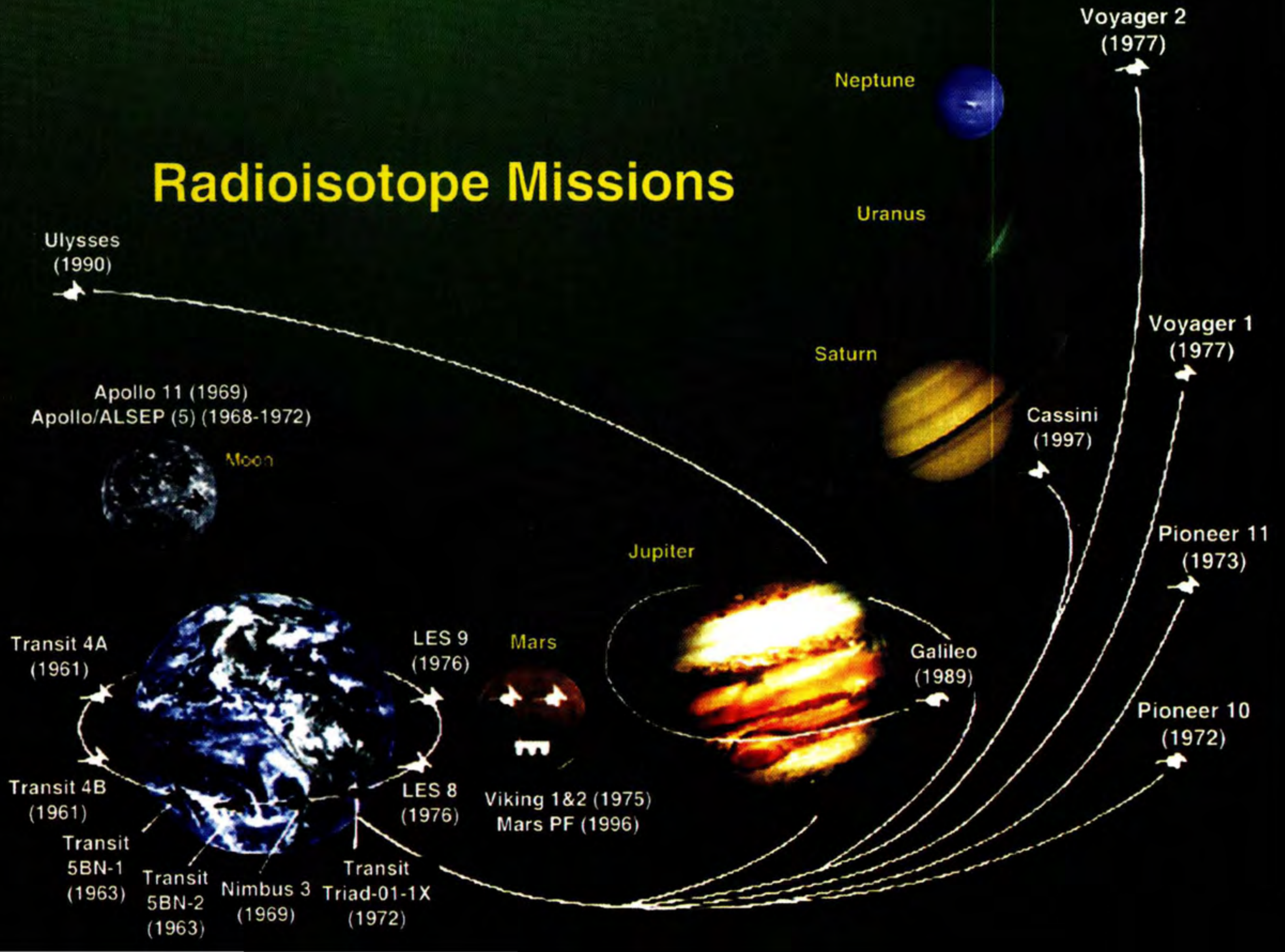
Advanced Radioisotope Power Systems Report

