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**INVESTIGATION OF THERMOPHOTOVOLTAIC CELL
DESIGNS FOR USE IN SPACE POWER SYSTEMS**

by

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FOR USE IN SPACE POWER SYSTEMS**

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ABSTRACT

As technology and spacecraft design advance, the demand for power is ever more prevalent. Photovoltaic cells continue to be the industry standard for space power, and their efficiencies have progressed significantly. However, as spacecraft become more exquisite, modest solar arrays are scaling into massive booms, and time spent in eclipse is becoming increasingly problematic. Tuning photovoltaic cells to the infrared region, coupled with a thermal source, removes the requirement for sunlight and allows the spacecraft to generate electricity even in complete eclipse. Thermophotovoltaic (TPV) cells also allow a spacecraft to operate at great distances from the sun so long as a heat source is present. This thesis explores the possibility of incorporating TPVs coupled with nuclear power systems into spacecraft design for future Earth-bound and interplanetary missions.

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LIST OF ACRONYMS AND ABBREVIATIONS

AC	Alternating Current
AEC	Atomic Energy Commission
AM0	Air Mass Zero
As	Arsenide
Au	Gold
BOL	Beginning of Life
Cd	Cadmium
CIGS	Copper Indium Gallium Selenide
cm	Centimeter
CME	Coronal Mass Ejection
CRU	Combined Rotating Unit
DC	Direct Current
DOD	Department of Defense
DoE	Department of Energy
DUFF	Demonstration Using Flat-top Fusion
EDA	Electronic Design Automation
ELINT	Electronic Signals Intelligence
ENS	Ensign
EOL	End of Life
eV	Electron Volt
FF	Fill Factor
FPSE	Free Piston Stirling Engine
Ga	Gallium
Ge	Germanium
GEO	Geostationary Equatorial Orbit
GLFC	Graphite Lunar Module Fuel Cask
GPHS	General Purpose Heat Source
H	Hydrogen
He	Helium
Hg	Mercury

hr	Hour
In	Indium
InP	Indium Phosphide
IP	Intellectual Property
IR	infrared
I_{sc}	Short Circuit Current
ISPM	International Solar Polar Mission
ISS	International Space Station
JIMO	Jupiter Icy Moons Orbiter
K	Kelvin
K	Potassium
kg	Kilogram
km	kilometer
KRUSTY	Kilopower Reactor Using Stirling Technology
kT	Kiloton
kW	Kilowatt
LEO	Low Earth Orbit
LES	Lincoln Experimental Satellite
Li	Lithium
LT	Lieutenant
m	Meter
MeV	Mega Electron Volt
MHW	Multi-Hundred Watt
MIM	Monolithic Interconnected Modules
MIT	Massachusetts Institute of Technology
MMRTG	Multi-Mission Radioisotope Thermoelectric Generator
mV	millivolt
Na	Sodium
NASA	National Aeronautics and Space Administration
Ni	Nickel
NOLH	Nearly Orthogonal Latin Hypercube
NREL	National Renewable Energy Laboratory

Pb	Lead
$P_{\max}$	Max Power Point
Pu	Plutonium
PV	Photovoltaic
RADAR	Radio Detection and Ranging
RCGA	Real Coded Genetic Algorithm
RTG	Radioisotope Thermoelectric Generator
Sb	Antimonide
Si	Silicon
SKD	Skutterudite
Sn	Tin
SNAP	Systems for Nuclear Auxiliary Power
TCAD	Technology Computer Aided Design
Te	Tellurite
TFE	Thermionic Fuel Element
TPV	Thermophotovoltaic
U	Uranium
USN	United States Navy
V_{oc}	Open Circuit Voltage
VWF	Virtual Wafer Software
W	Watt
Xe	Xenon
ZrH	Zirconium-Hydride
η	Efficiency
λ	Wavelength
μm	Micrometer

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I. INTRODUCTION

Photovoltaic cells have been the backbone of space power since the Sputnik-era, and continue to be the preferred option today. As achieving orbit has become more accessible and spacecraft have grown more sophisticated, they inherently require more power to operate. The cost of solar cell production has decreased over time, and for the most part increased efficiencies have kept pace with the increasing power demand. However, as solar arrays are being scaled up and deployment mechanisms are becoming increasingly more complex to accommodate them, designers are beginning to search for alternatives.

Nuclear power sources, such as radioisotope thermoelectric generators (RTGs), have traditionally been used only when solar arrays are unable to produce the power necessary for the mission, such as interplanetary missions. RTGs are self-contained power sources that use thermocouples to convert thermal energy from radioactive decay directly into electricity and operate completely independent of their environment. While RTGs are an attractive alternative for near-Earth missions since there is already flight heritage established, the thermocouple energy conversion technology is antiquated and their efficiencies are still in single-digit percentages. Space nuclear fission systems have been researched for decades, but with dwindling support and unreliable funding the United States has only flown one. In recent years, many alternative energy conversion methods have greatly benefited from the innovations of the terrestrial renewable energy movement, and their efforts carry forward to space applications. One such conversion method is thermophotovoltaic cells (TPVs), or solar cells that are tuned to accept thermal photons rather than light.

Thermophotovoltaics are not a new concept, in fact, they have been heavily researched since the 1960s. At the time, the Department of Defense was interested in a stealthier power source that was easily portable, and looked heavily into TPVs until the 1970s. The energy crises of that decade further propelled TPV research, and the industry greatly benefitted from the technological advancements made to conventional PV cells [1]. With the advent of modeling software such as Silvaco's Virtual Wafer

Fabrication Software, cells no longer need to be physically manufactured and tested in order to be optimized, further contributing the advancement of these cells. Today, TPVs have been found to achieve efficiencies of up to 39%, making them more than a viable option for integration into RTG or other nuclear systems destined for space [2].

This thesis investigates the use of thermophotovoltaic cells with a nuclear power system and discusses the potential benefits or drawbacks over conventional solar arrays from a size, weight, power, and reliability standpoint. Chapter II reviews historical space nuclear power programs, while Chapter III discusses the types of both dynamic and static power conversion methods that have been designed for space applications. Chapter IV provides background on the mechanics of conventional photovoltaic cells and moves into characteristics unique to thermophotovoltaic cells. Previous work and current state of the art research on TPVs is covered in Chapter V, along with successful optimization techniques that have been used for conventional PV cells. The thesis culminates in Chapter VI, which provides a complete spacecraft design analysis of the engineering tradeoffs associated with using a TPV power conversion system over conventional solar arrays. Chapter VII further discusses the results of this analysis and identifies areas of future work.

II. TYPES OF NUCLEAR DEVICES

In the 1950s, the United States Atomic Energy Commission (AEC) launched a program for the experimental development of nuclear-fueled power systems for both terrestrial and extra-terrestrial use, titled the Systems for Nuclear Auxiliary Power (SNAP) program. Odd-numbered SNAP designators were reserved for radioisotope-fueled devices, whereas even-numbered designations indicate a compact nuclear reactor system [3]. This chapter will provide an overview of the types of nuclear power systems that have been developed and flown on space missions thus far.

A. RADIOISOTOPE THERMOELECTRIC GENERATORS

For missions where solar cells cannot provide sufficient power, the default alternate choice thus far has been radioisotope thermoelectric generators (RTGs). First used in 1961 on the navigation satellite Transit 4A, these devices typically use plutonium-238 as fuel due to its high power density, long half-life, and low shielding requirements [1]. As the fuel decays it produces heat, which is transferred to one side of a thermocouple, and the other side is connected to a radiator exposed to the cold space environment. The temperature differential between the hot and cold “shoes” produces a current, which is then used to power the craft. A more detailed description of thermoelectric conversion can be found in Chapter III. Excess heat produced by the radioactive decay is used to maintain proper temperatures of the spacecraft instruments. Figure 1 shows a diagram of a typical RTG.

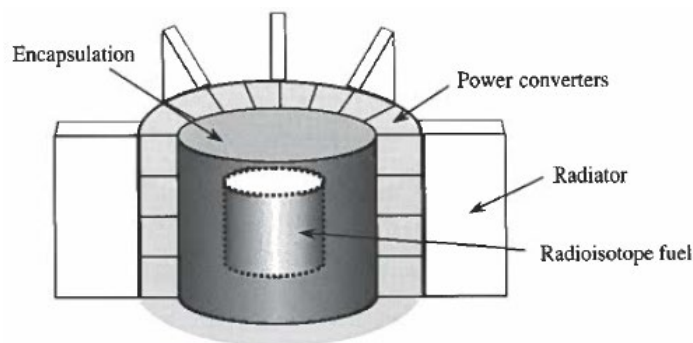


Figure 1. Cutaway View of Typical RTG. Source: [1].

The Transit 4A satellite was the first spacecraft to ever utilize radioisotope power, and Transit 4B followed close behind a few months later. Both satellites featured the SNAP-3B RTG, which used lead-tellurite (PbTe) thermocouples and had a beginning of life (BOL) output power of 2.7 W. In addition to SNAP-3B, Transit 4A/B also featured solar cells on their outer surfaces, and had the unprecedented ability to switch between the power supplies on command [1].

Following Transit 4A/B, Transit-5BN-1, -2, and -3 were launched in 1963 and 1964, though Transit-5BN-3 failed to achieve orbit and burned up upon reentry. These satellites featured the SNAP-9A RTGs, which were very similar to the SNAP-3B but were much larger, and had a BOL output power of 26.8 W [1]. Figure 2 shows a photograph of the SNAP-9A RTG.

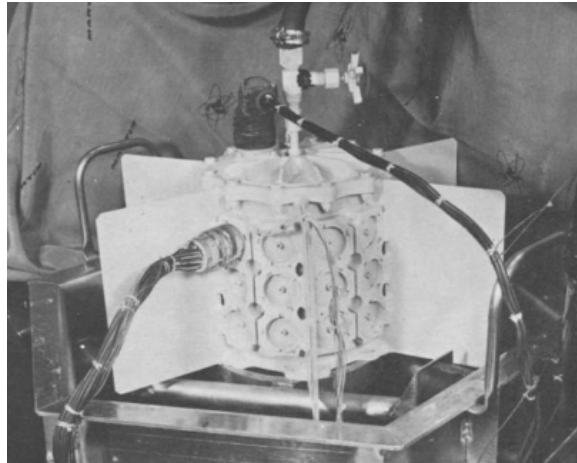


Figure 2. SNAP-9A. Source: [4].

When Transit-5BN-3 disintegrated in the atmosphere upon reentry, the segmented fuel block design allowed the fuel to separate from the main body and burn up in the atmosphere, and about one kilogram of Pu-238 was subsequently dispersed over the entire globe. Though the RTG operated as designed for mission abort conditions, this dispersal prompted a pause of all future nuclear launches and further research was conducted into alternative reentry safety procedures [1], [5]. In 1968, the SNAP-19B RTG was launched aboard Nimbus-B-1. Unfortunately, due to a malfunction in the

rocket's booster guidance system, this satellite also did not make it to orbit. However, thanks to the redesign of the RTG, the fuel elements were able to withstand reentry and ocean impact. This particular SNAP-19B RTG was able to be recovered, and the fuel was reused for the Nimbus-III satellite, which was launched less than a year later [5]. These RTGs could supply between 28 W and 43 W, and featured improved PbTe thermocouples. Though Nimbus-B-1 was ultimately a mission failure, it proved that the new RTG design was indeed safer and more reliable, and the SNAP programs continued to progress. The SNAP-19 RTGs were also used aboard the Pioneer 10 and 11 interplanetary missions in 1972 and 1973, respectively, and on the Viking 1 and 2 Mars landers in 1975. Though largely the same, these RTGs were modified slightly to incorporate a longer life span in the Pioneer missions, and made to withstand the transit to Mars and even more adverse conditions on the Martian surface [1]. Incredibly, the RTGs on the Pioneer missions surpassed their expected lifetime of six years and operated for over 30 years (Pioneer 10) and 22 years (Pioneer 11) until their communications links were lost [6]. Figure 3 shows a cutaway view of the SNAP-19 RTG from the Viking missions.

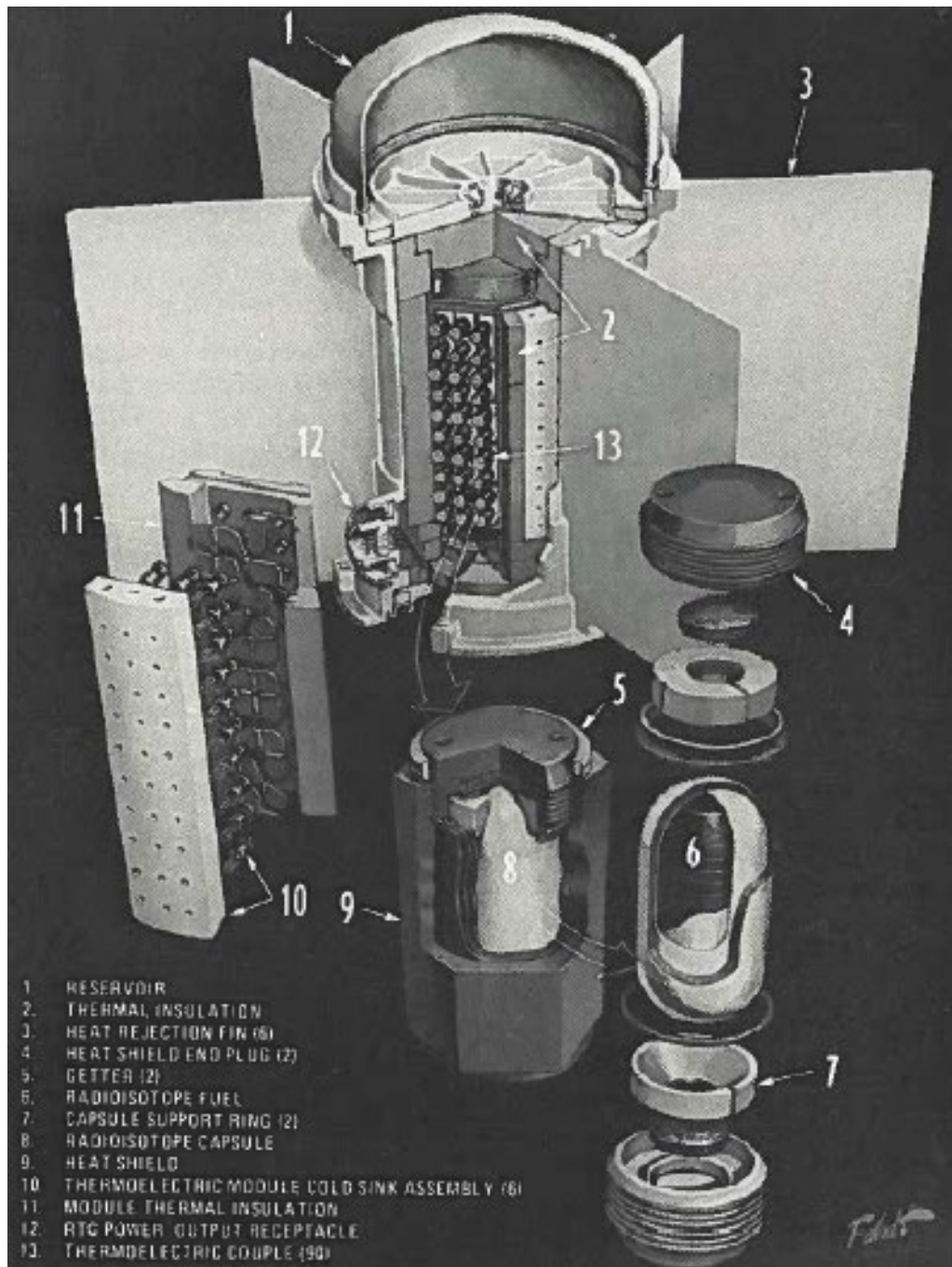


Figure 3. SNAP-19 RTG. Source: [1].

The SNAP-27 RTG was designed specifically for the Apollo missions to support experiments on the lunar surface. At BOL, this RTG could supply up to 63.7 W of power, which was more than double the earlier odd-numbered SNAP variants. Since astronauts

would be physically involved in unloading and setting up the RTGs once they landed on the surface of the moon, the SNAP-27 fuel was kept separately from the generator assembly in a graphite lunar module fuel cask (GLFC) to provide further protection and safety in the event of an unplanned reentry or launch pad mishap. Once safely on the moon and ready to set up experiments, the astronauts would unload the GLFC and place the fuel cell into the generator assembly [7]. These enhanced safety features were tested when Apollo 13's lunar module was jettisoned and reentered the Earth's atmosphere. The GLFC remained intact and today can be found at the bottom of the Tonga Trench, where no trace of leaked Pu-238 has been detected [8]. Figure 4 shows astronaut Alan Bean of Apollo 12 unloading the GLFC, and the generator assembly next to him on the surface of the moon.

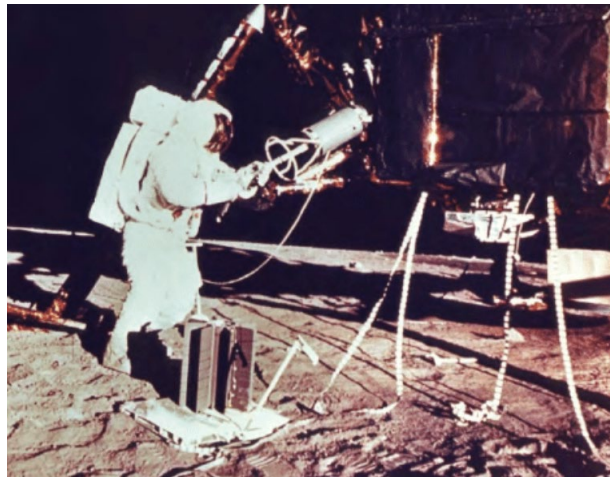


Figure 4. Astronaut Alan Bean with the SNAP-27 RTG. Source: [9].

Following the SNAP program, the Multi-Hundred Watt (MHW) RTG was developed to supply more power for interplanetary missions. It was first launched onboard Lincoln Experimental Satellites (LES) 8 and 9 in 1976, where each satellite carried two RTGs. This was the first time silicon germanium thermocouples were used instead of lead telluride, signifying a major advancement in RTG design and providing a boost in efficiency [10]. At launch, each MHW RTG could produce 150 W and operated around 6.7% efficiency, a significant improvement over the 4–5% efficiencies of the

SNAP RTGs [2]. In 1977 the MHW RTGs were launched aboard Voyager 1 and 2, where each satellite carried 3 RTGs. Originally designed to operate for 5–10 years, Voyager 1 and 2 are miraculously still operating with no major malfunctions over 44 years later [10].

Given the success of the Voyager missions, interest in RTGs for space power increased greatly, and the Department of Energy and NASA began planning and developing an RTG for their next mission: the International Solar Polar Mission (ISPM), later renamed to Ulysses. Originally coined the ISPM-RTG, the Department of Energy attempted to use a selenide-based material for its thermoelectric conversion. This later proved to be problematic, and the RTG did not pass the testing phase. After a major redesign, the new and improved ISPM-RTG was born. It featured the same SiGe thermocouples as the MHW RTG, but swapped the beryllium housing for aluminum alloy and featured a new modular heat source design vice the integral designs of all previous RTGs (except SNAP-27). When other projects, such as the Galileo project, saw the potential benefits of the improved ISPM-RTG, these units were requested for their missions, too, and thus the ISPM-RTG was renamed to the general-purpose heat source (GPHS) RTG [11]. Figure 5 shows a cutaway view of a typical GPHS RTG.

