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Reenergizing U.S. Space Nuclear Power Generation

Michael W. Obal

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About This Publication

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I N S T I T U T E   F O R   D E F E N S E   A N A L Y S E S

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Executive Summary

This report presents an argument for reenergizing space nuclear reactor development in the United States. Over the next 10 to 20 years, missions with high-power requirements (i.e., 200 kWe or greater) will become essential to maintaining U.S. freedom of action in space. Nuclear power provides the only practical method of achieving such power levels in the requisite timeframe. Specifically, this report proposes a rapid prototyping approach to the development of 200 kWe nuclear-powered electric propulsion (NEP) systems that would enable game-changing space survivability and missions. Examples of such missions include space-based power transmission, ultra capacity communications, high-performance intelligence, surveillance, and reconnaissance (ISR) collections, and high-altitude nuclear detonation (HAND) remediation. These types of missions are crucial to long-term space dominance. However, as this report illustrates, solar power systems cannot practically address power requirements above 100 kWe due to the size of the solar arrays and mass of the batteries.

Several issues have stalled the development of space nuclear reactor technology in the United States. Foremost among these are low government interest owing to a lack of (or perceived lack of) mission pull, the burden of previous failed space nuclear reactor programs, and the hostile political atmosphere produced by past terrestrial contamination incidents. The perceived lack of mission pull and the programmatic issues that hampered previous programs can be addressed readily. The fundamental technical hurdles inherent in space nuclear reactor development can be resolved in a non-nuclear test environment.

To address these challenges, two rapid prototyping approaches are presented, both building on pioneering work undertaken by NASA to fabricate and test complete NEP systems in a non-nuclear, simulated space environment. Both development approaches involve heat-pipe-cooled reactors with direct energy extraction to minimize moving components and provide power to several candidate electric propulsion (EP) systems. One approach, focused on minimizing cost and technical risk, begins with a 25 kWe reactor that grows to 200 kWe at a nominal operating temperature. The other approach, focused on minimizing NEP system mass, begins with a 200 kWe reactor and increases its operating temperature by 250 K to improve system efficiency and reduce mass. Under both approaches, system operation would take place in a thermal vacuum chamber and include a high-fidelity reactor control simulator developed from zero-power tests.[1]

Either of these two approaches would provide the detailed systems engineering data necessary to project performance, cost, and schedule accurately for a future flight program. They would also redevelop U.S. nuclear design and manufacturing talent bases and reestablish space nuclear reactor development facilities. Application of lessons learned from previous failed development attempts, coupled with the adoption of a rapid prototyping approach, would mitigate development risks while gradually eroding the negative political environment associated with space nuclear power. Further work is needed to assess the current state of U.S. design capabilities and facilities that would support such an undertaking.

[1] Zero-power testing is a method by which a nuclear reactor design is evaluated in a manner that produces very little radioactive material and is covered in detail in Appendix D.

Contents

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1. Introduction

The United States and Russia have been working on space nuclear power and propulsion technologies flight programs since the 1950s. This work has resulted in numerous successful space missions (80+) and a few failures. The two failures that resulted in terrestrial radioactive contamination (United States: Transit-5BN-3 (in 1964) and Soviet Union: Cosmos 954 (in 1978)) have hampered the development of future space nuclear systems in the United States by creating a politically hostile environment[2] based on misinformation that is still prevalent today (Aftergood 1989). This environment, coupled with the past failures of two major U.S. space nuclear reactor program initiatives (the SP-100 and the Prometheus/Jupiter Icy Moons Orbiter (JIMO)), has essentially eliminated any government or commercial investment interest (FY10 ~$10 million). Several excellent references (Angelo and Buden 1985; English 1987; Bennett et al. 1996; Kulcinski 2004; Taylor 2005; Bennett 2006) provide details on the goals, accomplishments, and demise of the SP-100 and Prometheus programs.

This study will only concentrate on the nuclear reactor component of space nuclear power technology. It suggests a path forward to rebuild this technology within the next decade. The key argument presented is the role that space nuclear reactors can have in sustaining U.S. space dominance. It will also show how this objective can be achieved with space electrical power availability above 100 kWe, which is not practical using solar power technology.

The Obama Administration published an updated *National Space Policy of the United States of America* (Office of the President of the United States 2010). A few key relevant paragraphs in this policy document are cited here:

Principles (p. 3; paragraphs 4 and 5)

- As established in international law, there shall be no national claim of sovereignty over outer space or any celestial bodies. The United States considers the space systems of all nations to have the right of passage through, and conduct of operations in, space without interference. Purposeful interference with space systems, including supporting infrastructure, will be considered an infringement of a nation's rights.
- The United States will employ a variety of measures to help ensure the use of space for all responsible parties and, consistent with the inherent right of

[2] President Carter banned U.S. space nuclear reactors during his tenure.

> self-defense, deter others from interference and attack, defend our space systems and contribute to the defense of allied space systems, and, if deterrence fails, defeat efforts to attack them.

Space Nuclear Power (pp. 8 and 9)

> The United States shall develop and use space nuclear power systems where such systems safely enable or significantly enhance space exploration or operational capabilities.
>
> Approval by the president or his designee shall be required to launch and use United States Government spacecraft utilizing nuclear power systems either with a potential for criticality or above a minimum threshold of radioactivity, in accordance with the existing interagency review process. To inform this decision, the Secretary of Energy shall conduct a nuclear safety analysis for evaluation by an ad hoc Interagency Nuclear Safety Review Panel that will evaluate the risks associated with launch and in-space operations.
>
> The Secretary of Energy shall:
>
> - Assist the Secretary of Transportation in the licensing of space transportation activities involving spacecraft with nuclear power systems;
> - Provide nuclear safety monitoring to ensure that operations in space are consistent with any safety evaluations performed; and
> - Maintain the capability and infrastructure to develop and furnish nuclear power systems for use in United States Government space systems.

From these excerpts, the current Obama administration is serious about space protection and the continuing the development of space nuclear power technology. If funding can be obtained, this administration is a proponent of maintaining U.S. space dominance.

Many counter arguments to nuclear space power have been presented during the decades of this technology development. One such recent argument is that the current U.S. launch approval process (see Appendix A) of a nuclear-powered spacecraft does not allow interested non-governmental representatives to participate and thereby neglects their input and support. Therefore, U.S. policies in this matter are not well understood and accepted by the public, which may lead to excessive mission delays due to protests and/or legal action (Voegeli 2007). To avoid these delays in the development of space nuclear power, Voegeli recommends that missions should be restricted to non-military use (i.e., not for introducing weapons into space) and that an additional oversight board should be established to encourage public input and discussion. This oversight board would make a recommendation to the president on whether to launch.

The strategies presented by this author are early public education and buy-in. However, these recommendations essentially neuter U.S. response options to aggressive space-control activities currently underway by China and Russia. Details of adversary space-control efforts are highly classified to protect sources and methods and cannot be shared with the public without putting these intelligence methods at extreme risk. Therefore, the public needs to trust the current congressional oversight for U.S. responses in this arena. An additional layer of flight safety discussion and approval could be put in place, but this action will only add to Department of Defense (DoD) and Intelligence Community (IC) sponsors' perception of increased programmatic risk (cost and schedule) and will reduce the investment appeal of this technology.

Until a comprehensive and verifiable ban on offensive and defensive space-control activities is enacted, the "electric power" available to a nation's earth orbiting spacecraft ultimately will decide which nation maintains space dominance in the future.

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2. High-Powered Space Missions

In the next 10 to 20 years, space dominance will belong to the country that has the ability to fly systems having at least 200 kWe power capabilities. This amount of power supplied by a space nuclear reactor will provide two key attributes for future spacecraft:

- By powering electric propulsion (EP) systems, it will contribute significantly to increased spacecraft survivability by changing orbits in manner that will degrade foreign Space Object Surveillance and Identification (SOSI) capabilities used for tracking and targeting.
- When not powering the EP system, it will power game-changing payloads. Four unclassified examples are provided in this study.

A. Spacecraft Survivability

The freedom to access space and to safely place and maintain U.S. and allied intelligence, surveillance, and reconnaissance (ISR) and commercial assets is critical to our national security during peace and wartime. China's successful series of anti-satellite demonstrations and the ground-based electronic warfare probing by unknown actors has challenged the control of the space assets of the United States and other countries. This new threat environment has lead to the development of space-control technology investments by most space-faring nations. Although the details of space-control activities are highly classified, the technology is available to degrade or neutralize each other's space assets (Wilson 2000). One key element of space control is awareness of where everything in space is at any given moment. A SOSI system is designed to provide that awareness using optical, radar, signal intelligence (SIGINT), and other sensors. Data from these sensors are fused with published object orbit tracking information, with the goal of identifying all space objects of interest. These objects can be man-made or natural debris, active and inactive satellites, and manned or unmanned spacecraft in low earth orbit (LEO), mid-earth orbit (MEO), geostationary orbit (GEO) or highly elliptical orbits (HEOs). They can also be located between the earth and the moon and beyond. Currently, the U.S. SOSI system tracks about 18,000 objects (Iannetta and Maltk 2009).

One major benefit of SOSI systems is collision avoidance. These systems predict the movement of all objects about which they have knowledge and assess the future risk of collisions. An object in orbit is continually perturbed by non-uniform gravitational and magnetic fields, solar pressure from the sun, aerodynamic pressure from the earth's upper atmosphere (if in low-altitude orbit), and, for satellites and spacecraft, self-initiated maneuvers. These perturbations will change the object's orbit parameters (satellite element sets) over time. Therefore, SOSI systems have to work continually to keep track of whatever they can. To avoid collisions, the United States and other countries publish orbit information for many of their assets, commercial satellites, and debris that they catalog. Despite these efforts, occasional collisions still occur (e.g., the recent one (February 2009) at 790 km between a Russian (Cosmos 2251 – 900 kg) and an U.S. commercial satellite (Iridium (Ir) 33 – 560 kg)). This event created about 500 detectable pieces of man-made migrating debris, which is threatening other satellites (Iannetta and Maltk 2009).

The other role a country's SOSI system performs is targeting an object in space. If so ordered, the targeting information is used to attack a satellite using a variety of means. Adversary SOSI systems are rapidly evolving with upgraded sensors and computational systems that fuse sensor and supporting information to provide them increasingly improved space situational awareness (SSA). To track an object as it departs the field of view (FOV) of a country's SOSI system's sensors, an estimate is made as to where the object will reappear as it completes its orbit. This estimation process does not perfectly model perturbations, including maneuvers, so maneuvering can introduce uncertainty into orbit estimates (i.e., degrade SOSI system performance as shown in Figure 2-1). Maneuvers can be executed to change altitude and inclination simultaneously. Given the amount of change to the orbit caused by a series of maneuvers, a SOSI system could lose track of an object and have to begin searching for it. If the SOSI system is continually searching for an object, it cannot target this object.

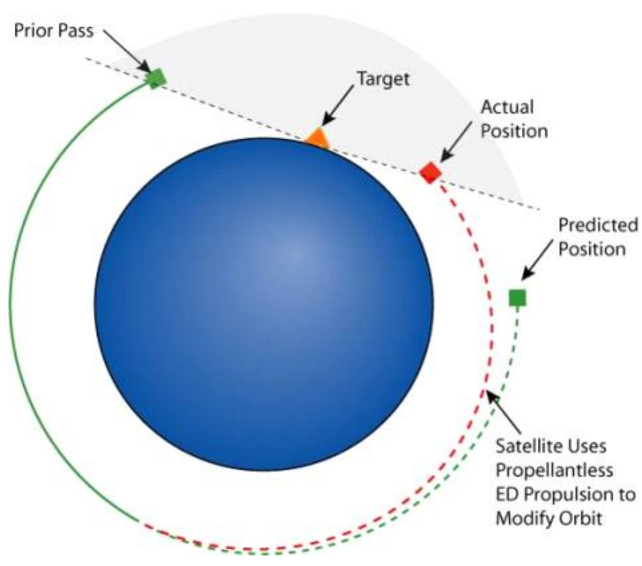


Figure 2-1. Example of SOSI Degradation

Source: Artwork courtesy of Tethers Unlimited Inc.

Changing the orbit of a satellite requires energy provided by its propulsion system. Changing the orbit enough for a SOSI system to lose track of the satellite depends on the capabilities of the SOSI system and the manner in which the orbit change was done. Table 2-1 presents a few examples of the types of orbit altitude and inclination changes possible with 5 N of thrust from a nuclear-powered electric propulsion (NEP) device and 17 N of thrust from an electrodynamic (ED) tether (a subset of EP systems). These changes in orbit are enough to cause SOSI system tracking problems.

Table 2-1. EP-Induced Orbit Changes

Mass (kg)	Thrust ± (N) for Entire Orbit Period	Initial Orbit (km)	Orbits per Day	Incline (deg)	Max. Change (Loss or Gain) per Orbit or Day (km) (Note 1)	Max. Difference in Time (Early or Late) to Some Point After One Orbit or Day (sec)	Max. Incline Change per Orbit or Day (deg)
EP (Note 2)							
				LEO			
15,000	5	1,000	~14	51.6	4/56	3/44	0.011/0.015
10,000	5	1,000	~14	51.6	6/89	4/59	0.0160/0.22
				MEO and GEO			
15,000	5	10,000	~4	28.7	46/185	44/177	0.053/0.021
10,000	5	10,000	~4	28.7	68/279	66/264	0.078/0.031
10,000	5	GEO	1	0	1,151/1,241	1,770/1,770	0.128/0.051
ED Tethers (Note 2)							
				GEO			
15,000	17	1,000	~14	28.7	14/197	16/225	0.014/0.091
10,000	17	1,000	~14	28.7	21/290	23/320	0.020/0.281

Note 1 for Table 2-1: Non-optimal thrusting (thrust pointed along fore and aft of vehicle).
Note 2 for Table 2-1: EP Iso ~7,000 sec; ED Tether Isp ~40,000 sec.

For example, SOSI at GEO is primarily done using narrow FOV optical systems that can be weather constrained. For a 10,000 kg vehicle, a 5 N thrust for 1 day changes its GEO orbit by 1,151 km and early/delay of arrival time by 1,770 sec. The distance traveled along track (circumference) would be 5,442 km/day or 7.4 deg/day (49 days to travel around the GEO belt). The satellite could also be repositioned in the 3 deg inclined dead satellite parking belt after 24 days assuming an estimated 0.128 deg/day inclination change. Therefore, the 5 N thrust activated at the right time (correlated with weather patterns) for a few days would degrade most SOSI systems.

SOSI at LEO is primarily radar with some optical and SIGINT support. These systems typically use “predictive techniques”—spot-checking objects as they enter and reenter certain sectors—but do not track continuously. A 5 N thrust activated at the right time can also degrade this SOSI approach. Nevertheless, the EP systems with an Isp[3] of ~7,000 sec would require 1,150 kg of Xenon to provide about 183 days of fuel.

Also from Table 2-1, note that the ED tethers offer the greatest potential for changing a satellite orbit characteristics in LEO. Unfortunately, ED tethers are limited in their operational envelop to LEO altitudes below ~2,000 km and at lower inclinations. This limitation is due to the requirement to complete the ED tether circuit using the higher density plasma of the ionosphere and optimize its thrust with the magnetic field orientation and strength. Details of EP system performance capabilities will be discussed later.

This study presents the technical feasibility of combining nuclear reactors with propulsion or NEP to defeat the tracking capabilities of foreign SOSI systems and to energize revolutionary space ISR and other missions' payloads. This study also describes a rapid prototyping development approach to demonstrate this technology cost effectively to investors, with the goal of enticing them to continue investing until such systems are operational.

B. High-Powered Mission Payloads

When not providing power to an EP system, the nuclear reactor would provide power to game-changing payloads. Table 2-2 lists a few examples. These mission payloads will revolutionize DoD and IC operations in space. Actual performance details of these payloads are sensitive and will not be discussed in this report.

Table 2-2. Space Mission Power Needs To Revolutionize Space Dominance

Missions
Space control – Constant orbit repositioning using electric propulsion
Power available for payloads when not thrusting
• GEO ultra-high bandwidth and secure communications/processing
• MEO Airborne Moving Target Indication (AMTI) against low observable (LO) unmanned aerial vehicles (UAVs)
• 10,000 km range high-power (kWe) transmission beaming
• Multiple high-altitude nuclear detonation (HAND) radiation remediation using ES tethers

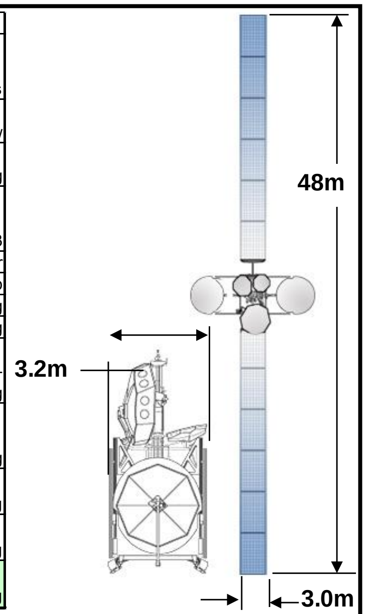
1. Ultra-High Capacity Space Communications in GEO

Demands on secure communication services from GEO to tactical users continue to drive U.S. space-based communication capacity. The ongoing command, control, communications, computers, intelligence, surveillance, reconnaissance (C4ISR) revolution requires continuous secured communications to a variety of ground, sea, and air-moving

[3] Isp = specific impulse, seconds.

ISR and unmanned sophisticated weapons platforms to meet user needs. Currently, wideband (high-capacity), protected (anti-jam, covertness, and nuclear survivability), and narrowband (mobile, voice, and low data rate) space-based communications are evolving rapidly as military satellite communications (MILSATCOM) deploy Wideband Gapfiller, systems like Advanced Extremely High Frequency (AEHF) and Advanced Narrowband. These systems, as well as commercial services, already require kilowatts of power. Figure 2-2 and Figure 2-3 show a few examples of large GEO commercial communications satellite designs.

Galaxy IIIC		
Total EOL Array Power - TJ GaAS	15000	watts
Array Specific Power (Loral)	16	kg/kw
Solar Array Mass	240	kg
Batteries - NiM IPV Assume 70% DOD	NiH	RNH1 60-3
Cell capacity	160	a-hr
Cell #	112	28s4p
Cell mass	4.18	kg
Total Batt	562	kg
Specific Capacity (70%DOD)	26	W-hr/kg
Batt/SA Management Electronics	160	kg
Total Mass 15 KW	**962**	**kg**
Mass for 100 kWe	**6413**	**kg**
Tmass for 200 kWe	**12826**	**kg**



- Using linear scalar mass for 100 kWe ~ 6413 kgs and for 200kWe ~ 12,826 for the power subsystem itself
- α ~64 kg/kWe

Figure 2-2. GEO Comsat Power Technology Extrapolated to 200 kWe

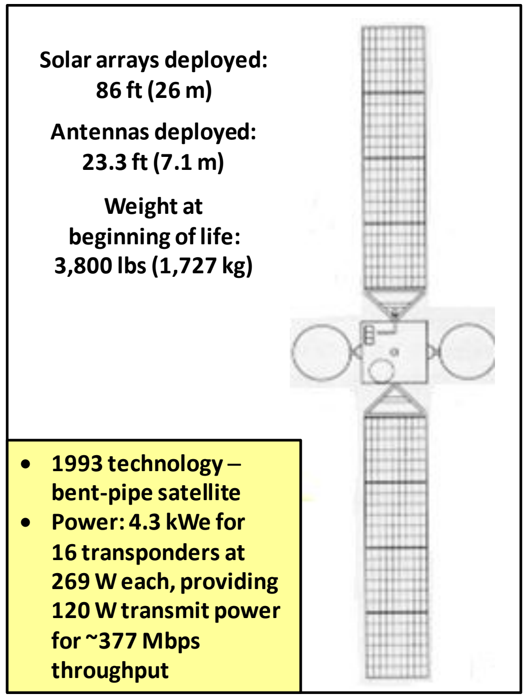


Figure 2-3. GEO DirectTV 1993 Capability

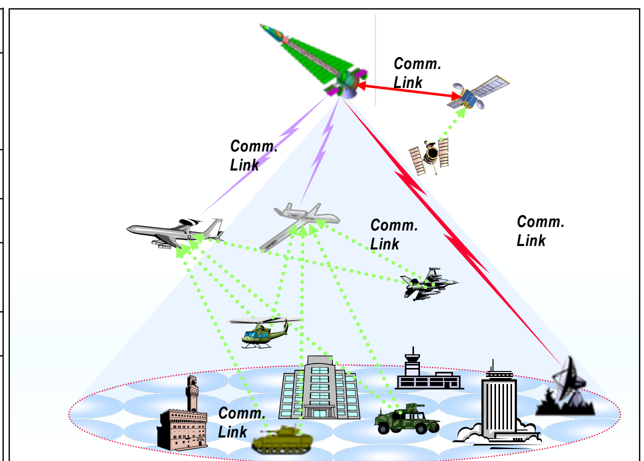
Figure 2-2 shows an extrapolation of the power system mass using a linear factor to 6,413 kg for 100 kWe and to 12,826 kg for 200 kWe. The 100 kWe power subsystem mass, coupled with the rest of the spacecraft mass (bus, antennas, station-keeping fuel, and so forth) exceeds the direct insertion launch capability for a Delta IV/Atlas heavy launch vehicles (~6,280 kg to GEO). The 200 kWe mass also exceeds the GEO transfer orbit capability (~12,000 kg) for these vehicles. So, as demand continues to increase and high-capacity signal processing (switching) and laser communications become key for robust anti-jam and ultra-secure point-to-point and broadcast communications, power loads on the satellite will begin to reach 100 kWe and beyond. At that a point, nuclear reactors will have to be included in the design mix to place these advanced systems into GEO orbit in a single launch.

Figure 2-3 demonstrates how much power was needed in 1993 (4.3 kWe) to power 16 transponders (120 W each) to move ~377 Mbps through bent pipe processing: an average of 0.2 Mbps/W performance. Transponder processing has improved greatly since 1993, from about 40 Mbps to 90 Mbps, with power of ~230 W at the spacecraft level needed to supply ~108 W at the transponder: 0.83 Mbps/W performance.

The world demand for 36 MHz space transponder equivalents is projected to exceed 7,150 by 2015 (Global Industry Analysts, Inc. 2011), requiring about 1.644 MWe in space. So, imagine that the entire world's capacity was handled by 200 kWe electric nuclear power satellites. One would only need 9 such satellites compared to 110 satellites at ~15 kWe each or 101 less launch vehicles at ~$300+ million each. This simple calculation demonstrates the immense savings potential of nuclear reactor power in space.

Figure 2-4 shows an ultra-high performing communication satellite concept with high-capacity point-to point, broadcast, cross-link capacity; high-speed cryo-electronic asynchronous transfer mode (ATM) switching; and on-board signal processing. A 200 kWe power capability would revolutionize communications to space and terrestrial deployed forces everywhere.

RF capability - space to ground		
Power (kWe)	Number of transponders (36 MHz equil. @ 230 watts each capable of 90 Mbps)	
40	174	15.6 Gbps
Laser Comm - space to space		
5	50 Mbps/watt	250 Gbps
Cryo electronic processing		
50	1000 to 1 cooling for 50 watts of 10 to 65 K electronics	On board latency free ATM switching and processing
Spacecraft		
105	Bus, propulsion, protection and reserve	



World demand in 2015 is 7150 transponders (36 Mhz equivalent at 90 Mbps) @ ~ 230 watts each requires ~ 1,644 kWe - all of space with nine nuclear reactor spacecraft instead of hundreds from a power point of view.

Figure 2-4. GEO Communication Capabilities With 200 kWe of Power

2. LO UAV Fence

Space-based AMTI has been a key feature in future space-based radar (SBR) constellation concepts (see Figure 2-5). To provide worldwide SBR coverage with a reasonable number of platforms, designers have focused on MEO operational orbits (~10,000 km) as opposed to LEO orbits (~1,000 km). Increasing the range from LEO to

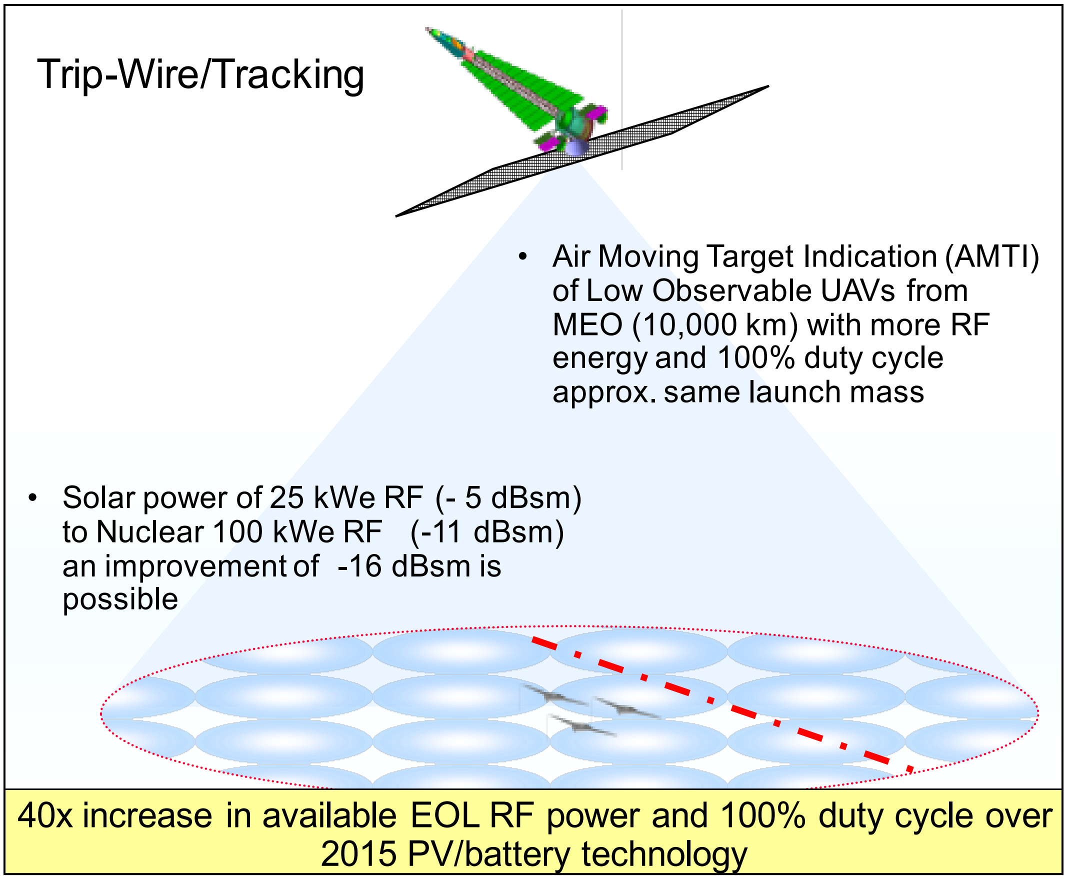


Figure 2-5. MEO Orbit AMTI and Tracking of LO UAVs and Cruise Missiles

MEO severely increases the power demand of the SBR system due to the range-squared losses. MEO SBR designs require a large aperture and kilowatts of power.

Using solar power subsystems, designers cannot allow 100% duty cycle (operations in eclipse) of such a radar payload. Such operations require a significant increase in battery mass, making the entire system too large/heavy to launch with a single vehicle. Nuclear reactors can operate in and out of eclipse. For example, a multi-beam, X-band, monostatic radar operating at 10,000 km would have a 40x increase in availability when compared to a solar power system meeting the same radio frequency (RF) requirements for detecting LO UAVs and cruise missiles.

3. Power Socket in Space

A nuclear reactor is an ideal way to provide the energy needed for space-based solid-state lasers. Solid-state lasers are becoming more efficient due to sustained DoD investments. They are projected to operate with 21% efficiency within the next decade. Figure 2-6 shows an estimated solid-state laser orbiting system that could produce 43 kW of laser power transmission potential from a 200 kWe power laser system. For example, a

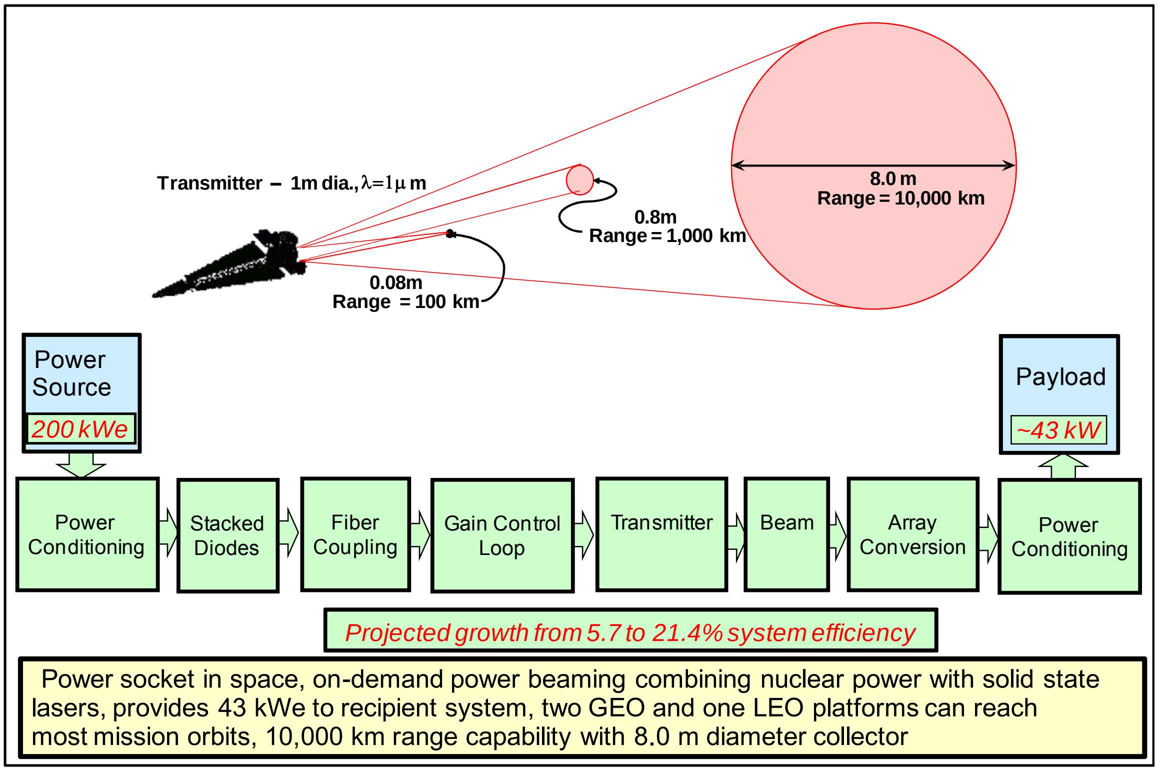


Figure 2-6. Space Power Beaming

design with a 1 m diameter primary transmission optic with λ = 1 μm could produce an energy spot-size of 0.8 m at 1,000 km and 8 m at 10,000 km. To be safe, the wavelength would be matched to the atmospheric absorption line, and the high-powered beam would be enclosed with a low-power beam used to align the transmitting and receiving spacecraft electro-optical systems.

