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Technical Track Chairs: Bhavya Lal & Susannah Howiesan

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MARS MISSION ARCHITECTURE DECISION THROUGH VALUE MODELS

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Crewed missions to Mars often focus on how long the crew stays on Mars and which propulsion system powers the mission. This analysis examines expected Net Present Value of Opposition and Conjunction architectures using both nuclear and chemical propulsion. It was found that the number of launches required is indeed a strong contributor to the difference in cost between using chemical and nuclear propulsion before considering development costs. The Conjunction architecture was found to have greater value than the Opposition architecture, but the difference in value between applying chemical and nuclear propulsion was greatest for the Opposition architecture. This difference in architecture value is due to a long stay on Mars generating more benefit than a short stay and this benefit is much greater than that of a returned sample or the time in space traveling between Earth and Mars.

I. Introduction

Crewed missions to Mars have been considered repeatedly since the Apollo program. Called Design Reference Missions and Design Reference Architectures (DRA), two broadly applicable mission categories have been identified as well as conceptual designs of vehicles for implementing them. Conjunction architectures require a long stay on Mars but allow for efficient transfers between Earth and Mars. Opposition class missions have shorter stays and total mission duration, but have transfer trajectories that require larger spacecraft and more crew time in-space than Conjunction architectures. Spacecraft design to achieve either of these varies greatly depending on choice of propulsion system.[1]

Two propulsion options under consideration are Advanced Chemical Rockets (ACR) and Nuclear Thermal Propulsion (NTP). While NTP has a significant specific impulse (Isp) advantage over ACR which can be leveraged to reduce required spacecraft propellant load we do not yet have NTP flight engines. In addition to requiring a development program the NTP engines may have higher unit cost than ACR engines. In light of this we require a means of determining at which point the cost of developing and deploying NTP exceeds the cost of conducting Mars missions with ACR. We also require means of comparing the value of the missions' architectures.

I.A. Value Models and the Value of Time

Value models are a means of comparing alternatives and deciding among them. By Expected Value Utility Theorem: "given a pair of alternatives, each with a range of possible outcomes and associated probabilities of occurrence… the preferred choice is the alternative that has the highest expected utility."[2] As development cost of space systems is a concern a convenient means of comparing values is by monetary equivalents and cash flows as defined in equation 1.

$$Net_N = \sum Benefits_N - \sum Costs_N \quad (1)$$

When cash flows are evaluated as in equation 1 this is the "Benefit-Cost" perspective. It requires a means of knowing or estimating the benefits of a project, but such benefits can be difficult to quantify for public sector projects such as spaceflight. This makes a "Cost-Effectiveness" approach tempting, one in which costs are considered without benefits and this is applicable when the potential benefits of alternatives are identical. However, for space flight upfront costs such as spacecraft and launch vehicles are large and missions can last for years so this encourages spending as little money as possible up front. This perspective also ignores the potential benefits to be realized which are more valuable if they are realized sooner due to the value of time. This has been demonstrated in the context of a robotic mission to Europa[3] and similar methods will be applied to a Mars mission to draw conclusions on long duration crewed space explorations missions.

I.B. Methodology

Core methodology from this past work also applies in the context of crewed Mars missions, but notable differences include

- In-space vehicles unique to mission architecture and choice of propulsion system
- Costs common to all architectures such as the transit habitat and associated launch are not considered
- Only using the Space Launch System (SLS, formerly Ares V at time of DRA 5.0) to place mission elements in orbit
- Two categories of mission time value
    - In-space (outbound and return)
    - On Mars (stay)
- Value of Mars samples returned to Earth

Net Present Value (NPV) is applied with estimates of the costs and benefits to ensure values of all mission options are conducted in the same fiscal year value of money and account for change of value of money over time assuming 7 % annual discount rate (converted to 0.57 % per 30 day period) for government projects.[4] Additionally, since we cannot guarantee 100% probability of mission success but crewed missions require very high reliability the expected NPV is evaluated assuming 99% reliability. However, the cost of achieving this 99% may be different for missions of different durations.[3] This expected NPV calculation is executed as defined in equation 2 using discount rate i for each cash flow period z and mission reliability R(t) representing probability of successful mission completion.

$$E(NPV) = \left(\sum_{z=0}^{N} \frac{Net_z}{(1+i)^z}\right) R(t) \quad (2)$$

Costs are estimated based on applicable examples from publically available literature. The Mars Transfer Vehicle (MTV) spacecraft designs for NTP and ACR from DRA 5.0 were used for the Conjunction class mission. For the Opposition class missions the larger MTVs required to meet mission ΔV requirements were assembled using MTV elements defined in DRA 5.0:

- NTP
  - 1 Core stage
  - 1 InLine stage
  - 2 Drop Tanks with trusses
- ACR
  - 7 TMI stages
  - 4 MOI stages

The ACR MTV cost is estimated by first assuming the cost of its TMI stage relative to the SLS.[3] This TMI stage is then used to estimate the cost of its engines and propellant tank structure based on literature.[5] These can be applied to make estimates of cost per engine and together with the propellant tank structure cost estimate make other ACR MTV stage cost estimates. The NTP vehicle is assumed to have similar tank structure costs, but each NTP engine is assumed to have production costs that are some multiple more expensive than its ACR engine counterpart. This does not include NTP engine development cost. Value of benefits are estimated from annual costs of historical examples of crewed spaceflight: in-space time from International Space Station (ISS)[6] costs and the value of time on another planet and of the samples returned from Apollo program[7] costs.

As much of the analysis depends on inputs that must be estimated, sensitivity and break-even analysis is conducted to gain additional insight. Sensitivity analysis varies inputs from the values used in the baseline to determine the extent to which inputs contribute to the value model output. Break-even analysis is then applied to determine at which point architecture and propulsion alternatives result in missions of equivalent value if possible. This information contributes to determining the value of different architectures and technologies so that we can gauge at which point one architecture is favorable to another and if NTP adds enough value to the mission to justify its development and unit production costs.

II. Results

II.A. Baseline Results

Baseline results of expected NPV are listed in table 1. Utilizing NTP results in higher value than ACR, but as this does not account for the NTP engine development cost this difference can be interpreted as how much can be spent on NTP development before ACR would provide higher value. Of note is that this difference is greater for Opposition than Conjunction architectures, but Conjunction architectures have much higher expected NPVs.

Table I. Baseline results compilation

	E(NPV) ($ Billion)		
Propulsion	Opposition	Conjunction	Δ(C-O)
ACR	86.6	848	761.4
NTP	98.1	852	753.9
Δ(NTP-ACR)	11.5	4	

To assist in putting these results in perspective, table 2 compiles the baseline spacecraft costs and table 3 the baseline values of samples and time used to generate these results. While the spacecraft are significant investments, the estimated value of Mars samples and time on Mars are large enough that a positive expected NPV can be achieved even for the short stay duration of the Opposition class missions. The longer stay time of the Conjunction class missions provides more time periods to generate value resulting in the very large difference in value between the classes of mission. Also of note is that for Opposition missions the cost of the NTP vehicle increases much less than the cost of the ACR vehicle. This difference is due to the ACR vehicle requiring many more stages to close the Opposition mission compared to the Conjunction mission compared to the required increase in spacecraft elements for the NTP vehicle.

Table II. Baseline spacecraft cost compilation

	Cost of Spacecraft and launch ($B)		
Propulsion	Opposition	Conjunction	Δ(O-C)
ACR	17.1	9.35	7.75
NTP	5.58	5.03	0.55
Δ(ACR-NTP)	11.52	4.32	

II.B Sensitivity Analysis

The model inputs were increased and decreased individually to determine how the model results changed as the variables changed. Tornado plots are provided as a means of visualizing these model sensitivities as they show the range of results generated from increasing and decreasing the model inputs. The lengths of the bars in the tornado plots reflects the degree to which the change in input value influences the result. Table 3 compiles the baseline values for each input and the low and high values they were adjusted to for the senility analysis.

Table III. Model Inputs

	Baseline	Low	High
SLS ($B)	1.00	0.50	2.00
TMI ($B)	1.00	0.50	2.00
NTPE unit ($B)	0.22	0.11	0.43
Space time ($B)	0.29	0.14	0.57
Mars time ($B)	54.08	27.04	108.15
Mars sample ($B)	54.08	27.04	108.15
S	0.4	0.8	0.2
Discount rate %	0.57	0.29	1.15

Figure 1 is a tornado plot for using ACR MTVs to perform Opposition class missions. The two strongest factors of note are the values of the Mars sample and time at Mars. If the discount rate is high value of future benefits such as time in space, on Mars, or the returned Mars samples is reduced. Alternatively, by equation (2), if the discount rate is small the denominator of the expected NPV calculation is close to 1 so it has little effect on the calculation. The Mars Sample's value is less impactful than the value of time at Mars due to the sample's value occurring in a later cash flow.

The value of time in space is a relatively weak factor since it is low compared to the value of time on Mars and the cost of the MTV and SLS. Lowering the cost of the TMI stage (and the rest of the MTV along with it) has a similar effect to lowering the cost of the SLS, but these costs have a more pronounced effect if they are larger than if they are reduced. The cost of the MTV's reliability (which scales with S) potentially allows for some savings on the cost of the MTV if the cost of improving reliability is high, but those savings are not realized if the cost of improving reliability is low. However, the mission's benefits are large compared to the cost of the MTV, reducing the strength of this factor.

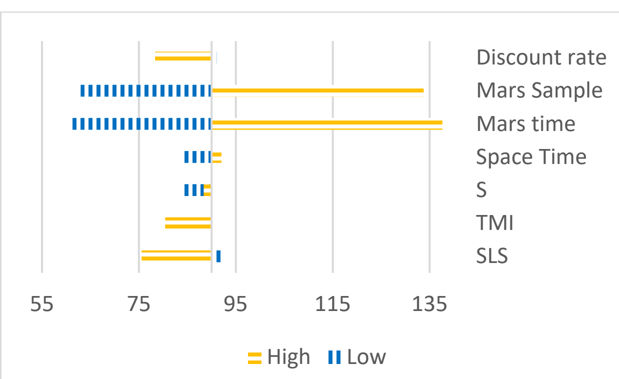

Fig. 1. Tornado Plot for Opposition class ACR in $B

Figure 2 is a tornado plot for using NTP MTVs to perform Opposition class missions. There are many similarities with the ACR option, but notable differences include the impact of the cost of the SLS and the NTP engines. The cost of the SLS is a weaker factor for the NTP option compared to ACR since fewer SLS launches are required. Also, since the element assumed to change value was specifically the engines, costs for the rest of the MTV were not impacted and the NTP engines were not expensive compared to the MTV as a whole.

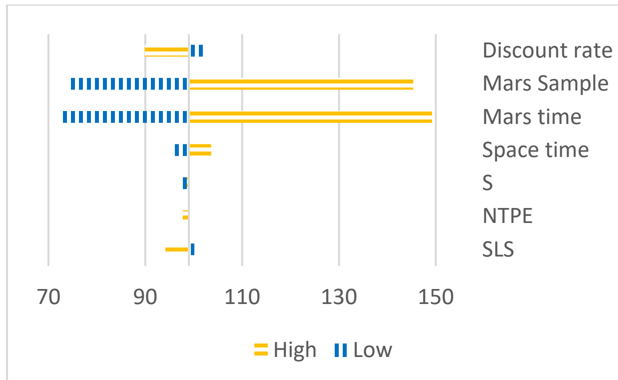

Fig. 2. Tornado plot for Opposition class NTP in $B

Figure 3 is a tornado plot for using ACR MTVs to perform Conjunction class missions. In this case, the mission is long enough and has a long enough stay time at Mars that the value of Mars stay time dominates. This is due to having many cash flow periods to generate value with time at Mars and this also reduces the influence of the value of Mars sample as the sample is returned later than in the Opposition class mission. This value of time at Mars is so great as to appear to make the cost of the MTV and its launches have very little influence on mission value. The same trends also hold for using NTP as seen in figure 4.

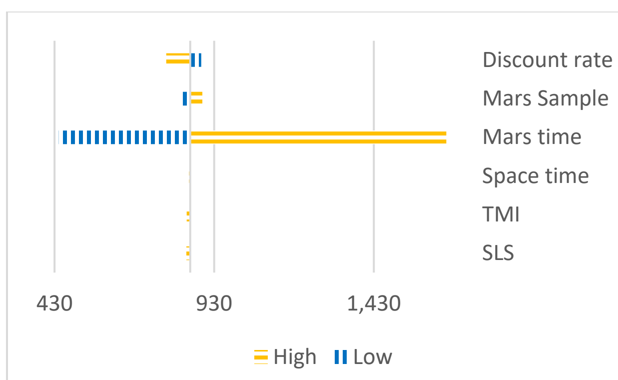

Fig. 3. Tornado plot for Conjunction class ACR in $B

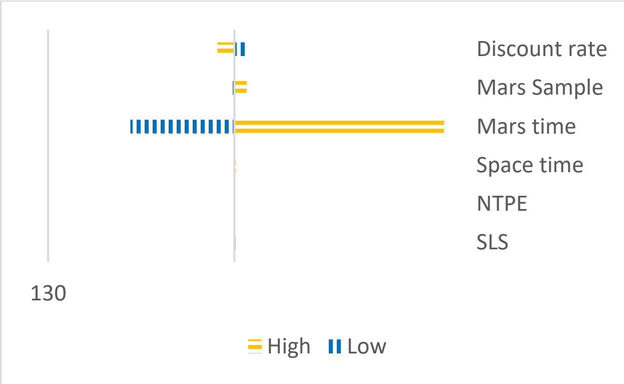

Fig. 4 Tornado plot for Conjunction class NTP in $B

II.C Breakeven Analysis

Where possible, it was determined what value model inputs drove some of Mars mission alternatives to have equivalent value. As factors that differentiate ACR from NTP this was performed for the Cost of the ACR's TMI stage and the NTP engines. It was found that for the Conjunction class mission if the TMI stage could have a dramatic reduction in price to about $6.5 million this would also result in the other stages of the ACR MTV being reduced enough to overcome the additional cost of its additional SLS launches. This could not be done for the Opposition class mission however as it required even more SLS launches. Similarly, to make the NTP option expensive enough to match the value of the ACR option each NTP engine would have severe unit cost increases $1.7 billion for Conjunction class missions to $6.3 billion for Opposition class missions.

As the major drivers differentiating Opposition and Conjunction class missions the values of time in space, on Mars and the value of the Mars samples were also considered for breakeven analysis. If the value of time on Mars was significantly lower than estimated, $261 million per 30 day period, then an NTP powered Opposition class mission would have the same value as a Conjunction class mission, but a similar reduction does not allow for an ACR Opposition class mission to do the same. Attempting to do so actually results in a mission architecture with negative NPV, so it would not be worth conducting. Large increases to the value of time in space ($119 to $120 billion per 30 day period) and the value of the returned Mars samples ($17 trillion) would be required to make Opposition and Conjunction class architectures equivalent.

III. Conclusions

Benefit-cost analysis has been applied to assist in deciding between alternatives on Mars mission architecture and propulsion options. Estimates on costs and benefits based on publically available literature resulted in the benefits derived from staying on and returning from Mars to be much greater than the costs of making the trip regardless of architecture and the longer the stay time the greater the benefits are. For the long stay Conjunction class architecture NTP provides a relatively small increase in value of the mission. The benefits of applying NTP to Mars missions are more pronounced for short stay Opposition class architecture as the cost growth for MTVs that can close the mission is much smaller when using NTP than ACR. However, these benefits may be negated if the cost of NTP engine development exceeds about $4 billion (Conjunction) to $11 billion (Opposition).

The value of time at Mars is the strongest factor in all cases considered. This factor was the only one of note for the longer Conjunction class missions, but the value of the sample is also important to the value of shorter Opposition class missions. However, for the Opposition architecture to have greater value than the Conjunction architecture the value of time on Mars must be much less than the value of the samples returned. Additionally, the MTV and launch costs when using NTP are not a significant factor for Opposition architecture, but these costs are important for ACR since the ACR MTV requires additional stages to be launched.

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Thank you Dr. Collopy for contributing to the past work on the Europa Clipper case study. This work is a continuation of that line of research seeking to determine if those findings are more broadly applicable in spaceflight.

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REACTOR SUBSYSTEM TRADES FOR A NEAR-TERM NUCLEAR THERMAL PROPULSION FLIGHT DEMONSTRATION MISSION

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Solid core nuclear thermal propulsion (NTP) is an enabling in-space propulsion technology capable of high specific impulse (850 - 1100 s) and thrust (10^1 - 10^2 klbf thrust). Because of these benefits, there has been interest by the National Aeronautics and Space Administration (NASA) in a near term flight demonstration mission. Beginning in fall 2019, Analytical Mechanics Associates Inc. (AMA) has been tasked to perform an independent industry study of near-term NTP concepts to enable a near-term in-space flight demonstration (FD). At the heart of this study was the proposal of various reactor subsystem designs by industry participants. Reactor designs spanned a large performance trade space of specific impulse (~750 – 950 s) and thrust (5 – 45 klbf) which allowed for the evaluation of value and risk associated with the NTP FD mission and flight system design. This paper overviews the methodology undertaken order to mature reactor designs and assess the feasibility and value of each designs for a NTP FD mission. Based on these designs, trade studies were performed to assess candidate reactor design attributes and weaknesses with regard to projected subsystem performance, operability, fuel technology, extensibility and programmatic elements (cost, schedule, risk). Industry inputs and trades were informed by and traceable to NASA stakeholder needs. Trade study metrics were recaptured from historic studies performed on NTP and other nuclear space fission power systems and modified to better assess mission specific needs from demonstration of a first-of-a-kind NTP system. Ultimately, the trades performed allowed for recommendations to be given to NASA stakeholders for future consideration when assessing reactor subsystem designs in future mission planning.

I. INTRODUCTION

Nuclear thermal propulsion (NTP) is an in-space propulsion technology capable of high specific impulse (Isp) and in-space thrust (10^0 – 10^2 klbf)[1]. NTP uses the heat generated from nuclear fission to directly heat a propellant and provide thrust. Through the use of a hydrogen propellant, NTP is capable of 850 – 1100 s, nearly double that or more than best performing chemical engines (RS-25, ~450 s)[2]. The combination of high in-space thrust and specific impulse allows for increased flexibility in mission planning including faster trip times, larger abort or launch windows, and payload mass growth allowance[3]. Therefore, NTP is considered by many to be a leading candidate for crewed interplanetary missions, such as Mars and has gathered significant interest from different government agencies, such as NASA, as a near term propulsion option for crewed missions in the mid-2030s or other unique in-space missions requiring high Isp and thrust.

I.A. NTP Flight Demonstration Industry Study Overview

Beginning in fall 2019, AMA Inc. has been tasked by NASA's Game Changing Directorate (GCD) to perform an independent industry study of near-term nuclear thermal propulsion concepts to enable an in-space NTP flight demonstration (FD) mission. The overall systems engineering approach utilized in this study is thoroughly overviewed in a complimentary paper by Collins, et.al.[4]. The goal of this study was to grasp a better understanding of a realistic trade space for near-term NTP FD missions which could be utilized to inform NASA stakeholders of associated technical, performance, and programmatic risks and/or benefits for different technology options for future mission planning. The study consisted of the development and trades of conceptual flight system designs (reactor, engine, spacecraft) and mission concept of operations (Con-Ops) for a FD. Design of the flight system consisted not only of identifying conceptual engineering designs and corresponding performance metrics, but also entailed soliciting information used to perform a programmatic assessment of each design. Programmatic assessment consisted of a survey of high-level cost, schedule, risk/opportunity drivers for each system.

In order to define the trade space as well as mature mission concepts and flight system designs, industry inputs on spacecraft, engine, and reactor designs were gathered from study participants. Each study participant is a company which represents an expertise in either spacecraft, engine, or reactor design. Each participant proposed designs and maturation strategies based on their area of expertise. Industry inputs were governed by study ground rules and assumptions (GR&As), key performance parameters (KPPs) and mission objectives agreed upon by NASA stakeholders. The KPPs and mission objectives applicable to the reactor subsystem are discussed further in[4]. Stakeholder inputs are applicable only to this study and may not be representative of future operational NTP mission needs.

I.A.1. Ground Rules

1. **Nuclear Regulations** – The project will follow the guidance of the Presidential Memorandum[3] and associated U.S. federal regulations (e.g. NRC, DOE, and DOT) for the development, launch, and in-space operations of space fission systems.

2. **Reactor Operations & Disposal Orbit** – The reactor will only operate and be disposed of at an orbit with a perigee greater than 2000 km altitude above Earth.

3. **NTP Fuel Enrichment** – The reactor fuel will use high-assay low-enriched uranium (HALEU, < 20% U^{235}).

I.A.2. Assumptions

1. **Interdependencies from Other Projects** – The FD project may not depend on other projects and hence other sources of funding in order to close on the FD objectives.

2. **Nuclear Facilities Development Schedule** – Any new facilities development for fission ground testing shall not drive the FD schedule (i.e. critical path), and any such facilities will be sub-scale / sub-power level.

I.A.3. Key Performance Parameters

1. Specific Impulse – Operational Mission (> 875 s), Flight Demo (> 700 s, > 900 s preferred)

2. Thrust – Operational Mission (15 – 25 klbf), Flight Demo (5 – 25 klbf)

3. Reactor Mass – Operational Mission (3,500 kg), Flight Demo (< 3,500 kg)

I.A.4. Mission Objectives

1. **Demonstration of NTP Capability**

2. **Demonstration of Regulatory Processes**

3. **Maturation of NTP Technology**

I.B. Reactor Subsystem Role in NTP Flight Demonstration Industry Study

The reactor subsystem design was a primary area of interest in this study since the design and maturation of the reactor subsystem has driven the cost and schedule, as well as overall vehicle performance in previous NASA NTP programs. It was desired that each participant be capable of proposing the design of a near term reactor subsystem capable of demonstrating the technology benefits of NTP for a FD, that could be leveraged with future development to be extensible to higher I_{sp} or thrust systems needed for operational missions, such as to Mars. Unique to this study and proposed mission (compared to traditional NTP development strategies) is that NTP technology is demonstrated in an integrated system for the first time in-space, in addition to demonstrating NTP reactor operation for the first time with HALEU fuel. Through a seven-month effort with inputs collected via four data calls, reactor designs were matured to the fidelity representative of a Pre-Phase A design that allowed for collection of qualitative but representative metrics corresponding to the characteristics of each design. This paper overviews the process undertaken for developing reactor designs, overviews trade study methodology and rationale, and identifies lessons learned to consider for future NTP reactor trades.

II. TRADESPACE DEFINTIONS

II.A. FUEL TRADESPACE DEFINITION

NTP reactor design is inherently tied to fuel design. Fundamentally, reactor operating conditions must ensure no catastrophic fuel failure during operation that could lead to loss of engine performance needed for the mission or loss reactor control. At the highest level, reactor performance, such as specific impulse and total operating time, is limited by fuel properties, such as fuel melting temperature and high temperature lifetime (endurance). Both fuel material (chemical make-up and composition) as well as geometry (physical shape and dimensions of fuel) govern operating limitations (performance) and functions of the fuel within the NTP reactor. Therefore, fuel material and geometric selection impact the operating characteristics and failure mechanisms of the reactor. Based on the potential failure modes, different operating margins may be imposed resulting in an impact to reactor performance. Because of this correlation between fuel design and reactor performance, in the first stage of the study, it was desired to:

1. Identify the trade space of all potential solid core NTP fuel options. This allows for a high-level understanding of their corresponding critical properties (fuel melting temperature and endurance) which can bound performance (specific impulse and system lifetime) of the NTP reactor design space.

2. Understand where in the trade space corresponded to fuels that could allow for the most promising flight reactor designs which could satisfy the GR&As of this study and best meet mission objectives.

To meet these goals, the definition of the fuel trade space for the FD reactor was undertaken in two steps. First, reactor participants were surveyed for inputs on available fuel forms to NTP. In addition to identifying potential candidates, recommendations were sought on corresponding performance limits (maximum use temperature or known endurance), material characteristics, as well as manufacture and performance technology readiness level (TRL) status. After these inputs were gathered, fuel candidates were assessed via a trade study to allow for AMA guidance to participants on defining the reactor design space for the FD mission.

A literature review of nuclear fuel and space nuclear system design considerations[5,6,7,8] was undertaken in order to identify figures of merit (FoMs) and desired property ranges to inform the fuel trades performed in this study. From this review and subject matter expert (SME) input ~50 metrics have been identified for use in the FD reactor fuel trades. The identified metrics were grouped into four categories (Section II.A.1): fissile fuel properties, fuel matrix properties, fuel functionality and design, and technology readiness & commercial assessment. Trades were undertaken using a weighted decision matrix approach. This allowed for a quantitative assessment each fuel system while accounting for multiple competing design features and property considerations. In general, if all categories were equally weighted, fuels which scored the highest had capability for the highest melting temperatures, desirable nuclear properties (i.e. low absorption and high moderation), as well as simple design features by which thermal-mechanical design and analyses tools already exist. Low melting temperature fuels which exhibited potential for cross-over development in pre-existing terrestrial nuclear programs also scored competitively despite low performance potential.

II.A.1. Fuel Attribute Categories

1. **Fissile Fuel Properties** – Fissile fuels are the uranium compounds which allow for spontaneous fission in a critical geometry. Fissile property metrics captured a wide range of material and nuclear properties applicable to fuel design including cross section and thermo-mechanical property data.
2. **Fuel Matrix Properties** (if applicable) – Two general types of fuels have been proposed for NTP application: structural matrix fuels and geometrically optimized fuels. Geometrically optimized fuels are composed of all net-shape uranium compound material with or without protective coatings. Structural matrix fuels are composed of a structural refractory matrix with impregnated fissile particles. Similar to fissile fuel properties, fuel matrix property metrics captured a wide range of material and nuclear properties of the matrix applicable to design.
3. **Fuel Functionality and Design** – As discussed, fuels perform a variety of functions within the reactor. Fuels can be moderating, provide structural support of the core, and most importantly maintain a coolable geometry for heat transfer. Fuel functionality and design metrics captured aspects of fuel compatibility, geometry, function, and resistance to degradation during operation.
4. **Technology Readiness & Commercial Assessment** – NTP maturation can benefit from leveraging pre-existing technical capabilities which can enable reliable, predictable fuel performance under NTP operating conditions. Whether it is utilizing a pre-existing fabrication infrastructure, established quality assurance methods, or reducing lead times, established fuel approaches can help with maturing NTP fuels to the readiness needed for a flight demo. The metrics in this category emphasized the need for pre-existing fuel fabrication, performance maturation and ties to on-going commercial efforts.

II.B. REACTOR TRADESPACE DEFINITION

Since many of the fuel designs (> 75%) that were proposed in the first phase of the study were not previously surveyed for LEU NTP application, it was not known prior to the conceptual design phase whether concepts would close or if the resultant reactor designs would have favorable performance or operating characteristics for the flight demo mission. For each design, different functional and performance parameters were requested to inform down selection of reactor concepts for further development and final reactor trade space. It was found near-term fuel forms with higher readiness, typically correlated to lower specific impulse engines due to lower temperature material limits. “Advanced fuels”, although requiring more extensive technology development, were able to meet study KPPs for both the flight demo and operational mission and enable more competitive thrust-to-weight ratios for larger thrust sizes. Reactor concept down selection was primarily driven to cover the range of thrust and specific impulse levels rather than performance and operating parameters (Fig. 1). Ultimately, eleven reactor designs spanning a wide trade space of nine impulse (750 – 950 s) and thrust (~ 5 – 45 klbf) capabilities were developed by four participating reactor subcontractors. This wide range in potential performance was desired to be investigated in this study in order to understand the impact of initial selection of FD key performance parameters (KPPs) on overall cost and value from the FD mission. Although covering the trade space was driving, reactor performance metrics were considered when trading various reactor designs which occupied the same regime in the design space. For example, some reactor designs may incorporate materials which impose more of a challenge to design for criticality for low thrust, small reactor sizes or some fuels may enable high temperature operation (high I_{sp}) but some may allow for intrinsically higher temperature operation while maintaining similar additional performance characteristics. All in all, a wide survey of potential reactor concepts was performed by industry. Beyond the specific-impulse thrust design space, reactor designs were surveyed over the range of the neutron spectrum (fast, thermal, epithermal), with and without moderating components, utilizing a wide variety of fuels in various active core configurations (no moderator, moderator block, tie-tubes, multiple fuel geometries and chemistries).

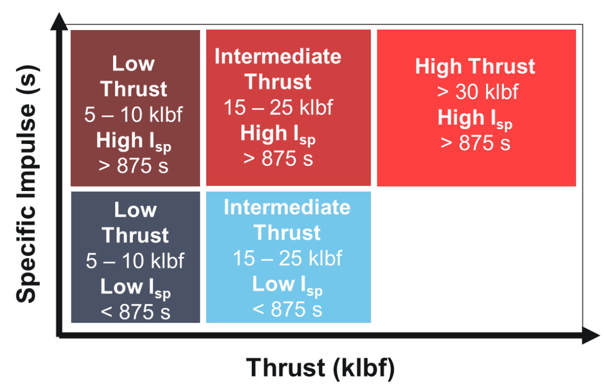

Fig. 1. The reactor and engine conceptual design trade space surveyed in this study spanned a wide range of specific impulse and thrust levels.

III. REACTOR TRADE STUDY AND KEY FINDINGS

In order to assess the attributes and limitations of each reactor subsystem (RSS), it was desired to identify figures of merit related to reactor subsystem design, including: performance, functionality, and related programmatic elements of: cost, schedule, risk/opportunity, and extensibility/strategic value. Each of these criteria are necessary for reactor designs to fulfill in order to satisfy performance and functional requirements as well as exhibit the project elements necessary to sustain a successful program. Each of these metrics were traceable to stakeholder informed mission objectives (Fig. 2). The performance and functional related metrics provided an understanding of the value of each design, while programmatic elements typically pointed to the cost of each design. Similar to the fuel assessment trade, reactor FoMs were obtained through literature review[7,8,9,10,11,12] and SME input. Reactor trades were strongly influenced by NERVA/Rover reactor[8,9] and engine[10] trades performed late in the development program prior to the Small Nuclear Rocket Engine Definition Study and similarly structured.

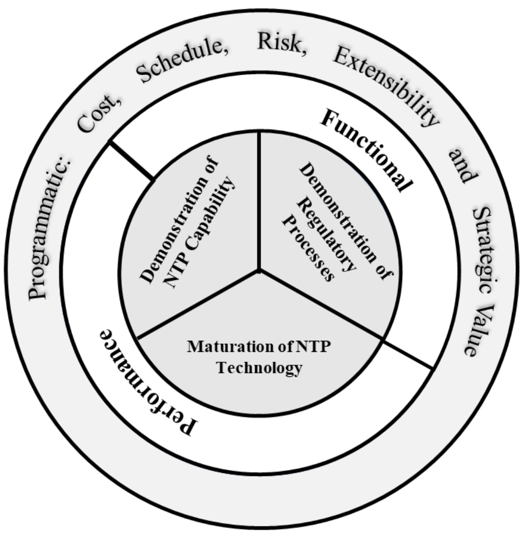

Fig. 2. Reactor trade categories: performance, functional, and programmatic (cost, schedule, risk, extensibility and strategic value) assess the characteristics of conceptual FD reactor designs with respect to mission objectives.