D-20757



Advanced Radioisotope Power System Team

2001 Technology Assessment and Recommended Roadmap for Potential NASA Deep Space Missions Beyond 2011 Report

Requested by
Joe Parrish NASA's Office of Space Science

Written by Team Members

Rao Surampudi, NASA JPL, Chairperson
Bob Carpenter, OSC (DOE support contractor)
Mohamed El-Genk UNM
Lisa Herrera, DOE
Lee Mason, NASA GRC
Jack Mondt, NASA JPL
Bill Nesmith, NASA JPL
Donald Rapp, NASA JPL
Robert Wiley, BA&H (DOE support contractor)

March 2001

The authors of this report gratefully acknowledge the receipt of presentations and written material used to prepare this report, as given in the following table:

Topic	Contributors
Mission Needs	NASA-JPL
AMTEC Technology	NASA-JPL, AMPS, DOE, and OSC
Thermoelectric Technology	NASA-JPL, Hi-Z, DOE, OSC, University of New Mexico
Stirling/Thermal Acoustic Technology	NASA-GRC, TRW, DOE/LANL
Thermionic Technology	New Mexico Energy Research Institute, DOE/ SNL, BAH
Thermal Photovoltaics	NASA-GRC

Part I - Executive Summary

Part II - Final Report

Appendices

Advanced Radioisotope Power System Team
2001 Space Technology Assessment and Recommended Roadmap for Potential NASA Code S Missions Beyond 2011

Part I - Executive Summary

Recommendation

The assessment team recommends that the Advanced Stirling Engine Converter (ASEC), Alkali Metal Thermal to Electric Converter (AMTEC) and Segmented Thermoelectric (STE) technologies be funded with detailed technical progress reviews yearly to decide whether to continue or not. Future *down selects* should be based on demonstrated technical progress instead of projections. There are uncertainties in the performance estimates of advanced radioisotope power systems (ARPS) conversion technologies, uncertainties in the future supply of Pu-238, and lack of detailed requirements of NASA missions beyond 2011 as to their sensitivity to mass, lifetime, electromagnetic interference (EMI), and vibration. Therefore, it is not appropriate at this time to down-select to one ARPS conversion technology that can fulfill the needs of most NASA deep space science and Mars missions beyond 2011.

The recommended ARPS technology roadmap (Figure ES-1) allows options to either develop Stirling 1.0 or 1.1 to flight readiness on an accelerated schedule, or to develop Stirling 2.0 roughly in parallel with AMTEC and Segmented Thermoelectrics for missions that launch beyond about 2011. Technology readiness gates are provided for each technology. A Formal Review Board would review each technology at these gates. The Board would recommend to continue the technology or not based on technical progress in meeting their requirements at the gates and the applicability to future NASA missions.

