant tungsten (W) cross sections (Sears, 1992)

Z (Atomic Number)	**A (Mass Number)**	$\sigma_S(b)$	$\sigma_A(b)$
W-74	Natural	2.97	18.3
	180	3.0	30.0
	182	6.10	20.7
	183	5.36	10.1
	184	7.03	1.7
	186	0.065	37.9

The fuel assembly design pitch and fuel diameters were optimized toward the MITEE-C design of only utilizing WUO2 cermet fuel sheets, starting with the pin-cell design below.

The revised pin cell design optimizes the repetitive lattice structure to enhance fission, neutron moderation, and subsequent thermalization (i.e., slowing neutron energies from MeV to 0.025 eV) using a single cermet fuel type: WUO2 with HALEU. The inter-assembly/ element pitch, correlated with pressure tube radii, is set at 4.54 cm to achieve a height-to-diameter ratio (L/D) of 1.75. For comparison, the baseline MITEE design features pitches ranging from 3.405 cm to 6.81 cm (L/D = 1), while MITEE-B employs a 5.0345 cm pitch (L/D = 1.0). A comparison of fuel assembly geometries across MITEE, MITEE-B, and the proposed MITEE-C is presented in Table 9 assuming a 61 fuel element configuration, with detailed geometric parameters of the new pin cell provided in Table 10. The nozzle (hot channel) dimensions are retained to preserve the divergence aspect ratio.

Table 9. Comparison of MITEE, MITEE-B (HEU - WUO2 & MoUO2), and MITEE-C (HALEU - WUO2) fuel element design parameters

Fuel Element Description	**MITEE**	**MITEE-B**	**MITEE-C**
Core Height to Diameter Ratio (L/D)	1.0	1.0	1.75
Pitch [cm]	3.405 - 6.81	5.0345	4.54
Fuel Element Diameter [cm]	2.27	2.00	2.27
Height [cm]	30.645 - 61.29	45.3105	71.505

Table 10. New Fuel Element Geometric Design Parameters

Fuel Element Description	**Design Parameter**
Fuel Element Radius (only WUO2)	1.235 cm
Fuel Element Cold Frit Radius	Fuel Element Radius + 0.01 cm
Cold Region	Fuel Element Cold Frit + 0.20 cm
Moderator Region Radius	2.07 cm
Pressure Tube Radius	2.27 cm
Height	71.505 cm (L/D = 1.75)

The pin cell design also exhibits an additional beryllium hexagonal structure within the moderator region. This serves as a structural component of the moderator region where it houses the yttrium hydride moderator, which is nominally sintered, produced, and handled in powdered form. The optimal hexagonal lattice was found regarding the neutronic design/Keff obtained. This varied the hexagonal lattice's pitch proportional to its edge lengths. A rudimentary optimization was conducted in OpenMC, with results listed in Table 11 below. A row was added to showcase the Keff for a moderator region with no hexagonal lattice structure as the threshold for an improved Keff.

Table 11. Simple Optimization on Hexagonal Lattice Structure within Moderator Region

Keff (OpenMC)	**Pitch (not fuel element pitch)**	**Edge Length (note: inner edge length is -0.01 cm)**
0.2974	1.00 cm	0.50 cm
0.2982	1.01 cm	0.505
0.2980	1.02 cm	0.51 cm
0.2982	1.03 cm	0.515 cm
0.2975	1.04 cm	0.52 cm
0.2983	1.05 cm	0.525
0.2971	1.06 cm	0.53 cm

Keff (OpenMC)	**Pitch (not fuel element pitch)**	**Edge Length (note: inner edge length is -0.01 cm)**
0.2967	1.07 cm	0.535 cm
0.2971	1.08 cm	0.54 cm
0.29705	1.09 cm	0.545 cm
0.2977	1.10 cm	0.55 cm
0.2973	1.11 cm	0.555 cm
0.2978	1.12 cm	0.56 cm
0.29899	1.13 cm	0.565
0.2970	1.14 cm	0.57 cm
0.29776	1.15 cm	0.575 cm
0.2971	1.16 cm	0.58 cm
0.2974	1.17 cm	0.585 cm
0.2968	1.18 cm	0.59 cm
0.2971	1.19 cm	0.595 cm
0.2956	1.20 cm	0.6 cm
0.2973	1.21 cm	0.605 cm
0.2974	1.22 cm	0.61 cm
0.29878	1.23 cm	0.615
0.2956	1.24 cm	0.62 cm
0.2971	1.25 cm	0.625 cm
0.2963	1.26 cm	0.63 cm
0.2967	1.27 cm	0.635 cm
0.2967	1.28 cm	0.64 cm
0.2966	1.29 cm	0.645 cm
0.2976	1.30 cm	0.65 cm

Keff (OpenMC)	**Pitch (not fuel element pitch)**	**Edge Length (note: inner edge length is -0.01 cm)**
0.29907	1.31 cm	0.655 cm
0.2979	1.32 cm	0.66 cm
0.2963	1.33 cm	0.665 cm
0.2987	1.34 cm	0.67 cm
0.2973	1.35 cm	0.675 cm
0.2977	1.36 cm	0.68 cm
0.29697	1.37 cm	0.685 cm
0.2962	1.38 cm	0.69 cm
0.2976	1.39 cm	0.695 cm
0.2970	1.40 cm	0.70 cm
0.2977	1.41 cm	0.705 cm
0.2972	1.42 cm	0.71 cm
0.29827	1.43 cm	0.715 cm
0.2979	1.44 cm	0.72 cm
0.2964	1.45 cm	0.725 cm
0.2970	1.46 cm	0.73 cm
0.2978	1.47 cm	0.735 cm
0.2970	1.48 cm	0.74 cm
0.29908	1.49 cm	0.745 cm
0.29797	1.50 cm	0.75 cm
0.2985	**No hexagonal/ Beryllium lattice**	**No hexagonal/ Beryllium lattice**

The variations in Keff for the fuel assemblies with a hexagonal lattice pitch are plotted in Figure 13. As shown in the figure, five Keff peaks exceed the baseline value of 0.2985 observed for the fuel assembly without a hexagonal lattice in the moderator region.

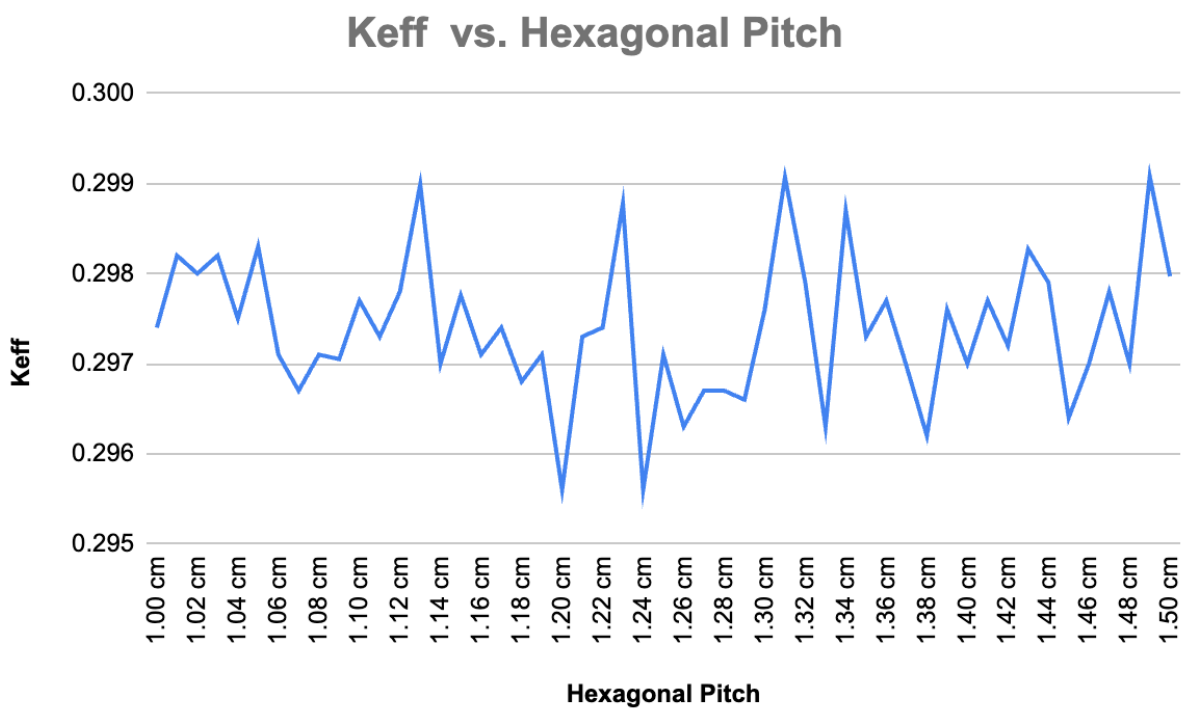


Figure 13. Keff of fuel assemblies (with hexagonal lattice) vs. hexagonal pitch

The highest Keff value within the pitch range of 1.0 to 1.5 cm was observed at a pitch of 1.49 cm, yielding Keff = 0.29908. Consequently, this fuel assembly design was selected for inclusion in the broader core design optimization study. The horizontal cross-section of this fuel assembly design is shown in Figure 14 below.

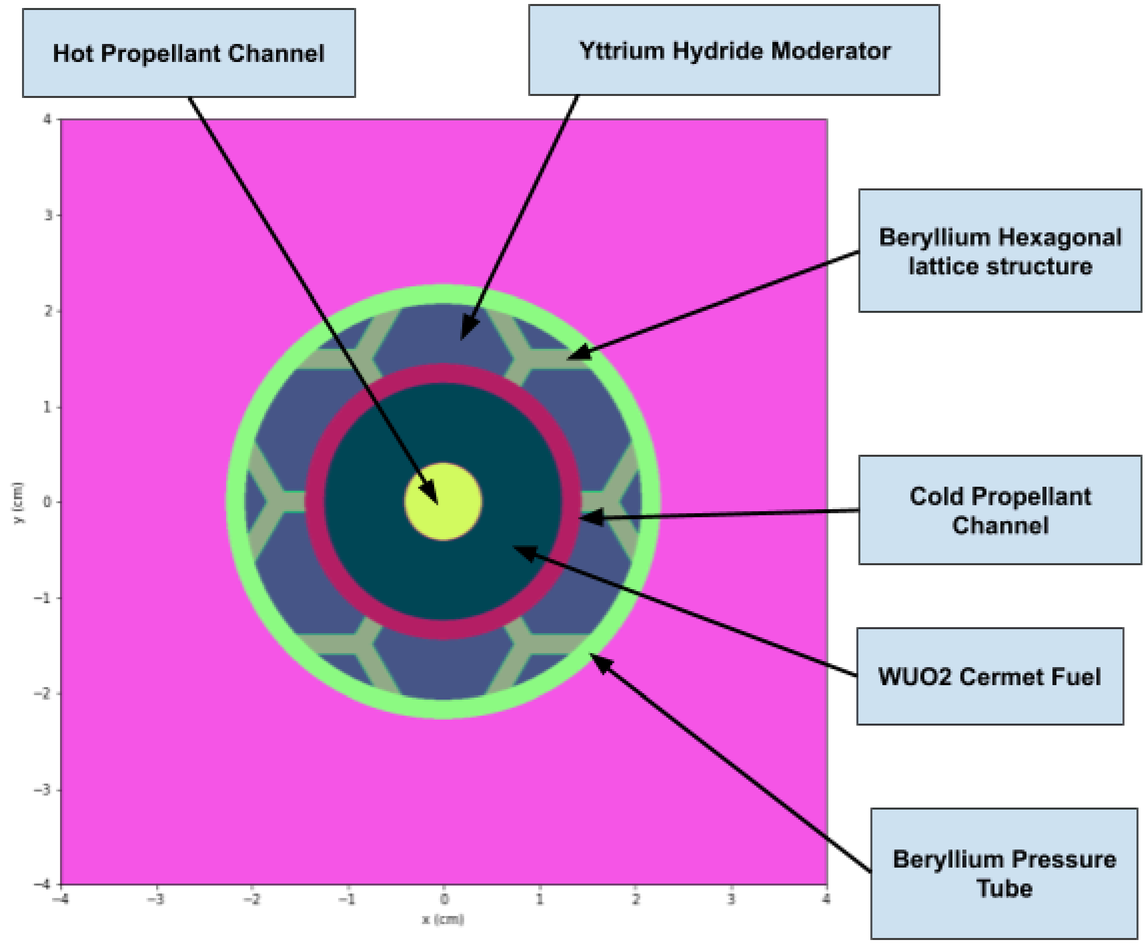


Figure 14. Horizontal cross section of fuel assembly (for hex pitch = 1.49 cm)

Explicit hydrogen coolant channels, each with a radius of 0.065 cm, were modeled axially through the individual upper axial beryllium reflector to allow flow from an external turbopump into the beryllium hexagonal lattice structures. Each channel is connected to three differential height radial flow channels, which ultimately direct coolant or propellant toward the cold propellant channel. Figure 15 below visualizes the cross-sectional locations of each of the six axial coolant channels in a single pin-cell located within the core geometry.

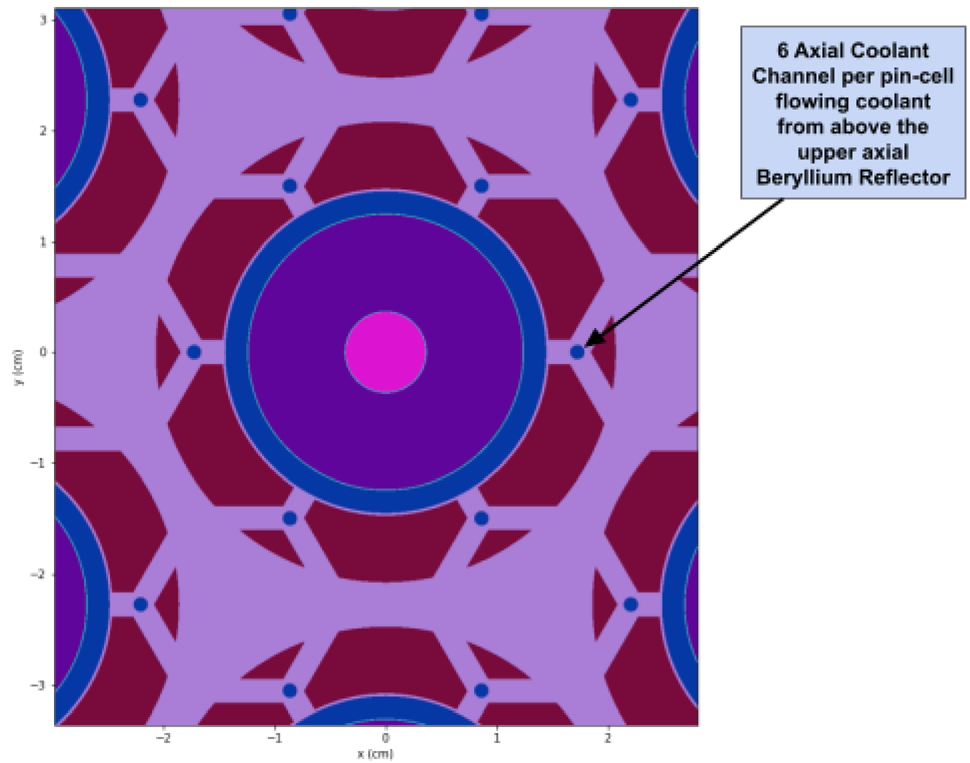


Figure 15. Horizontal cross section of pin cell/fuel assembly within core visualizing positions of axial coolant channels

The core diameter of MITEE-C was set to 65 cm to enable the addition of 7 cm diameter control drums. Control of the MITEE-C reactor relies on 12 additional beryllium control drums that were not previously evident in the MITEE and MITEE-B designs. This configuration provides a reliable method of starting up and shutting down the reactor, as previously tested on ROVER/NERVA, which achieved a technology readiness level of 6 (Borowski, 1991; Nelson et al., 2021). The control material used is 1.5 cm thick boron carbide with a 100-degree view angle. The 12 control drums extend through the upper axial beryllium reflector to facilitate mechanical-control integration and are spaced radially around the core with a 30-degree separation. The horizontal cross section of both the core geometry and the upper axial beryllium reflectors are shown in Figures 16 and 17 below.