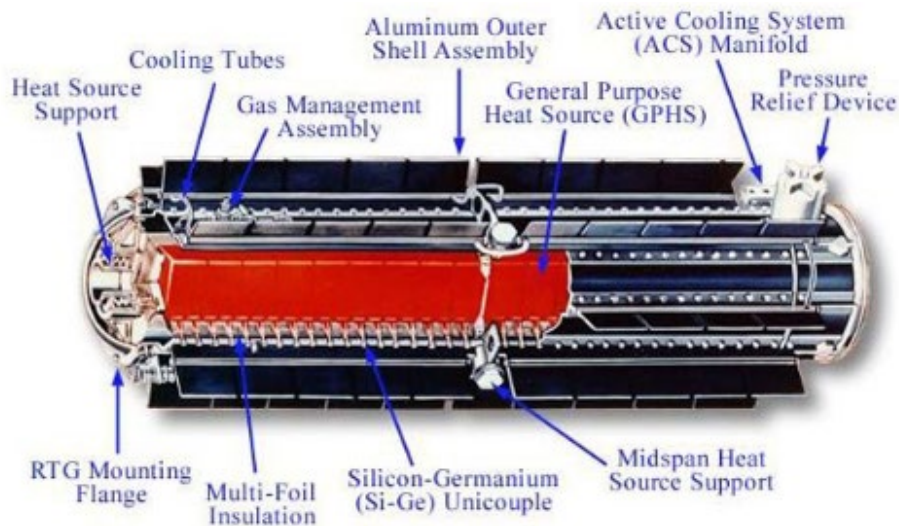


Figure 5. GPHS RTG. Source: [11].

The goal of the modular design was to safely contain the fuel in the event of an unexpected mission abort so that it could be retrieved (and potentially reused) to avoid contamination of the biosphere. Each GPHS RTG could provide approximately 300 W of power at BOL, and operated at around 6.8% efficiency—a further improvement from the MHW RTG [11]. Galileo was launched first in 1989, with two GPHS RTGs onboard, followed by Ulysses in 1990, which had only one. Following the successes of these missions, the GPHS RTG went on to power both the Cassini and New Horizons missions, and continued to be the “go-to” for spacecraft travelling outside the realm of solar cells until 2003, when NASA awarded the contract for the development of the new Multi-Mission RTG (MMRTG) [12]. Figure 6 shows New Horizons’ GPHS RTG, with the craft in the background.



Figure 6. New Horizons and its GPHS RTG. Source: [13].

The MMRTG was developed based on the SNAP-19 design with improved PbSnTe thermocouples, but continued to use the same GPHS fuel modules instead of the integral fuel design in the SNAP-19 [14]. The GPHS RTGs remained the best option for

interplanetary travel missions, however, because their SiGe thermocouples operate best in the vacuum of space and would not be able to operate on the surface of planets with atmospheres, such as Mars [12]. Since the MMRTG was designed with planetary roving in mind, it was made smaller and lighter, but could still supply 114 W of electrical power at BOL [14]. The MMRTG was first launched in 2011 on the Curiosity Mars Rover, and again in 2020 on the Perseverance Rover, both of which continue to operate today [15]. Figure 7 shows an exploded view of the MMRTG.

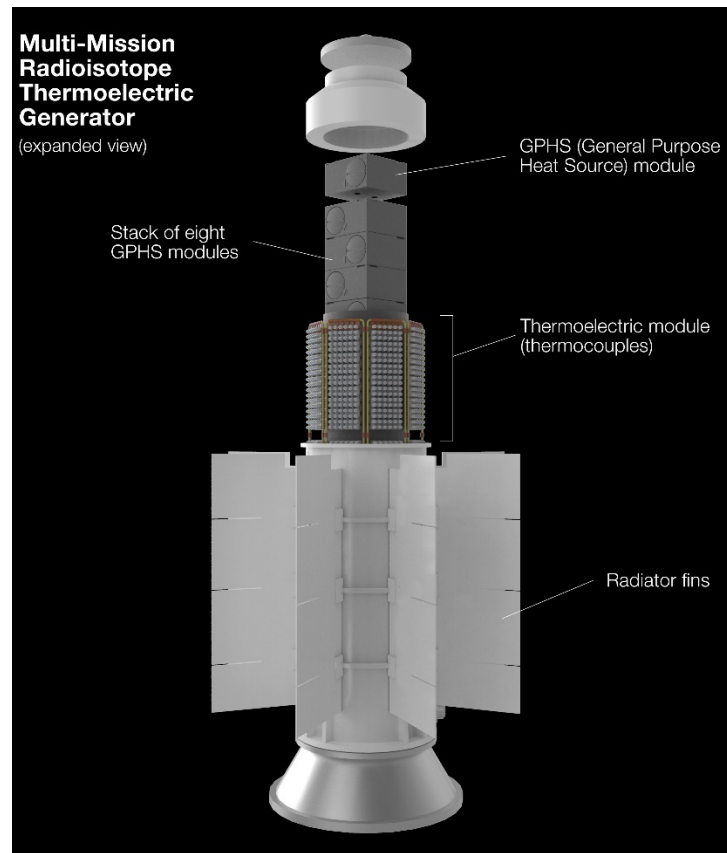


Figure 7. Exploded View of MMRTG. Source: [16].

Today, NASA is investigating upgrading the MMRTG design to incorporate skutterudite (SKD) thermocouples in place of the PbSnTe materials currently in use to boost efficiency and power conversion rates [17]. This upgrade is still in the testing

phase, and more research is required to fully understand the viability of using SKD materials in the vacuum of space as well as in unknown atmospheres.

B. NUCLEAR FISSION SYSTEMS

1. Legacy Systems

The even-numbered SNAP programs began concurrently with the odd-numbered programs in the late 1950s. In total, the United States only developed 3 reactors in the SNAP program: 2, 8, and 10A. Plagued by a myriad of budget cuts and administration changes, SNAP-10A, was the only reactor from the United States that made it to orbit [1]. Table 1 shows an overview of each SNAP reactor, all of which were fueled by highly enriched Uranium-235 [1]. As shown, each even-numbered SNAP program created an experimental reactor and a developmental reactor.

Table 1. Operating Specifications for SNAP Reactors. Source: [1].

	SNAP-2		SNAP-10A		SNAP-8	
	EXPERIMENTAL REACTOR (SER)	DEVELOPMENTAL REACTOR (S2DR)	FLIGHT SYSTEM FS-3	FLIGHT SYSTEM FS-4	EXPERIMENTAL REACTOR (S8ER)	DEVELOPMENTAL REACTOR (S8DR)
Critical Shutdown	September 1959 November 1960	April 1961 December 1962	January 1965 March 1966	April 1965 May 1965	May 1963 April 1965	June 1968 December 1969
Thermal Power (kW _t)	50	57	38	42	600	600-1000
Thermal Energy (MWh)	225	273	383	41	5,100	4,350
Electric Power (watts)	—	—	402	650 BOL 535 EOL	—	—
Electric Energy (MWh)	—	—	4.0	0.6	—	—
Time at Power and Temperature	1,800 hr at 920 K 3,500 hr above 750 K	2,800 hr at 920 K 7,700 hr above 750 K	417 days at approx 800 K	43 days above 785 K	100 days at 600 kW _t at 975 K; 1 yr above 400 kW _t at 975 K	429 hr at 1,000 kW _t ; 1,670 hr at 965 K; 6,680 hr above 810 K

Unlike the SNAP reactors, the SNAP RTGs were designed and built with specific missions in mind and received funding more readily through their respective mission budgets. The reactor programs struggled to gain traction and secure funding since they did not have a defined end goal or high-priority mission assigned to them [18]. Nevertheless, the even-numbered programs persisted, as the reactors could produce significantly more power for longer than their RTG counterparts. Additionally, as new rockets with heavier lift capabilities were developed, the increased shielding required for a reactor system became less of a barrier.

All three SNAP reactor systems were powered by a uranium-zirconium-hydride (U-ZrH) fuel mixture and used a sodium-potassium alloy (NaK) as their primary coolant. For power conversion, SNAP-2 and SNAP-8 utilized a Rankine cycle with mercury as the working fluid, whereas SNAP-10A used thermoelectric power conversion [1]. A description of the Rankine cycle can be found in Chapter III. Figure 8 shows a schematic of the SNAP-2 reactor system. The heat from the reactor was transferred to the Hg Rankine cycle loop via the NaK primary coolant loop through the boiler. The coolant pump was a DC conduction pump powered by a thermoelectric generator, which featured cromel-constantan thermocouples operating between the hot NaK reactor outlet temperature and the cool liquid Hg boiler inlet temperature. In the Rankine cycle loop, the alternator, turbine, bearings, and mercury pump were all connected via a single shaft in the combined rotating unit (CRU). This design allowed for increased efficiency and less complexity, as there was only one moving part. At BOL, SNAP-2 could supply 3 kW of electrical power at a conversion efficiency of 9% [1].

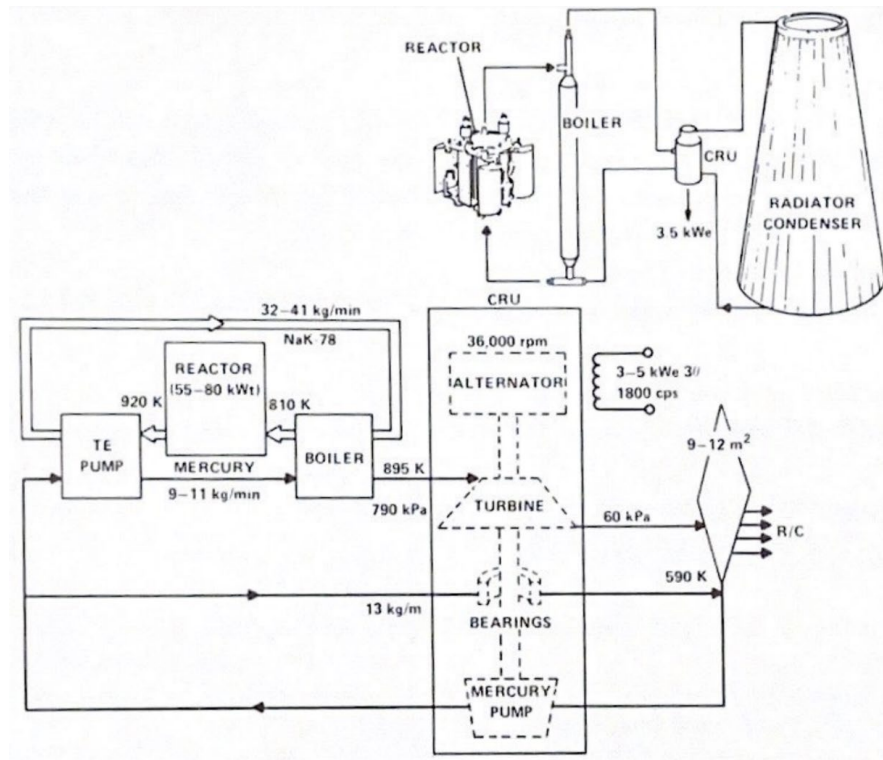


Figure 8. Schematic of SNAP-2 Rankine System. Source: [1].

The SNAP-2 design took almost a decade of research and development, all of which laid the foundation for both the SNAP-8 and SNAP-10A system designs. The SNAP-8 featured four main loops and separated the turbine-alternator shaft from the condenser and fluid pump. With the Hg pump separated from the turbine-alternator assembly, SNAP-8 utilized a self-contained, single-shaft centrifugal pump with an induction motor assembly. It also replaced SNAP-2's thermoelectric NaK pump with a single-shaft feed pump, motor, and recirculating pump assembly. Figure 9 shows the quad-loop design. At BOL it could produce 30 kW of power, with the ability to expand to 60 kW if needed for a particular mission, and its overall power conversion efficiency with the Rankine cycle was 9% [1].

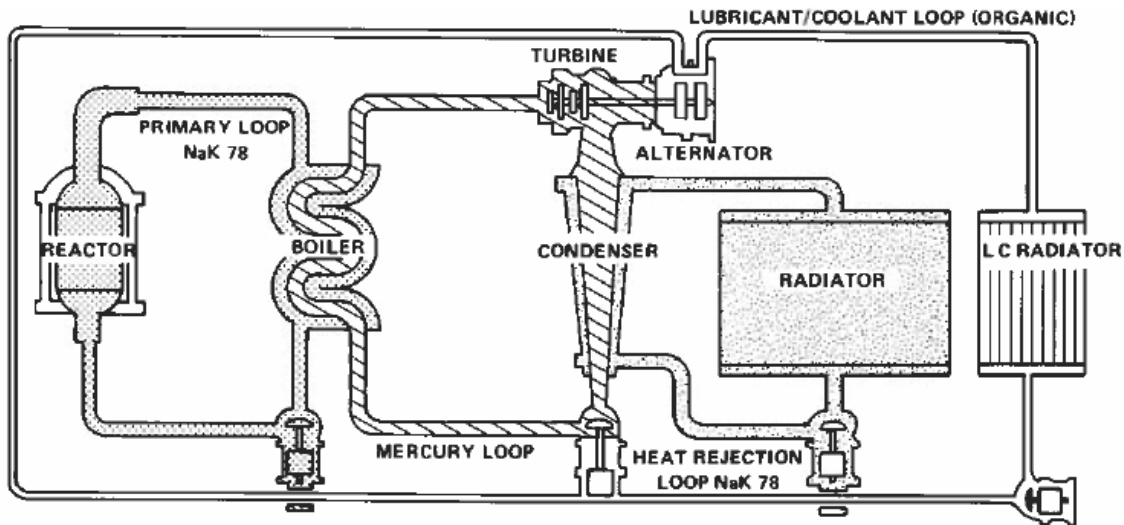


Figure 9. Schematic of SNAP-8 Rankine System. Source: [1].

SNAP-10A used the same reactor and primary coolant pump design as SNAP-2, but utilized a convection cone of SiGe thermocouples for its energy conversion instead of a Rankine cycle. Figure 10 shows a schematic of the SNAP-10A. Due to the radiative properties in the vacuum of space, the total radiating area of the cone was approximately 5.8 m^2 to allow for proper cooling and temperature control. Inside the cone, NaK flowed through 40 evenly spaced tubes, each lined with 72 SiGe thermocouples. The overall efficiency of the energy conversion was 1.43%, but nevertheless SNAP-10A could

produce 533 W of power on orbit. It was launched in 1965 under the SNAPSHOT program, and out of an abundance of caution the reactor was not started until it achieved its final orbit. Upon reactor startup, it operated successfully for just 43 days, when the voltage regulator on the bus of the spacecraft failed and the reactor system shut down. Though the planned year-long mission was cut short, it is still lauded as a great success for how well the reactor performed on orbit and safely responded to a component failure [1].

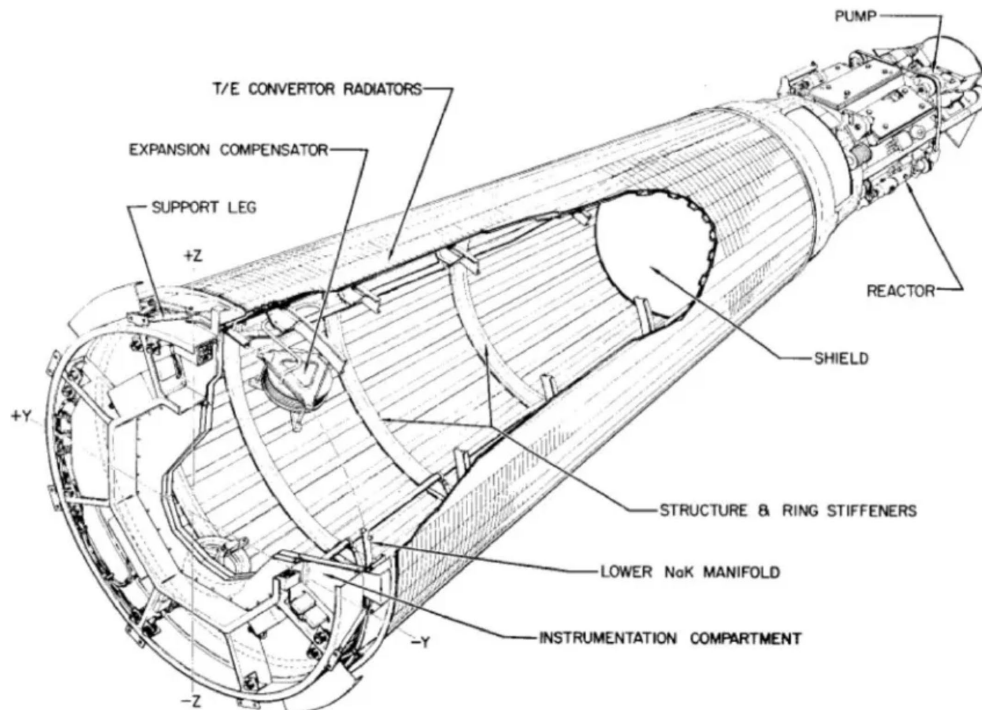


Figure 10. SNAP-10A. Source: [19].

While SNAP-10A remains the only nuclear reactor ever launched by the United States, the Soviet Union launched over 30 between 1965 and 1987. Their primary reactor system was the BES-5, or Buk, which was launched on 34 different satellites used for ocean surveillance, called RORSATs [20]. While the American reactors were all on the thermal neutron spectrum, the Bucs were fast neutron reactors, meaning the neutrons produced by the fission process had much higher energy levels and required more highly enriched fuel. The Buk reactors also used SiGe thermocouples integrated into the fuel

rods inside the core of the reactor, a design called “in-core” power conversion. In this configuration, and given that these neutrons had average energy levels of 1 MeV or higher, the Buk could supply 3 kW of power to its RADAR or ELINT payloads at BOL [21].

The second space nuclear reactor designed by the Soviet Union was a thermal reactor called the TOPAZ-I. This reactor used in-core power conversion as well, however its method was thermionic vice through thermocouples [22]. Figure 11 shows a schematic of the TOPAZ-1 fuel rod with integrated thermionics. A more detailed description of thermionic energy conversion can be found in Chapter III. At BOL the TOPAZ-I could provide 5 kW of power and could operate for up to five years. The Soviet Union successfully launched the TOPAZ-1 reactor on two RORSATs in 1987. These satellites were placed into a higher LEO orbit to avoid uncontrolled reentry and atmospheric burn-up in the event of a malfunction, as was the case with two previous RORSATs in 1978 and 1983. The TOPAZ-1 reactors operated for less than a year before their missions ended [22]. In 2008 and 2014, it was reported that the coolant tubes onboard each of the TOPAZ-1 RORSATs had cracked, and NaK coolant had leaked into space. It is unclear whether this was caused by equipment failure or an impact by space debris [23], [24].

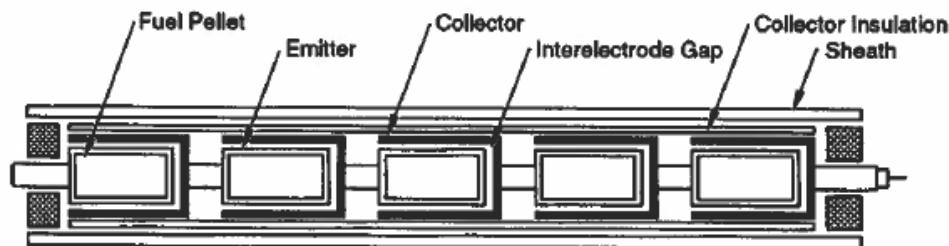


Figure 11. Schematic of the TOPAZ-1 Fuel Rod with Integrated Thermionics.
Source: [25].