III.A. Reactor Trade Categories

1. **Performance** – Reactor operating parameters strongly impact performance of the engine. The engineering design of the reactor also impacts engine performance by adding extra weight or inefficiency in heating the propellant. In this category, reactor operating parameters tied to engine performance such as specific impulse and thrust to weight were traded. Most important metrics with regards to the performance category included reactor exit temperature and weight breakdowns of different critical reactor components. Margins of the design at steady state temperature were also collected and traded.
2. **Functional** – Design attributes related to operation and functionality of fuel and reactor. This was further divided into secondary categories of reactor operability and fuel design. Reactor operability was assessed by comparing reactor temperature coefficients and reactivity worths to desirable regimes for stable reactor operation. This was considered an important aspect to the trade since stable reactor operation is anticipated to greatly impact whether reactor operation is authorized or not. Fuel design was assessed using the fuel trade criteria previously developed.
3. **Programmatic** – During the development of engineering systems performance, functional, and programmatic requirements need to be met in order to allow for successful system maturation. For this study, no requirements were imposed on designs beyond GR&As. However, attributes related to cost, schedule, high-level risk/opportunity, extensibility, and strategic value associated with each concept were included as FoM to inform stakeholders of anticipated programmatic trends to consider for each concept.

The trade was weighted and scored using the analytical hierarchy process[13] (AHP) for pairwise comparison of different metrics and their sub criteria to minimize bias during the weighting process. The quantitative values collected from each of the reactor designs were scored on a non-linear scale from 1 to 5 with 1 being the least desirable range and 5 being the most desirable range. The weightings were determined via AHP and weighted scores were determined for each of the designs to inform rankings. Because the scope of this study consists entirely of conceptual fuel and reactor designs, each concept was compared to metrics reported for the historic small nuclear rocket engine (SNRE) design (HEU, ~900 s, 15 klbf engine) as a reference.

It was found when surveying and equally trading all three elements: performance, functional, and programmatic, that near-term flight demos leveraging more mature fuel forms scored well regardless of thrust level if they were capable of achieving specific impulse values above 800s. When the trade was performed with performance as the most highly weighted criteria, advanced fuel designs looked more favorable indicating more extensive fuel technology development was necessary. There existed no design which universally satisfied all trade criteria. However, for each of the industry design approaches, feasible conceptual reactor designs were able to be matured which met study KPPs for the FD mission in the case of near-term fuels and the operational mission for advanced fuels within the same class.

VII. CONCLUSIONS

Over the course of this study, nine conceptual reactor designs were matured by four participating reactor companies. Reactor designs spanned a large range of operating conditions which satisfied engine thrusts of 5 – 45 klbf and Isps of 750 – 950 s. Designs matured by each company provided confidence that a NTP FD reactor design is feasible for a wide range of performance needs (Isp and thrust) with a variety of different fuel types and reactor configurations utilizing HALEU enrichment. To guide conceptual design development and recommendations reactor subsystem performance, functional, and programmatic metrics were gathered and an initial fuel assessment trade and final comprehensive reactor subsystem trade were performed.

As with any complex engineering system, KPPs and requirements are major design drivers that will ultimately govern project cost and schedule. A near term-flight demo was traded to have both technical and programmatic value which when wisely implemented to build up technical capability and programmatic confidence without lost cost and schedule. Near-term flight demos should incorporate more mature fuels capable of being extensible to higher performing advanced fuels. A NTP FD with lower KPP values of thrust and Isp than the operational mission requirements, can still provide value by allowing for maturation of critical reactor components, initial development of fundamental manufacture and test methods, as well as the establishment of critical infrastructure under less demanding operating conditions which may allow for a higher probability for success for both the flight demo and operational mission fuel and reactor technology development. Extensible fuels which utilize the same fabrication methods (same fuel class) as their higher performance counter parts and allow for similar method of reactor operation and control (similar spectrum and feedbacks) are recommended in the case a lower performance, steppingstone FD approach.

Cost, schedule, and risk are major drivers for programmatic decision making when considering a near-term (mid- to late 2020s) FD. Although cost-sharing, interproject, or interagency buy in is desirable to increase value to the taxpayer, and accelerate development through knowledge sharing, there was no reactor design surveyed which excels for all potential stakeholder needs (performance, functional, programmatic). Investment in critical infrastructure (manufacture or testing facilities) can allow for value and methods when appropriate extensibility exists between concepts, that can be maintained and remain in service to support multiple projects depending on changing national priorities.

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EVALUATING THE EFFECT OF VEHICLE SHAPE AND ASTRONAUT POSITION ON THE WHOLE BODY EFFECTIVE DOSE EQUIVALENT IN DEEP SPACE

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I. INTRODUCTION

As future crewed, deep space missions are being planned, it is important to assess how spacecraft design can be used to minimize radiation exposure. Collectively with shielding material[1], vehicle shape and astronaut position must be used shield astronauts from the two primary sources of space radiation: Galactic Comic Rays (GCRs) and Solar Particle Events (SPEs). GCRs, which are composed of highly energetic and fully ionized elements, are a chronic source of radiation exposure and account for the majority of the background radiation. SPEs, which originate from coronal mass ejections, are composed of mostly protons. SPEs are statistically, yet spontaneous in nature, and vary in magnitude, composition and duration[2]. For this paper, only the GCR source is analyzed.

The On-Line Tool for the Assessment of Radiation in Space (OLTARIS) version 4.1 analysis package [3, 4, 5] is used to evaluate and analyze this detailed radiation field. OLTARIS is a tool developed by the National Aeronautics and Space Administration's (NASA) Langley Research Center to enable engineering and research related space radiation calculations. OLTARIS utilizes a 1D particle transport code, HZETRN (High Charge and Energy Transport)[6, 7, 8], and has the ability to analyze 3D objects with this 1D code.

II. THEORY

The objective of this research is to evaluate how effective vehicle shape and astronaut position are at reducing the whole body effective dose equivalent, E_D absorbed in astronauts. To represent the average anatomy of astronaut, the Male Adult voXel (MAX)[9], 2005 version, human phantom is used, due to this being OLTARIS'conservative human phantom[10, 11]. A simple spherical and right circular cylinder(RCC) geometry are used. The ray distribution used in this study is the 1002 geodesic[1] arrangement of the rays on the sphere [12].

II.A. GCR Boundary Condition

GCRs are composed of fully ionized stable and metastable isotopes. Although GCRs include every naturally formed element, not all elements are in high abundance. Protons account for roughly 91% of the total flux, alpha particles account for approximately 8%, and heavier particles account for less than 1% of the total flux. Even though the abundance of heavy particles is relatively low, they contribute to approximately 86% of the total dose equivalent[13].

The 1977 solar minimum boundary condition is the event used in this research. The measured spectra at 1 astronomical unit for hydrogen, helium, and heavy ions up to ^{58}Ni are used. The abundances for species heavier than nickel (Z > 28) are typically four orders of magnitude less than that of ^{56}Fe[14] and therefore not included in this boundary condition.

II.B. Radiation Limits

NASA details in the Space Permissible Exposure Limits (SPELs) for Space Flight Radiation Exposure Standard [15, 16], that astronauts risk from ionizing radiation shall not exceed 3% Risk of Exposure-Induced Death (REID) for cancer mortality at a 95% confidence level to limit the cumulative effective dose received by an astronaut throughout his or her career. Short-term dose limits are also imposed to prevent clinically significant non-cancer health effects. Due to the large number of uncertainties that remain in knowledge of biological effects of GCRs, specifically heavy ions, no limits have been set. The goal of the research is to identify the materials or the characteristics future materials should have to limit human risk.

II.C. Whole Body Effective Dose Equivalent

The Ray-by-Ray method and lumped tissue model, as defined within OLTARIS is used in this analysis. The MAX human phantom is represented by approximately 1000 points of interest that represent the organs cited by National Council on Radiation Protection and Measurements (NCRP) Report #132[17]. These points are used to determine the E_D for each combination of material and thickness geometry.

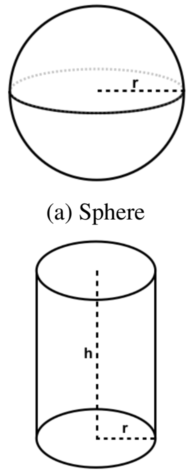
The E_D is defined in NCRP report 132 as a weighted sum of organ dose equivalents.

$$\mathrm{E_D} = \sum_{i=1}^{N_{\mathrm{organs}}} \mathrm{w_i H_i} \tag{1}$$

where the relative weights, w_i, are categorized by organ or tissue; the organ dose equivalents, H_i, are determined by calculating a dose equivalent at each point and then averaged. These values are then used, as described above, to determine the E_D.

For GCRs, OLTARIS calculates the E_D absorbed per day and multiplies by the number of days under investigation, therefore, E_D results, for GCRs, are reported in $\frac{mSv}{day}$.

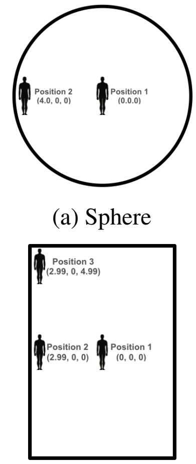
[1]Defined as a regular polygon with identical equilateral triangular faces. Rays are projected along the triangle sides and faces directly onto the shape



(a) Sphere

(b) RCC

Fig. 1: Space Vehicle Shapes



(a) Sphere

(b) RCC

Fig. 2: Human Phantom Positions

II.D. Vehicle Shapes

Two vehicle shapes are used in this study: a sphere and a right circular cylinder(RCC). The sphere is a hallow shell with a radius of 4.072m, as shown in Figure 1a, and was chosen due to each ray in OLTARIS experiencing the same amount of shielding, therefore effectively calculating the maximum E_D for each thickness geometry. The RCC was chosen due its similarity to the international space station sections, and has a radius of 3m and height of 10m, as shown in Figure 1b.

II.E. Astronaut Positions

Five human phantom locations, as shown in Figure 2, are investigated in this study: 2 in the sphere and 3 in the RCC. In the sphere, the first human phantom position is located at its center with coordinates (0,0,0), and the second is located near the sphere's shell with coordinates (4.0,0,0). In the RCC, the first human phantom position is located at its center with coordinates (0,0,0). The second human phantom position is located near a singular wall, in this study coordinates (2.99,0,0) is used. The third human phantom position is located near the intersection of two walls, in this study coordinates (2.99,0,4.99) is used.

II.F. Materials

For this study, aluminum, polyethylene, and liquid hydrogen were chosen to represent a metal and polymer[1]. Table I shows the density and line colors associated with each material. Aluminum shields will be represented by shades of red and pink. Polyethylene shields will be represented by shades of purple and blue.

III. RESULTS AND ANALYSIS

Due to the large number of data points, the results for each material, vehicle shape and astronaut position use a different color, as shown in Table I. When referencing the combinations of vehicle shape and astronaut position, the vehicle shape(position #) form is used. For example, for position 1 in the sphere, the abbreviation SPHERE(P1) is used. For this paper, only the Polyethylene and Aluminum are analyzed.

III.A. Polyethylene

Figure 3 shows the E_D as a function of the polyethylene thickness for each human phantom position in the sphere and RCC vehicles, using the GCR boundary condition. All combinations of vehicle shapes and astronaut positions begin similarly between 1.214 and 1.213$\frac{mSv}{day}$ at 0.01$\frac{\mathrm{g}}{\mathrm{cm}^2}$ and decreases to roughly 10$\frac{\mathrm{g}}{\mathrm{cm}^2}$. After which the lines begin to spread as they continue to decrease, this is due to secondary particle buildup from nuclear nuclear fragmentation in the RCC(P1) and SPHERE(P1). When the human phantoms are moved to positions 2 and 3, the larger decrease in E_D is because of the increased shielding experienced by each ray. After 30$\frac{\mathrm{g}}{\mathrm{cm}^2}$, each continues to decrease steadily to 1000$\frac{\mathrm{g}}{\mathrm{cm}^2}$ with values ranging from 0.006713$\frac{mSv}{day}$ for SPHERE(P1) to 0.002008$\frac{mSv}{day}$ for RCC(P3). The shielding ability and order of each vehicle shape and astronaut position is directly related to the mean amount of shielding each ray experiences. Though SPEs are not included in this paper, it is important to note that the same pattern is not seen using the SPE boundary condition.

Figure 4 shows the relative change of E_D at human phantom positions 2 and 3, when compared to position 1 for each vehicle shape with polyethylene shielding. The first maximum decrease in E_D occurs at 7.5$\frac{\mathrm{g}}{\mathrm{cm}^2}$, with the SPHERE(P2) showing the greatest decrease with 11.8% with RCC(P2) and RCC(P3) showing similar maximums of 9.9 and 9.4% respectfully. As the vehicle thicknesses increase, the relative changes decrease to 30$\frac{\mathrm{g}}{\mathrm{cm}^2}$, before increasing with relative changes from 52.97% for SPHERE(P2) to 39.58% for RCC(P2) at 1000$\frac{\mathrm{g}}{\mathrm{cm}^2}$. The continued increase in relative difference to position 1, shows that even at thicknesses of 1000$\frac{\mathrm{g}}{\mathrm{cm}^2}$, the increase in shielding when the astronauts are moved to positions 2 and 3 is still large enough to greatly effect the E_D.

TABLE I: Materials Analyzed

Material Name	Density ($\frac{g}{cm^3}$)	Sphere Line Color		RCC Line Color			Material Reference
		P1	P2	P1	P2	P3	
Liquid Hydrogen	0.071						[5]
Polyethylene	1.00						[5]
Aluminum	2.70						[5]

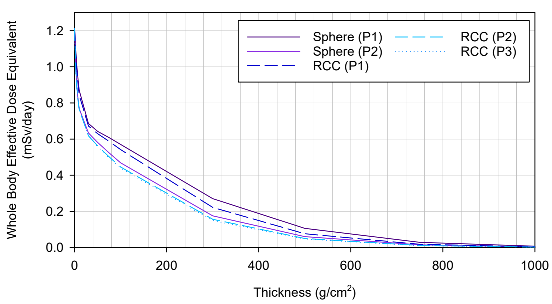


Fig. 3: Whole Body Effective Dose Equivalent as a function of Thickness of Polyethylene

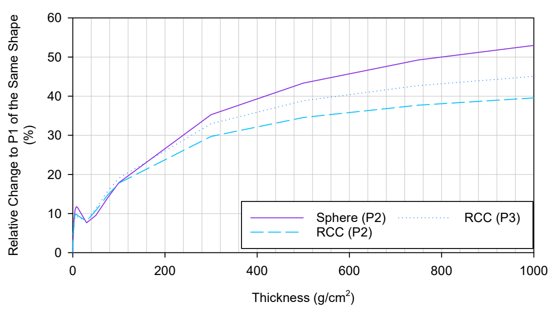


Fig. 4: Relative Change to P1 of the Same Shape as a function of Thickness of Polyethylene

III.B. Aluminum

Figure 5 shows the E_D as a function of an aluminum thickness for each position in the sphere and RCC vehicles, using the GCR boundary condition. All vehicle shapes and astronaut positions begin the same with E_D of $1.214\frac{mSv}{day}$ at $0.01\frac{g}{cm^2}$ and decreases to roughly $10\frac{g}{cm^2}$. Similar to polyethylene, the aluminum lines decrease with an increasingly less slope and but with a wider spread. The spread is larger than the one recorded for polyethylene, due to secondary particle build due to nuclear fragmentation is larger in aluminum[1]. This is because aluminum's increased density and mass number. After $30\frac{g}{cm^2}$, the SPHERE(P1) and RCC(P1) lines increase slightly, before following the pattern of the other lines and decreasing steadily to $1000\frac{g}{cm^2}$. This increase is due to the aluminum thickness not being wide enough to not allow the buildup of secondaries to escape the shielding. This is also witnessed at $100\frac{g}{cm^2}$ with the RCC(P1) separating from the SPHERE(P1), showing greater shielding ability because of the increased shielding thickness experienced in a RCC compared to a sphere. At $1000\frac{g}{cm^2}$, E_D values range from $0.02642\frac{mSv}{day}$ for the SPHERE(P1) to $0.007244\frac{mSv}{day}$ for RCC(P3). Also similar to polyethylene, The shielding ability and order of each vehicle shape and astronaut position is directly related to the mean amount of shielding each ray experiences.

Figure 6 shows the relative change of E_D at human phantom position 2 and 3, when compared to position 1 for each vehicle shape for aluminum shielding. The first maximum decrease in E_D occurs at $10\frac{g}{cm^2}$ with 7.79% change for SPHERE(P2) and $7.5\frac{g}{cm^2}$ with 6.42 and 6.00% change for RCC(P2) and RCC(P3) respectively. As the thicknesses increase, the relative changes decrease to $50\frac{g}{cm^2}$ for SPHERE(P2) and $30\frac{g}{cm^2}$ for RCC(P2) and RCC(P3), before increasing with relative changes from 65.64% for SPHERE(P2) to 48.53% for RCC(P2) at $1000\frac{g}{cm^2}$. Also similar to polyethylene, the continued increase in relative difference to position 1, shows that even at thicknesses of $1000\frac{g}{cm^2}$, the increase in shielding when the astronauts are moved to positions 2 and 3 is still large enough to greatly effect the E_D.

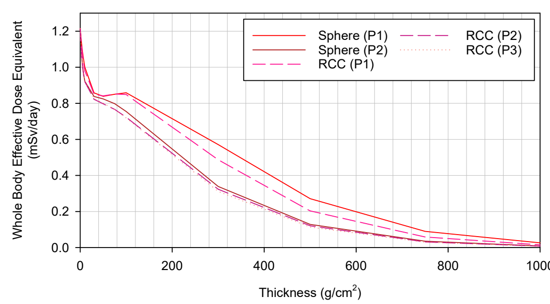


Fig. 5: Whole Body Effective Dose Equivalent as a function of Aluminum Thickness

IV. CONCLUSIONS

E_D vs. Thickness and the relative change when compared to the human phantom position 1 analyses show that, using OLTARIS, moving a human phantom closer to a wall does significantly decrease the E_D. This pattern is not dependent on material nor boundary condition, but the mean shielding thickness a source ray must travel through for the GCR boundary condition. SPHERE(P2) is found to have larger relative change, except between 30 and $100\frac{g}{cm^2}$ for polyethylene and aluminum. Also that for both materials, the continued increase in the relative difference to position 1 plots, show that even at

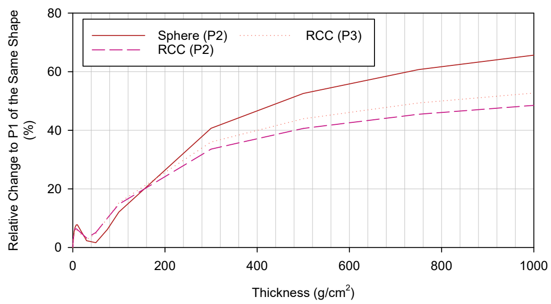


Fig. 6: Relative Relative Change to P1 of the Same Shape as a function of Aluminum Thickness

thicknesses of $1000\frac{\text{g}}{\text{cm}^2}$, the increase in the amount of shielding when the astronauts are moved to positions 2 and 3 is still large enough to greatly effect the E_D.

V. FUTURE & CURRENT WORK

Current work is focused on evaluating if these same trends are seen in MCNP6 using the same boundary conditions, materials and OLTARIS configuration; as well as when the simplifications inside OLTARIS are removed.

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REGULATIONS ON GROUND TESTING SPACE NUCLEAR SYSTEMS

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The current regulatory framework allows for different options to test a terrestrial based space nuclear reactor or integrated system which vary depending on the developer. This study begins by examining the existing authorities and relevant indemnification to clarify the existing pathways for space nuclear power and propulsion (SNPP) testing. Following an analysis of the current regulatory structure, this brief will explore future options and gaps of the current approach.

I. Current Licensing Framework

Developing a nuclear power or propulsion system for space will require some degree of testing on Earth. Testing can range from the irradiation of parts in existing test reactors to the full certification test of a nuclear thermal rocket (NTR) on a stand. These tests pose safety and environmental risks, for example from the radioactive emissions propelled from a NTR or the possibility of an accident in a power reactor. Any space nuclear power and propulsion (SNPP) testing on the ground will require assessment and mitigation of these risks. However, SNPP test regulations have changed since programs were licensed under the Atomic Energy Commission (AEC), the regulatory authority has since split between the Department of Energy (DOE) and the Nuclear Regulatory Commission (NRC) [1].

I.A. DOE Authority

I.A.1. DOE Licensing

Following the reorganization of the AEC in 1974, government missions using space nuclear power have been developed and fueled at National Labs, under the jurisdiction of the DOE, to facilitate space nuclear systems testing. The DOE licensing process consists of limiting risks. All DOE-authorized nuclear facilities must set an authorization agreement between the facility, contractor, etc. and the DOE. That agreement consists of all steps required to ensure that the facility is safe and environmentally aware, including those listed in Figure 1. The safety review process, captured in the Documented Safety Analysis (DSA), follows a series of steps, which are typically quantitative and risk-informed, aimed at ensuring the facility is safe [2].

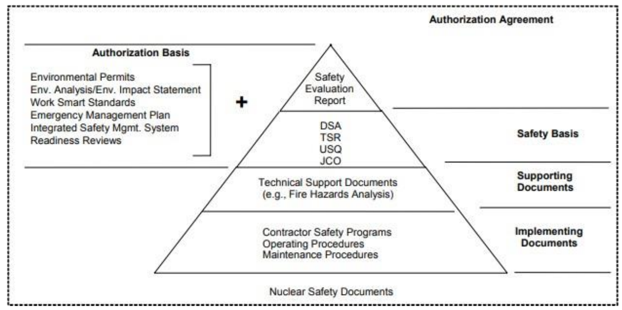

Fig. 1 Relationship of the Safety Basis, Authorization Basis, and Authorization Agreement [3]

The level of analysis required by the DOE is dependent on a categorization of the nuclear facility. Lower category facilities are expected to have greater risk and thus require additional levels of analysis, as shown in Table 1.

I.A.2 DOE Indemnification

The DOE can provide indemnification in each contract that involves risk of a nuclear accident [4]. This indemnification, under the Price-Anderson Nuclear Industries Indemnity Act (Price-Anderson), covers damage to persons and property damage for nuclear accidents. For Price-Anderson to apply, the contract (or license) must cover the activities under which the accident took place (e.g., the launch) and the owner of the system (i.e., the entity covered by Price-Anderson) must have a causal link to the accident. The indemnity covers not only the owner of the system but also all public liability extending above the financial protection required under the contract or license. The agreement of indemnification includes two tiers:

1. The contractor provides financial protection that the Secretary of Energy determines to be appropriate; and
2. Any additional liability (up to $10B above the required financial protection) will be indemnified by the Federal Government.

Additionally, the DOE (along with all executive agencies) can provide indemnity from Public Law 85-804 (50 U.S.C. § 1431-1435), which allows the President and delegated agencies to exercise contractual relief as long as it facilitates the national defense. The standard use for Public Law 85-804 "is when the risk arises from an activity that is unusually hazardous or nuclear in nature, with risk of loss so potentially great that the contractor's financial and productive capabilities would be severely impaired or disrupted" [5].

TABLE I. DOE Hazard Categorization for Safety Analysis Reports.

Categorization	Description	This Includes
Category 1	Potential for significant off-site consequences	Reactors that have a steady-state power level greater than 20 MWt
Category 2	Potential for significant on-site consequences	Potential for nuclear criticality events
Category 3	Potential for only significant localized consequences	Potential for worker exposure at acceptable thresholds[1]
Below Category 3	Consequence less than the basis for categorization as Category 3	

I.B. NRC Licensing

Under the Atomic Energy Act, NRC has the authority and responsibility to regulate all nuclear activities not authorized by DOE, including fuel production and transportation, product and utilization facilities, permitting for the construction of nuclear power facilities. At the time of writing, NRC has advised other Federal agencies on the development and launch of space nuclear systems, but it has never licensed such a new space system or its testing. Current regulations set two classes of NRC licenses: class 103 and class 104 [6].

I.B.1. NRC Class 103 Reactors

Class 103 licenses and its associated review process is for commercial and industrial facilities where "the facility is to be used…is devoted to the production of materials, products, or energy for sale or commercial distribution, or to the sale of services, other than research and development or education or training" [7]. The prototypical class 103 facility is a commercial power plant selling electrical power to the grid. The requirements for a class 103 facility are more involved and stringent than that for a class 104, including a demonstration of each safety feature, analyses or testing to show interdependent effects of each safety feature are acceptable, and access to sufficient safety design operating data. Typically, 103 licensed facilities are based on well-developed, tested, and demonstrated concepts ready for commercial deployment. A nuclear reactor to be operated in space commercially could possibly be licensed as a class 103 facility, although it is unlikely.

I.B.2. NRC Class 104 Reactors

Space nuclear reactors, especially their testing and development, would likely be licensed as class 104 facilities. Class 104 facilities are mainly used for research and development (e.g., a reactor to be tested on the ground). NRC describes these facilities as typically smaller than 20 megawatts-thermal and none of the active 104 reactors are larger than 20 megawatts-thermal [8], leaving it uncertain where a space reactor such as a multi-100 MW nuclear thermal propulsion reactor could receive class 104 licensing. Class 104 facilities follow similar top-level standards as power reactors, but their implementation is guided by a minimal regulatory approach to encourage widespread and diverse research and development [9]. NUREG-1537, or the *Guidelines for Preparing and Reviewing Applications for Non-Power Reactors*, lays out the standard review plan (SRP) and acceptance criteria including accident scenarios that should be discussed [10]. The SRP is relatively deterministic and based on power reactors cooled/moderated by light water.

I.B.3. NRC Indemnification

The NRC can provide indemnification under Price-Anderson. The NRC requires operators of nuclear reactors to demonstrate financial responsibility and in return provides indemnification; for other types of nuclear facilities, the NRC can enter into agreements at its discretion [4]. The indemnification has three tiers:

1. The owners of the nuclear system or material cover the first $450M of damages, typically via insurance;
2. Damages past the first tier are covered by a pool of funds made up of contributions from all operating reactors, currently at approximately $13B; and
3. If the second tier is depleted, Congress must determine whether to appropriate additional relief.

To the knowledge of this study the Plum Brook Reactor Facility, licensed by the AEC to be a nuclear propulsion test reactor for NASA, and since decommissioned by the NRC, is the only example of SNPP activity under the sole aegis of NRC. The NRC has yet to facilitate the licensing of a new facility developing space nuclear systems and NRC licensing a space nuclear ground testing facility (especially from a Federal agency) has little-to-no precedent.

1.C. Licensing for Different Space Nuclear System Developers

Who the user of the space nuclear system is and the type

[1] For more information on threshold levels see DOE-STD-1027-92.

of nuclear system the user designs both play an important role in determining how that user might navigate the regulatory process. For current SNPP projects within NASA or the DOD, National Laboratories provide infrastructure and the ability to license new capabilities to enable successful system testing. Idaho National Lab (INL) tested Radioisotope Power Systems (RPS) currently in space like Curiosity and New Horizons, and has the capability to test new RPS systems. Future RPS systems require the DOE to complete safety basis analyses, including the DSA and hazard analyses, but this is a process regularly done with projects at INL. Similar to RPS systems, fission-based nuclear power systems can use existing or modified facilities and follow established licensing procedures. Kilopower, a small fission powered space nuclear system, was demonstrated at the Nevada National Security Site from November 2017 to March 2018. National Laboratories currently lack the infrastructure to test a Nuclear Thermal Propulsion (NTP) system but if the sufficient infrastructure were developed (e.g., test stand and exhaust capture capabilities) the DOE could license this testing. DOE facility authorization is flexible, consisting of typically quantitative, risk-informed steps. A final DOE authorization comprises multiple analyses and ultimately an agreement between the facility, the contractor, and the DOE. An existing agreement could cover new SNPP tests, such as placing a space power reactor at Idaho National Lab, or could require a new authorization agreement, such as the DOE establishing a test facility at a NASA center. NASA could also look to the NRC for authority to test space nuclear systems at their own facilities, however this has yet to be put into practice.

The regulatory process does change if the developer is a commercial entity. Commercial entities can operate under DOE licensing authority at National Labs in the following circumstances: "the construction or operation of a production or utilization facility for the Department [DOE] at a United States Government-owned or controlled site…or the use or operation of a production or utilization facility for the Department in a United States Government vehicle or vessel: Provided, that such activities are conducted by a prime contractor of the Department [DOE], under a prime contract with the Department [DOE]"[11]. Commercial entities can operate under the authority of the DOE until the reactor is "operated…for the purpose of demonstrating the suitability for commercial application of such a reactor" [12], at which point the reactor would require NRC licensing. NRC licensing is required for commercial activities that take place during and after the demonstration phase. It is unclear what specific activities constitute demonstrating the suitability for commercial application of such a reactor but once at this point the commercial entity requires NRC licensing.

II. Limitations and Paths Forward

The licensing process for testing SNPP systems will be a major challenge if it requires a new facility or safety basis. Most space nuclear testing can likely be conducted in existing facilities, such as component-level irradiation or even system-level demonstration as illustrated by the Kilopower test. Licensing a power reactor demonstration may also be simple, as it fits closer to the expertise of regulators and especially because it may fit within the safety basis of a larger site like INL. Most of this testing, however, will fall under DOE authority. NRC may have licensed non-power reactors that could be used for component-level testing and may provide technical assistance in certain cases, but the NRC does not seem equipped or interested in licensing a space nuclear power test reactor.

For new facilities or tests, DOE authorization appears to be only tried and tested approach. DOE has existing facilities that can host new facilities. Furthermore, it can establish a test facility at a NASA site, similar to how it has established a Radioisotope Thermoelectric Generator (RTG) assembly and storage center at the Kennedy Space Center. DOE's licensing process is relatively flexible and technology- inclusive, perhaps even allowing a full-scale NTR test. Previous and current certification testing plans have baselined using DOE authorization. However, DOE authorization for a new testing facility will still be lengthy and complex, especially combined with environmental approvals.

The status quo points to one available regulatory option for SNPP test or demonstration missions, requiring all government and private entities to work through the DOE. Such a consolidated system may leverage DOE expertise, its broad indemnification authority, and help with testing and infrastructure commonality. However, if demand for DOE facilities is high, the DOE may also be slow and bureaucratic. The DOE also does not have to take on and license a company or other government agency. The NRC has the authority to license an SNPP test facility but its current licensing approach is not conducive to licensing space nuclear systems. This uncertainty could add expense and delay to any NRC licensing process.

Another option could be to set up a separate regulatory process specific to SNPP testing and demonstration missions. This process could be similar to that established for launch review and approvals under NSPM-20, including interagency review and potentially Presidential approval. However, since ground testing falls directly under the utilization of nuclear material, without legislative change, the Atomic Energy Act will still require that the NRC or DOE license the facility. A legislative change does not seem advisable given the few expected testing missions, especially since DOE and NRC would likely be involved in any such review process and both have the capabilities to license a testing facility without a new licensing process, unlike a launch. A review process similar to nuclear launch could possibly be combined with DOE or NRC authorization, but since one of the entities would still need to give final approval, it would likely only lengthen and add complexity to the licensing process.