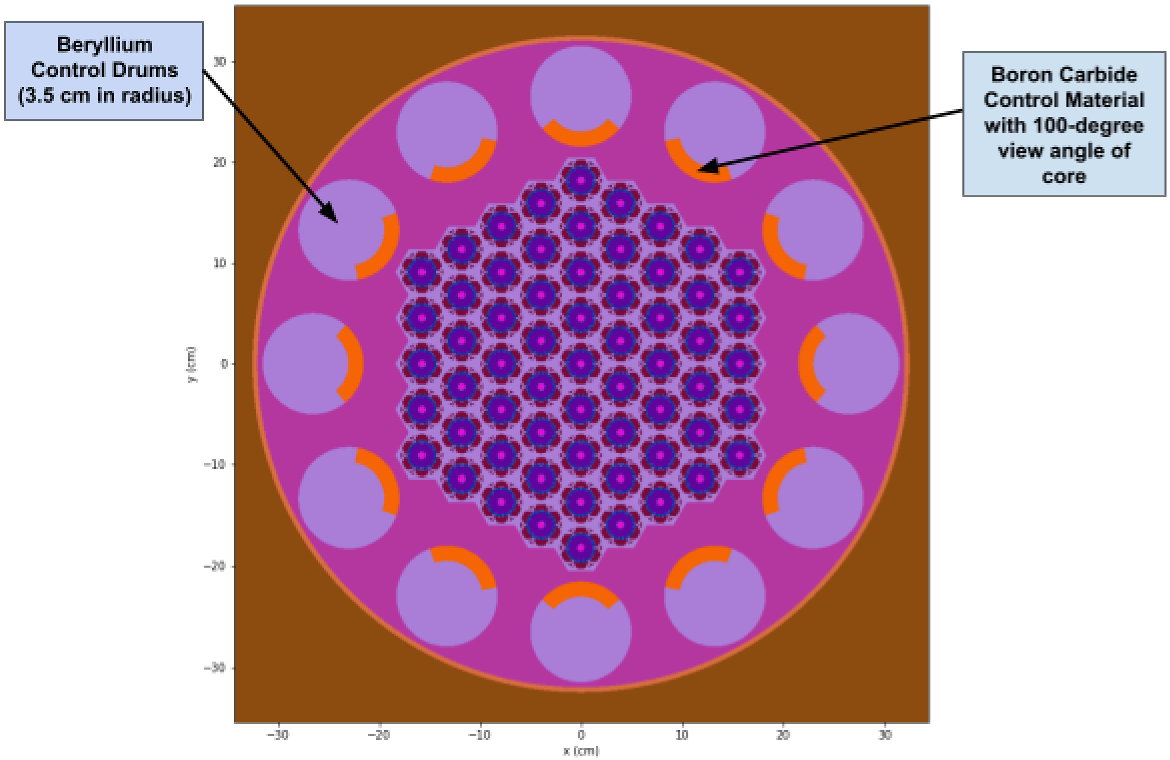


Figure 16. In-core geometry with control drum switched in on position (control material in view of core)

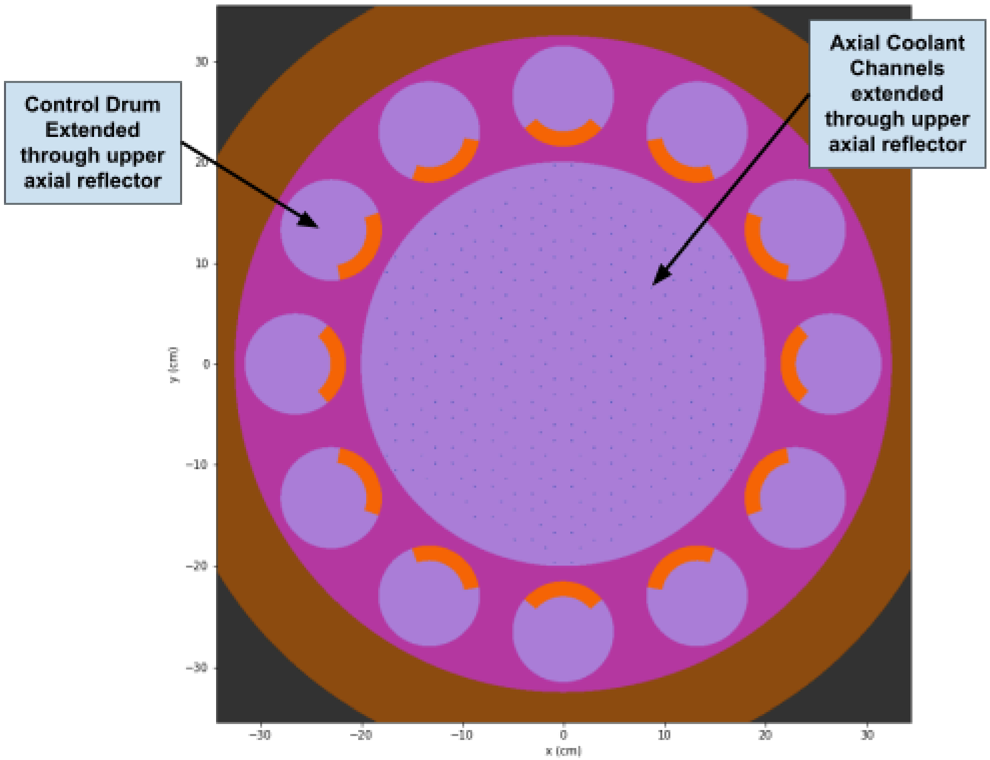


Figure 17. Reflector slice geometry with control drums switched in on position

During shutdown, the control drum layout is identical to that shown in Figures 16 and 17, and in the startup configuration, the control drums are rotated 180 degrees away from the core. The core geometry chosen for MITEE-C showcases a 61 fuel assembly/ element design similar to one of the MITEE and MITEE-B design configurations, as previously discussed. This was chosen in order to ensure that the reactor's Keff remains greater than unity for a HALEU-fueled system. Rounding out the MITEE-C design, each WUO2 cermet fuel region was further modeled as individual 0.025 cm thick fuel sheets. Each of the fuel sheets showcases porous channels 0.009 cm in radius that facilitate diffusive flow of propellant from the cold coolant channel towards the hot frit. This design ensures that the amount of flow channels in each fuel sheet occupies an approximately 25% volume fraction of hydrogen propellant to the 75% volume fraction of WUO2 fuel. Figure 18 below visualizes an example of individual fuel sheets as a cell region within OpenMC.

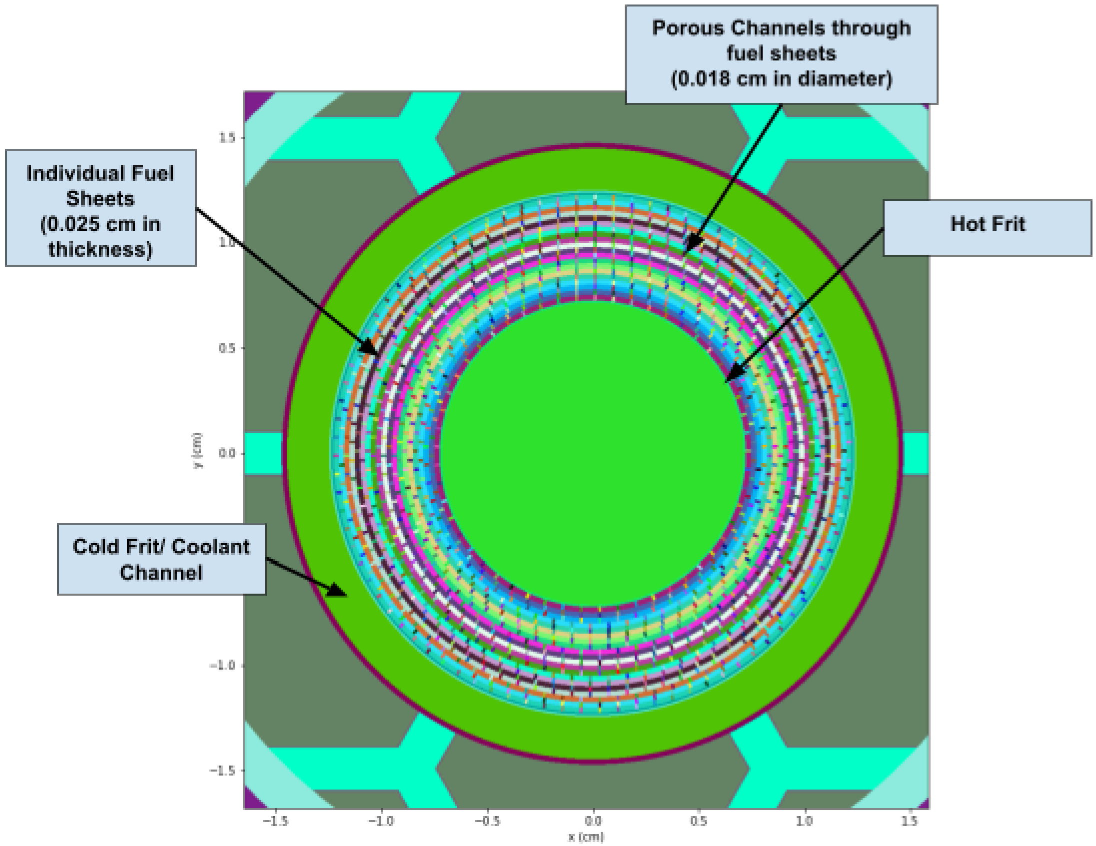


Figure 18. Individual Fuel Sheets Modeled in OpenMC facilitating flow from cold channel to the hot frit/ exhaust channel

In this iteration of the OpenMC simulated MITEE-C design, an initial Keff of 1.03335 was achieved with the reactor when the control drums are in the off position (the core is turned on in startup mode). When the control drums are in the on position, a Keff of 0.91258 was observed, signifying a reasonable Keff margin between startup and shutdown modes.

In all, the MITEE-C design has a total mass of 602 kg[74] and a total thrust of 21,160 Newtons, which yields a key metric of performance:

$$Thrust/weight\ ratio\ = \frac{21160\ [\frac{kg \cdot m}{second^2}]}{602\ kg \cdot 9.80665 [\frac{m}{second^2}]} \cong 3.58$$

[74] Not including auxiliaries, but the estimates in the Spaceplane section account for those separately from the engine, so those need not be included here. Auxiliaries would increase to about 705 kg.

It may be appropriate to consider thrust-to-mass including the mass of the radiation shielding. This study goes into more detail below about how to derive that mass for two options of radiation shielding material. For now, adding 122 kg for the 3 cm thick cylindrical ClearView option shell becomes:

$$Thrust/weight\ ratio = \frac{21160\ [\frac{kg \cdot m}{second^2}]}{724\ kg \cdot 9.80665[\frac{m}{second^2}]} \cong 2.98$$

Or, instead adding 222 kg for the 7 cm thick cylindrical cyanate ester shell option:

$$Thrust/weight\ ratio = \frac{21160\ [\frac{kg \cdot m}{second^2}]}{824\ kg \cdot 9.80665[\frac{m}{second^2}]} \cong 2.62$$

It should be noted that the assumptions may be more relevant to the 1,500 kg version of the engine detailed in the Spacecraft section. Regardless, while this is not the 4 assumed in the Spaceplane section of this report, 4 should be achievable with further optimization, as well as better materials than this report was allowed to cite. This also does not account for the bell nozzle assumed in the Spaceplane section, but that is estimated to have less mass than the radiation shielding, with a correspondingly smaller impact on the ratio. The main design in the Spaceplane section assumes a larger engine massing about 1,500 kg, while the Smaller Payload Option design assumes a slightly smaller engine massing about 500 kg. Of these two options, achieving a ratio of 4 at a total engine mass of 1,500 kg is more likely, in part since the radiation shielding only scales with the two-thirds root of the engine core's mass[75]. Another option, if optimization proves elusive, would be to run it at higher power levels than 100 MW: see the power levels noted for the 3.5 Thrust-To-Weight Ratio designs, which assume engines of 3,000 kg and 1,000 kg.

A rudimentary depletion study was also performed on OpenMC, subsequently on the MITEE-C design, to verify the 100 MW operational power capacity of the reactor core for 45 launch-relaunch cycles. A launch and orbital burn period of 665 seconds and a 275,400-second

[75] The core's mass scales with its volume, but since the radiation shielding's thickness stays constant, the radiation shielding's mass scales with the core's surface area.

cold shutdown period between launches[76] was assumed for the depletion results shown below in Figure 19.

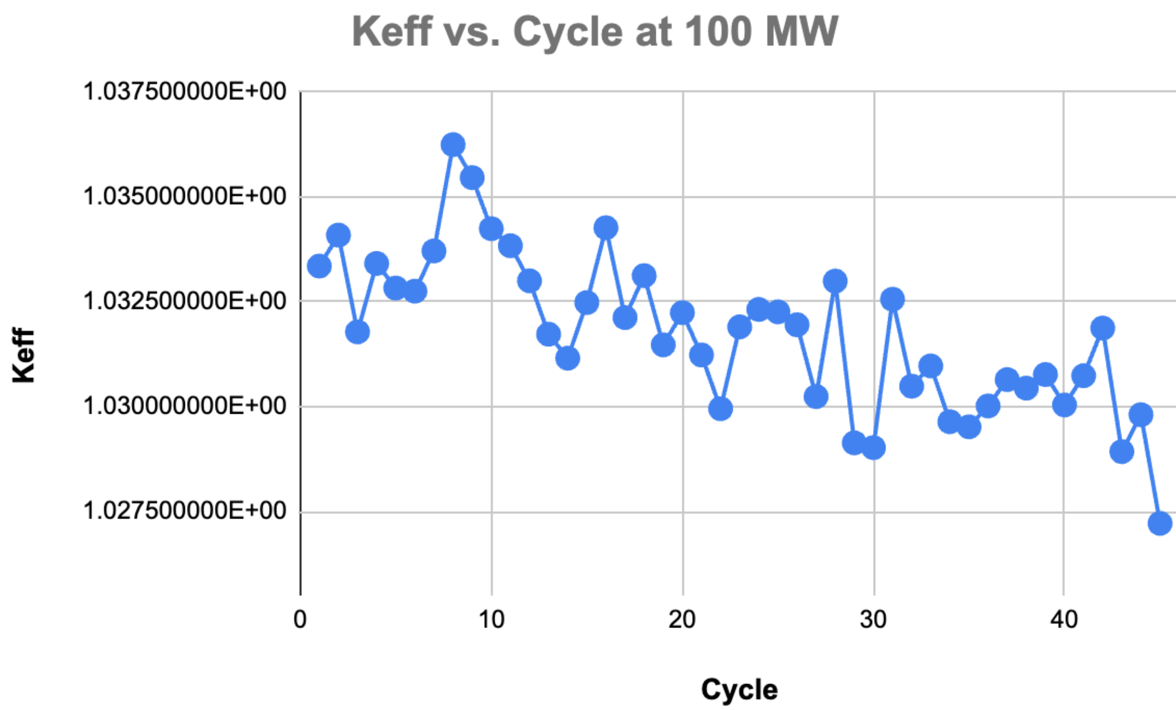


Figure 19. Depletion results across a period of 45 launch-relaunch cycles

[76] Slightly over 72 hours, as measured from the end of the deorbit burn for one mission to the beginning of takeoff for the next. The deorbit burn is short enough, preceded by a long enough cold shutdown period while the launch vehicle is in orbit, that it might be more realistic to measure from the end of one mission's orbital insertion burns ("Hohmann Transfer - Burn 2" in the Spaceplane section of this report) to the beginning of the next mission's takeoff, but the result is approximately the same in practice. This allows for 2 launches per week, which would achieve roughly 100 launches per year after accounting for maintenance, scheduling irregularities, weather-induced missed launch opportunities (assuming launch from a location with a fairly low incidence of such weather), and so forth.

MITEE-C External Flux and Dose

In this section, the unshielded and shielded flux spectra and dose of the MITEE-C design were investigated using OpenMC. For the unshielded study, particle energies and flux of both neutrons and gammas were obtained at a 1 cm thick shell defined at ~21 cm from the reactor's (Rx) radial boundary and ~15 cm from the Rx axial boundary. The core is assumed to be operating at 100 MW as verified by the depletion study. For the below studies, the control drums are in the off position, so the core is in operation.

Neutron Flux Determination

A volumetric normalization factor obtained flux values with units $[\frac{n}{cm^2 \cdot seconds}]$ as OpenMC provides flux values in $[\frac{n-cm}{source}]$. The following methodology was used to determine this normalization factor.

Heating Rate (Massachusetts Institute of Technology, UChicago Argonne LLC, & OpenMC contributors, 2025):

$$H' = 8.2572 \cdot 10^{7}[\frac{eV}{source}] \cdot 1.6022 \cdot 10^{-19}[\frac{Joules}{eV}] = 1.3230 \cdot 10^{-11}[\frac{Joules}{source}]$$

Normalization Factor (reactor core designed for 100 MW power output):

$$F = \frac{Power}{H'} = \frac{100 \cdot 10^{6}[\frac{Joules}{second}]}{1.3230 \cdot 10^{-11}[\frac{Joules}{Source}]} = 7.5588 \cdot 10^{18}[\frac{source}{second}]$$

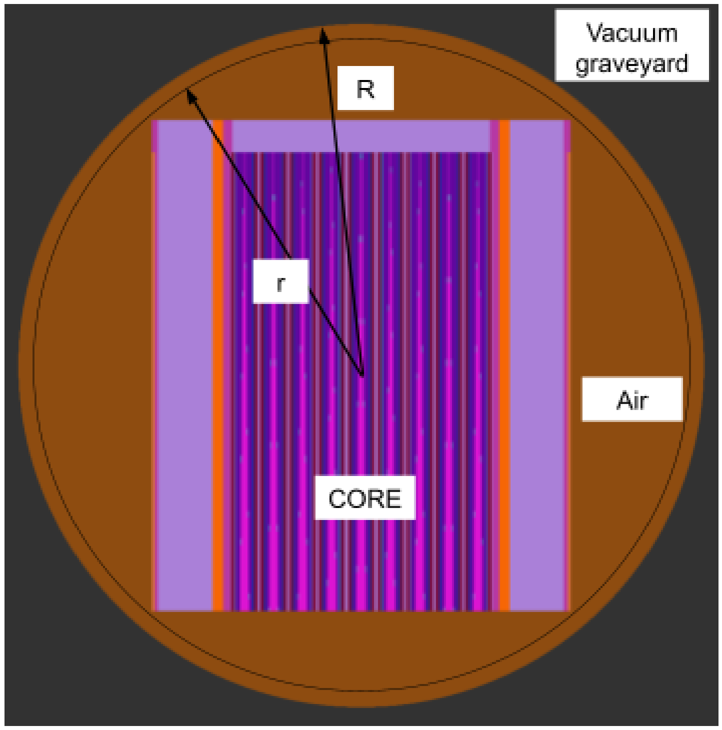


Figure 20. Definition of 1 cm spherical shell layer where flux tallies were defined (Image taken from OpenMC Plotter) - shell thickness not drawn to scale

R is defined as the half height of the reactor core + 15cm = 53.2575 cm. r is defined as the half height of the Rx core + 14 cm = 52.2575 cm. The volume of the 1 cm thick shell: $\frac{4}{3}\pi(R^3 - r^3) = 34977.7027[cm^3]$.