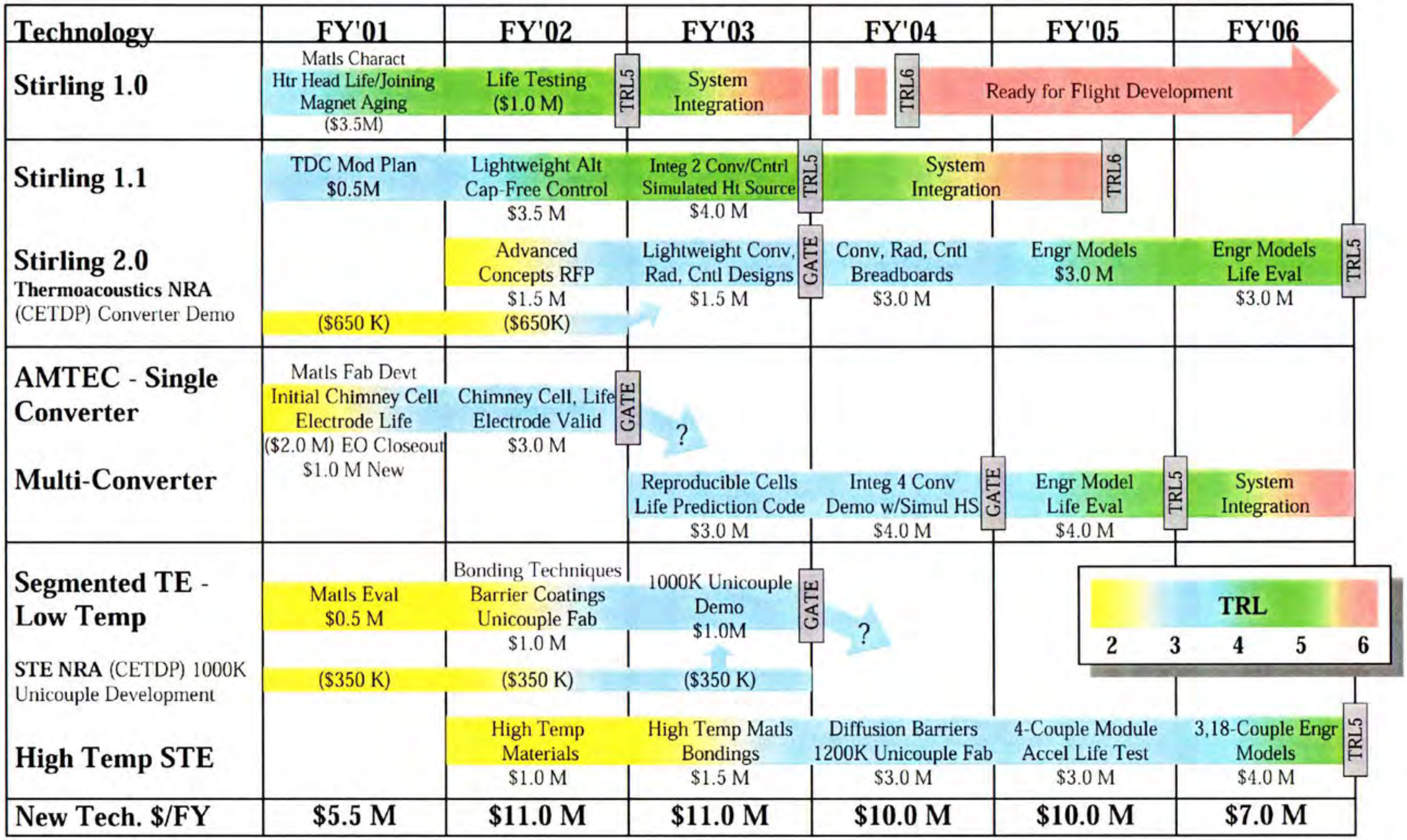
This document was prepared in late 2000. Some mission plans have changed since then. In early 2001, NASA's Office of Space Science and DOE's Office of Space and Defense Power Systems established an RPS Provisioning Strategy Team to recommend a strategy for provisioning of safe, reliable, and affordable RPSs to enable potential 2004-2011 space robotic missions. Some of the conclusions and recommendations of that team may differ from the recommendations of this report.

1.0 Introduction

NASA's Office of Space Science requested JPL to lead an assessment of advanced power source and energy storage technologies that could enable future (beyond 2006) NASA Space Science missions, and prepare technology road maps and investment strategies. The power source technologies to be reviewed are advanced radioisotope power sources, solar cells, and fuel cells. The energy storage technologies are batteries, regenerative fuel cells and flywheels. This summary report is the result of reviewing the power requirements for future NASA science missions and providing a technical assessment of the radioisotope power conversion technologies being considered for these future NASA missions.

1.1 Spacecraft Power Technology

All spacecraft require electrical power in order to accomplish their mission. Future reports will review solar cells and fuel cells. Currently, power is provided either by a photovoltaic (PV) array with batteries or by radioisotope thermoelectric generators (RTG). Over the years, the efficiency, specific power, and lifetime of PV arrays with batteries have steadily improved. PV arrays with batteries are the power source of choice for most space missions within 2 AU of the sun because of their high specific power, efficiency, and reliability.



System integration and flight development costs, EO and Mars funds, and CETDP costs NOT included in totals. Independent Review Board evaluates technology progress each year to determine whether to proceed or not.

Figure ES-1. Recommended ARPS Technology Roadmap

However, there are missions for which PV arrays with batteries are unsatisfactory. These include missions where the solar flux is too low or variable due to eclipses, or for which distances from the sun are large or changing. Examples include missions on Mars that require (a) operation in shadows, (b) extended lifetimes where seasonal variations and settling of dust on PV arrays would be deleterious, or (c) where power throughout the 24.66-hour diurnal cycle is essential.