III. CONCLUSIONS

Existing DOE licenses may provide sufficient authority to cover many tests and can be extended for most power testing. These solutions will require DOE authorization and oversight. For a new test facility (e.g., at a NASA site), mainly considered for NTR engine testing, DOE is much better positioned to provide authorization. As a fee recovery agency, NRC is not likely to develop the capabilities to license such sites, nor is it clear that they should. However, DOE should still be encouraged to provide expedient licensing for SNPP systems and to support the missions of private entities or other government entities.

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FISSION REACTOR INADVERTENT REENTRY

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NASA's Nuclear Power and Propulsion Technical Discipline Team has directed an effort to consider possible improvements to the launch approval process as it relates to fission reactors, including issues related to reentry risk. This paper includes a discussion of the issues associated with different types of inadvertent reentry, the possible consequences of those events, a review of previous work in the area, security and nonproliferation issues, and options for safety requirements that might be considered, including mission implications.

I. TYPES OF INADVERTENT REENTRY EVENTS

There are a number of types of reentry events that can potentially occur with missions containing fission reactors.[1] These include:

- During Ascent to Orbit
- From Low Earth Orbit
- From Mid and High Earth Orbits
- During an Accidental Flyby Reentry or a Long-Term Reentry due to Failure Away from Earth
- During a Direct Return-to-Earth Scenario

Each type of reentry event can produce a variety of possible adverse environments for fission reactors, depending on a number of factors. For example, the spacecraft containing the fission reactor may be attached to all or part of the launch vehicle or may still be encapsulated in the launch vehicle fairing.

When a reentry accident occurs, a number of physical phenomena may occur such as overpressures from bursting pressurant tanks, destruct mechanisms, and propellant explosions in-air. In addition, there may be launch vehicle and spacecraft components or fragments impacting the fission reactor along with adverse thermal environments due to propellant fires. The design and initial configuration of the fission reactor and these potential environments insulting the spacecraft and fission reactor will determine the configuration during reentry.

As reentry progresses, the spacecraft and/or fission reactor would experience aerothermal and aerodynamic loads. The reentry body's shape, mass, aerodynamic properties, tumble rate, altitude, latitude, longitude, azimuth, velocity and flight path angle, as well as the atmospheric properties will determine the amount of aerothermal and aerodynamic loading. Velocities can exceed 11 km/s for a flyby event. The reentry configuration of the spacecraft and the design of the fission reactor will determine the impact of these externally imposed loads. Depending on the severity of these loads and the ability of the reentry structure to withstand them, there is a potential for the reentry body to break apart further or ablate/burnup prior to impact. In addition, the reentry loads can potentially cause energetic material (e.g., fuel and pressurant tanks) to explode, burn, or become projectiles, which may impact the reactor.

II. POSSIBLE REENTRY OUTCOMES

There are three potential outcomes for a fission reactor during reentry. First, the fission reactor can burnup in the atmosphere. Second, it can impact the Earth's surface intact with or without additional spacecraft components. Finally, it can break apart during reentry, but its various components survive reentry and impact the Earth's surface (a scattered reentry).

II.1 Burnup in the Atmosphere

If the aerothermal loading on the fission reactor during reentry is great enough to ablate and/or vaporize its components, the fission reactor's material will be disbursed throughout the atmosphere, thus minimizing the radiation exposure to individuals. In the 1970s and 1980s, B. W. Bartram performed a detailed analysis of worldwide dispersion of aerosols and vapors.[2] For a cold reactor the dose to an individual would be many orders of magnitude less than a millirem. For a previously operated (hot) reactor, Bartram calculates maximum doses to an individual in the millirem range for a 1-MW thermal reactor with a 10-year space mission and a 1-year cool-down period before reentry. Because of the low consequences, early space reactor missions (such as Systems for Nuclear Auxiliary Power (SNAP)-10A) [3] strived to achieve complete burnup of the reactor in the upper atmosphere. However, because fission reactors are designed to utilize high temperature materials, it is difficult to verify that a system will sufficiently breakup and vaporize without special design features, such as a destruct mechanism.

II.B Intact Impact

If the fission reactor does not burn up or break up during reentry, then it and possibly parts of the spacecraft will impact the Earth's surface. Any additional structure(s) at impact may alleviate some of the stresses of impact or more likely put more stress on the fission reactor at impact. In addition, if any high-energy materials survive reentry and impact with the fission reactor, the fission reactor may experience additional overpressures, fragment insults and adverse thermal environments.

Intact re-entry is assumed to occur if an engineering solution (such as an aeroshell) is used to protect the reactor core and associated structure. If the impact occurs on land, radiation doses to the public can occur. If the impact occurs in the ocean, doses to the public are effectively zero, as a few meters of water provide sufficient shielding. Recovery in the ocean may be very difficult, but the public risk is effectively eliminated.

For a cold reactor, intact reentry is not an issue unless the reactor goes critical. Neither direct radiation nor dispersal from impact cause any serious dose.[1] An intact reactor impact may lead to fission product generation, either because it was critical during reentry or because criticality occurred upon impact. Without criticality, radiological impacts are limited to those from the fission products present from previous operation in space.

Hot reactor reentry for an intact reactor could yield radiation doses to the public from direct gamma radiation, from fission products released at impact, or from fission products released due to a criticality excursion. Calculations of direct radiation dose from an unshielded SP-100 reactor with zero decay time that lands without being buried varied from 80 rem/hr at 100 m to ~800,000 rem/hr at 1 m.[4] The same reactor core with 1 year of decay prior to reentry had a direct radiation dose of 20 mrem/hr at 100 m to 180 rem/hr at 1 m.

Public dose from an impact causing dispersion of fission products or a criticality excursion that destroys the reactor can produce doses to the public in the millirem to hundreds of rem range. Studies of fission product inventory for SP-100 produced a maximum inventory of 4.E7 curies after 7 years of operation.[5] Estimates of fission product inventory for the Rover/ Nuclear Engine for Rocket Vehicle Application (NERVA) nuclear thermal rocket program predicted a maximum peak inventory at 1.E9 curies.[6] Estimates for burst excursion for a small Kilopower space reactor predict a maximum peak inventory of 1.E7 curies assuming a 5.E18 fissions event.[7] With 1 day of decay, most fission product inventories will drop ~2 to 3 orders of magnitude and within 1 year they drop ~5 orders of magnitude.

II.C Scattered Impact

A scattered impact of a reactor is similar to a completely intact reactor impact except that the fission reactor breaks up during reentry, and the individual parts would separately impact the Earth's surface. While criticality is precluded, a scattered reentry of a hot reactor will have the potential to impact a greater number of people and a much larger area. The reentry of Cosmos 954 scattered debris over a wide area.[8] The total area searched for reentry debris was about 124,000 square kilometers. Scattered reentry makes it more difficult to control access to the crash site. Even relatively small pieces of a hot reactor can lead to elevated doses in the immediate vicinity. Scattered reentry is the least desirable outcome for a reentry event.

III. SECURITY IMPLICATIONS AND INTERNATIONAL AGREEMENTS

The security implications of an inadvertent reactor reentry are driven primarily by the form of the reactor fuel. The Department of Energy identifies four categories of material based on the nature and quantity of material.[9] The most sensitive material is Category I and includes materials that might be used directly in a nuclear weapon, such as highly enriched U-235 or Pu-239. Category IV material is of little concern from a security and nonproliferation standpoint and includes U-235 enriched to less than 20%. Category I materials, and to a lesser extent Category II and III materials, need to be secured as quickly as possible should an intact or scattered inadvertent reentry occur. Recovery may be very difficult for a scattered reentry or if the reentry occurs over the ocean. There is limited reentry guidance in international treaties and agreements. United Nations (UN) Resolution 47/68[10] provides guidance on the use of nuclear power in space; however, this resolution is nonbinding. The Outer Space Treaty, Article VII, indicates that a launching nation is liable for damages to other nations.[11]

IV. PREVIOUS APPROACHES TO REENTRY

Reentry issues have been considered for a number of previous space reactor programs, see Table I.[12] Reentry strategies have evolved over the years. Complete high-altitude burnup is desirable because of the low doses that result, and the potential difficulties associated with reactor recovery. However, based upon a review of past programs, it is clear that burnup cannot be achieved without an active system such as the core pusher deployed on the Radar Ocean Reconnaissance Satellite (RORSAT) missions.

Implementation of the core pusher resulted in the creation of a significant amount of space debris from the NaK coolant and fuel elements. It is unclear whether a high-temperature refractory nuclear rocket would burnup during reentry even if it were dispersed prior to reentry. The SP-100 program proposed the use of a reentry shield to ensure the reactor remained intact and could be retrieved, but this increased the overall system mass, added mission/operational complexities, and presented challenges relative to verifying the long-term integrity of the structure. The lesson learned is that reentry strategies require an integrated plan that includes consideration of the planned mission and operating space, accidental criticality, reentry exposure, and nuclear material security.

TABLE I. Overview of mission operating space, and proposed reentry strategy.[13]

Program	Proposed Mission Operation	Reentry Options
SNAP	LEO or GEO orbit, planetary, or deep space.	Hot reentry for some missions: boost to higher orbit, burnup or intact reentry.[14]
Rover	LEO or Earth-flyby operation.	High-orbit disposal. Active destruct and intact reentry considered.
SP-100	LEO-GEO, deep space or planetary.	High-orbit disposal. Cold or hot reentry intact with aeroshell.
Topaz	HEO operation with deep space disposal	Proposed cold reactor reentry dispersal.
Prometheus		Cold reactor reentry dispersal.
Kilopower	HEO, planetary, or deep space operation.	To Be Determined
NTP	HEO, LEO, medium Earth orbit (MEO) operation	Cold reentry dispersal.

V. POSSIBLE SAFETY REQUIREMENTS FOR REENTRY

V.A. General Design Criteria

General Design Criteria (GDCs) were used in all space reactor programs after SP-100, including Topaz II and Prometheus. GDCs provide guidance to designers and reflect good engineering practice. After review of previous programs, including the GDCs developed by Al Marshall for Topaz II,[15] the following are suggested GDCs for space reactor reentry.

Planned radiologically hot reentry shall be precluded from mission profiles.

For any credible radiologically hot reentry accident, the reactor fuel shall reenter essentially intact, or alternatively, shall result in essentially full dispersal as vapor or fine particles of radioactive materials at high altitude.

For the second GDC above, credible is recommended to refer to accidents with a likelihood greater than 1E-6.

V.B. Risk Criteria

A recently issued Presidential Memorandum specifies risk criteria for nuclear launches.[16] That memo states that authorities should ensure that:

the probability of an accident resulting in exposure in excess of 25 rem TED to any member of the public does not exceed 1 in 100,000

The Presidential Memorandum is believed to apply to an entire mission, that is, the sum of all possible accidents yielding 25 rem should not exceed 1 in 100,000. References 1 and 13 proposed risk criteria that could be applied to inadvertent reentry. Those criteria are more conservative than the presidential memorandum and separately address the probability of reentry, the conditional probability of criticality, and the doses that would result from an Earth impact. While no specific criteria have been set in the Presidential Memorandum for reentry alone, it makes sense that the reentry risk must be less than the allowable total risk. A guiding philosophy here, continued from Reference 13 is that both the likelihood of inadvertent reentry and the consequences should be addressed in the criteria, thus maintaining an element of defense in depth that is traditional in nuclear safety.

V.B.1 Likelihood of Inadvertent Reentry

As noted in Reference 1, we have conservatively recommended that the likelihood of an inadvertent reentry be $\leq$1E-4 over the mission life. This probability is summed over all credible accidents. Credible is not defined in the Presidential memorandum, but it is suggested here that accidents with probabilities greater than 1E-6 be considered credible. If the probability of reentry can be shown to be less than 1E-6, as was done for the Cassini flyby,[17] then the reentry can be considered "incredible," and thus consequence calculations are not necessary.

Reentry can occur with a reactor in a number of different states:

- Shut down and cold,
- Shut down and hot, or
- Operating and presumed hot

Reference 13 provided evidence that cold reactors represent little risk to the public unless criticality occurs.

Therefore, cold reactors can be excluded from further analysis if the probability of criticality is sufficiently low.

The definition of a "hot" reactor is somewhat arbitrary. Reference 10 indicates that reactors should decay down to the level of the actinides prior to reentry. That level can be different for different fuel types. A reactor that has never operated will probably contain less than 100 Ci of radioactivity. A radioactivity limit in Curies could be the basis for the definition of "hot," e.g., 1,000 Ci. Another possibility consistent with the Presidential Memorandum would be to show that, for a bounding calculation, the resulting dose to the public from an intact reentry would be less than 25 rem.

For many missions, the most likely impact site will be in an ocean. In that case, doses to the public will be effectively zero, with or without criticality, unless the site is adjacent to the shore. From purely a public safety standpoint, ocean impacts are not of concern; however, if recovery of the reactor is desirable, then an ocean impact can be problematic. There are approximately seven nuclear submarines, U.S. and Russian, that were sunk without recovering their reactors due to the difficulty of the recovery or the depth of the site. Because public risk is eliminated, it is recommended that ocean impacts be considered successes with respect to the risk criteria in the Presidential Memorandum. This is significant, because requirements to preclude criticality in water can significantly impact reactor design.

V.B.2 Reentry Considerations for Mission Types

Overall failure probabilities can be developed by combining a number of probabilities for particular events that might occur for a given mission. These might include the probabilities of:

- Reentry during ascent and orbital insertion
- Reentry from orbit
- Reentry during or after a departure burn
- Reentry during a fly-by
- Reentry during a return-to-Earth mission
- Reentry during disposal phase
- Land impact
- Hot or critical reactor during reentry
- Criticality upon impact

For certain missions, such as a space tug, events may occur multiple times during a mission lifetime and the probabilities must be added together.

V.B.3 Consequences of Inadvertent Reentry

Reference 1 discussed the need to localize the consequences of a space reactor reentry, for both safety and possible security concerns. Scattered reentry is highly undesirable for a number of reasons, including the difficulty of retrieval as was the case for Cosmos 954.[18] Therefore, it is recommended that a scattered reentry be largely precluded by requiring the combined probability of inadvertent reentry plus scattering to be less than 1E-6. For intact reentry, some scattering of radioactive components may occur due to breakup at the impact site, and such scattering should be confined to an impact zone with a radius of less than 1 km.

For a successful intact reentry of a hot reactor, radiation doses may be very high adjacent to the reactor as noted previously. Doses at a distance will be small unless there is a driving force sufficient to disperse the fission products, e.g., to cause melting or vaporization. Such a driving force can come from the reactor's own decay heat if not precluded by design or from criticality upon impact. The Presidential Memorandum does not specify a distance from the reactor for calculating a 25 rem dose. Per the guidance in Reference 1, it is recommended that a distance of 1 km be used for the calculation.

VI. Mission Implications

It is important to consider safety criteria early in the mission design process. Missions may be impacted in a variety of ways, including payload mass and configuration, orbital altitudes, and the ability to perform near-Earth operations. While no particular missions are intended to be precluded, certain missions are likely to require more restrictive safety measures.

For orbital missions, including Earth departure and return missions, the likelihood of inadvertent reentry decreases with altitude. In addition to altitude, the probability of inadvertent reentry can be reduced through increased reliability and redundancy of control systems, as well as greater resistance to micrometeoroid impact. Such design changes may increase the system mass.

For Earth-flyby missions, the trajectory biasing approach used in Cassini reduced the probability of Earth impact to acceptable levels. That approach requires more propellant to carry out the multiple trajectory changes. This approach may also apply to return-to-Earth orbit scenarios. Trajectory biasing may increase mission times for flyby or return-to-Earth scenarios.

If reentry is to involve complete burnup, then reactor and fuel design will be important, possibly including active means to eject and disperse the core. The velocity and angle of reentry will also be important. If reentry is to be intact, then an aeroshell may be required, impacting both reactor and spacecraft design and the overall mass.

Consequences can be managed through reactor and spacecraft design, reentry behavior or reducing the fission product inventory. The reactor design determines the number of fission products that might be released during a criticality event upon impact and also influences the reentry behavior. Design changes may impact the mass of the reactor system and/or spacecraft. Operationally, a spacecraft in a higher orbit will have greater time for decay of radionuclides in some scenarios. Intentionally moving a spacecraft to a higher orbit at the end of life requires additional propellant. For Earth-flyby scenarios, the reactor could remain off prior to the flyby. For return-to-Earth scenarios, operational profiles that minimize the radionuclide inventory prior to Earth approach could be considered. These measures will affect mission times and capabilities and require carrying additional propellant to perform needed maneuvers.

VII. CONCLUSIONS

In summary, general design criteria and risk criteria have been suggested for reentry accidents. The likelihood of inadvertent reentry should be kept as low as possible, but if reentry is to occur, either burnup or intact reentry is preferred over scattered reentry. Mission profiles may be significantly affected by the need to address reentry safety. Reentry safety should be considered early in the design process to avoid major design changes and/or adverse mission impacts.

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CONSIDERATIONS FOR IMPLEMENTING PRESIDENTIAL MEMORANDUM-20 GUIDELINES FOR NUCLEAR SAFETY LAUNCH AUTHORIZATION FOR FUTURE CIVIL SPACE MISSIONS

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National Security Presidential Memorandum-20 (NSPM-20) (Launch of Spacecraft Containing Space Nuclear Systems) [1] dated 20 August 2019 provides updated guidelines for launch authorization for three categories of proposed launches of spacecraft with space nuclear systems: Federal Government civil space including NASA, Federal Government defense and intelligence, and commercial. These space nuclear systems provide power, heat, and/or propulsion to the spacecraft.

NSPM-20 states: "For United States launches of space nuclear systems, the Federal Government must ensure a rigorous, risk informed safety analysis and launch authorization process" [1], primarily by examining the probabilities of potential launch and reentry accidents and their consequences. At the same time, for previous NASA missions, the launch approval process "has taken an average of six years and costs over $40 million" [2]. In an effort to streamline the process, and improve cost and schedule, NSPM-20 provides specific guidelines including the following: (1) "to the extent possible, safety analyses and reviews should incorporate previous mission and review experience" [e.g., Environmental Impact Statements (EISs), Records of Decision (RODs), Safety Analysis Reports (SARs), and Safety Evaluation Reports (SERs)], (2) "demonstrate that the mission is within the safety basis envelope established in the system-specific SAR, in which case it is not necessary to repeat the analysis supporting the system-specific SAR," and (3) "authorization for launches of spacecraft containing space nuclear systems shall follow a three-tiered process based on the characteristics of the system, the level of potential hazard, and national security considerations" (i.e., use risk-adjusted metrics for required level of effort and launch authorization authority).

A future example interplanetary mission (EIM) that plans to use a Multi-Mission Radioisotope Thermoelectric Generator (MMRTG) is covered by NSPM-20, and is used here as a proxy to illustrate potential considerations for implementing NSPM-20 guidelines. Assume that this EIM plans to use one or two Earth Gravity Assist (EGA) maneuvers in its mission trajectory and that it will use launch vehicle (LV) stages with solid propellant. Its LV will definitely have a flight termination system (FTS).

This paper investigates each of these three NSPM-20 guidelines for three accident categories associated with the EIM: (1) EGA (aka Earth flyby or Earth swing-by) reentry, (2) solid propellant fires, and (3) FTS functions and probabilities. This paper also identifies the components needed to implement each guideline in a rigorous fashion, then assesses whether the necessary components (e.g., analyses, reports, tests, reviews, risk communications, previous launch approvals) currently exist or would need to be produced or modified.

Although these NSPM-20 guidelines could be logical and appropriate approaches for evaluating the risk associated with a system-of-systems (i.e., launch of nuclear systems) that has reached steady-state, the current state of affairs for the EIM is likely still in the "start-up transient" phase. For example, past EISs and SARs for each successive mission were constantly updated with new test data, new technology, new knowledge, and new understanding, such that previous risk results could change [3]. Additionally, past SARs proactively considered review comments and findings from past SERs. Therefore, one potential future side effect of the proposed cost improvement approach is the stagnation of technological progress in nuclear safety analyses.

Because there are no cases of launch authorization of commercial launches of nuclear systems, and no previous unclassified detailed guidelines for launch authorization of defense or intelligence launches of nuclear systems, this paper refers to relevant NASA missions.

I. THREE ACCIDENT CATEGORIES

The three specific NSPM-20 guidelines for the three EIM accident categories are discussed next.

I.A. Earth Gravity Assist Reentry Accident

The last NASA radioisotope power systems (RPS) missions with EGA maneuvers were Galileo (1989 launch) and Cassini (1997 launch). Galileo successfully flew a Venus-Earth-Earth-Gravity-Assist (VEEGA) mission trajectory, and Cassini successfully flew a Venus-Venus-Earth-Jupiter-Gravity-Assist (VVEJGA) mission trajectory.

I.A.1. Previous Mission Analysis and Review Experience

Two issues are identified for direct use of the Cassini EGA accidental reentry analytical results: (1) review comments and (2) different designs. These are discussed next.

From the Interagency Nuclear Safety Review Panel (INSRP) Reentry SubPanel (ReSP) report [4]: "The ReSP found that the FSAR [Final Safety Analysis Report] results are not representative for the [VVEJGA] reentry phase. The subpanel found that the FSAR's analysis was non-conservative [i.e., favored survival of the General Purpose Heat Source (GPHS) / Graphite Impact Shell (GIS)] for three primary reasons: (1) an error in the surface energy balance for carbon ablation, (2) a non-conservative bias on heat transfer rates and an underestimate of aerodynamic heating uncertainties, and (3) the use of a GIS orientation favoring survival after release from the aeroshell. The subpanel estimates that the net effect is a factor of 20 increase in the mean number of GISs failing at altitude during a VVEJGA reentry. The ReSP analyses indicate that on average (integrating over all possible reentry path angles), VVEJGA reentries will result in the release of 19.9 kg of [plutonium dioxide or plutonia] fuel at 24 to 40 km altitude, with 8.5 kg of this reduced to respirable size."

After the 1997 Cassini launch, JHU/APL further investigated Earth atmospheric reentry physics in response to the ReSP comments. Several prominent national researchers were consulted, but disagreed on the level of uncertainty in radiation heating, the surface energy balance, and the wall boundary condition. The extreme nature of the aerothermal analyses and the lack of supporting test data or research in this regime made it difficult to substantiate the theories for the Cassini mission [3].

The Cassini FSAR analyzed the Step 0 GPHS module. Later efforts to improve the thermostructural integrity of the GPHS module to EGA reentry aerothermal loads led to first the Step 1 module design (as used on New Horizons' RPS), and then the Step 2 module design (as employed in the MMRTG). It is interesting to note that the same analysis methodology used in the FSAR, with ostensibly the same three shortcomings identified by the ReSP, was used to redesign the GPHS module and assess its survivability. Considering the issues identified with the analysis, the different module designs, and possibly different reentry conditions, using the Cassini FSAR to assess the response of the MMRTG's GPHS modules to a hypothetical EIM EGA reentry would not be straightforward.

I.A.2. System-specific SAR

Previous analyses predicted that accidental EGA reentry would produce severe environments: if a reentry had occurred for the Galileo mission, the estimated hypothetical reentry velocity was 14.3 km/s (47,000 ft/s), with heat fluxes on the GPHS approaching 204 MW/m^2 (18,000 Btu/ft^2-s), and surface temperatures greater than 4400 K (7500°F) for a steep reentry angle of -90° (Lucero 1994). For the Cassini mission, higher momentum needs translate into a higher estimated Earth flyby velocity of 19.5 km/s (64,000 ft/s). New techniques predict maximum heat fluxes on the GPHS of 397 MW/m^2 (35,000 Btu/ft^2-s), and maximum surface pressures of 0.9 MPa (131 psi) (Bhutta et al. 1996). Maximum decelerations are expected to be about 9800 m/s^2 (1000 g's). [5]

To confidently establish a safety basis envelope of response to EGA reentry, testing is probably the best way. However, it is not possible to properly test GPHS modules to these environments (e.g., shear force, cold wall heat flux, deceleration) in existing ground-based test facilities [14]. Using now 25- to 50-year-old aerothermal and thermostructural technology, the GPHS modules were predicted to survive for Galileo and to be breached for Cassini. However, review by the ReSP found that the analysis presented in the Cassini FSAR was non-conservative for the three reasons previously stated.

Given the ReSP's review findings of the FSAR's analysis, it would be difficult to ascribe failure thresholds or boundaries based on the FSAR's results. Therefore, it is unlikely that a safety basis envelope of the Step 2 module to EGA reentry loads can be established in a system-specific SAR either via testing or analysis.

The Cassini ReSP concurred with the separate lightweight radioisotope heater unit (LWRHU) FSAR's estimates of fuel release for VVEJGA reentries. These findings were based on analyses assessed to include several conservative assumptions. The Cassini ReSP also noted that a VEEGA reentry would substantially reduce the predicted number of failed GISs and LWRHU failures [4].

Based on these facts, EGA reentry aerothermal analysis should be conducted for the EIM SAR. The probability of an EGA reentry accident should be reduced to below 1E-6 by mission design, as was done for Cassini.

I.A.3. Three-tiered Process Based on Risk-adjusted Metrics

From the Safety Evaluation Report [6]: "The Cassini INSRP used an internationally accepted risk coefficient of 5% per Sv (0.05% per rem) to determine latent cancer fatalities [ICRP 60]; and by model extrapolation and inference, one latent cancer fatality per 20 person-Sv (2,000 person-rem). Thus, an Earth Gravity Assist (EGA) reentry accident, spreading plutonium over most of the Earth, would deliver a calculated mean 50-year collective dose of about 30,000 person-Sv (three million person-rem), and a hypothetical associated "collective cancer risk" of about 1500 latent cancer fatalities. However, the average individual 50-year dose would be about 10 microSv (1 mrem), a total dose each person repeatedly absorbs each day from natural background sources such as from inhalation of terrestrial radon daughter products, radionuclides whose emissions are quite similar to plutonium's alpha particles."

NSPM-20 states that Tier III parameters apply to the maximally exposed individual, not the average individual, at a probability greater than 1E-6. From the previous Cassini INSRP quote, the maximally exposed individuals would have suffered latent cancer fatalities. However, given the other EGA analysis and review issues described earlier, it is not conclusive that this quote implies that Tier III applies to the EIM.

I.B. Solid Propellant Fire Accident

Solid propellant fire accidents were analyzed for the Mars Pathfinder, Cassini, Mars Exploration Rovers (MERs) A and B, New Horizons (NH), Mars Science Laboratory (MSL; 2011 launch), and Mars 2020 missions, with progressively more test data, analyses, and knowledge and understanding [3] [7] [8].

I.B.1. Previous Mission Analysis and Review Experience

The Power Systems Working Group (PSWG) of the MSL INSRP in 2009 provided comments on the MSL FSAR, including the following: "The PSWG observes that the situations where large amounts of [plutonia] fuel are exposed to the solid propellant fire environment result in the largest biologically effective releases by far, and dominate the mean source terms, mean health effects, and mean mission risk more than any other single factor… Thus, the PSWG believes that the behavior of plutonia and MMRTG components in solid propellant fire environments should be the subject of a significant research and even experimental program prior to the next RTG [Radioisotope Thermoelectric Generator] launch in order to improve the accuracy of these release predictions and improve the accuracy and relevance of all related risk assessment results."

Indeed, following this recommendation, two independent solid propellant fire testing campaigns were conducted, but under different conditions. It was found in one-to-one comparisons that solid propellant fire characteristics tested in high-altitude vented chamber (HAVC) conditions were less intense than those tested in sea-level open-air (SLOA) conditions; example comparisons are typical measured temperatures of 2500 K vs. 2800 K, respectively, and heat fluxes of 0.85 MW/m^2 vs. 2 MW/m^2, respectively [8]. Note that the melting point of the iridium clad protecting the plutonia fuel pellets in the GPHS module is 2739 K. Test data, fire characterization, and environmental specifications under SLOA conditions are more representative of launch accident conditions at Cape Canaveral, FL, and were used in the SARs for MER, NH, and MSL.

I.B.2. System-specific SAR

Plutonia surrogates and iridium were exposed to solid propellant fires in the JHU/APL tests under SLOA conditions, and their responses were measured and characterized [3] [7], including new findings and measurements: iridium melted, mass of plutonia surrogates lofted into the air, gaseous plutonia surrogate chlorides formed in the plume, and fluorescence was detected (plutonia surrogates displace alumina) in aerosols. However, there are two issues: (1) intact MMRTG, GPHS, GISs, and fueled clads were not tested in solid propellant fires and (2) plutonia surrogates' thermochemical properties and responses are similar but not identical to plutonia. Based on test results and analyses to date, it is likely that a safety basis envelope for fueled clads and bare plutonia will be lower than measured solid propellant fire environments.

Responses of the intact MMRTG, GPHS, GISs, fueled clads, and plutonia should be evaluated analytically for the EIM SAR, plutonia or plutonia surrogates should continue to be tested in solid propellant fires, the intact MMRTG, GPHS, GISs, and fueled clads should be tested in solid propellant fires, and test conditions should be similar to those of Cape Canaveral.

I.B.3. Three-tiered Process Based on Risk-adjusted Metrics

It could reasonably be assumed that the risks of the EIM due to solid propellant fire accidents would be similar to the risks for the MSL and the Mars 2020 missions. The magnitudes of these risks can be found in the respective MSL and Mars 2020 safety documents. The MSL Final Environmental Impact Statement (FEIS) [9] gives an overall radiological risk of 9.11E-4. However, the Mars 2020 FEIS [10] gives an overall radiological risk of 2.9E-5, a factor of 31 lower, for essentially the same MMRTG launch configuration. The factor of 31 is outside the estimated uncertainty bands of x/25 and 25x [11]. An uncertainty analysis was planned for the Mars 2020 FSAR.

The recently released Final Supplemental EIS for the Mars 2020 Mission [12] revises the overall radiological risk for the Mars 2020 mission estimate to be 4.9E-4, approximately 17 times higher than estimated in the 2014 FEIS, and more in line with that of the MSL FEIS. The Final EISs and Supplemental EISs are public documents.

The MSL FSAR gives an overall risk similar to the MSL FEIS. Mars 2020 has not yet finished its launch approval process, whereas MSL has, and has launched. The Mars 2020 FSAR and SER should be reviewed when available, and their risk results compared to those in the MSL FSAR and SER.

Given this information, it is appropriate to use the earlier MSL FEIS results. The MSL FEIS states: "For very unlikely events involving ground impact of the entire launch vehicle or parts thereof, with a total probability of release ranging from 1 in 11,000 to 1 in 830,000, the maximally exposed individual could receive a dose ranging from a fraction of a rem up to about 30 rem."

This would be within the NSPM-20 Tier III parameters, which states "Tier III shall apply to launches of any spacecraft containing a space nuclear system for which the associated safety analyses determine that the probability of an accident during launch or subsequent operation resulting in an exposure in excess of 25 rem TED to any member of the public is equal to or greater than 1 in 1,000,000." Note that the probability of an accident is greater than the total probability of release. Given the assumption stated at the beginning of this section, Tier III parameters might be applicable to the EIM as well.

I.C. FTS Functions and Probabilities

I.C.1. Previous Mission Analysis and Review Experience

The LV's FTS provides the capability, in the event of an early launch or flight anomaly, to terminate thrust, disperse the propellants, and prevent LV fragments from threatening populations or property as they fall back to the Earth's surface. Crucial aspects of the design and analysis of an FTS include timing and relative timing for some obvious, some subtle, and some non-obvious reasons.