Thus,

$$F/V = \frac{7.5588 \cdot 10^{18}[\frac{source}{second}]}{34977.7027[cm^3]} = 2.1610 \cdot 10^{14}[\frac{source}{cm^3 \cdot second}]$$

When multiplied with the OpenMC provided flux, it will give:

$$\phi = \phi_{OpenMC}[\frac{n-cm}{source}] \cdot F/V[\frac{source}{cm^3 \cdot second}] = \phi[\frac{n}{cm^2 \cdot second}]$$

The flux spectrum is then plotted in Figure 18:

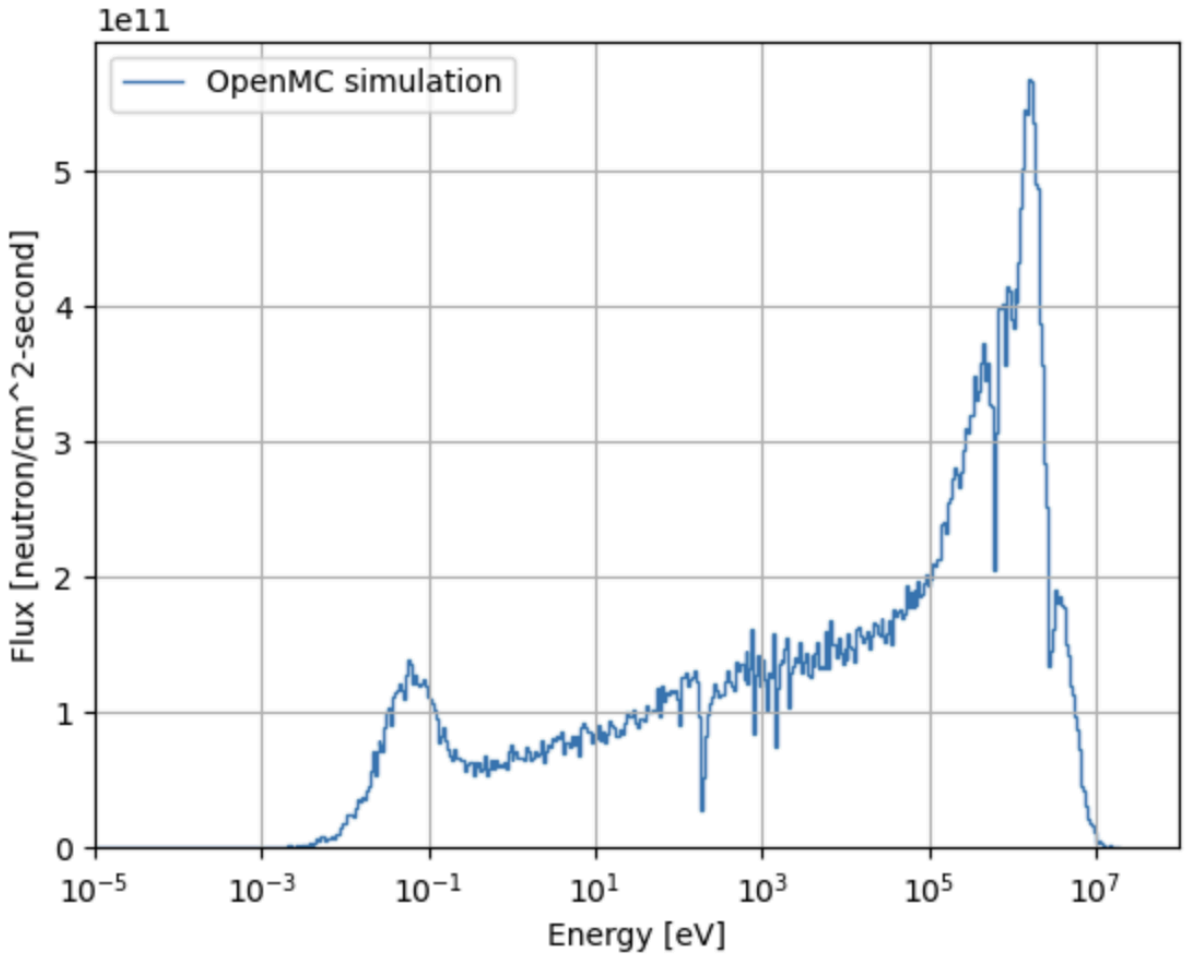


Figure 21. Neutron flux spectrum against neutron energy at the 1 cm thick shell surrounding Rx core at ~21 cm away from Rx radial boundary and ~15 cm away from Rx axial boundary

Gamma Flux Determination

The heating rate is assumed to be constant, and the thickness of the shell layer and distances of r and R were set as a constant in this study. Therefore, the following plot (Figures 22 and 23) was obtained for the gamma flux at the 1 cm thick shell boundary.

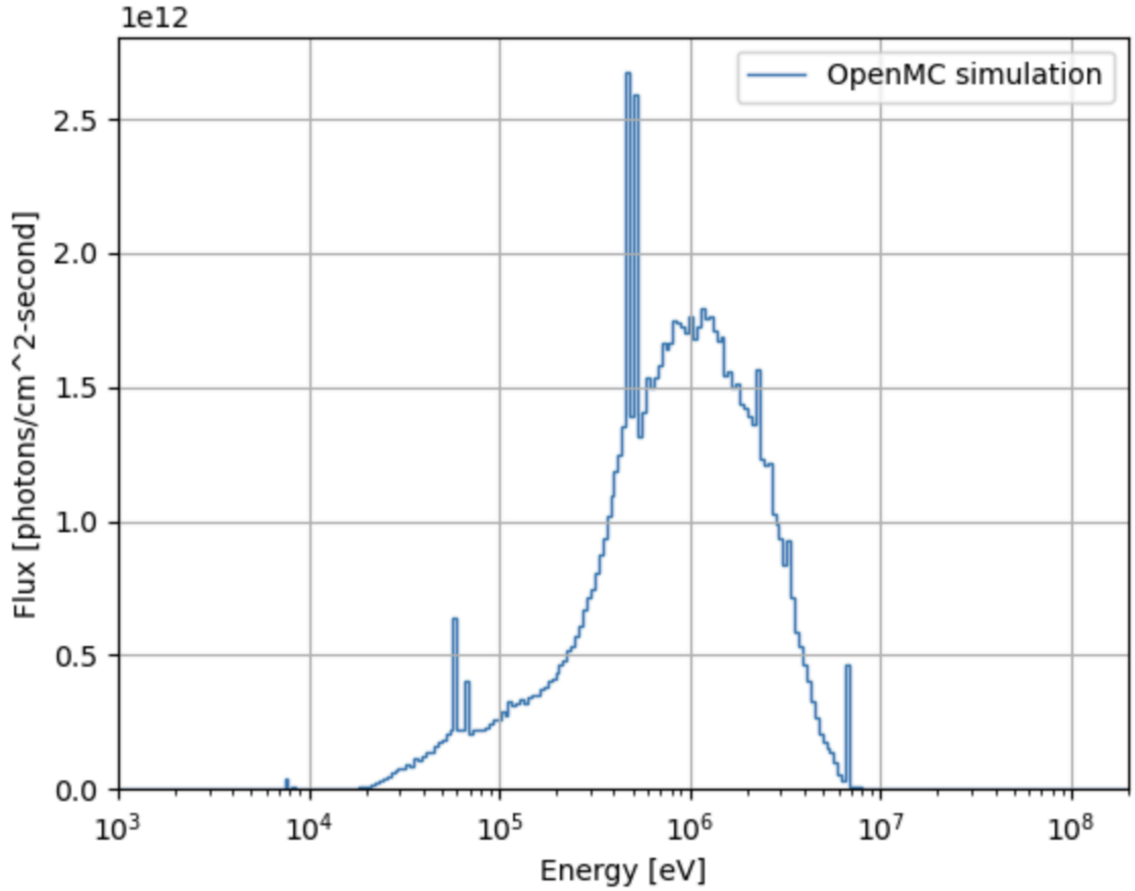


Figure 22. Photon flux spectrum against photon energy at the 1 cm thick shell surrounding Rx core at ~21 cm away from Rx radial boundary and ~15 cm away from Rx axial boundary (semilog)

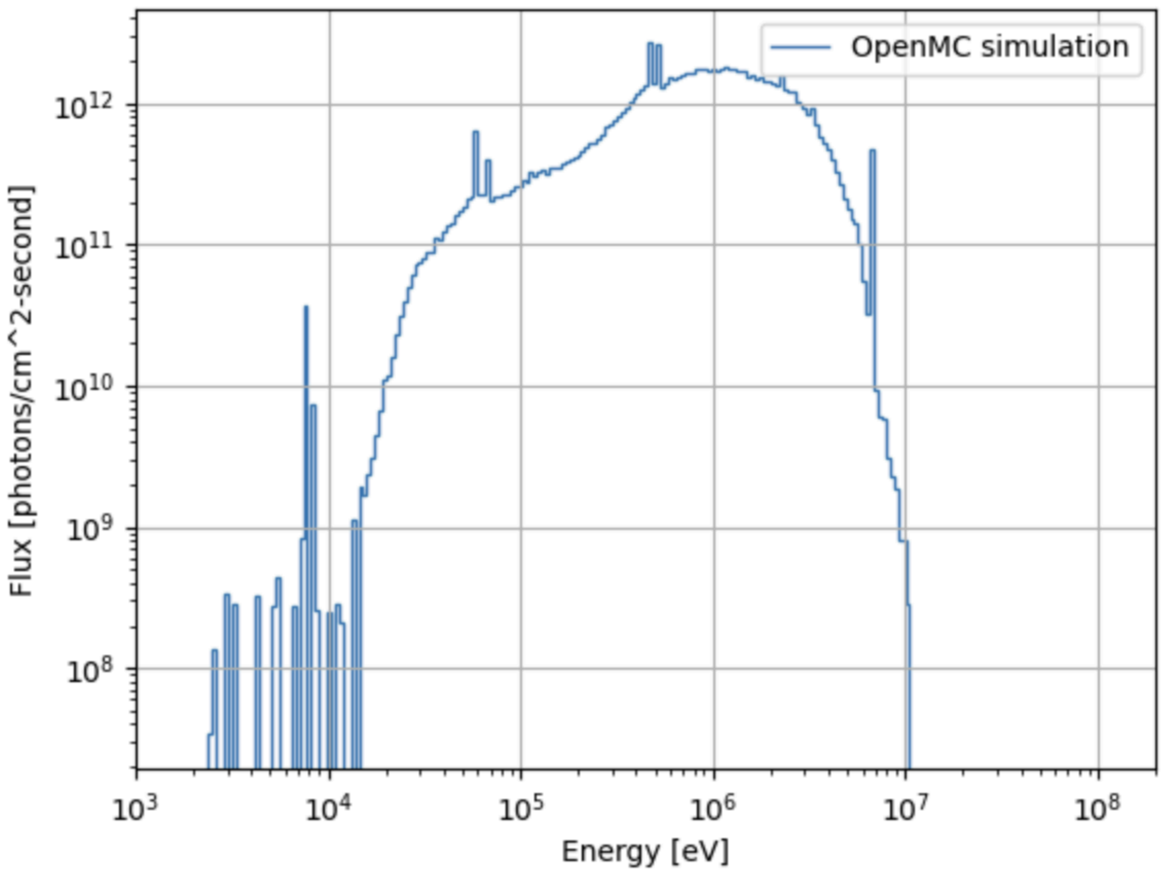


Figure 23. Photon flux spectrum against photon energy at the 1 cm thick shell surrounding Rx core at ~21 cm away from Rx radial boundary and ~15 cm away from Rx Axial boundary (log-log)

MITEE-C Radiation Shielding Study

Now that the dose and flux escaping the MITEE-C core geometry have been quantified, an approach towards determining the necessary radiation-shielding material, thickness, and mass can be taken. This approach assumed the core geometry as a point source, with shielding material modeled in OpenMC as a shell encapsulating the core. Two forms of commercial off-the-shelf radiation-shielding material were investigated in this study. The first of these is the Radium Inc. ClearView Radiation Shielding material (Abernethy & Bakshi, 2019; Bakshi & Chu, 2020). ClearView is a transparent, lead-free shield formed by encasing a liquid ammonium metatungstate solution within a polycarbonate shell. This product provides a non-toxic, non-hazardous alternative to conventional lead-based shielding. The second radiation shielding investigated acknowledges the fact that ClearView is designed for commercial, land-based nuclear power plants, so concern exists as to whether it could survive the harsh operational environment of space flight or launch. Thus, a subsequent radiation-shielding material studied was the cyanate ester resin (Toray, 2023). Carbon-fiber-reinforced cyanate ester resin is a flight-proven material used in the Artemis 1 mission as an aeroshell (backshell and heat shield) for the Orion spacecraft (Toray, 2025) with a current technology readiness level of 6 at minimum, making it suitable as a benchmark material.

Per the FAA's requirements, deriving from NSPM-20 and other regulations as detailed elsewhere in this report, it is desired to achieve a dose to the public of no more than 25 millirem per flight in a non-accident scenario. Virtually all of this will occur during takeoff, as the launch vehicle will be far enough away from the public after takeoff that distance will diminish radiation to a relatively negligible amount, and upon landing the engine will be in a subcritical state with greatly diminished emissions. As noted in the Spacecraft section, a sample takeoff is 8.58 seconds at less than full power; for simplicity and to account for pre-takeoff warmup, this radiation shield study models emissions over 10 seconds. Most of the designs in the Spacecraft section assume a maximum power level significantly above 100 MW, so 100 MW is modeled as being closer to the actual power level during takeoff. Furthermore, members of the public will not be immediately adjacent to the launch vehicle during takeoff, any more than members of the public would stand immediately behind a chemical rocket during takeoff. The only limit to the keep-away distance for the public is that it not be unreasonably large - which is an arbitrary limit, but since current (as of May 2025) rocket launches typically keep the public miles away, anything less than a kilometer should be within this limit.

Both options begin by modeling the radiation shielding as a spherical shell. In practice, a cylindrical shell conformal to the engine is more likely, designed to avoid straight-through paths between inside and outside the shell. This does not change the gamma or neutron flux reduction and resulting public keep-away distance: they are affected by the thickness and material of the shell, but not its shape.

It is simple to calculate the volume of this shell, given the shell's thickness and given that the engine is a cylinder 71.51 cm tall with a 65 cm diameter. For example, as 1 cm of thickness adds 2 cm to each of the height and diameter, the volume of a 1 cm thick cylindrical shell can be modeled as the difference between the volume of two cylinders, the outer one with 67 cm diameter and 73.51 cm tall while the inner one has a 65 cm diameter and is 71.51 cm tall:

$$Volume\,[cm^3] = 3.14159 \cdot ((73.51\,cm \cdot (\frac{67\,cm}{2})^2) - (71.51\,cm \cdot (\frac{65\,cm}{2})^2))$$
$$\simeq 21878.54 cm^3$$

Aside from the specific thicknesses, the volumes of the shells used below use the same formula.

Radium Inc. ClearView Radiation Shielding

The neutron and photon flux spectra leaving the modeled ammonium metatungstate and polycarbonate housing radiation shielding system were obtained with a methodology similar to the one used in the previous section. The cells through which flux tallies (defined as a 1 cm thick tally cell) were filtered are shown in the schematic below (Figure 24).