1.2 Goals for Advanced Radioisotope Power System Technology

The system efficiency of the state-of-art (SOA) SiGe RTG is 6.5%, and a 100 watt-class SiGe RTG has a specific power of ~4.5 watts/kg. The SOA SiGe RTG has a proven lifetime greater than 20 years.

The reasons to develop ARPS technologies are:

- To increase the specific power by about a factor of 2 (9 to 10 watts/kg),
- To increase the system efficiency by a factor of 2 to 4 (13% to 25%),
- To reduce the radioisotope power system (RPS) recurring cost, and
- To reduce the RPS fabrication time from project start through delivery to the launch site.

ARPS technologies that increase both the system specific power and efficiency are expected to have the widest applicability.

2.0 Study Overview and Description

JPL established a technical assessment team that included participants from the Department of Energy (DOE), Glenn Research Center (GRC), University of New Mexico, and the Jet Propulsion Laboratory (JPL). The goal of the study was to determine investment strategies for RPS conversion technologies that have the best potential to satisfy the requirements of future deep space science and Mars missions.

2.1 Objectives

The assessment team objectives were to:

- Review NASA future mission needs for advanced radioisotope power systems,
- Assess the status and potential performance of ARPS technologies,
- Estimate resources required to advance ARPS technologies to NASA TRL 5,
- Prepare road maps for promising technologies that have the potential to satisfy future mission requirements, and
- Recommend to NASA and DOE investment strategies for developing ARPS technologies.

2.2 Approach

The system efficiency of the state-of-art SiGe RTG is 6.5% and the SiGe RTG specific power is 4.5 watts/kg. The SOA SiGe RTG has a proven long lifetime greater than 20 years. The reasons to develop ARPS technologies are:

- To increase the specific power by about a factor of 2 (9 to 10 watts/kg),
- To increase the system efficiency by a factor of 2 to 4 (13% to 25%),
- To reduce the recurring cost of RPSs, and
- To reduce the flight RPS fabrication time from project start through delivery to the launch site.

ARPS technologies that improve both the system specific power and efficiency is the highest priority. A more efficient ARPS converter reduces the amount of Pu-238 fuel and the cost for any power level. The team used a scaled-down 100-watt version of the Cassini 285-watt SiGe RTGs as a baseline against which ARPS technologies were compared. Based on most future deep space science and Mars missions, the power requirements can be met with an RPS module that is sized to deliver 100 watts electric at the beginning of mission (BOM). Some missions may require several such RPS modules. In this assessment, we assumed that an RPS module uses a specific number of GPHS modules at an assumed BOM thermal power of 240 watts/module. The number of general purpose heat source (GPHS) modules was chosen so that the BOM power output of each ARPS module is approximately 100 watts or greater.

2.3 Team Schedule

The assessment team held three face-to-face meetings. The first meeting was held at JPL in September 2000, where the assessment team received input on power requirements for NASA's future Space Science Missions. The team reviewed mission requirements for power level, mass, lifetime, vibration, EMI, etc. The team also received input from the technologists developing AMTEC, thermoelectrics, thermionics and thermoacoustic technologies as applied to ARPS. The second meeting was held at GRC in October 2000, where the team received input on Stirling and thermal photovoltaics technology and more detailed information on the thermo-acoustics technology. The third meeting was held at DOE Headquarters in Germantown to obtain a consensus on the present NASA Technology Readiness Level of each technology reviewed. Also, the most promising ARPS technologies were agreed to, and the top level road maps and estimated costs were prepared to bring these promising technologies to NASA TRL 5. Most of the work on this report was done during the Fall of 2000, and this report does not take into account programmatic changes made by NASA since then.

2.4 Criteria

The parameters used to evaluate the advanced technologies are: (1) Safety, (2) Lifetime and Fault Tolerance, (3) Specific Power (watts/kg), (4) Conversion Efficiency, (5) Applicability to meet a wide range of mission requirements, (6) Development Risk, (7) Spacecraft Interface Issues, (8) Converter/GPHS Interface Issues, and (9) Feasibility of Validating the Lifetime Performance for 15 year missions.