An example of an obvious reason is that the FTS should be activated, either via commanded or automated means, prior to the possibility of the whole or parts of the LV falling back to Earth and causing potential harm to life or property.

An example of a subtle reason is on the breakup system of the NH third-stage rocket motor, where the detonation of the aft destruct charges were designed with a time delay of milliseconds so as to preclude the possibility of its blast effects prematurely severing the detonating cords that carry the destruct signal to the newly-added forward destruct charges.

An example of a non-obvious reason is an accident scenario where the LV destructs on or near the launch pad. If the destruct charges on the LV's strap-on solid rocket boosters (SRBs) ignite, large solid propellant fragments from their upper domes could be propelled vertically upward and then fall back on top of the spacecraft and/or MMRTG, with potential impact and crushing effects. Sandia National Laboratories (SNL), in developing the nuclear risk assessment (NRA) for the Mars 2020 mission's EIS [11], postulated that the impulse generated by a near-concurrent detonation of the central common core booster (CCB) could deflect these propellant fragments radially outward during their fallback, thereby lowering the probability of impacting the spacecraft and MMRTG [13]. In their memorandum, SNL noted there was "considerable uncertainty" in their calculations. Among these are the facts that (1) the detonation signals travel along the destruct lines at detonation velocities (4 to 10 km/s), not light speed (300,000 km/s), (2) the liquid propellant and liquid oxygen need time to mix to develop full impulse, and (3) the location of the CCB's so-called center of explosion changes and is dependent on the degree of mixing of the propellant and oxidizer. SNL used their lower calculated probabilities in the NRA, with the intent of requesting more rigorous assessment of this accident scenario and probabilities for the FSAR.

I.C.2. System-specific SAR

As alluded to in Section I.C.1, the FTS functions and probabilities can have significant effects on the subsequent disposition of the LV and its components, including possible RPSs. Past missions such as NH, MER A and MER B had their FTSs modified to account for the close physical proximity of a solid propellant stage to the RPSs, by incorporating a "breakup system". Also during the NH mission development, a new version of the FTS operating software was implemented. These examples show that some past missions had customized FTS features, so it would be difficult to provide a system-specific SAR for future FTS functions and probabilities.

I.C.3. Three-tiered Process Based on Risk-adjusted Metrics

The FTS functions and probabilities are a balance between safety and launch execution. Hypothetically, one could "tune" an FTS to improve safety and reduce risk, but at the expense of lowered probability of launch success. For instance, one could lower the threshold or criteria for automatic or commanded activation of the LV destruct charges. Discussion of any specific FTS design features and tradeoffs is beyond the scope of this paper.

II. CONCLUSIONS

The risks associated with three accident scenarios were documented in previous nuclear space missions' EISs and SARs. Review comments on the EISs were provided by the NASA Associate Administrator for the Science Mission Directorate (NASA/SMD/AA) in the ROD, and on the SARs by the INSRP in the SER for each mission. Decisions for each project's continuation were then provided by the NASA/SMD/AA in the ROD, and the Director of the Office of Science and Technology Policy in a launch approval letter to the NASA Administrator, respectively, for each mission.

Relevant analyses and reviews for EGA reentry accidents are the Galileo and Cassini missions. Relevant analyses and reviews for solid propellant fire accidents are the NH, MSL, and Mars 2020 missions. Review of the respective EISs, RODs, SARs, and SERs for these missions show that the supporting technology is still in flux and under constant improvement, as described in this paper. These documents' analytical results for these accident scenarios are not directly usable for the launch authorization processes of National Security Presidential Memorandum-20 for the EIM. However, they do provide a firm basis of departure. Establishing system-specific SARs was also found to be in the same situation. Lastly, some previous analyses and reviews indicated that Tier III requirements would hold for the EIM, whereas others were non-conclusive.

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SMALL PROBES FOR A SUBSURFACE OCEAN EXPLORATION MISSION TO EUROPA

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Europa is an icy moon of Jupiter with a subsurface ocean of liquid water. Since water is one of the fundamental ingredients for life, its study has gained a lot of interest within the science community. Recently, concepts for a probe, which could potentially melt though Europa's ice shell, have been discussed in the open literature. To date, these concepts have mainly focused on a single probe design that makes use of a relatively large radiogenic heat source based on plutonium-238. However, due to Europa's relatively low temperature (100 K), low atmospheric pressure (0.1 µPa) and low gravity (g=1.35 m/s^2), smaller probe designs with higher thermal power densities are could play a significant role to help overcome the challenges associated with ice sublimation and refreezing. In this paper, new smaller melting probe designs are examined that make use of radiogenic heat sources with higher thermal power density. Specifically, designs that utilize curium-244 and uranium-232 are assessed. Probes with relatively small lengths (0.20-1.50 m) and small radii (0.06-0.12 m) are considered and compared with different melting velocities and radioisotopes. Different concepts are studied for the communication problem.

I. INTRODUCTION

Jupiter's icy moon Europa is one of the best candidates for hosting some form of biological life. With its surface ice and relatively warm subsurface ocean, it has a large quantity of water. For this reason, it is considered to be the next frontier for future robotic probe missions (Ref. 1).

As a result of the thickness of the surface ice layer (between 5 km and 30 km), the subsurface oceans are not directly accessible. A melting probe has been proposed as one of the more promising technological approaches to breaching the ice sheet. However, there are two main challenges associated with melting through the Europa ice:

1. The first is low pressure on the surface, at 0.1 *µPa,* ice readily sublimates. The water vapour created is more thermally insulating than liquid water, thus demanding more thermal power to melt through to the deeper ice layers. The pressure induced by the water vapour also exerts an upward force on the probe, preventing the probe from successfully penetrating into the ice. This is exacerbated by the fact that Europa has a gravity which is only around 1.35 m/s^2.
2. The second challenge is the low temperature of the first ice layer. At around 100 K, melted water ice rapidly refreezes under no thermal load. Therefore, if inadequate levels of heat are distributed to the lateral sidewalls of the probe, it could result in the probe stalling in the ice.

To overcome these challenges, one could possibly send multiple probes which have relatively small cross-section and length. Additionally, by using radioisotopes with a high thermal power density, it will be possible to produce kilo of watts of thermal power within a small volume and for a small mass.

Using this design philosophy, different mission concepts have been outlined in this paper. For instance, it is envisaged that it will be challenging to send data through the ice from the probe to the lander. This could be solved by using a number of small melting probes distributed through the ice from 5 km in depth up to 25 km in depth (5 km intervals), each with a communication subsystem. In this way it would be possible for the main probe at 30 km to send data using these small communication probes.

II. RADIOISOTOPE SELECTION

Kramer et al. (Ref. 2), proposed the concept of using curium-244 or uranium-232 with power densities of: 2.5 W/g and 4.3 W/g, and similar densities of between 11 g/cm^3 and 12 g/cm^3 (in oxide form) as candidates for such melt probes. These radioisotopes would produce large quantities of thermal energy in a small volume and mass.

^{244}Cm is a by-product of civil nuclear power and in one year, it is possible to produce six kilograms of this radioisotope only in Sweden's nuclear plants (Ref. 3). The short half-life of curium-244 (18.1 years) makes it suitable only for relatively short missions. However, with only a small mass of this radioisotope, it is possible to produce a large quantity of thermal power for the melt probe, thermal subsystems and an electrical power source required to power instruments or probe payload.

For a Europa mission, curium-244 based ceramic radioisotope material, is one of the more suitable options because small probes with high thermal power could be a promising solution for a mission to Europa. Small volumes of curium-244 could produce enough energy to melt ice at different velocities (from 0.2 mm/s up to 1 mm/s) and produce enough thermal and electrical energy for the probe's subsystems. Uranium-232 with its 4.3 W/g of thermal power is another option. It has a greater thermal power output than curium-244.

To the best of the authors' knowledge uranium-232 would need to be obtained as a byproduct of the Th cycle or by neutron irradiation (Ref. 4).

III. FIRST MISSION CONCEPT

An assumption has been made that the probe would need to melt through 30 km of ice (worst case) in 3-5 years in order to reach the ocean and study the possibilities of life on Europa. This does not consider the 6 years (depending on launch dates) required to reach Europa. The data in Table 1 represent different challenges for the mission. The buoyancy force (F*) (*c.f.* Eq. 1) was identified as one of the key considerations when designing a melt probe for this type of mission (Ref. 5).

TABLE I. Europa Environment (Ref. 1).

Europa Environment	Temp. T [K]	Pressure [Pa]	Gravity [m/s^2]
Surface	100	0.1 x 10^{-6}	1.3
30 km Depth	273	~30 x 10^6	

$$F^* = mg - \pi R^2 \rho_L g L \quad (1)$$

ρ_L=water density

g=gravitational acceleration.

To avoid stalling caused by buoyancy, the corrected force (Eq. 1) needs to have positive values, thus probes need a high mass value (*m*) in small volume i.e. a small radius (*R*) and short length (*L*).

The radiation environment is dangerous for the spacecraft on the surface; however, the ice protects the probes. The ice temperature increases linearly from 100 K at the surface to 273 K at 30 km depth. The temperature is lower at the poles than at the equator (Ref. 1).

A large thermal power is needed for a probe for an icy moon mission. On Europa, the heat required to melt ice with medium to high velocities (0.6 mm/s to 0.9 mm/s) is around 7.5 kW (Ref. 2) (in this way is possible to melt 30 km in ~1.3 years). The power requirement could be achieved with only 3.789 kg of curium-oxide (Cm_2O_3). By mass, this means only 1/8 of the amount of fuel is required to meet this power need when compared to PuO_2 (~23 kg) for example.

Table II outlines some of the properties of the first probe used in the calculations.

TABLE II. Probe 1 properties.

Probe properties	Length [m]	Radius [m]	Melting velocity [mm/s]	Mass [kg]
Probe 1	1	0.06	0.9	[50-150]

Equation 2 shows that if the cross-sectional area of the probe (*A*) increases, melting velocity (*V*) must decrease for the same thermal power (*Q*).

$$Q \propto VA \quad (2)$$

IV. RESULTS

The thermal power needed at the head of the probe (QH) increases if the cross-section area of the probe increases, assuming a constant melting velocity (see Fig. 2).

An interval of mass between 50 kg and 150 kg has been studied. The resulting variation in corrected buoyancy force is shown in Figure 2.

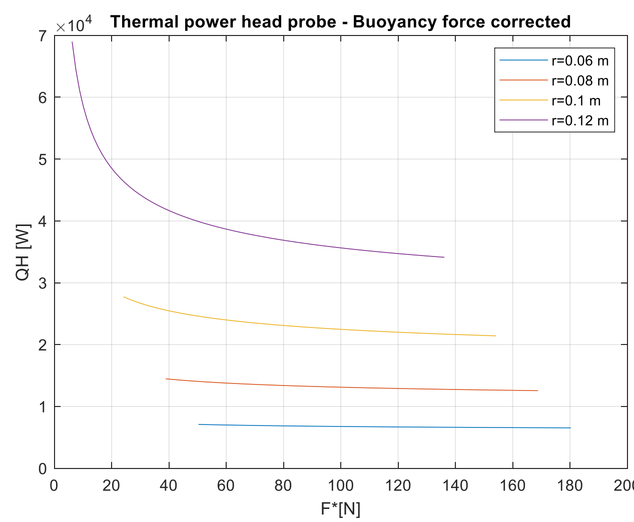

Fig. 2. Probes with different radii are shown considering melting velocity (V=0.9 mm/s) and different values of corrected force. QH is the heat power at the probe head.

Cylindrical probes with short length and small radii appear to be the best solution to reduce the quantity of thermal heat needed and to avoid refreezing, but this leads to having a small volume available for payloads or other instruments.

For 7.5 kW at the head of the probe, the refreezing length is ~0.15 m and the radius is 0.06 m. Additional wall heaters are required to avoid refreezing and a stall. The quantity of lateral heat depends by the melting velocity (*V*), the length (*L*) and the radius (*r*) of the probe (Ref. 5). With high values of these parameters, the heat needed for lateral walls (Eq. 3) increases (Ref. 5)

$$\dot{Q}_L = R^2 V (T_m - T_s) n \sigma^d \quad (3)$$

$$with\ \sigma = \frac{L}{WR^2}, n = 932\frac{Ws}{Km^3}, d = 0.726$$

Tm=melting temperature

Ts=ice temperature

Probe 1 requires 7.5 kW at its head and requires an additional 5 kW for lateral walls to avoid the stall. Thus, the total heat required is 12.5 kW i.e. 6.5 kg of curium-244 in oxide form. In order to reduce the quantity of curium-244 required, it is possible to reduce the probe dimensions and thus the melting velocities.

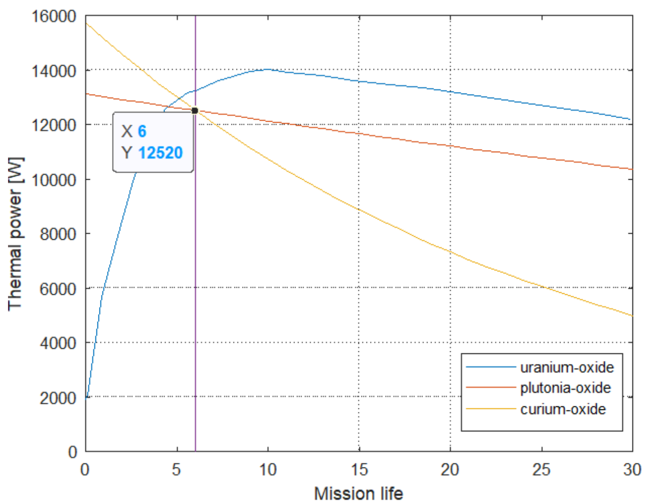

Fig. 3. Plutonium, uranium and curium based ceramic fuels are compared. The graph shows thermal power as a function of time.

In Figure 3 the variation in thermal power output versus time of for the ceramic fuels is shown. Included is the 6-year assumed cruise phase. This allows a mission lifetime sufficient for probe 1 to melt through the ice layer.

TABLE III. Radioisotopes properties

Ceramic Fuel	Power density [W/g]	Density [g/cm^3]	Mass needed Per 6.5 kW (At 6 years) [kg]	Volume filled by RPS (probe1)
Pu-238	0.5	11.5	38	100%
Cm-244	2.5	11.7	6.3	12%
U-232	4.3	12	3.2	5%

From Table III is possible to read the different quantities of radioisotopes needed to produce the required thermal power after the cruise phase. With curium and uranium is possible to use more volume for payloads and other subsystems in the probe.

V. TWO ARCHITECTURES TO ADDRESS THE COMMUNICATION CHALLENGE

A challenge for this mission architecture is communication. The melting probes need to send data to the lander, then the orbiter and, finally, to Earth. However, data transmission through the ice is very difficult transmit information by antenna systems. One solution consists of using small antennas that work in the radio frequency range (100 MHz) each requiring 5 We. The mission would need relays every 2 to 5 km distributed through the ice (Ref. 6). A number of small probes, with integrated communication subsystems, would need to be deployed into the ice layer.

These probes include a radioisotope thermoelectric generator and antenna to communicate. For this situation two different architectures are explored, each described in Table IV and Table V.

TABLE IV. Architecture 1: probe properties

Architecture 1	Length [m]	Radius [m]	Melting vel [mm/s]	Mass [kg]
Main probe	0.50	0.06	0.40	25
Small probe	0.20	0.06	0.06-0.3	10

The first architecture consists of five small probes and one main probe. In order to communicate with the lander, the main probe needs relay probes every 2 to 5 km. Thus, small probes are distributed at increasing depths in 5 km steps with the main probe in the ocean at a 30 km depth. Each small probe has a mass of ~10 kg and each has a different melting velocity by design. The main probe (see Table IV) comprises a communication system and the main payload with multiple sensors. For this design, considering the temperature variation with increasing depth, ~7.8 kg of curium-244 are needed for the entire mission (5.7 kg in total for small probes and 2.2 kg for main probe) or ~3.5 kg of uranium-232.

TABLE V. Architecture 2: probe properties.

Architecture 2 -Probes	Length [m]	Radius [m]	Melting Vel. [mm/s]	Mass [Kg]
Main probe	0.16	0.08	0.40	25
Small probe	0.12	0.06	0.06-0.3	10

Architecture 2 is similar with different melting velocities and different probe sizes. The mission duration is 3.7 years (not considering the cruise phase) and the probes have the same diameters and lengths to improve the stability during the melting phase.

The main probe has an average melting velocity of 0.255 mm/s to reach 30 km in 3.7 years and each small probe has different quantity of curium with different velocities. Only 5.04 kilograms of curium-oxide or 3.3 kg of uranium-oxide are needed for this concept.

VII. CONCLUSIONS

For a melt probe mission to Europa, small probes with high thermal power in the smallest volume are needed. For this purpose, it is possible to use radioisotopes. A number of options have been considered. These radioisotopes include: curium-244 and uranium-232, with respectively 2.5 W/g and 4.3 W/g. A number of probe architectures have been proposed.

A mission concept with multiple small probes deployed into the ice it could provide an option to solve the communication challenge.

Melting the ice with a lower velocity than the concepts considered could be another solution, but the mission time would grow to over 10 years in addition to the 6-year cruise phase.

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A SYSTEMATIC APPROACH TO DEFINING A NUCLEAR THERMAL PROPULSION FLIGHT DEMONSTRATION

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Analytical Mechanics Associates (AMA) was tasked by the National Aeronautics and Space Administration (NASA) to lead a nuclear thermal propulsion (NTP) flight demonstration (FD) study with the support of various industry companies specializing in spacecraft, engine, and nuclear reactor development. Inputs from participating companies were used to define an overall design space with multiple near-term FD concepts to provide NASA with a comprehensive assessment of the FD value proposition in terms of capability versus cost and schedule. In order to ensure consistency and traceability to the FD mission objectives and stakeholder needs, the study leveraged a top-down systematic approach. The results of the study suggest technically and programmatically feasible FD concepts that will effectively demonstrate NTP capability, exercise nuclear regulatory processes, and mature NTP technology.

I. INTRODUCTION

The National Aeronautics and Space Administration (NASA) is currently investigating solid-core nuclear thermal propulsion (NTP) to enable crewed missions to Mars. As demonstrated in over twenty ground tests during the 1950-70s Rover/NERVA program[1], NTP achieves greater architectural robustness compared to traditional chemical systems by utilizing a nuclear fission reactor to heat up liquid hydrogen for rocket thrust. Due to the lighter atomic mass of hydrogen, NTP produces more than twice the specific impulse (Isp) compared to chemical propulsion, reducing propellant mass and enabling faster transit times. Additionally, NTP offers architectural and programmatic flexibility, such as payload mass growth tolerance, higher launch vehicle insertion orbits, wider launch windows, and mission abort options.[2] These capabilities are vital to facilitate crewed missions to Mars.

In the fall of 2019, Analytical Mechanics Associates (AMA) was tasked by the NASA Game Changing Development (GCD) program to lead a meaningful and independent NTP flight demonstration (FD) study with the support of various industry companies specializing in spacecraft, engine, and nuclear reactor development, including Aerojet Rocketdyne, Blue Origin, Boeing, BWX Technologies, General Atomics, United Launch Alliance (ULA), Ursa Major, Ultra Safe Nuclear Corporation (USNC), and X-Energy. Inputs from participating companies were used to define an overall design space with multiple near-term FD concepts. Providing NASA with a comprehensive assessment of the value proposition and system design space enables informed decision making as the prioritization between capability, cost, schedule, and risk evolves. The study recognizes that maturing reactor technology is not the only consideration for a successful NTP FD. The FD must provide demonstrable evidence of the propulsion capability to enable future missions and keep stakeholders engaged. The FD must also alleviate programmatic and regulatory concerns regarding the development, launch, and in-space operations of a nuclear reactor engine.

In order to ensure consistency and traceability to the FD mission objectives and stakeholder needs, the study leveraged a top-down systematic approach analogous to traditional systems engineering derivation. An abbreviated version is given in Figure 1, which will guide the remaining sections in this paper. Although the diagram suggests a serial progression through the work elements, the mission and flight system architectures were iterated upon as capability, risk, cost, and schedule were assessed in order to optimize the FD value as time allowed within the six-month study.

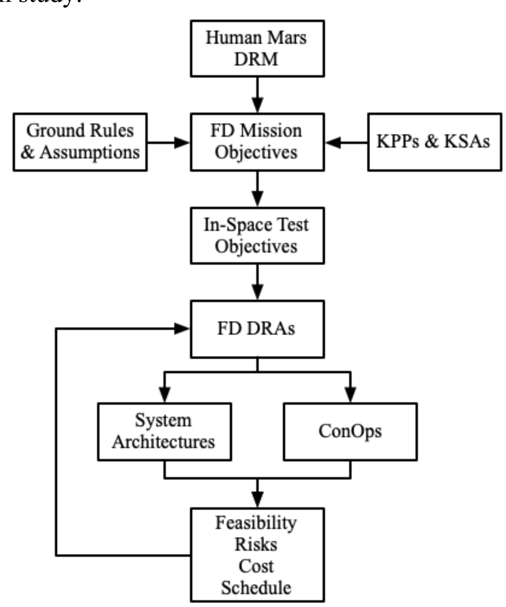

Fig. 1. NTP FD study systems engineering traceability.

II. MISSION OBJECTIVES

Understanding stakeholder expectations provides the foundation upon which all other system engineering work depends. Defining stakeholder expectations begins with identifying the goals and objectives the mission is to achieve.[3] Understanding mission objectives ensures the team is working toward a common vision.

The industry study with concurrence from NASA customers identified the following mission objectives for the NTP FD.

1. **Demonstration of NTP Capability** – An NTP FD provides demonstrable evidence of NTP capabilities for advanced in-space propulsion applicable to a range of future NASA, DoD, and/or commercial missions. The evidence substantiates the performance and feasibility of NTP in an operational environment, with the understanding that the FD mission is only a demonstration and may not represent the full performance or capabilities of an operational engine but is extensible to the operational performance.
2. **Demonstration of Regulatory Processes** – An NTP FD demonstrates the verification processes of federal regulatory requirements for space fission systems. The process of executing a FD provides an initial standard of analysis and reporting to meet current or new regulations that will be leveraged for future systems. This includes the development, launch approval, in-space operations, and disposal of such fission systems.
3. **Maturation of NTP Technology** – An NTP FD will require development of NTP technologies culminating in an in-space demonstration that will provide data that verifies the technology readiness level of these technologies. The FD will be a part of a larger technology maturation plan (TMP) toward an operational NTP engine for a NASA crewed mission to Mars. Technology maturation needs from DoD or commercial entities may inform the TMP but are secondary priorities. The FD technology maturation value (benefit vs. cost/schedule) will be greater than a ground development and testing only approach.

Ultimately, the goal of the FD is to lower barriers for adopting NTP as an enabling technology to get humans to Mars. The first barrier is the lack of demonstrated capability in space. Although NTP was demonstrated via ground testing by the Rover/NERVA program, future mission planners need verification of the capability of a modern NTP engine as a high-Isp in-space propulsion system. Furthermore, the current regulatory environment and lack of available facilities necessitate significant cost and schedule overhead to perform full-scale, full-power ground testing, which could result in loss of interest in adopting NTP. The second barrier is the lack of experience in navigating regulatory approval to transport, handle, and launch a nuclear-reactor-powered propulsion system for use in space. Although NASA has effective procedures for preparing, launching, and operating nuclear systems in space, future space mission planners will face unique regulatory procedures for fission nuclear reactors. The third barrier is not having a clear technology development roadmap to advance NTP systems for crewed mission to Mars applications. Future mission planners need evidence that there is a clear and affordable path for developing and testing this technology.

The value of the NTP FD mission will be reduced if the mission does not work as intended, if the demonstration is ambiguous, or if it is not disposed of in a manifestly safe way. The value of the NTP FD mission will be enhanced if no further flight-demonstration-only missions are needed before operational NTP systems are used in space and ultimately qualified via full-scale, full-power ground testing for crewed missions to Mars.

III. GROUND RULES AND ASSUMPTIONS

Ground rules are the governing principles of the mission. The following ground rules for the NTP FD are based on programmatic constraints and expectations levied by NASA. They only apply to this FD study and do not necessarily reflect ground rules for an operational NTP mission.

1. **Nuclear Regulations** – The project will follow the guidance of the Presidential Memorandum[4] and associated U.S. federal regulations (e.g. NRC, DOE, and DOT) for the development, launch, and in-space operations of space fission systems.
2. **Launch Vehicle** – The selected launch vehicle shall have credible evidence that it will be operational at the time period of the FD launch.
3. **Reactor Operations & Disposal Orbit** – The reactor will only operate and be disposed of at an orbit with a perigee greater than 2000 km altitude above Earth.
4. **Fuel Enrichment** – The reactor fuel will use high-assay low-enriched uranium (HALEU).
5. **Extensibility** – The FD engine will use a nuclear fuel that is ultimately extensible to the fuel for operational NTP engines.

The regulatory environment, particularly with respect to launching of nuclear materials, is currently uncertain. While the Presidential Memorandum released in 2019[4] helps to quantify safety requirements in terms of probability, additional guidance and processes are needed. The approach for this study has been to develop recommendations to NASA with respect to regulatory considerations. NASA will need to monitor new guidance and work with regulators to ensure that the regulatory requirements do not drive the schedule or cause unforeseen complications.

Furthermore, current guidance from NASA restricts fuel enrichment to less than 20% (HALEU) in order to alleviate programmatic and regulatory overhead. Moreover, only reactor concepts with an evolutionary pathway to reach operational performance are considered for the FD. The value of the FD mission is only realized if the NTP fuel and engine control approach are ultimately extensible to the operational system. Developing and demonstrating extensible NTP systems matures the technology, buys-down risk, and proves feasibility.

Assumptions are suppositions used to formulate the mission and system architecture. The following high-level assumptions drive the mission and system design. Additional assumptions were identified to further constrain the design space and associated engineering analyses.

1. **Interdependencies from Other Projects** – The FD project may not depend on other projects and hence other sources of funding in order to close on the FD objectives. Opportunities may arise to "on-ramp" technologies or activities, but the FD project schedule shall not depend on such on-ramps until they become baselined and are absorbed into the project.
2. **Nuclear Facilities Development Schedule** – Any new facilities development for fission ground testing shall not drive the FD schedule (i.e. critical path), and any such facilities will be sub-scale / sub-power level.

Upgrades, modifications, or new facility development will be required in order to fully test and certify the fuels and reactor. The level of testing and amount of facility development required is a trade-off with accepted risk. More capability and reliability testing results in more confidence in the success of the FD mission. This will need to be weighed against schedule constraints in order to develop an effective TMP. Additionally, NASA should outline the development path from the FD mission to a near-term Mars mission including the development of a full-scale, full-power ground testing campaign that may need to occur in parallel to the FD.

IV. KEY PERFORMANCE PARAMETERS (KPP) AND KEY SYSTEM ATTRIBUTES (KSA)

Key performance parameters (KPPs) define the technical performance goals essential to satisfying the technology demonstration. A threshold value is established for the minimal acceptable performance, and a goal value is specified as the intended value to be achieved.[3] Key system attributes (KSAs) are lower priority attributes of the system that are critical for an effective demonstration. Many factors are involved in setting KPPs and KSAs including current state-of-the-art and future applications.

The primary application for the NTP FD is the NASA crewed mission to Mars. Table I captures the KPPs and KSAs for both the Mars mission and the FD mission as well as the rationale for the FD threshold values. These values are extensible to and establish the feasibility of the primary operational application and are used to inform the FD system design and concept of operations (ConOps).

TABLE I. NTP Key Performance Parameters (KPPs) and Key System Attributes (KSAs).

Metric	NTP Mars Mission	NTP FD Mission	Rationale
Key Performance Parameters (KPPs)			
Isp	> 900 sec	> 700 sec, > 900 sec preferred	A 900 sec Isp is minimum to close an opposition-class Mars mission. A 700 sec Isp demonstrates capability beyond chemical systems and represents an advance that indicates extensibility to 900 sec.
Thrust	15,000 – 25,000 lbf	5,000 – 25,000 lbf	A thrust of ~5,000 lbf is the minimum anticipated for sustained criticality that could reasonably scale to a Mars mission. Although higher thrust demonstrates operational engine performance, lower thrust system provides longer burn durations and restarts, which is important for demonstrating and characterizing reactor operations.
Key System Attributes (KSAs)			
Reactor Mass	~ 4,500 kg	< 4,500 kg	Optimize mass based on cost, schedule, and risk while still proving feasibility of operational engine.
Burn Duration (Single/Lifetime)	15 min / 45 min	Maximize	Maximize duration that reactor is at steady-state criticality to demonstrate reactor operations and extensibility to a Mars mission.
Number of Burns	~ 6	> 2	Two burns are the minimum to demonstrate restart capability. Additional burns demonstrate controllability and reliability but are limited based on available propellant, risk tolerance, etc.
Startup/Shutdown Duration	35 sec / 30 sec	< 45 sec	Minimize to maximize effective Isp, minimize propellant loss, and demonstrate reactor controllability. Potential to vary across ConOps.

V. TEST OBJECTIVES

The test objectives define the mission ConOps and identify the technology maturation benefit of the FD. Figure 2 shows the traceability of the flight test objectives to the overall mission objectives.

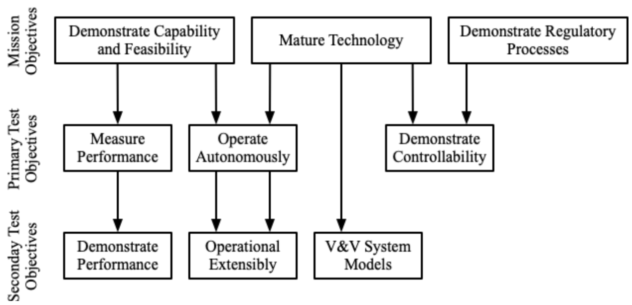

Fig. 2. NTP FD study test objectives traceability.

The first primary test objective is to **measure system performance** in order to prove the capability and feasibility of NTP engines. The is achieved by measuring Isp and thrust during a full-power, full-thrust burn with multiple, independent data sources. The second primary test objective is to **operate in-space autonomously** by performing a complete burn cycle and restart using reliable control systems. The final primary test objective is to **demonstrate reactor controllability** robustness and characterize reactor kinetics based on propellant inlet and outlet conditions, control drum worth, and the neutron flux spectrum.

Secondary test objectives include performing burns at the rated Isp and thrust to **demonstrate system performance**, collecting telemetry to **verify and validate (V&V) models** of the NTP system to further mature the technology, and performing multiple burn cycles to **demonstrate operational extensibility** to the Mars mission. Additional test objectives that are likely to be achieved based on available propellant and instrumentation include characterizing reactor integrity by monitoring plume inventory and changes in performance; understanding nuclear heating of engine components and propellant; characterizing fuel element heat transfer; demonstrating engine throttleability; and demonstrating engine operations at low-power extensible to an orbital maneuvering system (OMS) or bi-modal power system. Furthermore, there is the potential to demonstrate reactor safety robustness by simulating off-nominal events and conditions including a fast restart, stuck control drum, high/low propellant flow rates, emergency shutdown, etc.