In the 1980s, the United States began researching space nuclear power sources more seriously again. Given the recent successes of the Soviet TOPAZ program, the U.S. decided that rather than design, build, test, and fly its own thermionic space nuclear reactor, it would be more cost-effective to purchase some from the Soviet Union. In 1989

leaders from the Soviet space nuclear program attended the Albuquerque Space Nuclear Power Symposium, and a tentative purchase agreement was made. The U.S. agreed to purchase six of the new TOPAZ-II reactors, which featured a single-cell thermionic fuel element (TFE) in place of TOPAZ-I's multi-cell design. This TFE was easier to manufacture and had a higher reliability than the multi-cell TFEs, and at BOL the TOPAZ-II could supply 6 kW of power [26]. Figure 12 shows the TOPAZ-II. The six reactors were delivered to the United States, two in 1992 and four in 1994, and were extensively ground tested. Unfortunately, due to budget cuts and a change in the national nuclear posture, the TOPAZ-IIs were never flight tested, and were eventually returned to Russia [26].

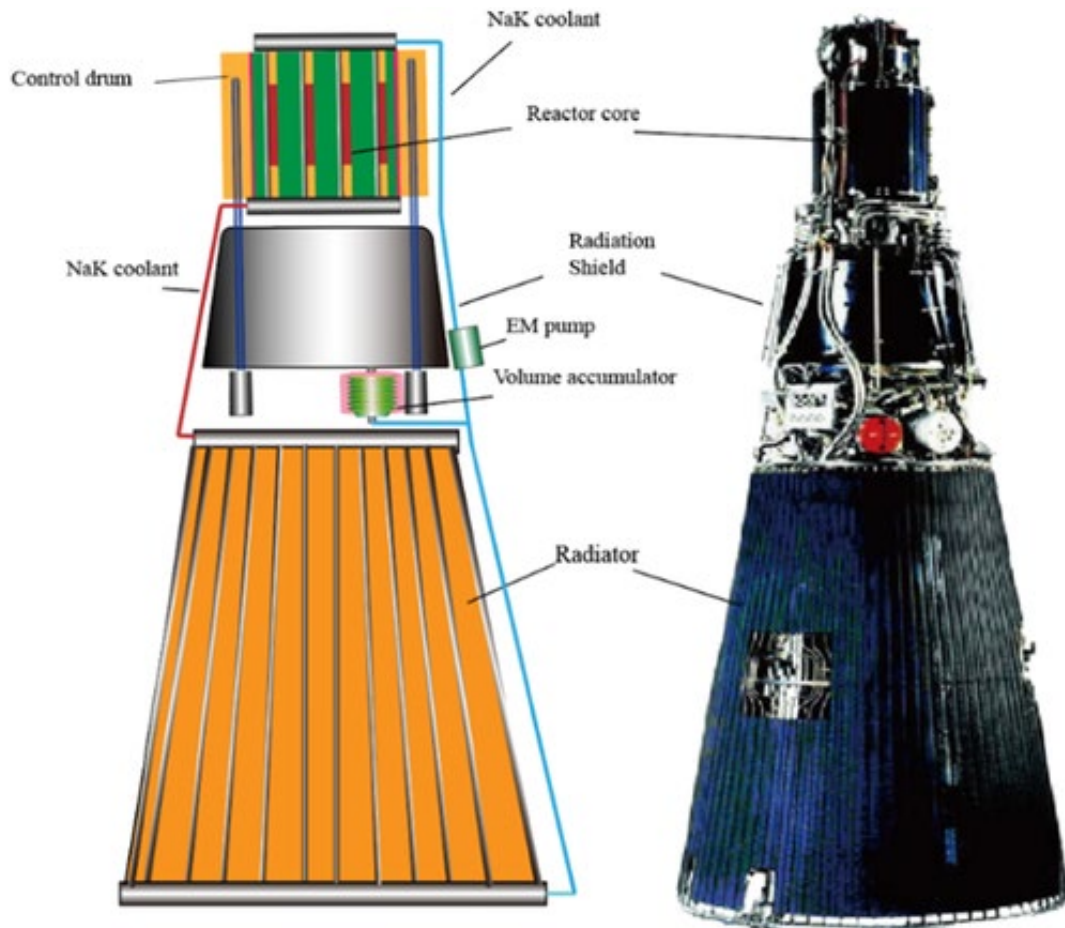


Figure 12. TOPAZ-II Reactor System. Source: [27].

2. Current Systems

After a long lull in nuclear space development, Congress approved funding for NASA's Prometheus Project in 2003. Its objective was to develop a nuclear reactor capable of supporting power and electric propulsion demands for an interplanetary science mission to three of Jupiter's moons, later coined the Jupiter Icy Moons Orbiter (JIMO) mission [28]. A conceptual design of the JIMO craft can be found in Figure 13. The design featured a 200 kW fast fission reactor at the nose of the craft in conjunction with a Brayton cycle power conversion system. A detailed description of Brayton cycle power conversion can be found in Chapter III. In the middle was the main boom assembly, which consisted of several large radiator panels, and at the tail of the craft resided the science experiments, antennas, and electric thrusters. Though the program lost funding in 2005, its research on the use of nuclear power to fuel propulsion as well as power requirements paved the way for future bimodal reactor systems [29].

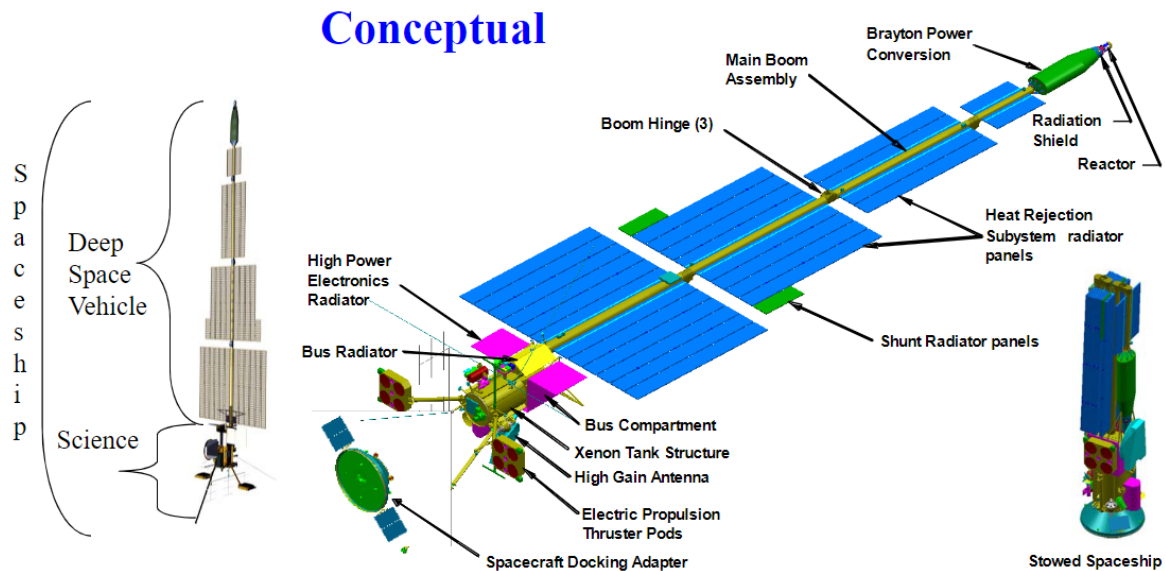


Figure 13. Conceptual Design of JIMO Spacecraft. Source: [28].

In 2012, the Demonstration Using Flattop Fusion (DUFF) test was conducted to prove that heat from a sample nuclear source, in this case a reactor assembly named Flattop, could be transferred via heat pipes and used to power Stirling engines and

generate useful electrical power [30]. A detailed description of Stirling engine power conversion can be found in Chapter III. This proof-of-concept test was completed less than 6 months after it was proposed with a modest budget utilizing only existing equipment and rudimentary design principles. This test not only successfully proved that Stirling engines could be used in conjunction with a nuclear power source, but also that testing for a new fission reactor system did not have to be done all at once, and that low-budget incremental tests could provide invaluable data to be used in further research [31].

Following the DUFF test, interest around Stirling engine systems grew, and in 2015 NASA began the Kilopower Program. This program aimed to develop scalable reactor systems for use in space applications, capable of production between 1 and 10 kW of electricity for ten years. From this program stemmed the Kilopower Reactor Using Stirling Technology (KRUSTY) test reactor. It featured a solid U-235 core design, vice individual fuel rods as in previous reactor designs. Liquid sodium flowed around the core through heat pipes and transferred the heat to Stirling engines, creating up to 5 kW of electricity [31]. Figure 14 shows the Kilopower flight concept and the KRUSTY configuration. The KRUSTY reactor system was successfully ground tested for five months beginning in November 2017. Throughout the testing various failure modes and mission conditions were simulated, to include a full power run and a scram test, and the KRUSTY system responded as designed each time [32].

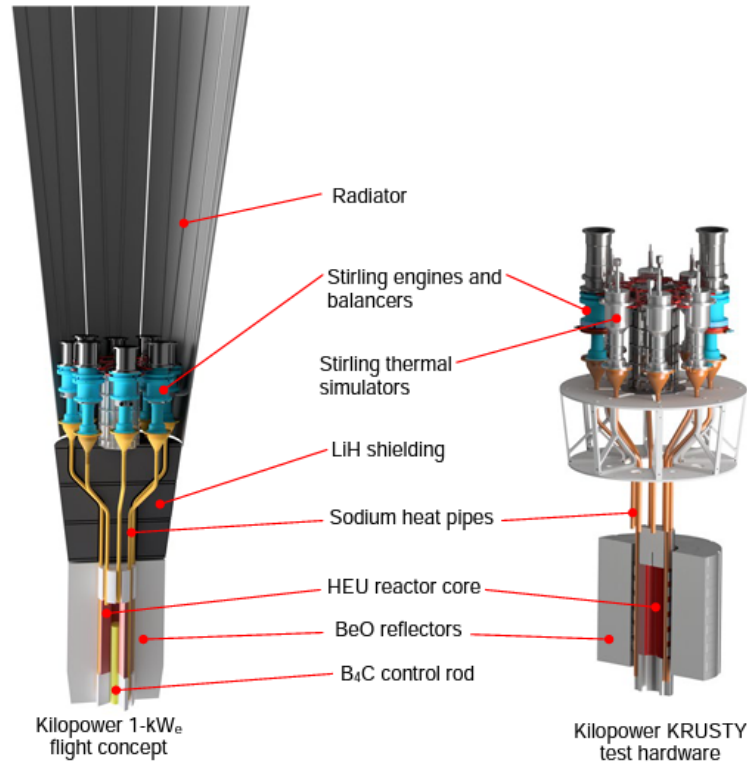


Figure 14. Kilopower and KRUSTY Configurations. Source: [33].

The Kilopower Program was discontinued in 2018, but its success lives on through NASA's Fission Surface Power Project, which seeks to explore energy solutions for use on the lunar surface [34]. It is projected that four KRUSTY systems could feasibly create enough power for a small colony on the moon [33]. Figure 15 shows a concept design for lunar power. In this image the reactor is buried underground, and the modules near the surface are the Stirling engines. The disk on top is the radiator, which would be folded and stowed much like the Kilopower design in Figure 14 for the transit to the moon. This project is ongoing, with hopes to launch a demonstration model to the moon by 2031 [34].

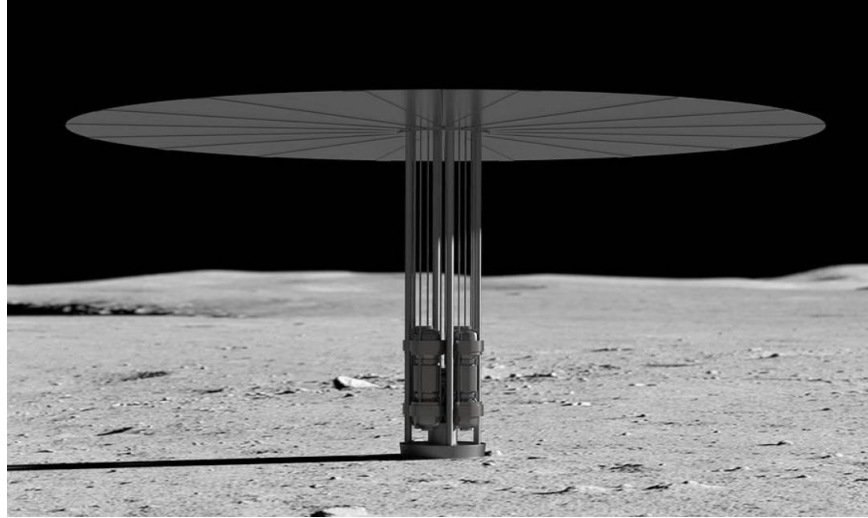


Figure 15. Concept Design of a Fission Surface Power System. Source: [34].

This chapter provided context on historical nuclear power systems for space applications and the current state of ongoing research. While fission systems are capable of producing more power than RTG systems, the simplistic design and proven flight heritage of RTGs makes them the natural choice for spacecraft. That said, fission systems could be a great option to provide stable power for outposts on other planets. Chapter III will provide an overview of the different types of thermal to electric power conversion methods and how their efficiencies compare.

III. TYPES OF THERMAL-ELECTRIC CONVERSION

In general, thermal to electric power conversion can be separated into two general categories: dynamic and static. Dynamic power conversion involves a mechanism that physically moves in order to produce electricity, whereas static convertors remain stationary and rely on the movement of ions, for example. This chapter will serve as a brief overview of the main types of power conversion methods in each of these categories and how they have been, or could be, employed in space.

A. DYNAMIC

1. Micro-Brayton Generator

Brayton cycles are most commonly used in gas turbine or jet engines due to their high torque or thrust qualities at the turbine outlet and favorable efficiencies. These engines consist of a compressor and a turbine coupled together on a single shaft that are either connected to a generator assembly to produce electricity, in the case of a gas turbine generator, or a nozzle to produce thrust, in the case of a jet engine. In the vacuum of space, an ideal closed Brayton cycle is used, typically with a Helium-Xenon working fluid [1]. Figure 16 shows a schematic of Brayton cycle designed for space applications. In this model the heat source depicted is an RTG, but fission systems can be used as well. The recuperator is not a mandatory component, but it increases efficiency by using heat from the turbine outlet to preheat the gas prior to interaction with the heat source [1]. Overall, the system depicted in Figure 16 has an efficiency of 26.8% [35]. Currently there are no spacecraft using a Brayton cycle for power generation, however, the Hubble Space Telescope is using this cycle in reverse as a cryocooler for one of its near infrared cameras, and it was proposed for used on the discontinued JIMO mission mentioned in Chapter II [36]. In general, Brayton cycles are an attractive solution because they do not require the use of liquid metal working fluids or active vibration control systems since their motion is only rotational.

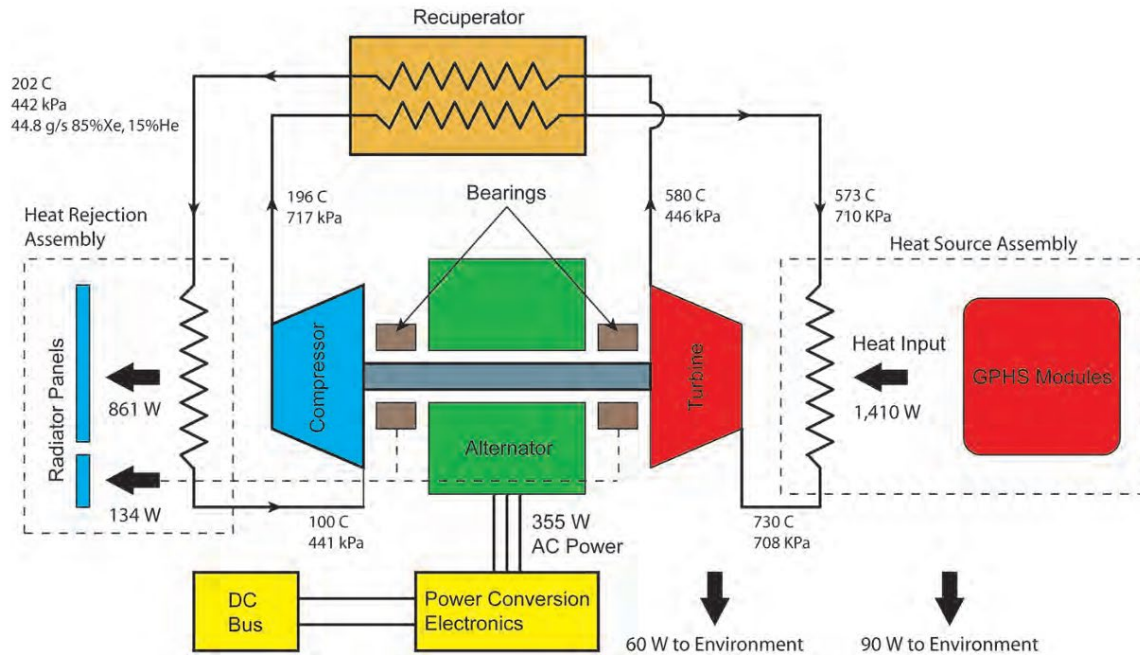


Figure 16. Schematic of a Brayton Generator for Space Applications.
Source: [35].

2. Stirling Engine

The Stirling engine has been considered for space applications since the 1970s, and continues today in NASA's Fission Surface Power Project. The type of Stirling engine used in space is called a Free Piston Stirling engine (FPSE). An example of this type of Stirling engine can be found in Figure 17. The FPSE operates purely off of the temperature differential between the top and bottom plates, passing a plunger and a piston back and forth in a perpetual motion so long as a heat source (the reactor or radioisotope) and a cold source (outer space) are present. These engines are very efficient, typically around 25%, and can continue to operate seemingly infinitely [1]. In 2020, one of NASA's RTG-powered Stirling engine prototypes reached its 14th year of continuous, maintenance-free operation [37]. Due to its non-contact seals and bearings, the internal mechanisms have no signs of wear, and the system continues to operate today.

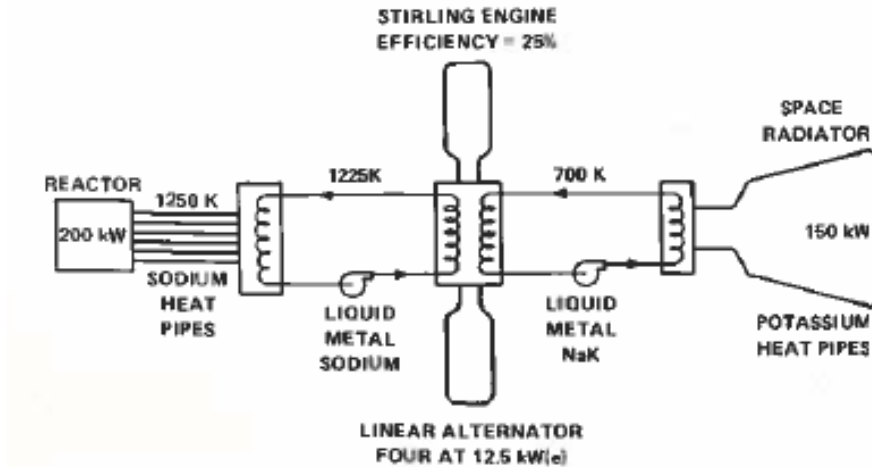


Figure 17. Schematic of a Stirling Engine for Use in Space. Source: [1].