2.5 NASA TRL Descriptions

The team's interpretation of the NASA Technology Readiness Level scale is described in Figure ES-2. NASA TRLs 3, 4, and 5 include some demonstrated lifetime performance. NASA TRLs 1 and 2 refer to new technologies that are in the early stages of emergence. Technology development efforts for NASA space missions involve advancing a technology from NASA TRL 2, 3, or 4 to TRL 5.

TRL	Accomplishment
1-2	Concept and application are formulated. Basic phenomena are observed in a laboratory environment.
3	Critical functions are tested in a laboratory environment of breadboard configuration to validate proof-of-concept's potential performance and lifetime.
4	A breadboard system (or at least the major components of the system) is tested in the laboratory and verify that components will work together effectively in a system. At this level, preliminary analytical and experimentally efforts are made to calculate lifetime performance of critical components.
5	A realistic breadboard portion of the system is thoroughly tested in a relevant environment that demonstrates the flight system design. Lifetime performance predictions of critical components are validated with accelerated tests.
6	System engineering model with approximate "form, fit, and function" of a flight systems or prototype demonstration tested in a relevant environment on ground or in space. System lifetime performance prediction validated based on accelerated life testing of components and subsystems.

Figure ES-2. The NASA Technology Readiness Levels for ARPS

When a technology reaches the NASA TRL 5, successful prototype flight configuration and accelerated lifetime tests of major subassemblies and components have been tested on the ground in a relevant space environment. Subsequently, the technology is transferred to a system integrator to develop a flight RPS that meets specific space science mission requirements. NASA-JPL projects require that a technology reach TRL 6 by spacecraft preliminary design review (PDR).

The ARPS assessment team estimated the resources needed to advance selected candidate ARPS from TRL 2, 3, or 4 to TRL 5. The team qualitatively described the development risk that the technology may not reach TRL 5 with the estimated resources, as high, medium, or low. A critical element of technology development is assessing the lifetime of the ARPS. This requires accelerated testing of components, subassemblies and systems to validate lifetime prediction codes. In-depth analysis of failure modes and accelerated tests are required to validate ARPS lifetime performance prior to launch.

3.0 Current Technologies for Radioisotope Power Systems

The present state-of-the-art radioisotope power systems is the SiGe RTG. The SOA SiGe RTG includes a radioisotope heat source and a thermoelectric heat-to-electrical power converter. Radioisotope heat sources and thermoelectric converters have played an important role in NASA's space exploration missions.

The present radioisotope fuel state-of-the-art is a space-qualified GPHS that can be used as a single module or in multiples to supply heat at temperatures up to 1373 K for future RPS. The GPHS module has the physical form of a rectangular block approximately 9.7 cm x 9.3 cm x 5.3 cm. Each module has a mass of 1.445 kg, which includes 440 grams of plutonium 238. With fresh plutonium 238 radioisotopes, a module produces ~240 W of thermal power that decays with an 87.74-year half-life. The GPHS module can be operated in a vacuum or an inert cover gas environment. For outer planet missions the GPHS is operated in a vacuum, and for Mars or other planetary atmospheres the GPHS modules have to be sealed to maintain a vacuum or an inert gas atmosphere. Disposing of the He produced by radioactive decay in a sealed heat source poses a challenge for the thermal insulation.

Space-proven RTGs have used two different unicouple materials, SiGe and PbTe-TAGS. The PbTe-TAGS generator requires an inert gas pressure over the PbTe-TAGS to inhibit sublimation and does not require venting the helium to space. The helium is produced from the plutonium radioactive decay process and vented from the GPHS module. A 100-watt PbTe-TAGS RTG would have an estimated specific power of about 3 W/kg and a system efficiency of 6.2%. The Viking Landers on Mars used a smaller PbTe-TAGS RTG with an inert cover gas. A 100 watt SiGe RTG is expected to have a specific power of about 4.5 W/kg and a system conversion efficiency of 6.5%.