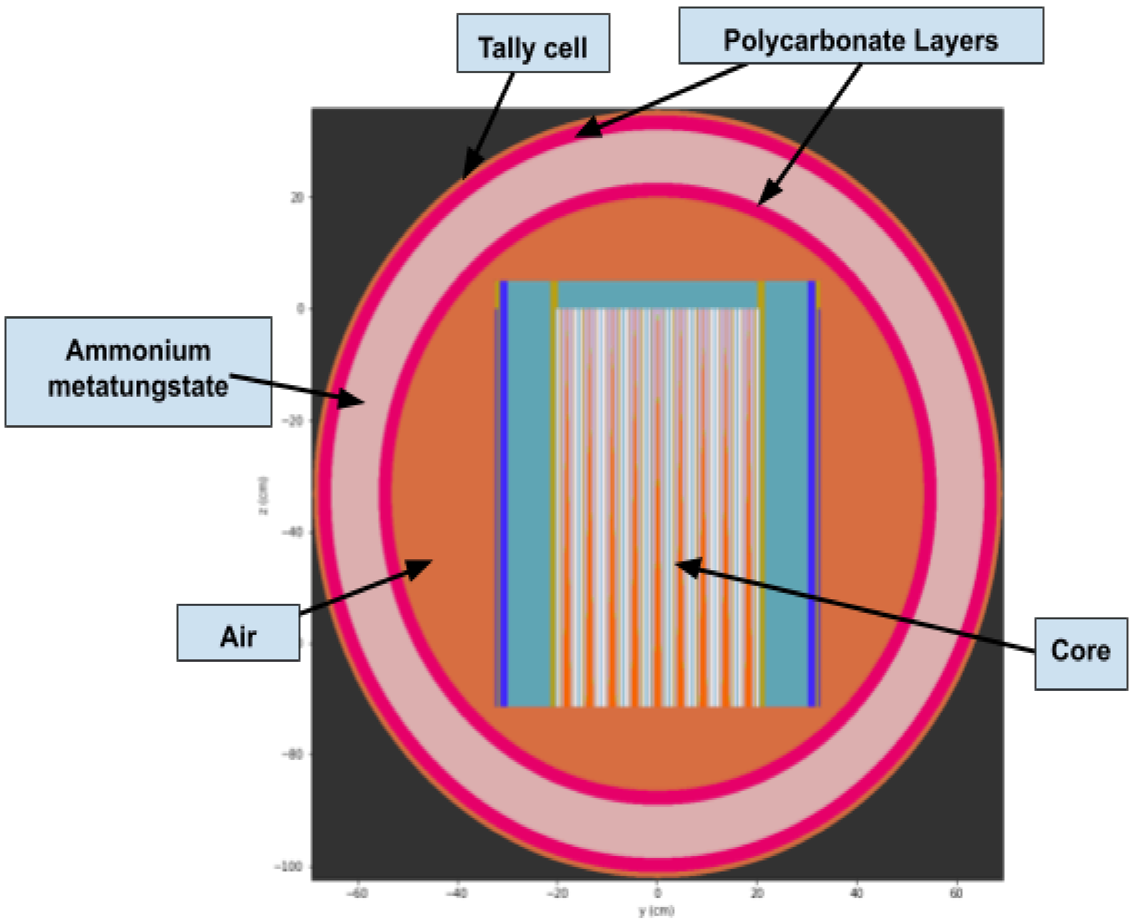


Figure 24. Modeling of ClearView radiation shielding (ammonium metatungstate sandwiched between polycarbonate) against the core geometry as a point source

The polycarbonate and ammonium metatungstate materials were defined by their densities and atomic fractions as outlined in Table 12 below:

Table 12. Material definitions of ClearView model used in radiation shielding study

Material	Density [g/cc]	Atomic Fraction	
Polycarbonate $(C_{16}H_{14}O_{3})_n$	1.21	Carbon	0.485
		Hydrogen	0.424
		Oxygen	0.091
Ammonium Metatungstate $(NH_4)_6H_2W_{12}O_{40}•H_2O$	2.3	Hydrogen	0.594
		Oxygen	0.361
		Tungsten	0.03
		Nitrogen	0.015

Heating rate:

$$H' =$$
$$8.2572 \cdot 10^{7}[\frac{eV}{source}] \cdot 1.6022 \cdot 10^{-19}[\frac{Joules}{eV}] = 1.3230 \cdot 10^{-11}[\frac{Joules}{source}]$$

Normalization factor (reactor core designed for 100 MW power output):

$$F = \frac{Power}{H'} = \frac{100 \cdot 10^{6}[\frac{Joules}{second}]}{1.3230 \cdot 10^{-11}[\frac{Joules}{Source}]} = 7.5588 \cdot 10^{18}[\frac{source}{second}]$$

R is defined as the radius of the outer boundary of the tally shell outside of the shielding = 68.8625 [cm]. r is defined as the radius of the outer boundary of the shielding = 67.8625 [cm]. The volume of the 1 cm thick tally shell: $\frac{4}{3}\pi(R^3 - r^3) = 58729.12\ [cm^3]$.

Thus,

$$F/V = \frac{7.5588 \cdot 10^{18}[\frac{source}{second}]}{58729.11829[cm^3]} = 1.2871 \cdot 10^{14}[\frac{source}{cm^3 \cdot second}]$$

When multiplied with the OpenMC provided flux, it will give:

$$\phi = \phi_{OpenMC}[\frac{n-cm}{source}] \cdot F/V[\frac{source}{cm^3 \cdot second}] = \phi[\frac{n}{cm^2 \cdot second}]$$

The neutron and photon flux spectrum departing the radiation shielding is thus plotted in Figures 19 and 20.

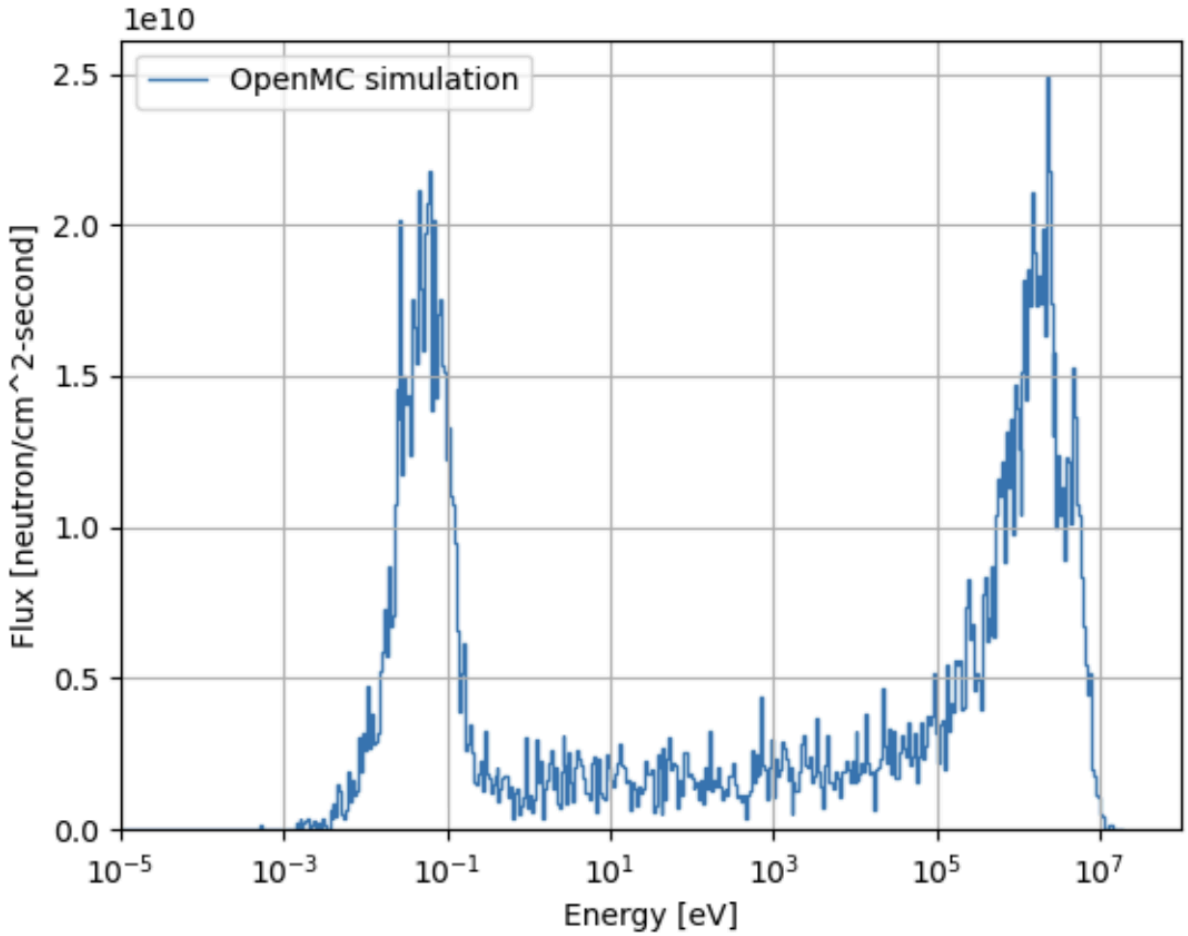


Figure 25. Neutron flux spectrum against neutron energy at the 1 cm thick shell surrounding Clearview radiation shielding at ~22 cm away from Rx radial boundary and ~16 cm away from Rx axial boundary

Comparing the non-shielded incident neutron flux in Figure 21 to the flux departing the shielding in Figure 25, the fast neutron peak is drastically reduced from a magnitude of $\sim 6 \cdot 10^{11}$ to $\sim 2.5 \cdot 10^{10}$. There was also a reduction in the thermal neutron peak, however, not as drastic as the fast flux reduction ($\sim 1 \cdot 10^{11}$ to $\sim 2.0 \cdot 10^{10}$).

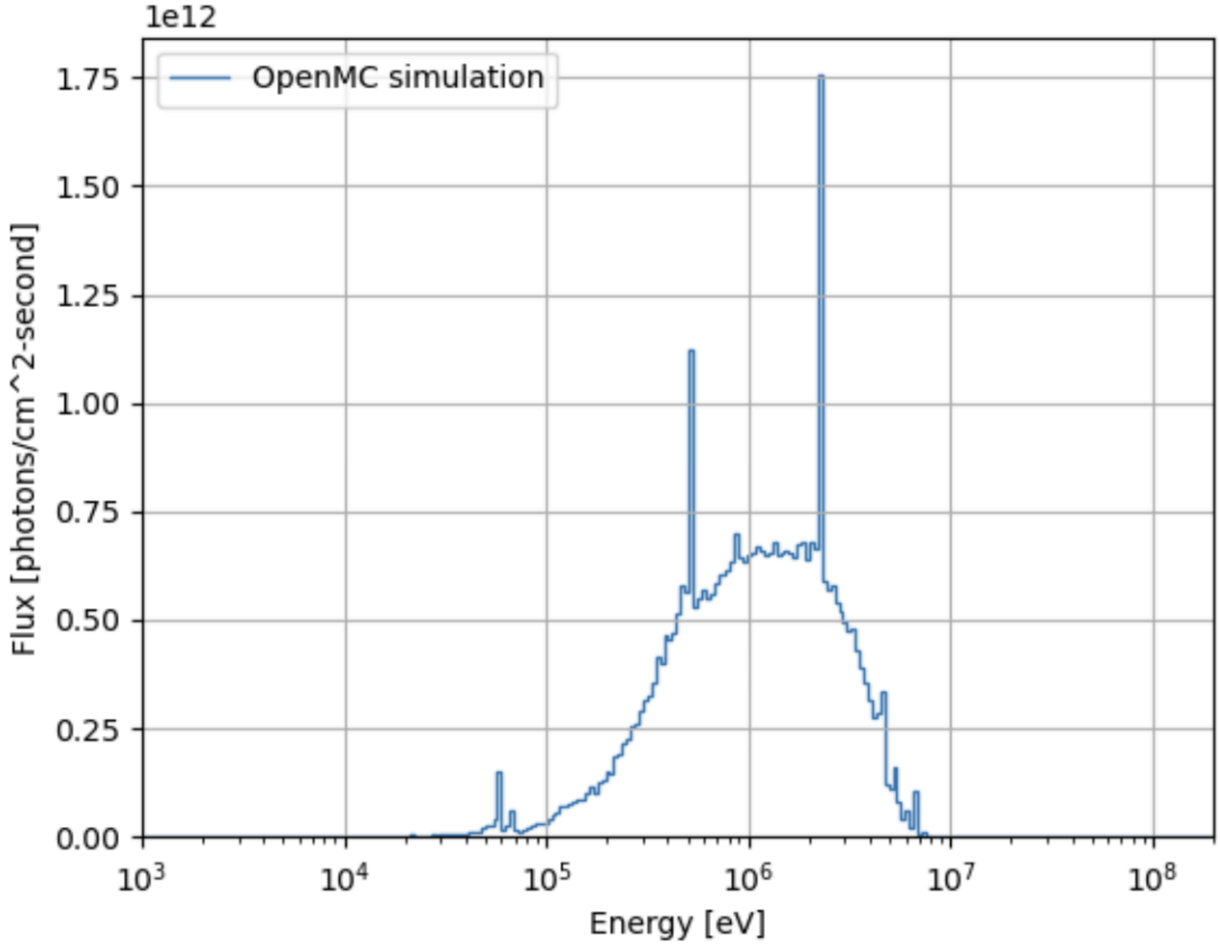


Figure 26. Photon flux spectrum against photon energy at the 1 cm thick shell surrounding Clearview radiation shielding at ~22 cm away from Rx radial boundary and ~16 cm away from Rx axial boundary (semilog)

Now, comparing the non-shielded incident gamma flux in Figure 22 to the flux departing the shielding Figure 26, the bulk peak flux was reduced $\sim 1.5 \cdot 10^{12}$ to $\sim 7.5 \cdot 10^{11}$. Therefore, one could postulate that the effectiveness in gamma shielding is less effective compared to neutron shielding for the simulated Clearview model.

Dose value results from OpenMC are provided in units of $pSv - cm^2$. From the simulation results provided, it is evident that the modeled thickness could reduce the photon dose and lower the OpenMC-derived neutron dose values by approximately an order of magnitude.

$Impact\ Photon\ Dose = 1.9911\ pSv - cm^2$

$Leaving\ Photon\ Dose = 1.5781\ pSv - cm^2$

$$Impact\ Neutron\ Dose\ =\ 26.337\ pSv\ -\ cm^{2}$$
$$Leaving\ Neutron\ Dose =\ 2.0992\ pSv\ -\ cm^{2}$$

However, neutron dose is nominally reported in units of rem. Thus, the fast neutron flux peak values, as shown in Figures 15 and 19, were divided by the respective energy-dependent fluence per unit dose equivalent values ("Units of...," 2017) to achieve dose in rem per seconds:

The photon and neutron dose incident on modeled shielding material for neutrons at 2.5 MeV:

$$Photon\ Dose\ [rem/s] \simeq 1.5\ \cdot\ 10^{12}\frac{photons}{cm^{2}-second}\cdot\ 2.5MeV\ \cdot\ 5.76\ \cdot\ 10^{-9}\frac{cm^{2}-rem}{MeV-photon}$$
$$=\ 2.16\ \cdot\ 10^{4}\frac{rem}{seconds}$$
$$Neutron\ Dose\ [rem/s] \simeq \frac{6\cdot 10^{11}[\frac{n}{cm^{2}-seconds}]}{29\cdot 10^{6}[\frac{n}{cm^{2}-rem}]} =\ 2.0689\ \cdot\ 10^{4}\frac{rem}{seconds}$$

The photon and neutron doses leaving modeled shielding material for neutrons at 2.5 MeV:

$$Photon\ Dose\ [rem/s] \simeq 7.5\ \cdot\ 10^{11}\frac{photons}{cm^{2}-second}\cdot\ 2.5MeV\ \cdot\ 5.76\ \cdot\ 10^{-9}\frac{cm^{2}-rem}{MeV-photon}$$
$$=\ 1.08\ \cdot\ 10^{4}\frac{rem}{seconds}$$
$$Neutron\ Dose\ [rem/s] \simeq \frac{2.5\cdot 10^{10}[\frac{n}{cm^{2}-seconds}]}{29\cdot 10^{6}[\frac{n}{cm^{2}-rem}]} =\ 862.07\frac{rem}{seconds}$$

Over 10 seconds, this is a total of 116,620.7 rem at R = 68.8625 cm. As rem diminish by the square of the distance, and it is desired to achieve 0.025 rem:

$$Distance\ [meters]\ =\ 0.688625\ m\ \cdot\ \sqrt{\frac{116620.7\ rem}{0.025\ rem}} \simeq 1487m$$

This exceeds the 1,000 meter target. Doubling the thickness would reduce the gamma and neutron doses by the ratios above, but this would only achieve a distance of 1,015 meters. Tripling the thickness, which reduces the doses by the ratios squared, is necessary:

$$Photon\ Dose\ [rem/s]\ =\ 1.08\ \cdot\ 10^{4}\frac{rem}{seconds}\ \cdot\ \left(\frac{1.08\cdot 10^{4}\frac{rem}{seconds}}{2.16\cdot 10^{4}\frac{rem}{seconds}}\right)^{2}$$

$$=\ 2.7\ \cdot\ 10^{3}\frac{rem}{seconds}$$

$$Neutron\ Dose\ [rem/s]\ =\ 862.07\frac{rem}{seconds}\ \cdot\ \left(\frac{862.07\frac{rem}{seconds}}{2.0689\cdot 10^{4}\frac{rem}{seconds}}\right)^{2}\simeq 1.50\frac{rem}{seconds}$$

Over 10 seconds, this is a total of 27,015 rem at R = 68.8625 cm, reducing the public keep-away distance to:

$$Distance\ [meters]\ =\ 0.688625\ m\ \cdot\ \sqrt{\frac{27015\ rem}{0.025\ rem}}\simeq 716m$$

So, using this solution, if the public is kept at least 716 meters away from the launch vehicle during takeoff, the maximum permitted radiation dose to the public would not be exceeded.

The volume of a 3 cm thick cylindrical shell is:

$$Volume\ [cm^{3}]\ =\ 3.14159\ \cdot\ \left(\left(77.51\ cm\ \cdot\ \left(\frac{71\ cm}{2}\right)^{2}\right)\ -\ \left(71.51\ cm\ \cdot\ \left(\frac{65\ cm}{2}\right)^{2}\right)\right)$$

$$\simeq 69584.77cm^{3}$$

Although the authors of this study lack data for the proportionate thicknesses of the polycarbonate shell and the ammonium metatungstate interior, assuming the complete volume of shielding has an average density of 1.755 g/cc[77] should get a fairly accurate estimate of the resulting mass for a 3 cm thick cylindrical shell:

$$Mass\ [kg]\ =\ 1.755\ \frac{g}{cm^{3}}\ \cdot\ 0.001\ \frac{kg}{g}\ \cdot\ 69584.77\ cm^{3}\simeq 122\ kg$$

It should be noted that the public keep-away distance and mass of radiation shielding are minimums to achieve the regulatory limit within the assumptions of this study. The minimums in practice may be less.