With three such 200 kWe fission reactor laser power-beaming stations (one LEO, two GEO satellites), tens of kilowatts of electrical energy (kWe) of instantaneous usable power would be available for 5 to 10 min. One LEO power-beaming satellite in a 1,500 km circular orbit at an inclination of 0 deg would provide access to all LEO assets but would be time constrained by earth shadowing. Two GEO power-beaming satellites with an inclination of 175 deg circular orbit slightly inside GEO orbit (retrograde) would have access to all GEO assets. However, these satellites would be time constrained by a maximum range requirement of 10,000 km. Satellites that receive energy from power-beaming satellites at 10,000 km range would need an ~8 m diameter ultra-lightweight receiver array and high-capacity flywheel energy storage. Such an architecture would provide indiscernible operations and geometric and timing agility power to transfer concept of operations (CONOPS) to receiving assets.

4. HAND Radiation Mitigation

A nuclear weapon detonation in LEO used as an anti-satellite (ASAT) weapon and a mechanism to disrupt communications or as a force demonstration between warring third world countries will produce a high-intensity, man-made radiation belt (see Figure 2-7). The energetic particles (electrons and ions) from the HAND event will become trapped due to the dipole structure of the earth's magnetic field, creating a man-made radiation belt that will spread around the earth like a thin shell (Hoyt 2008). This intense belt of primarily megaelectronvolt (MeV) energetic electrons will rapidly age all existing DoD, IC, allied, and commercial LEO satellites not destroyed by the initial blast. One way to mitigate this man-made radiation belt rapidly is to develop a method to perturb the energetic electrons in such a manner that they scatter (i.e., by reducing the "pitch angle" between their velocity vector and the geomagnetic field vector). Depending on how the pitch angle is perturbed, the energetic particle either will spiral into the "loss cone" or be reflected back along the magnetic field lines. Particles that enter the loss cone dissipate harmlessly into the earth's upper atmosphere.

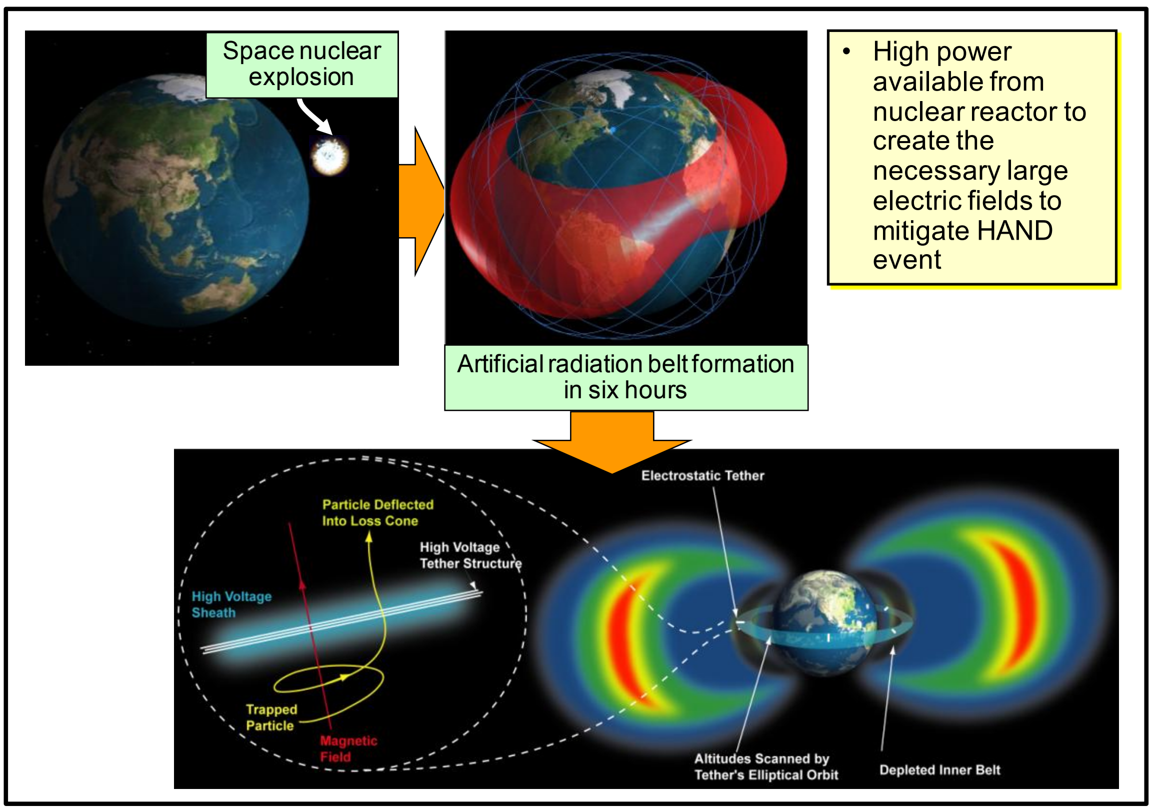


Figure 2-7. HAND Radiation Belt Remediation Using Electrostatic (ES) Tethers

Source: Hoyt and Minor 2005; Hoyt and Cash 2007.

Perturbing these energetic electrons traveling along the magnetic field lines could be done with a large (260 m in diameter 50 km long) LEO ES field (sheath within ionosphere plasma). Since the energetic particles travel back and forth between the poles and around the earth's circumference near the speed of light, they eventually all interact with this ES field. As they travel through the ES field, they are perturbed into the loss cone or are reflected back along the magnetic field lines. Such a field can be generated by an ES tether system if enough power is available. Figure 2-8 shows details of an ES tether concept. Hoyt (2008) describes how 10 such unique orbiting ES systems (50 km long tether with an egg beater design of 25 wires and at 100 kV potential) powered by 40 kWe could reduce the radiation flux of a 50 kT HAND event that produced a 1,000 km wide man-made radiation belt to 1/e of its initial value in just 12 days.

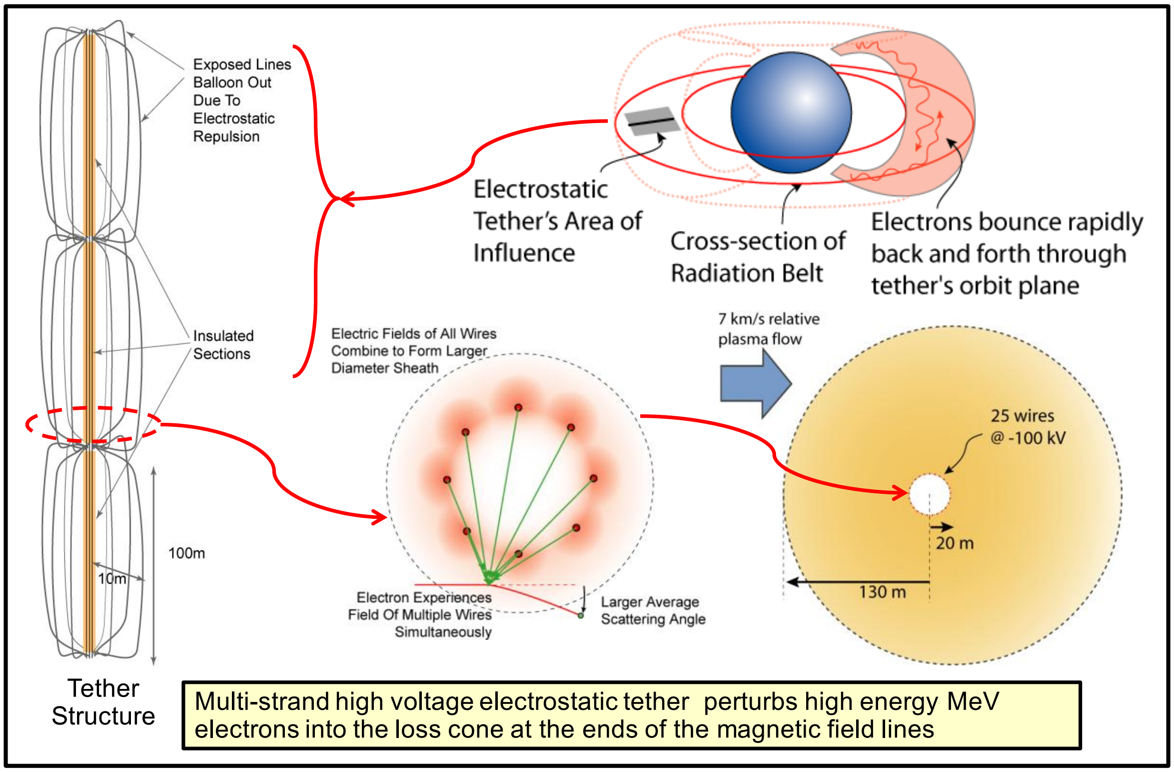


Figure 2-8. ES Tethers Concept

Source: Hoyt and Minor 2005; Hoyt and Cash 2007.

Given 200 kWe of power available to the ES system design described by Hoyt (2008) and assuming a 50% efficient high-voltage power supply with a deployed tether system bias voltage increased to 500 kV, the ES sheath diameter would grow from ~260 m to ~800 m for structure that is 50 km long. With three deployed nuclear-powered systems available in LEO, the same 50 kT HAND man-made 1,000 km wide radiation belt would now be reduced to 1% of its original flux in 1 day instead of 12 days. This architecture could handle multiple HAND events, assuming the three nuclear-powered spacecraft survive the local detonation effects.

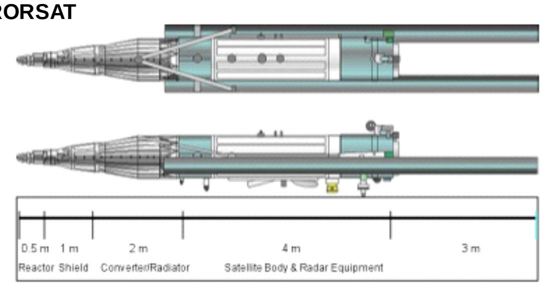
Reviewing and extrapolating from various experimental conceptual designs of such an ES system (Hoyt and Minor 2005; Hoyt and Cash 2007), such a system would have a maximum estimated mass of 750 kg. It could conceivably serve as a secondary payload on a LEO-based, NEP-driven AMTI phase array radar or power-beaming spacecraft discussed earlier.

Without a strong mission pull and an innovative design, development, test, and evaluation approach, space nuclear reactor technology will not have the support to overcome the residual political resistance or to secure the necessary funds. Russia and China do not have these political restrictions and could field space nuclear reactors within the next decade if they so desire. The Former Soviet Union (FSU) has flown about 31+ reactors for military intelligence applications in Radar Ocean Reconnaissance Satellites (RORSATs) since 1965, while the United States has flown only 1 (the Systems for Nuclear Auxiliary Power (SNAP)-10) for a short period of time (Aftergood 1989) (see Figure 2-9). When the Soviet Union dissolved into the Russian federation, their program was put on hold (around 1987). However, recent statements by Russia imply that it is rejuvenating its space nuclear reactor program:

- May have invested $170 million in 2010 for space nuclear propulsion and power generation in the mega-watt class and is considering additional investments of ~$600 million over the next 9 years (Madrigal 2009).
- Starting to work on standardized modules with nuclear-powered propulsion systems involving 150 to 500 kWe devices.
- Planning for concept designs around 2011 leading to testing in 2018 and possible launch in 2020 (World Nuclear Association 2011).

	USA	USSR
	SNAP-10A	RORSAT
Initial Flight	1965	1967
Number Flown	1	31+
Power KWt	46	~100
Power KWe	0.65	~5
Conversion	Thermoelectric	Thermoelectric
Fuel	U-ZrHx	U-Mo
kg^{235}U	4.3	25 - 35
Mass (kg)	435	~390
Coolant	NaK	NaK
~ α (kg/kWe)	670	78

RORSAT



SNAP-10A

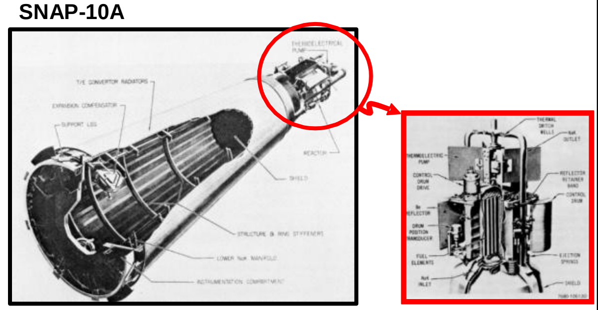


Figure 2-9. Nuclear Space Reactors That Have Actually Flown in Space

Source: Angelo and Buden 1985; Kulcinski 2004.

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3. Solar vs. Nuclear for High-Power Applications

A comparison must be made between solar photovoltaics and nuclear-power approaches before exploring NEP systems as a critical technology for space dominance. The sun provides 1,367 W/m^2 for collection and conversion into electric power. A reasonable size array (13 × 13 m) would appear to solve the problem of collecting 200 kWe of power. However, this is not the case if solar cells are used to collect the energy to power the satellite electronics and recharge its batteries. The need to recharge batteries comes from requiring power when the satellite is in the eclipse of the earth. Depending on the orbit, the eclipse time can be periodic and short, aperiodic and long, or somewhere in between. Whatever the case, the solar arrays have to be large enough to recharge the batteries and to run the payloads.

Solar cells that make up the arrays have improved greatly over the last 3 decades. They had conversion efficiencies of about 7% in 1970 and are predicted to approach 35% efficiency in the near future. Over the last decade, cells at around 28% have been economically manufactured due to large government and industry investments (Sharps et al. 2010). However, solar cells deteriorate due to the harsh space environment (radiation) and can lose their efficiency with improper thermal management. For example, depending on array design and space environmental effects, a 28% efficient cell would most likely operate at 18% at end-of-life (EOL) (Wertz and Larson 1999).

Figure 3-1 depicts the operating regimes for different types of space power sources. This figure, developed decades ago, is still valid even though solar cell system capabilities have steadily improved over the last 5 decades. At this time, solar cell systems do not appear to be practical much above 100 kWe because they require huge solar array structures and, depending on their orbit, massive amounts of batteries.

An important parameter when comparing space power systems is “α,” or “specific mass kg/kWe” (Angelo and Buden 1985). The smaller the α, the less mass the power system needs to generate 1,000 W (1 kWe) of electric power, resulting in immense cost savings for launching the system into orbit. Table 3-1 is a comparison of solar and nuclear power systems’ projected mass and solar array or radiator size and α for a LEO satellite operating 100% of the time with a ~30 min eclipse per orbit. Two solar power systems’ capabilities are noted: one that can be built today and a future system that

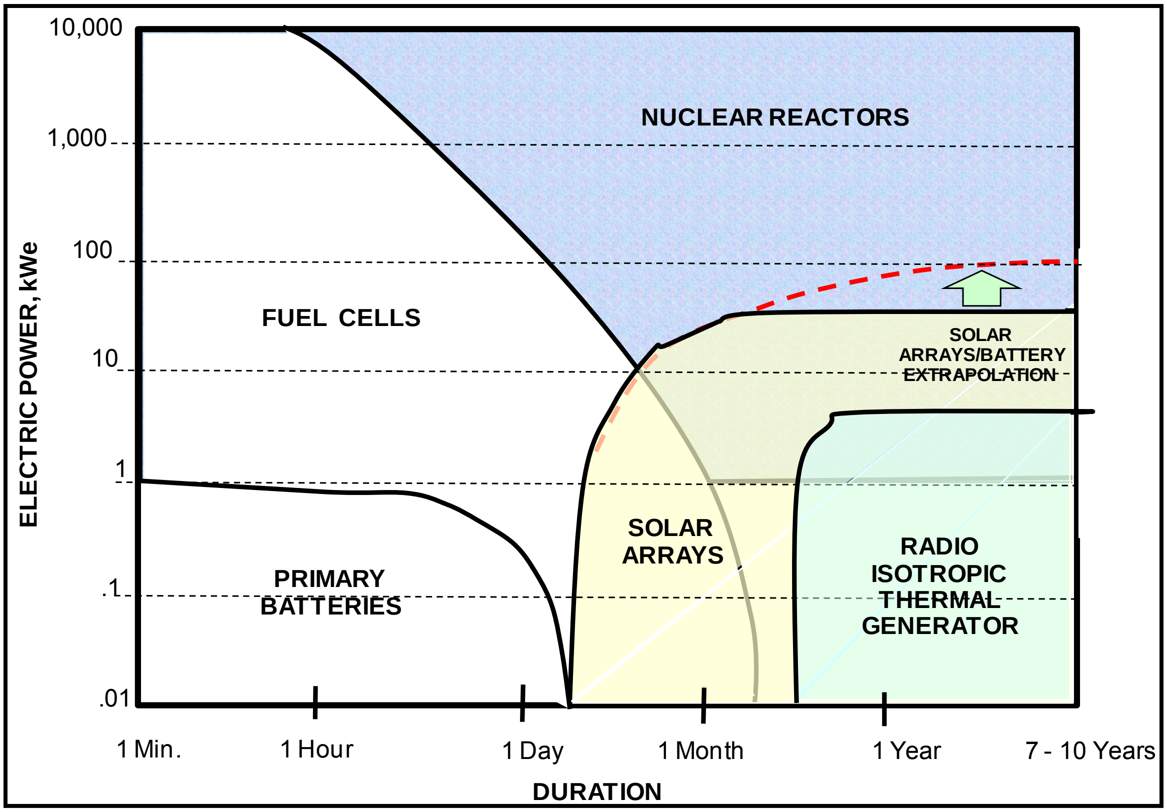

Figure 3-1. Where Nuclear Reactors Fit in Space Power Applications

Source: Aftergood 1989.

assumes 35% efficient cells and advanced 150 W-hr/kg lithium (Li) batteries. What is noted by Table 3-1 and shown in Figure 3-2 is that the solar array size becomes unwieldy compared to the nuclear power radiator size. As the solar array becomes very large, the deployment and support structure parasitic mass degrades the areal mass (kg/m^2) of the array, making the entire subsystem heavy. In addition, the packaging of such a large structure into a single launch vehicle becomes a critical issue to avoid because of the prohibitive cost of multiple launches and on-orbit assembly (e.g., the International Space Station (ISS)).

Another extremely challenging problem with huge array structures is the ability to design the satellite control system with confidence. Such extremely large structures are affected by solar pressure, non-uniform heating/thermal shock as the satellite passes through eclipse, and a multitude of inertial forces. These large structures are difficult—if not impossible—to test in a 1 g terrestrial environment. Such testing is required to provide confidence in the structural dynamic modeling that predicts their 0 g stability behavior during space deployment and operations. Finally, designing appropriate electrical power collection architectures for such large arrays to minimize ohmic losses becomes challenging.

Table 3-1. Comparison of Solar and Nuclear Power System Projected Capabilities

	25 kWe (100 kWt)			100 kWe (400 kWt)			200 kWe (800 kWt)			
	Solar		Nuclear	Solar		Nuclear	Solar		Nuclear	
	Today	Future		Today	Future		Today	Future		
Mass (kg)	1,955	815	1,000	7,819	3,261	4,000	15,638	6,521	8,000	6,000
Operating Temperature (K)	NA	NA	1000	NA	Na	1150	Na	NA	1150	1400
Solar or radiator size (m^2)	211	108	~81	844	434	~132	1,688	868	~265	~128
α (kg/kWe)	78	33	40	78	33	40	78	33	40	32

Note 1 for Table 3-1: Solar Array Technology Assumptions
Today ~ NiH_2 batteries: 50 W-hr/kg with 40% depth-of-discharge; 18% gallium arsenide (GaAs) cells (80% EOL)
Future ~ Advanced Li batteries: 150 W-hr/kg with 40% depth-of-discharge; 35% multi-junction cells (80% at EOL)
Note 2 for Table 3-1: Radiator Technology Assumptions
ROM ~ Emissivity 0.85, 4 to 1 thermal/electric watts overall nuclear system conversion
Radiator temperature: 25 kWe system (400 K), 100 kWe system (500 K), 200 kWe system (500–600 K)
Note 3 for Table 3-1: Nuclear power systems become more practical above 100 kWe at 100% duty cycle, even with advances in solar power technology, primarily due to radiator vs. solar array size
Note 4 for Table 3-1: Gray shading indicates superior value.

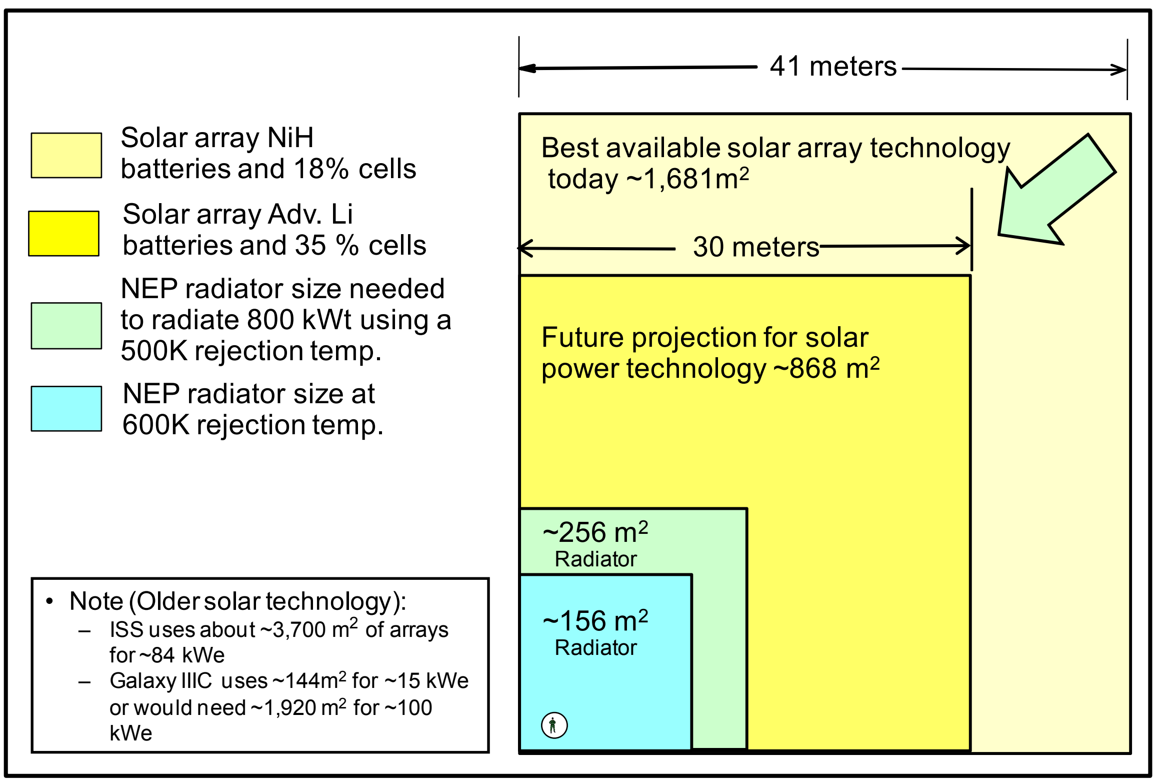

Figure 3-2. A 200 kWe Solar Array Size vs. NEP Radiator Size
Source: Angelo and Buden 1985; Wertz and Larson 1999.

One way to reduce the size of the huge solar arrays is by using solar-concentrator-type photovoltaics or thermal designs. Prototype solar thermal systems have shown promise in developing hundreds of kilowatts of electrical energy of power in space, but these systems require constant high-precision pointing and tracking of their concentrator optics toward the sun. Such systems also need batteries to operate in eclipse. Given these constraints, such systems have not yet evolved into designs over a few kilowatts of electrical energy that would satisfy the greater than 100 kWe requirements with the operational flexibility offered by nuclear power.

For the moment, nuclear reactor systems appear to be the only viable space power source for providing more than 100 kWe. What follows is a suggested approach to reenergize this technology within the next decade, thereby providing the United States a capability for space dominance for the foreseeable future.

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4. Recent Attempts To Revive Nuclear Space Reactors

Two significantly funded U.S. programs (greater than $100 million) over the past 30 years have attempted to orbit space nuclear reactors with estimated power output from 10 kWe to a mega watt. Figure 4-1 is a rough timeline of aircraft and space nuclear power programs since the 1950s. At present, advanced design and manufacturing work in the United States is limited to low power (less than 500 We) Radioisotope Thermoelectric Generators (RTGs).

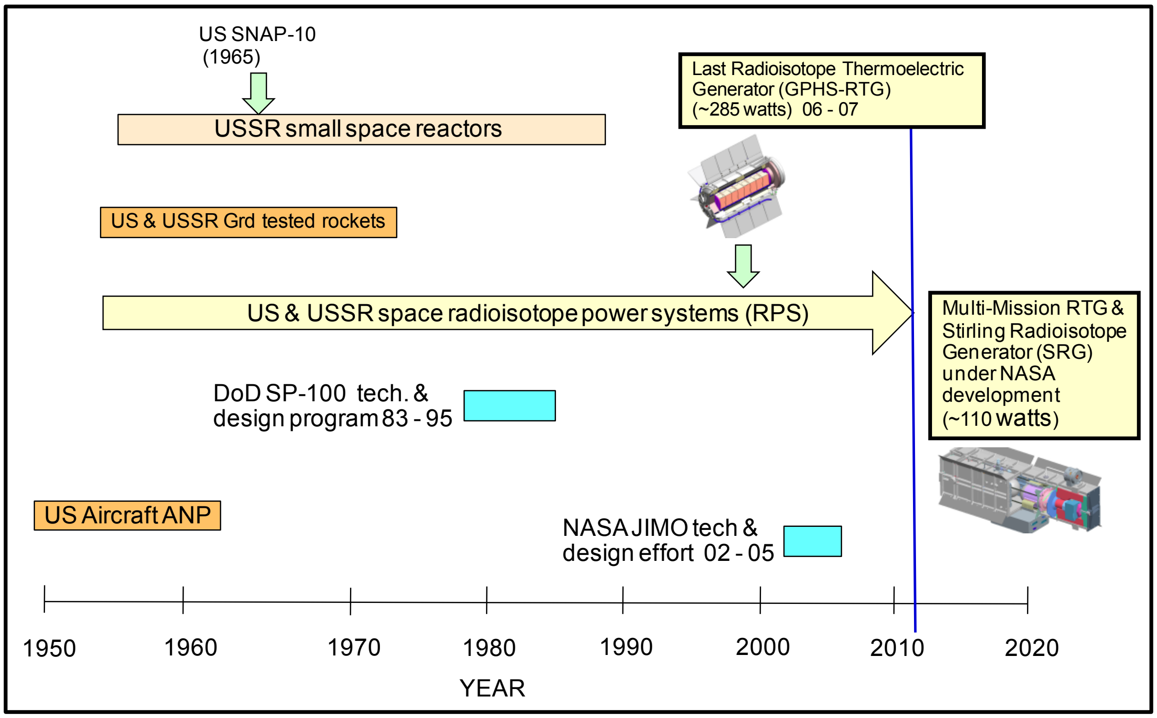


Figure 4-1. Nuclear Aerospace Programs Over the Last 6 Decades

Source: Abelson et al. 2005.

The goal of the SP-100 program was to develop a modular nuclear reactor design that could provide from 10 kWe to 1 MWe power. The program was funded from about 1983 to 1995. The goal of the Prometheus program (initially known as the JIMO program) was the development of a 200 kWe system and funding of the program 2002 to 2005. These programs had some common features and notable differences that drove their design requirements, costs, schedules, and program management and acquisition approaches. A brief description of each program, with observations on what may have contributed to investor disinterest, follows.

A. The SP-100 Program

The SP-100 program had its roots in the Space Power Advanced Reactor (SPAR) program initiated in 1979. In 1983, the SPAR program became the SP-100, with initial requirements of a modular design from 10 to 100 kWe, a 7 to 10 year life, and sized to fit in the shuttle including room for a kick-stage with an α of 20 to 30 kg/kWe (Angelo and Buden 1985; Buden 1993). Early reactors had a typical α of 670 to 130 (Angelo and Buden 1985). However, advancements in reactor design, materials, energy extraction, and waste-heat removal technologies and reactors that can operate at higher temperatures may eventually result in α's from 30 to 40. At these α levels, nuclear space power technology is superior to solar-based systems above 100 kWe.