3. Rankine Cycle

A Rankine cycle for space applications consists of a liquid metal working fluid, a heat source, a power turbine which turns an alternator, a condenser, and a fluid pump. SNAP-2 and SNAP-8, as described in Chapter II, both featured a Rankine cycle with a mercury working fluid, though many other potassium or sodium-potassium fluids have been proposed. Figure 18 shows a schematic for a basic Rankine cycle for space power. One thing to note about this type of power cycle is if the turbine operates in or near the condensing region of the working fluid, there is a greater risk of premature turbine blade wear due to condensate droplets. In terrestrial applications steam is used as the working fluid and can simply be heated to a higher temperature to mitigate this. When using a liquid metal working fluid, superheating may not be possible or practical due to physical material properties of the system components. One way to moderate the condensation is to utilize a condensate removal mechanism, but further zero-gravity testing is required [38]. SNAP-2 and SNAP-8 both had overall power conversion efficiency of 9%, but more recent testing indicates efficiencies of up to 19% can be achieved [1], [38].

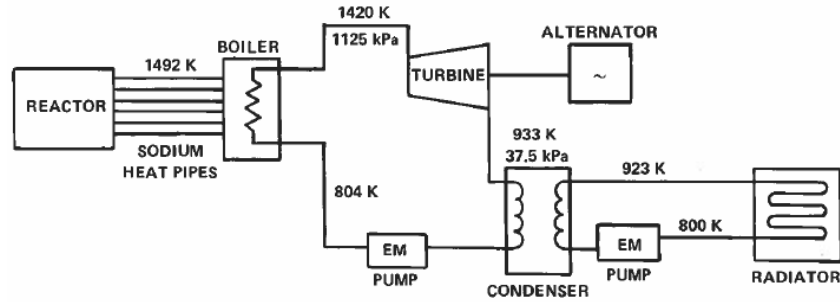


Figure 18. Schematic of a Rankine Cycle for Use in Space. Source: [1].

B. STATIC

1. Thermionic

Thermionics involves inducing electron emission from an emitter surface placed near a heat source, such as a reactor, and retrieving those electrons on a collector material where heat is then released. A current flow is created by connecting the emitter and collector plates. Essentially, thermionics can be thought of as a heat engine that uses electrons as its working fluid [1]. Figure 19 shows a simplified diagram of thermionic power conversion. These devices can either be placed outside of the reactor core to collect ambient heat, or inside the core to collect heat more directly as in the Soviet TOPAZ designs described in Chapter II. In-core TFEs are able to achieve higher conversion efficiencies, but the integrated fuel rods can be complex to manufacture. Overall, thermionic converters are generally between 15–20% efficient, but this varies greatly depending on the configuration and materials used [38].

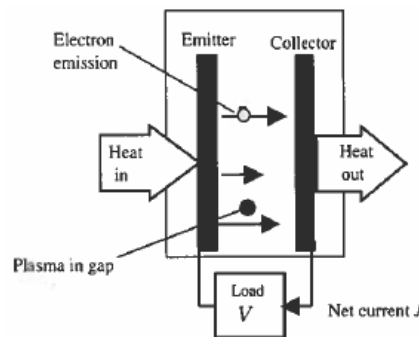


Figure 19. Simplified Diagram of Thermionic Energy Conversion.
Source: [25].

2. Thermoelectric Conversion

The basis of thermoelectric conversion, also known as thermocouples, is the Seebeck effect. This phenomena occurs when two dissimilar conductors are connected at two junctions of different temperatures and produce a resulting electromotive force [1]. Figure 20 shows a schematic of a simple thermocouple circuit. As discussed in Chapter II, this type of power conversion technology has been the powerhouse of space nuclear power since its inception in the 1950s. Their simple design and proven flight heritage on RTG and fission systems alike make them a popular choice, however, their conversion efficiency still hovers around 5–8% due to material limits of the semiconductors [38].

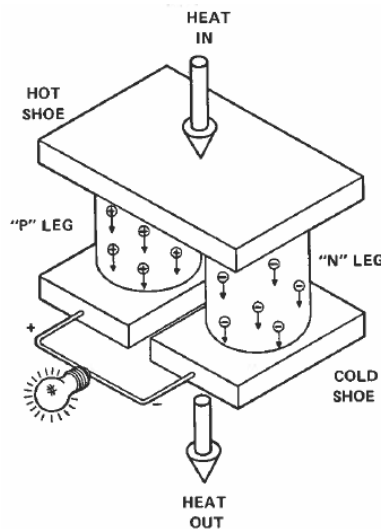


Figure 20. Simplified Diagram of a Thermoelectric Converter. Source: [1].

3. Thermophotovoltaics

Photovoltaic (PV) cells, or solar cells, consist of several layers of semiconductor materials that are artificially “doped.” This means they are injected with atoms that have either one more or one less electron in their outer shell than the base material, resulting in an overall positive or negative charge. When placed together, the electrons in the N-region join with the holes in the P-region. As they combine, excess electrons and holes are driven out of the junction, and these positive or negative charges begin to build up on either side of the region. This region is often called the depletion region or the P-N

junction. The buildup of opposing charges results in an electromagnetic field. When photons hit the P-N junction, electrons break free leaving behind holes, both of which are then pushed out of the region by the electromagnetic field. When the layers are connected via metal contacts at the top and bottom of the cell, electrons and holes are able to move between the layers and recombine, producing a current [39]. Figure 21 shows a diagram of a typical solar cell.

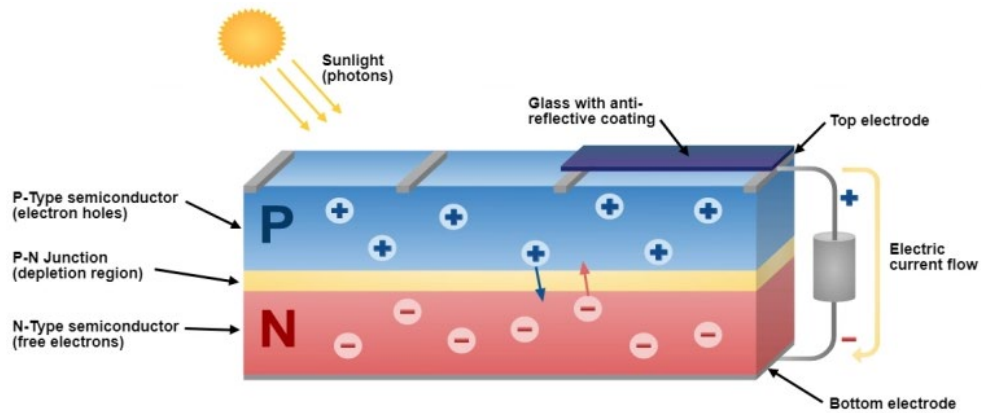


Figure 21. Diagram of a Typical Solar Cell. Source: [40].

Thermophotovoltaic (TPV) cells work in a similar way, however the photons used are not in the visible light spectrum, but rather the infrared spectrum. Thus, a heat source, such as a nuclear source, can be used instead of the sun. This enables the cell to produce electricity continuously, even in times of eclipse, which is a huge advantage over conventional PVs. While low efficiencies and high manufacturing costs have historically made TPVs a less than desirable solution, recent developments in materials science and PV manufacturing are bringing them back into focus, with some cells approaching up to 40% efficiency, which is on par with current conventional PV efficiencies [2], [41]. Chapter IV provides a more in depth look at TPVs and how these types of cells can be used in future space applications.

IV. THERMOPHOTOVOLTAIC CELLS

Conventional photovoltaic cells have provided safe, reliable energy to spacecraft for decades, and have become the industry standard for space power. As described in Chapter III, these cells only produce power in sunlight, requiring either energy storage devices to continue powering payloads while in eclipse or complete system shutdown in those regions of orbit. In some cases, the regions in eclipse may be of high interest to the payload, but power constraints prevent it from operating. A possible solution to this and many other complications arising as the demand for space power increases is to use TPVs. Tuning PV cells to the IR region completely removes the requirement for sunlight, allowing the spacecraft to generate uninterrupted, stable power even in complete eclipse or at great distances from the sun so long as a heat source is present. This chapter breaks down the mechanics of PV cells and describes the components of a typical TPV power system.

A. BACKGROUND ON PHOTOVOLTAICS

As discussed in Chapter III, TPVs operate just as conventional PVs do, just at different wavelengths of light. Solar cells for space applications are designed to respond to the Air Mass Zero (AM0) spectrum, shown in Figure 22, which corresponds to the spectrum from a 5800 K blackbody emitter (the sun) at the edge of Earth's atmosphere. The photon wavelength (λ) and energy (E) are directly related through Equation 1, where h is Plank's constant and c is the speed of light.

$$\lambda = \frac{hc}{E} \quad (1)$$

In this Equation E can be replaced with the bandgap, which is the energy required by the photon to break an electron free in a given material. If the wavelength of light is known, this Equation can be used to solve for the bandgap of the ideal material to be used. It is important that the photons have enough energy to break electrons free in the material, but

not too much since any excess energy will be given off as heat within the cell, which degrades its performance.

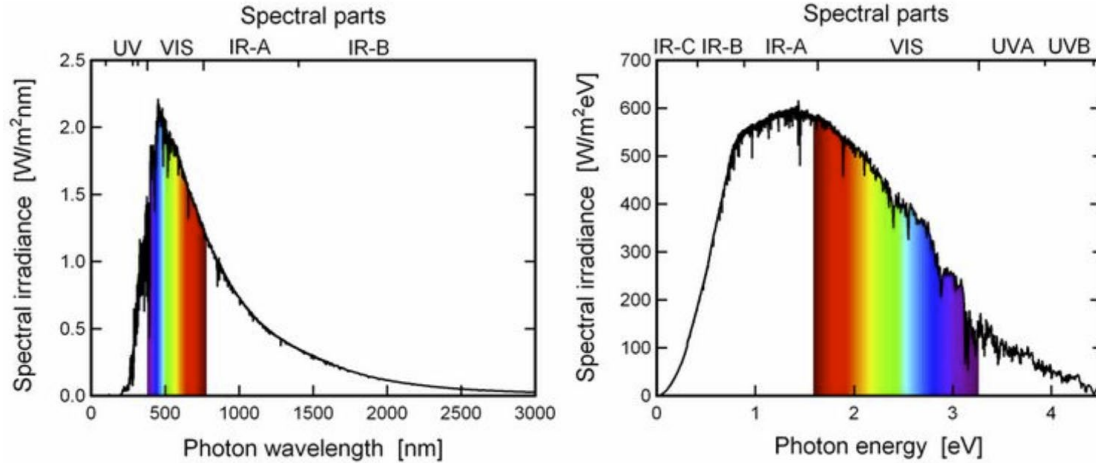


Figure 22. AM0 Spectrum. Source: [42].

PVs can vary almost all of their parameters to improve their performance, such as doping level, materials used, layer thickness, manufacturing methods, special coatings or cover glass, how the metal contacts are integrated, and even the texture of the cell. In order to be able to directly compare them, there are a few key performance parameters that are measured. The first is the short-circuit current (I_{sc}). This is the maximum current the cell is capable of producing, and is measured by shorting the two leads of the cell together. The next key performance parameter is the open-circuit voltage (V_{oc}), which is the maximum voltage the cell can produce. To measure this, the cell must be exposed to light, but the leads remain disconnected from the circuit. The photons will continue to break the electrons free, but they will be unable to recombine with holes since the circuit is not complete. Eventually the voltage across the P-N junction will become so strong that it will restrict any further movement of charged particles. This voltage is the V_{oc} [43].

The I_{sc} and the V_{oc} are typically plotted against each other to form the I-V curve. The maximum power point (P_{max}) can be found at the “knee” of this curve, and is the

desired operating point for any PV cell. The last key performance parameter is the fill factor (FF), which can be found through Equation 2 [43].

$$FF = \frac{P_{\max}}{I_{sc} V_{oc}} \quad (2)$$

A higher FF is desirable since it yields a higher $P_{\max}$. Figure 23 is a typical I-V curve showing the key performance parameters.

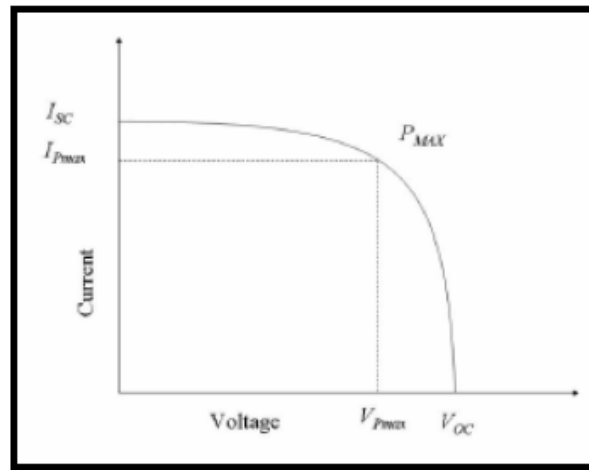


Figure 23. Typical I-V Curve. Source: [44].

Along with the key performance parameters mentioned above, overall cell efficiency is an important metric as well. On principle, efficiency is a measure of power output versus power input expressed as a percentage. For solar cells, Equation 3 can be used.

$$\eta = \frac{P_{\max}}{P_{in}} * 100\% \quad (3)$$

In this equation, P_{in} is the incoming solar irradiance, which varies based on Earth's position in orbit around the sun. For simplicity an average value of 1350 W/m^2 can be used, which is often referred to as the solar constant [43]. Figure 24 shows the progression of solar cell efficiency through the years. This plot is courtesy of the National

Renewable Energy Laboratory (NREL), in Golden, Colorado. Typical solar cells currently used in space are around 30% efficient [45]. It is important to note that the cells certified and tested by NREL in Figure 24 were operating at a reference temperature of 25°C. On average all types of PV cells experience a degradation of output voltage as temperatures rise, typically around 2 mV per degree Centigrade, and operating temperatures in space can be much higher than 25°C [42]. Thus, in order to accurately predict cell efficiency and true output power of an array, the operating temperature must be accounted for [43].

Best Research-Cell Efficiencies

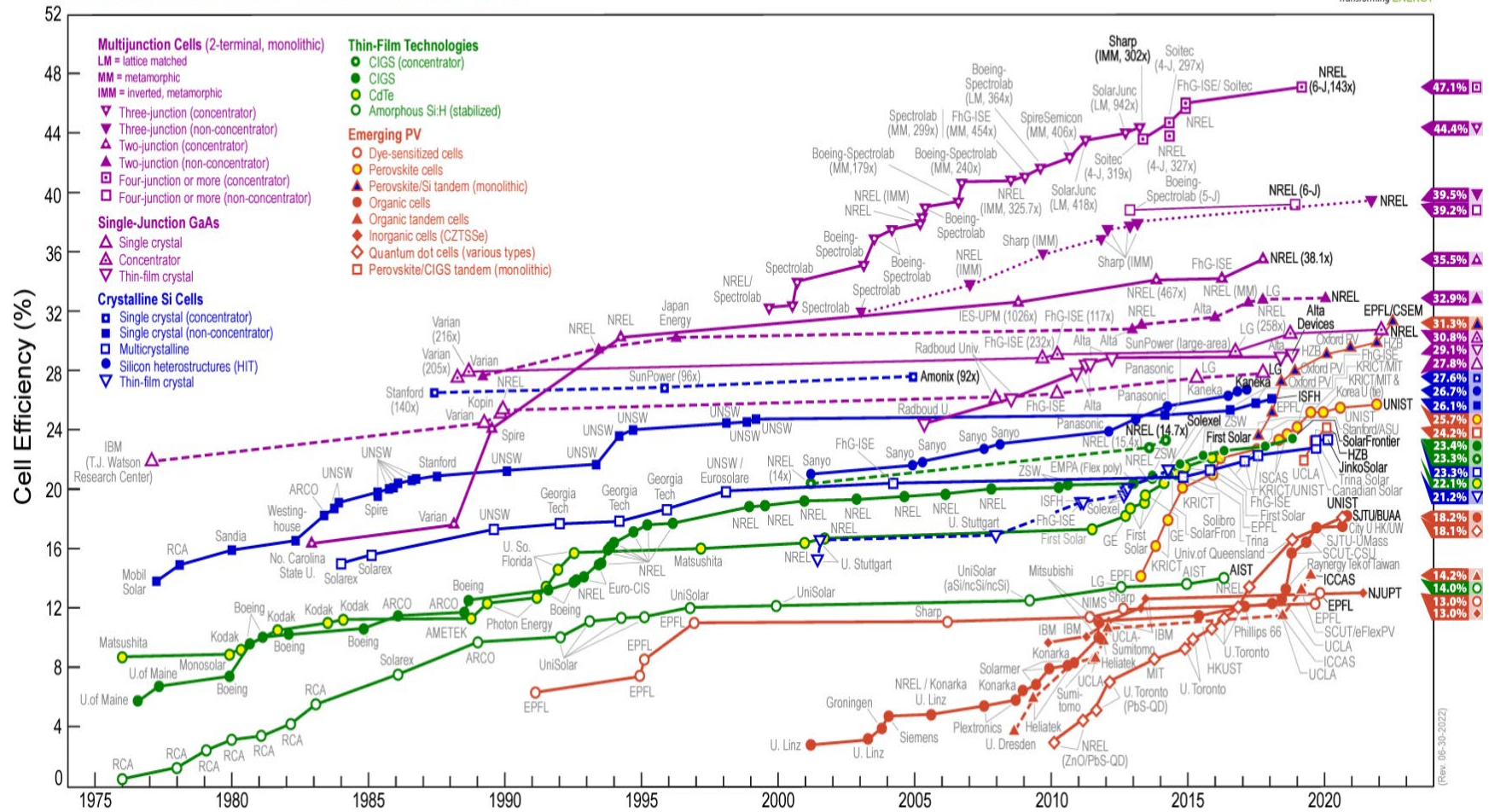


Figure 24. Solar Cell Efficiency Through the Years. Source: [41].

The cells identified as “concentrator” cells in the legend consist of a lens or mirror that amplifies the incoming solar energy and projects it onto the solar cell. The cells marked as “thin film” are alternative, flexible cells that are much thinner than conventional PVs. They can range between a few nanometers to around ten micrometers thick, whereas a conventional solar cell has a thickness around 200 micrometers [46]. This thesis focuses primarily on conventional, non-concentrator type cells.

1. Single-Junction Cells

Single-junction cells are the most basic form of PV cell, as described in Chapter III. They are the easiest to manufacture and are the least complex. Currently, the most efficient single-junction cell is 26.1% for Silicon and 39.5% for Gallium Arsenide under ideal conditions [41]. Figure 25 shows the conversion capability of a single Silicon cell under the AM0 spectrum. Since the cell is comprised only of Silicon, which has a bandgap of 1.1eV, only the wavelengths of light 1.1 micrometers and shorter possess enough energy to free an electron (using Equation 1). This means that the remaining portions of the spectrum go unused and introduce heat into the cell. In an effort to avoid this loss and increase efficiency, multijunction cells were created.

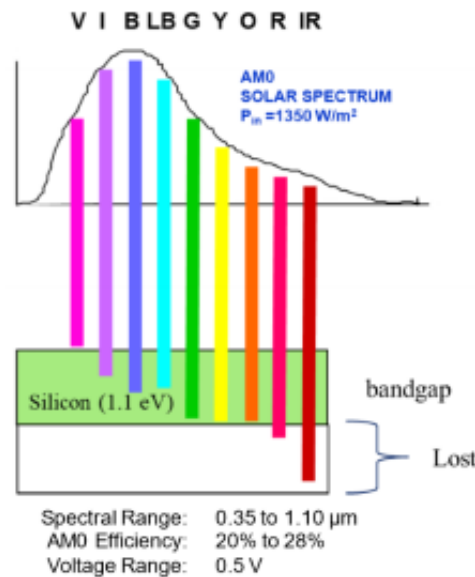


Figure 25. Single-Junction Cell Usable Energy Conversion. Source: [47].