VI. MISSION AND FLIGHT SYSTEM DESIGN

Unlike NASA exploration and science missions, technology demonstrations focus on developing and maturing transformative space technologies to enable future missions. Therefore, the mission and vehicle architectures need only to achieve that which satisfies the mission and test objectives.

Weighted decision matrices aid in quantitatively evaluating architectural options with competing criteria such as performance, cost, and risk. This method was used to inform many decisions within the NTP FD design space, including the mission destination, vehicle architecture, and engine/reactor designs.

Given that all mission and test objectives can be satisfied by any mission destination, cost, schedule, and risk result in the highest weighted criteria. Operating in Earth orbit scores the highest since it significantly reduces mission cost and complexity. Other missions such as escape trajectories and abort scenarios do add value by demonstrating capability but score lower given the lower weight on strategic value and extensibility.

Many vehicle architectures were considered that span the mission value space in terms of capability versus cost, ranging from modifying existing architectures such as launch vehicle upper stages and large satellite buses to developing a custom spacecraft that maximizes capability (i.e. available propellant). By spanning the design space while maximizing value, NASA can down-select based on desired capability or cost. Additionally, each vehicle concept has associated risks. For example, the modified upper stage concept benefits from available hydrogen propellant loading capability but introduces a higher risk of launch failures due to the modification of the upper stage. Given the nuclear payload, launch failures are a significant risk that must be sufficiently mitigated.

Similarly, the reactor designs also span the mission value space in terms of performance, feasibility, and extensibility versus cost, schedule, and risk. Although NASA prefers to demonstrate an operational Isp above 900 sec, providing more near-term reactor designs facilitates a lower-cost and reduced-schedule FD that is still extensible to the operational system. Additionally, ensuring feasible designs that reduce operational and programmatic risks is critical to realizing a successful FD mission. See Kelsa Benensky's paper[5] for more information on the metrics used to trade the reactor designs.

A significant challenge for this study was developing technically feasible concepts given the limited time and communication among industry participants. Furthermore, the companies had varying levels of experience with NTP and varying ability to assess the system elements to which they interface. To enable the closure of integrated NTP system designs, AMA provided target performance parameters and interface value ranges representative of the best parings between engine and reactor concepts. The key interfaces provided are given in Figure 3. Given that reactor development is more technically challenging than the non-nuclear engine aspects, the interface definition was intentionally constructed to avoid over constraining the reactor designs, while still ensuring integrated system feasibility.

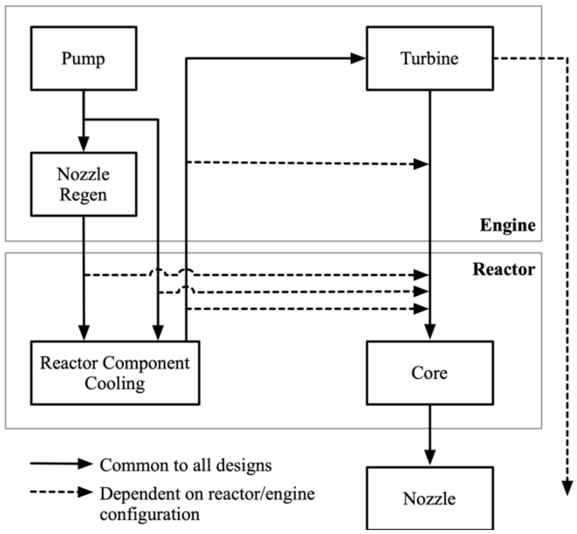

Fig. 3. Generalized engine/reactor interface diagram.

VII. IN-SPACE CONOPS & OPERATIONAL RISKS

The in-space ConOps defines the NTP burn profile throughout the FD mission. The test objectives prioritize demonstrating reactor control and buying down risk by performing zero-power reactor tests to characterize criticality; longer, controlled startups to characterize reactivity; and multiple restarts versus longer duration burns to characterize reliability and robustness. Many burn ConOps were developed that consider risk tolerance, reactor/fuel readiness, and available propellant, while still ensuring satisfaction of all primary test objectives. Given that the NTP FD will require an accelerated technology development phase that relies heavily on digital analysis and modeling to verify reactor material properties, performance characteristics, and control approaches, there is the possibility that anomalous behavior will occur during reactor operations that could result in hardware damage or loss of the reactor. It is critical to the success of the FD mission that all operational risks are identified, well understood, and sufficiently mitigated in order to effectively demonstrate NTP feasibility and prevent development setbacks.

VIII. COST, SCHEDULE, AND RISK

In order to close on the value proposition of the NTP FD, the cost, schedule, and risk of the mission must be established. However, the unique programmatic and technical challenges that contribute to the cost, schedule, and risk of the FD and NTP more generally are out of scoop for this paper.

IX. CONCLUSIONS

The results of the industry study suggest technically and programmatically feasible NTP FD concepts that will effectively demonstrate NTP capability, exercise nuclear regulatory processes, and mature NTP technology. The performance, operability, and programmatics for a range of reactor, engine, and spacecraft designs were surveyed and traded to inform NASA stakeholders of the major drivers within the FD mission design space. By maximizing the mission value across the design space in terms of capability and performance against cost, schedule, and risk, NASA is equipped with the information needed to facilitate a successful NTP FD.

The scope of the study was sufficient to close on the value proposition of the FD concepts in terms of overall cost, schedule, and risk; however, additional detailed design is needed to close on the flight system design, particularly for the integrated engine system. Defining interfaces enabled pairing of engine and reactor concepts to ensure feasibility; however, closing on the integrated engine cycle was limited due to time and communication constraints between companies.

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OPERATIONAL CONSIDERATIONS FOR SPACE FISSION POWER AND PROPULSION PLATFORMS

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As consideration of launching reactors into space moves ahead, it is important to consider the issues related to the operation of various types of space nuclear power and propulsion reactors. This paper discusses some of these considerations, including possible human and equipment radiation exposures that might occur during different types of missions and the operational stages within those missions, managing the approach to and working around space reactors, maintaining reactors for long-duration operations, controlling reactors and monitoring their availability and health, evaluating possible reactor accident scenarios, planning for planetary protection due to their operation and post operation decommissioning and disposal.

I. INTRODUCTION

The general applications of a nuclear reactor in space typically include Nuclear Electric Propulsion (NEP), Nuclear Thermal Propulsion (NTP), Fission Surface Power (FSP) and In-Space Nuclear Power (INP). The mission categories to be considered can initially be split into a few general categories as seen in Table I. These mission categories can be further characterized depending upon the specific mission needs and profiles. Some of the variations that can be considered might include purely robotic missions, whether or not the spacecraft includes a human crew or is a robotic mission that could interact with a crewed mission; whether the mission is a single deployment mission or can be considered an outpost or space station mission that could provide multiple opportunities for human interaction with the spacecraft or surface outpost that utilizes the nuclear reactor; or whether or not maintenance and repair activities could be considered for the nuclear reactor.

Any discussion of the operational considerations for space reactors should cover many of the possible interactions between humans and space nuclear power and propulsion systems, but because there are no firm missions defined at this point that could utilize a nuclear reactor it is impossible to cover all of the possible missions and applications for space nuclear power and propulsion that mission planners can envision. The issues and concerns for the different types of missions for nuclear reactors in space applications is discussed along with the potential efforts that can be made to address some of these concerns and considerations.

TABLE I. General categories of space nuclear power and propulsion missions.

Mission Category	Brief General Description	Expected Power Range
Nuclear Electric Propulsion (NEP) Transport Missions	Utilization of a nuclear reactor to produce and supply electrical power to electric propulsion technologies	Greater than 10 kWe
Nuclear Thermal Propulsion (NTP) Transport Missions	Utilization of a nuclear reactor to directly heat a propellant to provide a direct thermal propulsion capability	Greater than 100 MWt
In-space Nuclear Power (INP) for Electrical Power Missions	Utilization of a nuclear reactor to produce and supply electrical power for mission activities and housekeeping for an in-space or orbital mission	1 kWe to greater than 1 MWe
Fission Surface Power (FSP) for Electrical Power and Surface Outpost Missions	Utilization of a nuclear reactor to produce and supply electrical power for mission activities on the surface of a planet or other astronomical objects	1 kWe to greater than 1 MWe

II. RADIATION EXPOSURE

One of the responsibilities of the mission and reactor designers, as well as the mission operations team, will likely be to keep both human and equipment radiation exposures acceptably below the limits for human and equipment radiation exposure. Consideration should be given to the possible pathways through which humans and equipment could be exposed to radiation and radioactive material during transit, operation, shutdown and disposal operations.

II.A. Radiation Exposure Limits

Humans and equipment associated with space exploration and travel will be affected by the natural radiation fields in space and by radiation sources carried aboard a spacecraft or included in a surface outpost operation. NASA has performed research and developed a set of radiation exposure standards for astronauts[1], and these may be applied to future space missions including fission power and propulsion sources. The basis for the standard is related to the planned career exposure for any astronaut that "shall not exceed 3 percent Risk of Exposure-Induced Death (REID) for cancer mortality at a 95 percent confidence level to limit the cumulative effective dose (in units of Sievert) received by an astronaut throughout his or her career." The resulting dose limits depend upon the mission's length, astronaut's age, sex and other considerations. For example, the Effective Dose Limit for a 1-Year Mission for a never-smoking, 40 to 60-year-old male astronaut would range from 0.88 Sv to 1.17 Sv. The Effective Dose Limit for similar aged female astronauts is about 20 percent lower.

Electronic equipment for payloads and system controls, which tend to vary widely depending upon the application, have been developed and tested against the radiation fields that they are likely to see and standards have been developed for radiation exposure for these sensitive instruments and equipment.

II.B. Potential Human Exposure Pathways

Each particular utilization of space fission power likely will bring its own specific potential ways to expose humans and equipment to the radiation emitted from the reactor. Clearly, as particular missions and utilizations of space fission power are envisioned, more specific detail will be applied to those spacecraft and surface outpost designs.

The primary and most expected radiation path for radiation exposure from a space fission power reactor could come from the direct shine of neutrons, betas and photons emitted from the reactor during operation. All of these come directly or indirectly from the fission processes themselves and can be readily shielded to reduce the exposures to manageable levels. Neutrons from the reactor core can interact with many materials and cause the formation of radioactive activation products. When these activation products decay their gamma and beta rays can be transported to the locations where they could cause an increase in the human radiation dose. After shutdown of the reactor, the materials in and around the core will remain radioactive for years to come.

Accidents may open additional pathways to radiation exposure to both humans and equipment. Reasonable consideration should be given to the possible accident scenarios and mechanisms, as well as the driving forces and available energies for distribution of radioactive materials, as well as to any particular local material transport phenomena. Determination of the magnitude of possible radioactive material releases and how might they impact access, rescue, cleanup, and disposal should all be important considerations for determining the potential radiation exposure to crew and equipment from reactor accidents.

III. APPROACH TO SPACECRAFT AND REACTORS

One special design consideration for space reactors could be accessibility to the cargo or surface outpost area while avoiding any high radiation areas. Each space reactor configuration should be designed to provide sufficient shadow shielding in order to protect any human habitation areas or equipment and payload areas. Putting multiple reactors on either a space platform or at a surface outpost could complicate the designs of these missions.

The need to approach a hot reactor by either personnel or robotic systems should be minimized. Key factors to consider include:

- Minimize the time needed to carry out planned operations, e.g., simplify tools and procedures
- Utilize physical controls and barriers to prevent inadvertent entry into a radiation hot zone
- Enable means to construct temporary shielding using in situ materials
- Employ simple dosimetry and warning systems
- Establish safe paths for ingress and egress
- Create decontamination zones.

Radiation monitoring is a normal part of nuclear operations, and it is anticipated that radiation monitors may be needed around reactors to provide warnings of possible problems. The commonly used ALARA (As Low As (is) Reasonably Achievable) principle is appropriate for developing docking strategies that minimize radiation exposure from on-board reactors[2].

IV. MANAGING REACTOR MAINTENANCE

Ideally, manual maintenance requirements for reactors and associated systems should be minimized. To date there have been no crewed space missions involving a nuclear reactor, and un-crewed missions have not allowed for maintenance. Manual maintenance activities may incur a number of risks to astronauts either in space or on a surface. There is increased potential for radiation exposure, along with the normal risks of astronauts performing activities outside a spacecraft or habitat.

The amount of maintenance required, if any, may be very design and mission specific. Both the reactor and the power conversion system, including heat rejection, must be considered. Small, simple designs with few moving parts, such as KiloPower[3], tend to require less maintenance than a large reactor with a complex power conversion system. Similarly, missions to deep space without a crew may not allow for significant maintenance of the reactor.

Certain reactor system components, such as the fuel, reactor internals, and primary cooling system, are likely to have potential single point failures and are not amenable to maintenance activities. Thus, the quality control and design margins may be particularly important for these components. Table II provides a high-level summary of maintenance possibilities for various missions.

TABLE II. Maintenance Possibilities by Mission Type.

	Reactor Core and Passive Reactor Components	Active Primary System Components	Power Conversion or Propulsion System
Deep Space (Un-crewed)	No	No	No
Orbital Missions (Un-crewed)	No	Unlikely, Possibly Robotic	Possible Robotic
Crewed Space Missions	No	Unlikely, Possibly Robotic or after delay	Possible
Surface Power (Robotic Missions)	No	Unlikely, Possible Robotic	Unlikely, Possible Robotic
Surface Power (Crewed Missions)	No	Possible after Delay	Possible

For cases where maintenance is possible, it is important to distinguish between components enabled for planned maintenance and those that have the capability for maintenance. The former is generally to be avoided, as the goal should be to design systems requiring little or no intervention by astronauts during normal activity. On the other hand, the capability to perform maintenance when necessary is consistent with human rating requirements that the crew be able to intervene when necessary to execute the mission or prevent a catastrophic event[4]. Providing the capability for maintenance may add complexity to the design, e.g., by designing for access, and adds mass through the need for tools and spare parts.

Refueling or maintenance of reactor core components is unlikely to be feasible in any space mission due to radiation levels and inaccessibility. Thus, sufficient fuel must be provided for the entire mission. All materials in and around the reactor must be designed for anticipated radiation effects and thermal loads assuming that maintenance may not be possible. Instrumentation should provide sufficient redundancy to allow for failed sensors.

Active components can include reactivity control systems and various types of pumps and valves. Maintenance on the active neutron absorbing part of a control system, such as a control rod or drum, is unlikely to be practical. Drive motors and pumps or valves outside the core region may be more amenable to maintenance.

There are many different designs for power conversion systems, from direct passive thermionic conversion to a variety of thermal cycles and engines. Designs with high reliability and redundancy may require less maintenance and are preferred.

In theory, robotic maintenance could be performed during most missions. Robotic operations can be designed to withstand high radiation environments and eliminate risks involved in human activities. Robotic capabilities can be sent with the mission at a cost of additional mass in robotics and spare parts.

Maintenance on a surface, such as the Moon or Mars, may be necessary for long duration missions. Maintenance on the reactor itself should be avoided, but those on the power conversion system may occasionally be necessary. On Mars, dust storms may occasionally require cleaning of radiators, which might be done manually or remotely. Providing redundancy through multiple reactors may allow time for the failed system to cool off prior to beginning maintenance.

V. REACTOR CONTROL AND MONITORING

All nuclear reactors require constant monitoring and control prior to launch and initial operation, during pre-startup testing, during the various phases of operation including startup, ascent to power operation, steady-state power operation, changes in power level, through shutdown (both hot and cold) and restart, and during the final shutdown and management of the disposal of the reactor system. Additional considerations include the utilization of autonomous and/or remote control of a space reactor, a determination of the need for a continuously occupied and managed control room either on Earth or close to the reactor, and control of multiple reactors on a spacecraft, space station or surface outpost.

In cases where fission reactors are used for propulsion or critical life-support functions, the ability to rapidly change power level or even restart following an unplanned shutdown may be important. Reactor restarts can be affected by the need to control temperature transients throughout the cooling and power conversion systems and by the buildup of fission products, such as Xenon-135, which act as neutron absorbers.

Some space nuclear fission systems may experience long times of shutdown or minimal maintenance power levels such as NTP or NEP reactors used in tug or bus operational modes. FSP reactors might also experience long shutdown times as outposts or spacecraft sit dormant for periods of time. Each of these systems could face risks and uncertainties if a long shutdown is needed.

Autonomous control may be needed for some, if not all, space missions that use nuclear reactors. This probably will be an imperative for any robotic missions. Currently, there is little experience with autonomous control of terrestrial nuclear reactors as these systems always have a dedicated control room and reactor operators that are close enough to the reactors so that any electronic time delays could be minimized. There is limited experience with the design or specific development of autonomous control systems for space reactor applications[5-8].

VI. REACTOR ACCIDENT SCENARIOS

Reactors in space could have the same potential accidents as their terrestrial counterparts and therefore space reactor designs must account for these potential accidents. Examples of the types of accidents that can be postulated include:

- A reactor failure leading to the reactor not providing electricity to critical systems;
- An accident leading to the partial or full melting of the reactor core;
- A reactivity insertion accident leading to thermal shock of the core destroying all or part of the reactor;
- An accident leading to the release of fission products.

Some or all of these accident types may be of concern for the safety of astronauts or completion of a mission. The reactor design must therefore consider all issues for each specific mission, although the importance of each accident type may change based on the mission characteristics. A systematic study of initiating events and failure modes should be undertaken using standard methods such as Failure Modes and Effects Analysis (FMEA) or a probabilistic risk assessment using such tools as fault trees[9]. After failure modes have been determined, the end state or accident progression needs to be calculated. The end state that results may be a fully functioning reactor, a partially functioning reactor, a dead reactor, or fission product release up to full core melt with fission product release. Finally, contingency/response plans need to be developed for what to do if an accident occurs. It is important to note that space reactor accidents can be preventable and their consequences minimized through good engineering, design and controls.

VII. DECOMMISSIONING AND DISPOSAL

Very few requirements or international agreements exist to guide post-operational decommissioning and disposal (D&D). The Outer Space Treaty[10] implies that siting of reactors and waste storage systems should not interfere with planned activities of other nations. Previously, RTGs or other small radioactive sources have been left in place on the Moon or Mars without any particular disposal strategy. For fission power systems more specific and intentional strategies may be warranted.

Not every mission could need to consider D&D. These are missions where the reactor may not pose a future threat to personnel or equipment when left in its current location. Examples include:

- Reactors operating in sufficiently high orbits that fission products will decay to the actinide levels prior to reentry or impact.
- Reactors orbiting bodies where future impact is of no consequence, e.g., the Sun or Jupiter, and collision with other orbiting bodies, e.g., a moon, is not of concern.
- Deep space missions where the reactor is not expected to return.

Other missions may require specific D&D plans. Except perhaps for deep space missions, it is assumed that fission system should be shut down at the end of life, thus beginning a decay process that renders the reactor safer over time. D&D activities need to be considered early in the mission design.

Some missions that include orbits below the sufficiently high orbit criteria can be addressed in a number of ways. Typically, for Earth orbits and other bodies, the disposal could include boosting to a sufficiently high orbit per UN criteria[10]. Fuel to accomplish the final boost must be available. One alternative is to provide a positive dispersal/destruct system to allow for planned burnup on reentry, or a second alternative could be to provide for a targeted, or planned, reentry, such as was applied during the accident response for the RTG on Apollo 13 in 1968[11]. However, planned reentry is generally not preferred due to political, as well as technical, considerations.

For missions involving NEP or NTP where only part of the mission profile involves orbiting, then the propulsion system could be used to direct the reactor into the Sun or another acceptable body or to direct it into deep space. The best choice may depend upon the mission profile and the capability of the spacecraft. Such approaches could be considered for reactors used for out and back transport missions to the Moon or Mars.

Surface power missions that require D&D are different because the reactor may not leave the body it resides on or disposal elsewhere. There are three possibilities for surface D&D:

- Allow the reactor to remain in place with no intentional D&D activities beyond safe shutdown and establishing an exclusion area.
- Allow the reactor to remain in place with deliberate D&D activities such as reactor burial in place or providing shielding.
- Move the reactor to a safe location. Significant transport capability might be necessary, including hoists or cranes and a designed disposal location.

In the event of an abnormal reactor termination, e.g., a reactor accident, other measures may be necessary for D&D. In some cases, the event may render D&D impossible.

VIII. CONCLUSIONS

This paper identifies considerations for the operations of space nuclear fission power and propulsion systems from their startup through their several stages of operation and to their eventual decommissioning and disposal. A few specific conclusions can be drawn.

1. There are many details that need to be identified before any fission system could be launched on a mission in space; however, it is important to immediately identify the safety and design requirements, criteria and standards that will guide the further development of space fission power and propulsion systems.

2. Due to the emanations of radiation from all fission power and propulsion systems, the early establishment of radiation exposure guidelines, criteria and standards for both people and equipment will enable both mission and reactor designers to move forward with detailed plans and designs.

3. The need for maintenance on and around a fission power or propulsion system should be minimized.

4. Reactor control and instrumentation must include and integrate both human operator and autonomous control technologies to safely startup, operate and shutdown space power and propulsion reactors.

5. Considering, analyzing, preparing for, and establishing standards and criteria for accident analysis may be necessary for all fission power and propulsion systems early in the design process for these systems.

6. It is important to consider the possible planetary impacts and the mechanisms for decommissioning and disposal of all fission reactor systems early in their design consideration.

7. The complete development of operational procedures for testing, startup, radiation protection, power management, emergency management, shutdown, extended dormant periods, decommissioning, disposal, etc. will be needed, as well as development of training and certification programs for both remote and local operators.

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LEU NTP FLIGHT DEMONSTRATION VEHICLE AND APPLICATIONS TO OPERATIONAL MISSIONS

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Nuclear thermal propulsion (NTP) has been extensively researched as a potential main propulsion option for human Mars missions. NTP's combination of high thrust and high fuel efficiency makes it an ideal main propulsion candidate for these types of missions, providing architectural benefits including smaller transportation system masses, reduced trip times, increased abort capabilities, and the potential for transportation infrastructure reuse.

Since 2016, AR has been working with NASA and members of industry as part of the NASA Space Technology Mission Directorate Game Changing Development Nuclear Thermal Propulsion Project. The overall goal of this project is to determine the feasibility and affordability of a low enriched uranium (LEU)-based NTP engine with solid cost and schedule confidence.

Having shown feasibility and affordability, program planning has been underway for follow-on activities to continue to mature the LEU NTP engine technology. These activities include program planning for reactor fuels testing, reactor component design, engine component technology development, test facility design and demonstration, and a demonstration engine available for ground test and potentially flight test. These follow-on activities would set the stage for full scale development of a human rated NTP flight engine for use in human exploration missions.

This paper presents details of a potential LEU NTP prototype flight test and corresponding first flight vehicle along with potential applications of an evolved vehicle for subsequent operational missions.

NOMENCLATURE

AR = Aerojet Rocketdyne
CFM = Cryogenic Fluid Management
CLV = Commercial Launch Vehicle
DoD = Department of Defense
E-M = Earth-Moon
L1 = First Lagrange Point
GCD = Game Changing Development
Isp = Specific Impulse
LEO = Low Earth Orbit
LEU = Low Enriched Uranium
LH2 = Liquid Hydrogen
MEO = Medium Earth Orbit
MLI = Multilayer Insulation
MMOD= Micro-meteoroid Orbital Debris
MSFC = Marshall Space Flight Center
NASA = National Aeronautics and Space Administration
NTP = Nuclear Thermal Propulsion
RAAN = Right Ascension of the Ascending Node
RCS = Reaction Control System
SLS = Space Launch System
SOFI = Spray-on Foam Insulation
STMD = Space Technology Mission Directorate
TDRS = Tracking and Data Relay Satellite
ULA = United Launch Alliance

I. INTRODUCTION

Since 2016, AR has been working with NASA, the Department of Energy, and members of industry as part of the NASA Space Technology Mission Directorate (STMD) Game Changing Development (GCD) Nuclear Thermal Propulsion (NTP) Project. The overall goal of this project is to determine the feasibility and affordability of a low enriched uranium (LEU)-based NTP engine with solid cost and schedule confidence.

Having shown feasibility and affordability, program planning has been underway for follow-on activities to continue to mature the LEU NTP engine technology. These activities include program planning for:

1. Initial NTP engine system technology development including reactor fuels testing, reactor component design, engine component technology development, test facility design and demonstration;
2. Prototype NTP engine development including potential testing either on the ground or in flight;
3. Human rated NTP flight engine full scale development for the full scale flight engine for human exploration mission.

As seen in Figure 1 below, the prototype NTP engine development and testing provides a path, along with the initial NTP engine system technology development activities, to a human rated NTP flight engine system.

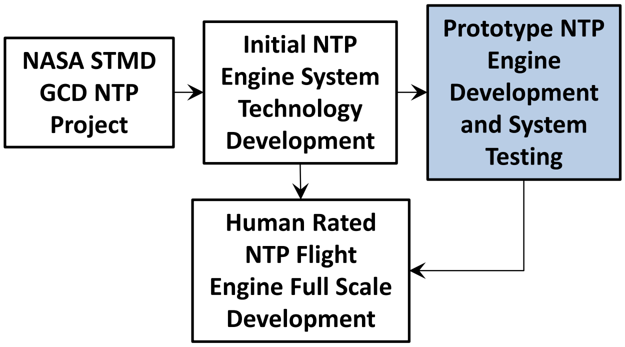

Fig. 1. Prototype NTP engine development and system testing, either on the ground or in flight, provides a path from STMD GCD NTP feasibility assessment to NTP flight engine full scale development

AR is currently performing preliminary definitions of potential prototype NTP engine flight test options that can reduce technical risk for the larger human rated NTP flight engine full scale development. The following sections will discuss these flight test options along with potential options for evolved vehicles based on the flight test vehicle to perform operational missions.

II. PROTOTYPE ENGINE FLIGHT TEST OPTIONS

In 2019, AR started examining various approaches for prototype NTP engine flight test vehicles that would have lineage to a NTP-based human Mars architecture.

An initial screening of flight test mission concepts examined missions that provide information on NTP operational verification, demonstration of integrated cryogenic systems versus non-cryogenic systems, NTP integration with a cryogenic stage similar to the Mars vehicle, packaging capability for launch on a commercial launch vehicle (CLV), and many other attributes.

Based on these initial screening activities, the best NTP and stage flight test approach appears to be one that achieves the following goals:

1. Have drop-off orbit that provides safety - independent of prototype NTP engine operation;
2. Demonstrate operation of a NTP engine (reactor) in space: Perform multiple burn sequences (start-up, main stage, shutdown, cooldown) with burn times to demonstrate NTP capability;
3. Demonstrate processes for a safe launch and operation of a nuclear reactor into space via commercial launch similar to Department of Defense launches;
4. Demonstrate passive cryogenic fluid management (CFM) for an extended period of time applicable to Lunar and Mars missions and apply data to design of robust passive/active CFM technologies;
5. Demonstrate launch of a cryogenic stage in the payload fairing of a launch vehicle (Figure 2).

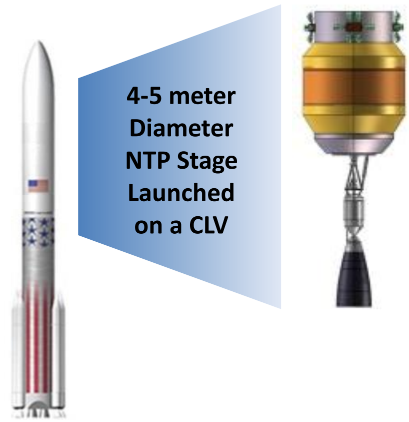

Fig. 2. A prototype NTP flight test vehicle can be sized to fit on existing or near-term future CLVs

Many potential flight test mission options can satisfy these goals, including low Earth orbit (LEO) plane changes (either changes in inclination or right ascension of the ascending node (RAAN)), LEO-to-medium Earth orbit (MEO) altitude changes, and LEO-to-Earth-Moon (E-M) L1 transfers.

The LEO plane change demonstration mission (Figure 3) was selected for further study because it has several operational advantages:

1. Flexibility in the final orbit allowing for shorter or longer than nominal NTP burn times;
2. Continuous nuclear-safe orbit throughout the mission;
3. Altitude is kept below the global continuous coverage provided by the Tracking and Data Relay Satellite (TDRS) system.

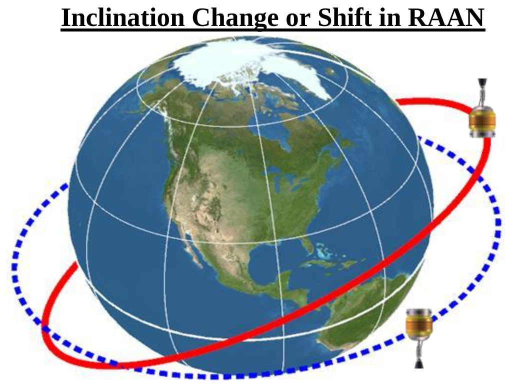

Fig. 3. LEO plane change flight test mission

III. LEO PLANE CHANGE MISSION PARAMETRIC TRADES

Parametric trades of different prototype flight vehicle sizes and different prototype NTP engine thrust levels are provided in Section III.

A primary flight test mission goal is to demonstrate NTP operability over several main engine burns. In order to achieve this goal, the LEO plane change mission is envisioned to consist of two burns, each with a minimum burn time of six minutes (30 second startup, minimum of 5 minute main stage, 30 second shutdown). This results in the need for a stage large enough to permit up to 10 minutes of NTP main stage burn time.

A nuclear safe LEO starting orbit of 2,000 km x 2,000 km x 25° is selected. This orbit is advantageous for several reasons, including:

1. Low orbital debris spatial density;
2. Negligible atmospheric drag;
3. Continuous tracking and data relay coverage provided by TDRS.

Figure 4 provides a sensitivity of burn time to stage gross mass for two different prototype NTP engine thrust levels (12.5 klbf and 15.0 klbf) with different launch vehicle class capabilities called out. A prototype NTP engine flight test vehicle with a 12.5 klbf NTP engine and a 20 mT vehicle gross mass is highlighted with the blue star as it provides sufficient total NTP main stage burn time and can be launched on the Delta IV Heavy or a future medium CLV such as Vulcan or Omega.

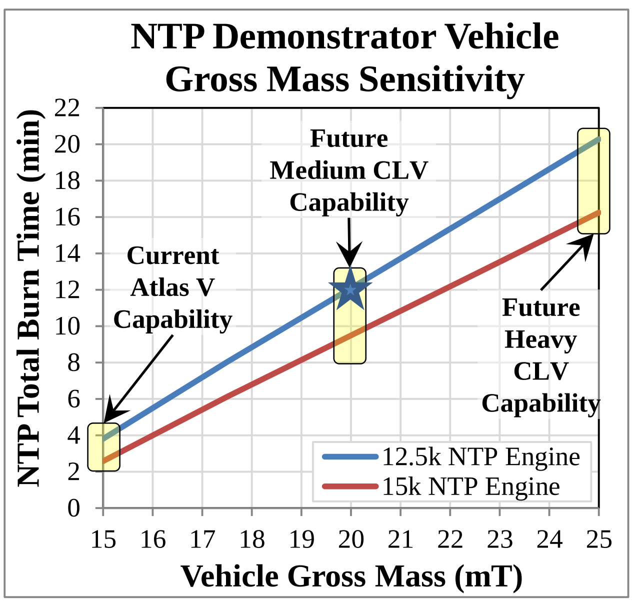

Fig. 4. Several launch vehicle options are capable of launching a prototype NTP engine flight test vehicle sized to achieve the prototype engine total burn time goal of >10 min

Figure 5 shows the orbital changes envisioned for the prototype NTP engine flight test mission. The mission details are for an example 12.5 klbf prototype NTP engine thrust and an initial test flight vehicle gross mass of 20 mT. This engine and vehicle size combination allows for over 12 minutes of main stage burn time.