In 1983, the SP-100 program support came from a coalition of National Space and Aeronautics Administration (NASA), Department of Energy (DOE), and Defense Advanced Research Projects Agency (DARPA) sponsors interested in jointly developing space nuclear power technology (see Figure 4-2). To make this partnership work, the SP-100 program created a steering committee made of representatives from all the sponsors. The steering committee provided programmatic direction to the program director, who forwarded the guidance to the program manager. The program director also received guidance directly from the Strategic Defense Initiative Organization (SDIO) director. DoD (SDIO) and NASA provided the missions, NASA developed the power conversion, and DOE handled the nuclear parts of the program (Demuth 2003).

The strategy of the program was stated as identifying potential users and space mission applications that required large amounts of power, determining the power system functional requirements from the mission requirements, and, developing system concepts that are bounded by safety and other design constraints. Any critical technical issues that arose from the design concepts were identified through analysis and experiment. This approach made the SP-100 design relevant to many organizations driven by different mission requirements that were in flux, although this tactical strategy may have contributed to adding a degree of complexity to the design that adversely affected program cost and schedule.

As the program evolved, the upper power design limit was changed from 100 kWe to one megawatt to satisfy some of the SDIO mission requirements shown in Table 4-1 (Aftergood 1989). The configuration stabilized around 1985, as shown in Figure 4-3 (Kulcinski 2004). Numerous references (Angelo and Buden 1985; Bennett et al. 1996; Kulcinski 2004; Bennett 2006) detail all the design features, subsystem technologies, and test programs conducted during SP-100 program.

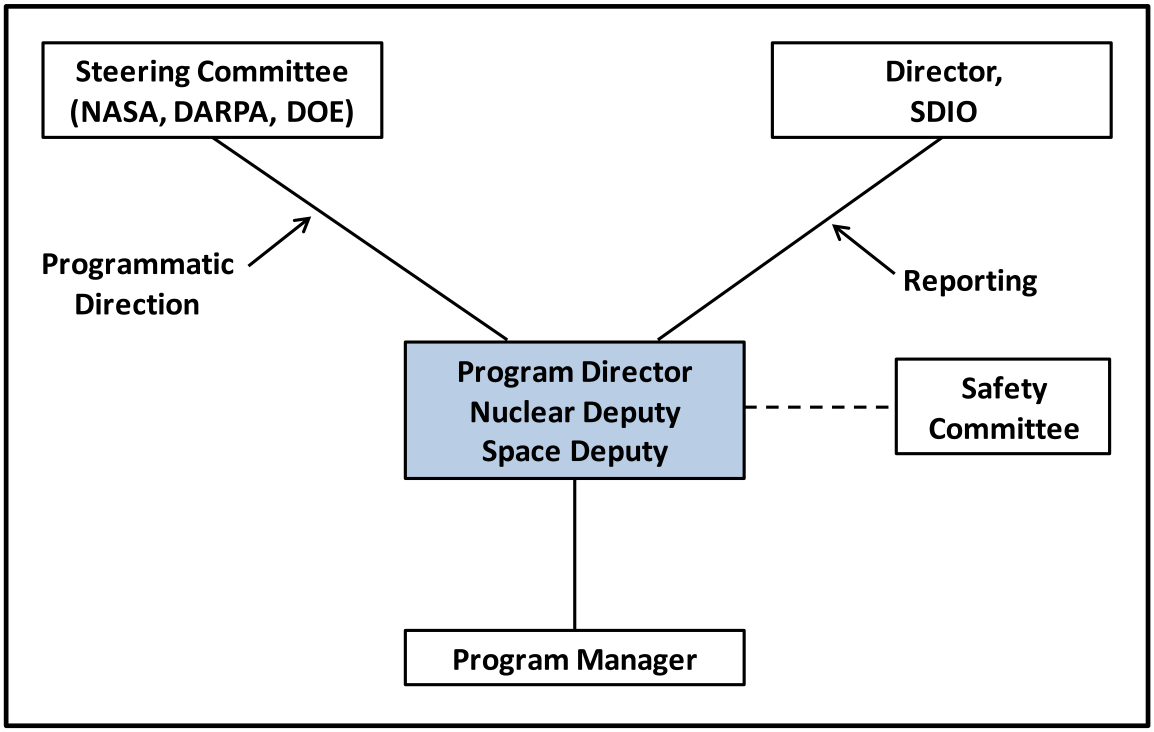


Figure 4-2. SP-100 Program Organizational Relationships

Source: Angelo and Buden 1985.

Unfortunately, the SDIO missions were the first to go away, and then the NASA planetary missions were put on hold. As the government missions began to evaporate and key stakeholders lost interest, a 20 to 40 kWe variant design (for commercial communication applications in GEO) was pursued up to detailed design. The program was terminated in 1995 (Demuth 2003). Depending on the reference cited, somewhere between \$734 million[4] to \$1.4 billion[4] were spent on the program across a complex infrastructure of industry and national laboratories.

Key features of the SP-100 (100 kWe) design after several system tradeoff studies (Angelo and Buden 1985; Demuth 2003; Bennett et al. 1996) were as follows:

- Uranium nitride (UN)-fueled core reactor sealed in rhenium (Re)-lined niobium with 0.1% zirconium (Nb-1Zr) cladding or, as actually done later (improved high-temperature creep performance), PWC-11 (Nb-1Zr with 0.1% carbon cladding).
- Operating with a fast-neutron spectrum; the reactor design used about 190 kg of fuel in 984 fuel pin-like structures. Note that the Re was used to act as a neutron poisoning agent (reduce reactivity) if the reactor core was submerged in water due to an accident.

[4] 1995 to 2008 1.34 inflation multiplication factor.

Table 4-1. Space Power Requirements for the SDIO From 1970 to 1990

SDIO/ Ballistic Missile Defense Organization(BMDO)	Modes of Operation Base/Alert/Burst (Est. Avg. Kilowatts of Electrical Energy)
Boost and space surveillance/tracking satellite	5/10/50
Laser radar (LADAR)/imager	15/20/500
Laser illumination Doppler LADAR	5/10/100 15/20/600
Space-based interceptor carrier	2/50/100
Directed energy weapon (DEW) (chemical laser, fighting mirror, neutral particle beam/space-based free electron laser (FEL)	10/10,000/5 x 10^5
Railguns	10/10,000/5 x 10^6

Source: U. S. Congress 1988.

Note for Table 4-1**:** In addition to SDIO power requirements, the NASA deep space travel to Jupiter and outer planets and the Soviet space supremacy challenge were drivers of space power needs.

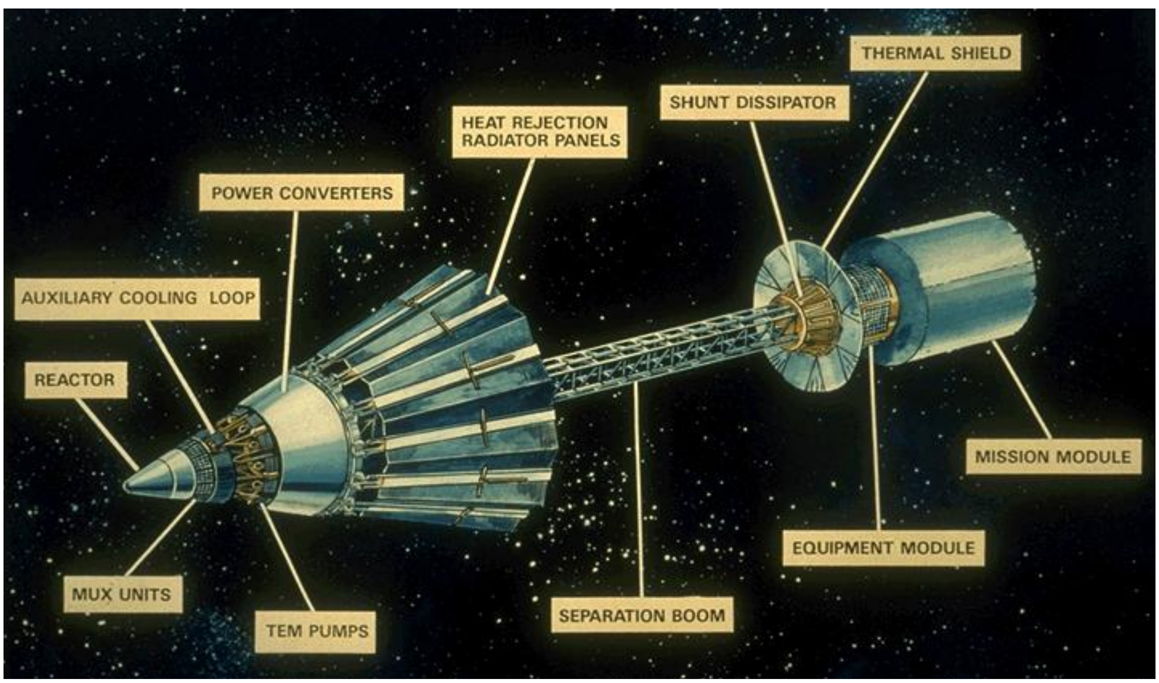


Figure 4-3. One of the Final Design Concepts for the SP-100 Nuclear-Powered Satellite

Source: Bennett 2006 (Original source: The Jet Propulsion Laboratory (JPL).

- Liquid Li metal-cooled reactor operating at 1,375 K. The reactor is launched with the Li metal frozen solid. Upon reaching orbit, the metal is heated to a liquid. A similar concept was used on the only U.S. reactor to fly, the SNAP-10 reactor.
- Passive energy extraction with no moving parts, using thermoelectric conversion with 12 distributed panels. Each panel consisted of 720 cells that produced 1.5 kWe per panel.
- Revolutionary satellite heat-rejection radiator design using liquid Li metal in titanium (Ti) heat pipes surrounded by carbon-carbon (C/C) for protection against micro-meteorite and debris. The radiator was designed with a heat-rejection temperature of 800 K, requiring about 106 m^2 in area. A reactor power of 2.3 MWt was needed to convert to 100 kWe of electric power.
- The reactor core was put in a safe mode during launch using three boron carbide (B_4C) control rods, but the design eventually grew to seven rods that were locked in place.
- Control during operations was accomplished using 12 sliding beryllium (Be) reflector elements. The final design had two independent shutdown mechanisms: spring-loaded reflector elements and safety rods that activate upon loss of electrical power.
- Shielding was accomplished using a structure consisting of alternate layers of B_4C, lithium hydride (LiH), and tungsten (W) and had a mass of about 1,000 kg.
- Mass goal in 1992 was about 4,000 kg, but various references (Angelo and Buden 1985; Kulcinski 2004; and Bennett et al. 1996) put the mass at 4,600 kg or even 5,422 kg. As mentioned previously, the projected reactor system mass is critical to determining α (kg/kWe).
- Overall system reliability goal was 0.95.
- At the end of the power system's life, the SP-100 was to be parked permanently in a high orbit so that it would not deorbit and reach the earth for thousands of years.

The SP-100 program development effort spent a lot of time and effort designing the nuclear reactor's major components and subsystems as modular systems. The goal was to make the entire concept adaptable in several different ways and to vary the performance of the design to meet different mission requirements. Figure 4-4 is a sketch of how the SP-100 concept might have looked for different power ratings (Demuth 2003). The idea was to couple the reactor, the shield, and the heat-rejection radiators to different energy-extraction devices to get different power conversion efficiencies. SP-100 was also

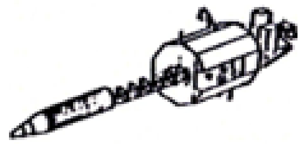
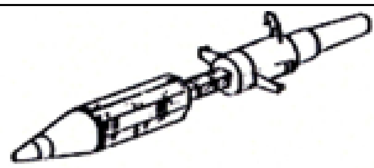
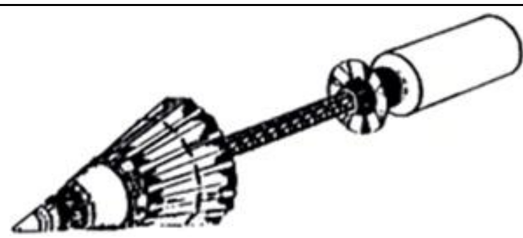
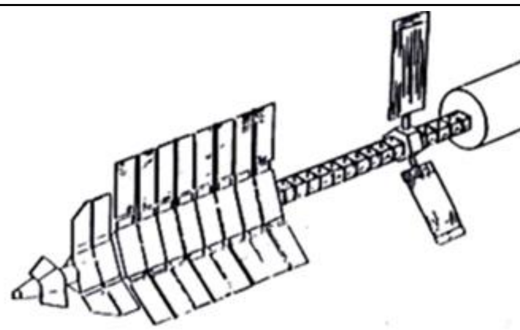
	**10 kWe Low-Power Military Mission**
	**30 kWe EP**
	**100 kWe Generic Flight System**
	**1,000 kWe Stirling Conversion Generic Flight System (GFS) Reactor**

Figure 4-4. SP-100 Design Variations From 10 kWe to 1 MWe

Source: Artwork courtesy of JPL.

designed to be verified functionally at full operational temperature before launch without operating the reactor, be compatible with a wide variety of space environments, and be launched on several different vehicles. All of these requirements to meet the different mission needs of NASA and the DoD, coupled with the modular construct, led to a highly sophisticated design that demanded significant technology developments. These factors increased program cost and schedule.

This programmatic approach may have also contributed to a classic “time-to-market” failure. Because the SP-100 took so long to develop, its critical mission pull evaporated. The SP-100 program never had a complete end-to-end[5] prototype system built for system capabilities testing in a space-simulated vacuum environment. Such a test could have been a way to “wow” stakeholders and investors and demonstrate full system-level capabilities as a return on their multi-year investment. Historically, creating

[5] For example, reactor plus power extraction, thermal management, power management and distribution (PMAD), and simulated payload subsystem loads.

conditions that stretch the development of an expensive space program increases the risk of premature cancellation. The program may not only lose its relevance to the initial mission requirements, but also lose key government stakeholders as they move on to other assignments. This systemic problem in executing long-term space acquisition programs will be addressed next. Using a rapid prototyping approach to get to full system prototype testing quickly may be a way to maintain investor keen interest as they see the system take shape and undergo test and evaluation (T&E).

B. Prometheus/JIMO Project

NASA's Nuclear Systems Initiative had funded some pre-project work under the title of the Jupiter Icy Moon Orbiter, or JIMO, project starting in September of 2002 and completed several initial mission studies, detailed technical analyses, and industry surveys by early 2003. The results of these studies were encouraging, and JIMO was slated for a FY04 (October 2003) start. Congress was excited about JIMO and provided $20 million of FY03 funds to jump-start the program in March 2003. The project was renamed Prometheus in March 2003 and eventually transferred to the newly established NASA Exploration Systems Mission Directorate in February 2004. The goal of this program was to provide power and EP for deep solar system exploration, where minimal solar power is available and chemical propulsion limits the maneuverability and destinations. At the time, the Bush administration was committed to supporting nuclear EP and power for deep-space NASA missions (Taylor 2005).

The primary goal was to develop a Deep Space Vehicle (DSV) for outer solar system exploration by combining a space nuclear reactor with EP, thereby creating a safe and reliable NEP. The DSV was to contain a payload accommodation envelope of about 1,500 kg for science instruments and support subsystems. Figure 4-5 is an example of a 100 kWe design concepts for the Prometheus/JIMO mission.

Included in the DSV design features were nuclear reactor technologies that would also be extendable to Lunar and Martian surface operations. Due to the almost 7 year gap between the demise of the SP-100 program and the trickle start of the Prometheus program, NASA felt that these design goals required significant nuclear reactor technology advances in reactor, energy conversion, heat rejection, and EP (Taylor 2005). Furthermore, NASA soon realized that no one organization possessed all the required skills in these critical areas. This realization forced it to reach out to others to build the Prometheus team. These early program decisions resulted in a large team that consisted of NASA HQ, the JPL, the DOE Office of Nuclear Energy, 4 DOE Naval Reactor (NR) laboratories, 8 other NASA centers, Northrop Grumman Space Technology (NGST), 6 supporting DOE laboratories, 31 universities, and 14 industrial subcontractors. This

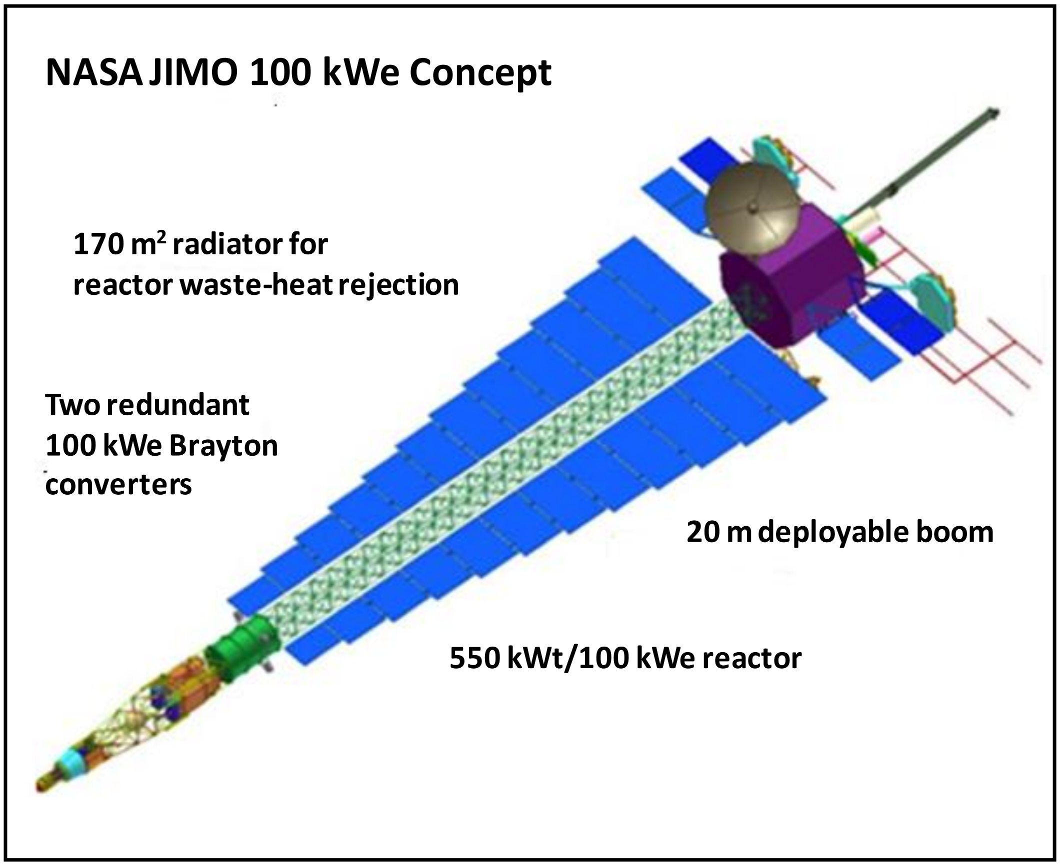


Figure 4-5. Prometheus Candidate Design

Source: Taylor 2005.

team eventually completed a Phase A study, spending in the neighborhood of $128.5 million in project and mission planning, analysis of alternatives, design, subsystem analysis, conferences, and component experimentation. NASA's decision to join with DOE NR labs to produce the nuclear reactor was interesting. Although the NR organization had built many successful submarine and surface ship reactors, it had never built a space nuclear reactor. This factor would have required the NR designers to make a serious adjustment to their design approach since they did not have the oceans for getting rid of the reactor and energy conversion system waste heat. They would have been restricted to space radiators with stringent mass and launch vehicle packaging constraints, micrometeorite and debris survivability, and radiative cooling sensitivity driven by the 4th power of the radiator temperature.

In FY05, NASA reprioritized its major programs and postponed its nuclear initiatives. Ongoing nuclear technology efforts were affected because NASA put NEP beneath nuclear surface power and nuclear thermal propulsion. These budgetary actions resulted in the termination of the Prometheus program in October 2005.

Figure 4-6 displays the complexity of the team's relationships and the project organization chart. All of these management interactions can be inherently costly and can cause excessive delays in critical decision making. The Prometheus team appeared to grow too quickly and eventually had to develop a Responsibilities Assignment Matrix that specified which organization was responsible for each element within the work breakdown structure (Taylor 2005).

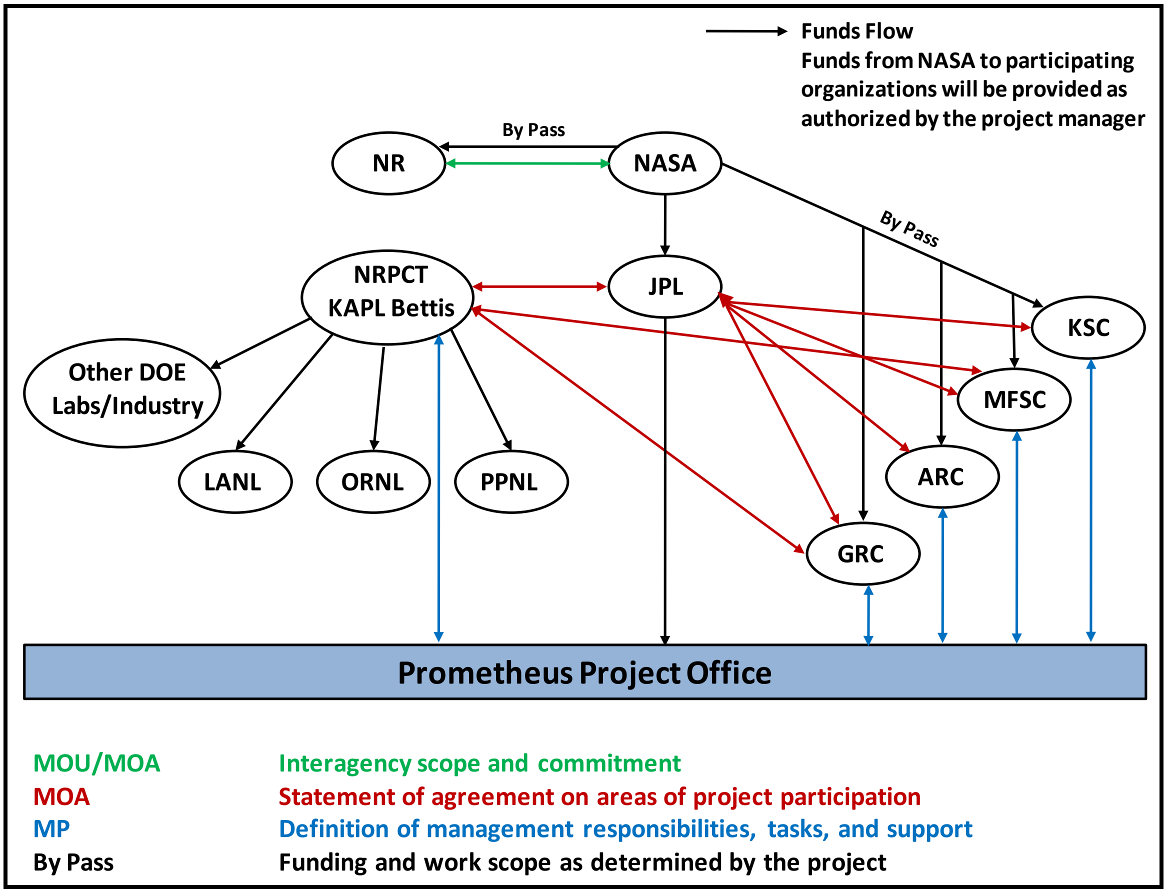


Figure 4-6. Prometheus Multitude of Government Interactions

Source: Figure adapted from Taylor 2005.

The integration of so many government, industry, and academic organizations cannot be conducive to creating a cost-effective, agile, efficient organization. One way to minimize this issue in the future would be to require prime industry bidders to build the required expertise within their teams. Part of the proposal evaluation would concentrate on the breadth and depth of their technical expertise for such a project. As will be discussed later, doing as much end-to-end system-level prototyping in a non-nuclear space simulation environment before committing to a flight program that requires the launch safety review process would help to reduce the team's size in the early program phases.

The JIMO mission was planned to be launched in 2015 or almost 12 years after the Phase A study. This planned amount of time might have been excessive given today's turbulent budgetary environment. In addition, key administration and congressional stakeholders would most certainly have changed over that period. A way to shorten the development time for space nuclear power and propulsion subsystems has to be found, or these systems may never be developed in the United States.

Another interesting observation from Taylor (2005) was NASA's decision at the program onset to address a large number of JIMO mission technology challenges simultaneously. That decision may have resulted in too many diverse component and subsystem technology investments, which quickly dried up the initial funding. Efforts in technology areas such as high-power telecommunication, radiation-hardened electronics, and low-thrust trajectory tools could have been delayed to focus the initial investments on the reactor and power conversion engineering. Without the NEP, no one was going anywhere. Mature off-the-shelf heat-rejection and EP systems were available at that time and could have been used for a scaled engineering unit, end-to-end NEP demonstration.

Large, diverse, and sometimes competing government organizations forced to form a team, a long product development time, excessive academic and industrial participation, and broad brush initial technology investments may have added high risk to the entire effort no matter how talented the NASA management team was. One wonders whether a different project management approach, such as initially decoupling the actual JIMO mission detail requirements from the NEP, could have provided resources to build a prototype rapidly for a scaled NEP demonstration. Such a demonstration consisting of an end-to-end non-nuclear (electric heaters) NEP engineering unit may have made the case to NASA stakeholders that the NEP development had reached a level of momentum worth continuing.

Some of the key features of the DSV (Taylor 2005) were as follows:

- A high-temperature, gas-cooled reactor was directly coupled with redundant pairs of Brayton turbo-alternators produced 200 kWe of power.
- Similar to the SP-100, a radiation shield concept was designed to produce a conical shadow over the rest of the vehicle, protecting it from the reactor.
- A long structural boom supported the two-dimensional (2D) array of heat-rejection radiators and separated the payload electronics from the reactor.
- The thruster power goal was 180 kWe, using Ion and Hall thrusters mounted on two deployable thruster pods (Isp goal was 5,000 to 8,000 sec.).
- A docking interface was available for on-orbit docking activities when in earth orbit.
- The estimated mass of the reactor module was 3,309 kg, and the heat-rejection segment mass was about 2,566 kg. Therefore, α would have been ~30.

- The conceptual design wet mass was 36,375 kg (including a 30% margin). That mass included 1,500 kg for the payload and 12,000 kg for Xenon EP propellant.
- Three launches were necessary to assemble the complete spacecraft in orbit.
- An Aero-shell (jettisoned before reactor start up) was designed as a safety device to keep reactor together during an unplanned reentry back to earth.
- The nuclear reactor was designed with a life goal of 20 years—10 years at full capability plus 10 years at reduced power.

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5. Space Nuclear Reactors and Electric Propulsion

Figure 5-1 compares the SP-100, the Prometheus/JIMO spacecraft, and a typical design layout of a nuclear reactor spacecraft. Common characteristics are the reactor, shield, energy conversion, waste-heat disposal radiators, and the power management and distribution (PMAD) subsystems separated by a boom-like structure from the payloads. For the SP-100, the thermal radiators form a three-dimensional (3D) cone-like structure from the shield. The JIMO design has the radiators spread out in a 2D flat-panel-like geometry.

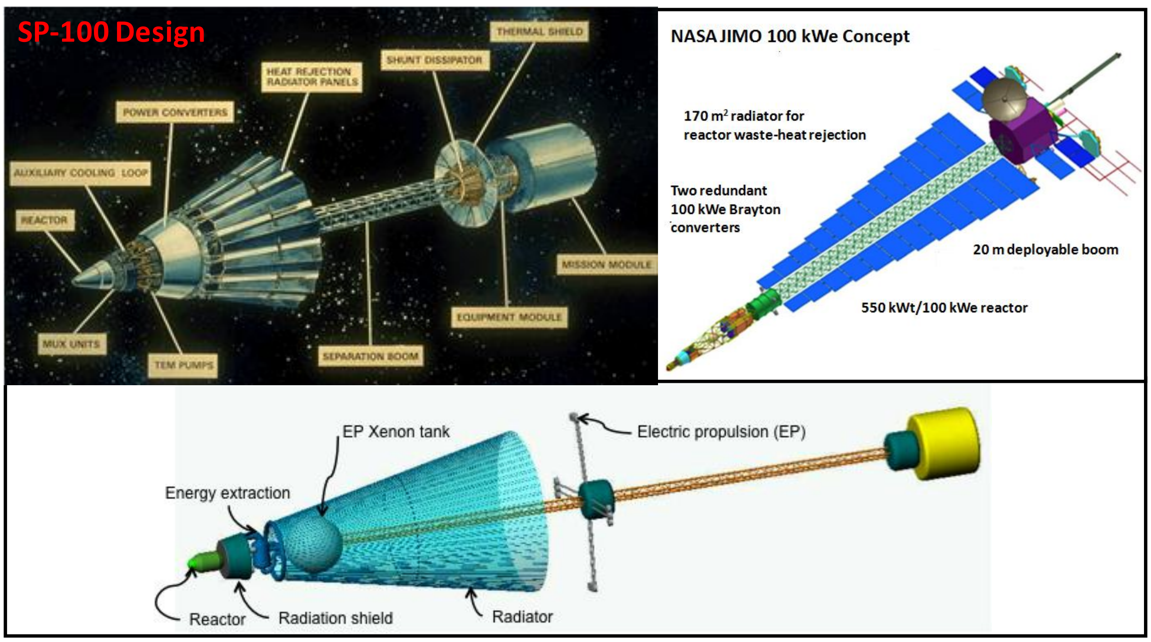


Figure 5-1. Typical Space Nuclear Power Design Concepts

Source: Angelo and Buden 1985; Taylor 2005. Artwork courtesy of Los Alamos National Laboratory (LANL).