[77] The average density of the two materials.

Cyanate Ester Radiation Shielding

The neutron and photon flux spectra leaving the modeled cyanate radiation shielding system were obtained with a methodology once again similar to the one used in the previous section. The cells through which flux tallies (defined as a 1 cm thick tally cell) were filtered shown in the Figure 27 schematic below.

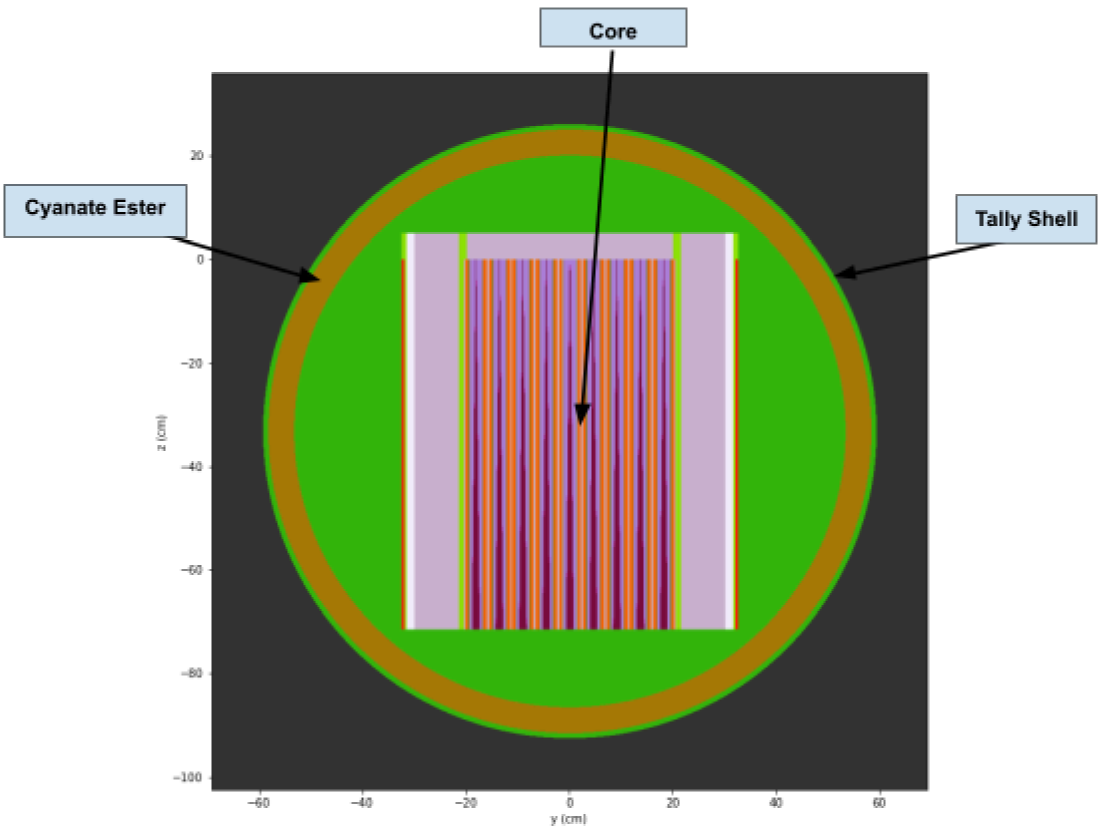


Figure 27. Modeling of cyanate ester radiation shielding against the core geometry as a point source

The cyanate ester was similarly defined by its density and atomic fractions as outlined in Table 13 below:

Table 13. Material definitions of cyanate ester model used in radiation shielding study

Material	**Density [g/cc]**	**Atomic Fraction**	
Cyanate Ester $C_{17}H_{14}N_2O_2$	1.22	Carbon	0.486
		Hydrogen	0.4
		Oxygen	0.057
		Nitrogen	0.057

Heating rate:

$$H' = 8.2572 \cdot 10^{7}[\frac{eV}{source}] \cdot 1.6022 \cdot 10^{-19}[\frac{Joules}{eV}] = 1.3230 \cdot 10^{-11}[\frac{Joules}{source}]$$

Normalization factor (reactor core designed for 100 MW power output):

$$F = \frac{Power}{H'} = \frac{100 \cdot 10^{6}[\frac{Joules}{second}]}{1.3230 \cdot 10^{-11}[\frac{Joules}{Source}]} = 7.5588 \cdot 10^{18}[\frac{source}{second}]$$

R is defined as the radius of the outer boundary of the tally shell outside of the shielding = 59.2575 [cm]. r is defined as the radius of the outer boundary of the shielding = 58.2575 [cm]. The volume of the 1 cm thick tally shell: $\frac{4}{3}\pi(R^3 - r^3) = 43385.7356\ [cm^3]$

Thus,

$$F/V = \frac{7.5588 \cdot 10^{18}[\frac{source}{second}]}{43385.7356[cm^3]} = 1.7422 \cdot 10^{14}[\frac{source}{cm^3 \cdot second}]$$

When multiplied with the OpenMC provided flux, it will give:

$$\phi = \phi_{OpenMC}[\frac{n-cm}{source}] \cdot F/V[\frac{source}{cm^3 \cdot second}] = \phi[\frac{n}{cm^2 \cdot second}]$$

The neutron and photon flux spectrum departing the radiation shielding is thus plotted in the figure below

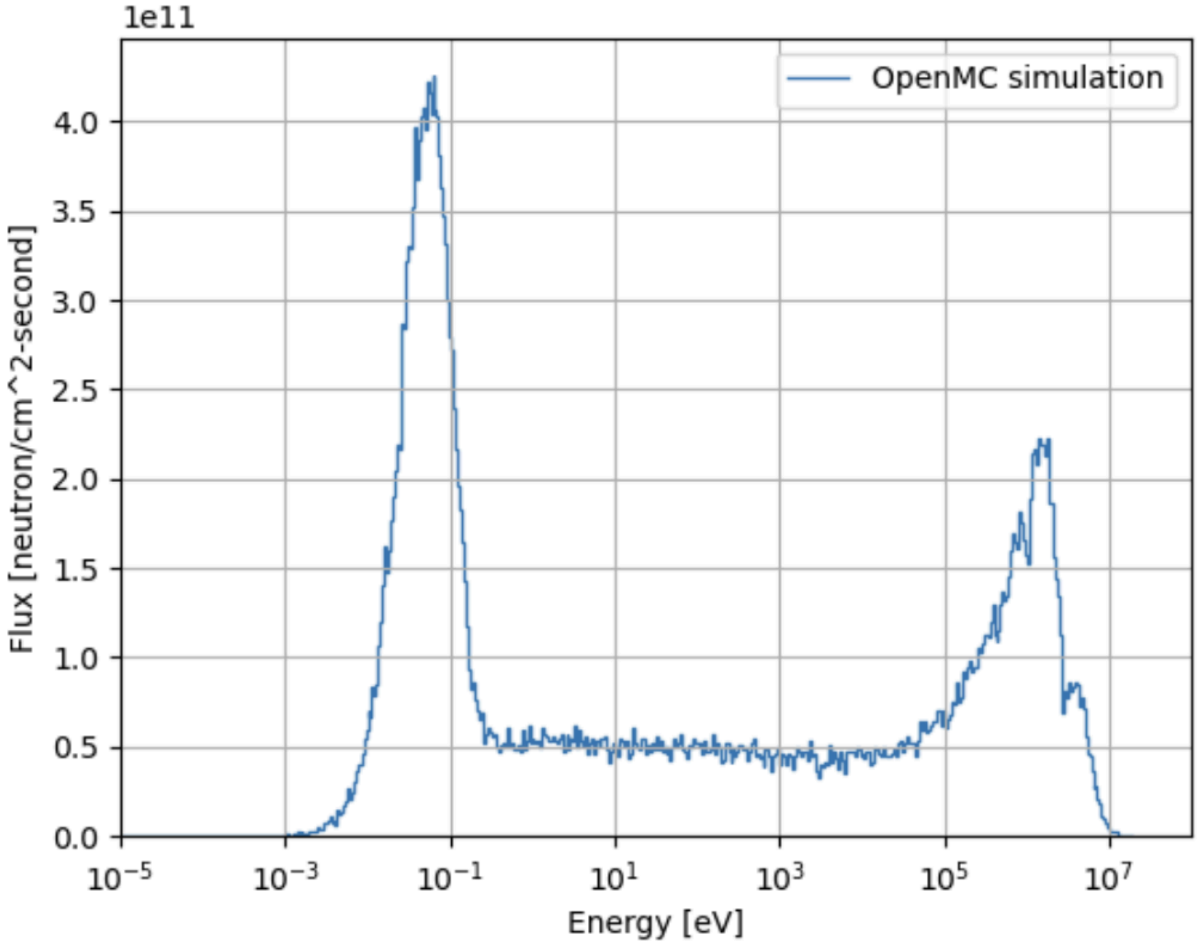


Figure 28. Neutron flux spectrum against neutron energy at the 1 cm thick shell surrounding cyanate ester radiation shielding at ~22 cm away from Rx radial boundary and ~16 cm away from Rx axial boundary

Comparing the non-shielded incident neutron flux in Figure 18 to the flux departing the shielding Figure 28, the fast neutron peak is reduced from a magnitude of $\sim 6 \cdot 10^{11}$ to $\sim 2.2 \cdot 10^{11}$. There was also a slight increase in the thermal neutron peak value ($\sim 1 \cdot 10^{11}$ to $\sim 4 \cdot 10^{11}$), which could be associated with neutron moderation.

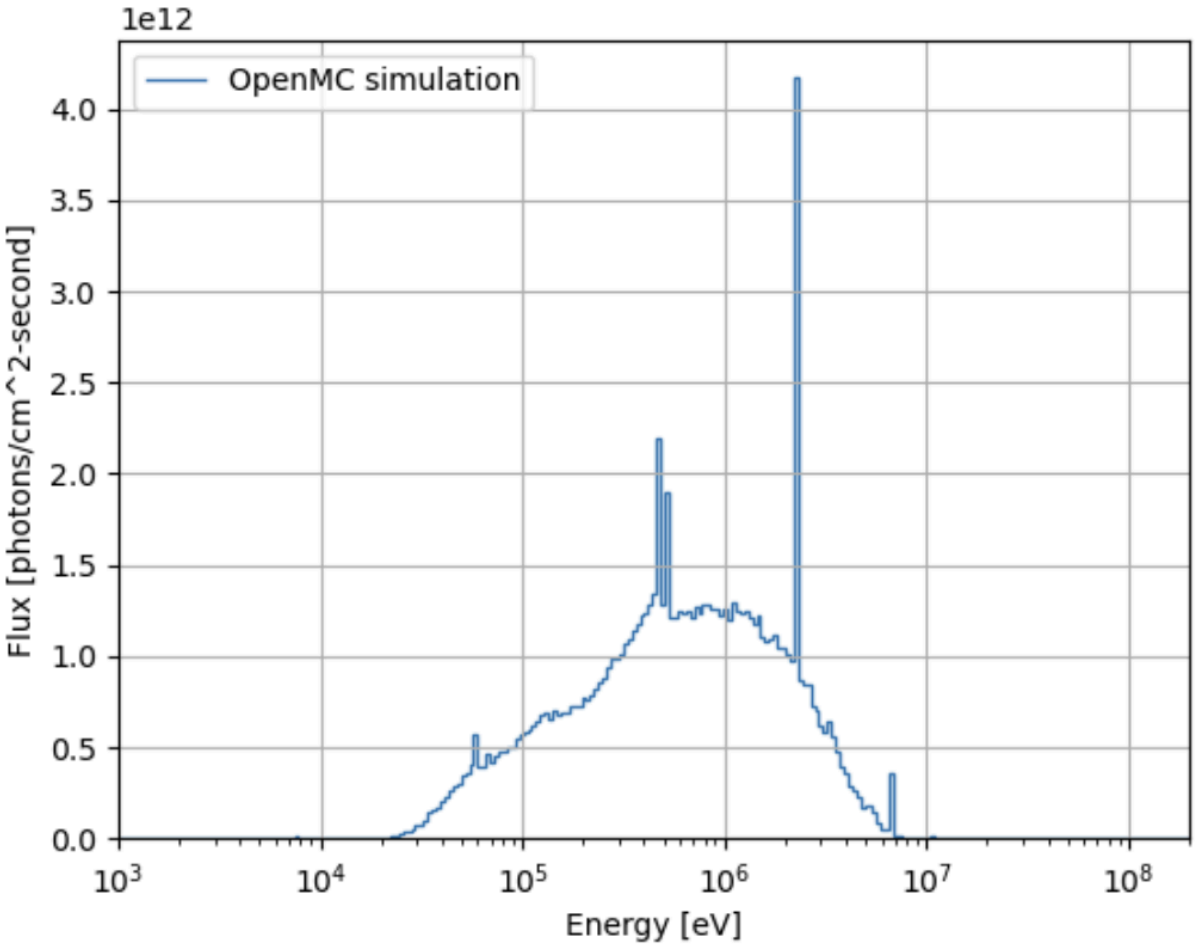


Figure 29. Photon flux spectrum against photon energy at the 1 cm thick shell surrounding cyanate ester radiation shielding at ~22 cm away from Rx radial boundary and ~16 cm away from Rx axial boundary (semilog)

Once again, comparing the non-shielded incident gamma flux in Figure 19 to the flux departing the shielding Figure 29, the bulk peak flux was mildly reduced from $\sim 1.5 \cdot 10^{12}$ to $\sim 1.25 \cdot 10^{12}$.

The dose value results from OpenMC are noted below in units of $pSv - cm^2$. From the simulation results provided, it is apparent that cyanate ester, even though having a higher technology readiness level, performs less efficiently when compared to an equal thickness of ClearView in shielding neutrons.

$$Impact\ Neutron\ Dose\ = 26.337\ pSv\ -\ cm^2$$

$$Leaving\ Neutron\ Dose = 15.405\ pSv\ -\ cm^2$$

The photon and neutron doses are once more reported in units of rem below. The fast neutron flux peak values, as shown in Figures 21 and 28, were divided by the respective energy-dependent fluence per unit dose equivalent values ("Units of...," 2017) to achieve dose in rem per seconds. The photon and neutron doses incident on modeled shielding material for photons and neutrons at 2.5 MeV:

$$Photon\ Dose\ [rem/s] \simeq 1.5 \cdot 10^{12} \frac{photons}{cm^2-second} \cdot 2.5MeV \cdot 5.76 \cdot 10^{-9} \frac{cm^2-rem}{MeV-photon}$$

$$= 2.16 \cdot 10^4 \frac{rem}{seconds}$$

$$Neutron\ Dose\ [rem/s] \simeq \frac{6 \cdot 10^{11}[\frac{n}{cm^2-seconds}]}{29 \cdot 10^6[\frac{n}{cm^2-rem}]} = 2.0689 \cdot 10^4 \frac{rem}{seconds}$$

The photon and neutron doses leaving modeled shielding material for neutrons at 2.5 MeV:

$$Photon\ Dose\ [rem/s] \simeq 1.25 \cdot 10^{12} \frac{photons}{cm^2-second} \cdot 2.5MeV \cdot 5.76 \cdot 10^{-9} \frac{cm^2-rem}{MeV-photon}$$

$$\simeq 1.81 \cdot 10^4 \frac{rem}{seconds}$$

$$Neutron\ Dose\ [rem/s] \simeq \frac{2.2 \cdot 10^{11}[\frac{n}{cm^2-seconds}]}{29 \cdot 10^6[\frac{n}{cm^2-rem}]} = 7.586 \cdot 10^3 \frac{rem}{seconds}$$

Over 10 seconds, this is 256,860 rem at R = 59.2575 cm. As rem diminish by the square of the distance, and it is desired to achieve 0.025 rem, the public keep-away distance is:

$$Distance\ [meters] = 0.592575\ m \cdot \sqrt{\frac{256860\ rem}{0.025\ rem}} \simeq 1899m$$

As with ClearView, 1 cm is not enough. It turns out that even 6 cm will not achieve the target but 7 cm, which reduces the doses by the above ratios to the 6th power, will:

$$Photon\ Dose\ [rem/s] = 1.81 \cdot 10^4 \frac{rem}{seconds} \cdot (\frac{1.81 \cdot 10^4 \frac{rem}{seconds}}{2.16 \cdot 10^4 \frac{rem}{seconds}})^6$$

$$\simeq 6.27 \cdot 10^3 \frac{rem}{seconds}$$

$$Neutron\ Dose\ [rem/s] = 7.586 \cdot 10^3 \frac{rem}{seconds} \cdot (\frac{7.586 \cdot 10^3 \frac{rem}{seconds}}{2.0689 \cdot 10^4 \frac{rem}{seconds}})^6$$

$$\simeq 18.44 \frac{rem}{seconds}$$

Over 10 seconds, this is a total of 62,884.4 rem at R = 68.8625 cm, reducing the public keep-away distance to:

$$Distance\ [meters] = 0.592575\ m \cdot \sqrt{\frac{62884.4\ rem}{0.025\ rem}} \simeq 940m$$

So, using this solution, if the public is kept at least 940 meters away from the launch vehicle during takeoff, the maximum permitted radiation dose to the public would not be exceeded. The volume of a 7 cm thick shell is:

$$Volume\ [cm^3] = 3.14159 \cdot ((85.51\ cm \cdot (\frac{79\ cm}{2})^2) - (71.51\ cm \cdot (\frac{65\ cm}{2})^2))$$
$$\simeq 181849.49 cm^3$$

Given the density and volume, the mass is easily obtained for a 7 cm thick cylindrical shell:

$$Mass\ [kg] = 1.22\ \frac{g}{cm^3} \cdot 0.001\ \frac{kg}{g} \cdot 181849.49\ cm^3 \simeq 222\ kg$$

As with the ClearView study, the public keep-away distance and mass of radiation shielding are minimums to achieve the regulatory limit within the assumptions of this study, and the minimums in practice may be less.