Burn #1 is envisioned to operate at a lower reactor temperature, providing additional reactor temperature margin for the first use of the reactor in space, resulting in an initial Isp of 800 seconds. Burn #2 is then envisioned to operate at the 2700 K nominal reactor operating temperature, resulting in an Isp of 900 seconds.

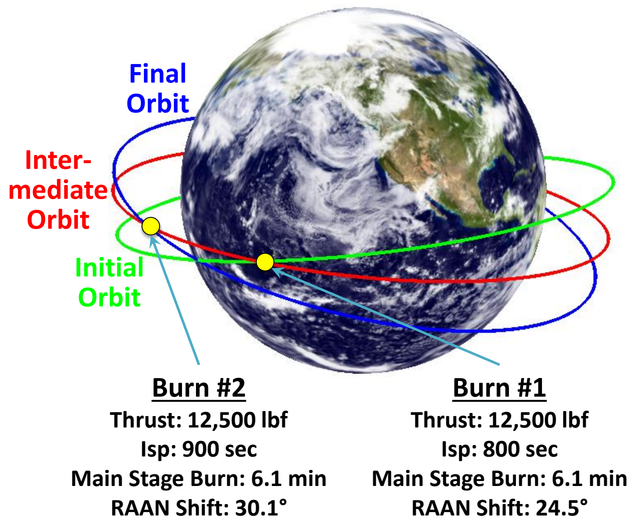

Fig. 5. Two-burn prototype NTP engine flight test mission achieves key demonstration goals (multiple burn sequences, >10 minutes of NTP main stage burn time) while staying within a nuclear safe orbit

Figure 6 provides the envisioned concept of operations for the example 12.5 klbf / 20 mT prototype NTP flight test vehicle mission shown in Figure 5. The near-24-hour mission consists of an initial checkout of approximately 6 hrs, a first 6-minute main stage burn, a 6-hr coast / cooldown, a second 6-minute main stage burn, a second 6-hr coast / cooldown, and a final approximate 6-hr for mission closeout and stage safing and monitoring.

12.5k NTP Engine / 20 mT Gross Mass Vehicle			
	$T_{Initial}$ (hr)	ΔT (hr)	T_{Final} (hr)
Launch to 2,000 km circ @ 25 deg	0.0	0.5	0.5
Spacecraft Checkout (3 orbits)	0.5	6.0	6.5
First Burn (Startup / Main Stage / Shutdown)	6.5	0.1	6.6
Coast / Engine Cooldown (3 orbits)	6.6	6	12.6
Second Burn (Startup / Main Stage / Shutdown)	12.6	0.1	12.7
Engine Cooldown / Checkout (3 orbits)	12.7	6	18.7
Mission Closeout / Monitoring	18.7	6	24.7

Fig. 6. Near-24-hour prototype NTP engine flight test vehicle mission duration is sufficient to achieve mission goals

Figure 7 shows the example 12.5 klbf / 20 mT prototype NTP flight test vehicle within the United Launch Alliance (ULA) Vulcan launch vehicle 5.4m diameter payload fairing. The NTP test vehicle can be sufficiently sized with enough liquid hydrogen (LH2) to permit at least 10 minutes of NTP main stage burn time while still fitting within the dynamic envelope of the Vulcan payload fairing.

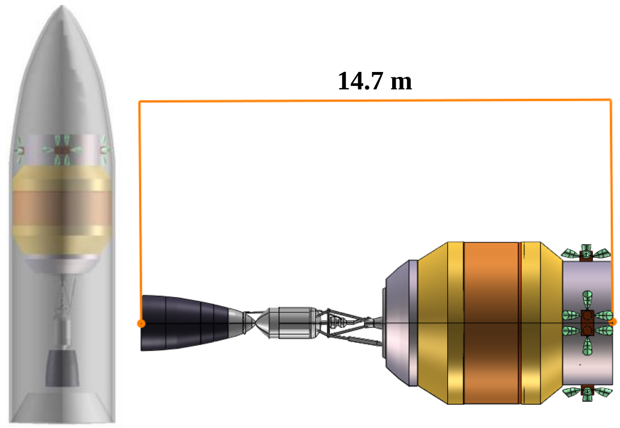

Fig. 7. Prototype NTP flight test vehicle sized to fit in the ULA Vulcan 5.4m diameter payload fairing

Figure 8 provides a summary mass roll-up for the 12.5 klbf / 20 mT NTP flight test vehicle. The flight test vehicle is envisioned to leverage CLV upper stage subsystems to the greatest extent possible, including: LH2 tank and primary structures, storable reaction control systems (RCS), batteries, command and data handling, guidance, navigation, and control, communications, passive CFM (spray-on foam insulation (SOFI), multilayer insulation (MLI)), and micro-meteoroid orbital debris (MMOD) shielding.

Subsystem	Predicted Mass (kg)
1.0 Structures	**3,018**
2.0 Propulsion	**5,625**
MPS	5,511
RCS/OMS	114
3.0 Power	**252**
4.0 Avionics	**405**
5.0 Thermal (SOFI, MLI, MMOD)	**685**
Dry Mass	**9,986**
6.0 Non-Propellant Fluids	**450**
Inert Mass	**10,436**
7.1 MPS Usable Propellant	**5,935**
7.2 RCS Usable Propellant	**180**
Gross Mass	**16,550**
Payload	**1,000**
LV Payload Attach Fitting	**500**
LV Payload Margin	**1,950**
LV Payload System Mass	**20,000**

Fig. 8. Summary mass roll-up of example NTP flight test vehicle sized to launch on a ULA Vulcan launch vehicle

IV. EVOLVED VEHICLE OPTIONS FOR OPERATIONAL MISSIONS

In addition to providing risk reduction for a full scale human rated NTP flight vehicle, the prototype NTP flight test vehicle can also provide an initial starting point for an evolved operational stage (Figure 9).

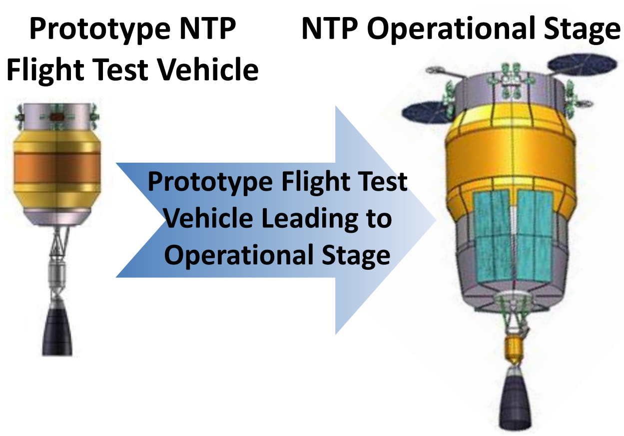

Fig. 9. A prototype NTP flight test vehicle can be evolved for use on future operational missions[1]

Missions such as outer planetary science, Cislunar cargo delivery, and Earth orbit altitude / plane changes could potentially benefit from an operational NTP in-space propulsive stage.

Example outer planetary science mission trade results are provided in Figures 10 and 11 for Jupiter and Uranus deep space science missions[2]. This NTP operational stage is sized to fit, along with a payload, within the Long SLS 8.4m diameter payload fairing. This stage has an estimated gross mass of 31 mT, approximately 50% larger than the NTP flight demonstrator vehicle discussed in Section III.

Unlike the NTP flight test vehicle, the NTP operational stage would require active CFM utilizing cryocoolers to maintain the LH2 in a liquid state for the duration of the potentially multi-year mission.

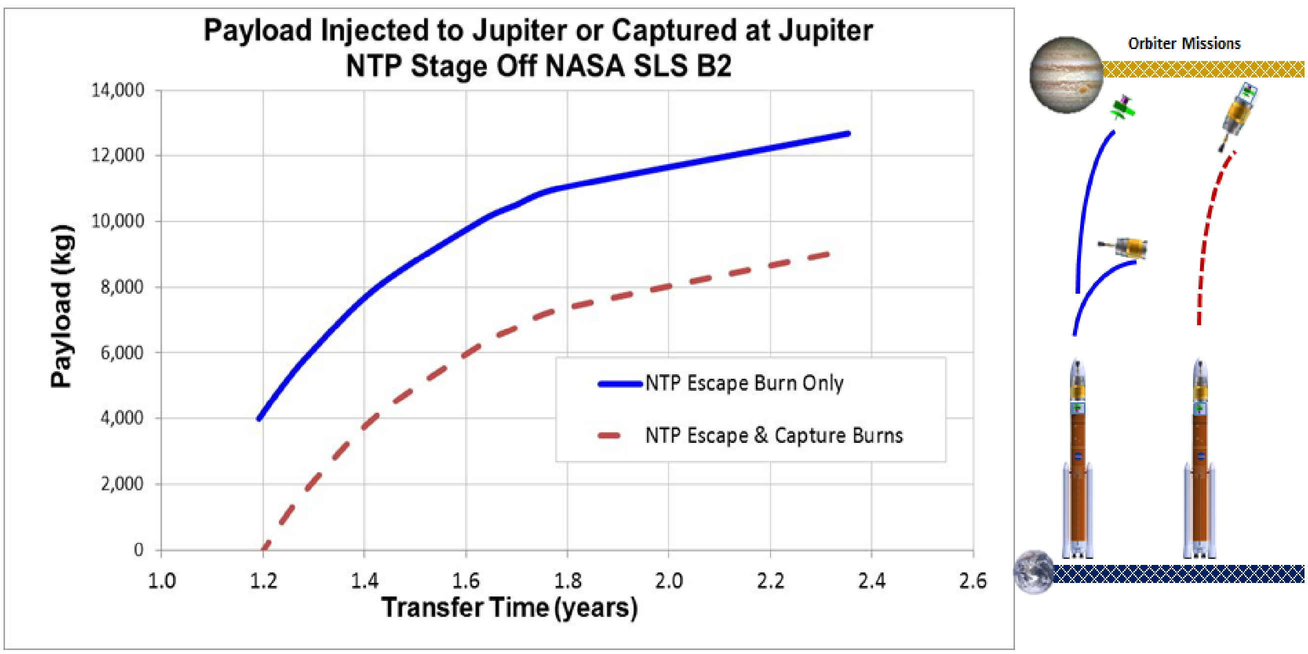

Fig. 10. Deep Space NTP Operational Stage for Jupiter Orbiter Missions using NASA SLS Block 2 Launch Vehicle

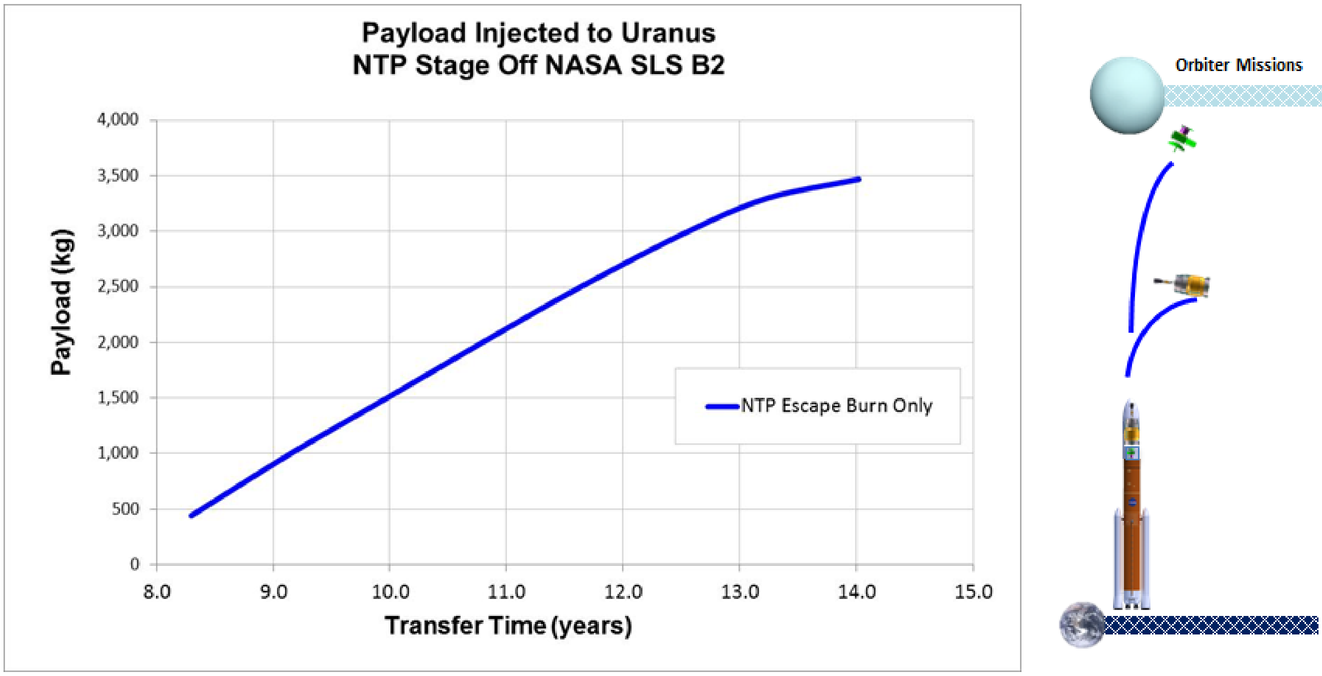

Fig. 11. Deep Space NTP Operational Stage for Uranus Orbiter Missions using NASA SLS Block 2 Launch Vehicle

V. CONCLUSIONS

Prototype NTP flight test vehicle options with applications to operational missions were defined. Flight test missions were identified that allow for the safe testing of the NTP flight test vehicle in space, demonstrate NTP engine operation in space with multiple engine burns with sufficient burn times to demonstrate main stage capability, provide risk reduction on NTP and stage CFM systems, and launch on CLV's. Furthermore, operational missions were identified that utilize an evolved NTP in-space propulsion stage to provide significant mission benefit. Examples were provided for outer planetary science missions to both Jupiter and Uranus.

ACKNOWLEDGMENTS

The authors would like to acknowledge the support of NASA Marshall Space Flight Center and Glenn Research Center, NASA Space Technology Mission Directorate (STMD) Game Changing Development (GCD) and the Department of Energy engineers that continue working NTP for Mars crewed missions and other missions that can benefit from NTP.

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NUCLEAR PROPULSION FOR FUTURE PLANETARY MISSIONS

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Nuclear propulsion can be the game changing technology for outer planet exploration and beyond. This paper presents a literature review on conceptual mission design and trajectory analysis to planetary science missions using nuclear thermal and nuclear electric propulsion. The paper discusses the enhanced and new enabling capabilities for deep space science missions using nuclear propulsion system. The paper will also present mission design and trajectory analysis for a rendezvous mission to a selected outer planet using nuclear thermal propulsion system.

I. INTRODUCTION

Nuclear propulsion can be the game changing technology for outer planet exploration and beyond. One of the many reasons for not yet having a dedicated planetary mission to ice giant planets Uranus and Neptune is the large energy requirements which increases Initial Mass in Low Earth Orbit (IMLEO). A mission using chemical propulsion system would not be possible without a super heavy lift launch vehicle and thereby increases the overall cost of the mission significantly.

With chemical propulsion systems reaching their maximum performance limit, nuclear propulsion is the next best option for outer planet exploration. High thrust and high specific impulse (over twice the best chemical propulsion engine) nuclear propulsion system can enable missions which have been limited due to the large ΔV requirements. A nuclear power propelled spacecraft with high ΔV would reduce the trip time by up to a factor of two when compared with a chemical propulsion system.

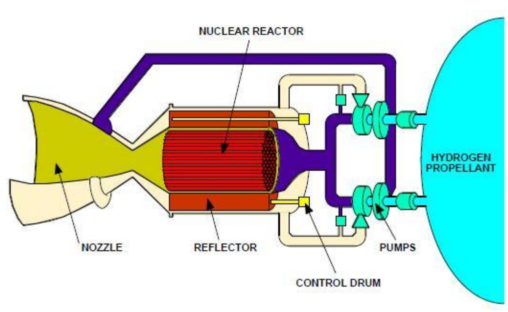

Fig. 1. Major elements of a nuclear thermal propulsion system[1].

Nuclear propulsion can be classified into two types, Nuclear Thermal Propulsion (NTP) and Nuclear Electric Propulsion (NEP). In NTP system, the liquid propellant is heated through the reactor core and is forced through a nozzle to produce thrust for the spacecraft. This type of propulsion system can produce high thrust and high specific impulse[2]. Fig. 1 shows major elements of a nuclear thermal propulsion system. In NEP system, the reactor is used to generate electric power which in turn is used to run electric thrusters to provide thrust to the spacecraft. This type of propulsion system has much higher specific impulse but has very low thrust[3].

II. PLANETARY MISSIONS TO OUTER PLANETS

Table I below shows the rendezvous missions to outer planets using chemical propulsion system. It can be noted that all the mentioned missions required various gravity assist trajectories in order to achieve the required ΔV to reach the destination planet. The requirement for gravity assist also limits the launch window of the spacecraft.

TABLE I. Rendezvous science missions to outer planets.

Spacecraft/ Destination	IMLEO (kg.)	Trajectory	Trip time (yrs.)
Galileo/ Jupiter	2380	V-E-E-G-A	6.14
Juno/ Jupiter	3625	2+ dv-E-G-A	4.92
Cassini/ Saturn	5712	V-V-E-JG-A	6.71

Previous studies using nuclear propulsion system have demonstrated that trip time for rendezvous missions to outer planets can be reduced by a factor of two. Table II below shows the trip times for rendezvous mission to outer planets using an NTP system.

TABLE II. Rendezvous missions to outer planets using nuclear thermal propulsion system[4].

Mission	IMLEO (kg.)	Trajectory	Trip time (yrs.)
Jupiter orbiter	3395	E-J	2
Saturn orbiter	4170	E-S	3
Uranus orbiter	6750	E-U	7
Neptune orbiter	10100	E-N	9

An NTP powered spacecraft can also enable new class of missions which would be almost impossible using a conventional chemical propulsion system. For example, a round-trip sample return mission to outer planet would require prohibitively large IMLEO. However, an NTP system with over twice the ΔV capability would be best suited for such missions. Another advantage of NTP system is that it can be used for onboard electric power generation. A bi-modal nuclear engine for propulsive thrust and electric power for the control of the spacecraft would eliminate the requirement of Radioisotope Thermoelectric Generators (RTGs).

III. SPACECRAFT MISSION DESIGN

Deep space missions use a number of trajectory options such as direct transfer using hohmann trajectory or planetary gravity assists. Some of the ballistic trajectory options used by planetary science missions are shown in fig. 2.

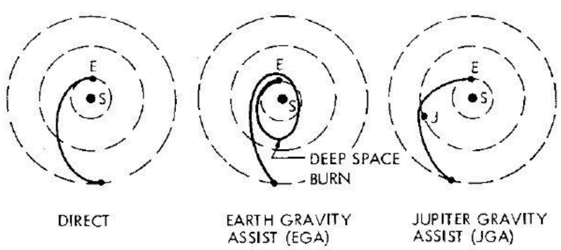

Fig. 2. Planetary ballistic trajectories[5].

With the development of NTP system, it is equally important to perform detailed mission design and trajectory analysis. A direct transfer rendezvous mission trajectory would require an Earth escape burn either by utilizing upper stage of the launch vehicle or spacecraft's onboard propulsion system. Once the NTP system injects the spacecraft into escape trajectory with sufficient C3 energy, the spacecraft will now enter into the coast phase. During the coast phase NTP system is used for onboard power generation for spacecraft command and communication purpose. During the planetary capture the spacecraft's NTP system would again be restarted to provide the ΔV for orbit insertion. Fig. 3 below shows preliminary trajectory analysis to Neptune using moderate thrust and high specific impulse propulsion system.

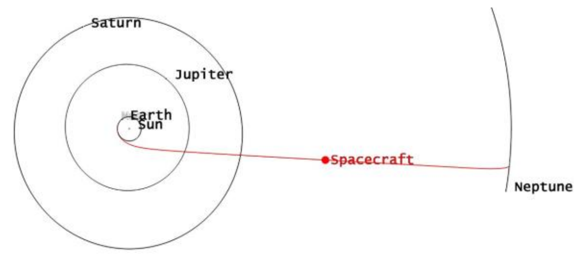

Fig. 3. Direct transfer to Neptune

IV. TRAJECTORY DESIGN METHODOLOGY

The first step in trajectory design requires calculating the approximate trip time using field-free approximation and understand various mission constrains. However, a realistic trajectory design requires numerical propagation including all the forces acting on the spacecraft. For this purpose, Analytic Graphic's Inc.'s System Tool Kit was used to determine and optimize a high fidelity trajectory design for an NTP powered system.

The trajectory design for rendezvous mission was divided into three phases. The first phase consists of acceleration phase. During the acceleration phase, spacecraft departs from a circular low Earth orbit (NASA Orbital debris guidelines) using its onboard NTP powered system. The thrust vector are specified in Cartesian axes in order to provide the departure ΔV and plane change with respect to the destination planet. The second phase of the spacecraft is coasting phase. During this phase, heliocentric propagator is used without any active propulsion system to determine the spacecraft's expected trajectory. Trajectory correction maneuvers may be required during this phase to make sure the spacecraft is continuously oriented towards the destination planet. The last phase consists of planetary capture and orbital insertion phase. During this phase the spacecraft's heliocentric velocity is reduced by orienting the thrust vector in anti-velocity vector and continuously updating the spacecraft's attitude during the long finite burn maneuver. Highly inclined polar orbit around the destination planet is targeted during the analysis which is usually preferred for the planetary science missions.

Most of the studies for a nuclear propulsion spacecraft design have concentrated towards determining the mission trip time during heliocentric phase or using gravity-free approximation. This paper would present a literature review on conceptual mission studies to outer planets using NTP and discuss gaps to mission analysis and trajectory design. End-to-end high fidelity trajectory analysis for a rendezvous mission to a selected outer planet would be presented which would highlight the complexities of Earth escape maneuver and planetary orbital insertion phases.

V. PRELIMINARY RESULTS

Jupiter rendezvous mission was selected to perform a preliminary trajectory analysis based on a miniature reactor engine concept from literature studies. The spacecraft's initial mass in low Earth Orbit is restricted to 3350 kg. The dry mass of the spacecraft is 961 kg including payload, reactor engine and propellant tank mass. The remaining 2389 kg was allocated for fuel for Earth departure and Jupiter capture maneuvers. The preliminary results have demonstrated that the total trip time of the spacecraft is about two years which in comparison to chemical propulsion spacecraft reduces the trip time by about a factor of two. Figure 4 below shows heliocentric trajectory for Jupiter rendezvous mission.

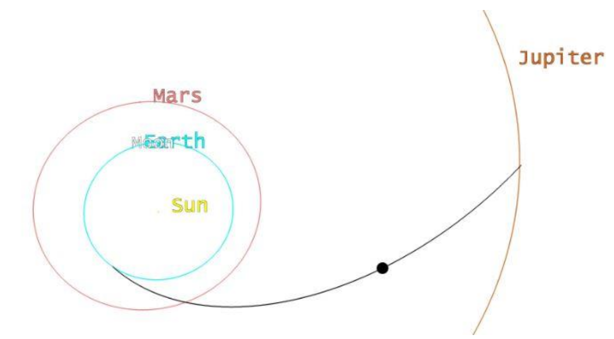

Fig. 4. Preliminary trajectory result for Jupiter rendezvous mission

VI. CONCLUSION

A spacecraft using chemical propulsion limits the capability for an ambitious outer planet mission such as a lander mission or sample return missions. The direct trajectory analysis to outer planet Jupiter have demonstrated that a highly efficient bi-modal NTP system can reduce trip time considerably when compared to chemical only propulsion system. A bi-modal system would be perfectly suitable for a sample return mission due to it high ΔV capability. The full paper will include the detailed high fidelity trajectory analysis results for a discovery class mission.

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TRADE OFFS BETWEEN HIGH AND LOW ENRICHED URANIUM FUELED SPACE NUCLEAR POWER AND PROPULSION SYSTEMS

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Abstract

When compared across seven key criteria, highly enriched uranium (HEU) and low-enriched uranium (LEU) each offer advantages and disadvantages: (1) Performance: HEU space nuclear power and propulsion (SNPP) systems demonstrate better performance characteristics at lower power levels.(2) Safety: There is disagreement in the community. Advocates of HEU systems note that HEU systems marginally safer because increasing LEU performance often requires complex designs, which reduces their inherent safety during launch and operations. Advocates of LEU systems believe LEU systems can be designed to meet all safety criteria. (3) Security: LEU is accepted domestically and internationally as more proliferation resistant, although given U.S. security procedures, the actual risk of theft of fuel during or after a launch accident may be small. (4) Timeliness: At least one HEU-based system is further along on the development continuum; there has been no development or testing of LEU systems. Depending on the level of funding invested, both systems can be developed on time scales required. (5) Fuel availability: HEU and is available when required without any modification, whereas LEU fuel would either need to be downblended or enriched. On the other hand, LEU may be more sustainable over the long term, since there are plans to enrich high-assay LEU (HALEU) commercially. (6) Cost: Costs of either HEU or LEU systems are unknown and application dependent. HEU systems are likely to have higher security costs, while LEU systems could have higher development and launch costs. (7) Public-private partnerships: Given security concerns, developing and working with LEU systems could enable greater participation from the private sector. Overall, the trade-off between HEU and LEU systems is complex. One area where LEU systems have the absolute advantage is geopolitical: using HEU in civil applications may be perceived as a double standard, since the United States encourages other countries to switch to LEU systems wherever possible.

I. Background

All SNPP systems to-date have used HEU—enriched from its naturally occurring 0.7% concentration of U-235 isotope to a concentration greater than 20% (Ref. 1)—and recent NASA testing for a surface power system used uranium enriched higher than 93%, considered weapons-grade (Ref. 2 and 3). Recent analyses have proposed using LEU for SNPP systems, specifically uranium enriched to 19.75% U-235 content, referred to as HALEU to differentiate it from lower enrichments (Ref. 4).

The discussion on enrichment levels in space systems has become controversial and fragmented, often focusing exclusively on separate criteria that support either HEU or LEU SNPP systems (Ref. 5 and 6). It is difficult to make any definitive comparisons between HEU and LEU systems: HEU and LEU designs are almost never directly comparable and relevant policy implications are accompanied by significant uncertainty. However, we can still identify the relevant criteria for trading between HEU and LEU and begin to understand how to come to a policy that treats both options evenly.

II. HEU and HALEU System Trade Off Criteria

There are seven key policy-relevant criteria pertinent to enrichment levels in space systems.

II. A. Performance

HEU generally offers performance advantages over LEU at lower power levels; these advantages may be reduced at higher power levels.

The primary determinants of power systems performance in space are **power** and **mass,** mass being a function of power. A heavier nuclear reactor (and system) results in higher launch costs, heavier-class landers for surface power systems, and lower ΔV (change in velocity) capability for propulsion applications. Volume also poses a concern; if the system does not fit in the fairing of a rocket or lander, it faces a transport challenge. It is important to note that launch is not paid for by the kilogram. Since customers typically buy the entire launcher, transportation costs rarely scale linearly with mass—they are flat to a threshold, and then jump if one has to use a larger vehicle or do multiple launches. Depending on the outcomes of the upcoming lander competitions and surface system architectures, nuclear system mass difference is not necessarily a major cost discriminator.

Fundamentally, HEU has a much lower critical mass and can provide more power per unit mass and volume. This difference is most pronounced in small, simple power systems, such as the Kilopower architecture for a lunar system, where both the 1 kWe and 10 kWe LEU Kilopower systems would be about 650 kg heavier than an HEU variant (Ref. 7). This corresponds to a mass increase of about 200% at 1 kWe (a tripling), and 40% at 10kWe. At higher power levels, the difference in mass becomes less pronounced due to the increasing prominence of other mass factors (shielding, power conversion, and thermal control). Above 100 kWe, higher enrichment levels likely do not provide as significant of a mass advantage. This makes HEU uniquely suitable for low-power surface systems than for most propulsion applications, which often call for significantly higher power levels (Ref. 8).

Several secondary performance criteria can also be important. HEU may provide more power over a greater system **lifetime** due to slower depletion of fissile material (Ref. 9). For certain low-power designs, HEU also enables reduced reactor design **complexity.** LEU systems may require moderators, dynamic reactor control, and rare material enrichments, especially in order to become mass-competitive with HEU (Ref. 10). Previous reactors used a moderator; however this lesson does not apply, as these prior systems aimed for shorter lifetimes (few months) as compared with the required lifetimes of future systems (years to decades). As with mass, differences in complexity become less pronounced at higher power.

The decision to use HEU or LEU can determine if missions are effective or even feasible. Although Human Landing System class landers may have sufficient payload capacity to land an LEU system, moving the reactor once landed may require a small mass. Second, some class of science missions may only be feasible with HEU-based nuclear electric propulsion (NEP) systems. A recent NASA study found that four of the seven considered missions (Neptune/Triton, Dual Centaur, Neptune Orbiter, Pluto Orbiter) could only be conducted with the HEU variant (Ref. 11). As a final example, Honeybee Robotics is designing an SNPP based surface melt probe to drill through ice and access the ocean on Europa. The requirement for a small melt probe diameter likely requires the volume decreases from of HEU (Ref. 12).

II. B. Safety

HEU systems offer a slight radiological safety advantage because they actually include a smaller mass of U-235. When considering potential launch accident scenarios, the greater complexity of many LEU designs may make it more difficult to preclude inadvertent criticality under all accident scenarios (e.g., full submersion) (Ref. 13). At high-power levels where both LEU and HEU systems may be moderated, both face the same criticality preclusion challenges and are therefore not significantly different in terms of launch safety.

It is important to note that both HEU and LEU systems can be designed to meet the launch safety criteria set forth in NSPM-20.

II. C. Security and Proliferation Concerns

HEU systems present greater security challenges relative to LEU. Most HEU space fission systems would contain more than 25 kg of HEU fuel, which the International Atomic Energy Agency defines as "the approximate amount of nuclear material for which the possibility of manufacturing a nuclear explosive device cannot be excluded" (Ref. 14). For example, the 1 kWe Kilopower design contains about 27 kg of HEU (Ref. 15).

The first security concern for nuclear space systems is potential **theft or loss of the fuel**. Fuel for an SNPP system could conceivably be stolen during development, transportation, storage, after a launch or reentry accident, or even from space. However, the probability of any HEU fuel in a space system being lost or stolen is likely very small. U.S. Government HEU systems are likely to use the same infrastructure or security procedures that have long been successfully applied to HEU used for national security purposes, though this may require additional infrastructure at the launch facility. Material is most vulnerable to loss during a launch or reentry accident, but both are low probability incidents, and historically countered with mitigation plans (e.g., preplaced response and recovery for launch and active dispersal for reentry) (Ref. 16). These security concerns and mitigations will result in increased cost, summarized in the next section. It is impossible to eliminate all risk of theft or diversion, and some argue that no U.S. civilian system should be allowed to use HEU (Ref. 17, 18, and 19).