For discussions that follow, Figure 5-2 is a NEP-like system with key subsystems identified in a block-like subsystem sketch.

In general, space nuclear reactors can be grouped into roughly three categories: liquid metal-cooled, gas-cooled, and heat-pipe-cooled devices. Figure 5-3 contains artist sketches of the three types and a summary description of the SP-100 and JIMO key program design features.

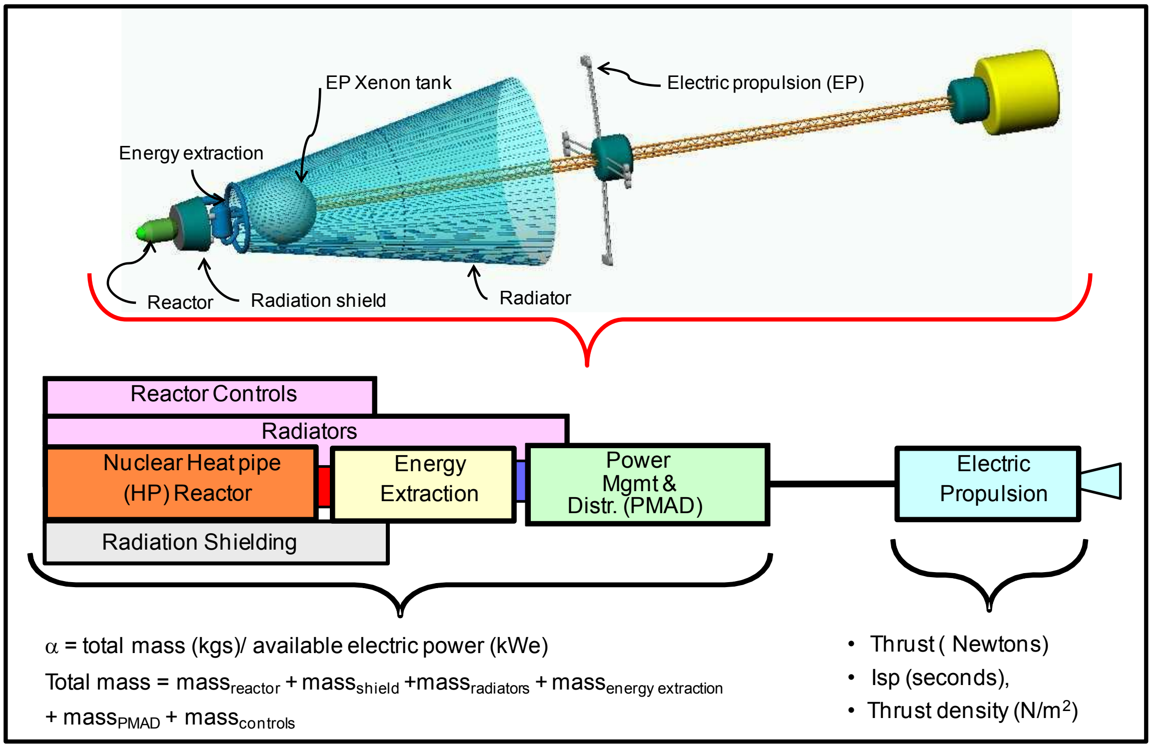


Figure 5-2. Block Subsystem Mode of a Typical Space Nuclear Power Electric Design

Source: Artwork courtesy of LANL.

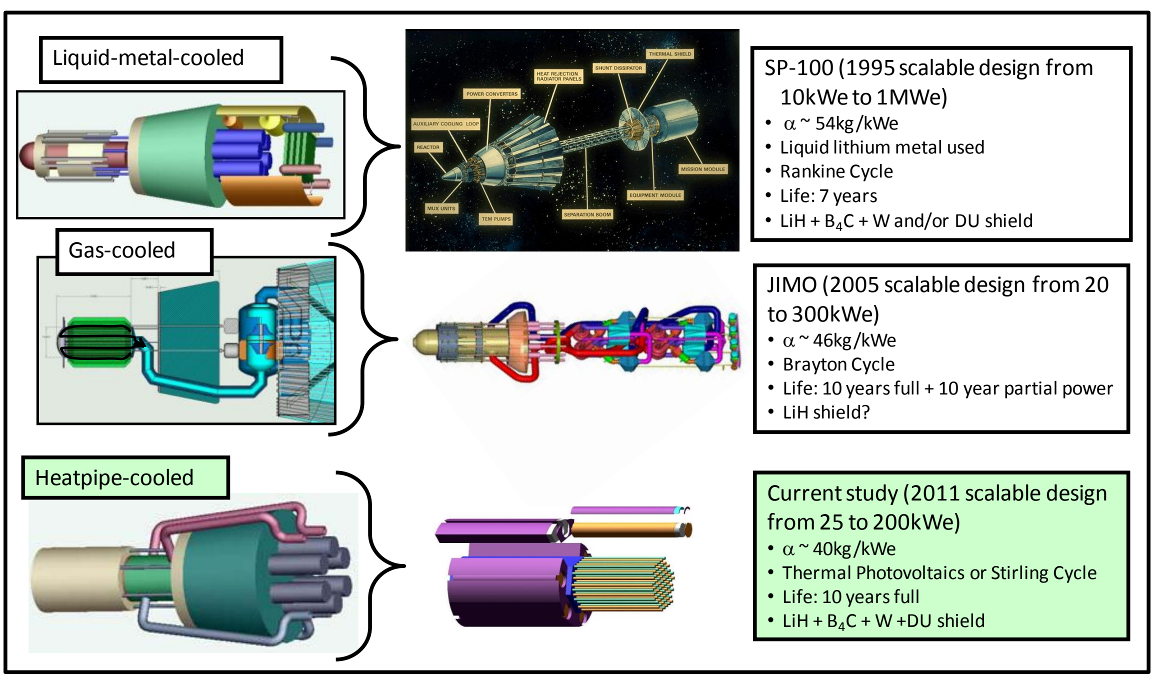


Figure 5-3. Space Nuclear Reactor Types

Source: Artwork courtesy of LANL.

The typical monoblock reactor-like geometry consists of an assembly of stationary pipes held together and containing nuclear material (fuel pins), neutron absorbing materials (B_4C) movable rods, empty tubes/spaces allowing liquid metal or gas to flow through or heat pipes for cooling the system (see Figure 5-4 and Figure 5-5).

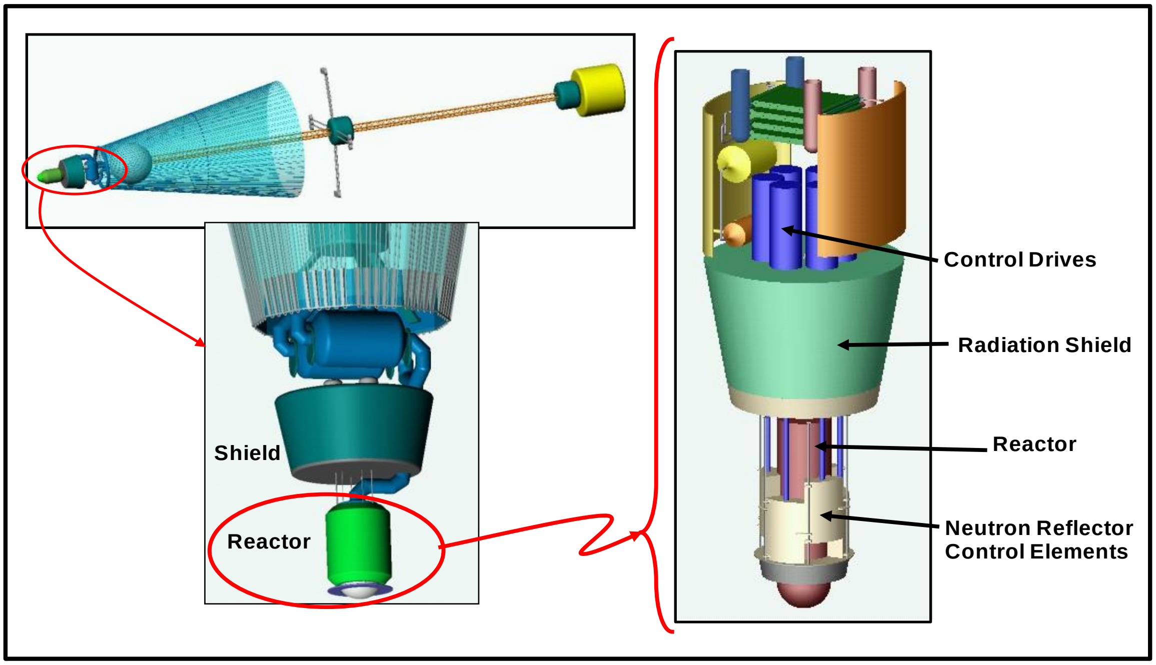


Figure 5-4. Space Nuclear Reactor Details

Source: Artwork courtesy of LANL.

Many references (Angelo and Buden 1985; Bennett et al. 1996, Kulcinski 2004; Bennett 2006) offer great insight into engineering considerations for designing a space nuclear reactor. The changing geometry of the B_4C and Be structures relative to the nuclear-fuel pins controls the reactor by controlling the neutron flux. The heat-pipe reactor system was chosen for further study because it had fewer moving parts than the liquid metal- and gas-cooled systems. Heat-pipe-cooled reactor systems also have inherently graceful degradation, which might be a desirable feature for a military system. If the heat pipes are sized properly and one fails, the surrounding heat pipes can automatically pick up the additional thermal load.

Figure 5-6 and Figure 5-7 are sketches of various passive and active energy conversion approaches considered for nuclear space power. Only the thermophotovoltaic (TPV) and Stirling cycles were considered for this study since they have a minimum of moving parts, which keeps this proposed NEP design less complex and potentially less costly. The Stirling cycle is envisioned to consist of multiple independent power heads providing an additional level of redundancy.

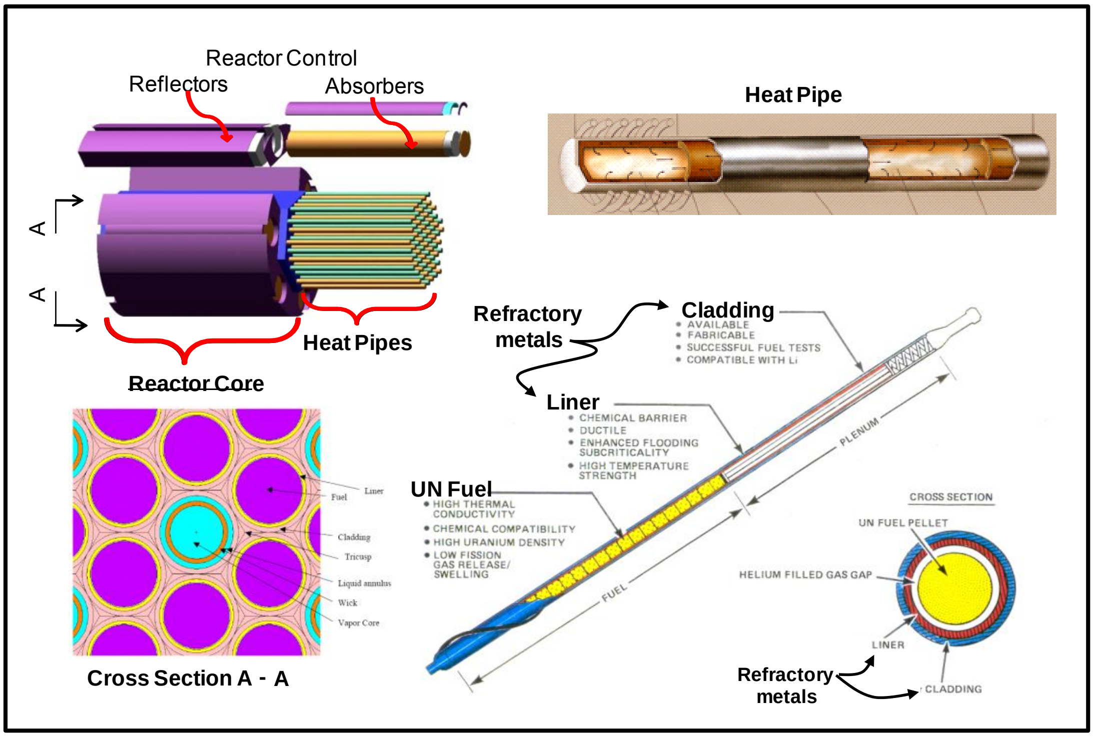


Figure 5-5. Details of a Heat-Pipe Reactor

Source: Greenspan 2008. Artwork courtesy of LANL and the University of California.

Note for Figure 5-5: Surrounding the construct this figure are several moveable Be-like structures to reflect neutrons back into the core.

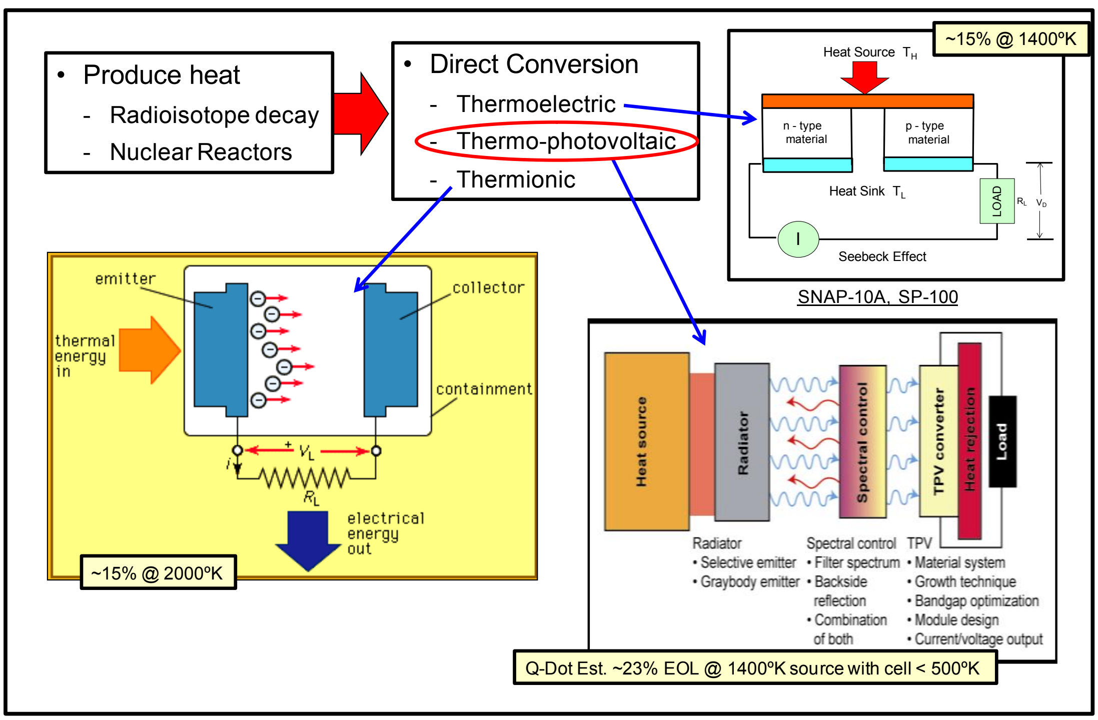


Figure 5-6. Direct Energy Extraction Approaches

Source: Angelo and Buden 1985; Teofilo et al. 2008.

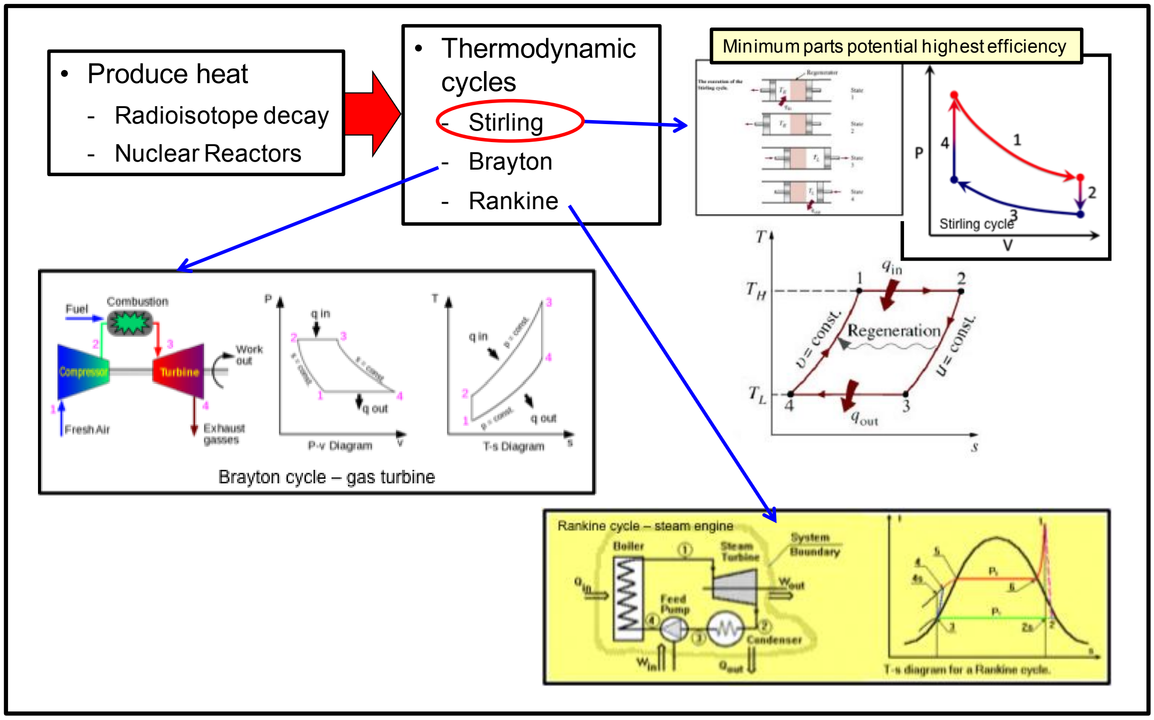


Figure 5-7. Indirect Energy Extraction Approaches

Source: Angelo and Buden 1985; Teofilo et al. 2008.

A. Electric Propulsion

Figure 5-8 provides a snapshot of performance data for several candidate EP approaches that were considered for the NEP. Cassady et al. (2008) and Auweter-Kurtz et al. (2005) provide exhaustive details on the performance, capabilities, and technology readiness of different EP technologies. Strong government and commercial investment and flight experience in space EP continues to mature this technology.

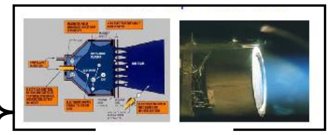
Figure 5-8 also identifies the Isp (specific impulse, seconds) and the maximum thrust (Newtons) for different EP approaches developed in the past given an electric power demand (kWe). Isp, an extremely important parameter that can be thought of as gas mileage, is defined as the thrust-to-weight flow rate. It is a measure of the energy content of the propellant and how efficiently it is converted into thrust (Wertz and Larson 1999).

The challenge for the optimum survivability application is to have an Isp as high as possible with the greatest thrust. For example, a chemical rocket may have millions of Newtons of thrust but have an Isp from 150 to 380 sec; therefore, it requires a large fuel tank for extended operation. On the other hand, an ion thruster may have only 0.460 N of thrust but an Isp of about 6,000 sec; therefore, it needs a much smaller amount of mass per operating unit of time.

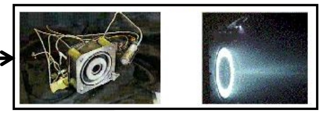
Isp = Thrust/weight flow rate (seconds)

Type	Isp (sec.)	Thrust (max)(N)	Power (kWe)
NASA GRC HIPEP (low)	6000	0.460	25
NASA GRC HIPEP (high)	9620	0.670	39.3
NASA NEXIS	8700	0.517	27
VHITAL 160 (Mode 1, two stage tech.) [1]	6000	0.650	25
Electro dynamic tether	40,000	4.5	25

(1) VHITAL Mode 2: Isp, 8000, Thrust 0.710N at 36kWe

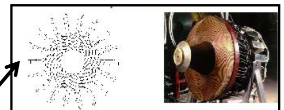


Ion thruster

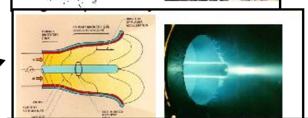


Hall thruster w/induction

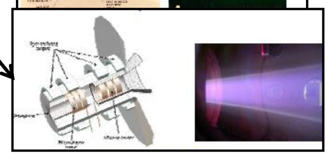
Type	Isp (sec.)	Thrust (max) (N)	Power (kWe)
Pulsed Inductive thruster (NuPIT)	3000 - 10000	5	200
Li-MPDT ALFA	4250	4	192
VASIMR (VX-200)	5000 - 10000	5.7	200



Pulsed inductive thruster



Magneto plasma dynamic thruster



VASIMR thruster

Figure 5-8. EP Concepts

Source: Auweter-Kurtz et al. 2005; Cassady et al. 2008. Photo courtesy of NASA.

Unfortunately, to maneuver sufficiently to negate accurate tracking of foreign SOSI systems within a reasonable amount of time, 0.460 N may not provide enough thrust. The goal then is to provide a space propulsion system with very high Isp (~7,000 sec) at a workable thrust (~5 N) to negate foreign SOSI tracking capabilities with a reasonable amount of consumables. A NEP system can achieve this capability at the right power levels.

Figure 5-8 shows a few types of EP engines with ~5 N of thrust and an Isp range from 3,000 to 10,000 sec. If one knows the thrust, power input, and Isp, the overall thruster efficiency can be estimated using Eq. 2.5-8 from Goebel and Katz (2008).

Eq. 2.5-8 was rewritten to solve for the total thrust efficiency of an EP engine, η_T, as:

$$\eta_T \approx ((g_o * Isp * Thrust)/(2 * Power)) * 100\ [\%]. \qquad (5\text{-}1)$$

According to Eq. (5-1), 5 N at an Isp of 3,000 sec would need a thruster efficiency of 36.75%. Likewise, at 10,000 sec, η_T is about 122%, which does not appear possible and demonstrates the limitations of Eq. (5-1). An Isp of 7,000 sec gives an efficiency of about 85%, which seems high but reasonable and therefore the goal of this program.

From Eq. 2.4-2 in Goebel and Katz (2008), another first-order estimate can be made for the amount of consumables used per second (Q) necessary for EP engine(s) using

Xenon producing 5 N with an Isp of ~7,000 sec. To calculate the Xenon mass flow, start with solving for Isp:

$$\text{Isp} = (1.037 \times 10^6)^*(\text{Thrust [N]})/(\text{Q [sccm]}^6)\ \text{[Seconds]}, \tag{5-2}$$

where Q = (mass flow rate of Xenon)* g_o.

Eq. 2.4-2 can then be rewritten, solving for the approximate Xenon flow rate, Q, necessary to create this thrust level. For example, given a thrust of 5 N and an Isp of 7,000 sec, the mass flow per second would be

$$\text{Q [mg/sec]} = ((1.02 \text{ x } 10^5)^* \text{ (Thrust [N]))/(Isp [sec])} \approx 73 \text{ mg/sec.} \tag{5-3}$$

Therefore, from Eq. (5-3), the mass used per day of constant EP thrust would be ~6.3 kg or 394 g per 90 min LEO orbit or 1.05 kg per 4 hr MEO orbit. So, for example, if the mission was planned for 5 years and the amount of Xenon was restricted to ~1,150 kg, the spacecraft could thrust for a constant 24 hrs for ~183 days (10% duty cycle) resulting in the orbit changes noted in Table 2-1.

B. ED Tether Propulsion

A unique form of EP is the ED tether (Hoyt 2008). Propulsion is produced from the ED tether by generating current flow through a long (up to 50 km) gravity-gradient conducting wire. This wire interacts with the earth's magnetic field to produce a Lorentz force aligned along the orbit track of the satellite to which an ED tether is attached (see Figure 5-9).

Depending on the direction of the current, the satellite can change orbit altitude and inclination. ED tethers have an estimated Isp of over 40,000+ sec and have the potential to produce as much as 5 N of thrust for a 25 kWe powered system. The key challenge for ED tethers is to produce sufficient current in the wire by efficiently coupling the current in the wire through the ionosphere.

To produce 5 N of thrust requires an electric current in the tether of at least 4 to 5 ampere (A). The state-of-the-art today is about 1 A current flow by using a Hollow Cathode Plasma Contactor (HCPC). The HCPC creates a high-density local plasma using Xenon gas that diffuses out into the ionosphere, thereby creating a low impedance path for the electrons in the tether to be emitted or collected (Hoyt 2008). This approach

[6] The term "sccm" is defined as the standard cubic centimeter per min to convert sccm to mg/sec for Xenon. One starts with Eq. B-4 in Goebel and Katz (2008), where 1.0 [sccm $_{\text{Xenon}}$]) = 7.435 × 10^{-4} × Ma [mg/sec]/0.9931468. (Ma is the atomic number for Xenon, which is 131.293.) The value of 0.9931468 is a correction for compressibility at standard temperature and pressure. Therefore, 1.0 [sccm $_{\text{Xenon}}$] = 0.0983009 [mg/sec].

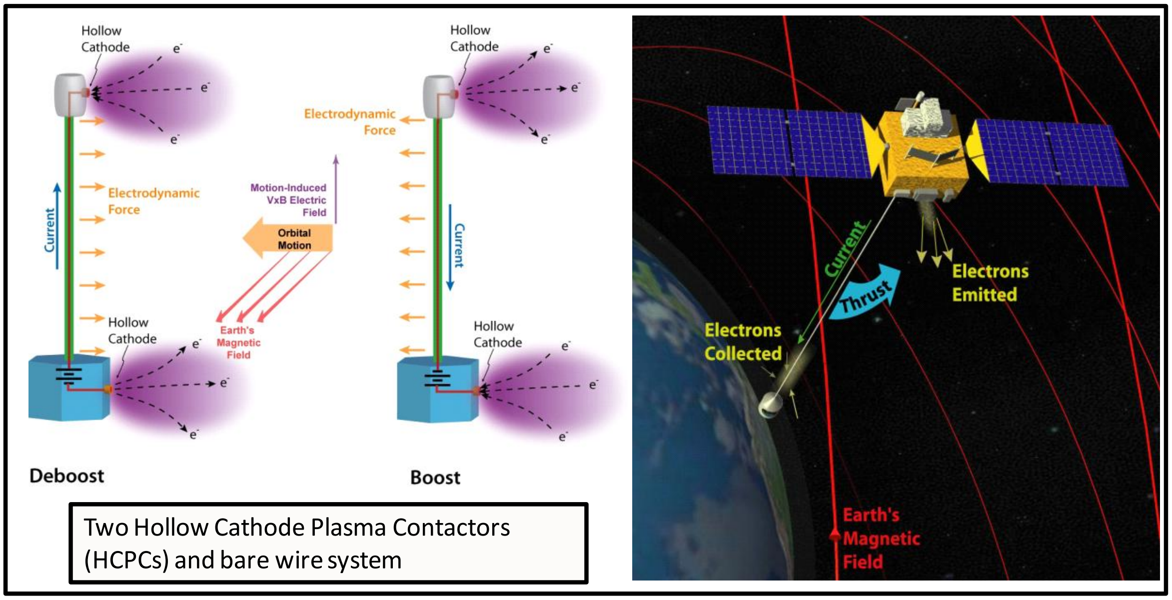


Figure 5-9. ED Tether Propulsion

Source: Hoyt 2008.

restricts ED tethers to orbit LEO altitudes from 250 to 2,000 km. The lower altitude limit is driven by the upper-atmosphere drag on the tether that would deorbit it, and the upper limit is due to the ionosphere density. ED tethers can work at any inclination but have more optimum thrust (Lorentz force) for the satellite when the tether's orbit track is perpendicular to the earth's magnetic field lines. This orientation favors tether operations at lower inclination orbits. Nevertheless, given these operational restrictions, no other EP system today can match the performance of ED tethers. With a 200 kWe nuclear reactor, the satellite could drive as many as eight 25 kWe ED tether systems. The challenge for such a design would be to keep the deployed tethers stable and away from each other.

With HCPC-type devices positioned at least 16 m apart, a variety of ED tethers configurations can be explored to maximize tether current and, therefore, thrust. The long length of multiple tethers (~50 km) may create a problem of entanglement; therefore, a single tether with multiple distributed HCPCs may be more feasible.

6. Rapid Prototyping Nuclear Electric Propulsion

The idea of a non-nuclear vacuum test environment to evaluate a space NEP system is not new and was pioneered by NASA/JPL and Marshall Space Flight Center (MSFC) with the Safe Affordable Fission Engine (SAFE-30) program in 2003 (see Figure 6-1). A 30 kWt sodium-filled stainless-steel heat-pipe reactor heated by electric heaters to 1023 K was coupled directly to the heater head of a commercially available free-piston Stirling cycle engine operating at 923 K. It produced 350 We @ 175 volts direct current (VDC) that was stepped up to 1,000 VDC (using a direct current (DC) to DC converter) for powering a 17 cm NASA Solar Technology Application Readiness (NSTAR)-like-designed ion thruster. This experiment was the first end-to-end NEP system evaluation using a non-nuclear testing approach (Hrbud et al. 2003). The SAFE-30 core was to be the building block for the next phase of the program, SAFE-400, which was a 400 kWt device producing 100 kWe of power. Poston et al. (2002) addresses the design and analysis of the SAFE-400 testbed, which was never built. This pioneering approach was key to demonstrating a low-cost and rapid method of designing, fabricating, testing, and evaluating a complete heat-pipe-based NEP system without the need for an actual nuclear reactor.