4.0 Missions Requiring RPSs

4.1 Previous NASA Missions

The Pioneer 10 and 11 missions and the Viking Mars Landers used PbTe-TAGS RTGs, each providing about 40 watts per generator at specific power at about 3 watts/kg. The RTGs deployed on the moon by the Apollo 12, 14, 15, 16, and 17 missions used PbTe thermoelectrics and produced about 70 watts per generator. The Voyager mission used Multi-Hundred Watt (MHW) SiGe RTGs. Each Voyager spacecraft has three MHWs. Each MHW produced 158 watts at BOM and are still working after 22 years. Galileo, Ulysses, and Cassini missions all use SiGe GPHS RTGs. Each GPHS RTG produced 285 watts at BOM. Two GPHS RTGs power Galileo. Ulysses has one GPHS RTG. Three GPHS RTGs power Cassini. The Voyager, Galileo, Ulysses, and Cassini missions are still active and the RTG power source performance has been as predicted.

The SiGe RTG uses multi-foil thermal insulation in vacuum for high thermal efficiency between the heat source and the radiator. The helium produced by the plutonium decay is vented directly to space. Therefore, the SOA SiGe RTG thermal insulation design would have to be changed to operate efficiently on Mars or any other planetary body with an atmosphere.

4.2 Future NASA Missions

Spacecraft within about 2 AU of the sun use solar photovoltaics power with the sun as the energy source. Solar cells are proven to be very reliable. Missions to the surface of Mars require special consideration because Mars is close enough to the sun to allow use of solar cell power. The disadvantages of solar cell power on Mars are as follows:

1) Power is only available roughly one-third of a diurnal cycle,
2) Power availability varies widely with season at higher latitudes,
3) The lander or rover requires long-life rechargeable batteries for overnight survival,
4) Solar cell power does not function in canyon shadows, and
5) The power output gradually deteriorates due to dust accumulation on the solar cells.

Short-term Mars surface missions can use direct solar cell power. However, some form of dust mitigation may be needed for long-term ambitious scientific surface missions planned for 2007 and beyond. Missions beyond 2 AU and long duration Mars surface missions may require radioisotope power systems.

The NASA Mars Exploration Program Office (MEP) enterprise's future plans include the possibility of a Mars surface mission for 2007, a Mars sample return for 2011, and Mars surface lander missions every four years beyond 2011. It is possible to carry out limited science versions of these surface missions using solar cell power. However, the potential use of RPSs would provide the longevity and versatility required to accomplish ambitious scientific objectives for these missions.

NASA Exploration of the Solar System (ESS) enterprise's future plans includes the possibility of a Europa Orbiter, Pluto-Kuiper Express, Europa Lander, Titan Explorer and Neptune/Triton Orbiter missions. These are 6 to 15 year missions far from the sun that appear to require radioisotope power systems.

The NASA Sun-Earth Connection (SEC) enterprise's plans include the possibility of a Solar Probe, Interstellar Probe, Interstellar Trailblazer, and the Outer Heliosphere Radio Imager missions, each of which would likely require radioisotope power sources. These missions are in the early stages of planning and the projected power levels are 200 to 300 watts, with desired lifetimes of up to 30 years.

It is planned that between 2011 and 2022, up to six space missions may be launched, each potentially requiring at least 200 watts of electric power from an RPS. If two 100 We SiGe RTGs were to be used for each of these missions they would require 14 GPHS modules per mission. The cost of each GPHS module at current Pu-238 prices is about $1 M, that is $14 M for a 200 watt SiGe RTG powered mission. Six missions would require about $84 M for the GPHS modules. High-efficiency ARPS have the potential to reduce these costs. For example, an advanced Stirling ARPS would require four GPHS modules per 200-watt mission for a total GPHS module cost of $24 M for six missions. Therefore, the GPHS module cost for six 200 watt advanced Stirling ARPS powered missions would be $60 M less than six 200 watt SiGe RTG powered missions. An advanced S-TE or AMTEC ARPS would require 8 GPHS modules per 200-watt mission, that is $8 M per mission and $48 M for six 200 watt missions for a cost reduction of $36 M.

4.3 Fuel Availability

Radioisotope heat source availability is limited by the quantity of fuel that can be produced and processed or which is already in the inventory. Pu-238 had been produced in U.S. reactors and processing facilities at the DOE's Savannah River Site (SRS). With the shutdown of the last production reactor at the SRS, the U.S. no longer has a Pu-238 production capability.