Another security or proliferation concern is that other nations would use the U.S. example of using HEU in space systems to justify their own HEU enrichment and utilization efforts (not just for space). Reducing the use of HEU in some civilian applications (e.g., research reactors and medical isotope production) has been an ongoing U.S. and international effort for decades. The United States has invested hundreds of millions of dollars into converting its research and medical isotope production reactors into LEU. Using HEU for space applications while discouraging others from doing so may appear hypocritical, and some non-proliferation advocates worry this might weaken non-proliferation efforts. Some have gone so far as to argue that use of HEU in the U.S. space program would lead to increased risk of nuclear war (Ref. 20).

Other experts note that HEU's proliferation concerns may be overstated. In the past, even U.S. allies have not always followed our example, such as the decisions of the French, British, and Japanese to reprocess spent fuel despite U.S. abstinence on nonproliferation grounds. Other nations may point to a U.S. decision as justification for their own programs, but the converse is not necessarily true: abstaining from HEU in space systems may not dissuade other motivated actors or near-peer nations from pursuing HEU for space or other applications.

II. D. Cost[1]

Costs of a specific HEU or LEU SNPP system are unknown and application dependent. HEU systems are likely to have higher security costs, while LEU systems would likely have higher development and launch costs.

It is challenging to make any definitive conclusions about cost. For example, some reactor programs would likely operate under the DOE umbrella (as has Kilopower) and would not see any direct costs for fuel procurement, transportation, facilities, or security prior to the launch site because these would be included within fixed costs under another budget. Alternatively, a private company developing the ability to handle and use HEU would require hundreds of millions in facility licenses alone. Furthermore, choosing an LEU system could enable broader private financial participation in the program, which could offset some taxpayer funding.

Several cross-cutting cost conclusions are worth mentioning. First, the cost difference between HEU and LEU is likely a relatively small fraction of the total cost of any space nuclear development program, which is expected to be hundreds of millions (the cost of advancing Kilopower from its current state to a flight ready system has been estimated at $150–600M). Second, the higher launch costs of LEU are externally fixed, while the increased security costs of HEU are internally determined through security procedures. Third, the cost difference decreases with overall reactor power or size—launch costs will be high for a Mars nuclear thermal propulsion system regardless of enrichment. Cost will be more sensitive to enrichment levels for small systems (e.g., Kilopower).

1 Cost is a complicated criterion because "costs to whom" is important to determine. For example, HEU may be available free to a NASA system; however, there is still cost to producing it (and diverting it from other use). To what extent should this cost be considered if it is free to the user? Especially if the production is for a different purpose (and would be carried on even if there is no use in space).

II. E. Timeliness of Availability of System

A 1 kW HEU system has the advantage of being further along on the design and testing continuum via the Kilopower reactor design concept. The most urgent need for space nuclear fission power may be a power system to support a lunar presence. According to current NASA plans, fission power (~10–40 kWe) will likely be needed on the Moon by the mid to late 2020s. Fielding an operational system will therefore require near-term development and demonstration.

The recent tests of a 1 kWe HEU variant at zero power along with the fact that that the fuel is available and certified increases the likelihood that it can be ready by 2028. However, if a 10 kWe unit size is desired, significant system changes will be required, and the relative schedule difference between developing an LEU or HEU system would be driven by programmatic factors. Also, while a 1 kWe HEU system is further along on the Technology Readiness Level (TRL) scale, HEU systems may require about an extra year of lead launch time for preparing the launch facility and other approval processes.

II. F. Availability of Fuel

HEU is more available in the near term, because it is directly obtained from DOE's NNSA stockpile of weapons-grade U-235, and because only small quantities of HEU (tens of kilograms) would likely be required for surface reactors fielded in the near term (Ref. 21). Producing HALEU would either require downblending from the NNSA HEU stockpile or enriching LEU.

In the longer term, however, HALEU likely has a better path to fuel sustainability than HEU. A number of other applications are currently calling for HALEU, namely DOD's proposed small modular reactors and some commercial advanced reactor power concepts. DOE has begun a demonstration program with Centrus Corporation to enrich to HALEU levels for use by next generation commercial nuclear reactors, (Ref. 22) as well as other use such as Army's development of small modular reactors. The current enrichment facility in the United States, Louisiana Energy Services (LES), could be expanded to build HALEU capability.[2] However, NNSA does have long-term plans to establish HEU enrichment capability as the naval fleet will require new fuel by 2060 (under the current allocation) (Ref. 23).

2 LES is owned by foreign entities (based in France) and the centrifuges are internationally sourced, meaning that it would not meet DOD requirements of "unobligated" material. Furthermore, the facility is restricted under its operating agreement to only be used for "producing enrichment for peaceful non-explosive purposes only."

II. G. Prospects for Cost-Sharing Partnerships with the Private Sector

Use of HEU would limit, though not eliminate, opportunities for private sector participation. If LEU is more likely to be available to private entities for development of space nuclear systems outside the government, use of LEU might reduce costs and shorten timelines. It may also provide opportunities for lower-cost testing at universities and commercial facilities. Finally, LEU technology, systems, knowledge, and infrastructure can be more readily and effectively transferred to commercial application.

Government use of HEU for a space mission does not necessarily preclude private sector participation in that mission, and does not preclude commercial companies from developing their own LEU systems. However, fewer companies are likely to participate. BWXT is currently the only commercial entity licensed to handle HEU materials, but other companies can participate in design activities that do not require handling nuclear material.

III. Conclusion

The preliminary analysis above shows the decision to use HEU or LEU is a complex trade-off among criteria with varying degrees of importance, and there is far less information available than would be required to make decisions. One area where LEU systems have the absolute advantage is geopolitical: using HEU in civil applications may be perceived as a double standard, since the United States encourages other countries to switch to LEU systems wherever possible.

Whether the United States continues to use HEU or switches to HALEU (or any other level of enrichment) in SNPP systems will ultimately be a policy decision. [i]

IV. Acknowledgements

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AN OVERVIEW OF COMMONALITIES IN THE SPACE NUCLEAR POWER AND PROPULSION SYSTEMS

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The United States may develop multiple space nuclear systems simultaneously, raising total cost but potentially decreasing program-specific costs by leveraging commonality across systems. This commonality will be weighed alongside with heritage research, terrestrial reactors, future needs, and multiple mission applications. This paper examines commonality in space nuclear systems, historical examples, as well as what options may be prudent for future development programs. We conclude that commonality proposes large upside, but always adds complexity and perhaps risk of program failure.

I. COMMONALITY AND NUCLEAR SYSTEMS

A space nuclear system development, test, and evaluation program will be expensive. Previous development efforts such as Rover or SP-100 have cost billions in today's dollars (Ref. 1). A decision to develops multiple space nuclear power and propulsion (SNPP) systems simultaneously (e.g., nuclear thermal propulsion [NTP], nuclear electric propulsion [NEP], and surface power) would raise total cost, but could decrease program-specific costs by leveraging commonality across systems.

Commonality, in this context, refers to the use of common materials, components, or subsystems in more than one system, ranging from sharing the same components to building on a common, pre-determined platform. Commonality can reduce integrated costs and development time if done correctly, but can negatively affect application-specific performance. Commonality and modularity are prevalent in most engineering fields, especially consumer product design (Ref. 2).

II. MISSION DIFFERENTIATION

To inspect potential for commonality between space nuclear systems, we can first investigate the reasons which might require them to be different. Ultimately, all differentiation will be driven the profile of the missions, which will require unique parameters. This section is a high-level discussion of these differentiators.

II.A. Propulsion

Nuclear power could benefit multiple propulsion missions that are of interest to NASA, the Department of Defense (DOD), and private entities. Appendix A provides a list of potential mission needs. Proposed missions can be primarily differentiated by the amount of mass to be moved and the required change in distance as a function of time:

- Higher payload mass requires greater reactor power to achieve the same acceleration.
- The mass of the propulsion system will be important to all categories (since mass will directly affect the acceleration abilities), but low payload mass missions will be more sensitive to changes in reactor mass.
- All missions will seek fuel efficiency, or specific impulse (Isp) to reduce mass, but missions with high acceleration requirements will prioritize thrust over Isp.

For example, a NASA mission transporting astronauts to Mars prioritizes an NTP systems with an Isp of ~900s, while the NTP interceptor developed by the DOD in 1990s prioritizes propellant storability and thrust, while maximizing Isp within those constraints. Both applications benefit from a reactor temperature of ~ 2,800K to achieve the high Isp, but they differ drastically in thrust requirements—110 kN vs. 10 kN of thrust—due primarily to differences in mass. Despite size differences, it is still possible for the systems serving these missions to have much in common; for example, the DOD mission could use the same nuclear fuel (e.g., type, form, and operating temperature) as the NASA NTP system but fewer fuel elements.

The reactor operating time is an even more critical distinction between these two missions, even more so if compared to other potential missions such as a bimodal tug for science missions. Some missions require only short duration reactor operation (less than an hour in total), while others require or are enabled by continuous or intermittent operation. While the reactors might use the same nuclear fuel, their optimal designs will be very different.

II.B. Power

Nuclear systems can also generate electrical power that can then be used to power a human habitat, scientific instruments, or an electric propulsion (EP) system (see Appendix A). Power systems requirements are defined by power level, mass requirements, and operating time. Like propulsion, power systems also have a number of secondary design drivers including safety, maintainability, operability, and simplicity. The broad subsystem components are driven by the power levels across four major ranges (see Table 1).

TABLE I. Ranges of Space Nuclear Power Levels.

Power	Optimal Design Characteristics	Example missions
< 1 kWe	Radioactive decay (e.g., Pu-238) and static thermoelectric generators	Scientific orbiters and landers; ALSEP Apollo experiments
1 – ~50 kWe	Fission system, U-235 dense fuel (e.g., uranium metal), fast-neutron or thermal spectrum, Stirling generators	Early human surface occupation, NEP for science mission
> ~50 kWe	Fission system, fast-neutron or thermal spectrum, and Brayton cycle	Late surface occupation, large in-space power
> 1 MWe		Industrial power, large NEP for Mars or tug operations

Most space power reactor concepts are far lower in power than nuclear thermal propulsion reactors, which would typically produce hundreds of megawatts thermal. These size differences challenge commonality both between terrestrial and space power reactors and among space power systems themselves. The primary challenge comes from extremely small space fission systems (e.g., < 50 kWe), where (1) reactor designs begin to bump up against critical mass of fuel requirements required to even create a critical reaction, driving unique design decisions toward U-235 dense fuel elements (e.g., uranium metal); and (2) a completely different power conversion system is required. An *optimal* design for a small fission system may look drastically different from an optimal 100 kWe system.

These differences have significant implications. The most likely near-term drivers for power systems might be a lunar surface system for NASA (10–100 kWe),[1] and in-orbit power (< 100 kWe). Publicized design efforts seem focused on the 10 kWe level. However, an optimized 10 kWe system (e.g., Kilopower) does not have much in common with larger systems (although multiple reactors can combine as power modules),[2] though using a less optimized system can increasing commonality. The long-term future of nuclear power systems in space may see MW-scale reactors for industrial applications or nuclear electric propulsion (NEP), which may have little in common with a 10-kWe system. However, nearer term surface missions are likely to be lower power (<50 kW) placing an emphasis on the best systems that meet that requirement.

Mass requirements depend upon both required power and type of operation. For example, landed power systems or those in a static orbit must fit within the vehicle mass budget. These transportation operations will be critical to defining the reactor system design, and the mass will not have a linear effect on performance. On the other hand, power systems on actively propelled systems are dynamically mass sensitive. Both surface and propulsion applications will seek to reduce mass, but mass will likely be more important in the latter application. A reactor optimized for NEP will likely be higher performing.

II.C. Summarizing Mission Differentiation

Differences between space nuclear missions and those using terrestrial nuclear power raise the cost for commonality. Reactors and systems designed for propulsion will typically be overdesigned for power applications. They will be designed for higher operating temperatures, total thermal power, and lower mass than needed for surface or on-orbit power applications. In some cases, this could be advantageous for power: e.g., a low specific power reactor for NEP could better serve a surface power mission. However, many advances will not be relevant for power; for example, power reactors cannot operate at the high temperatures sought by NTP systems (e.g., ~3,000 K), due in part to current limitations in efficiencies of power conversion systems. The high operating temperature of NTP will be a unique challenge, though if it is successfully met the high temperature fuel could also operate at the lower temperatures of a power reactor.

As discussed in the previous section, all differences in fission systems will be exacerbated at low power levels, where unique challenges drive mass and other design challenges. At higher power levels (e.g., > 50-kWe), the ease of commonality increases.

Finally, terrestrial power application systems designs appear to have limited applicability to space designs, though again they could use common materials and nuclear fuels as long as they meet the space design requirements. Terrestrial reactor systems operate at lower temperatures, much higher thermal power, and with much more mass. Advanced and micro reactor systems

1 This power level has not been confirmed, and NASA studies vary widely. Those provided here range from minimal life support to significant ISRU activities.

2 The synergy and competition between commonality and modularity is very important. One of the primary benefits of commonality is that it enables larger scale production, benefiting from marginal production and economies of scale. A modular system accomplishes the same thing in a different way.

approach the operating regimes of space nuclear systems but any comparison is limited. Additionally, terrestrial systems do not require considerations of launch safety, thermal control in a vacuum, or low power conversion.

III. POTENTIAL AREAS OF COMMONALITY

Despite differences in mission and optimized-operating environments, it is possible to find commonality at the subsystem or component levels. This section does not provide an exhaustive list of commonalities nor a prescription on which are worthwhile (which will require a system-specific analysis), but rather exemplifies areas that should be considered for commonality. The findings here are based on a literature review and interviews with space nuclear experts.

III.A. Components and Materials

Commonality is most achievable at the component level, especially in fuel elements and materials.

Fuel is one of the most important and expensive parts of the traditional nuclear lifecycle. Existing U.S. nuclear plants all use a similar fuel element. A variety of fuels have been tested in the United States and abroad for space nuclear systems, and even more have been considered. Several factors seem to drive technical discussions on fuel decisions: (1) the temperatures the fuel is required to withstand; (2) U-235 density; and (3) R&D heritage.

Three parameters define fission power system fuel types: *enrichment*, *type*, and *form*. Enrichment has major consequences on reactor architectures. Three fuel *types* have been proposed or used for nuclear systems: (1) uranium metals (e.g., Molybdenum-alloyed fuel); (2) uranium oxides (e.g., UO2 common in terrestrial power facilities); and (3) other ceramic types (e.g., uranium nitride or carbide). Each have somewhat different manufacturing processes, and only UO2 is commercially used at scale in the United States. Fuel types are then manufactured into fuel *forms*, for example ceramic pellets dispersed or embedded in a metallic matrix (e.g., CERMET) or coated particles of a fuel such as U02 embedded in a matrix (often graphite).

Each of these options has varying levels of heritage, temperature resistance, material interactions, and accident tolerance, making it challenging to determine which is best. Metal fuels are ideal for low-temperature operations (for example a low power reactor for surface missions) whereas ceramics require additional development but enable high-temperature applications (important for propulsion because Isp is directly proportional to reactor temperature). Of course, high temperature fuels could also be used in low temperature applications with mass increases. Beyond these technical factors, the key tradeoffs are between technical maturity and experience, and the promise of higher operating temperatures and corrosion resistance; e.g., between the carbide-composite fuel elements tested during Rover and TRISO fuel-bed proposed by the current NASA NTP project and DOD's small reactors programs (Ref. 3; Ref. 4). Establishing a new fuel production line is expensive and time-consuming; there is always benefit to leveraging existing production capabilities or at least limiting the number of fuels required.

Another component-level commonality closely related to fuel is material research and development. New materials needed for space reactors are mainly driven by higher temperatures as well as in some cases an attempt to improve reactor efficiency through reactor components that absorb less neutrons. Some of these materials will be shared by most designs, such as a beryllium or beryllium-oxide reflectors or coated zirconium-hydride as a moderator. Some will be specific to certain subsystems, such as using refractory metals in high temperature turbines for power conversion.

III.B. Subsystem

Entire subsystems may in some cases be common between top-level systems, introducing a level of modularity to U.S. SNPP systems. There may be options for reusing subsystems within a similar type of nuclear system, and/or among SNPP system types.

Different iterations of the same type of nuclear system (e.g., NTP) can use common subsystems. This includes both variations on power, temperature, size, etc., but also commonality with future iterations of the same system; designing subsystems and the system architecture to continue to use similar pieces reduces overall cost and the challenge of upgrading system performance. An example of this would be a scaled-up surface power system using the same instrumentation measurement and control (IM&C), fluid storage, and waste heat management system as a smaller demonstrator, with simple scaling factors to account for increased size/flow/thermal energy. Subsystems not significantly affecting critical performance characteristics (e.g., not the reactor or power conversion system) can more likely be common between different system variations.

A principal subsystem extending across all power systems is power conversion.[3] Stirling generators are a

[3] There are three major types of technologies considered for space nuclear power conversion: thermoelectrics, dynamic Stirling generators, and dynamic Brayton and Rankine cycles. Thermoelectrics have the greatest flight heritage, but are limited to about ~4 kWe.[3] Stirling generators have been studied extensively at NASA, and are applicable from tens of Watts up to about 10 kWe.[3] Brayton and Rankine cycles show lower performance in the 1–10 kWe range, but will be required once power levels begin to

valuable investment for RPS and low-power FPS, while high-power FPS requires more advanced conversion systems. If lower performance at very low powers can be tolerated, then starting with a Brayton cycle to increase commonality may provide significant benefits as power is increased to meet the needs to lunar propellant production.

Similar logic extends across the types of space nuclear systems, where entire subsystem architectures (e.g., IM&C) could be extended for all space nuclear applications. An example of this might be shielding systems. All types of nuclear systems are likely to use Tungsten (W) or Lithium hydride (LIH) as the shield material, and most applications can use the same shielding designs or variations on a central theme. Developing common subsystems may require a more proactive design approach, but would drastically increase commonality. Conversely, an over-engineered or underperforming subsystem has greater cost or performance consequences than any component.

III.C. Infrastructure

Another dimension over which to consider commonality is the infrastructure that the development, manufacture, and operation of different systems can share. All space nuclear systems will have points of overlap in their lifecycles where commonality is inherent or can be leveraged. We identified five points of commonality for the terrestrial infrastructure and operations enabling for space nuclear power and propulsion.

- Fuel Supply and Manufacture: acquiring enriched uranium or sharing fuel supply lines with terrestrial programs such as the DOD small reactors program
- Reactor/System Manufacture: Commonality can lead to decreases in manufacturing costs, based on a common system/subsystem/component that can use the same equipment or a more flexible manufacturing platform that serves more components. Some manufacturing investments can be leveraged across multiple manufacturing lines or modularized for different systems.
- Testing, Models, and Simulation: While not major cost drivers, there is a strong need for validated physics-based models of space nuclear systems and the underlying codes will have significant commonality across the various systems. Common materials data and models will facilitate cheaper and faster nuclear system development and validation.
- Logistics: For example, designing transportation containers to fit multiple types of nuclear systems and even terrestrial micro reactors could save cost and time. Similarly, retrofitting storage at launch, test, or manufacturing sites that can hold and secure multiple types of systems will enable lower launch costs over multiple campaigns.
- Launch: Early missions can be pathfinders for later deployments, such that development missions prove and improve the launch approval process for the scaled deployment of operational systems. This should apply to both government and commercial processes.

IV. CONCLUSIONS AND POLICY OPTIONS

The primary goal of commonality is to reduce the cost and time of development programs while supporting more missions. Commonality can decrease lifecycle costs, enabling bulk component purchasing and leveraging fixed infrastructure costs. It can also have other long-term benefits, such as establishing inventories of shared components or subsystems, reducing long-term R&D needed to upgrade unique components, and increasing operational experience with common elements. Finally, and perhaps most importantly, commonality can lead to a reduction in the number of major development projects, decreasing fixed overhead, committed resources, and (potentially) development risk (Ref. 5).

Commonality also has a number of potential costs and drawbacks. Designing systems to be common (e.g., modular systems) results in significant up-front engineering challenges, in recent case studies adding 12–50% to development costs (Ref. 6). It can also decrease performance for specific missions, result in overdesigned systems for some applications, and reduce diversity across the entire system portfolio. Investing in commonality also introduces risks including that multiple systems can be sensitive to common points of failure and that a consolidated supply chain can reduce competition. Perhaps most importantly, spending more time and money to maximize commonality might delay results from a program, increasing the risk of it being canceled or failing without ever fielding any system. As a final challenge, intending to introduce commonality or even designing for it does not ensure that the constructed systems will have much in common; divergence in the missions and products can reduce the benefits anticipated from commonality (Ref. 7). The benefits of commonality scale with the number of systems to be manufactures while space nuclear systems have historically only supported one mission per decade.

There is potential for the United States to leverage commonality in developing space nuclear systems. These include:

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exceed 10 kWe. From: Mason, L. 2018. “A Comparison of Energy Conversion Technologies for Space Nuclear Power Systems.”

1. Encourage commonality: create a policy goal and thus an artificial incentive for programs to pursue. Such a policy goal might highlight the importance of commonality and direct agencies to identify and leverage opportunities for commonality, as appropriate.
2. Demonstrate traceability: ensure that any near-term demonstrators have commonality with longer-term missions, which effectively means traceability to multiple missions.
3. Require subsystem commonality and modularity: policy can also direct agencies to share subsystems between development programs. This can be done explicitly through policy that directs an agency or multiple agencies to develop and use a subsystem across their systems.
4. Require platforming: develop one or more platforms (e.g., a reactor) that will serve the maximum number of missions possible.
5. Focus on feasibility: do not to encourage commonality *a priori* but develop the system or a portfolio of systems incrementally. This approach would prioritize initial demonstration before adding extra requirements for long-term efficiency.

Successfully deploying and demonstrating a space nuclear system would be beneficial across systems—all space nuclear programs will benefit from increased reputation.

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IMPROVEMENTS TO THE NUCLEAR LAUNCH APPROVAL PROCESS AND OPPORTUNITIES FOR NEW MISSIONS

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Abstract

Improvements in the U.S. nuclear launch approval process may enable new missions that were not feasible before. The regulations and processes that previously existed were onerous enough to discourage, or even make infeasible, small scale missions. The improvements include changes to presidential directives, NASA environmental review requirements, and DOE safety reviews. Taken together, these improvements will significantly simplify compliance, especially for missions which use radioisotope heater units (RHUs).

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INTRODUCTION

Radioisotope Power Systems (RPS) support missions that need autonomous, long-duration power. RPS have a proven record of operation in the most extreme cold, dusty, dark, and high-radiation environments, both in space and on planetary surfaces.

RPS technologies offer potential to serve a wide range of missions. The NASA RPS Program has an established relationship with the DOE and current agreements and processes are in place to support mission requirements.

Safety is an integral part of any nuclear system, and it encompasses the entire system lifecycle.

The strategy used to meet safety objectives for any U.S. space nuclear heat source or system is to:

Design and build safety into each nuclear heat source and system at the outset, considering its potential applications;

Demonstrate the safety of each nuclear heat source and system through rigorous analysis and testing; and

Separately and quantitatively assess the environmental impact as well as the level of risk for each proposed nuclear system and nuclear-powered space mission for use in decision making and approval processes.

I. REGULATORY REQUIREMENTS

The goal of the nuclear launch approval process is to understand the risks (environment and public) associated with the launch of radioactive materials. The former process evolved from existing federal requirements, which included:

I.A. Presidential Directive/National Security Council – 25 (PD/NSC-25)

Entitled "Scientific or Technological Experiments with Possible Large-Scale Adverse Environmental Effects and Launch of Nuclear Systems into Space," this directive was first issued in 1977 under President Jimmy Carter. The directive addresses a range of actions that could have international impacts. One paragraph specifically addresses launches involving nuclear material. It specifies that certain launches require Presidential approval, and requires an Interagency Nuclear Safety Review Panel (INSRP) (including DOD, DOE, NASA, EPA and the NRC) to evaluate the risks associated with missions requiring the President's approval.

I.B. 2010 National Space Policy

The National Space Policy has been updated a number of times. The 2010 version largely deals with the issues associated with commercial space. It has a short section addressing nuclear launches and requires Presidential/designee approval for nuclear-powered spacecraft launches; and directs DOE to conduct a nuclear safety analysis and produce a safety analysis to be evaluated by the ad hoc INSRP.

I.C. National Environmental Policy Act (NEPA)

NEPA requires federal agencies to analyze potential environmental impacts during program and project decision making. It specifically requires an environmental impact statement (EIS) for "major federal actions significantly impacting the quality of the human environment."[1] The Council on Environmental Quality (CEQ) has regulations which expand upon the requirements in the statute and provide guidance on its implementation. NASA also has issued regulations (14 CFR 1216) for its implementation of NEPA.

I.D. National Response Plan (NRP)

The NRP provides the mechanisms for a comprehensive coordinated response to all Incidents of National Significance. Incidents of National Significance are high-impact events that require an extensive and well-coordinated multiagency response to save lives, minimize damage, and provide the basis for long-term community and economic recovery. As the principal Federal official for domestic incident management, the Secretary of Homeland Security declares Incidents of National Significance (in consultation with other departments and agencies as appropriate).

In order to prepare the response in the event of an accident associated with a nuclear launch, a radiological contingency plan (RCP) team is established and includes all appropriate federal, state and local agencies that may be involved in the response.

I.E. NASA Procedural Requirement (NPR) 8715.3 "General Safety Program Requirements", Chapter 6 "Nuclear Safety for Launching Radioactive Materials"

NASA's Nuclear Launch Safety Approval (NLSA) process is captured in NPR 8715.3 "General Safety Program Requirements", Chapter 6 "Nuclear Safety for Launching Radioactive Materials." The NPR includes NASA procedural requirements for implementation of PD/NSC-25, and is managed by Office of Safety and Mission Assurance (OSMA). It includes requirements to designate a Nuclear Flight Safety Assurance Manager (NFSAM) and an INSRP Coordinator, and calls for the OSMA to provide assistance to the cognizant NASA Mission Directorate and project office(s) in meeting nuclear launch safety analysis/evaluation requirements and review all radiological contingency and emergency planning.

II. PROGRESS ON IMPROVEMENTS

II.A. PRESIDENTIAL MEMORANDUM

In June 2018, the National Science and Technology Council (NSTC) Subcommittee on Space Hazards and Security formed a Nuclear Safety Launch Process (NSLP) Working Group (hereafter, "the WG"). The WG was tasked with reviewing the existing launch approval process and considering potential policy and process adjustments, possibly including revisions to PD/NSC-25.

Areas that the work group felt needed to be addressed included:

- Trigger levels – i.e. the establishment of a revised threshold for triggering the launch approval process.

- Bounding – i.e. an acceptable risk or exposure level that is determined to be sufficiently safe for launch approval, and

- Processes, i.e. the establishment of standards and procedures to guide the INSRP in the conduct of their reviews. For example, a charter or terms of reference could be developed for the INSRP to outline what is and is not expected from the review.

The WG's efforts resulted in the issuance on August 20, 2019 of a new National Security-Presidential Memorandum (known as NSPM-20).

The updated policy:

- Replaces portions of Presidential Directive/National Security Council (PD/NSC)-25 and the 2010 National Space Policy;

- Seeks to ensure that the United States can safely and efficiently develop and use space nuclear systems to enable or enhance exploration or operational capabilities, and that safe application of space nuclear systems is a viable option for commercial space activities;

- Provides consistent safety guidelines, clarifies roles and responsibilities across government agencies, and delineates requirements for analysis based on the relative risks of different missions.

Some of the key changes the NSPM-20 makes to the previous government nuclear launch approval process include:

- Establishes safety guidelines to assist mission planners and launch approval authorities in ensuring launch safety across the full range of space nuclear systems.

- Directs that safety analysis incorporate past experience to maximize effectiveness and efficiency.

- Replaces the mission-specific ad hoc Interagency Nuclear Review Panel (INSRP) with a standing Interagency Nuclear Safety Review Board (INSRB). This change will facilitate engagement early in the mission planning process, as well as between missions when space nuclear systems are under development.

- Expands the membership of INSRB to fully include DOT and NRC.

- Directs NASA to ensure that INSRB is available to review any potential commercial launch of a space nuclear system under review by DOT.

- Structures launch approval for space nuclear systems to follow a tiered process based on system characteristics, level of potential risk, and national security considerations

- While launches in all tiers require safety analysis, review, and reporting, only Tier II and Tier III launches require INSRB review, and only Tier III launches require the President's approval.

- Directs the Secretary of Transportation to issue guidance describing the process DOT will use to evaluate any application for a license involving a space nuclear system.

The NSPM-20 establishes three tiers for the launch approval process, based on the quantity of material being launches and the mission risk. The three tiers are defined as:

Tier 1

- launches of spacecraft containing radioactive sources of total quantities up to and including 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. For Pu-238, this is equivalent to the amount of material in approximately 55 RHU's.

Tier 2

- (i) launches of spacecraft containing radioactive sources in excess of 100,000 times the A2 value referenced above;

- (ii) any Tier I launches where the associated safety analyses determine that the probability of an accident during launch or subsequent operation resulting in an exposure in the range of 5 rem to 25 rem TED to any member of the public is equal to or greater than 1 in 1,000,000;

- (iii) any launches of spacecraft containing nuclear fission systems and other devices with a potential for criticality (defined as the condition in which a nuclear fission chain reaction becomes self-sustaining), when such systems utilize low-enriched uranium (less than 20 percent uranium-235 enrichment).

Tier 3

- launches of any spacecraft containing a space nuclear system for which the associated safety analyses determine that the probability of an accident during launch or subsequent operation resulting in an exposure in excess of 25 rem TED to any member of the public is equal to or greater than 1 in 1,000,000.

- Due to potential national security considerations associated with nuclear nonproliferation, Tier III shall also apply to launches of spacecraft containing nuclear fission systems and other devices with a potential for criticality when such systems utilize any nuclear fuel other than low-enriched uranium.

II.B. NEPA

NASA's regulations implementing NEPA list types of actions that "normally require an EIS," including "Development and operation of a space flight project/program which would launch and operate a nuclear reactor or radioisotope power systems and devices using … a total quantity of radioactive material for which the A2 Mission Multiple (see definitions in Appendix A to this subpart) is greater than 10."2

NASA has begun discussions with the Council on Environmental Quality (CEQ), the executive agency which oversees NEPA implementation across the federal government. The initial indications are that CEQ is supportive of making NASA's NEPA regulations less proscriptive and more flexible.

NASA has also nearly completed the development of a Programmatic Environmental Assessment (PEA) for missions that would use only RHU's. The PEA would satisfy NASA's obligations under NEPA for missions that fit within its parameters. The PEA would cover spacecraft launched from Kennedy Space Center (KSC) and Cape Canaveral Air Force Station (CCAFS), Florida. The U.S. Department of Energy (DOE), U.S. Air Force (USAF) and Federal Aviation Agency (FAA) are cooperating agencies on the PEA. The DOE's cooperating agency role stems from its responsibility in producing and controlling the radioisotope material used in RHUs; DOE maintains the ownership of RHUs throughout their life cycle and allows for their use in NASA missions. The USAF is a cooperating agency because it manages the launch facilities at CCAFS and has expertise in launches using RHUs. The FAA is a cooperating agency because it issues launch operator licenses and experimental permits for commercial spacecraft activities at KSC.