Recently, Glenn Research Center (GRC) and a coalition consisting of MSFC and four DOE labs (Los Alamos National Laboratory (LANL), Oak Ridge National Laboratory (ORNL), Sandia National Laboratory (SNL), and Idaho National laboratory (INL)) have undertaken a "non-nuclear" fission power testbed (see Figure 6-2). The Technology Demonstration Unit (TDU), located in the GRC Vacuum Facility #6, is a hardware program to develop a simulated nuclear reactor for system-level T&E (Mason et al. 2011).

The nuclear reactor simulator is based on a liquid metal cooling design using NaK (a sodium-potassium compound) to provide heat to a pair of free-piston Stirling engines with the goal of producing 10 kWe to a simulated electrical load. The reactor core simulator is heated using 48 kWe of electrical heaters, bringing the NaK liquid metal to 850 K at the heat exchanger of the Stirling engines. The engines are rated to produce 12 kWe of power and provide 375 K heated water to a series of space radiators inside the vacuum chamber. These radiators reject about 36 kWt of waste heat. The vacuum chamber has liquid-nitrogen-cooled walls for space thermal environment simulation. When completed, this testbed will primarily evaluate liquid metal heat transport, electric power generation, and waste-heat removal and compare results to analytical models (Mason et al. 2011).

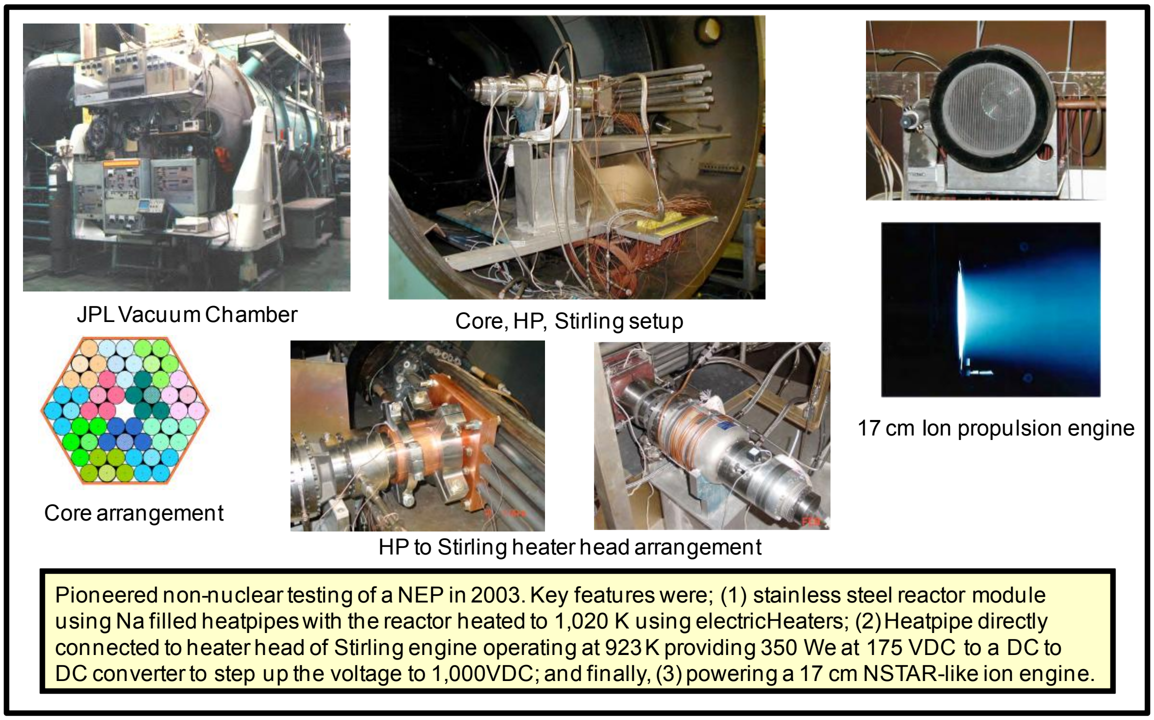


Figure 6-1. NASA SAFE-30 Test Program

Source: Hrbud et al. 2003. Photos courtesy of NASA.

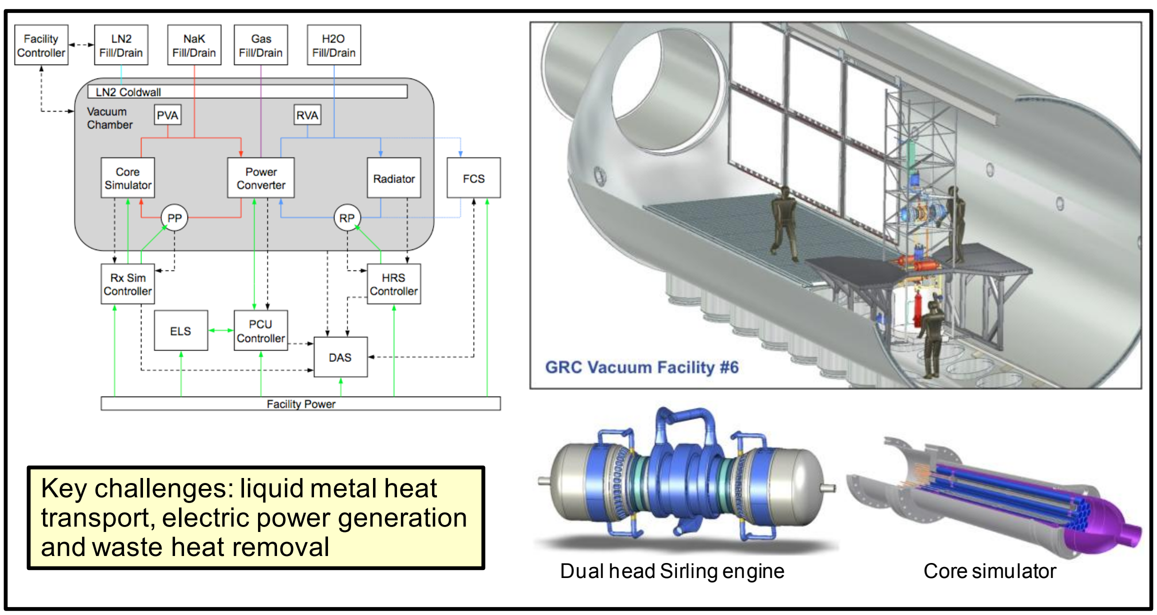


Figure 6-2. NASA/DOE Fission Power Systems Technology Demonstration Unit

Source: Mason et al. 2011. Artwork courtesy of NASA.

Two NEP system development approaches that are important to this work (as shown in Table 6-1) are based on lessons learned from the SP-100 and Prometheus programs' experiences. The genesis of these approaches builds upon exploratory work done in 2004 as a seedling and presented to DARPA management. This project was not approved for further funding. Nevertheless, the author believes that the two approaches presented next have merit for further consideration by the DoD and/or IC to provide space power and to ensure space dominance for the foreseeable future.

Table 6-1. Two Ways To Proceed

	Approach A: Low Risk (Low to High Power)	Approach B: High Risk (Warm to Hot Temperature)	Comments
α (kg/kWe)	Constant: 40	Decreasing: 40, 35, 30	
Mass (kg)	Increasing: 1,000 to 8,000	Decreasing: 8,000 to 6,000	
Temperature (K)	Slight increase: 1,000 to 1,150	Large increase: 1,150 to 1,400	
Energy Extraction	TPV or Stirling choice for cycles #1 to #3	TPV or Stirling for cycle #1; Stirling for cycles #2 and #3	
Power (kWe)	Increasing: 25, 100, 200	Constant: 200	
Thrust	Increasing: 500 mN to 5 N	Constant: 5 N	
Cost	First cycle: $440 million	First cycle: $800 million	Total for either ~1.2 billion to 1.3 billion
Cycle Duration	36/36/36 months	48/36/36 months	
Benefit	High Technology Readiness Level (TRL) cycle #1 for earliest low-cost flight demonstration	Most thrust are cycle #1; mass-optimized after cycle #3	

Approach A, which focused on minimizing initial cost and technical risk, begins with a 25 kWe reactor that grows to 200 kWe at a nominal operating temperature. Approach B, which focused on minimizing NEP system mass, begins with a 200 kWe reactor and increases its operating temperature by 250 K to improve system efficiency and reduce mass.

The two approaches have a few general features in common. They each contain three rapid prototyping cycles. At the end of each approach are slightly different NEP products but with more or less similar performance features. A well-defined scope in each cycle ends with a full-up, ground engineering design unit (EDU) NEP for full-scale testing in a simulated space thermal vacuum environment. The EDU provides a high-fidelity testbed (see Figure 6-3) for the end-to-end demonstration of a space fission reactor including the following:

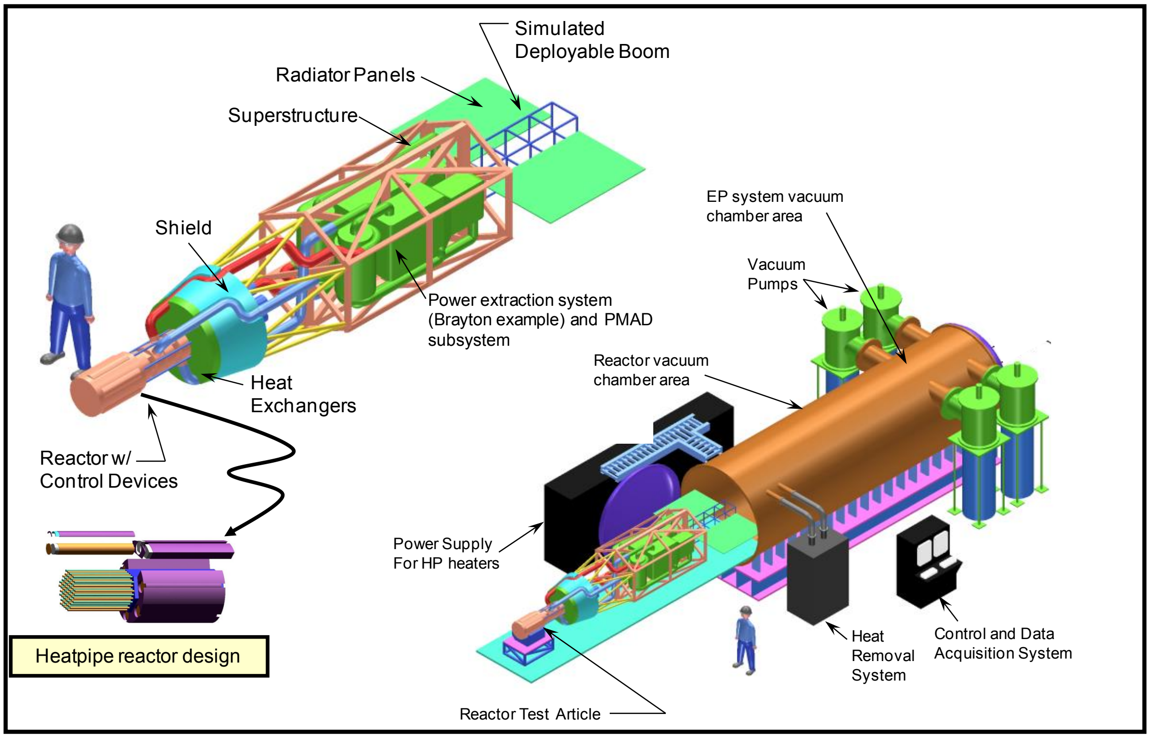


Figure 6-3. NEP Engineering Design Unit (EDU) Ground Test Concept

Source: Mason et al. 2011. Artwork courtesy of LANL.

- Simple yet elegant design for continued mass optimization (maximize α)
  - Electrically heated (non-nuclear) monoblock heat-pipe reactor (minimal moving parts), with the design assumption of using UN fuel with a prototype shield
  - Passive and active energy conversion subsystems with direct energy extraction from reactor cooling heat pipes to minimize system complexity
  - NEP PMAD subsystem capable of high-voltage and high-current switching for different EP engines
- High-fidelity EDU
  - High-voltage power management and distribution system for the electric heaters
  - Simulated prototype waste-heat thermal radiator
  - EP units with a minimum Isp of 7,000 sec
  - EDU NEP built to specific launch vehicle requirements
  - Liquid nitrogen (LN_2)-cooled surfaces for space environment simulation

  - High-performance vacuum chamber to maintain sufficient vacuum conditions during short-duration EP operation or separate vacuum chamber for EP operations
- Fast track program management
  - Small/lean engineering and program management team managed by a single agency
  - Stringent scope control within each rapid prototyping cycle
  - Single prime contractor integrator with minimum government-furnished equipment (GFE) contributions
  - Energy-conversion-device enhancement developed in parallel
  - Buy whenever possible; design and build only what is necessary
  - Continued value-added assessments using a companion virtual testbed for qualification of the entire NEP system with virtual candidate payloads and competing solar space power systems
  - Extensive and continuous stakeholder analysis
  - Facility for "zero-power" testing to develop confidence in reactor controls simulator

A. Approach A: Constant α and Isp; Increase Power, Mass, and Thrust

Rapid prototyping "Approach A" strives to build the initial lowest cost and risk, end-to-end launch, and space-qualified prototype NEP EDU (using the highest TRL subsystems and components) as soon as possible and then continue to grow its capability in performance and size.

1. First Rapid Prototyping Cycle

Figure 6-4 is a block diagram of the design by the end of the first 36 month cycle. It consists of a 25 kWe Li-filled heat-pipe reactor electrically heated and operating at a maximum of 1000 K, with a full complement of control devices and software.

This EDU would also have a

- Radiation shield mass thermal management optimized by using stainless-steel-encapsulated LiH, W, depleted uranium (DU) and B_4C;
- TPV passive direct energy conversion;
- Simulated waste heat radiators; and
- PMAD subsystem driving the EP devices.

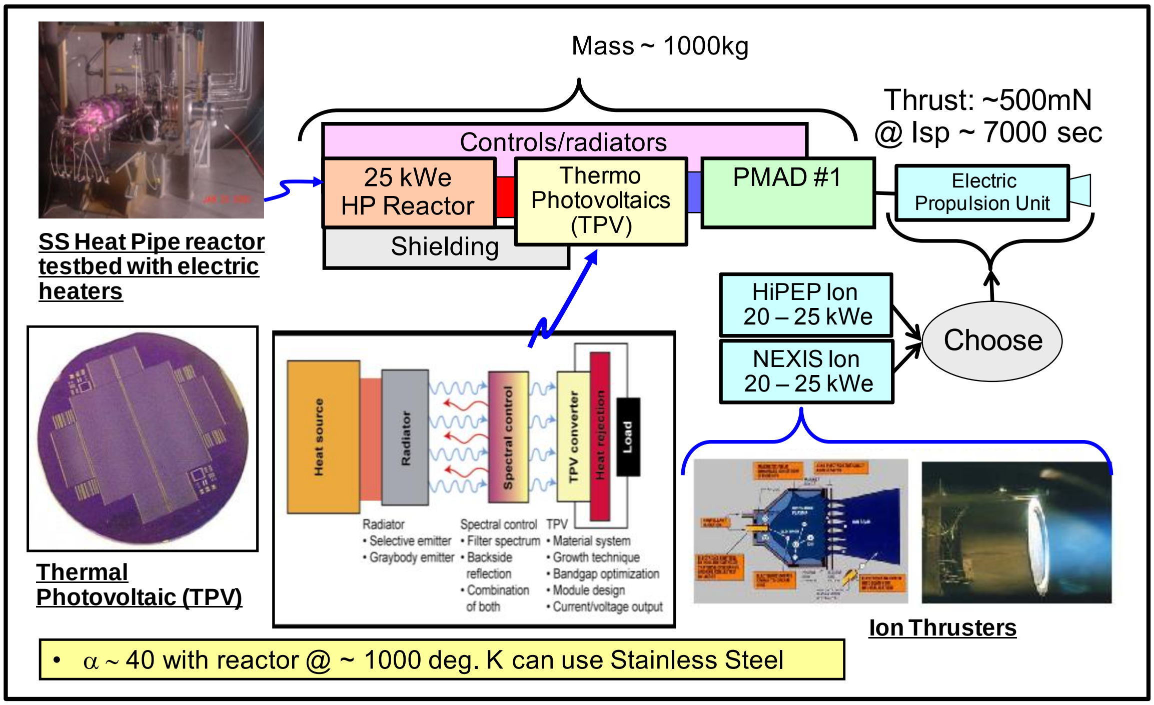


Figure 6-4. Approach A: NEP Rapid Prototyping Cycle #1

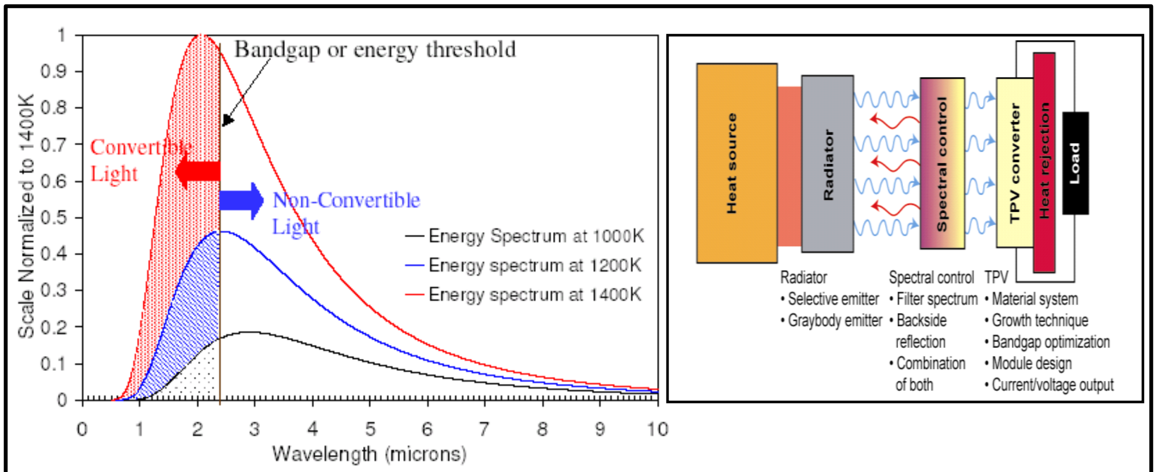
Source: Cassady et al. 2008; Teofilo et al. 2008. Artwork courtesy of NASA.

Two candidate EP systems developed under the NASA JIMO program were the High Power Electric Propulsion (HiPEP) and the Nuclear Electric Xenon Ion System (NEXIS) thrusters. These two systems were advertised to provide an Isp of greater than 7,000 sec and a thrust between 0.517 to 0.670 N (Auweter-Kurtz et al. 2005; Cassady et al. 2008). EP devices similar to these thrusters would be down-selected and integrated into the EDU. The selected EP device would provide an Isp of 7,000 sec and thrust of approximately 500 mN. All components would be designed and built to survive a particular launch environment to the degree possible. If necessary, the EP engine could be operated in a separate vacuum chamber in the vicinity of the EDU to reduce test complexity.

Although the operating temperature of this first NEP configuration is ~1000 K, the Li heat pipe was chosen instead of the NaK design to begin the process early in the program to address the fabrication and contamination control technologies needed for the eventual higher temperature operation. The Li-based heat-pipe designs are extremely sensitive to nitrogen and oxygen contamination, which could affect the solubility of different refractory metals (i.e., niobium (Nb), tantalum (Ta), molybdenum (Mo)) in Li causing early pipe failure from corrosion (Vasil'kovskii et al. 2000).

Working with Li heat pipes at the start of the program will develop the non-contaminating fabrication process technologies necessary to build long-life, high-temperature heat pipes. Vasil'kovskii et al. (2000) provide a good general overview of the complexity in developing a welding fabrication-based process that minimizes contaminants.

TPV technology (see Figure 6-5) was chosen for this first rapid prototype cycle since it is a passive energy conversion approach that involves no moving parts. TPV is a class of solid-state devices in the form of p-n diode that can convert radiant thermal (infrared (IR)) photons directly into electricity.



- The key features are no moving parts, inherent reliability and simplicity.
- Limitations include: Conversion efficiency at high operating temperature and large scale manufacturing
- Quantum dot concepts may resolve many of the issues at high temperatures

Figure 6-5. TPV Power Extraction

Source: Teofilo et al. 2008.

Low-bandgap TPV cells are usually made with binary, ternary, or quaternary semi-conductors such as indium gallium arsenide (InGaAs), gallium antimonide (GaSb), indium arsenide (InAs), or indium gallium arsenide antimonide (InGaAsSb) (Teofilo et al. 2008). To make the TPV devices efficient, radiation from the hot heat pipes is filtered to pass only the optimum wavelengths suitable for absorption by the TPV cell. The filter also reflects unwanted wavelengths back to the source, thereby reducing heating of the TPV cell. The filter performance sets the requirement for the optimum diode bandgap. A high spectral efficiency requires filters with very high reflectivity and low absorptivity for photons below the bandgap and high transmissivities and low reflectivity for photons above it (Teofilo et al. 2008).

Currently, the operating temperature of the TPV cells must be less than 400 K, which drives the thermal management system radiator performance down because of the low waste-heat rejection temperature. Cells produced by Emcore Corporation operate at about 25% efficiency at 300 K and with a 1350 K emitter source but will decrease to less than 10% efficiency when operating at 415 K. The reason for this efficiency drop is that as the temperature increases, so does the TPV cell dark current, thereby reducing its efficiency. To address this issue, investments continue to be made (e.g., improvements in the metal oxide chemical vapor deposition (MOCVD) manufacturing processing, the use of dual junction cells to convert below-bandgap photons, and growth of nano-structured super lattice material in the junction of these cells). The goal is 40% efficiency at 500 K via reducing the dark current by up to 60% (Teofilo et al. 2008).

The overall system design goal would be a mass of less than 1,000 kg providing an α less than 40. This cycle would be 36 months long, including 4 months for the T&E assessment of the complete NEP system performance. Supporting these activities would be a comparative analysis to solar power systems and system performance level studies to identify high-value missions that could use this NEP design. Also cost and schedule estimates would be available for investors interested in actually building a flight article.

2. Second Rapid Prototyping Cycle

Figure 6-6 shows the second rapid prototyping cycle. Given the successes of the first cycle, the design would evolve to a 100 kWe heat-pipe reactor operating at 1150 K, which requires refractory metals.

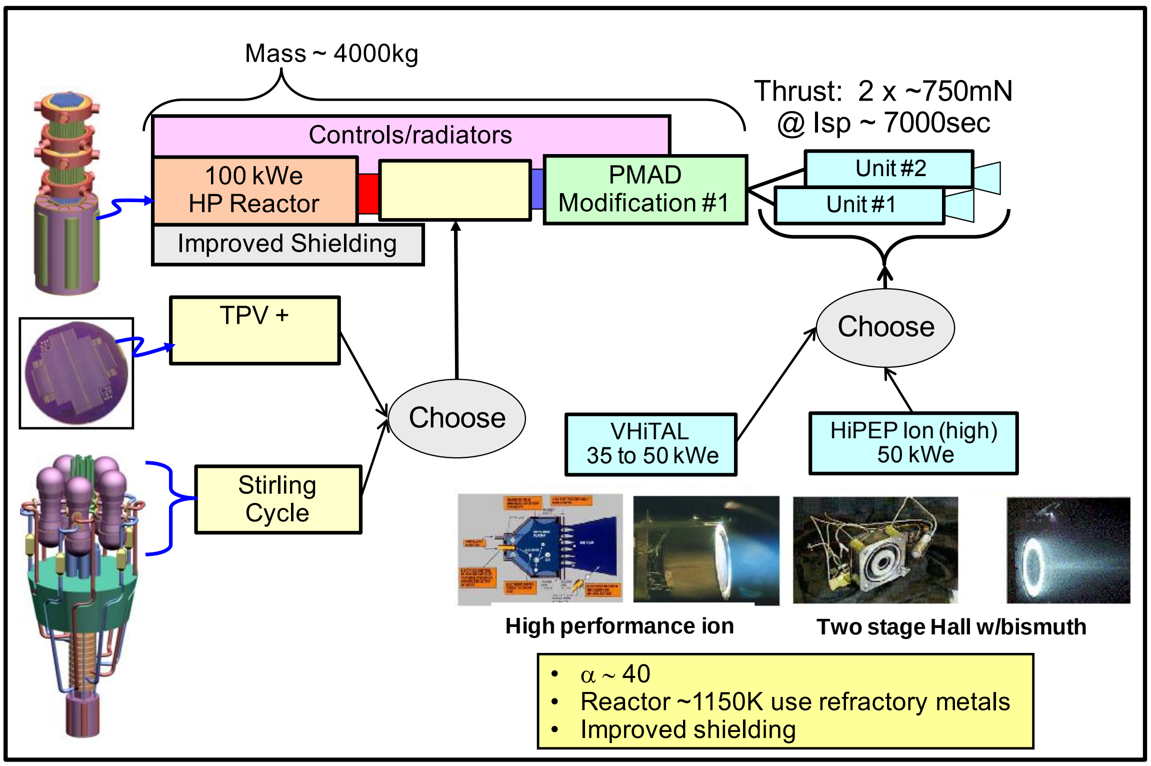


Figure 6-6. Approach A: NEP Rapid Prototyping Cycle #2

Source: Cassady et al. 2008; Teofilo et al. 2008. Artwork courtesy of LANL. Photos courtesy of NASA.

Refractory metals are a class of metals (Nb, Mo, Ta, W, Re, Ti, vanadium (V), chromium (Cr), Zr, hafnium (Hf), ruthenium (Ru), osmium (Os), and Ir) that have high melting temperatures (greater than 2128 K) and low wear. Appendix B provides more details on this class of metals, their performance at elevated temperatures, and fabrication issues. Subsystems would include a shield optimized as in the first rapid prototyping cycle and a down-select between an upgraded TPV or Stirling cycle engine active energy conversion subsystem with direct contact with the reactor heat pipes (similar to the NASA SAFE-30 configuration).

The 100 kWe reactor would provide power to a variety of 30 to 50 kWe EP systems similar to the very high Isp thruster with anode layer (VHITAL) and HiPEP ion-type thrusters. Selected earlier (after the first year) in Phase 1, these type of thrusters would have had as much as 48 months to mature and be available for the EDU testing.

After 72 months, this program would have demonstrated two classes of NEP flight prototypes to stakeholders and potential investors, identified the potential missions and technology shortcomings, and refined the cost and schedule planning data for flight articles.

3. Third Rapid Prototyping Cycle

Figure 6-7 shows the final prototyping cycle for Approach A. Given the successes of the first two cycles, the design evolves to a 200 kWe Li heat-pipe reactor operating at 1150 K. In this cycle, further improvements in TPV and the Stirling energy conversion developments will be assessed, and one system will be down-selected based on the least mass and highest efficiency. Depending on the parallel investments made in these energy-extraction technologies, one or both of the conversion approaches could be viable. Similar to the previous cycles, the waste-heat thermal management system would be simulated, the PMAD would be improved, and, if possible, a lower mass and thermal management optimized shield using advanced materials would be developed (see Appendix C).

Based on their maturity today and investments during the 72 month period of the first two cycles, three 200 kWe EP-like devices could be ready for down-select. They would be similar to the NASA Nuclear-Electric Pulsed Inductive Thruster (NuPIT), Lithium-fed Magnetoplasmadynamic Thruster (Li-MPDT), and the Variable Specific Impulse Magnetoplasma Rocket (VASIMR) engines currently in development (Auweter-Kurtz et al. 2005; Cassady et al. 2008). The requirement again would be to demonstrate an Isp of greater than 7,000 sec with a thrust of ~5 N. The total mass of the reactor and supporting subsystems would be allow to grow to 8,000 kg while maintaining an α of less than 40.