Radiation Shielding Summary

All of the alternatives examined add less than 250 kg to the engine's mass. Arguably this should be factored into the engine's thrust-to-weight ratio. As seen in the calculations above for the thrust-to-weight ratios, doing so does not drop said ratio below 1. Some have claimed it is not possible to achieve this level of protection with less than multiple tons of shielding, but a close examination of such arguments presented during this study revealed no accounting for the small size of the engine core. Keeping the minimum required shielding mass low is one of the reasons why this study proposes a fairly miniature design.

In both the ClearView and cyanate ester cases, public minimum keep-away distances for launch of under 1 kilometer were achieved. As noted elsewhere in this report, noise concerns from supersonic exhaust may dictate larger keep-away distances in practice.

It may be possible to get away with less or no radiation shielding on the dorsal side, as that should never be pointed at any member of the public during a non-accident scenario. The ventral side will be pointed at the runway and ocean at various points during takeoff, while the engine will be shut down and thus not need shielding during landing, which would argue for its exclusion from radiation shielding needs as well. However, these options potentially increase radiation danger in an accident scenario and for that reason are not preferred. That said, it may be possible to optimize the shape. For instance, this study assumes a perfectly cylindrical shell, which results in greater effective thickness at the corners; rounding the corners to achieve uniform thickness would slightly reduce the volume and thus mass.

As noted elsewhere in this study, the authors are aware of better-performing materials, but did not have permission to cite their performance for this study. Therefore, this study focuses on whether it is possible to achieve the regulatory threshold for public exposure limits, assuming the public is kept at least 1 kilometer away, with a reasonable mass using materials this report was allowed to cite the performance of. Any practical effort would likely use said better materials. Further, this study considers placing the engine outside the propellant tank so the hot hydrogen has a short path to exit via the bell nozzle. If the engine was in the middle of the tank, the propellant would provide much more shielding during the takeoff phase before most of it was used, though the exit path would need to be insulated from the surrounding cryogenic hydrogen.

The focus of this study is on safety to the public, but three further considerations deserve a bit of examination:

1. The payloads will, with the current design, incur significantly higher doses than the public due both to their closer proximity and longer duration of exposure. While the payloads would be at the far end of the launch vehicle from the engine, in or near the nosecone, providing additional shielding from the propellant tank, the smallest of the designs contemplated in the Spacecraft section of this report would have the payload only

about 10 meters from the center of the engine. This may present difficulties for certain payloads, but most payloads should be powered off and generally inert until after deployment, which should mitigate these difficulties. Additional shielding specific to the payload compartment, a configuration commonly named "shadow shield", can be installed in future versions, such as larger launch vehicles, which might carry human passengers or crew. Alternatively, a mission-specific shadow shield can be provided for any particularly sensitive payloads[78], coming out of the payload's mass allowance: for instance, a 12 kg, 6U CubeSat could be launched with 12 kg of shadow shield, summing to the allowed 24 kg of payload, on the 12U launcher.

2. The spacecraft's electronics will have to be designed to operate under the expected radiation dose. This can be accomplished by use of radiation-hardened commercially-available hardware.
3. As noted elsewhere in this report, repeated exposure to high amounts of radiation can degrade a spaceport's infrastructure. The Economic Analysis appendix includes consideration of replacing the runway every 4 years. Proper material selection can minimize the problems this causes for the spaceplane itself, but this may be a limiting factor on the spaceplane's operational lifetime[79]. A higher amount of radiation shielding, so long as it does not reduce performance to less than that needed to achieve orbit, might be justified to reduce maintenance costs, which would have to be paid from the launch vehicle program's revenue.

[78] As noted in the Economic Analysis appendix, the likely majority customer would be paying for the launch as a whole, not just for a slot on a ride to whatever orbit they can get. A customer with a 6U seeking a dedicated launch would pay the same as a customer with a 12U. The price of radiation shielding is low enough that filling unused payload capacity with shadow shielding appears to be a financially practical option in these cases, if the payload would significantly benefit from it.

[79] A prototype spaceplane would likely only operate for a few decades anyway, before being replaced with better design. Material degradation from high radiation exposure is not likely to drop its operational lifetime below this range (though some components will need replacing before then), but may help determine the specific retirement date.

Spaceplane

An engine by itself is not an entire launch vehicle. So as to confirm what performance would be needed, this study models what a launch vehicle using this engine and its launch mission may look like. This presents notional values a future version of the engine should aim to achieve.

The resulting spaceplane[80] is a horizontal takeoff, horizontal landing vehicle, so as to allow an accelerated stop should there be a failure on engine startup[81], accepting the mass penalty of wings and landing gear in trade for greater safety, and consisting of a reusable single stage to orbit[82]. A NTP launch vehicle may require[83] a spaceport separate from any other spaceport, but a spaceplane's spaceport can be converted[84] from existing airports allowing launch over unpopulated areas, such as the ocean. This model assumes the engine can be improved to achieve a vacuum thrust-to-engine-weight[85] of at least 4, and uses an aerospike nozzle to reduce specific impulse losses to the atmosphere[86].

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[80] Visually, this would look like the Space Shuttle but smaller, taking off horizontally without solid rocket boosters or an external propellant tank. A low-polygon render is part of Figure 1 in the ConOps section. Like the Space Shuttle it would land on a runway, though unlike the Space Shuttle it would ideally land where it took off from. The biggest difference is internal: mostly propellant tank rather than mostly cargo space, with only a small payload section up front and no crew space.

[81] "Failure on engine startup", for a vertical takeoff launch vehicle, would be particularly hazardous for this vehicle as it would smash the engine into the ground, under the mass of a full propellant tank. A good defense is to eliminate that failure scenario entirely, at least for early prototypes. Later generations of this launch vehicle, if no failures upon engine startup are observed, might be vertical.

[82] See definition in the Introduction section. Two-stage-to-orbit approaches such as air launch would be possible as well, but they would be more complex than, and thus harder to render as safe as, single-stage-to-orbit.

[83] More to allay fears and reduce risk - in other words, as part of regulatory feasibility - than from any technical requirement.

[84] In the United States as of May 2025, a spaceplane could not simply take off from an ordinary airport and proceed into space. While such an act may be physically possible (if the airport has a long enough runway and facilities to load the spaceplane's propellant), it would be illegal in the United States: launch to orbit is only allowed from specifically licensed spaceports. It is possible to amend the relevant regulations and legislation to permit this, and there are (as of May 2025) countries where this would be legal, but this report considers regulatory feasibility under the regulations in effect in the United States as of this report's publication.

A Department of Defense licensed NTP spaceplane would presumably be launched from existing military bases, but while this would sidestep the FAA's spaceport licensing requirement, it would still require construction of liquid hydrogen fueling facilities at each launch site, so one could not simply launch from any military airstrip. In most cases a shielded hangar and dedicated runway with cooling systems would be needed at each site too, though operational models have been discussed where a launch vehicle lands at a remote base, reloads its propellant, launches again, and returns to its main base, requiring a hangar only at its main base; this still requires liquid hydrogen to be supplied at the remote base. There has also been discussion of launching to space from an aircraft carrier, but despite the existence of proponents within the Navy, the authors of this report are, as of May 2025, unaware of any unclassified summary of said approach's advantages. In any case, the need for a shielded hangar space and liquid hydrogen production or storage - along with the greater dangers of carrier landings even in calm weather, the increased hazard should an accident occur, and dealing with neutron activations by the carrier's runway - appear to make carrier-based operations impractical.

[85] Including control drums, a gimbaled bell nozzle, and all necessary radiation shielding as part of the engine's mass.

[86] Without an aerospike nozzle, atmospheric specific impulse might be only 850 seconds. This would require enough extra propellant, propellant tank mass to store the extra propellant, et al to substantially increase the required minimum vacuum thrust-to-engine-weight.

As it is desired to launch as often as possible in order to amortize fixed costs, the payload mass target is intended to use some existing standard to maximize the number of payloads at the target, which can be commercially sourced. The primary design selected for this report was sized for a single secondary ESPA class payload, at 180 kg. A secondary design, listed further below, was sized for a single 12U CubeSat payload, at 24 kg[87]. These small payload masses, along with safety issues, preclude any serious consideration of having a human on board the prototype spaceplane during flight, as crew or passenger.

The mission analysis was performed to explore and model the transit of the vehicle from the runway to low Earth orbit, where it deployed a payload, and returned back to the space port to prepare for the next mission. To achieve this, the mission was split into 5 separate stages for analysis: takeoff, atmospheric transit, apogee raise, circularization, and deorbit[88]. These stages included different parameter sets and boundary conditions, including atmospheric effects when applicable. A mission model was then created using Python to calculate propellant consumption, transit times, masses, and other values to determine if the architecture would close for a given configuration and attempt to optimize execution when possible.

Given the design and validation of the vehicle engine may require iterations and further development in future phases, the computational model addresses the performance of a spaceplane which has assumed operational parameters. These parameters will act as the goals for future designs, setting specific points for operational temperature, reactor power, etc.

The parameters used in the calculations are outlined below:

Craft Parameters	**Values**
Dry Mass[89]	2141.5 kg
Payload Mass	180 kg

[87] Sizing for a single 3U CubeSat, at 6 kg, was considered as well. Scaling issues - mainly, square-cube and other effects resulting in a higher vacuum thrust-to-engine-weight being required - appear to make payload reduction to this size less viable, unless substantial improvements to the engine design can be achieved: a launch vehicle with 6 kg payload would take close enough to the same engine as one with 24 kg payload, given the current design, that the rest of the launch vehicle - including the total price per launch: see the Economic Analysis appendix - would be about the same anyway.

[88] Most of the orbital velocity would be shed via aerobraking, after the deorbit burn, but this does not directly impact the rocket's performance. The spaceplane design includes sufficient heat shielding for this aerobraking maneuver. The deorbit burn and subsequent atmospheric reentry would happen after fission products from the earlier burns had decayed, to quote SPD-6, "to a level of radioactivity comparable to that of uranium-235" (Trump, 2020). If a spaceport along the eastern United States was employed, reentry might be pointed at the Atlantic Ocean, so a complete systems failure would result in splashdown some distance from shore rather than crashing into land. In this case, there would be a banking maneuver - only possible if the vehicle is not breaking up from reentry or otherwise in danger of crashing - to turn toward the landing strip after the vehicle had slowed to subsonic velocity.

[89] 1500 kg engine, radiation shielding, and regeneratively cooled bell nozzle; 119 kg propellant tank; 225.5 kg spaceplane skin and heat shield; 25 kg electronics, cold gas valves and pipes for attitude control, cryopumps, and payload dispenser; 270 kg avionics (flight computer, transceiver antenna, power supply, et al), internal bracing, and landing gear; and 2 kg wiring. Total wingspan 7.33 m, length 15.35 m.

Propellant Mass	3,678 kg
Atmospheric Specific Impulse (sec)	950 sec
Vacuum Specific Impulse (sec)	1,000 sec
Drag Coefficient (subsonic, transonic, supersonic)	0.02, 0.05, 0.025
Lift Coefficient	1
Lift Area	28 m^2
Drag Area	8.6 m^2
Mass Flow Rate[90]	6 kg/s
Mission Parameters	**Values**
Desired Aerodynamic Altitude	50 km
Payload Altitude (km)	400 km
Re-entry Altitude (km)	200 km
Takeoff velocity (m/s)	60 m/s

Mission Outline

1. The craft accelerates on the ground from zero to takeoff velocity, at full capacity. This provides the craft enough lift to takeoff vertically, similar to an aircraft. To account for the friction and drag the craft might experience during this acceleration, the following equation is assumed:

   Friction + Drag = ¼ Thrust

2. The craft travels through the atmosphere and increases altitude until it reaches 50 km. The craft then accelerates horizontally until it reaches orbital speed. Accelerating to orbital speed in the upper atmosphere means less drag on the craft.
3. Once at orbital speed, the craft performs a Hohmann transfer to increase its altitude to the desired payload altitude.
4. Once the payload is deployed (and fission products in the engine decayed to necessary levels), the craft performs another Hohmann transfer to lower its altitude to the re-entry altitude.

After multiple iterations, the mission sequence was reanalyzed using updated specifications, resulting in the following outcomes:

[90] Based on 4:1 thrust(vacuum) to engine weight ratio. Engine weight = 1500 kg * 9.81 m/s^2 = 14,715 N. Vacuum thrust = 4 * 14,715 N = 58,860 N. Mass flow rate = thrust / (Isp * G_0) = 58,860 / (1,000 * 9.81) = 6 kg/s.

Results

Total Initial Craft Mass	6,000 kg
Max Atmospheric Thrust	55,917.00 N
Max Vacuum Thrust	58,860.00 N
Initial Launch Angle	38 degrees
Total Propellant Burned	3,574.27 kg
Leftover Propellant	103.73 kg

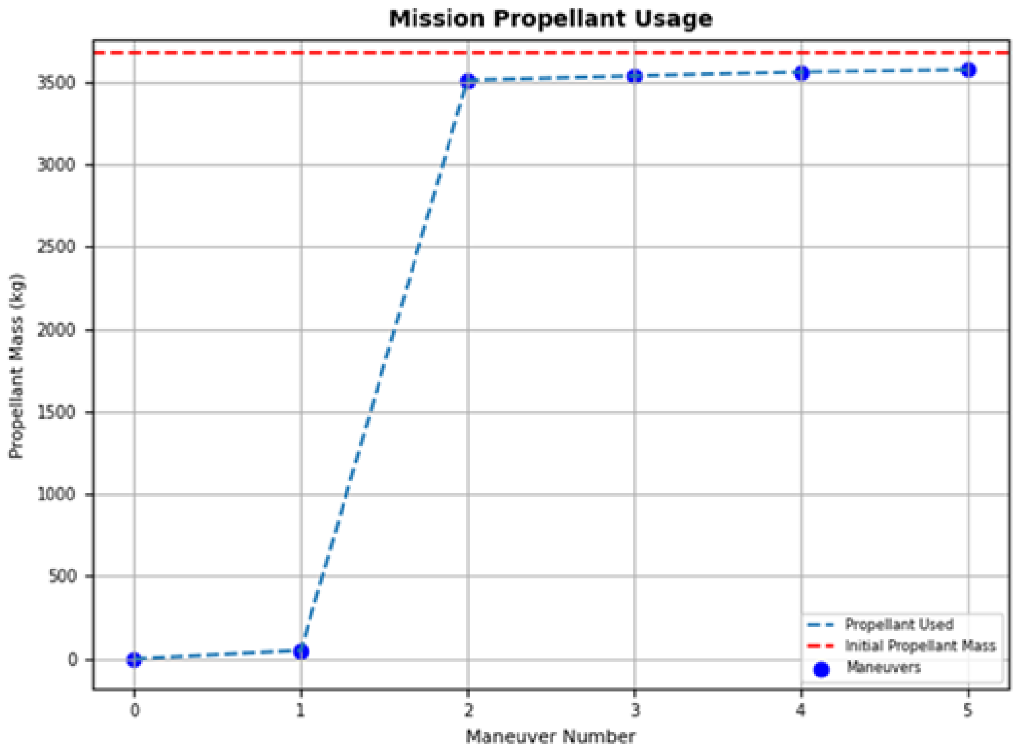

Figure 30. Propellant mass expended for each maneuver

Results for Individual Maneuvers

1. Takeoff

Takeoff Velocity	60.00 m/s
Takeoff Acceleration	6.99 m/s^2
Propellant Burned	51.50 kg
Time Taken	8.58 sec

2. Atmospheric Maneuver

Final Altitude	50.23 km
Final Velocity	7.88 km/s
Propellant Burned	3458.40 kg
Time Taken	576.40 sec (~9.6 min)

3. Hohmann Transfer – Burn 1

dv-1	101.39 m/s
dv-1 Propellant Burned	25.60 kg

4. Hohmann Transfer – Burn 2

dv-2	102.23 m/s
dv-2 Propellant Burned	25.55 kg
Final Altitude	400 km
Total Transfer Time	44.50 min

5. Deorbit

Final Altitude	200 km
Deorbit dv	57.63 m/s
Propellant Burned	13.23 kg
Deorbit Time	45.26 min

Thermal Power Calculation

The thermal power required to be generated by the engine for a successful mission can be determined by calculating the energy needed to heat the propellant, liquid hydrogen. Liquid hydrogen, stored at approximately 20K in the propellant tanks, must be heated to around 3000K to achieve the required specific impulse of 950s in the atmosphere[91]. This process involves two stages:

1. Phase Change: Hydrogen transitions from liquid to gas at 20K, which requires energy determined using the latent heat of vaporization.

Latent Heat of Vaporization of Hydrogen = 446 kJ/kg

2. Temperature Increase: Once in the gaseous state, hydrogen must be further heated from 20K to 3000K. The energy required for this depends on the specific heat of hydrogen gas, which

[91] Atmospheric specific impulse would only be 850s, but this report assumes an aerospike nozzle to reduce 15% loss to 5%.

varies with temperature. To account for this variation, the temperature range (20K–3000K) is divided into several segments, with an average specific heat value used for each segment, as shown:

- *20K – 1000K, Average specific heat = 14.05 kJ/kgK → 13769 kJ/kg*
- *1000K – 1500K, Average specific heat = 15.50 kJ/kgK → 7750 kJ/kg*
- *1500K – 2000K, Average specific heat = 16.60 kJ/kgK → 8300 kJ/kg*
- *2000K – 2500K, Average specific heat = 17.4 kJ/kgK → 8700 kJ/kg*
- *2500K – 3000K, Average specific heat = 18.01 kJ/kgK → 9008 kJ/kg*

This gives the total specific energy as 47,524 kJ/kg.