A key question in the development of the PEA was how many RHU's should be considered to be within its scope. The decision was to base the upper limit of RHUs on the projected need as determined by NASA; we then performed an impact analysis on alternatives developed based on the potential need and disclose the risk through the NEPA process. This process follows a traditional NEPA approach of agencies determining their proposed action based on their needs and then determining the environmental effect of that need. The programmatic EA examined up to 130 RHUs (or the Curie equivalent).

The PEA was issued for public comment in September, 2019.

III.C. NATIONAL SPACE POLICY

The NSPM-20 states that it supersedes the 2010 National Space Policy requirement that the Secretary of Energy "shall conduct a nuclear safety analysis" for launches that require Presidential approval. However, it still requires that a safety analysis report (SAR) be prepared by the launching agency.

DOE management has reviewed the process for producing SAR;s for past missions and has found that:

- Excellent engineering provides significant mitigation of Pu release

- No formal regulations to establish requirements and acceptable level of risk acceptance (exempt from 10 CFR 830)
- The space nuclear safety analysis methodology differs from other current DOE approaches
- The process has remained relatively unchanged for several decades
- There was a lack of prioritization and binning to assess risk importance, as it was easier to accept and make changes versus analyzing the technical merits of risk impacts resulting from these changes

DOE has considered several options for nuclear launches subsequent to Mars 2020. One option is to prepare a documented safety analysis (DSA), centered around the RPS or RHU's, using DOE published standards as guidelines or references. Once such a DSA is in place, it can be used to bound the conditions for launch in a technology and mission independent fashion. For example, the DSA could show that the RPS or RHU has a very low probability of releasing plutonium-238 under a given set of pressures, temperatures, and shock limits. If future missions do not result in conditions that exceed these established limits, then further analysis of the accident scenario, or any modification to the DSA, would be unnecessary.

A DSA for RHU-only missions has now been kicked off by DOE. When completed, it would function as described above and could significantly streamline the launch approval process for missions within its scope. Assumptions made for this DSA include:

- A single generic launch vehicle / spacecraft configuration will be defined to provide bounding values for fuel, etc.
- The hazard analysis will consider accidents associated only with the launch phase(s) that contributed the majority of risk per the MSL and Mars 2020 FSARs (e.g., pre-launch, early launch)
- A bounding nuclear payload consisting of LWRHUs will be established
- Acceptance criteria (e.g., health effects) other than current DOE Evaluation Guidelines may be needed for evaluation and risk binning of potential hazardous events
- Environmental effects of potential radioactive material releases will be adequately addressed in pertinent NEPA documentation (i.e., the safety analysis methodology does not address environmental impacts)
- DSA may be used to support integration of safety into mission and launch decisions

II.C. FISSION SYSTEMS

In addition to RPS, efforts are also underway to improve the nuclear launch approval process for potential fission systems. Fission systems have been a consideration in the OSTP's work on the process, and NASA has commissioned a work group to make policy and technical recommendations in this regard.

II.D. NASA SAFETY POLICY

As mentioned above, NASA Procedural Requirement (NPR) 8715.3 "General Safety Program Requirements", Chapter 6 "Nuclear Safety for Launching Radioactive Materials" is the NASA internal policy that dictates how nuclear launches are managed. The current policy was written to reflect the old PD/NSC-25; the policy is now in the process of being revised to reflect the new NSPM-20. In addition, a NASA standard will be developed to describe in detail NASA's safety practices and procedures for nuclear launches.

III. EXAMPLE – APPLICATION TO A MISSION

This is to give an example of the impact of all the changes mentioned above on a potential mission. The most immediate impacts would be to missions that utilize RHU's, as the NEPA and DOE safety processes will now have programmatic "umbrella" coverage for future RHU missions, and they also fall in Tier 1 of the NSPM-20.

Consider a potential mission using RHU's as heat sources to allow a lunar rover to survive the lunar night. Table 1 outlines the review and approval requirements under past policies and regulations, and the new requirements after process improvements have been made.

Table 1 – Example

Notional Lunar Rover with <50 RHU's to Survive Lunar Night

Area	Prior Policy/Approach	New Policy/Approach
NEPA	**Mission specific EIS**	**Covered by Programmatic EA**
Review	**Full INSRP**	**NASA review**
Safety Analysis	**DOE – mission specific Safety Analysis Review (SAR)**	**DOE Documented Safety Analysis (DSA) for the system**
Approval	**President**	**NASA Administrator**

It is expected these improvements could save the example mission over $30 million in review, analysis and approval costs.

V. CONCLUSIONS

The nuclear launch approval process encompasses a number of complimentary safety and environmental reviews, building on an extensive technical basis of information regarding launch vehicles, spacecraft, and RPS/RHU's. This technical basis is the key to maintaining safety and environmental protection. The "how" for the reviews should be updated to ensure they reference current standards, make use of past analysis, and are commensurate in scope to the risks being addressed.

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BIOGRAPHY

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HOW WILL A NASA DECISION ON LOW VS HIGH ENRICHED URANIUM FOR A NUCLEAR FISSION SPACE POWER REACTOR EFFECT THE COMMERCIAL SECTOR?

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Nuclear fission reactors for the generation of electrical power in space have promising applications for both government and commercial use. Due to technical, financial, political and material security concerns the choice of reactor fuel enrichment level must be carefully considered. The National Aeronautics and Space Administration (NASA) is currently leading the development of a reactor for civil purposes, and has the opportunity to set the expectations and standards for the commercial space sector. In this paper, we consider the impact of NASA's decision on fuel enrichment level, on the commercial sector, and how this decision fits within the scope of current U.S. space policy. We suggest that the development of a low-enriched-uranium fueled reactor would be favourable towards stimulating development of nuclear power technologies by the commercial sector, which would in turn reduce costs to NASA through future public-private partnerships.

I. INTRODUCTION

NASA has identified nuclear Fission Power Systems (FPS) as an enabling technology for the realization of a human mission to Mars[1]. In the nearer future, FPS are also being considered for a 2028 human base placed within the permanently shadowed craters of the lunar south pole[2]. Nuclear-enabled technologies have also received interest from the military, as well as the commercial space sector[3]. The seriousness about development of this technology is perhaps best evidenced by the recent presidential memorandum revising and streamlining regulatory policy on the launch of nuclear technologies into outer space[4].

Within the trade-space of possible FPS reactor configurations is the consideration of whether to use low-enriched uranium (LEU - less than 20% U-235) or high-enriched uranium (HEU - ≥ 20% U-235)[5] as a reactor fuel. The full breadth of consequences resulting from this decision are beyond the scope of this paper, however they encompass issues such as reactor mass[6] (with HEU leading to a lighter reactor), security costs associated with HEU use,[6-8] concerns of nuclear proliferation[9] and the associated international politics[10].

The purpose of this paper is to focus on NASA's collaboration with the commercial space industry. How will NASA's decision on fuel for its FPS affect its ability to collaborate with the commercial sector, and in turn, how will this decision affect the success of commercial nuclear endeavors?

II. NASA'S GROWING RELIANCE ON THE COMMERCIAL SECTOR

Up until the late 1980s, NASA's relationship with the commercial sector was mainly that of a contractee, with industry providing hardware and comparatively minor services to fulfil NASA mission goals. This changed with the Reagan administration, whose 1988 Space Policy increased the importance of the commercial sector in enabling government space activities[11]. Subsequent administrations have either maintained, or strengthened this approach.[13-16] As of the end of 2019, U.S. Space Policy was that departments and agencies should "purchase and use commercial space capabilities and services to the maximum practical extent"[15,16]. Indeed, the policy states that departments and agencies should "develop governmental space systems only when it is in the national interest and there is no suitable, cost-effective U.S. commercial or, as appropriate, foreign commercial service or system that is or will be available," and they must even "refrain from conducting United States Government space activities that preclude, discourage, or compete with U.S. commercial space activities"

Therefore, rather than monopolizing civilian space activities, the modern NASA frames itself as a pioneer of new space capabilities.[17] NASA's role is to lead in areas where commercial entities cannot realistically turn a profit, that is to mature new technologies and demonstrate capabilities whose long development timeline would deter typical investors. These technologies are expected to trickle down into the commercial sector through contracts, partnerships, and transfer.[18] Once it is economically viable for the industry to take its place, NASA can push off this new foothold to the next frontier, technological or otherwise.

Perhaps the greatest shift towards this new paradigm occurred when NASA began purchasing launch services, rather than building and operating its own rockets. The poster child for this model has been the Commercial Orbital Transportation Services (COTS) program, where NASA is partnering with two commercial suppliers

(SpaceX and Northrop Grumman) to provide cargo delivery services to the International Space Station (ISS)[19]. COTS demonstrated "that the space agency could rely on non-government providers for safe, reliable, and cost-effective cargo delivery services" and "the partnerships also provided a boost to the nation's economy by creating jobs in the commercial space sector and enabled the U.S. to recapture a share of the global launch market" [19]. Perhaps more importantly, "the COTS program saved taxpayer funds." For example "the final cost for developing and demonstrating the Falcon 9 rocket was only about $400 million—up to 10 times less than projected."[19] NASA utilizes these commercial capabilities to cost effectively reach and operate in low Earth orbit, providing a springboard to the Moon and eventually Mars.

These agreements between NASA and commercial partners take the form of public-private partnerships (PPPs) rather than typical contractee/contractor agreements. There exist numerous historical analogues for how such partnerships can benefit both the government and commerce[20], and NASA has now embraced this idea wholeheartedly, relying heavily on PPPs for future human missions to the ISS[21] as well as numerous services to support its return to the Moon[22-24]. It is clear that PPPs provide a way for NASA to satisfy the evolving space policy requirements of the last three decades, as well as meet tightening budget requirements. Nor is NASA alone in this endeavor, with the air force seeking to build lasting partnerships with both start-ups and established contractors[25].

III. LEU VS HEU FOR COMMERCIAL SPACE NUCLEAR POWER SYSTEMS

Recently, a number of commercial companies have expressed interest in developing, or utilizing FPS in space[3]. Perhaps most notable are the Ultra Safe Nuclear Corp's Space division, which is currently developing a compact reactor for surface power called Pylon[26], as well as Atomos Space, which plans to utilize nuclear electric propelled space tugs to reposition satellites in Earth orbit[27]. This is clearly a nascent industry, however, requirements for both high power and high-power density systems will likely grow significantly in the coming decades. In particular, industries such as in-space manufacturing[28], in-situ resource utilization[29] and asteroid mining, will depend heavily on access to high levels of power.

With respect to the question of LEU vs HEU for commercial entities, it is unclear whether a commercial company would be able to license and launch an FPS fueled by HEU. A 1986 Nuclear Regulatory Commission rule "Limiting the Use of Highly Enriched Uranium in Domestically Licensed Research and Test Reactors" strongly discourages the development of new HEU reactors.[30] A possible pathway around this may be for the U.S. government, through the Department of Energy (DOE), to maintain responsibility for the security of HEU fuel during the lifetime of an FPS.

However, there are other reasons why commercial entities would shy away from using HEU. The first is the prohibitive cost of facilities and security, which would be relayed to reactor manufacturers by the fuel fabricators. Furthermore, the cost of preparing the Kennedy Space Center to handle HEU payloads has been estimated at $28 million, plus $42 million in security costs per launch[8]. Second is that, as per the recent memorandum[4], the regulatory procedure is significantly more rigorous for HEU, requiring presidential approval for launch. Third is the concern for potential Congressional and non-government organization backlash against the use of HEU[31], and the deterring effect this may have on investors and potential reactor customers. The development of nuclear technologies for space is already a risky business, and choosing HEU fuel would significantly add to that risk; through increased costs, increased regulatory requirements and adverse political pressure. Reflecting these considerations, there currently exist no U.S. companies seeking to develop FPS technology which rely on HEU fuels.

IV. CONSEQUENCES OF A NASA DECISION ON LEU VS HEU

NASA is currently actively considering the LEU vs HEU fuel choice for development of an FPS reactor[32]. One of the most pressing constraints tipping this decision is the requirement for technology readiness prior to a 2028 deadline for the proposed lunar base. This has made the recently demonstrated Kilopower reactor[33], developed jointly by NASA and the DOE, a strong contender for further development. Kilopower is fueled by HEU, whereas alternative concepts use LEU, including a proposed LEU variant of Kilopower[6] and designs from commercial suppliers. [26]

Any decision on LEU vs HEU should consider NASA's ability to save costs by developing lasting partnerships with the commercial sector. Due to their enabling capabilities, yet technical and regulatory risk, an FPS fits perfectly into the description of a pioneering space technology. NASA could develop a reactor in partnership with the commercial sector until it becomes commercially viable for companies to manufacture and market their own reactor technologies. This will, in turn, enable NASA to save further costs by engaging in future PPPs, which may come in the form of purchasing in-space power contracts. Perhaps more importantly, the agency will be assisting in the development of infrastructure for space capabilities that can only be enabled by access to large quantities of power.

Given the justified skittishness of the commercial sector about HEU-fueled reactors, it is clear that development of an LEU-fueled reactor would facilitate the process of commercialization while development of an HEU-fueled reactor would not lend itself to future technology transfer. There are very few nuclear companies that include HEU-fueled reactors technologies in their line of business, and fewer still who would be likely to take on the cost and risk associated with developing such capabilities. This would leave NASA in a position to collaborate with only a few partners, and in ways that would require the government to continue to play a critical role in regulating the nuclear material, mitigating the benefits of a truly commercial solution. Eventually, if commercial nuclear capabilities were to be established around the use of LEU fuel, this would leave NASA out in the cold with a costly, highly regulated, and politically dubious technology.

In the current U.S. space policy framework, NASA is encouraged to build partnerships with industry, and rely on their services wherever possible. Choosing to continue development of an HEU FPS would go against this practice, which has been adopted throughout the rest of the agency. Indeed, the construction of a more compact and efficient HEU reactor may be in direct competition to any LEU reactors developed under purely commercial auspices, and would potentially violate the policy of not competing with U.S. commercial space activities[15].

V. COMPARING COSTS OF LAUNCHING HEU VS. LEU-FUELED REACTORS TO TRANS-LUNAR-INJECTION

The nearest term planned application of an FPS is on the lunar surface. It is therefore important to consider the cost of delivering a reactor to the Moon. Little is confirmed regarding NASA's plans for landing payloads on the lunar surface, however realistic estimates exist for launching into trans-lunar-injection (TLI), which constitutes a significant portion of the delta-V budget for lunar cargo delivery.

NASA plans to rely on its Space Launch System (SLS) [34], as well as partnerships through the Commercial Lunar Payload Services program (CLPS)[22] to deliver cargo to the lunar surface. The CLPS program plan is that landers will be delivered to lunar orbit via commercial launchers. Launch of a payload containing nuclear material would therefore require the provider to manage the regulatory and security process of handling the payload. Clearly, launch of HEU fuel will result in significantly more stringent requirements than that of LEU fuel, and may tip the scales to favor the use of a government launch vehicle, such as SLS, for such a payload.

Table 1 shows the estimated specific cost of delivering payloads to TLI for SLS and for a comparable commercial heavy lift launcher (such as the Falcon Heavy). Making the conservative assumption that a 10 kW LEU reactor is double the mass of an HEU-fueled reactor[7], this would put the launch cost to TLI of an HEU reactor on the SLS at around 4 times more expensive than the LEU reactor delivered on the Falcon Heavy. Despite the sweeping assumption made on launch vehicle selection, this exercise highlights the potential uncertainty in cost savings for delivering a lower-mass HEU reactor. Perhaps most importantly, unlike in many areas of space mission design, it demonstrates that, in the case of highly regulated material, mass minimization is not always king.

TABLE I. Values for estimating reactor launch costs.

Reactor Fuel:	**LEU**	**HEU**
Assumed launch vehicle	Falcon Heavy	Space Launch System
Cost per launch	$150 million (expendable)[35]	$2 billion est.[36]
Max payload to TLI	16.8 tons[*] [37]	26 tons[38]
Price per kg to TLI	$8,900	$76,900

VI. CONCLUSIONS

Nuclear fission space power reactors are an enabling technology for future NASA missions, and the establishment of a commercial space enterprise. Since the late 1980s, U.S. Space Policy and NASA decision making has evolved to encourage an increasing dependence on and integration with commercial partners. Current policy, regulations, cost and risk make it infeasible for commercial companies to consider the use of HEU fuel within a space reactor, with LEU as the logical alternative. NASA should pursue technology choices that enable and enhance their ability to collaborate with the commercial sector in this area, including the development of LEU as opposed to HEU-fueled reactors. This will not only reduce cost to the government, but pave the way for the commercial sector to develop this technology into a mature and marketable product. Future returns on this technology investment will benefit both the broader commercial space sector and NASA itself.

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[*] This value is taken from the maximum Falcon Heavy trans-Mars injection payload, used as a conservative estimate for TLI payload (which is not provided).

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US POLICY ON THE USE OF HIGHLY ENRICHED URANIUM IN SPACE NUCLEAR POWER

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The US space reactor program began in the early 1950's when a small, compact nuclear reactor was identified as a potential candidate to power earth reconnaissance missions. This led to the initiation of the Systems for Nuclear Auxiliary Power (SNAP) program. Around the same time, the US considered the development of a nuclear rocket to meet Intercontinental Ballistic Missile (ICBM) mission requirements for the US Air Force. Highly enriched uranium (HEU) was selected to fuel the reactor cores for both early space nuclear missions to meet the mission requirements. HEU was later proposed for later space reactor systems including the SP-100, Timberwind, Jupitor Icy Moons Orbiter (JIMO)/Prometheus, Fission Surface Power (FSP) and Kilopower. The current Nuclear Thermal Propulsion (NTP) program has considered the use of low enriched uranium (LEU), less than 20% uranium-235, in their reactor design. Russian space reactors including Buk, Topaz I and Topaz II used HEU and as did their nuclear rocket system. The reason HEU is used is simple – to meet mission requirements the power system must be as small and light as technically feasible. Any increase in mass impacts mission viability, the scope of the scientific payload, and costs. Historically US policy has endorsed the use of HEU for space reactors both domestically and internationally. Recently this decision has come under question by a few in the nonproliferation community. This paper explores US policy on the use of HEU in space power and other reactor applications including Naval reactors and civilian research reactors.

I. OVERVIEW

Since their initiation in the 1950's to the present, space reactors have undergone many mission and technology changes but until recently all reactors were designed with HEU. This includes the SNAP 2, 8 and 10, SP-100, JIMO/Prometheus, FPS and Kilopower power systems, and the Rover/NERVA and Timberwind nuclear thermal rocket systems. The Soviet Union likewise used HEU in the design of their space nuclear reactors Buk, Topaz I and Topaz II, and in the design of their nuclear thermal rocket systems. The systems were designed to meet mission requirements including power, mass, performance, lifetime and safety. Cost has also been an important factor as the launch costs per unit mass are substantial and for surface missions the cost to land is compounding.

The use of HEU in the reactor design was baselined to meet the mission requirements. Designs based upon low enriched uranium (LEU), of 20% U235 or less, require significantly more uranium resulting in a larger core, heavier external reflectors and radiation shield, and for some designs, it can complicate the safety by minimizing the effectiveness of the external reflector and control drums. These changes result in significantly more system mass.

Since the 1970's, as concerns about nuclear proliferation have increased, US domestic and international policy on the use of HEU in nuclear reactors has been under greater scrutiny and its use has been reduced. The primary concern is that a terrorist organization or rogue nation could divert tens of kilograms of HEU to be used to fabricate an improvised nuclear device (IND). HEU is the primary special nuclear material under consideration because it has the broadest use in civilian and military reactors, fresh HEU is difficult to detect, and it can be made into a relatively simple IND. Use of an IND would have a devastating impact on the global community. Other forms of special nuclear material can also be used for IND's or radiological dispersal devices (RDD) but there is significantly less material in use.

During the 1990's, nuclear security studies were an integral part of the SP-100 and US/Russian Topaz II space reactor programs. Analyses were completed to estimate the potential risk of diversion due to an accidental reentry for systems designed to reenter intact, disperse or burnup upon reentry. Based upon the results of these analysis each program established safety design requirements consistent with nuclear security guidelines.[1,2,3] Nuclear nonproliferation concerns have been an integral part of space nuclear power and the risk of diversion had identified to be low.

Due to the increased scrutiny by the nonproliferation community into the use of HEU as fuel in nuclear reactors questions have been raised about the justification for the use of HEU in US space reactors for power and propulsion. To understand US policy on the use of HEU in space reactors it is important to look at the use of HEU in all reactor applications. This paper provides an overview of US policy on the use of HEU for space, naval and civilian research reactors and how it has evolved over time.

II. US NONPROLIFERATION POLICY

The US nonproliferation policy has expanded significantly since the breakdown of the Soviet Union and 9/11 terrorist attacks. Concerns that terrorist would be able to divert nuclear material has focused US nonproliferation policy on reducing, and if technically feasible, eliminating the use of HEU in domestic and international civilian research reactors. The question has been raised whether or not to extend the US policy on HEU reduction in civilian research reactors to include US naval vessels and space reactors. The following is an examination of US policy on the use HEU in space reactors, naval reactors, and civilian research reactors.

II.A US Space Reactors

To-date there has been no official US domestic policy on the use of HEU in space nuclear reactors for power or propulsion. The only official US government policy on this issue has been the *United Nations 47/68 Principles Relevant to the Use of Nuclear Power Sources in Outer Space* (23 February 1993, Section 2 Nuclear Reactors) that states "Nuclear reactors shall use only highly enriched uranium 235 as fuel. The design shall take into account the radioactive decay of the fission and activation products."

After the reentry and dispersal of the Soviet Cosmos 954 reactor on January 24, 1978, the United Nations (UN) convened the UN Committee on the Peaceful Uses of Outer Space (COPUOS) to address the use of nuclear power in space.[4] The above UN principle to use only HEU reflected the shared belief that only HEU could meet the stringent mass and lifetime requirements for use in space.

Nuclear power system requirements are set by the mission parameters including mass, lifetime and costs. The nuclear reactor design is impacted by each of these parameters and in large part dictates whether or not HEU or LEU is required. Up until the recent design work under the current NASA NTP project, the use of LEU was not considered as a possible fuel, and all other power and propulsion reactors had been designed with HEU fuel including the most recent space power reactor system Kilopower.[5] Therefore while there has been no specific US policy on the use of HEU it has been de facto the fuel of choice to meet the extraordinarily difficult requirements for space and planetary exploration.

The NASA NTP projects has referenced specific nuclear nonproliferation policy as justification for using LEU.[6] Yet the US policy they cite is specific to the civilian research reactor program and the production of molybdenum-99, a medical isotope. It is not a reference to space nuclear propulsion or power.[7]

In summary, there has not been an explicit US policy on the use of HEU for space nuclear reactors for power and propulsion. US nonproliferation policy encourages but does not require that LEU, rather than HEU, be considered to reduce the potential security risks. A two-fold path of completing the low power Kilopower system with HEU for nearterm use, while assessing LEU designs for future missions is recommended.

II.B US Naval Reactors

According to the World Nuclear Association, in 2017 the US Navy had 81 nuclear power ships, 11 aircraft carriers and 70 submarines, with 92 reactors.[8] US naval reactors use HEU as do the United Kingdom, Russia and India. It has been reported, but not confirmed, that China uses LEU in their nuclear vessels.

In 1995 and 2014, the US Office of Naval Reactors submitted to Congress reports on the use of LEU as fuel for naval nuclear reactors.[9] [10] According to the 2014 report: "US Navy warship requirements determine naval fuel system design features that require HEU fuel to deliver optimum performance. These Navy requirements include ruggedness, endurance, stealth, maneuverability and compactness that are necessary to deliver safe, effective operation of nuclear reactors on board Navy warships. While LEU is used in commercial and most research reactors, naval requirements are far more demanding than those in land-based reactors."[10] A similar justification could be made for the need for HEU in space reactors.

The 2014 Office of Naval report acknowledges that an advanced fuel system could be developed but success was not assured and could impact lifetime, size and ship costs.[10] Based upon the 2014 conclusions, the navy was funded to initiate work on a conceptual LEU fuel research and development (R&D) plan for Congress by 2016. The Office of the Navy estimated it would require at least 15 years and $1B to develop the LEU fuel for a future aircraft carrier reactor and several billion to deploy, although success was not assured.[11]

In 2018, the Office of the Navy was funded to determine if the LEU fuel R&D should continue or not. In 2018 the US Navy and Department of Energy sent letters to Congress stating that they would not consider LEU fuel due to the impact on performance, cost, and its unsuitability in current submarines.[12]

Therefore, while proliferation concerns have been raised about the use of HEU fuel in US nuclear naval aircraft carrier and submarine reactors, the US government has decided not to develop an LEU fuel alternative. This is driven primarily by the need to meet Navy warship requirements.

II.C US and International Civilian Research Reactors

In 1978, the US DOE initiated the Reduced Enrichment for Research Reactors (RERTR) program to develop "technology necessary to enable the conversion of civilian facilies using high enriched uranium (HEU) to low enriched uranium (LEU) fuels and targets."[13] The program was initiated as concerns over potential diversion of fuel from civilian research reactor sites were raised. The following US policies were specific to the use of HEU in civilian research reactors:

1) 1986 NRC 50.64[14] Limitations on the use of highly enriched uranium (HEU) in domestic non-power reactors states that for all non-power reactors: The Commission will not issue a construction permit after March 27, 1986 for a non-power reactor where the applicant proposed to use HEU fuel, unless the applicant demonstrates that the proposed reactor will have a unique purpose as defined in § 50.2.

   Per 10CFR50.2 Definitions:

   Unique purpose means a project, program, or commercial activity which cannot reasonably be accomplished without the use of HEU fuel, and may include: (1) A specific experiment, program, or commercial activity (typically long-term) that significantly serves the U.S. national interest and cannot be accomplished without the use of HEU fuel; (2) Reactor physics or reactor development based explicitly on the use of HEU fuel; (3) Research projects based on neutron flux levels or spectra attainable only with HEU fuel; or (4) A reactor core of special design that could not perform its intended function without using HEU fuel.

2) 1992 Schumer Amendment to the Energy Policy Act, H12103[15] on civilian research reactors enacted to restricted the US export of HEU fuel or target material for research or test reactors unless:

   a) there was no alternative fuel or target material,

   b) the recipient had assured that they would convert to LEU when possible, and

   c) the US government was actively developing an alternative LEU fuel or target for that reactor.

3) 2005 Burr Amendment[16] to the 2005 National Energy Policy Act allowed the US to export HEU for medical isotope production to Canada, Belgium, France, Germany, and the Netherlands without requiring their conversion to LEU. This amendment countered the 1982 Schumer Act and was put in place to support the production of Molybdenum-99 (Mo99) for medical use. All of US supplied Mo99 is provided by foreign suppliers, most of which use HEU targets for its production. Mo99 is used in over 40,000 US medical procedures per day.[17]

4) 2012 American Medical Isotopes Production Act (AMIPA)[18] of 2012 required the US to establish a technology neutral program for Mo99 production without the use HEU – still in progress, without disqualifying the continued operation of existing reactors with HEU when there was no alternative LEU fuel and if they were seeking to convert to LEU. Section 134 of the Atomic Energy Act of 1954 was amended to no longer allow the NRC to issue a license for the export of HEU from the US for the purposes of medical isotope production, (although this period could be extended 6 years if there was insufficient global support of Mo99 produced without HEU) from the US.

5) 2016 March 31 Joint Statement on European Union-US HEU Exchange: "HEU currently remains important for a variety of peaceful scientific applications and for the production of critical medical isotopes, while at the same time HEU constitutes a significant security risk in the hands of unauthorized actors. Hence, the Participants encourage conversion of European research reactors and isotope production industries to non-HEU-based fuel and targets, where technically and economically feasible. At the same time they acknowledge that, in some facilities, HEU is still indispensable during the transition period to conduct peaceful scientific research or to produce medical radioisotopes used for radiopharmaceutical products."[19]

The US program to develop LEU fuel and target material for the production of medical isotopes has resulted in significantly less use of HEU in civilian research reactors both within the US and internationally. As shown in Figure 1 the US export of HEU has decreased significantly since the implementation of the RERTR program in 1978.

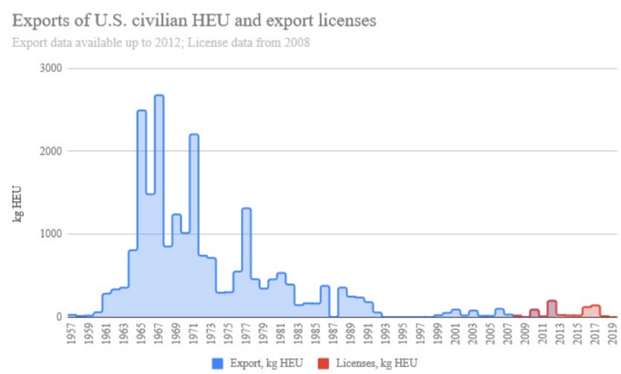

Fig. 1. US export of HEU and export licenses for civilian research reactors.[20]

The US LEU conversion program required that conversion would not significantly affect a civilian research reactor's safety, performance or operations.[21] In 2016 there were 74 civilian research reactors around the world that used, or were planning to use HEU, of which 8 are operating within the US.[21] Russia has the largest number of HEU fueled research reactors and critical assemblies in operation.[21]

The US and international efforts to convert civilian research reactors to LEU has significantly reduced the proliferation risks at a number of sites. The US and international policy has supported the conversion of civilian research reactors where technically and economically feasible and has worked to establish viable LEU fuel to support future conversions. There is strong international support for the conversion of civilian research reactors to reduce the risk of HEU diversion.

III. INTERNATIONAL PROLIFERATION CONCERNS

In addition to concerns that HEU presents a diversion risk, there are also concerns that other nations seeking to produce HEU circuitously for nuclear weapons will reference US use of HEU as justification to produce HEU for their future space reactor, naval reactors or civilian research reactors. The use of HEU in space systems is so specific and limited, and the requirements are especially stringent, that it may be understood by other nations that for this purpose it is clearly required. Future nonproliferation challenges must be met on a nation-by-nation basis as they arise with political and economic incentives and pressures just as they are today.

IV. CONCLUSIONS

The US does not have any policy limitations on the design, testing or use of HEU in US space nuclear power nor do other nations. All past space reactors for power and propulsion were designed with HEU. The present NTP project is considering LEU in its baseline design.

Space nuclear reactors for power and propulsion must meet a unique set of challenges to provide transformative capability for deep space and planetary missions. The set of mission requirements for these highly specialized challenges are primarily met through the use of HEU fuel in the nuclear reactor system. LEU may be considered for some missions but in general, will not meet the mass, lifetime, costs, and safety requirements for the majority of NASA's missions. A two-fold approach that allows the current HEU-fueled 1-10 kWe Kilopower to advanced forward while beginning the design and testing of key technologies for LEU systems would provide a win-win path forward for future space missions for NASA, DoD and civilian applications.

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