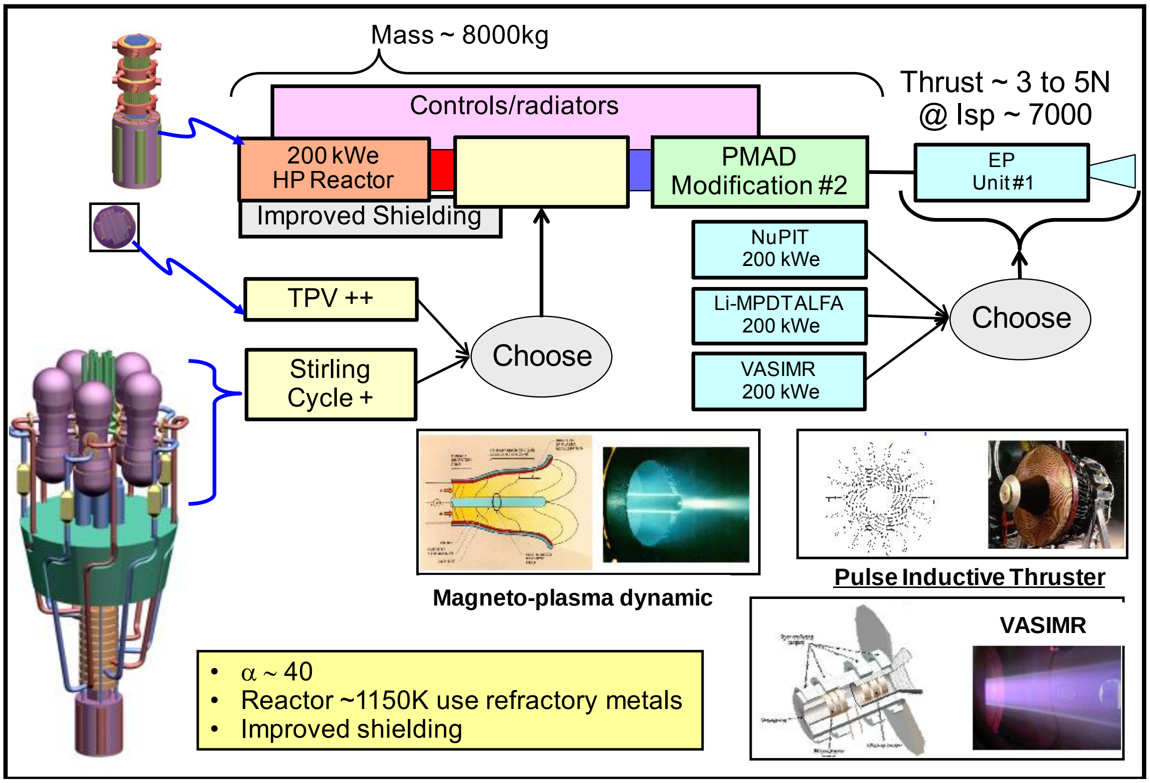


Figure 6-7. Approach A: NEP Rapid Prototyping Cycle #3

Source: Cassady et al. 2008; Teofilo et al. 2008. Artwork courtesy of LANL. Photos courtesy of NASA.

At the end of the third cycle of Approach A (9 years into the program), stakeholders and investors would have seen three end-to-end NEP systems that could have been be flown. This rapid prototyping feature is what sets this idea apart from previous efforts, such as the SP-100 and Prometheus/JIMO programs. Complementing these demonstrations would be an analysis of identified, innovative, high-value missions and improved cost and schedule data for the development of a flight article. Knowledge gained from these hardware demonstrations and simulations would have improved industry design environments critical for the development of NEP-based space flight systems.

B. Approach B: Constant Power and Thrust; Increase Reactor Temperature

This rapid prototyping approach strives to build the lowest mass, end-to-end, 200 kWe NEP EDU by continually raising the reactor operating temperature, thereby increasing its waste-heat rejection and the energy conversion subsystem efficiencies.

1. First Rapid Prototyping Cycle

Figure 6-8 is a block diagram of the design through the end of the first rapid prototyping cycle. It consists of an electrically heated 200 kWe refractory metal Li heat-pipe reactor operating at a maximum of 1150 K, with a full complement of control devices and software.

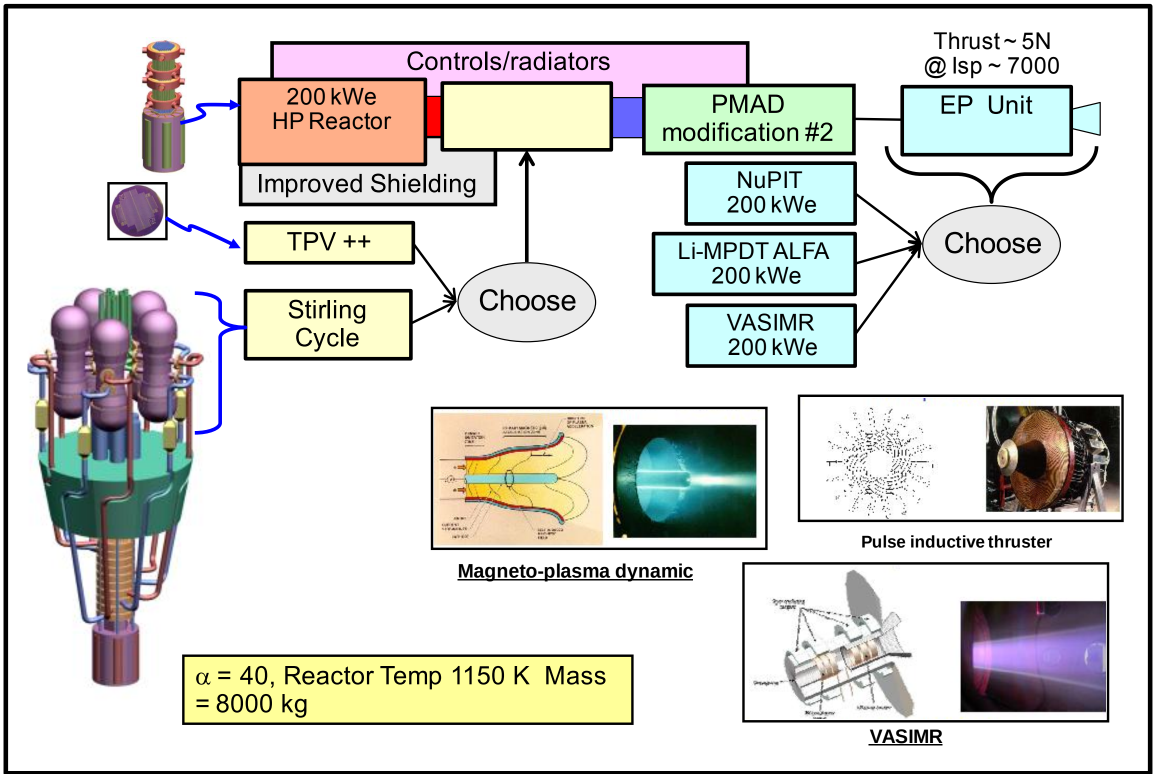


Figure 6-8. Approach B: NEP Rapid Prototyping Cycle #1

Source: Cassady et al. 2008. Artwork courtesy of LANL. Photos courtesy of NASA.

This EDU would also have (as mentioned previously in Approach A)

- An optimized mass and thermal performance shield using LiH, W, DU, and B_4C;
- Down-select between a TPV passive and Stirling Cycle active direct energy conversion;
- Simulated waste-heat radiators; and
- A reliable PMAD designed for high-voltage and high-current switching, driving the high-performance EP systems

The three 200 kWe EP-like engines (NuPIT, Li-MPDT, and VASIMR), based on their maturity today and investments during the development of the reactor system, would be candidate designs for down-select. The requirement, again, would be to demonstrate an Isp of 7,000 sec with a thrust in the range of ~5 N.

All components would be designed and built to meet a specific launch environment. The design goal would be a mass of less than 8,000 kg, providing an α less than 40. This first cycle would be 48 months long, including 4 months for the T&E assessment of the complete NEP system. The additional year for this approach is due to materials availability for and fabrication complexity of the first refractory metal reactor. These issues are discussed in Appendix B.

As noted previously, studies to identify high-value missions that could use this NEP design would be undertaken. These studies would include an analysis to provide flight article cost and schedule estimates. The key point of this approach is that at the end of the first rapid prototype cycle, a 200 kWe NEP prototype system would be available as a baseline design for a flight article.

2. Second Rapid Prototyping Cycle

Figure 6-9 shows the second rapid prototyping cycle. Assuming success during the first cycle, this next cycle undertakes the modification of the 200 kWe heat-pipe reactor to operate at 1250 K and the reduction of the overall system mass by 1,000 kg to achieve an α of less than 35. To raise the reactor operating temperature 100 K requires advancing Li heat pipes and Stirling engine technologies and, potentially, fabricating a new reactor core.

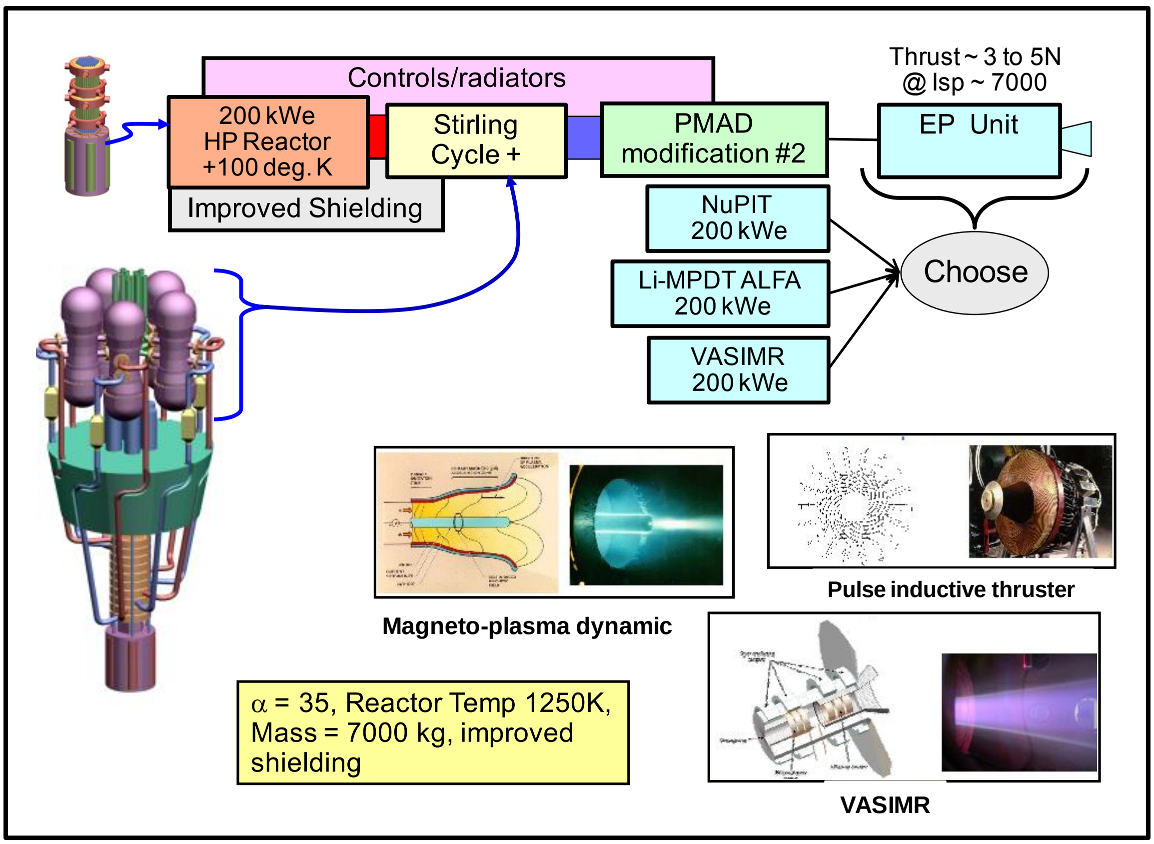


Figure 6-9. Approach B: NEP Rapid Prototyping Cycle #2

Source: Artwork courtesy of LANL. Photos courtesy of NASA.

Advanced Li heat pipes and Stirling cycle engine systems enhancements to operate at this elevated temperature would have occurred during the first cycle to be ready for integration into the higher operating temperature reactor. As in the previous approach, this EDU would have an advanced shield designed to overcome the limitations of LiH material (see Appendix C). TPV passive direct energy conversion would be dropped, and the higher temperature operating Stirling cycle engines would be used. Also, as noted previously, the EDU would contain a simulated waste-heat thermal management system, controls, and a high-voltage, high-current switching PMAD system. Power would be provided to one of the EP system designs considered in the first cycle. This second cycle would be 36 months long, including 4 months for the T&E of this NEP system. Mission studies and flight article cost and schedule estimates would be refined as necessary.

3. Third Rapid Prototyping Cycle

Figure 6-10 shows the third and last prototyping cycle for Approach B. Assuming successes during the first two cycles, the operating temperature of the system is now raised to 1400 K. Similar to the previous cycles, the waste-heat thermal management system would be simulated. If necessary, a new PMAD would be developed to handle the EP loads along with improved alternative material shield (see Appendix C). A new reactor core may have to be fabricated to operate at this temperature.

The best candidate 200 kWe-like EP systems mentioned previously would be demonstrated to provide an Isp of 7,000 sec and a thrust of ~5 N. Continued advancement in Li heat pipes and Stirling cycle engine performance—including an ability to operate at the elevated temperatures—would have occurred during the second cycle for integration into the 1400 K reactor. The total mass goal of the reactor and supporting subsystems would be ~6,000 kg, while maintaining an α of less than 30.

At the end of this last cycle (10 years into the program), stakeholders and investors would have seen three end-to-end demonstrations of a 200 kWe NEP system that continually improves the reactor α, possibly with a combination of different high-performance EP engines.

Approach B is higher risk than Approach A because it requires more investments for higher temperature operations of the reactor core elements, Li heat pipes, Stirling engines, and a radiation shield. Any one of these prototypes could be used as a high-fidelity baseline for a flight article. Approach B offers the option of flying a 200 kWe system with an α of 40 after the first 48 months but at a steep initial cost profile. Approach B requires 9 years.

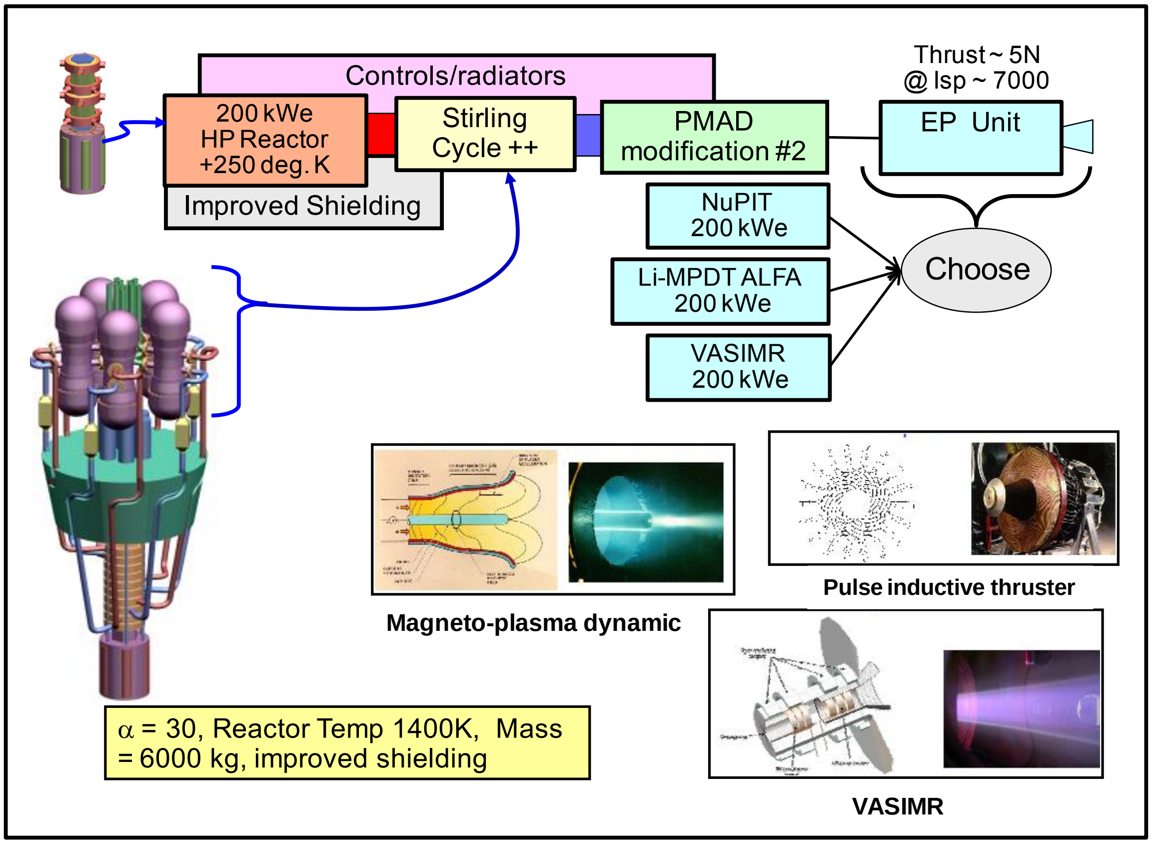


Figure 6-10. Approach B: NEP Rapid Prototyping Cycle #3

Source: Artwork courtesy of LANL. Photos courtesy of NASA.

C. Technology Challenges in Raising Reactor Temperature

For Approach B, the operating temperature of the 200 kWe reactor increases 250 K, from 1150 K to 1400 K. This modest increase in temperature will present several challenges to the reactor internal heat-pipe and nuclear-fuel components, to the shielding, and to the external energy-extraction methods. Nevertheless, successfully operating the NEP at 1400 K will increase the efficiency of the Stirling cycle energy-extraction process, reduce the external waste-heat radiator mass, and, therefore, reduce the overall system α.

The heat-pipe nuclear-fuel-rod design under consideration for this series of reactor modifications is shown in Figure 5-5. This particular traditional design uses UN fuel with refractory metals for the liner and fuel-rod cladding. The SP-100 design used Re liners and PWC-11 cladding, which demonstrated good strength and thermal conductivity and allowed for minor swelling of the UN pellets. The temperature increase of 250 K should not degrade the UN pellets. However, the PWC-11 material does exhibit creep[7] at these

[7] Creep occurs when a material at elevated temperature under constant load continues to deform. The rate of deformation can be constant over a long time but eventually accelerates resulting in structural fracture and failure.

higher temperatures. New refractory metal alloys would have to be investigated to improve the tensile and thermal creep strength limit. Additional refractory metal candidates are discussed in Appendix B. Creep data are somewhat limited for these metals. In 2004, no commercially available refractory metal with a design database sufficient for fabrication of a space nuclear reactor was available (Buckman 2004). In addition to performing well at elevated temperatures, these metals must survive the severe dynamic and acoustic loads in the launch environment.

As mentioned previously, the elevated operating temperatures of the reactor will also affect the performance and lifetime of the heat pipes that transport the thermal energy to the energy-extraction subsystems (TPV or Stirling engine). Li heat pipes were chosen because of their inherent high-temperature performance. However, as noted earlier, at elevated temperatures, any contaminates within a Li heat pipe will become a key source of accelerated corrosion and pipe failure. Therefore, manufacturing methods for purifying Li, pipe fabrication, and non-destructive inspection (NDI) processes will have to be improved during the 9 or 10 year effort to ensure the longevity of the design (Vasil’kovskii et al. 2000).

Appendixes B and C further identify higher temperature refractory metals and shielding materials suitable for the 1400 K operating reactor.

D. Nuclear Reactor Control System Simulator Design and Testing

Approaches A and B will only take the design of a NEP so far. Some level of nuclear testing must be accomplished to ensure that the appropriate design of the heat-pipe-cooled reactor is sound and to develop a control system with as much on-orbit autonomy as feasible. Previous efforts have used the concept of “zero-power” testing. Zero-power testing is designed to verify nuclear operation and safety characteristics of the reactor and is covered in detail in Appendix D.

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7. Cost Estimate and Available Facilities

Cost data are quite scarce for space nuclear reactor programs since the United States has only flown one, the SNAP-10A. The SP-100 program evaluated several reactor and power extraction subsystems but never really developed and evaluated an end-to-end prototype system that could survive the launch and space environment. Table 7-1 shows some cost data available from Kulcinski (2004).

Table 7-1. Historical SNAP-10 and SP-100 Funding Data

Only US reactor to fly SNAP-10A, no LV costs											
Program	GFY ($M)										
	1957	1958	1959	1960	1961	1962	1963	1964	1965	1966	
Reactor/shield subsystem	1.0	1.5	1.7	1.9	2.0	2.0	3.6	2.0	0.9	1.0	17.6
Power conversion subsystem				0.5	1.0	3.0	1.6	1.0	0.6		7.7
System integration ans other subsystems					0.9	8.3	10.0	7.2	7.0	2.4	35.8
											61.1
Reactor/shield subsystem	5.8	8.8	9.9	11.1	11.7	11.7	21.0	11.7	5.3	5.8	102.9
Power conversion subsystem	0.0	0.0	0.0	2.9	5.8	17.5	9.4	5.8	3.5	0.0	45.0
System integration ans other subsystems	0.0	0.0	0.0	0.0	5.3	48.5	58.5	42.1	40.9	14.0	209.3
	5.8	8.8	9.9	14.0	22.8	77.8	88.9	59.6	49.7	19.9	357.3
Notes:											
1966 to 2008 inflation multiplier 5.847											
Ref. "Space Reactor Electric Systems, Subsystem Technology Assessment, "ESG-DOE-13398, March 29, 1983 (page V-6)											

SP-100 Cost Data													
Year	1983	1984	1985	1986	1987	1988	1989	1990	1991	1992	1993	1994	1995	
Approximate Amount	16.0	16.0	20.0	27.0	64.0	94.0	76.0	59.0	50.0	46.0	35.0	19.0	1.0	523.0
Inflation multiplier of 1.39, 1995 to 2008	22.2	22.2	27.8	37.5	89.0	130.7	105.6	82.0	69.5	63.9	48.7	26.4	1.4	727.0

The SNAP-10A data are from an internal DOE assessment noted in Table 7-1. Both sets of funding numbers were multiplied by different factors to account for inflation since 1966 and 1995. As noted in this table, such technology endeavors could range from the hundreds of millions to almost a billion dollars.

In the spirit of those values plus experience by the author in developing research and development (R&D) programs, first-order estimates were made for the funding resources needed for the two rapid prototyping approaches (Approaches A and B; see Chapter 6) suggested for the development of a prototype NEP engineering unit (see Table 7-2 and Table 7-3). The final NEP engineering design unit would be a system designed for the launch (e.g., components and subsystems would have been subjected to qualification level testing) but not "hot" tested at full electrical load with nuclear fuel. Zero-power testing would be used to evaluate the different reactor designs (see Appendix D). The EP devices would be part of the EDU for full end-to-end T&E. Both approaches presented result in investment totaling \$1.2 billion and \$1.3 billion dollars for a 9 or 10 year effort. The annual funding distribution is different for the two approaches for the first rapid prototyping cycle.

Table 7-2. Approach A Funding Requirements

Approach A ($M)											
Year	1	2	3	4	5	6	7	8	9		
Rapid Prototyping Cycle 1	25 kWe system										
Cycle 2		100 kWe									
Cycle 3					200 kWe					Item Totals	FTE/year avg.
Item											
Prgm mgmt	2	2	3	3	3	3	3	3	3	25	8
GFE services	2	2	2	2	2	2	2	2	2	18	6
Radiators	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	5.4	2
Thermal	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	5.4	2
PMAD	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	5.4	2
System trades	1.2	1.2	1.2	1.2	1.2	1.2	1.2	1.2	1.2	10.8	4
Govt mgmt, safety	1.2	1.2	1.2	1.2	1.2	1.2	1.2	1.2	1.2	10.8	4
EP	3	8	12	12	8	8	8	8	2	69	23
TPV	5	10	10	10	10	10	5	5	5	70	23
Stirling system	5	15	55	35	35	35	20	10	5	215	72
Facility 500 kWt	10	40	30	10	10	10	10	5	5	130	43
25 kWe system	15	40	5							60	20
100 kWe system	5	15	60	80	30	5				195	65
200 kWe system				5	15	60	80	30	5	195	65
Yearly Total	51.2	136.2	181.2	161.2	117.2	137.2	132.2	67.2	31.2	1014.8	338
plus 20% reserves	61.44	163.44	217.44	193.44	140.64	164.64	158.64	80.64	37.44		
sub total 25kWe			442.32								
sub total 100 kWe						941.04					
sub total 200 kWe									1217.76		
Notes											
Minimal NEP flight safety											
Minimal program and govt. mgmt											
20% reserve added to each year											

Approach B requires more resources up-front than Approach A (~\$800 million for 48 months vs. ~\$440 million for 36 months) because at the end of the first cycle, Approach B has a 200 kWe NEP demonstrator vs. a 25 kWe system. The key difference from previous efforts is that a prototype NEP that could be launched is available at 36 months (Approach A) or 48 months (Approach B) from the start of the program. The concept presented here suggests that by demonstrating an end-to-end NEP, the TRLs of this type of engine will be raised quickly to a point where cost and schedule projections for space applications are much more certain. Either of the suggested approaches should result in continued investment by end users and the opportunity to fly an intermediate NEP configuration without waiting for the follow-on rapid prototyping cycles.

Table 7-3. Approach B Funding Requirements

Approach B ($M)												
Year	1	2	3	4	5	6	7	8	9	10		
Rapid Prototyping Cycle 1	1150 K reactor											
Cycle 2			1250 K reactor									
Cycle 3						1400 K reactor					Item Totals	FTE/year avg.
Item												
Prgm mgmt	2	2	2	3	3	3	3	3	3	3	27	9
GFE services	2	2	2	2	2	2	2	2	2	2	20	7
Radiators	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	6	2
Thermal	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	6	2
PMAD	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	6	2
System trades	1.2	1.2	1.2	1.2	1.2	1.2	1.2	1.2	1.2	1.2	12	4
Govt mgmt, safety	1.2	1.2	1.2	1.2	1.2	1.2	1.2	1.2	1.2	1.2	12	4
EP	3	15	15	10	8	6	2	2	2	2	65	22
TPV	5	10	10	2	0	0	0	0	0	0	27	9
Stirling system	15	55	45	40	30	55	20	20	5	5	290	97
Facility 500 kWt	10	40	30	20	5	5	5	5	5	5	130	43
200 kWe @ 1150K sys	15	80	110	50							255	85
200 kWe@ 1250 K sys		5	10	40	40	10	5				110	37
200 kWe@ 1400 K sys.					5	10	40	40	10	5	110	37
310												
Yearly Total	56.2	213.2	228.2	171.2	97.2	95.2	81.2	76.2	31.2	26.2	1076	359
plus 20% reserves	67.44	255.84	273.84	205.44	116.64	114.24	97.44	91.44	37.44	31.44		
sub total 1150 K sys				802.56								
sub total 1250 K sys							1130.88					
sub total 1400 K sys										1291.2		
Notes												
Minimal NEP flight safety												
Minimal program and govt. mgmt												
20% reserve added to each year												

Undertaking such a rapid prototyping program requires the modification of existing U.S. facilities. Three facilities still fully operational: MSFC, GRC, and Arnold Engineering Development Center (AEDC) have the necessary infrastructure to provide a heater power source and vacuum chamber that are large enough for the NEP non-nuclear testing.

The challenge comes in selecting a facility for the reactor “zero-power” testing to ensure that the reactor control simulator has minimal uncertainty. Appendix D details some of the zero-power test configurations used by Argonne National Laboratory for the SP-100. Unfortunately, the Argonne Laboratory facility used for evaluating the SP-100 design is no longer available. Manufacturing the UN pellets for zero-power testing would most likely come from LANL. INL would be a strong candidate for all necessary irradiation testing and “zero-power” testing for the subsection module of the nuclear reactor. Further work is required to assess these T&E facility options appropriately and to refine the cost estimates.

8. Summary

This report has attempted to summarize the current state of affairs for space nuclear reactor technology in the United States. An argument has been made to invest in this technology since future important space missions in the next 10 to 20 years will require power above 200 kWe. A few examples of such missions that relate to survivability and space dominance were presented. At 200 kWe, this work shows that solar power is not practical due to the size of the arrays and, depending on the orbit, the mass of the batteries. Unfortunately, nuclear reactor space technology must still overcome the negative political environment caused by two accidents that occurred many decades ago and resulted in minor terrestrial contamination. Two previous attempts—efforts that failed for a variety of reasons—to reenergize this space power technology were also presented. Currently, very little U.S. government or commercial funding is directed toward this critical space technology.

To address these issues, two rapid prototyping approaches were presented. These approaches build on the pioneering work undertaken by NASA to fabricate and test complete nuclear reactor power systems in a non-nuclear space simulated environment. Either of the approaches provides for a completed end-to-end system evaluation of launch- and space-qualified nuclear reactor EP systems in 36 to 48 month cycles. One approach begins with a 25 kWe NEP system and grows it to a 200 kWe system operating at nominal temperatures. The other approach starts with a nominal operating temperature 200 kWe NEP system and increases its temperature by 250 K. This increase in temperature improves the system's overall efficiency and reduces the system's α and, therefore, its mass by 2,000 kg. The second approach has higher technical risk but results in a more optimized NEP. These end-to-end system demonstrations, executed in a ground thermal vacuum environment (simulating space), would be complemented by a high-fidelity reactor control simulator developed from zero-power tests conducted in an appropriate facility.

Either one of these approaches would provide the detailed systems engineering data necessary to undertake a flight program with low uncertainty in performance, cost, and schedule estimates. Furthermore, the costly nuclear safety approval process could be delayed while system-level analysis and subsequent optimizations were completed. This degree of system-level testing was not done in the two previous attempts to reenergize this technology. The lessons learned from the previous failed attempts, coupled with that application of rapid prototyping acquisition and programmatic approaches, will also decrease the development risk for the investors and help to mitigate the negative political reactions for deploying this power technology in space. This combination should lead to continued investor interest and support to build flight articles to undergo the launch safety approval process. It will also redevelop U.S. engineering design and fabrication talent and reestablish the space nuclear reactor production facilities. However, further work is required to assess the current state of U.S. facilities and design capabilities for such an undertaking.

Appendix A. Flight Safety Design Considerations

If this rapid prototyping program is to be successful, the nuclear-powered electric propulsion (NEP) engineering design unit (EDU) systems must be designed to survive the launch environment and anomalies from the start. The eventual flight article must be able to meet the stringent U.S. safety review process necessary to obtain launch approval without major modifications. This process is well established and is summarized in several documents such as Presidential Directive/National Security Council Memorandum No. 25 (PD/NSC-25), the National Environmental Policy Act, and the Interagency Nuclear Safety Review Panel (INSRP) (Lenard 2006). In addition to this process, the United States will usually abide by international guidelines such as the *Principles Relevant to the Use of Nuclear Power Sources in Outer Space* (United Nations 1992). These guidelines follow the U.S. flight safety constraints very closely.