2. Multijunction Cells

Multijunction cells are designed such that solar cells of different materials and bandgaps are stacked on top of each other in order to absorb more portions of the solar spectrum. Figure 26 shows an example of a multijunction solar cell for space applications. The cells are layered such that the material with the highest bandgap is on top, and the lowest bandgap material is on the bottom. This is so that the highest energy photons are absorbed first, before they lose any energy, and the remaining weaker photons pass through to the next layer, and so on. The tunnel junction sections shown are highly doped layers that provide passage for electrons and minimize voltage losses between the cells [48]. Multijunction cells are capable of achieving efficiencies much higher than single-junction simply because a larger portion of the spectrum is able to be absorbed. It is important to note, however, that while single-junction solar cells experience approximately a 2 mV/°C degradation with an increase in temperature, the effect is cumulative in multijunction cells. For example, the cell in Figure 26 would experience approximately a 6 mV/°C loss as temperature increases [43].

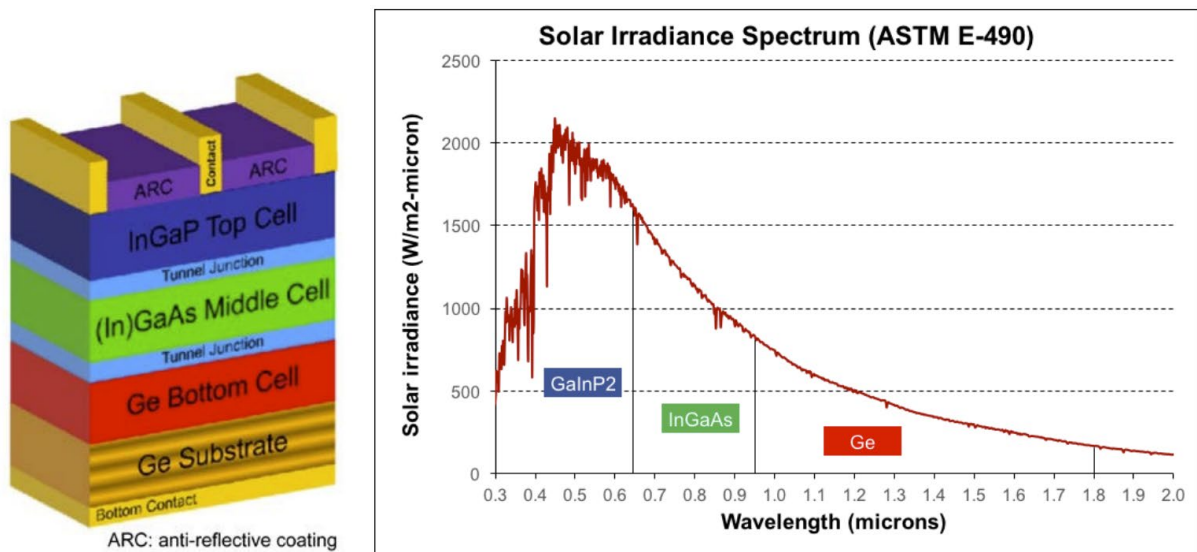


Figure 26. Example of a Multijunction Solar Cell for Space. Source: [45].

B. THERMOPHOTOVOLTAICS

Solar cells must rely solely on energy from the sun, whereas TPVs can utilize any type of radiant heat source. Additionally, since TPVs are focused on the IR regions of the electromagnetic spectrum, their bandgaps are typically much lower than those of conventional PV cells. This section serves to describe the components of a typical TPV power system and important considerations for TPV cell material selection. The key components, shown in Figure 27, are a radiating heat source, a spectral control device to shape the incoming energy spectrum, the TPV cell, and a cold reservoir such as a space radiator.

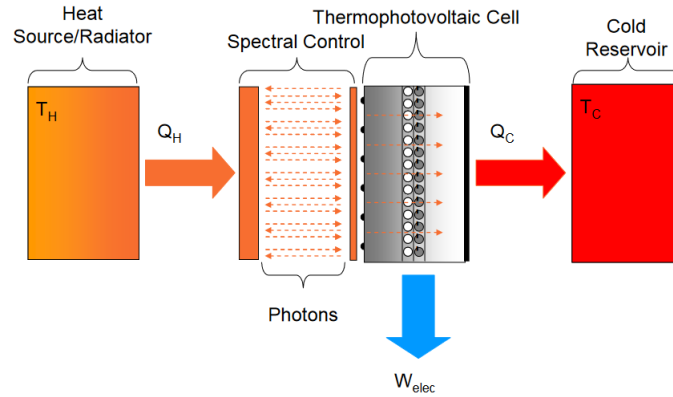


Figure 27. Schematic of a Typical TPV System. Source: [48].

1. Heat Source

In a TPV system, power generation begins with a heat source. This is a very important factor for overall system efficiency and an important consideration for system design. Planck's law of distribution states that the magnitude of the spectral radiance of a body increases with temperature [49]. It is known that the radiated energy from the sun, closely modeled by a 5800 K blackbody, is much greater than any lower temperature heat source. However, given that the sun is a considerable distance away from the Earth, the full magnitude of the spectral radiance is not received by conventional PV cells in orbit. TPVs, on the other hand, can be placed in close proximity to their emitters, allowing them to receive the full magnitude of the energy they radiate.

The spectral radiance at any temperature can be found using Equation 4, where B is the output power ($\text{W}/\text{cm}^2\text{-}\mu\text{m}$), T is the temperature (K), λ is the wavelength (μm), k is the Boltzmann constant, h is the Planck constant, and c is the speed of light.

$$B(\lambda, T) = \frac{2hc^2}{\lambda^5} \left[\exp\left(\frac{hc}{k\lambda T}\right) - 1 \right]^{-1} \quad (4)$$

Given the temperature for any heat source, this Equation will provide the spectrum of radiation across all wavelengths. Figure 28 shows the spectral radiance of typical RTG and fission sources overlaid with the bandgaps of common PV and TPV materials. More on these materials will be covered later on in this section.

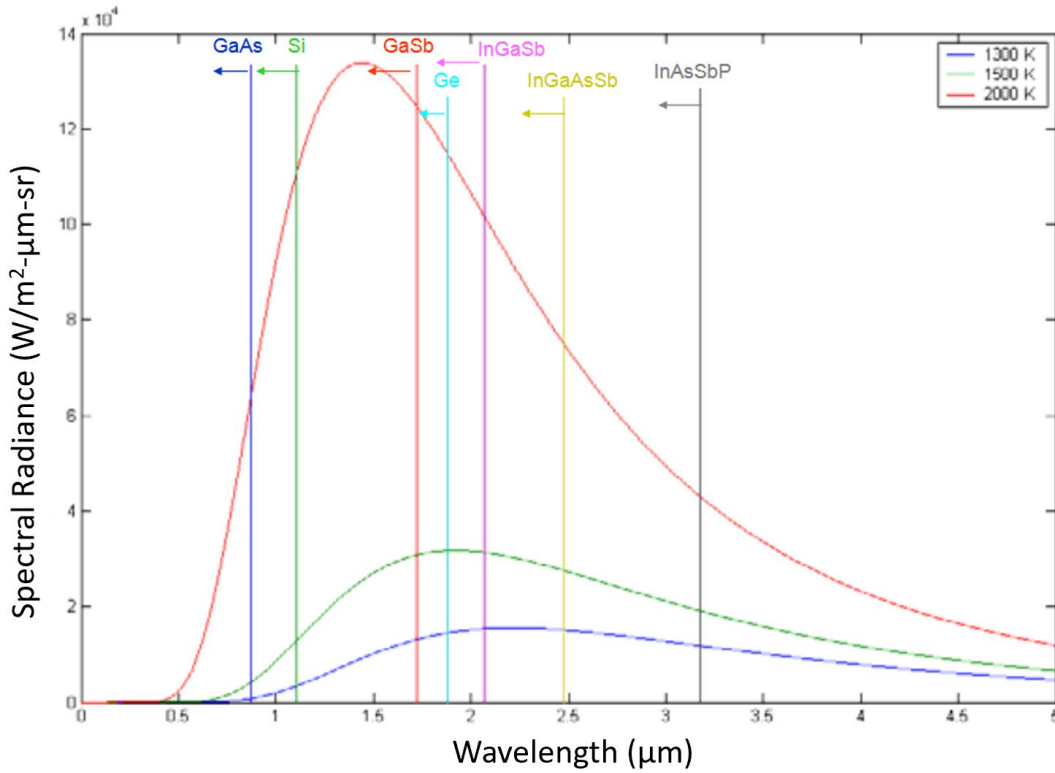


Figure 28. Spectral Radiance and Bandgaps of Common TPV Materials.
Source: [48].

2. Spectral Control

The spectral control portion of the TPV system is not a mandatory component, however, these devices greatly improve the cell efficiencies and therefore are typically employed either individually or in conjunction with one another. Generally, spectral control consists of a selective emitter, a filter, and/or a back surface reflector. Figure 29 shows the locations and basic functions of these devices.

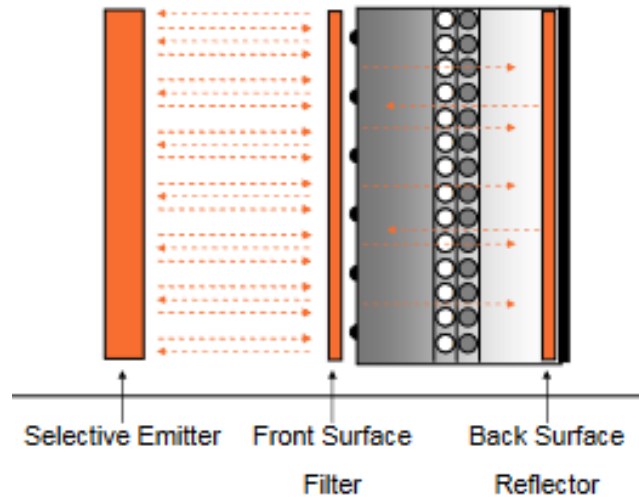


Figure 29. Spectral Control Devices. Source: [48].

The selective emitter shapes the spectrum of radiation from the heat source so that only the portion of the spectrum that is usable by the TPV cell is transmitted. This helps to keep the cell cooler and lessen the burden on the heat rejection system since unusable spectral energy is given off as heat within the cell. By limiting the radiation that reaches the cell, the intensity is slightly decreased, however the cell efficiency is better protected from temperature effects. The front surface filter works in a similar way by limiting the flow of below bandgap photons to the cell and reflecting them back to the heat source for recouperation [50]. A back surface reflector is located at the back of the TPV cell and serves to recycle photons that have passed through all layers back into the cell for absorption or back to the emitter to recover energy that would have otherwise been lost.

Since conventional PV cells are not in close proximity to their emitter (the sun), these features are not considered in Equation 3 for efficiency. TPVs, however, are close enough to their emitters that energy that is reflected away from the cell can be reabsorbed by the emitter, thus increasing efficiency of the overall system by keeping the emitter hot. Equation 5 gives the TPV efficiency equation, where P_{out} is the output power of the cell, Q_c is the heat absorbed in the cell, P_{inc} is the incident energy, and P_{ref} is the power reflected back to the emitter [2].

$$\eta = \frac{P_{out}}{P_{out} + Q_c} = \frac{P_{out}}{P_{inc} - P_{ref}} \quad (5)$$

3. TPV Cell

As previously discussed, TPVs are focused on the IR region of the electromagnetic spectrum, which has longer wavelength photons. Using Equation 1, in order for these photons to cause photogeneration, the bandgap of the materials used in the cells must be significantly reduced. Unfortunately, lower bandgap materials also typically result in lower V_{oc} . To combat this, TPV cells are typically placed in series or configured as multijunction cells. Table 2 lists the bandgaps and peak wavelengths of the materials found in Figure 28. Some materials indicate a range of values since the mole fractions of the individual elements can be adjusted to fine tune the cell as needed. As shown in the figure, materials typically utilized in conventional PV cells, such as Si or GaAs, have bandgaps that are much too high for the IR spectrums given. Ideally, a bandgap between 0.5-0.6 eV would maximize photogeneration in these spectrums. More on this topic will be discussed in the Chapter V.

Table 2. Bandgaps of Common Semiconductors. Source: [48].

Material	E(eV)	λ (μm)
GaAs	1.42	0.87
Si	1.12	1.11
GaSb	0.72	1.72
Ge	0.66	1.88
InGaSb	0.6	2.07
InGaAsSb	0.50-0.74	2.48-1.68
InAsSbP	0.39	3.18

4. Cold Reservoir

The cold reservoir most commonly takes the form of a radiator for space applications, though some designs do incorporate an active cooling system. Since convection is not possible and conduction is limited to spacecraft design, radiation is the only means of heat rejection in space. Equation 6 gives the radiator size, A (m^2), needed to reject a given amount of heat in space.

$$A = \frac{Q}{\epsilon \sigma T_{rad}^4} \quad (6)$$

In this Equation, Q is the heat to be rejected (W/m^2), T_r is the temperature of the radiator (K), ϵ is the emissivity of the radiator surface, and σ is the Stefan-Boltzmann constant. Even minute changes in radiator temperature can drastically change the size of radiator needed to cool the system. TPV efficiency can degrade drastically with temperature just as it does with conventional PVs, so heat rejection must be prioritized when striving for higher efficiencies. This must be coupled with realistic design as well, however, since radiators can make up a large part of the overall mass budget for a spacecraft.

This chapter has given an in-depth explanation of how TPV cells and TPV power systems directly convert thermal energy into usable power. The next chapter will provide information on the modeling software used at the Naval Postgraduate School and the current state of TPV technology for space applications.

V. PERFORMANCE OF TPVS

Manufacturing any type of PV cell can be costly and time consuming considering they are inherently complex and operate at an atomic level. For this reason, advancements in PV technology seemingly stalled for several years, as can be seen in Figure 24. With the advent of computers, researchers were able to begin simulating and modeling their work instead of having to build-and-check their physical prototypes, saving valuable time and money. The Naval Postgraduate School has pioneered the use of Silvaco Virtual Wafer Fabrication software to accurately model both PV and TPV cells. This chapter serves to analyze the performance of TPVs from previous NPS theses and the current state of TPV technology.

A. SILVACO

Founded in 1984, Silvaco Incorporated is a software company dedicated to providing electronic design automation (EDA), technology computer aided design (TCAD), and semiconductor design intellectual property (IP) to users worldwide. There are many programs within the Silvaco software suite that support a wide range of semiconductor modeling and simulations. To model PVs and TPVs, the Virtual Wafer Fabrication (VWF) toolbox should be used in conjunction with Atlas. Separately, MATLAB can be used to iterate runs in Atlas to support optimization efforts. Figure 30 shows how Atlas integrates with the VWF tools.

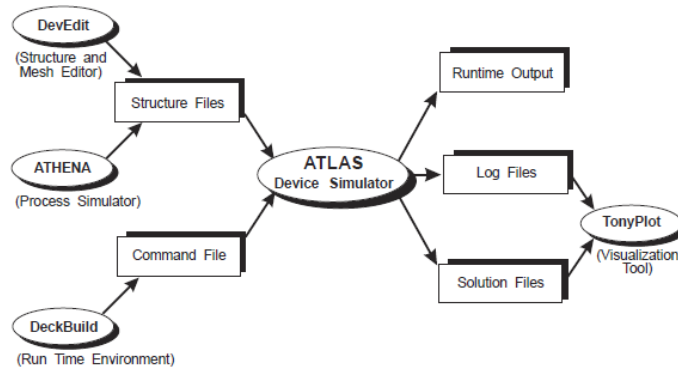


Figure 30. Atlas Inputs and Outputs. Source: [51].

Atlas is a physically-based device simulator, meaning that it can predict the electrical characteristics and responses of physical structures through the use of two- or three-dimensional mesh grids comprised of user-defined characteristics and nodes. This sets it apart from other modeling and simulation software, which are typically empirically-based. Through the DeckBuild interface users can define any and all features of a PV or TPV cells, such as the materials and thickness of the cell layers, doping levels, meshing, light beams to be used in the simulation, and even radiation exposure. Through this program it is also possible to model a multijunction cell. While Atlas does have an extensive library of materials, the user can define their own material and its characteristics. Once the simulation has been run, results can be displayed graphically using TonyPlot [51]. Silvaco models will be covered more in the following sections.

B. GALLIUM ANTIMONIDE CELL OPTIMIZATION

In June 2004, Ensign Bradley Davenport, USN, completed his thesis entitled “Advanced Thermophotovoltaic Cells Modeling, Optimized for Use in RTGs for Mars and Deep Space Missions” [52]. Using Silvaco, ENS Davenport was able to model single-junction Gallium Antimonide (GaSb) and Indium Gallium Arsenide (InGaAs) cells under the spectrum of a 1300 K blackbody. His best performing design, shown in Figure 31, was the optimized GaSb cell constructed by varying thickness and doping levels for each layer in the cell and iterating the Atlas software with MATLAB.

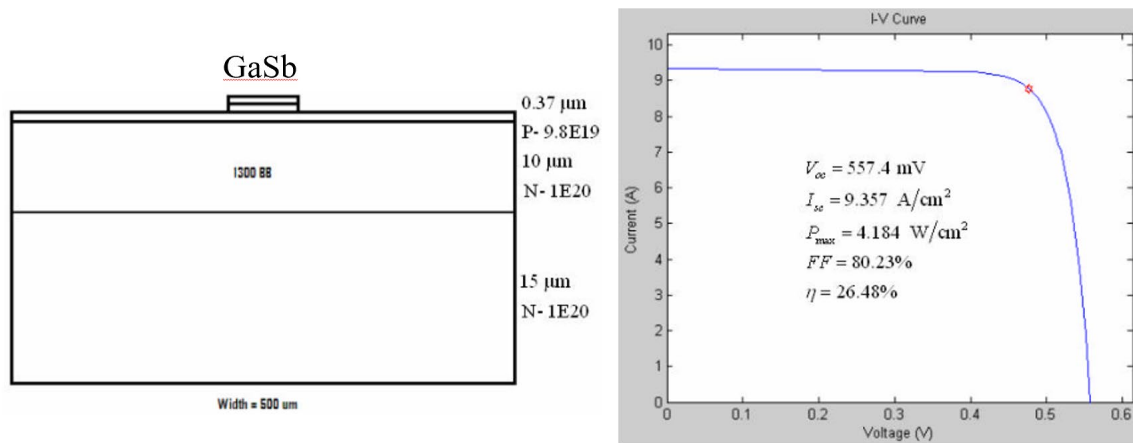


Figure 31. GaSb Silvaco Models. Source: [52].

As shown, ENS Davenport's optimized GaSb cell was able to achieve 26.48% efficiency when exposed to a 1300 K blackbody spectrum. This temperature was chosen because it closely matches the expected temperature and resulting spectrum from an RTG. When compared to the expected efficiency of a thermocouple, this cell has the potential to generate over three times as much power. This is significant because a simple swap of thermocouples to TPV cells within an RTG system could result in a three fold increase in power generation without any additional components.

C. MONOLITHIC INTERCONNECTED MODULES

Lieutenant Andrew Presby, USN, completed his thesis in December 2004 entitled "Thermophotovoltaic Energy Conversion in Space Nuclear Reactor Power Systems" [48]. Through his research with Bechtel Bettis, Inc., LT Presby explored the use of Monolithic Interconnected Modules (MIM) in TPV cells. A MIM design connects several PV or TPV cells in series on a common insulating substrate [53]. Since TPV cells typically have high current densities and low output voltages, by connecting the cells in series the cumulative output voltage is increased while the current is preserved. Figure 32 shows a cutaway view of Bechtel Bettis's cells and how they would be connected via MIM. The deep groove allows the bus bar to connect the conduction layer of the first cell with the top contact of the second cell, and continues on in the same way until the desired number of series cells are connected [48].