Using the mass flow rate, the required engine output can be found as follows:

$$\text{Engine Power Output} = 47{,}524\ \text{kJ/kg} * 6\ \text{kg/s} = 285\ \text{MW}$$

This would be slightly higher than the design proposed earlier in this report, but the design can be scaled up to this level. This stays under the regulatory limit of 1,000 MW thermal for a small modular reactor[92].

Additional Plots for the Atmospheric Maneuver

The following plots provide additional information on the horizontal and vertical motion and forces on the craft with respect to the altitude:

[92] See 10 CFR § 50.2 ("General Provisions...," 2025). Although the Department of Energy recognizes a smaller classification for microreactors, the relevant portions of 10 CFR do not make this distinction.

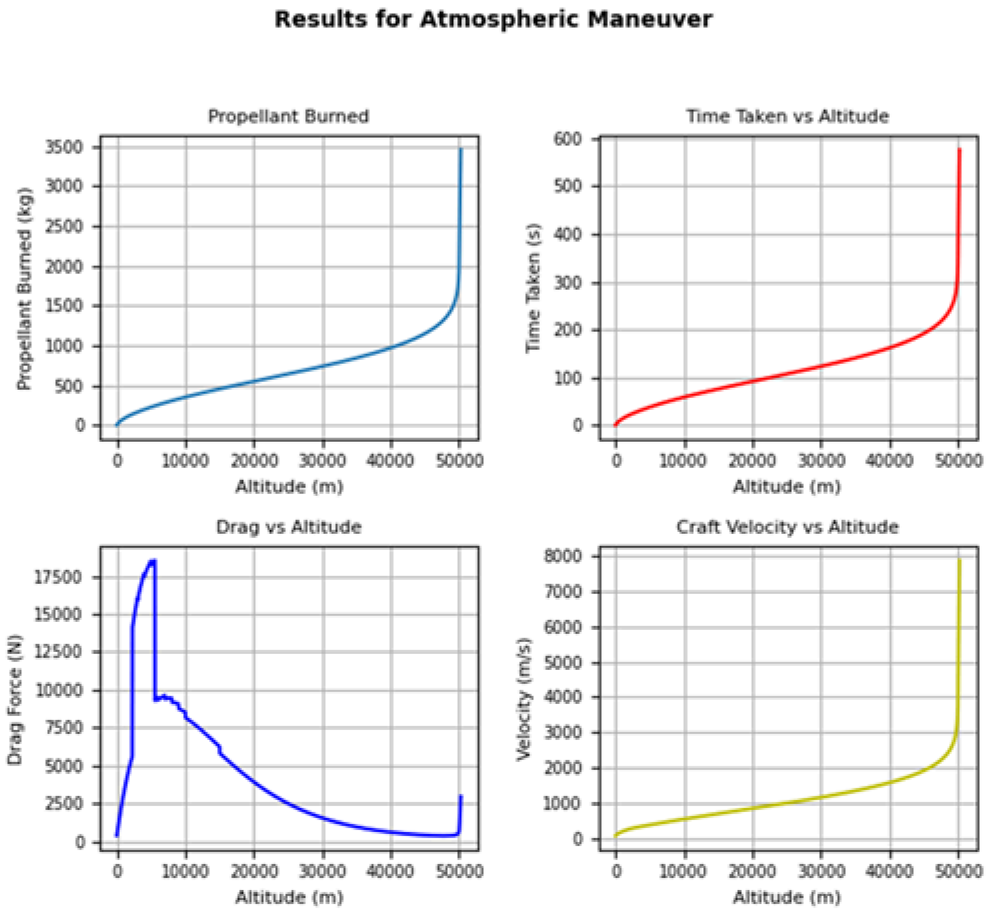

Figure 31. Results for atmospheric maneuver

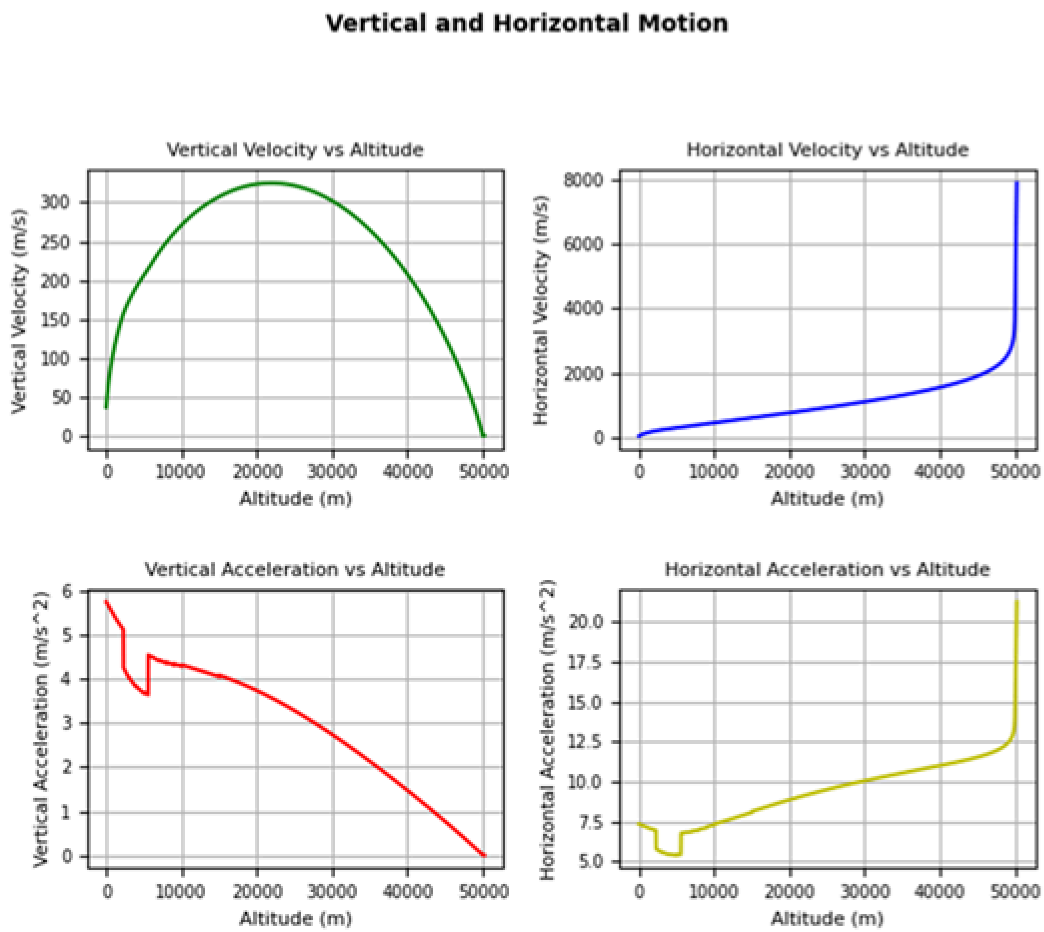

Figure 32. Vertical and horizontal motion

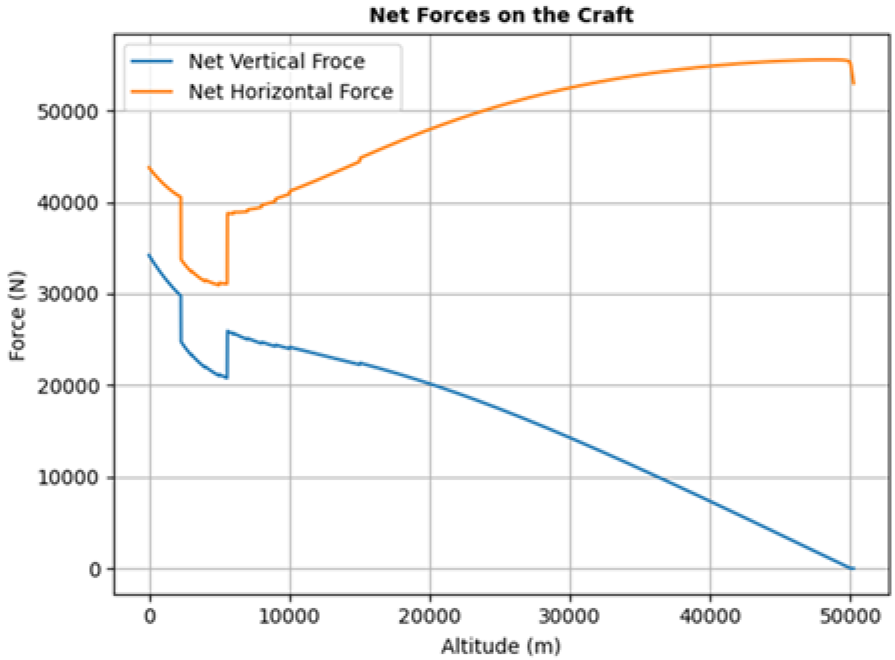

Figure 33. Net forces on the craft

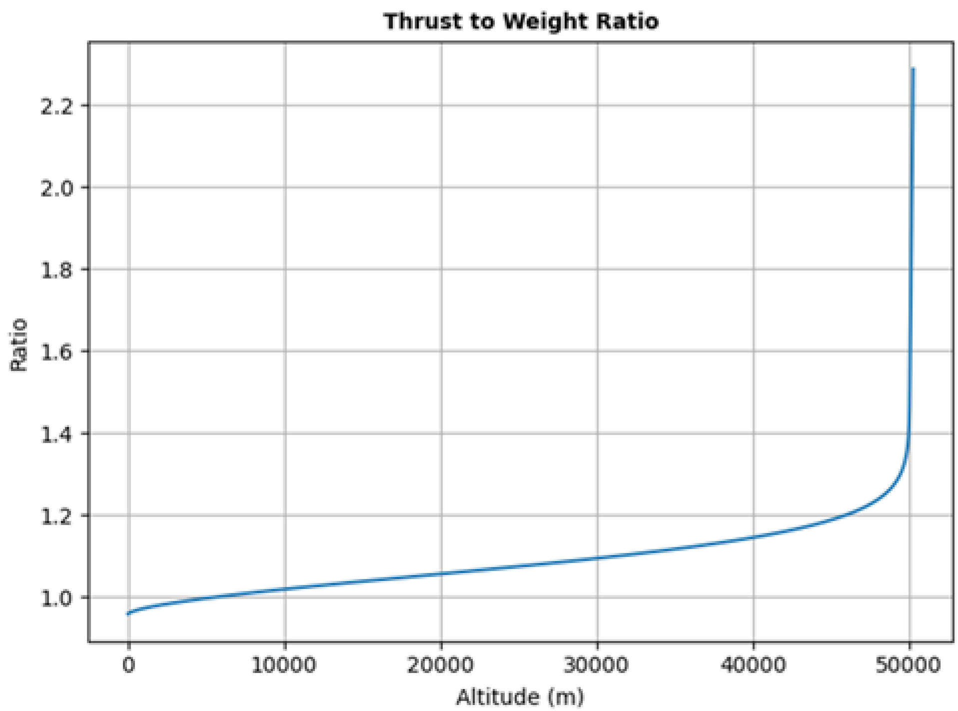

Figure 34. Thrust to weight ratio

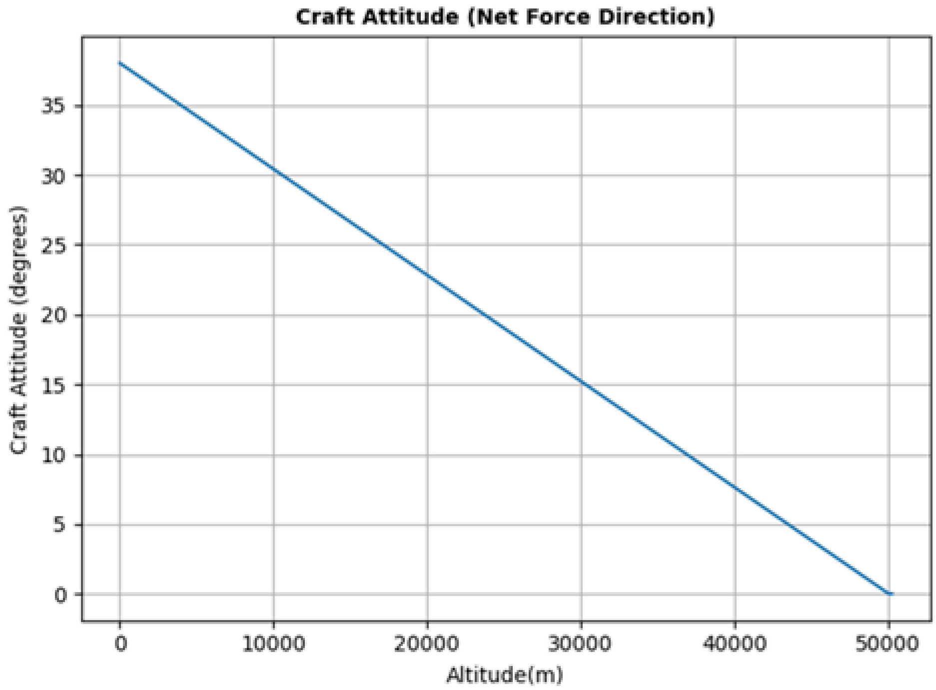

Figure 35. Craft altitude (net force direction)

Smaller Payload Option

Although an ESPA class payload was selected for primary analysis, this study also examines whether a smaller payload - and thus a smaller spaceplane, with lower development costs, first-of-a-kind risk analysis, et al - was viable. There do not appear to be broadly commercially supported payload standards with masses between the secondary ESPA class and the largest of the CubeSat standards. As of May 2025, the largest class of CubeSats in the CubeSat Design Specification[93] is 12U, at 24 kg (Johnstone, 2022). Assuming a 24 kg payload, simulations resulted in this design:

Craft Parameters	**Values**
Dry Mass[94]	771 kg
Payload Mass	24 kg
Propellant Mass	1,205 kg
Atmospheric Specific Impulse (sec)	950 sec
Vacuum Specific Impulse (sec)	1,000 sec
Drag Coefficient (subsonic, transonic, supersonic)	0.02, 0.05, 0.025
Lift Coefficient	1

[93] Revision 14.1, the current official version as of May 2025.
[94] 500 kg engine, radiation shielding, and regeneratively cooled bell nozzle; 57 kg propellant tank; 97.5 kg spaceplane skin and heat shield; 25 kg electronics, cold gas valves and pipes for attitude control, cryopumps, and payload dispenser; 90 kg avionics (flight computer, transceiver antenna, power supply, et al), internal bracing, and landing gear; and 1.5 kg wiring. Total wingspan 3.97 m, length 10.85 m.

Lift Area	9 m^2
Drag Area	3.72 m^2
Mass Flow Rate[95]	2 kg/s
Mission Parameters	**Values**
Desired Aerodynamic Altitude	50 km
Payload Altitude (km)	400 km
Re-entry Altitude (km)	200 km
Takeoff velocity (m/s)	60 m/s

This yields the following results:

Total Initial Craft Mass	2,000 kg
Max Atmospheric Thrust	16,775.10 N
Max Vacuum Thrust	17,658.00 N
Initial Launch Angle	33 degrees
Total Propellant Burned	1,203.73 kg
Leftover Propellant	1.27 kg

It is evident from the fact that less than 2 kg of propellant remains[96] that scaling down much further with the same assumptions is unlikely to yield a viable result. Indeed, attempting to optimize for a 6U (12 kg) total payload did not yield a viable configuration at substantially smaller total masses.

Given the same total specific energy for 20K to 3,000K, and only changing the mass flow rate, comes to 95 MW thermal.

3.5 Thrust-To-Weight Ratio Options

So far, this section has considered optimizing the engines to reach a thrust-to-weight ratio of 4. In case that proves excessively difficult, this study presents two options for the payload sizes considered above with a thrust-to-weight ratio of only 3.5, approximately what MITEE-C delivers.

The takeoff velocity had to increase to 90 m/s. More significantly, ending with a positive payload amount proved difficult until the mass flow rate - and thus, engine power - was substantially increased.

First, the 180 kg payload option:

[95] Assuming thrust scales linearly with engine mass. As seen earlier, mass flow rate scales linearly with thrust. This example has an engine mass of one third of the main example, so the thrust and mass flow rate are also one third.
[96] The “remaining” propellant is used for attitude adjustment and reserves. A reserve that is around 0.1% of the total is likely insufficient in practice, but this paper addresses whether these missions could be done at all, and this portion of the analysis is intended to illustrate an extreme case. Achieving better engine performance would need to be achieved in practice to enable this case.