The work of Bennett et al. (1982) is the pioneering safety document created for the SP-100 program and lists in detail the safety design requirements for space nuclear reactors. Satisfying these requirements and possibly others will be necessary to complete the launch approval process. The SP-100 program had an extensive engineering effort to ensure the safety of the system to meet, as a minimum, the requirements shown in Table A-1.

Space nuclear reactors do not contain highly radioactive fuel like a Radioisotope Thermoelectric Generator (RTG). The two main safety concerns for launching a space reactor are that it goes critical and that the fresh fuel is dispersed. To address criticality, the reactor design has to be evaluated to prove that it will remain subcritical despite launch pad explosion, impact, water immersion, and/or burial. To ensure that no significant radioactive material is created throughout the program, the rapid prototyping and testing of the engineering unit heat-pipe reactor design is accomplished in a non-nuclear fashion (electric heaters). In addition, the reactor neutronics design characteristics and accident configurations must be evaluated using zero-power testing (see Appendix D). Therefore, the fueled flight reactor will begin to create fission products only when it resides in a stable orbit and is turned on for power (Lenard 2006).

To make this technology successful, the overall program safety goal is to ensure that minimal terrestrial contamination will occur if a launch failure or mission anomaly occurs or for end-of-life disposal. Figure A-1 depicts several possible situations that may occur

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Table A-1. Department of Energy (DOE)
Nuclear Safety Criteria and Specifications for Space Nuclear Reactor

Safety Design Requirements (Necessary but not Sufficient)	Comments
The reactor shall be designed to remain subcritical if immersed in water or other fluids to which it may be exposed.	Other fluids: rocket propellant
The reactor shall have a significantly effective negative power coefficient of reactivity.	Reactivity is positive when a reactor is supercritical, zero at criticality, and negative when the reactor is subcritical. Negative power coefficient of reactivity is defined as the rate of change of reactivity to the rate of change of the reactor thermal power.
The reactor shall be designed so that no credible launch pad accident, range safety destruct action, ascent, or reentry from space resulting in Earth impact could result in a critical or super critical geometry.	
The reactor shall not be operated until a stable orbit or flight path is achieved and must have a reboost capability from low earth orbit if it is operated in that orbit.	Except for zero-power testing yielding negligible radioactivity at the time of launch
Two independent systems shall be provided to reduce reactivity to a subcritical state.	Cannot be subject to common cause failure
The reactor shall be designed to ensure that there is an independent shutdown heat-removal system or independent heat removal paths within the heat-transport system to provide decay heat removal.	
The unirradiated fuel shall pose no significant environmental hazard	

Source: Bennett 1982.

on launch and in orbit. The term "Nuclear Safe Altitude" is not specifically identified by U.S. or United Nations' guidelines. It is loosely defined as an orbit that is greater than ~1,000 km, which appears to meet the requirement that the reactor fuel will decay sufficiently before reentry.

In Figure A-1 for event #1, the launch vehicle has immediately encountered a problem and must be destroyed by launch safety. In this case, the reactor core has to be separated from the launch vehicle and protected by a reentry shroud such that the fuel is localized at the impact site. After the shroud-protected core has been separated from the launch vehicle, range safety has to destroy the launch vehicle. Premature destruction of the launch vehicle would spread the core nuclear material over a wide area since the fuel is traveling too slowly to burn up. Though for a reactor the nuclear material is not radioactive at this point, fuel dispersal will still cause a serious public safety concerns and negative reactions.

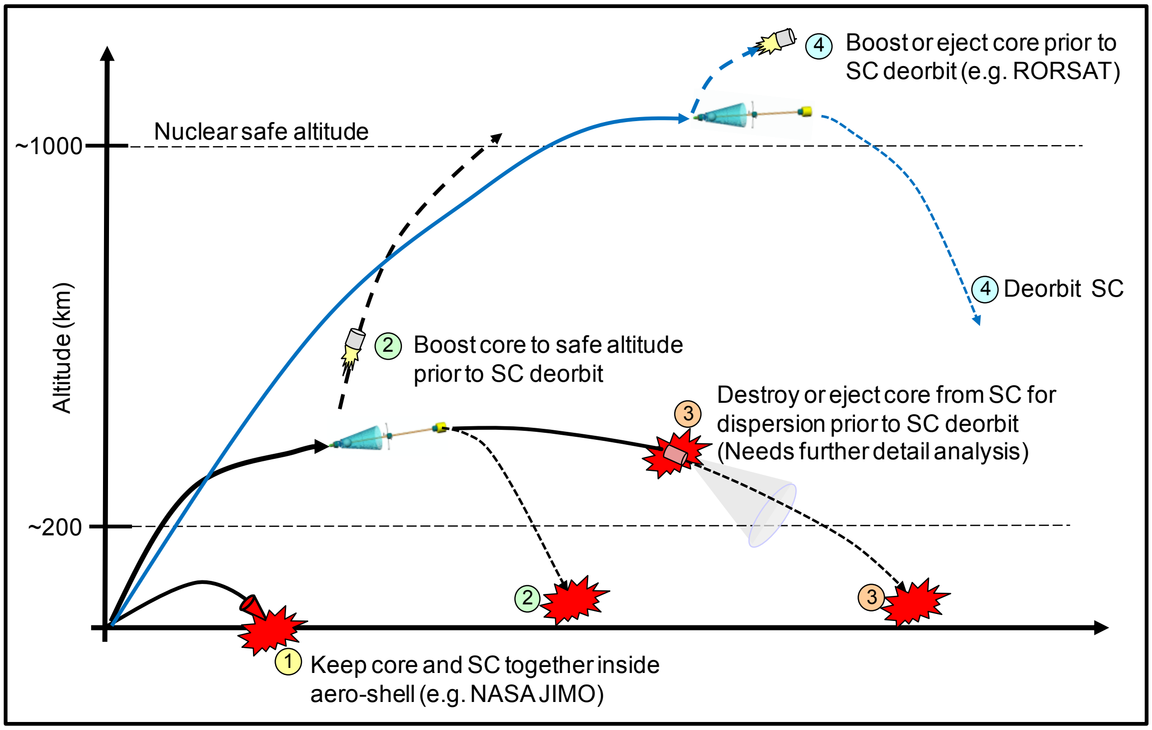

Figure A-1. Safe Reactor Core Placement Strategies

In Figure A-1 for event #2, the spacecraft is intact in orbit and is traveling ~7 km/sec, but the resultant orbit is below the nuclear safe orbit of ~1,000 km. In this case, the design could eject and boost the core to the safe altitude, or, as shown in event #3, it could eject and destroy the core and disperse the nuclear material. Since the fuel is traveling at orbital velocity, it would continue to break up and burn up as it falls to earth, thereby minimizing terrestrial contamination. As mentioned previously, since nuclear space reactors do not use plutonium like the U.S. Transit-5BN-3 RTG, worldwide contamination of a low half-life material would not occur.

The widespread radioactive contamination caused by the Soviet Cosmos 954 Radar Ocean Reconnaissance Satellite (RORSAT) spacecraft reactor was due to the fuel being temporarily protected from the reentry environment by the spacecraft/reactor structure and the very low mission altitude (less than 300 km). Given the worldwide negative reaction to this contamination event, the Soviets redesigned their RORSAT satellites. When Cosmos 1402 began its reentry, it ejected the reactor core from the spacecraft, which burned up in the atmosphere (Voegeli 2007). Nevertheless, properly designing the spacecraft to destroy/eject the core in a manner that optimally dispenses the nuclear material for reentry destruction needs to be investigated further to allow this accident mitigation option to be viable in today's political environment. Ideally, as shown in event #2, the best option would be to boost the core past the 1,000 km altitude.

Ejecting and boosting the core in a controlled manner requires a complex design with its own reliability challenges. Interesting to note, when Cosmos 1900 failed on orbit, it did succeed (September 30, 1988) in boosting its reactor core to ~720 km using a fail-safe backup system since its primary core boost system had also failed (Aftergood 1989).

In Figure A-1 for event #4, the spacecraft has reached the minimum nuclear safe orbit (greater than 1,000 km, circular or perigee). If the spacecraft fails or its mission end-of-life has occurred, the core can be ejected and/or boosted before deorbiting the spacecraft. Keeping the core intact at this or any parking orbit is important to avoid creating a debris cloud that will endanger other spacecraft at a variety of attitudes as the debris migrates around the planet.

Figure A-2 indicates top-level SP-100 safety design features. The most notable feature is the carbon-carbon (C/C) reentry shield designed to keep the reactor core at less than 300 K while the outside surface temperature reaches 3200 K. This reentry shield concept was also designed to keep all the core fuel together in the crater or water location upon impact. The National Space and Aeronautics Administration (NASA) Prometheus/ Jupiter Icy Moons Orbiter (JIMO) program followed the same design philosophy. Figure A-3 shows its reentry shroud.

Buden (1993) provides an in-depth overview of the additional SP-100 safety design features and lists several references involved in developing the engineering design details of this work.

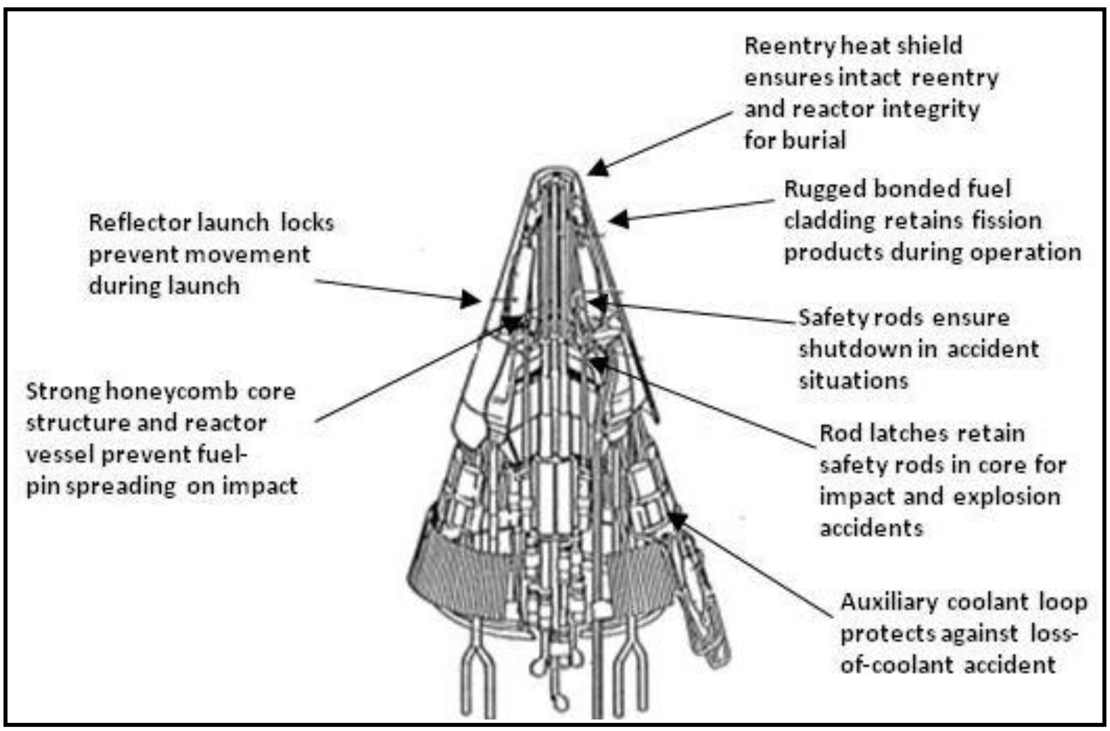

Figure A-2. SP-100 Reentry Safety Design

Source: Buden 1993. Photo provided by GE.

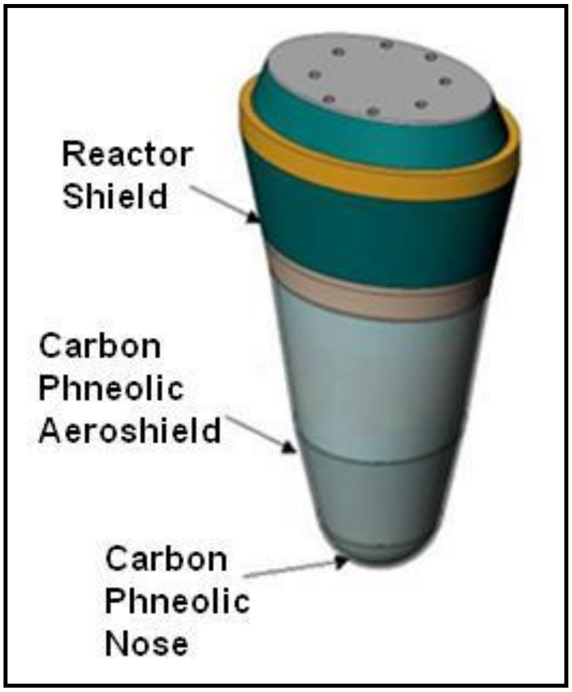

Figure A-3. NASA Prometheus/(JIMO) Reentry Shield Design

Source: Taylor 2005.

Appendix B. Heat-Pipe Reactors

Materials

The high-temperature heat-pipe reactor discussed in Chapters 5 and 6 and shown in Figure B-1 consists primarily of nuclear-fuel rods welded to lithium (Li) heat pipes, with filler material in the gaps. This traditional design concept has integral moving neutron reflectors and absorbers for reactor control. The operating temperature of the core will drive the requirements for the materials and fabrication techniques used. In this study, the minimum operating temperature was 1000 K and the maximum was 1400 K. As mentioned in the report, stainless-steel alloys are appropriate for operation around 1000 K, but refractory metals are required for temperatures above 1150 K. Therefore, for the fuel rods and the heat pipes, several refractory metals are available and have been explored in various references (e.g., Angelo and Buden 1985; El-Genk and Tournier 2004; Kapernick and Guffee 2002; Greenspan 2008; Hickman et al. 2010).

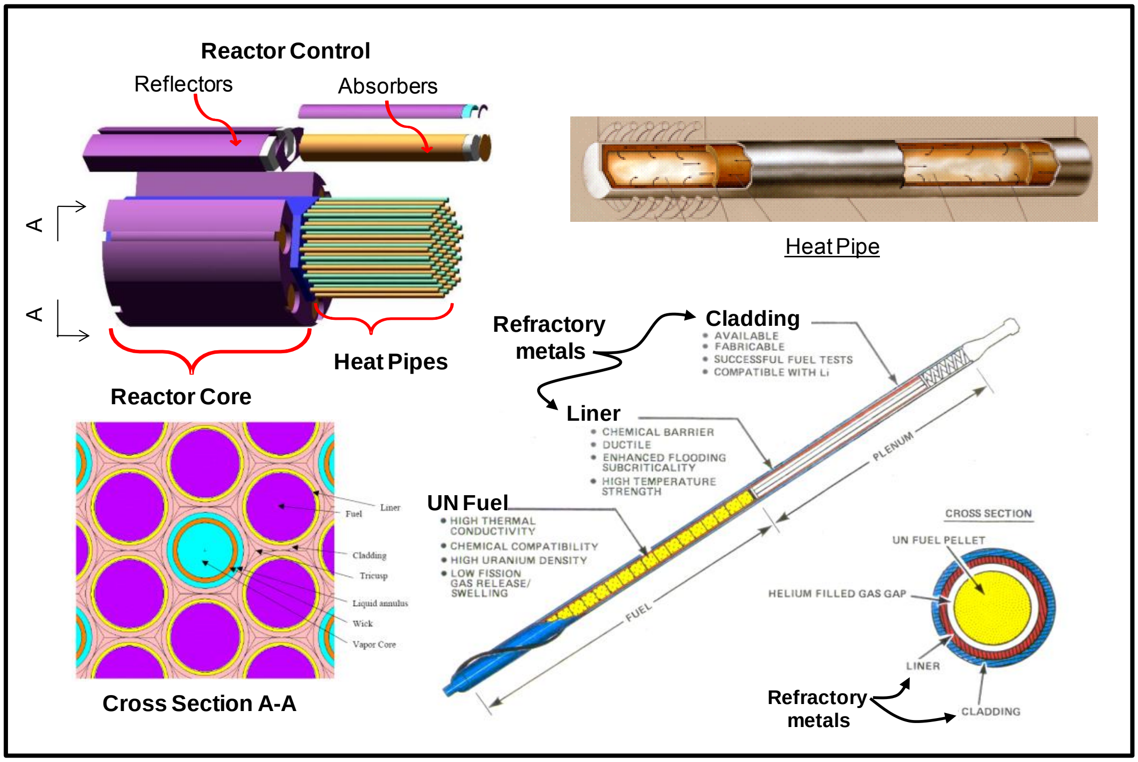

Figure B-1. Details of a Heat Pipe Reactor

Source: Greenspan 2008. Artwork courtesy of the Los Alamos National Laboratory (LANL) and the University of California.

The other key thermal issues for the materials used in the reactor core and associated subsystems are not to raise the ductile-to-brittle transition temperature (DBTT) into the range where launch will probably occur: 290 to 305 K. Raising the DBTT of refractory metals is usually the result of embrittlement (recrystallization) caused by welding in the presence of contaminates. The launch environment will generate severe dynamic stress environments at the weld points, and fracture may occur if the DBTT has increased.

The traditional space reactor design pioneered by the SP-100 program (see Figure B-1) initially used uranium nitride (UN) fuel, a rhenium (Re) liner, and niobium with 1% zirconium (Nb1Zr) cladding. This design allowed for good strength and thermal conductivity and minor swelling of the UN pellets. Operating the reactor at 1400 K does not degrade the UN pellets, but the Nb1Zr material exhibits creep at these higher temperatures. After a series of creep tests in 1986, the SP-100 program determined that Nb1Zr was an inadequate cladding, and PWC-11 (Nb-1Zr with 0.1% carbon) was chosen as a replacement.

For operations at 1400 K, new refractory metal alloys candidates will need to be considered due to their improved tensile and thermal creep strength limits at this elevated temperature (see Table B-1). Unfortunately, long-duration (multi-year missions) creep data are not available for most of these metals. This issue will have to be addressed before building an engineering qualification test article and an eventual flight article reactor. From the limited elevated temperature (1400 K) creep data taken in 2004, the highest performing metals were ASTAR-811C and T-111 (Buckman 2004).

At an operating temperature of 1400 K, the heat-pipe design used in this type of reactor contains liquid Li metal for the heat-transfer working fluid. Its structure will need to be made out of refractory metal. Past designs of the reactor core have the key components welded together in a controlled atmosphere to avoid contaminants that degrade the refractory metals. When a vacuum is available, electron beam welding is preferred. If no vacuum is available, gas tungsten arc welding is used with an inert gas blown over the welded area.

The reactor core is a precision mechanical design. Numerous references (Angelo and Buden 1985, Hrbud et al. 2003; El Genk and Tournier 2004; Greenspan 2008) have detailed heat-pipe reactor designs with corresponding thermal stress and neutronic analyses. To make the discussed rapid prototyping concept viable, the fuel-rod simulators using electric heaters would need to be engineered in a manner that precisely matches the thermal load interface (watts/cm) predicted for the reactor fuel rods and heat pipes. The design goal of the engineering unit test reactors would be to minimize any design modifications for the flight article.

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Table B-1. Candidate Reactor Core Materials for High-Temperature Operation Greater Than 1200 K

Material	Vendors	Reference	Comments
Nb-1Zr	Plansee Group	Angelo and Burden 1985	Nb1Zr to increase strength for temperatures above 873 K
PWC-11	Unknown	Buckman 1986	Nb-1Zr with an addition of 0.1% carbon; better creep performance than Nb-1Zr
Molybdenum (Mo)	Plansee Group, PMTI	Angelo and Burden 1985	Mo refractory metal; 2896 K melting, very low coefficient of thermal expansion
Mo-rehnium (Re)	Plansee Group, Rhenium Alloys	Hickman et al. 2010	Mo with 44.5 to 47.5% Re (good combination of high-temperature strength (up to 1800 K); ductility and resistance to oxygen embrittlement
T-111	Wah Chang	Hickman et al. 2010	Tantalum (Ta) alloy (tungsten (W)-8%, hafnium (Hf)-2%); good high-temperature performance, ductile following welding, easier to get, resistant to liquid metals given low impurities; has not been made for decades
ASTAR-811C	Wah Chang	Hickman et al. 2010	Ta alloy (W-8%, Re-1%, Hf-.7%, carbon); increased temperature performance but has not been made for decades
Mo-TZM	Plansee Group	Greenspan 2008	(Mo (~99%), titanium (Ti) (~0.5%), zirconium (Zr) (~0.08%) and some carbon; is a corrosion resisting Mo super alloy.; twice the strength of pure Mo and more ductile and weldable
Hastelloy XR	Magellan metals	Greenspan 2008	Developed by the Japan Atomic Energy Agency (JAEA); good for outer vessel applications, corrosion resistant
Li	Many vendors		Fluid used in heat pipe
Be, BeO	Many vendors		Neutron reflectors
B_4C	Many vendors		Neutron absorbers

As mentioned previously, the engineering unit test reactor core design is required to tolerate the extreme launch loads and acoustic environment without generating high static and dynamic stress concentrations at the welds and locations where large changes in cross-sectional inertia are found (i.e., corners). No reference could be found on whether any of the SP-100 components were subjected to launch vibration and acoustic analytical finite element or other stress analysis or actual laboratory testing. The SP-100 design was targeted initially to fly on the shuttle, which has a well-defined vibration and acoustic launch environment. However, since the space shuttle is being retired, a new launch spectrum will have to be identified for the reactor core and other nuclear-powered electric propulsion (NEP) subsystems. Candidate launch vibration and acoustic environments would most likely be from the Delta IV (heavy) or Atlas V heavy lift vehicle (HLV) due to the overall estimated mass of the NEP and candidate payloads (10,000 to 15,000 kg) and the minimum safe altitude of 1,000 km (perigee or circular).

The heat-pipe fast reactor concept in this study uses the traditional approach of movable control drums consisting of reflectors and absorbers to regulate the fission reaction by controlling the neutron density and, therefore, the power of the reactor (Angelo and Buden 1985). The baseline neutron reflectors chosen for this study are made out of beryllium (Be) or beryllium oxide (BeO), and the absorbers are made out of boron carbide (B_4C). Be and BeO radial reflectors have issues in a high-flux neutron environment. The Be radial reflector temperature should be below ~823 K to minimize void swelling, and the BeO axial reflector temperature should be kept above ~600 K to minimize microcracking. (Ragheb 2007) These temperature constraints warrant an investigation into other advanced Be alloys for reflectors such as the class of Be inter-metallics ($Be_{12}Ti$, $Be_{17}Ti_2$, $Be_{12}V$, $Be_{12}Mo$, Be_2Cr, and so forth.). The goal would be to increase the overall reactor temperature-operating environment and reduce degradation due to the high radiation environment (Sater 2003).

Special Fabrication Requirements

Hickman et al. (2010) provide a good assessment of the refractory materials noted in Table B-1 for use in a space reactor. They have an extensive list of references for work done during the SP-100 program. Key points in this reference are as follows:

- Fabrication of refractory metals can be complicated. The process involves cleaning and acid pickling, annealing and heat-treating, gas tungsten arc (in inert atmosphere) and electron beam welding (in vacuum), and extensive non-destructive evaluation (NDE).
- Vacuum welding has the additional problem of contaminants out-gassing in the chamber. Therefore, conditions in the vacuum chamber have to be monitored carefully.
- Refractory alloys are susceptible to cracking following welding due to low recrystallization ductility (raising the DBTT). One approach to resolving this complication is preheating the parts before welding and then post-heating afterwards to relieve stresses. Preheating/post-heating can be complicated because it must be done in a protective atmosphere to avoid contamination.

Li heat pipes were chosen in this study's baseline for reactor design because of the high-temperature environment. However, at elevated temperatures, any contaminates within a Li heat pipe can become a key source of accelerated corrosion and pipe failure. Therefore, manufacturing techniques for purifying Li and non-destructive inspection

(NDI) processes will have to be improved to ensure that less than 10 parts per million (ppm) of contaminates are contained within the heat pipe (Vasil'kovskii et al. 2000).

Nuclear-Fuel Availability

In this study, UN fuel was selected based on the work done in the SP-100 program. UN is a high-density, high-thermal-conductivity material (four to eight times that of uranium dioxide (UO_2)), which helps keep the temperature of the fuel low and allows the fuel to operate at a higher overall temperature. Although UO_2 is a more commonly available fuel, UN fuel would be preferred because of its lower thermal conductivity and greater mass.

A large quantity of UN fuel was made decades ago at LANL for the SP-100 program. To manufacture it today would probably require redeveloping this process either at the same place or perhaps at Idaho National Laboratory (INL). In support of the SP-100 program, the fabrication process made approximately 50,000 fuel pellets for about 75 fuel pins. These fuel pins were evaluated using irradiation testing. Cladding of the fuel pellets with PWC-11 was also accomplished (Demuth 2003).

Technology Issues

- Irradiation creep coefficient of refractory materials' data at high temperature.
- Expanded materials performance database for Mo-Re alloys, especially welded material to support fabrication.
- Updated American Society for Testing and Materials (ASTM) specifications for refractory materials and process controls (Hickman et. al 2010).
- Long-term refractory metals creep test data beyond 61,000 hr (2.8 years) at elevated temperature.
- Avoidance of raising the DBTT due to welding of the Mo-Re alloys. **Note:** The higher percentage of Re in Mo tends to mitigate this issue and keep the DBTT low, which is why the 47.5% alloy was investigated by the National Space and Aeronautics Administration (NASA) (Hickman et. al 2010).
- Embrittlement of tantalum (Ta) alloys due to of oxygen and hydrogen contamination.
- Extreme sensitivity of Zr and Hf to contamination by iron, copper, nickel, and cobalt (Hickman et al. 2010).
- U.S. vendor capability to go from bulk material production to final fabrication of refractory materials for nuclear components
- Contamination potential of refractory metal manufacturing annealing furnace (Hickman et. al 2010).

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Appendix C. Radiation Shield

Design and Fabrication

One of the most critical and complicated components of a space nuclear reactor is the radiation shield. The primary job of the shield is to protect the rest of the spacecraft from alpha and beta particles, energetic neutrons, gamma rays, and heat. This complex multi-functional structure protects the materials used in the reactor controls, energy-extraction subsystem, power management and distribution electronics, electric propulsion (EP) components, and whatever payload materials and electronics are onboard. Various concepts are discussed in the literature: boron carbide (B_4C) and lithium hydride (LiH) are used to absorb neutrons; and tungsten (W) or depleted uranium (DU) is used to slow the neutrons down for absorption and to attenuate the gamma radiation. During this process, the shield will heat up due to radiation, as will the reactor core itself (El-Genk and Tournier 2004). This waste heat must be transported from the shield (e.g., by heat pipes) and rejected into space using radiative heat transfer by the radiators.

To minimize the mass of the shield, a cone-like geometric structure that puts most of the spacecraft in a protected shadow, as shown in Figure C-1, must be built. This shadow concept works because there is no atmosphere to back-scatter the energetic neutrons. The Systems for Nuclear Auxiliary Power (SNAP)-10, SP-100, and National Space and Aeronautics Administration (NASA) Prometheus/Jupiter Icy Moons Orbiter (JIMO) designs used this approach. Furthermore, the SP-100 and Prometheus designs also used a structure for physical separation between the reactor and spacecraft payload. This separation helps due to the distance inverse square law for attenuating radiation. Figure C-1 is a sketch of typical space reactor configuration using a radiation shield to create a protected shadow zone and a separation boom to displace the spacecraft payload.

Materials Availability

As mentioned in Chapters 5 and 6, four primary materials have been explored in the past for the space nuclear reactor shield: LiH and B_4C as neutron absorbers and W and DU for gamma ray shielding and slowing down energetic neutrons to less than 1 MeV. LiH is a favored material because of its high hydrogen density (used today for hydrogen storage), low mass density, and reasonably high melting temperature (Angelo and Burden 1985).

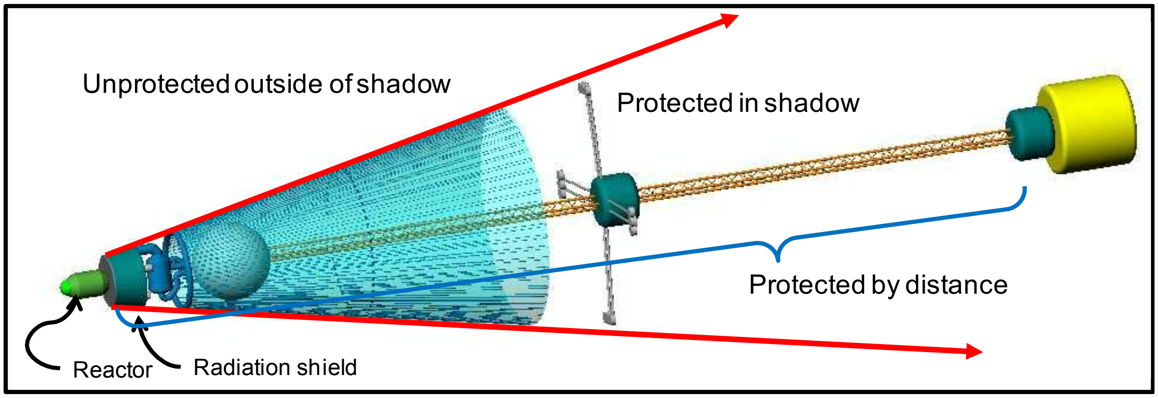

Figure C-1. Typical Geometry for a Shadow-Type Radiation Shield
Source: Artwork courtesy of the Los Alamos National Laboratory (LANL).