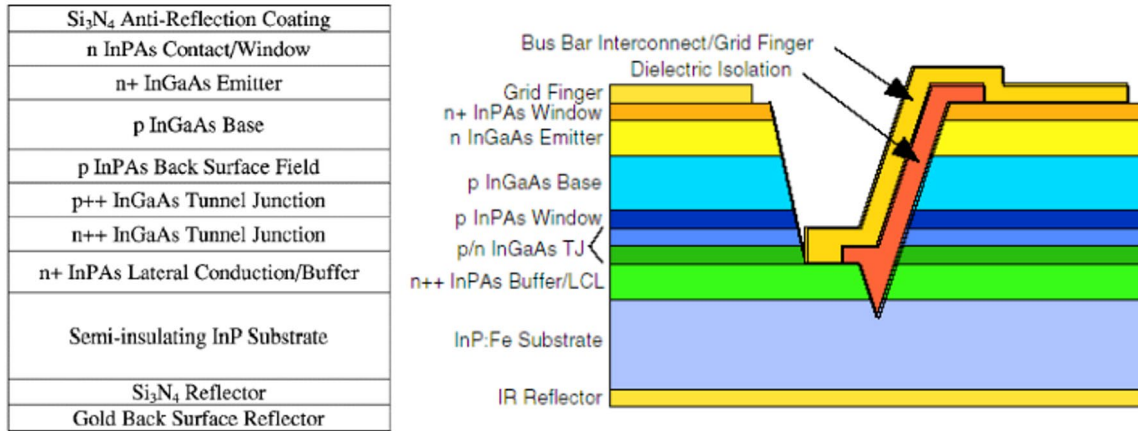


Figure 32. MIM TPV Cell Provided by Bechtel Bettis.
Source: [48].

The array built and tested by Bechtel Bettis was 2 cm², and each MIM within the array consisted of 30 cells. As shown in Figure 32, their cells have a built-in back surface reflector to provide some efficiency improvements. In the practical testing of the arrays, Bechtel Bettis implemented a front surface filter for additional spectral control. Overall the prototype array was able to achieve 23.9% efficiency operating under a 1331 K grey body emitter with spectral control measures in place [48].

This research is significant because the cells were physically built and tested under realistic operating conditions by Bechtel Bettis based on their modeling and optimization efforts. This further proved the validity of their models and verified this concept as a viable energy conversion method. The efficiency noted above is realistic for operation in an RTG system as the TPV cells were held at an elevated temperature of 400 K. As expected, higher efficiencies were recorded at lower temperatures. The construction of MIM cells is more complex and expensive and thus may not be appropriate for every application. Designers will need to decide whether the significant improvement in output voltage, and resulting output power per cm², is worth it.

D. TPV PERFORMANCE AT ELEVATED TEMPERATURES

A year later in December 2005, Lieutenant Burt Canfield, USN, completed his thesis entitled “Advanced Modeling of High Temperature Performance of InGaAs

Thermophotovoltaic Cells” [44]. As discussed previously, all PV cells experience degradations in output power as temperature increases. Additionally, heat rejection is more difficult in space due to lack of convection. These two principles were the basis of LT Canfield’s research. His thesis explored the potential benefits of operating TPVs at less than optimum temperatures in favor of a more favorable overall system design. Using Silvaco’s Atlas program in conjunction with MATLAB, LT Canfield was able to iterate the different cell parameters and create an optimized cell design. As shown in Table 2, the bandgap for an InGaAs cell can vary between 0.5 eV and 0.74 eV depending on the mole fractions chosen. In his research, LT Canfield optimized both a 0.55 eV cell and a 0.74 eV cell separately, and then estimated how they would perform together in a dual-junction configuration. Each cell was optimized individually for use at five different operating temperatures.

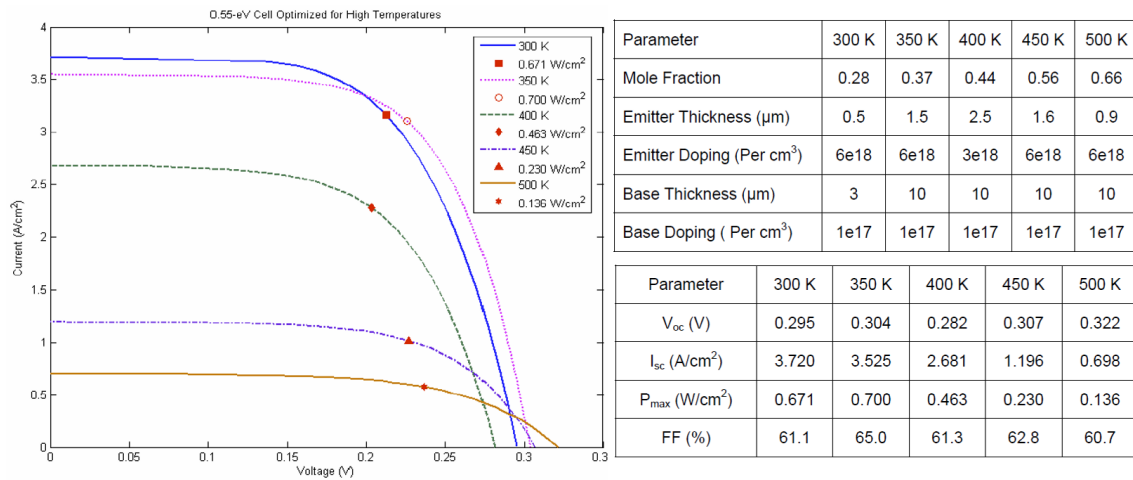


Figure 33. 0.55eV InGaAs Optimized Cell. Source: [44].

Figure 33 shows the performance results of the 0.55 eV optimized cell, and Figure 34 shows the results of the 0.74 eV optimized cell. Overall, the 0.74 eV cell was more affected by the increases in temperature and less affected by the changes in mole fraction than the 0.55 eV cell. In other words, even the most optimized 0.74 eV cell for each temperature did not perform much better than the baseline cell he used to start the optimization algorithm. On the other hand, the optimized 0.55 eV cell performed well

above the baseline cell at every temperature. These results show that for an InGaAs cell there exists a truly optimum mole fraction with an ideal bandgap that can be used at elevated operating temperatures, but above a certain mole fraction the higher bandgap will adversely affect the performance at high temperatures.

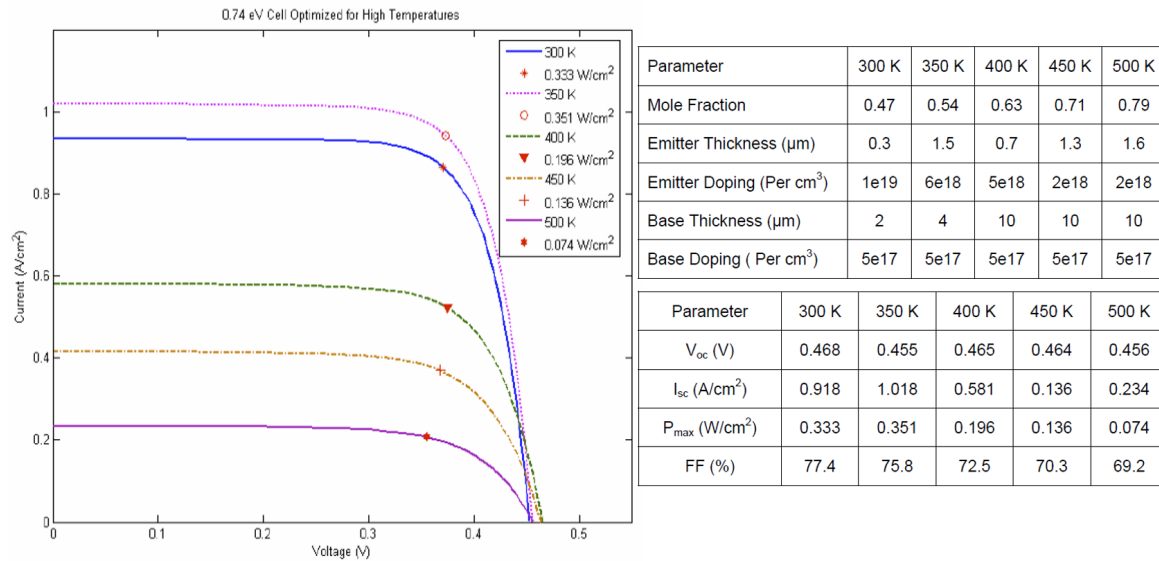


Figure 34. 0.74eV InGaAs Optimized Cell. Source: [44].

At the time of LT Canfield's thesis, the Silvaco Atlas software was not capable of processing multiple mole fractions of InGaAs or modeling the tunnel junctions found in multijunction solar cells. Instead, he extrapolated the performance of a double-junction cell based on the individual single-junction performances, noting that this would be a gross estimation. Though the maximum power output of the combined cell is likely overestimated here, the data is still valuable. The results of this model are shown in Figure 35.

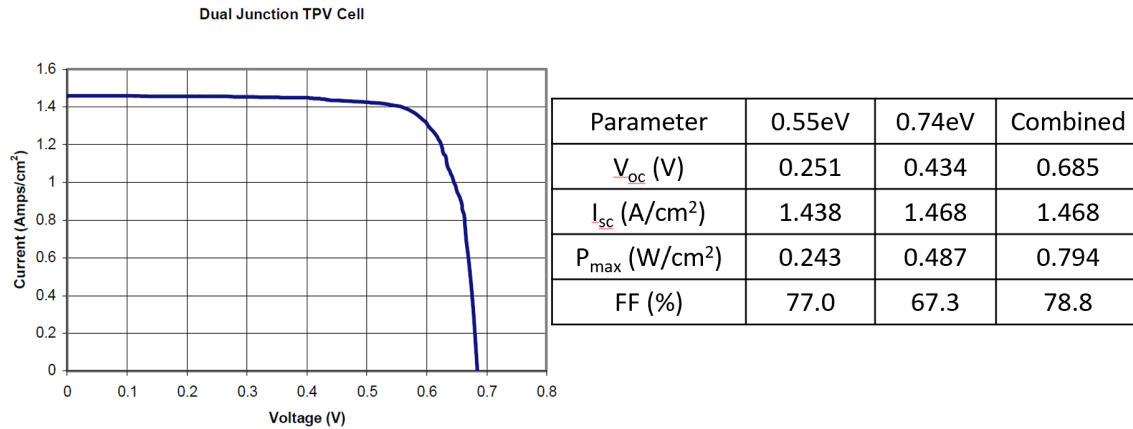


Figure 35. Dual-Junction TPV Performance. Source: [44].

LT Canfield's research is one of the few studies that is centered entirely around TPV performance at elevated temperatures. Most of the research that exists focuses on extracting the highest possible efficiencies to make TPV technology seem more appealing, however this is not particularly useful for space applications where the operating environment is more difficult to control. His thesis culminates in an analysis of the space radiator area and mass savings that would be accomplished if the TPVs were optimized to operate at 400 K vice 300 K. In Equation 6, these temperatures are referring to T_{rad} term in the denominator. Based on his model, at 300 K the cells are 20% efficient, but would require a 174 m² radiator to stay cool. If the cells were to operate at 400 K, the system would only be 14% efficient due, but the radiator would only need to be 84 m². This is over a 50% decrease in radiator size, and the resulting mass savings cannot be ignored. Even though the cell efficiency is only 14% at 400K, this is still almost double the efficiency of the thermocouples being used in current RTG systems [44].

E. THERMAL ENERGY GRID STORAGE

In 2021, a joint study on TPV energy conversion by the Massachusetts Institute of Technology (MIT) and the National Renewable Energy Laboratory (NREL) in Golden, Colorado was conducted to investigate thermal energy grid storage for terrestrial use. The basis of their study was to devise a system to use electricity at off-peak hours to create heat, store the heat, and then convert that heat back into electricity for peak usage hours

using TPVs. Figure 36 shows a diagram of this concept. For example, solar cells could be used to power the grid and create heat to be stored during the day, and TPVs could convert the heat back into electricity at night to continue to power the grid. Though this thesis is primarily concerned with TPVs for space applications, the study's improvements in efficiency and design are notable.

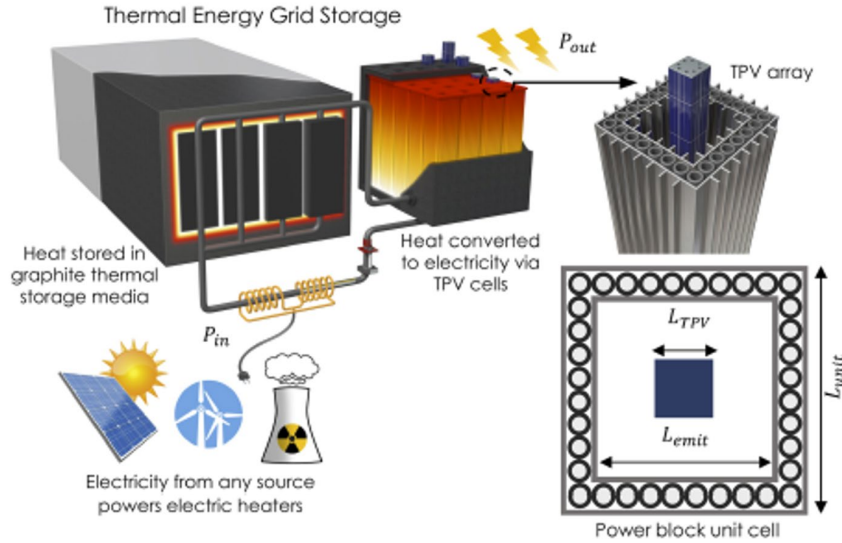


Figure 36. Thermal Energy Grid Storage System with TPVs. Source: [2].

The study focused on two different cells designs, both high-bandgap, double-junction TPV cells optimized for a 1,900-2,400 °C emitter. While these temperatures may not be suitable for an RTG system, they could be great candidates for use with a fission or solar concentrator system. The cells designed had bandgaps of 1.2/1.0 eV and 1.4/1.2 eV. Figure 37 shows the basic cell designs with the associated spectral response and IV-curves. For spectral control measures, these cells used only a highly reflective gold mirror back surface reflector.

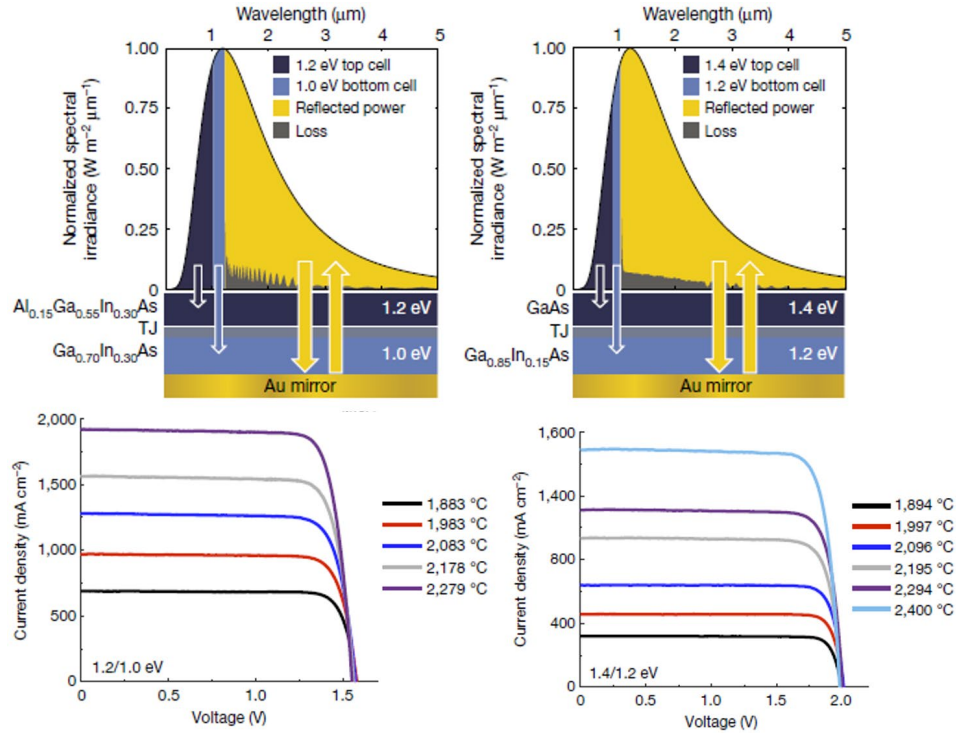


Figure 37. 1.2/1.0eV and 1.4/1.2eV TPVs with Spectral Responses and IV-Curves. Source: [2].

Overall the noted efficiency averaged over the temperature range was 39.3% for the 1.2/1.0 eV cell and 36.2% for the 1.4 eV/1.2 eV cell [2]. It is important to note that since this system was designed for terrestrial use, there is an integrated cooling system which relies heavily on convection to maintain the cells at approximately 25°C. This is why even as emitter temperature increased by 500°C, the V_{oc} hardly degraded. For space applications, these cells would be used in conjunction with a radiator for heat rejection. Given that radiation is not as efficient as convection, the cell operating temperatures would likely be much higher, degrading the V_{oc} and reducing the overall efficiency. That said, these optimized cells would still be a great option for use with a nuclear fission reactor or a solar concentrator system since even with temperature degradation they would still offer at least a threefold improvement in efficiency over what current thermocouples can provide.

F. SOLAR CONCENTRATOR SYSTEMS

When the International Space Station (ISS) concept was first being developed in the 1980s, the original design featured large solar arrays like the ones found on the ISS today, however, it also included a planned expansion utilizing solar concentrator arrays and a dynamic power conversion module to meet the anticipated increased demand for power once the station was established [54]. Figure 38 shows a diagram of the proposed solar dynamic concentrator array. A Rankine cycle, Brayton cycle, and Stirling engines were all considered in the original designs for this module, but the solar dynamic concept was ultimately abandoned over concerns regarding the longevity and reliability of a dynamic system providing power to a manned module, and today's ISS is powered solely by solar cells.

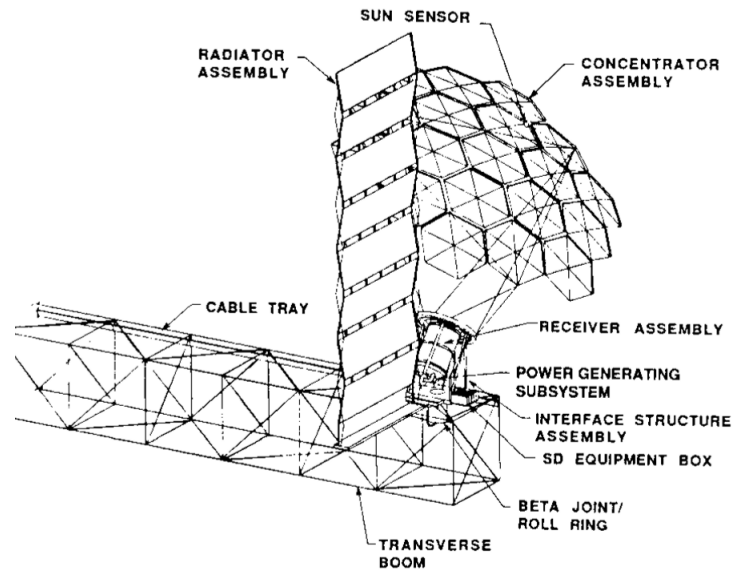


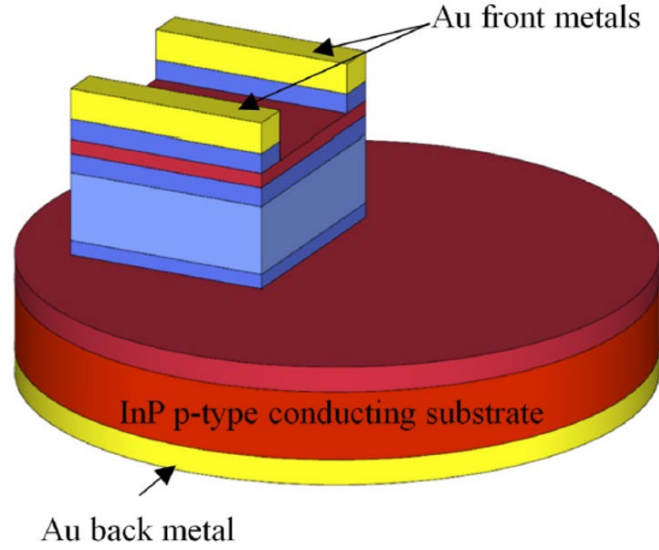
Figure 38. Solar Dynamic Power Module. Source: [54].