Craft Parameters	**Values**
Dry Mass[97]	3,965 kg
Payload Mass	180 kg
Propellant Mass	6,355 kg
Atmospheric Specific Impulse (sec)	950 sec
Vacuum Specific Impulse (sec)	1,000 sec
Drag Coefficient (subsonic, transonic, supersonic)	0.02, 0.05, 0.025
Lift Coefficient	1
Lift Area	29 m^2
Drag Area	12.03 m^2
Mass Flow Rate	21 kg/s
Mission Parameters	**Values**
Desired Aerodynamic Altitude	50 km
Payload Altitude (km)	400 km
Re-entry Altitude (km)	200 km
Takeoff velocity (m/s)	90 m/s

This yields the following results:

Total Initial Craft Mass	10,500 kg
Max Atmospheric Thrust	195,709.50 N
Max Vacuum Thrust	206,010.00 N
Initial Launch Angle	52 degrees
Total Propellant Burned	6,203.71 kg
Leftover Propellant	149.29 kg

Given the same total specific energy for 20K to 3,000K, and only changing the mass flow rate, comes to 998 MW thermal. While an even higher mass flow rate would yield more optimal results, this is just below the 1,000 MW thermal threshold for the definition of a small modular reactor, so this analysis ended there.

Second, the 24 kg payload option:

Craft Parameters	**Values**
Dry Mass[98]	1,399 kg
Payload Mass	24 kg
Propellant Mass	2,077 kg

[97] 3,000 kg engine, radiation shielding, and regeneratively cooled bell nozzle; 170 kg propellant tank; 295 kg spaceplane skin and heat shield; 25 kg electronics, cold gas valves and pipes for attitude control, cryopumps, and payload dispenser; 473 kg avionics (flight computer, transceiver antenna, power supply, et al), internal bracing, and landing gear; and 2 kg wiring. Total wingspan 7.42 m, length 18.05 m.

[98] 1,000 kg engine, radiation shielding, and regeneratively cooled bell nozzle; 84.9 kg propellant tank; 129.85 kg spaceplane skin and heat shield; 25 kg electronics, cold gas valves and pipes for attitude control, cryopumps, and payload dispenser; 157.5 kg avionics (flight computer, transceiver antenna, power supply, et al), internal bracing, and landing gear; and 1.75 kg wiring. Total wingspan 3.61 m, length 13.95 m.

Atmospheric Specific Impulse (sec)	950 sec
Vacuum Specific Impulse (sec)	1,000 sec
Drag Coefficient (subsonic, transonic, supersonic)	0.02, 0.05, 0.025
Lift Coefficient	1
Lift Area	8 m^2
Drag Area	4.71 m^2
Mass Flow Rate	12 kg/s
Mission Parameters	**Values**
Desired Aerodynamic Altitude	50 km
Payload Altitude (km)	400 km
Re-entry Altitude (km)	200 km
Takeoff velocity (m/s)	90 m/s

This yields the following results:

Total Initial Craft Mass	3,500 kg
Max Atmospheric Thrust	111,834.00 N
Max Vacuum Thrust	117,720.00 N
Initial Launch Angle	78 degrees
Total Propellant Burned	2,065.37 kg
Leftover Propellant	11.63 kg

Given the same total specific energy for 20K to 3,000K, and only changing the mass flow rate, this comes to 570 MW thermal.

These results suggest an alternative pathway to implementation, in case getting the engine's thrust-to-weight ratio to 4 turns out to not be viable.

Conclusion

This study has demonstrated, as best it can without actually starting preliminary hardware fabrication, a nuclear thermal launch vehicle is feasible from regulatory and technical points of view. The FAA and NRC will require much more thorough proof of safety, based in part on hardware testing, to license such a thing. Engine thrust-to-weight performance will need to be improved. But these things are possible. The logical next step is to focus on the technical side: in the plan below, NRC licensing is not needed until attempting to achieve Technology Readiness Level (TRL) 6 and FAA licensing is not needed until attempting to achieve TRL 8[99], although some further regulatory investigation will likely occur in the meantime.

As of this report, the technology is at low TRL[100] 3: more simulation and refinement needs to be done, but there is a conceptual design and physics-based simulations demonstrating analytical proof-of-concept. The next development phase would involve refining the design, then testing a mockup of the system at low temperatures using chemical or electric heaters instead of nuclear fuel to reach TRL 4. Testing this mockup under full temperature (3000K), vacuum, and vibration would advance to TRL 5. Replacing the heaters with nuclear fuel and repeating the tests, including verifying the control drums' reliable control over k-effective under expected operating conditions, would achieve TRL 6. Integration with a launch vehicle[101] and complete ground testing would reach TRL 7, followed by a demonstration orbital flight for TRL 8, ultimately leading to commercial deployment and TRL 9. Once commercial operation of the prototype demonstrates safety and reliability, work could then begin on scaled-up versions, most likely via license to existing developers of large launch vehicles.

The proposed steps to achieve TRL 6 and above will require the construction of a hot cell to perform testing at[102]. Fortunately, an environmental impact statement and certain other licensing-relevant work for a similar project has already been prepared (Haslett, 1995, p. 4.2 - 4.11), which should reduce the amount of effort required to license it. One notional design

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[99] In both cases, much of the work in the immediately preceding TRL consists of gathering data to support the licensing effort.

[100] The TRL definitions used here are a blend of the Department of Energy's and NASA's. For example, the DOE's TRL 8 requires first-of-a-kind commercial demonstration (U.S. DOE, 2009), while NASA's TRL 8 emphasizes flight qualification (NASA, 2017). In this case, the first demonstration of commercial capability - actually getting to LEO for the first time - happens to satisfy both definitions simultaneously.

[101] Which would be designed and constructed in a parallel effort.

[102] The normal procedure for testing a first-of-its-kind reactor would be to test at the national labs. However, conversations with Oak Ridge, Idaho, and Sandia National Labs revealed this is apparently not possible for the specific case of this project: a national lab might advise, but the actual testing must happen elsewhere. Details vary, but the main reasons are that they are not set up to capture or deal with exhaust (non-radioactive or otherwise) and that this reactor is too small for their facilities. Given this limitation, it makes sense to construct a hot cell at the location where, if all goes right, the first prototype spaceplane will fly from. This location will need to be able to also be licensed as a spaceport, which - along with launch vehicle licensing, to permit initial test flights - will likely happen in parallel to the efforts to reach TRL 6 then 7 before finishing as part of the TRL 8 effort.

would involve one or more concrete domes[103], usually sealed but open to admit a spaceplane. When sealed, the domes could only be accessed via airlocks and would operate like clean rooms[104], with the ability to be purged of air to test vacuum performance or filled with nitrogen to safely test exhaust against atmospheric pressure[105]. Ideally, operations inside these domes would be primarily or entirely robotic, controlled from outside the domes[106].

As part of the effort to reach TRLs 4 and 5, the liner will need to be tested and possibly redesigned to avoid hydrogen embrittlement, as well as to avoid cracking from thermal expansion of the nuclear fuel, and thickened to the desired design lifetime of the nuclear fuel[107].

Among the upgrades future work may attempt, so as to improve engine thrust-to-weight, depletion or other factors that limit the number of flights between changing the nuclear fuel, and other metrics; involve the use of better materials[108]; adding helium coolant or heat pipe channels to remove decay heat without venting propellant[109]; routing the hydrogen past the nuclear fuel multiple times[110] before entering through the frits; redesigning the frits to increase heat transfer so as to increase exhaust velocity, running at higher temperatures[111]; and optimizing the core

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[103] Likely 1 to 2 meters thick (plus any voids), depending on the exact emissions of the engine as built. To take an extreme disaster scenario: if all the cladding evaporated and the HALEU wound up collected at one point on the floor, and some member of the public leaned against the nearest spot on the outside of the wall, the concrete would be thick enough said member of the public would not receive more than 1 mSv in several hours (well before which time, said member of the public would presumably be removed by security).
[104] Specifically including cleanroom-grade HVAC to trap any radionuclide particles escaping the engine.
[105] The engine will also have a system to purge atmospheric oxygen prior to exhausting hot hydrogen, but the engine may be tested before this system is installed, and said system will need to be tested before being relied upon. Filling the room with nitrogen will enable said tests to be conducted safely.
[106] With no straight-line voids through the concrete for the power and data lines, not to mention the airlocks.
[107] 1.4 mm to last 4 years, as noted in the Introduction section, or whatever other thickness is dictated by further modeling of the nuclear fuel's lifespan.
[108] As noted in the Introduction section, there exist materials that could not be cited in this report. In most cases, this was due to the public nature of this report or because working out the licensing would have taken longer than this study's limited duration. These materials would be available for use in a prototype engine.
[109] Specifically, setting up the aerobraking heat shield as a "heat sponge", absorbing heat during aerobraking then passively radiating it away, can then also absorb decay heat from the reactor and passively radiate it away prior to aerobraking.

The firm Advanced Cooling Technologies has proposed an alternative, which would feature a thinner heat shield that did not need to absorb the full heat of aerobraking and use heat transfer pipes (similar to but more mass-efficient than helium coolant channels) to rapidly transfer the heat - whether from aerobraking or decay heat - to a central bank, from which it would be slowly released to be radiated away. Research into this alternative (not part of work funded by the DOE) is still ongoing as of this report's publication, with the aim of reducing overall mass and enabling more rapid removal of decay heat.
[110] See Nikitaev (2021, p. 10) for a similar design, or possibly just coiling around the bottom of a pin cell before entering through the frits just below the beryllium layer.
[111] 3,000K was chosen as an arbitrary target. Running at 3,600K - which would involve making sure that nothing other than the nuclear fuel, the tungsten liner, and the hydrogen reach that temperature, with cooling systems intercepting the heat before it can melt any other component - would boost vacuum specific impulse to 1,200 seconds, with a corresponding increase in atmospheric specific impulse. One analysis using the 12U launcher design from the Smaller Payload Option section, increasing vacuum specific impulse to 1,200 seconds and atmospheric specific impulse to 1,140 seconds, was able to achieve orbit with 1.36 kg of propellant left, starting

geometry to maximize heat per unit mass. As of May 2025, there is an ongoing effort[112] to evaluate genetic algorithms, and other artificial intelligence models trained on the many published nuclear thermal propulsion designs, for their ability to generate better configurations while satisfying constraints on k-effective, power levels, and thrust-to-weight ratio. Given the well known tendency of artificial intelligences to hallucinate physically impossible designs, these would be treated as suggestions, which must pass functionality and safety reviews before potentially being seriously considered.

with 1,065 kg of hydrogen and 652 kg of engine (with extra mass from both the lower amount of propellant and the smaller fuel tank), which only requires achieving a 3.07 thrust-to-engine-weight ratio. This ignores potential improvements to thrust, both from the temperature and from increased dissociation of hydrogen at that temperature. Achieving this higher temperature would entail running at higher power - for instance, the 285 MW configuration in the Spaceplane section would increase to 350 MW - and thus require more massive radiation shielding, but preliminary estimates suggest the propellant savings would more than outweigh this.

[112] Not part of work funded by the Department of Energy.

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Appendix A: Economic Analysis

More so than in the rest of this report, everything in this appendix is a rough estimate of what might eventually be achievable, not a specific prediction of exact values.

As detailed in the Spaceplane section, a spaceplane might be made consuming about 3,600 kg of hydrogen to deliver 180 kg to 400 km LEO. Another variation may be possible which consumes about 1,200 kg of hydrogen to deliver 24 kg to 400 km LEO.

As of 2019, the cost of hydrogen was about \$10/kg (Elgowainy et al., 2019, p. 2)[113] assuming bulk production of hydrogen at the spaceport this spaceplane would be launching from. Jet fuel accounts for 20-40% of commercial airliner costs (Mitchell & Simple Flying Staff, 2023), so let us assume 20% is achievable[114] for a spaceplane program. This ignores minimum fixed range costs, nuclear servicing[115], and other such factors[116], but also ignores post-2019 decreases in the

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[113] Ignoring the delivery portion, since this would be on-site production.

[114] Including overhead, profit, et al. This assumes airline-like operations involving no more than a dozen people directly in each launch between the launch range and launch operator - radar & traffic controller, propellant loader, payload handler/customer interface, trajectory programmer, safety officer/overseer, and so on - with the potential for some tasks, such as propellant loading, to be automated. The annual launch volume is high enough that roles such as weekly or monthly inspections, which happen at the same frequency and consume the same hours regardless of the number of launches, are not included in this dozen, but instead part of the fixed costs.

Given the high launch cadence, it would be best to tailor any program-specific government-imposed requirements to be on an annual or other such basis rather than per-launch, to the greatest practical extent. For example, consider the matter of inspections and accident preparedness that the NRC may desire. It has been established that the FAA has jurisdiction from the moment launch operations begin to the moment they end, which includes any pre-launch and post-launch inspections, so the NRC may not specifically require an inspection before, after, or as part of each launch. However, the FAA requires a safety officer to be part of the launch crew. It may make sense to allow a NRC-employed inspector to be dual-hatted as said safety officer, especially considering the likely overlap in required skill set, assuming the NRC was okay (such as having no conflict of interest concerns) with said person being under the launch company's command during launch operations, though this would only permit that one NRC observer to be around during the launch. Hazmat and other emergency response personnel requirements may seem like ideal candidates for further such dual-hatting, but in practice these roles would likely be fulfilled from dedicated local sources instead.

[115] HALEU costs about \$25,000 per kg (White & Cothron, 2023). 160 kg, as might be used by the smaller spaceplane's engine, would thus be \$4,000,000. Replacing the tungsten liner - around 10% of the HALEU's mass - along with the HALEU is estimated to cost a small fraction of this amount, such that it might be included in the HALEU's price. Assuming 4 years between replacement (accounting for a much lighter duty cycle than in a power plant), the program would need to generate \$1,000,000/year to cover this. Roughly triple this cost can be assumed for the larger spaceplane's engine. This would be one of the larger fixed costs, but charging \$50/kg of hydrogen used and flying 100 times per year generates annual budgets large enough to cover this. This is one of the key reasons why a high flight rate (and thus, flying standardized payload sizes, so there is substantial demand for flying a payload of that exact mass) is required for commercial viability: the annual budgets generated by only 1, or even 10, flights per year would struggle to afford replacement HALEU.

[116] If one ignores that replacement HALEU is only paid for once every 4 years, to make it seem like the largest line item next to true annual costs, the second largest might be replacing the runway asphalt every 4 years, as proposed elsewhere in this study. Cost estimates for asphalt for runways vary, but \$3-5/square foot (Palmetto Asphalt Service, 2025) is around the midpoint of the prices found; this analysis takes the \$4/square foot midpoint of that range. A 5,000 foot long runway, 75 feet wide, comes to 375,000 square feet and \$1,500,000 to replace. Increasing this to \$2,000,000 to account for disposal and other special handling means the program would have to set aside another

cost of hydrogen. Thus, the cost to reach LEO using this architecture might be $50 per kg of hydrogen consumed[117].

For the larger spaceplane, this comes to $180,000 per flight, or $1,000/kg; at 100 flights per year, this would generate an annual income of $18,000,000. For the smaller spaceplane, this is $60,000 per flight, or $2,500/kg, with a 100-flight-per-year annual income of $6,000,000. These numbers are competitive with the best price current launch options on a $/kg basis. However, $/kg is the wrong way to evaluate these spaceplanes, for two main reasons.

1. Less hydrogen is used per kg of payload at larger scales, which translates directly to better $/kg. Of these two examples, the smaller uses 50 kg of hydrogen per kg of payload, while the larger uses only 20 kg of hydrogen per kg of payload. So, if $/kg was the only figure of merit, one should scale the launch vehicle to a roughly equal payload to another launch vehicle being compared to. Extending the above simulations to 100,000 kg payload found one solution using 594,000 kg of hydrogen (thus, 5.94 kg of hydrogen per kg of payload), for $29,700,000 per flight[118] and $297/kg. At that scale, further efficiencies can further reduce $/kg[119], most likely under $100/kg.
2. There exists a large class of customers who care less about the $/kg and more about the size of the check someone needs to write, and other hurdles, to make the launch happen. Being able to buy a launch where your payload fills the rocket's capacity so no other customers for the ride, and thus no fear some other customer will cause the launch to slip or kick your payload off, can be seen as a premium feature CubeCab heard consistent commercial demand for since its founding in 2014. Few launch vehicles attempt to provide this dedicated[120] service for secondary ESPA and smaller satellites, so these customers have had to settle for rideshare service as their only option to get to orbit.

Likewise, the annual budgets are too small to interest large aerospace companies such as Lockheed-Martin or SpaceX. They are not too small for small businesses such as CubeCab - and once these small spaceplanes prove viable, licenses can be negotiated to allow large aerospace companies to develop and operate much larger versions, and thus achieve low $/kg.

$500,000 per year to cover this. Unlike replacement HALEU, this cost could eventually be shared among multiple launch vehicles using the same runway, although at first there would only be a single prototype launch vehicle.

[117] Other sources claim that $5/kg or $2.5/kg for hydrogen is more reasonable. In these cases, assume that fuel is 10% or 5% of the cost, respectively, to arrive at the $50/kg of hydrogen total cost.

[118] At this size, obtaining customers for 100 flights per year would be difficult at first, but a 10-flight-per-year annual income of $297,000,000 can likely cover the fixed costs. That said, if this option became available, industrial uses such as solar power satellites might then be able to serve as "anchor customers", providing guaranteed payloads for most of the flights, effectively subsidizing other flights that could be offered to other customers at the same low cost.

[119] For example, using vertical launch so as to do away with wings and landing gear. By the time a launch vehicle of that scale is constructed, the prototypes should have proven the safety and reliability of this engine architecture, making vertical launch more viable. Also, at that scale the per-kg cost of hydrogen will decrease.

[120] There exist launch operators who mount several CubeSats on a launch vehicle and call it a "dedicated CubeSat launch" because that launch only serves CubeSats, but that is not this report's meaning of "dedicated". This report strictly means one payload for the entire launch, and classifies two or more payloads on the same launch as a "rideshare" launch.