The shield for the SNAP-10 reactor was LiH and had a mass of 98 kg, or 22% (Angelo and Burden 1985) of the total reactor mass. For the SP-100 concept (depending on the reference), a multi-layer shield design consisted of layers of LiH, B_4C, W, or DU and had an estimated mass of 970 kg, or 21% of the total reactor mass (Demuth 2003). Using what appears to be the worst-case mass estimates for the NASA Prometheus/JIMO concept, the shield was 807 kg, or 13% of the total reactor mass (Taylor 2005). Given these percentages of total reactor mass, the shield is an area where mass saving could contribute significantly to overall mass reduction of the reactor and, therefore, a higher α (specific mass kg/kWe).

Special Materials Processing

W, DU, and B_4C are relatively easy to get and can be readily fabricated by many vendors into a variety of shapes. LiH is difficult to make and is created by heating lithium (Li) in a hydrogen gas at high temperatures. For the SP-100 program, this manufacturing was done at Oak Ridge National Laboratory (ORNL) (Ragheb 2007). Today, it can be purchased in large quantities in powdered form from American Elements. To make the shield, a cold press technique is used to form the material. However, since LiH reacts with moisture, manufacturing must be done in a very dry environment (Angelo and Burden 1985).

Technology Issues

LiH has a very narrow temperature operating range and has poor thermal conductivity (0.05 W/cm/K from 400 to 800 K). Early work in the SP-100 program showed that at temperatures below 600 K, their samples had as much as ~25% swelling when irradiated. This swelling was caused by chemical dissociation and phase changes due to heating and by gamma attenuation and helium production due to neutron plus Li reactions. Operations above 700 K caused dehydrogenation in the shield, which decreases its neutron absorption. At 962 K, the material begins to melt. One way to increase the operating temperature of the LiH shield is to enclose it in a thin shell of stainless steel. The operating temperature can be increased to 800 K before internal pressure causes the hydrogen to permeate the stainless-steel shield. The SP-100 design settled on an operational range between 600 to 700 K (Ragheb 2007). This narrow temperature range requires complex thermal management design since several penetrations by heat pipe and control structures through the shield may add to the heating load.

The SP-100 shield design evolved to meet this critical temperature range requirement by pressing the LiH into a stainless-steel honeycomb-like structure to separate the LiH from beryllium (Be) to resolve material incompatibilities. To reduce the thermal load further and keep the LiH material at less than 700 K, multi-layered walls consisting of W, Be, and B_4C were positioned between the stainless-steel-enclosed LiH and the reactor. Temperature control of the shield was one of the top 10 design issues of the SP-100 in 1987 (Buden 1993). Since the goal of this approach is to increase the temperature of the reactor further (up to 1400 K) to reduce α (kg/kWe), the overall mass of the system per power output investigation of new shield materials is warranted to widen the system design margin.

A relatively recent study (Anghaie et al. 2006) showed that enrichment of natural lithium borohydride ($LiBH_4$) to $^{6}Li^{10}BH_4$ enhances neutron (less than 1MeV) shielding. Other material candidates include lithium deuteride (LiD), LiZrH, $Be^{2}H_4$, LiH infiltrated with high-conductivity graphite foam, Li^6, Li^7, and hydrogen-infiltrated carbon nanotubes and boron fiber composites. To initially slow neutrons below 1 MeV, candidate substitute materials instead of W and DU include WB_2, TaB_2, and zirconium hydride (ZrH).

El-Genk and Tournier (2004) provide a detailed discussion of a shield design for a 110 kWe heat-pipe reactor. In their concept, graphite is used in addition to W, LiH, ^{7}LiH, Molybdenum (Mo), and 316 stainless steel to optimize radiation shielding performance.

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Appendix D. Zero-Power Testing

Zero-power testing is designed to verify nuclear operation and safety characteristics of the reactor. This testing is called “zero power” because the reactor is operated at a power low enough that it does not become significantly radioactive. The key assumption in this type of testing is that a fission reactor performs identically at any power (from milliwatts to megawatts) provided its geometry and temperature are the same. Therefore, zero-power operations can provide data pertinent to all modes of operation, including accident mitigation studies.

The best example of zero-power testing was done by Argonne National Laboratory (ANL) for the SP-100 program (LeSage 2001). ANL developed a facility that supported testing a variety of nuclear reactor configurations from 1963 to 1990. This facility, called the Zero-Power Physics Reactor (ZPPR), was used to study the design of the SP-100 space nuclear reactor. Those tests, identified as ZPPR-16 and ZPPR-20, were conducted from April 1988 to January 1989. This unique facility tested six different configurations of the SP-100 design, including launch vehicle failed conditions such as submersion in water (Phase C) or being buried in the ground (Phase D).

Figure D-1 shows a sketch of the ANL ZPPR facility, which allowed for a wide variety of reactor configurations to be tested. Each critical reactor configuration was labeled an “assembly,” and testing could last as long a month or years depending on the reactor under study.

An “assembly” is created by inserting nuclear fuel, reactor structural materials, coolant, reflectors, and control materials in “drawers” that were positioned in such a manner that they represented the reactor core. For the SP-100 program, the drawers contained rhenium (Re), zirconium (Zr) alloy, lithium hydride (LiH), high-purity Li^7, and hafnium (Hf). Figure D-1 shows the distribution of the drawers in the wall of stainless-steel square tubes for the various SP-100 test configurations. In particular, the Phase B “assembly” is shown for the reactor configuration with all the beryllium (Be) reflectors closed. Phase C “assembly” is for evaluating the reactor performance if it were totally submerged in water due to a launch vehicle failure. CH_2 gas was used as the water simulator. After each assembly is created on the opposing walls, the walls are pressed together hydraulically. This arrangement represents the complete reactor core simulator. As experiments are performed, measurements are taken throughout the ZPPR facility. The measurements are compared to a variety of reactor design tools to assess their uncertainty (LeSage 2001).

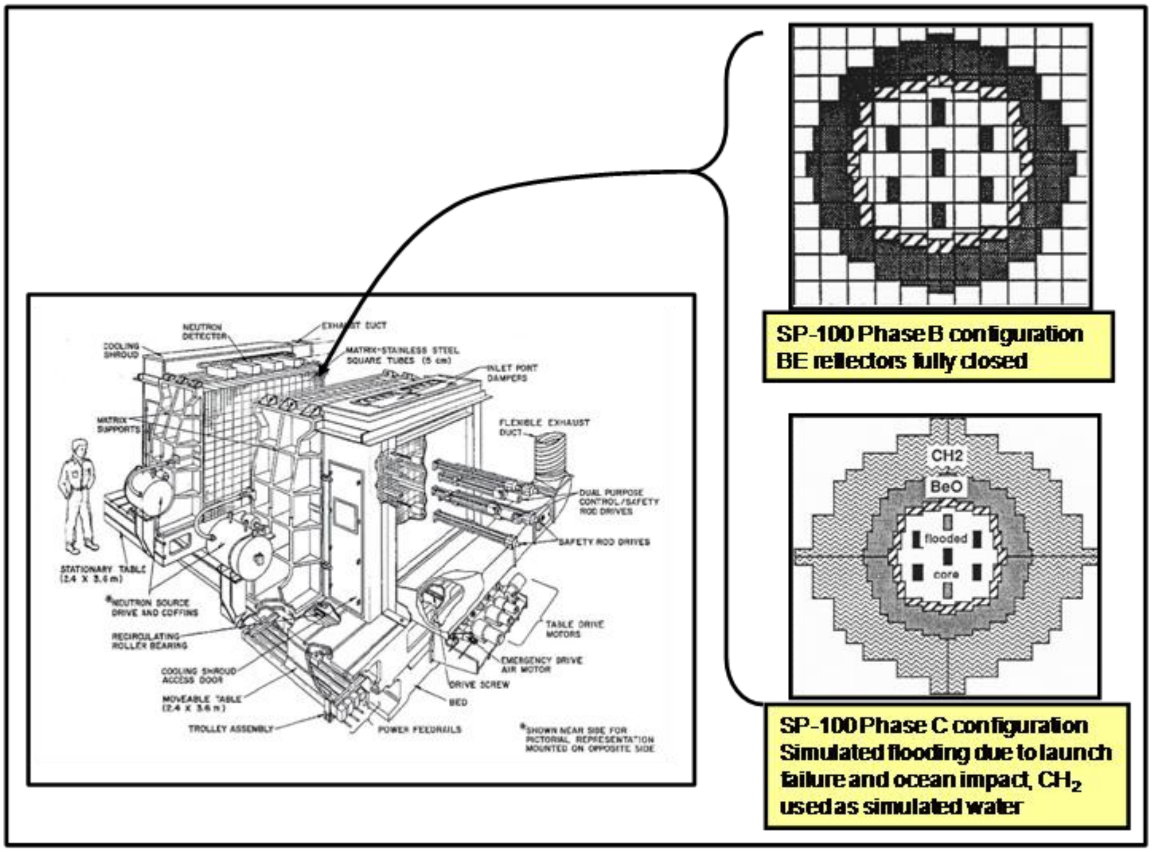

Figure D-1. Zero-Power Physics Reactor #20
Assembly (SP-100) With Phase B and C Configurations

Source: LeSage 2001.

Examples of a few measurements are as follows:

- Spatial measurements: confirm spectral convergence in the test zone, small sample reactivity worths,[8] gamma heating, fission and capture rates;
- Central reaction rate ratios;
- Central material reactivity worths for the constituent materials (e.g., control rods);
- Neutron spectrum and kinetic parameters; and
- Detailed fission and capture rate distributions for different configurations and at specific locations (e.g., for different control rod patterns).

[8] The function describing the relative reactivity effect of a given material in various positions in a reactor is the "reactivity worth" function of that material.

Examples of a few types of tests done are as follows:

- Subcritical and startup measurements (source-detector transfer function);
- Static and dynamic measurements at ambient temperature (reactor control system transfer function);
- Static and dynamic measurements with external heating;
- Temperature coefficient of reactivity; and
- Control rod performance.

Since the ANL facility was decommissioned in 1990, determining locations for such testing today is an important decision in the design of this entire rapid prototyping program concept. As noted previously, this testing is not only crucial for the developing the nuclear reactor subsystem controls, but also proves that the core design will remain subcritical under certain accident conditions. These demonstrations are required to obtain launch approval.

A space nuclear reactor is different from a terrestrial reactor in that it will have to operate semi-autonomously in space for any rapid degradation due to micro-meteorite and debris damage, attack, loss of communication and/or other nuclear-powered electric propulsion (NEP) or satellite subsystem failures. During the previously discussed rapid prototyping cycles (see Chapter 6), a control system would be developed and demonstrated. This control system would be capable of performing several different autonomous control decisions with the electrically heated operating nuclear reactor coupled to a virtual simulation of the nuclear reactor system. Zero-power testing is necessary to minimize the uncertainty in the reactor control simulator. Parry et al. (2009) compare zero-power testing data results to analytical models being developed for future reactor designs.

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Illustrations

Figures

Tables

References

Abelson, R. D. (ed.), T. S. Balint, K. Coste, J. O. Elliott, J. E. Randolph, G. R. Schmidt, T. Schriener, J. H. Shirley, and T. R. Spiker. 2005. *Expanding Frontiers With Standard Radioisotope Power Systems*. Report JPL D-28902. Pasadena, CA: Jet Propulsion Laboratory, California Institute of Technology. http://www.pdfcari.com/Expanding-Frontiers-with-Standard-Radioisotope-Power-Systems.html.

Aftergood, S. 1989. "Background on Space Nuclear Power." *Science and Global Security* 1: 93–107.

Angelo, J. A., and D. Buden. 1985. *Space Nuclear Power*. Malabar, FL: Krieger Publishing Company.

Anghaie, S., A. Charmeau, and L. Padilla. 2006. "Lithium Borohydride for Hydrogen Propellant Storage." In *Hydrogen Research for Spaceport and Space-Based Applications: Hydrogen Production, Storage, and Transport*, edited by Tim Anderson and Canan Balaban, 199–206. NASA Contractor Report NASA/CR—2008-215440/Part 3. Cleveland, OH: Glenn Research Center. http://gltrs.grc.nasa.gov/reports/2008/CR-2008-215440-PART3.pdf.

Auweter-Kurtz, M., C. Bruno, D. Fearn, H. Kurtz, T. J. Lawrence, and R. A. Lenard. 2005. "IAA Commission III: SG-2 – "Nuclear Space Power and Propulsion." Prepared for Commission 3 (Space Technology and System Development) of the International Academy of Astronautics (IAA). http://iaaweb.org/iaa/Studies/nuclearpropulsion.pdf.

Bennett, G. L., R. E. Smith, Jr., J. J. Lombardo, and B. J. Rock. 1982. *Nuclear Safety Criteria and Specifications for Space Nuclear Reactors.* Office of Space Nuclear Projects Report ONSP-1, REV. O. Washington, D. C.: U. S Department of Energy, Office of Space Nuclear Projects. http://www.fas.org/nuke/space/osnp-1.pdf.

Bennett, G. L., R. J. Hemler, and A. Schock. 1996. "Status Report on the U.S. Space Nuclear Program." *Acta Astronautica* 38 (4–8) (February–April): 551–560.

Bennett, G. L. 2006. "Space Nuclear Power: Opening the Final Frontier." Paper (AIAA 2006-4191) presented at the *4th International Energy Conversion Engineering Conference (IECEC)*, June 26–29, San Diego, California. http://www.fas.org/nuke/space/bennett0706.pdf.

Buckman, R. W. 2004. "Nuclear Space Power Systems Materials Requirements." In *Proceedings of the American Institute of Physics (AIP) Conference* 699: 815–820.

Buden, D. 1993. *Summary of Space Nuclear Reactor Power Systems 1983–1992.* Idaho National Engineering Laboratory. Prepared for the Tenth Anniversary Book for Symposium on Space Nuclear Power and Propulsion, Albuquerque, New Mexico, January 10–14.

Cassady, R. J., R. H Frisbee, J. H. Gilland, M. C. Houts, M. R. LaPointe, C. M. Maresse-Reading, S. R. Oleson, J. E. Polk, D. Russell, and A. Sengupta. 2008. "Recent Advances in Nuclear-Powered Electric Propulsion for Space Exploration." *Energy Conversion and Management* 49 (3) (March): 412–435.

Demuth, S. F. 2003. "SP-100 Space Reactor Design." *Progress in Nuclear Energy* 42 (3): 323–359.

English, R. 1987. *Atomic Power In Space, A History.* Department of Energy Report DOE/NE/32117-H1. Washington, D. C: Planning and Human Systems, Inc. http://www.fas.org/nuke/space/history.pdf.

El-Genk, M., and Tournier, J-M. 2004. "Conceptual Design of HP-STMCs Space Reactor Power System for 110 kWe." In *Proceedings of the American Institute of Physics (AIP) Conference* 699: 658–672.

Global Industry Analysts, Inc. 2011. "World Satellite Transponder Demand to Exceed 7,150, 36 MHZ Transponder Equivalents (TPEs) by 2015, According to New Report by Global Industry Analysts, Inc." http://www.strategyr.com/pressMCP-1211.asp.

Goebel, D. M., and I. Katz. 2008. *Fundamentals of Electric Propulsion: Ion and Hall Thrusters*. Hoboken, NJ.: John Wiley and Sons.

Greenspan, E. 2008. *Solid-Core Heat-Pipe Nuclear Battery Type Reactor.* Summary Report, University of California Department of Nuclear Engineering. Award Number DE-FC07-05ID14706. http://www.osti.gov/bridge/servlets/purl/940911-sHwPXv/940911.pdf.

Hickman, R. R., J. J. Martin, G. R Schmidt, T. J. Godfroy, and A. J. Bryhan. 2010. *Cost Estimate for Molybdenum and Tantalum Refractory Metal Alloy Flow Circuit Concepts.* NASA Report NASA/TM—2010–216431. Huntsville, AL: Marshall Space Flight Center.
http://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20100021082_2010021557.pdf.

Hoyt, R. P., and B. Minor. 2005. "Remediation of Radiation Belts Using Electrostatic Tether Structures." Aerospace Conference, 2005 IEEE, 5–12 March. IEEEAC Paper #2.0801.
http://ieeexplore.ieee.org/stamp/stamp.jsp?tp=&arnumber=1559348&isnumber=33126.

Hoyt, R. P., and M. Cash. 2007. *Reduction of Trapped Energetic Particle Fluxes in Earth and Jovian Radiation Belts, Electrostatic Tether Radiation Belt Remediation*. Tethers Unlimited, Inc. Final Report on Subaward 07605-003-061 on Prime Contract NAS5-03110, NIAC. Prepared for NASA Institute for Advanced Concepts (NIAC). http://www.tethers.com/papers/Hoyt_ES_RBR_Final.pdf.

Hoyt, R. P. 2008. "System Design and Performance Analysis of a Propulsive Electrodynamic Tether for Orbital Maneuvering." JANNAF Paper SPS-VII-01 presented at the 6th Modeling and Simulation Subcommittee/4th Liquid Propulsion Subcommittee/3rd Spacecraft Propulsion Subcommittee Joint Meeting, December 8–12, Orlando, Florida.

Hrbud, I., M. Van Dyke, M. Houts, and K. Goodfellow. 2003. "Non-Nuclear NEP System Testing." In *Proceedings of the American Institute of Physics (AIP) Conference* 654: 533–539.

Iannetta, B., and T. Maltk. 2009. "U.S. Satellite Destroyed by Space Collision." http://www.space.com/5542-satellite-destroyed-space-collision.html.

Kapernick, R., and R. M. Guffee. 2002. *Thermal Stress Calculations for Heat-Pipe-Cooled Reactor Power System.* Los Alamos National Laboratory Report LA-UR-01-6386. Paper submitted to the Space Technology and Applications International Forum, Albuquerque, NM. http://library.lanl.gov/cgi-bin/getfile?00796595.pdf.

Kulcinski, G. L. 2004. "Lecture 23-Nuclear Power in Space." Fusion Technology Institute, University of Wisconsin – Madison. http://fti.neep.wisc.edu/neep533/SPRING2004/lecture23.pdf.

Lenard, R. X. 2006. "Nuclear Safety, Legal Aspects and Policy Recommendations for Space Nuclear Power and Propulsion Systems." *Acta Astronautica* 59 (1–5) (July–September): 398–412.

LeSage, L. G. 2001. *An Overview of the Argonne National Laboratory Fast Critical Experiments 1963–1990*. Nuclear Technology (NT) Technical Memorandum ANL-NT-175. Argonne, IL: Argonne National Laboratory. http://irphep.inl.gov/d/anl_nt_175.pdf.

Madrigal, A. 2009. "Russia Leads Nuclear Space Race After U.S. Drops Out." *Wired.* http://www.wired.com/wiredscience/2009/11/nuclear-propulsion-in-space/.

Mason, L. S., D. Palac, M. Gibson, M. Houts, J. Warren, J. Werner, D. Poston, A. L. Qualls, R. Radel, and S. Harlow. 2011. "Design and Test Plans for a Non-Nuclear Fission Power System Technology Demonstration Unit." Paper presented at the Nuclear and Emerging Technologies for Space (NETS-2011), February 7–10, Albuquerque, NM.

Office of the President of the United States. 2010. *National Space Policy of the United States of America*. http://www.whitehouse.gov/sites/default/files/national_space_policy_6-28-10.pdf.

Parry, J. R., J. D. Bess, B. T. Rearden, and G. A. Harms. 2009. *Assessment of Zero-Power Critical Experiments and Needs for a Fission Surface Power System*. Idaho National Laboratory Report INL/CON-08-15138. http://www.inl.gov/technicalpublications/Documents/4282343.pdf.

Poston, D. I., R. J. Kapernick, and R. M Guffee. 2002. "Design and Analysis of the SAFE-400 Space Fission Reactor." *Proceedings of the American Institute of Physics (AIP) Conference* 608: 578–588.

Ragheb, M. 2007. "Space Power Reactors." In *Nuclear, Plasma, and Radiation Science: Inventing the Future* (Web Text), Part VI, Space Nuclear Power, Chapter 2. https://netfiles.uiuc.edu/mragheb/www/NPRE%20402%20ME%20405%20Nuclear%20Power%20Engineering/Space%20Power%20Reactors.pdf.

Sater, J. M. 2003. "Trip Report to ORNL." *Memorandum for Record*. Alexandria, VA: Institute for Defense Analyses (IDA).

Sharps, P. R., D. Aiken, B. Cho, D. Chumney, A. Cornfeld, J. Diaz, P. Patel, M. Stan, F. Newmann, and T. Varghese. 2010. "High Efficiency, Multi-Junction Solar Cells: Present Status and Future Directions." Briefing presented at the Advanced Concepts in Semiconductor Materials and Devices for Energy Conversion (ACSMDEC) Workshop sponsored by the Army Research Laboratory, December 7–8, Beltsville, Maryland. http://arlevents.com/energy/agenda.html.

Taylor, R. 2005. *Prometheus Project Final Report*. NASA JPL Doc. 982-R120461. Pasadena, CA: Jet Propulsion Laboratory, California Institute of Technology. http://trs-new.jpl.nasa.gov/dspace/bitstream/2014/38185/1/05-3441.pdf.

Teofilo, V. L., P. Choong, W. Chen, J. Chang, and Y.-L Tseng. 2008. "Thermophotovoltaic Energy Conversion for Space." *Journal of Physical Chemistry C* 112 (21), 7841–7845. http://pubs.acs.org/doi/pdfplus/10.1021/jp711315c.

U. S. Congress, Office of Technology Assessment. 1988. *SDI: Technology, Survivability, and Software,* OTA-ISC-353. Washington, D.C.: U.S. Government Printing Office. http://govinfo.library.unt.edu/ota/Ota_3/DATA/1988/8837.PDF.

United Nations. 1992. *Principles Relevant to the Use of Nuclear Power Sources in Outer Space.* U.N. Document A/47/49. http://stage.tksc.jaxa.jp/spacelaw/kokusai_utyu/kokuren_so_kaigi/D_gensiryoku.pdf.

Vasil'kovskii,V.S., V. P. Smetannikov, V. A. Evtikhin, I. E. Lyublinskii, V. A. Vertkov, and A. N. Chumanov. 2000. "High-Temperature Heat Pipes for Power Modules of Space Nuclear Power Systems." *Atomic Energy* 89 (1): 591–596.

Voegeli, N. 2007. "Space Nuclear Reactors: History and Emerging Policy Issues." *The Nonproliferation Review* 14 (1): 163–175. http://pdfserve.informaworld.com/703400_731261154_772325295.pdf.

Wertz, J. R., and W. J. Larson (eds.). 1999. *Space Mission Analysis and Design*, 3rd ed. London: Springer-Verlag Ltd.

Wilson, T. 2000. "Threats to United States Space Capabilities." Prepared for the Commission To Assess United States National Security Space Management and Organization. http://www.fas.org/spp/eprint/article05.html

World Nuclear Association. 2011. "Nuclear Reactors for Space." http://www.world-nuclear.org/info/inf82.html.

Abbreviations

α	specific mass kg/kWe
2D	two-dimensional
3D	three-dimensional
A	ampere
ACSMDEC	Advanced Concepts in Semiconductor Materials and Devices for Energy Conversion
AEDC	Arnold Engineering Development Center
AEHF	Advanced Extremely High Frequency
AIAA	American Institute of Aeronautics and Astronautics
AIP	American Institute of Physics
ALFA	advanced lithium-fed applied-field Lorentz force accelerator
AMTI	Airborne Moving Target Indication
ANL	Argonne National Laboratory
ANP	Aircraft Nuclear Propulsion
ARC	Ames Research Center
ASAT	anti-satellite
ACSMDEC	Advanced Concepts in Semiconductor Materials and Devices for Energy Conversion
ASTM	American Society for Testing and Materials
ATM	asynchronous transfer mode
Bettis	Bettis Atomic Power Laboratory
BMDO	Ballistic Missile Defense Organization
C4ISR	command, control, communications, computers, intelligence, surveillance, reconnaissance
CONOPS	concept of operations
CR	Contractor Report
DARPA	Defense Advanced Research Projects Agency
DBTT	ductile-to-brittle transition temperature
DC	direct current
DEW	directed energy weapon
DoD	Department of Defense
DOE	Department of Energy
DSV	Deep Space Vehicle

ED	electrodynamic
EDU	engineering design unit
EOL	end-of-life
EP	electric propulsion
ES	electrostatic
FEL	free electron laser
FOV	field of view
FSU	Former Soviet Union
FY	Fiscal Year
GEO	geostationary orbit
GFE	government-furnished equipment
GFS	Generic Flight System
GPHS	general-purpose heat source
GRC	Glenn Research Center
HAND	high-altitude nuclear detonation
HCPC	Hollow Cathode Plasma Contactor
HEO	highly elliptical orbit
HiPEP	High Power Electric Propulsion
HLV	heavy lift vehicle
HP-STMC	Heat Pipe–Segmented Thermoelectric Module Converter
IAA	International Academy of Astronautics
IC	Intelligence Community
IDA	Institute for Defense Analyses
IECEC	International Energy Conversion Engineering Conference
IEEE	Institute of Electrical and Electronics Engineers
IEEEAC	IEEE Aerospace Conference
INL	Idaho National laboratory
INSRP	Interagency Nuclear Safety Review Panel
IR	infrared
Isp	specific impulse, seconds
ISR	intelligence, surveillance, and reconnaissance
ISS	International Space Station
JAEA	Japan Atomic Energy Agency
JANNAF	Joint Army Navy NASA Air Force
JIMO	Jupiter Icy Moons Orbiter
JPL	Jet Propulsion Laboratory
KAPL	Knolls Atomic Power Laboratory
KSC	Kennedy Space Center

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kWe	kilowatts of electrical energy
LA-UR	LA-URs are unclassified documents that have been reviewed by SAFE-1, the Classification Office
LADAR	laser radar
LANL	Los Alamos National Laboratory
LEO	low earth orbit
Li-MPDT	Lithium-fed Magnetoplasmadynamic Thruster
LO	low observable
MEO	mid-earth orbit
MeV	megaelectronvolt
MILSATCOM	military satellite communications
MOA	Memorandum of Agreement
MOCVD	metal oxide chemical vapor deposition
MOU	Memorandum of Understanding
MSFC	Marshall Space Flight Center
NASA	National Space and Aeronautics Administration
NDE	non-destructive evaluation
NDI	non-destructive inspection
NEP	nuclear-powered electric propulsion
NETS	Nuclear and Emerging Technologies for Space
NEXIS	Nuclear Electric Xenon Ion System
NGST	Northrop Grumman Space Technology
NIAC	NASA Institute for Advanced Concepts
NR	Naval Reactor
NRPCT	Naval Reactors Prime Contractor Team
NSC	National Security Council
NSTAR	NASA Solar Technology Application Readiness
NT	Nuclear Technology
NuPIT	Nuclear-Electric Pulsed Inductive Thruster
ONSP	Office of Space Nuclear Projects
ORNL	Oak Ridge National Laboratory
OTA-ISC	Office of Technology Assessment–International Security and Commerce
PD	Presidential Directive
PMAD	power management and distribution
PNNL	Pacific Northwest National Laboratory
ppm	parts per million
PV	photovoltaic
R&D	research and development

RF	radio frequency
RORSAT	Radar Ocean Reconnaissance Satellite
RPS	Radioisotope Power System
RTG	Radioisotope Thermoelectric Generator
SAFE	Safe Affordable Fission Engine
SBR	space-based radar
SDIO	Strategic Defense Initiative Organization
SIGINT	signal intelligence
SNAP	Systems for Nuclear Auxiliary Power
SNL	Sandia National Laboratory
SOSI	Space Object Surveillance and Identification
SPAR	Space Power Advanced Reactor
SRG	Stirling Radioisotope Generator
SSA	space situational awareness
T&E	test and evaluation
TDU	Technology Demonstration Unit
TPE	Transponder Equivalent
TPV	thermophotovoltaic
TRL	Technology Readiness Level
U.S.	United States
UAV	unmanned aerial vehicle
USSR	Union of Soviet Socialist Republics
VASIMR	Variable Specific Impulse Magnetoplasma Rocket
VDC	volts direct current
VHITAL	very high Isp thruster with anode layer
ZPPR	Zero-Power Physics Reactor

Chemical abbreviations:

B_4C	boron carbide
Be	beryllium
BeO	beryllium oxide
C/C	carbon-carbon
CH_2	methylene
Cr	chromium
DU	depleted uranium
GaAs	gallium arsenide
GaSb	gallium antimonide

Hf	hafnium
InAs	indium arsenide
InGaAs	indium gallium arsenide
InGaAsSb	indium gallium arsenide antimonide
Ir	iridium
Li	lithium
LiBH4	lithium borohydride
LiD	lithium deuteride
LiH	lithium hydride
LN_2	liquid nitrogen
Mo	Molybdenum
NaK	a sodium-potassium compound
Nb	niobium
Nb-1Zr	niobium with 0.1% zirconium
NiH_2	nickel hydrogen
Os	osmium
Re	rhenium
Ru	ruthenium
Ta	tantalum
Ti	titanium
UN	uranium nitride
UO_2	uranium dioxide
V	vanadium
W	tungsten
Zr	zirconium
ZrH	zirconium hydride

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