Although this concept was never built, it was thoroughly designed and ready for production. The concerns regarding dynamic cycles in the unpredictable space environment are not unfounded, yet this design could still be utilized today. Instead of a dynamic power conversion system, TPVs could offer an alternative stable, static power conversion method. In Figure 38, power conversion module shown could be replaced by

a heat pump of sorts, and the TPVs could line the given radiator assembly. In this configuration, the solar concentrator would produce heat, which would be transferred to the TPVs for power generation. The TPV cells would stay cool due to their positioning on the radiator assembly, and thus maintain ideal efficiencies. This concept is provided in this section to show that TPV cells are not limited to use with nuclear devices, but rather can be used in conjunction with anything that produces a significant amount of heat.

G. REAL CODED GENETIC ALGORITHM OPTIMIZATION

Concurrently with the MIT-NREL joint study, the Universiti Tenaga Nasional in Malaysia and the University of Sheffield in the United Kingdom partnered in the optimization of InGaAs TPVs. Their technique was unique in that it used a real coded genetic algorithm (RCGA) which varied multiple cell parameters under different emitter temperatures simultaneously, whereas previous works optimized a single variable at a time. Similar to LT Canfield's work, the specific mole fraction chosen for the study was $\text{In}_{0.53}\text{Ga}_{0.47}\text{As}$, which has a bandgap of 0.74 eV. Due to its crystalline properties, this cell can be grown lattice-matched onto an InP substrate, which can be more cost effective than GaSb or GaAs on a large scale [55]. Figure 39 shows the cell structure and the ranges for each parameter that was tested.



In _{0.53} Ga _{0.47} As cell	Thickness (μm)		Doping concentration (cm ⁻³)	
	Lower limit (x_L)	Upper limit (x_U)	Lower limit (x_L)	Upper limit (x_U)
Cap layer	0.02	0.14	8×10^{17}	6×10^{19}
FSF layer	0.02	0.14	8×10^{17}	6×10^{19}
Emitter layer	0.05	0.32	5×10^{16}	7×10^{18}
Base layer	2.00	20.0	5×10^{16}	1×10^{18}
BSF Layer	0.02	0.14	8×10^{17}	6×10^{19}
Buffer layer	0.02	0.14	8×10^{17}	6×10^{19}

Figure 39. Cell Structure and Parameters for RCGA Optimization.
Source: [55].

The model was built and tested using Silvaco VWF software in conjunction with the RCGA. Like many of the other studies, it was assumed here that the system would have an adequate heat rejection system, and the cells were modeled at 27 °C for all emitter temperatures. Table 3 shows the efficiencies of the RCGA optimized cells for each temperature tested. The RCGA_{Practical} column takes actual manufacturing abilities into account for the optimum parameters determined by the algorithm. For example, greater efficiencies were produced when the base layer thickness was between 16 and 18 micrometers, however it is extremely difficult to produce a high quality InGaAs layer that thick. As a result, the base layer thickness was capped at 8 micrometers, which did affect the cell's overall efficiency.

Table 3. Results of the RCGA Optimization. Source: [55].

Radiation Temperature (K)	η using RCGA (%)	η using RCGA _{Practical} (%)
800	18.4	16.73
1000	21.92	20.07
1200	23.1	21.63
1400	23.18	22.06
1600	22.46	21.74
1800	20.83	20.48
2000	19.76	19.76

Although it may appear that the efficiencies noted in Table 3 may be inflated since the cells were kept at an optimum temperature, it is noteworthy that no spectral control measures were implemented. If even a back surface reflector were implemented for photon recycling it would increase the overall efficiency of the system. Like LT Canfield's cell, this design would be a good candidate for space applications with spectral control measures in place. This study proves that a RCGA used in conjunction with Silvaco is an effective method of optimizing cell parameters, which are not limited to TPV designs.

H. NEARLY ORTHOGONAL LATIN HYPERCUBE OPTIMIZATION

In June 2022, LT Brandy Allain completed her thesis on thin-film copper indium gallium (di)selenide (CIGS) solar cells [56]. These cells were unique in that they featured a vertical p-n junction and back surface contacts to eliminate inefficiencies from the top contact shading. To optimize the CIGS cells, LT Allain utilized a nearly orthogonal latin hypercube (NOLH) to vary cell parameters simultaneously across multiple runs in Silvaco. Like the RCGA mentioned in the previous section, this allows for improved trend recognition over single-varying parameter methods. The input to the NOLH is the number of parameters to be varied and their upper and lower bounds. From there, the NOLH creates a matrix of "design points" that have evenly spaced the parameter values across each column. These matrices of parameters were then modeled using Silvaco Atlas software, and further analyzed using JMP, which is a statistics analysis tool. Overall,

three different data sets generated from the NOLH were run. The initial set was based on a “best guess” set of initial parameters from a previous work with wide upper and lower boundaries for each. The next set used the JMP software to further analyze each parameter’s effect on efficiency and on each other to create a new “best guess” with slightly narrower boundaries. The same process was used to generate the last dataset, but with more restrictive boundaries. Figure 40 shows efficiency percentage for each trial.

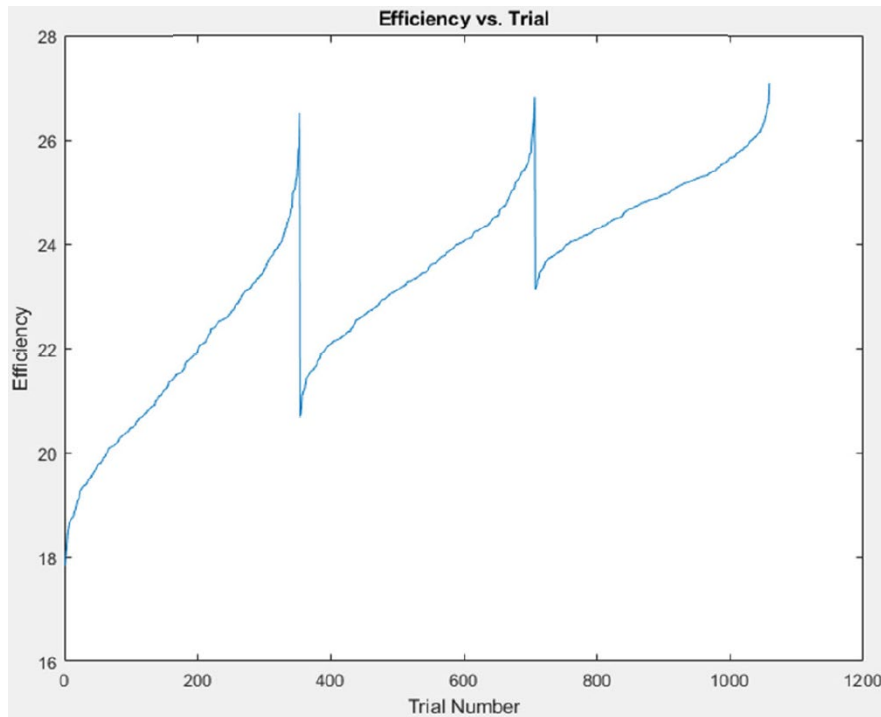


Figure 40. Efficiency vs. Trial. Source: [56].

Figure 40 shows that with each subsequent run the cells grew more efficient. Using the NOLH optimization, LT Allain was not only able to attain an optimized cell design, but she was also able to attain efficiencies that had never been achieved by any CIGS cell before: 27.09%. The purpose of mentioning this research was to highlight the potential benefits of using a NOLH for future optimization of TPV cells. While the design and use cases for TPVs differ greatly from CIGS cells, the same optimization process could be implemented with different parameters in place.

This chapter has explained the Silvaco software suite and provided examples of previous work on TPV cells and effective optimization techniques. ENS Davenport, LT Presby, and LT Canfield proved in their theses that TPV efficiencies rival those of conventional PV cells, and even with losses at elevated temperatures there are still clear benefits from an efficiency standpoint to incorporating TPVs into RTG system designs going forward. The MIT-NREL study furthered this point with their research into terrestrial use for TPVs, which resulted in a TPV cell that is almost 40% efficient under ideal conditions. The Tenaga-Sheffield joint study, along with LT Allain's thesis, proved that TPV cells can be further optimized using creative algorithms and Silvaco's modeling tools to simulate their performance. While extensive work on TPVs for space applications is lacking, the optimization techniques described have the potential to yield surprising results to the already promising future of TPVs. Chapter VI will focus on the potential benefits or drawbacks of implementing TPVs into spacecraft design from a size, power, weight, and reliability standpoint.

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VI. SPACECRAFT DESIGN

Space has become more accessible than it's ever been as the overall cost of launch has decreased and companies such as SpaceX have begun offering rideshare programs. This has allowed smaller companies to more easily flight test their experimental designs, contributing to a surge of growth within the industry. As solar cell efficiency has begun to stagnate in recent years, designers have had to get creative in designing deployable solar arrays large enough to support their increasingly complex payloads. This chapter will focus on the potential benefits or drawbacks of utilizing TPVs in conjunction with an RTG in place of traditional solar arrays. Table 4, taken from the Space Mission Analysis and Design handbook, shows the most common methods for space power systems and how they compare to one another across a range of categories. These topics will be discussed in depth in the following sections.

Table 4. Comparison of Common Space Power Sources. Source: [57].

EPS Design Parameters	Solar Photovoltaic	Solar Thermal Dynamic	RTG	Nuclear Reactor	Fuel Cell
Power Range (kW)	0.2–300	5–300	0.2–10	5–300	0.2–50
Specific Power (W/kg)	25–200	9–15	5–20	2–40	275
Specific Cost (\$/W)	800–3,000	1,000–2,000	16K–200K	400K–700K	50K–100K
Hardness – Natural Radiation – Nuclear Threat – Laser Threat – Pellets	Low–Medium Medium Medium Low	High High High Medium	Very high Very high Very high Very high	Very high Very high Very high Very high	High High High Medium
Stability & Maneuverability	Low	Medium	High	High	High
Low-orbit Drag	High	High	Low	Medium (due to radiator)	Low
Degradation Over Life	Medium	Medium	Low	Low	Low
Storage Required for Solar Eclipse	Yes	Yes	No	No	No
Sensitivity to Sun Angle	Medium	High	None	None	None
Sensitivity to Spacecraft Shadowing	Low (with bypass diodes)	High	None	None	None
Obstruction of Spacecraft Viewing	High	High	Low	Medium (due to radiator)	None
Fuel Availability	Unlimited	Unlimited	Very low	Very low	Medium
Safety Analysis Reporting	Minimal	Minimal	Routine	Extensive	Routine
IR Signature	Low	Medium	Medium	High	Medium
Principal Applications	Earth-orbiting spacecraft	Interplanetary, Earth-orbiting spacecraft	Interplanetary	Interplanetary	Interplanetary

A. SIZE

As spacecraft have grown increasingly more sophisticated, their demand for power has risen exponentially. Early satellites, such as DSCS II, were spin-stabilized and could be adequately supported with only body-mounted solar cells and batteries for power. Altogether these rudimentary satellites only required a few hundred watts of power to support their missions, whereas today's satellites may require several kilowatts [57]. For example, the GOES-R satellite launched in November 2016, features a vast 5 ft by 13 ft deployable solar array capable of producing up to 5 kW of power [58]. For missions beyond Earth's orbit, solar arrays must be scaled appropriately to the solar intensity available at their final destination. NASA's Psyche spacecraft, which will travel to the asteroid of the same name located between Mars and Jupiter, features two massive deployable solar arrays that are each 37 ft long. In total the spacecraft will be capable of producing 21 kW of power near Earth, but once at the asteroid the panels are projected to only produce 2 kW due to the increased distance from the sun [59].

Utilizing TPVs with an RTG heat source offers an overall more compact design than solar cells, requires no deployable components, and is a completely self-contained system since TPVs are agnostic to the sun. The GPHS RTG mentioned in Chapter II measures only 1.38 ft in diameter and 3.74 ft tall and can produce up to 300 W of power. Similarly, the MMRTG measures just 2.1 ft in diameter by 2.2 ft tall and can produce up to 125 W of power. For a mission like Psyche, 6–7 GPHS RTGs would need to be carried in order to provide the equivalent amount of power in their current configuration. However, both the GPHS and the MMRTG still use thermocouples instead of TPVs for energy conversion. At best, thermocouples are no more than 8% efficient, whereas TPVs have been proven to achieve efficiencies of well over 20%. Therefore, if TPVs were integrated into the GPHS RTG, one unit could potentially produce 750 W or more instead of 300 W based solely on the increased energy conversion efficiency. For the Psyche mission, this means that only 2–3 TPV-GPHS units would be needed to produce the necessary 2 kW of power, mirroring the Cassini spacecraft which carried three GPHS RTG units on its mission.

B. POWER

Another benefit to using TPVs over conventional PVs is the complete elimination of the need for an energy storage device. Historically, Nickel-Cadmium (NiCd) or Nickel-Hydrogen (NiH₂) batteries have been used for space applications, however advancements in Lithium-ion (Li-ion) battery technology have made them a popular choice. Table 5 shows the characteristics for each battery type. As with any household battery, its capacity for charging degrades slowly with each charge-discharge cycle until it can no longer hold a charge, typically around 15–18 years for Li-ion [57]. In many cases the batteries are the first component to fail, leaving the spacecraft usable only when the solar arrays are in sunlight and effectively ending the mission prematurely. RTG-powered spacecraft are not limited by these factors, and in many cases, like the Voyager missions, have continued to operate well beyond their life expectancy.

Table 5. Characteristics of Typical Batteries for Space Applications.
Source: [57].

Performance Characteristics for Rechargeable Batteries	Ni-Cd	Ni-H ₂	Li-Ion	System Impact
Energy Density (W-hr/kg)	30	60	125	Mass savings and vehicle center of gravity
Energy Efficiency (%)	72	70	98	Reduction of charge power reduces solar panel mass & size
Thermal Power (scale 1–10)	8	10	1	Reduction of radiators, heat pipes sizes
Self Discharge (% per day)	1	10	0.3	Simple management at launch pad & more margin during transfer
Temperature Range (°C)	0 to 40	–20 to 30	10 to 25	Management at ambient and thermal control reqs
Memory Effect	Yes	Yes	No	No reconditioning management
Energy Gauge	No	Pressure	Voltage	Easier state of charge assessment
Trickle Charge	Yes	Yes	No	Balancing need prior to eclipse. Li-Ion typically requires cell equalization circuitry.
Modularity	No	No	Yes	One cell design
Heritage	Yes	Yes	Yes	Risk Assessment. Continued Li-Ion LEO testing and missions establishing heritage.

As shown in Table 5, batteries are also extremely sensitive to temperature changes, Li-ion being the most delicate of the three. For this reason, it is common for designers to incorporate heaters to maintain sensitive components at proper operating

temperatures. By using an RTG system, the need for energy storage devices and heaters is completely eliminated. Furthermore, if there are other sensitive components onboard that require heaters, like a fuel tank for example, waste heat from the RTG can be siphoned off to help maintain proper temperatures of these components. This is yet another benefit to TPV+RTG systems. The potential mass savings from the elimination of these components will be discussed further in the next section.

Another factor to consider when designing a spacecraft power system is its mission. For systems requiring high-intensity power output, such as a high-fidelity imagery system or laser communications system, the power module must be able to support rapid discharging. In present systems, this is only possible if the batteries are able to maintain a high enough charge to support these operations. Oftentimes high-intensity operations can occur simultaneously, such as slewing and imaging, followed closely by transmitting to a ground station. Designers must consider the orbital parameters, time in view of the sun versus eclipse, and annual solar intensity variations to determine appropriate battery and solar array sizing requirements to support the spacecraft's mission.

In a TPV+RTG system, given that power is continuously being generated at a steady rate, the operation of a high-powered payload would be less of a concern. For reference, Spectrolab, Inc.'s best triple-junction solar cell has a $P_{\max}$ of 0.05 W/cm^2 at BOL, whereas LT Canfield's and the MIT-NREL TPV cells discussed in Chapter V have a $P_{\max}$ of approximately 0.7 W/cm^2 [60]. If the steady output power were not enough to support the necessary operations, capacitors or small batteries could be integrated to provide a quick release of energy when needed. These components would immediately be recharged by the RTG power system and ready for use in a much shorter amount of time than a conventional PV system since it would not be constrained to time in view of the sun.

C. WEIGHT

As discussed in Chapter II, the MMRTG currently in service weighs around 45 kg, and the GPHS RTG weighs approximately 57 kg. According to the Space Mission

Analysis and Design handbook, the mass of a solar array (M_{sa}) can be estimated using Equation 7, where P is the required output power of the array in Watts at EOL.

$$M_{sa} = 0.04P \quad (7)$$

This equation assumes the lower boundary for specific power for solar cells from Table 4. Similarly, the mass of batteries can be found using Equation 8, where C is the capacity of the battery in W-hrs and U_E is the energy density in W-hr/kg.

$$M_{batt} = \frac{C}{U_E} \quad (8)$$

For the GOES-R satellite, which produces 5kW of power through its solar array, the estimated mass of its solar array using Equation 7 is 200 kg, and the mass of its batteries is 87 kg using Equation 8. For the Psyche spacecraft, producing 21 kW, its solar array weighs around 840 kg, and its batteries 58 kg. Assuming that there would not be a significant change in weight by swapping thermocouples for TPVs, a TPV+RTG system could offer huge savings to the overall spacecraft mass budget. Table 6 summarizes these potential mass savings. For comparison, an analysis on the Iridium-NEXT satellite, which will remain in Earth's orbit, was also included.

Table 6. Typical Mass of Power Components and Potential Savings.
Sources: [57], [61]–[63].

Spacecraft	Power Required (W)	Mass of PV Array (kg)	Mass of Batteries (kg)	Mass of TPV GPHS (kg)	Mass Savings (kg)
GOES-R	5,000	200	87	399	-112
Psyche	21,000/2,000	840	58	171	727
Iridium-NEXT	2,000	80	27	171	-64

*Does not include mass of heaters.

It is important to note that the data in Table 6 is not all inclusive as it does not include the potential mass savings from the elimination of the heating units or other auxiliary components associated with solar arrays. As expected, there may a threshold

where conventional PVs are the more logical choice over a TPV+RTG system purely from a mass perspective. In general, for near-earth satellites that require 2 kW or more of power, conventional solar arrays “cost” less mass because multiple RTG units would be needed. That said, a TPV+RTG system could easily be scaled-up to support an increased power demand, and would be very competitive in this sector. For interplanetary missions, the advantage of using an RTG system is clear from the start as solar intensity drops off rapidly with increased distance from the sun.

D. RELIABILITY

Special considerations must be made when designing space power systems since oftentimes these units cannot be serviced after launch. Reliability must be maximized in terms of reduction of moving parts, decreased system complexity, and overall hardness or resistance to degradation caused by radiation or extreme temperature cycling. Given these factors, it is easy to see why static power conversion methods are almost always chosen over dynamic, as was the case for the ISS discussed in Chapter V. While dynamic methods have the ability to produce more power at a higher efficiency, there is inherently a greater risk of mission failure due to equipment malfunction. Conventional PVs and thermocouples have dominated the space power segment for decades because of their proven reliability and flight heritage, however, with TPV efficiencies approaching that of conventional PVs there is renewed interest in their use for potential space applications since they use the same technology as the solar cells that the industry is familiar with and require only a source of heat.

1. The Space Environment

The Space Mission Analysis and Design handbook defines “survivability” as “the ability of a space system to perform its intended function after being exposed to a stressing natural environment or one created by an enemy or hostile agent,” and “hardness” as “an attribute defining the environmental stress level which a space system can survive” [57]. TPVs offer the same tried and trusted static energy conversion methods as conventional PVs, and there are significant benefits from a survivability and hardness standpoint to using TPVs with an RTG unit over solar arrays in space.

Since conventional PVs must be exposed to the sun, this means they are also vulnerable to the increasingly unpredictable space environment. Earth is constantly being bombarded with charged particles from the sun in the form of solar wind, solar flares, and even coronal mass ejections (CMEs). Some of these particles, typically protons and electrons, are deflected by the Earth's magnetosphere, but others become trapped in Earth's magnetic field and continue to circulate in belts such as the Van Allen Belts. Figure 41 depicts this concept. These belts can change shape and size depending on solar activity, but in general the inner belt extends between 1,000 km to 10,000 km above the Earth's surface, and the outer belt 13,000 km and 60,000 km [57].

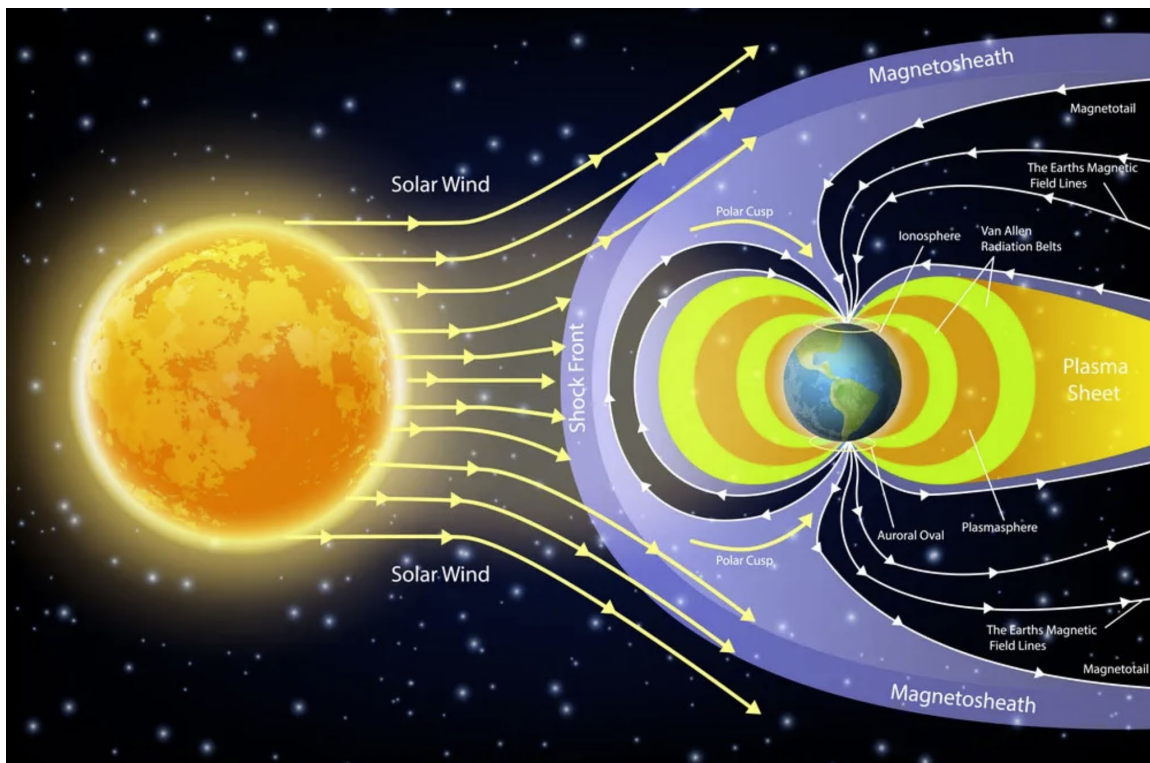


Figure 41. Earth's Magnetosphere and Van Allen Belts. Source: [64].

On average the duration of a complete solar cycle is around 11 years, with the next solar maximum expected in 2023 [57]. As solar maximum approaches, there is a marked increase in sunspots, solar flares, and CMEs, all of which can affect satellites inside and outside of the radiation belts and Earth's magnetosphere. Conventional PV

arrays must be designed to withstand such conditions since they are completely exposed and are prone to irreparable damage from this radiation. To help protect against the incoming charged particles, most space solar arrays are shielded with a doped cover glass and additional shielding on the back surface, however, this protection is only minimal. The expected degradation in output power for shielded single-junction Si cells is 15% after exposure to a total fluence of 10^{16} of 1 MeV equivalent electrons. For shielded triple-junction GaAs cells, the expected degradation is 11% after the same exposure [57]. Depending on the level of solar activity at the time, the solar arrays could encounter this level of electron fluence from natural sources annually, resulting in an 11–15% annual degradation in output power. Designers must take the specific orbit and expected conditions into account when selecting their solar arrays to ensure that the EOL power output will still meet mission requirements after the estimated radiation exposure, which is why BOL power output is much higher than EOL output.

Another element of the space environment that is becoming increasingly problematic is space debris. This debris originates from launch vehicles, satellite collisions, equipment malfunctions, or from natural sources such as micrometeoroids, and the size can range from a speck of dust to a rocket body. As space has become more accessible, the amount of manmade space debris has grown exponentially. For LEO, currently only debris that is 10 cm or larger is able to be catalogued and regularly tracked, however that does not come close to encompassing everything inadvertently stuck in Earth's orbit. It is estimated that at least one debris object exists for every cubic kilometer in space [65]. Given that a typical satellite in LEO travels at a speed of 7.5 km/s, that figure translates to potentially thousands of debris impacts per hour [57]. Unfortunately as solar arrays are scaled up to support higher power requirements, the probability of colliding with space debris and incurring damage increases as well. Figure 42 shows a solar array taken from the Hubble Space Telescope that was severely damaged under normal operations by space debris.

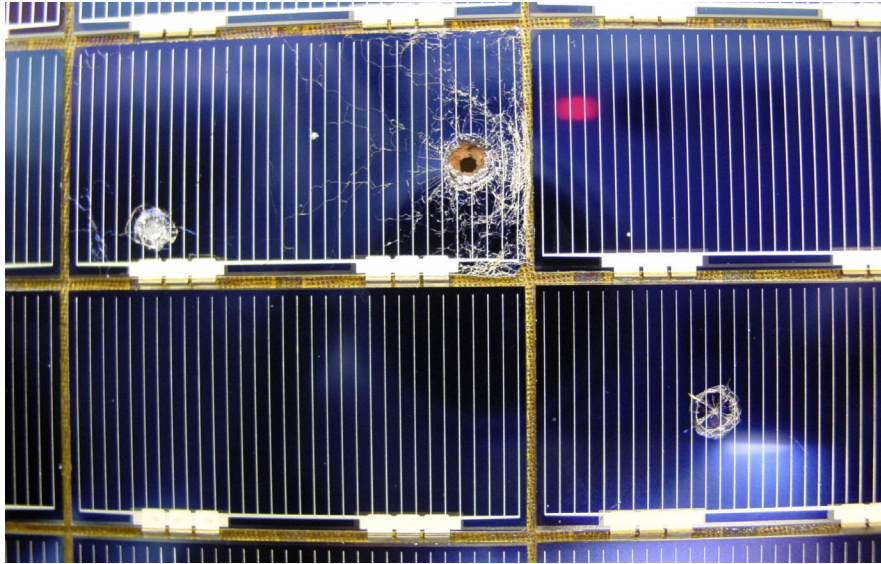


Figure 42. Solar Array from the Hubble Space Telescope. Source: [65].

Regrettably, not everything designers must plan for in the space environment is natural or inadvertent. Nuclear weapons and antisatellite weapons have been, and continue to be, a serious threat to all space systems. Just as solar cells degrade with exposure to natural electron fluence, the total fluence resulting from a nuclear event could be exponentially higher. According to a study conducted by the Defense Threat Reduction Agency, a 50 kT nuclear burst at an altitude of 120 km could result in a 67%-99% reduction in lifetime for non-military satellite systems [57]. Antisatellite weapons, such as laser systems or pellet weapons, are also a concern for exposed vital components such as solar arrays. Similar to how a naval weapon targets the propulsion system of a vessel to incapacitate it, an antisatellite weapon could easily target the spacecraft's solar arrays, leaving it unable to charge. Unfortunately, there is not much that can be done to protect solar arrays from these threats since their shielding must be transparent to allow access to sunlight.

A TPV+RTG power system offers a unique solution to many of these evolving threats, natural or otherwise, since the cells would not be exposed to the space environment at all. Instead, the TPVs would be located just as the thermocouples are in current systems: between the radioisotope heat source and the outer shielding of the module. In this way, the TPVs would be completely protected from incoming charged

particles from the sun, radiation from a nuclear blast, and especially well protected from space debris. It is for this reason that RTG systems are marked as “very high” on the hardness scale in Table 4. Considering the evolving threats of today’s space environment, this should be reason enough for spacecraft designers and mission planners to rethink their space power systems.

2. Mechanics

Just as airplanes experience drag while flying, LEO satellites in particular constantly experience atmospheric drag and must frequently readjust their orbit to avoid orbital decay and eventual reentry. The force of atmospheric drag can be found through Equation 9.

$$F_d = 0.5[\rho C_d A V^2] \quad (9)$$

In this equation, ρ is the atmospheric density, C_d is the coefficient of drag, A is the surface area in the direction of travel, and V is the orbital velocity. This equation is only able to approximate the drag force since the atmospheric density is difficult to measure or predict and changes with solar activity. Table 7 shows the drag coefficients for common satellite shapes. The “Surface” column refers to the surface condition of the spacecraft, specular meaning smooth and diffuse meaning rough. For solar panels, specular can be assumed [57].

Table 7. Equations for Drag Coefficients of Typical Satellite Shapes.
Source: [57].

Shape	Surface	Drag Coefficient
Flat Plate, Face First	Specular	4
	Diffuse	3
Flat Plate, Angle of Incidence α	Specular	$4 \cos^2 \alpha$
	Diffuse	$2 + \cos \alpha$
Sphere	Specular	2
	Diffuse	$8/3 = 2.667$
Circular Cylinder, Side First	Specular	$8/3 = 2.667$
	Diffuse	$2 + \pi/4 = 2.785$
Octagonal Cylinder, Side Face First	Specular	$2\sqrt{2} = 2.828$
Octagonal Cylinder, Side Edge First $\alpha = \cos 22.5^\circ$ $\beta = \cos 67.5^\circ$	Specular	$4 \frac{\alpha^3 + \beta^3}{\alpha + \beta} = 2.586$
Octagonal Cylinder, Side First, Average Over All Side-first Orientations	Specular	$C_D A = 2.812 A_{\text{face first}}$
Circular Cone, Half-angle α, Point First	Specular, Diffuse	$\frac{4 \sin^2 \alpha}{2 + \sin \alpha}$

Solar arrays are modeled by the “flat plate” cases and often have a large surface area, resulting in a large atmospheric drag force. A TPV+RTG system, however, would have a smaller surface area and could be modeled by the “circular cylinder” or “octagonal cylinder” cases, resulting in a lower C_d . Thus, with the reduction of both A and C_d , the overall the effect of atmospheric drag on an RTG system would be much lower than for a conventional PV system, as also denoted in Table 4. This would result in fewer readjustments in orbit for station keeping, and thus less fuel necessary for the mission.

While conventional PVs are considered a static power conversion method, unless the cells are body-mounted to the spacecraft there are, in fact, moving parts involved. Since a spacecraft must fit inside of a rocket fairing for launch, the solar arrays must be stowed and subsequently deployed once released into space. Furthermore, once the arrays are deployed the spacecraft’s instruments must adequately track the sun and move the arrays (if equipped) to absorb the sunlight at the optimum angle to produce the most power. These details add layers of complexity to the system and bring about questions

regarding the reliability of deployment and tracking mechanisms. Figure 43 shows the number of known anomalies for all deployable spacecraft elements. Of these anomalies, 58% of them were for solar array systems [66]. While the percent failure for deployable solar arrays remains low, these mechanisms inject inherent risk that would not exist for TPV+RTG systems, which require no deployable components.

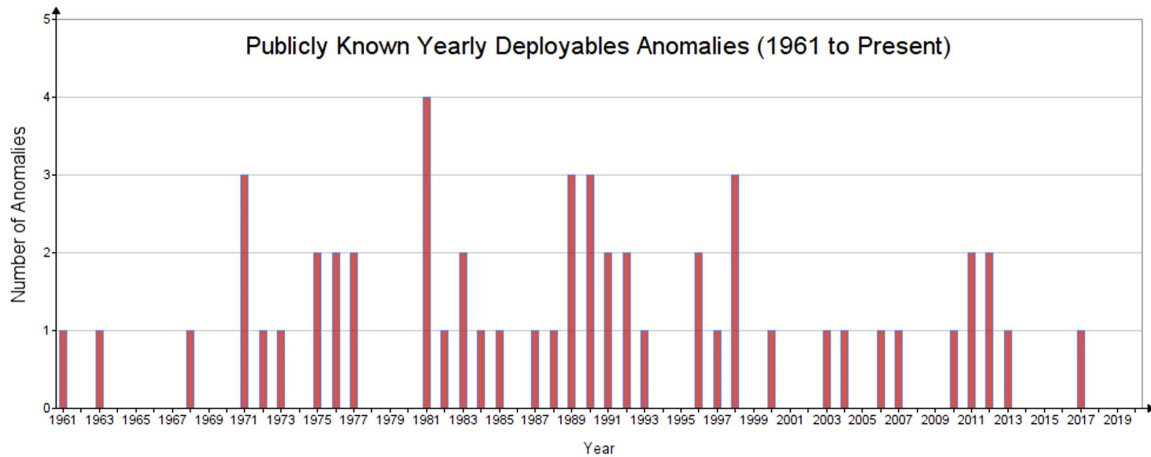


Figure 43. Deployable Element Anomalies 1961–2019. Source: [66].

Fundamentally, spacecraft are put into orbit or sent on interplanetary trajectories to perform a function or provide a service to those on Earth. Designers must balance positioning of equipment vital to the spacecraft’s mission, such as antennas or optical sensors, with its necessary life support equipment, such as the solar arrays. With solar arrays increasing in size this has become increasingly more difficult, resulting in long booms for either the arrays or payload equipment. In addition, the solar arrays can cast shadows onto the bus and payload equipment, or vice versa, creating massive temperature differentials that adversely affect these systems. In some cases, it is not possible to achieve the ideal sun angle and simultaneously complete the craft’s mission due to pointing requirements, so the operators must choose between charging the batteries or completing the mission. Since TPV+RTG systems continuously produce power independent of position, the mechanisms needed to rotate and reposition the spacecraft to either achieve a desired sun angle for the solar arrays or pointing angle for the payloads

can be scaled down or completely eliminated. This decreases overall system complexity, mass, and size, and increases reliability.

3. Safety

Public opinion on nuclear systems has long wavered, as noted in Chapter II with the constant funding and defunding cycles of nuclear-powered space programs. Some of this was due to program cost and budgeting constraints, however, a large portion of these decisions appear to fear-based. Following nuclear incidents such as the Transit-5BN-3 disintegration discussed in Chapter II, it is not difficult to see where the fear originates. Unfortunately, the triumphs of nuclear power systems have been completely overshadowed by their limited shortcomings. Since their redesign, RTGs have successfully survived reentry and subsequent impact to such a degree that the Nimbus-B-1 RTG was able to be recovered from the ocean floor and reused on the Nimbus-3. Additionally, according to NASA, a radioactive power source has never been the cause of a spacecraft failure [67]. Over sixty years of engineering, analysis, and testing has been devoted to safety regarding space nuclear power sources, and after multiple safe reentries it is time to give them credit for their proven flight heritage.

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VII. CONCLUSIONS

Thermophotovoltaic cells have been thoroughly investigated for terrestrial applications as the industrial sector has longed for a device to harvest waste heat from power or manufacturing plants and convert it into usable electricity. Oftentimes, however, the research indicates that the cost of manufacturing and implementing these cells far outweighs the benefits, and these programs have been abandoned. While there may not be much utility (yet) to using TPVs terrestrially, in space the benefits are clear.

TPV cells use the same familiar static energy conversion technology without having the drawbacks of orbital considerations for adequate sun exposure, shading effects, or radiation degradation. When used in conjunction with an RTG, there is no need for batteries, heaters, or deployment mechanisms, and the overall footprint of the power system is much smaller than the immense solar arrays of today's satellites and spacecraft. Inherently the radioisotope fuel in the RTG will degrade over time, but the half-life of Pu-238 is almost 88 years, making the annual degradation around half a percent. Moving forward in today's ever-changing space environment, a TPV+RTG system provides more protection from space debris and space weapons since it already has robust shielding—enough so that these units have survived reentry and impact.

While this thesis focused mainly on a TPV+RTG system, TPVs have also been suggested for use with a space nuclear reactor system. These systems are capable of producing much more power than an RTG system, but at the cost of increased risk. Only one nuclear reactor has ever been flown in space by the United States, and thus an immense amount of research, development, and funding would be required. In the same vein, Stirling engines offer a unique alternative to conventional solar arrays as well, and as discussed in Chapter II the combination of a nuclear fission system with Stirling engines is being considered for the NASA's Fission Surface Power Project. While highly efficient, Stirling engines are not ideal for a satellite or spacecraft use since the motion of their pistons act as a torque on the space vehicle, making accurate pointing more difficult. In addition, the added weight of the Stirling engine, about 20 kg without its heat source, further degrades the benefits of incorporating them into spacecraft design [68].

Further research is needed in the proper integration of TPV cells into an RTG system to validate the assumptions made in this thesis. It was assumed that the thermocouple and TPV cell would be a one for one swap with little change to weight or radiator fin design, however reality may suggest that TPVs require a larger radiator area to properly dissipate the heat and maintain cells at an optimum temperature. Additionally, even with the significant improvements in TPV efficiencies, there is still plenty of room for advanced optimization techniques discussed in Chapter V to be implemented into the TPV cell design, to include modeling and optimizing a multi-junction TPV cell. As with any model, the fidelity of the output data is only as good as the fidelity of the input parameters. ENS Davenport noted in his thesis that many material properties, especially optical parameters for longer wavelengths of light, are difficult or impossible to find, making it difficult to produce a realistic model of a TPV cell. More research is needed in these areas to promulgate usable parameters to further this technology.

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