There is sufficient Pu-238 on hand for about 18 GPHS modules, or the equivalence of one GPHS RTG. The DOE has a contract with Russia to procure up to 5 kg of Pu-238 every 6 months through 2002. Therefore, the DOE can purchase up to 10 kg in 2001 and another 10 kg in 2002 assuming a purchase is made every 6 months starting in January 2001. The DOE believe that additional Pu-238 can be purchased from Russia if needed.

The DOE recently completed a programmatic environmental impact statement (EIS) which includes the evaluation of alternatives for re-establishing a domestic Pu-238 production and processing capability. In its Record of Decision, the DOE announced its decision to produce Pu-238 at the Advanced Test Reactor in Idaho and at the High Flux Isotope Reactor in Tennessee.

The future availability of Pu-238 will have a significant effect on prioritization of ARPS technologies. If it is anticipated that we can still purchase Pu-238 from Russia beyond 2002 or re-establish domestic production facilities, then specific power and lifetime will be the most vital governing characteristics of ARPS technologies. If, on the other hand, it is possible that supplies beyond 2002 will be difficult to obtain, conversion efficiency is likely to be the most important characteristic.

5.0 Advanced Technologies Evaluated

Technologies evaluated in this study are listed in Table ES-1 and described after the table. Advanced Stirling Engine Converter (ASEC), Alkali Metal Thermal to Electric (AMTEC), Segmented Thermoelectric (STE), Low Temperature Thermionic (LTI), Thermoacoustics (TA) and Thermal Photovoltaics (TPV) technologies as applied to ARPS were reviewed and evaluated as to their potential to satisfy requirements of future NASA space missions. Estimates of system masses and efficiencies were made for each technology and compared to a scaled-down design of a 100-watt SiGe RTG.

Table ES-1. Technologies Evaluated

Technology	Specific Technology	Comments
Thermoelectric	SiGe RTGs	Used on Voyager, Ulysses, Galileo, and Cassini Missions.
	PbTe-TAGS	Used on Viking and Pioneer Missions.
Stirling Engine Converter	Version 1.0	Present design; efficiency is very good but mass is high; lifetime is not certain; most mature of the ARPS technologies; only one with a chance of being made ready for Mars 2007.
	Version 1.1	Make several near-term improvements to reduce mass of Version 1.0.
Advanced Stirling Engine Converter	Version 2.0 or Thermoacoustic	Low Mass Stirling Engine, Alternator, Radiator and Controller Potential for high efficiency long life Stirling Engine. Considered as an advanced form of Stirling Technology.
AMTEC	Refractory Metal Chimney	Potential for Low Mass and Medium Efficiency. Being developed under existing DOE contract.
Segmented Thermoelectric	Advanced Materials/ Segmented Unicouple	Potential for low mass and medium efficiency. Solid state device configuration, operations and handling are similar to SiGe RTG.
Thermionic	Cesiated triode	Low Temperature 1300 K Thermionic Early Stage of technology development.
	Microminiature	Early stage of research.
Thermo-photovoltaic	Thermal Photovoltaic Cell	Early stage of research.

Advanced Stirling Engine Converter (ASEC)-ARPS technology development approach uses a reciprocating free-piston Stirling heat engine or a Thermoacoustic heat engine with a linear alternator that is a low-mass version of the Stirling engine alternator now under development by DOE and NASA. The ASEC-ARPS has the principal advantage of almost four times the system efficiency of the SOA SiGe RTG. However, near-term versions of the ASEC-ARPS approach have specific power that is essentially the same as the SiGe RTG.

The ASEC-ARPS major technical issues are:

1) Validating the system lifetime for 6 to 15 year missions,
2) Developing an efficient, low-mass, highly reliable long life controller,
3) Reducing the EMI for space missions that measure very small magnetic fields, and
4) Reducing the Stirling engine alternator vibration for very sensitive seismic instruments.

Alkali Metal Thermal to Electric Converter (AMTEC)-ARPS produces electric power by the pressure-induced flow of sodium ions through a beta-alumina solid electrolyte (BASE) which produces dc current and voltage. AMTEC delivers dc power with no vibration and very small EMI. AMTEC is a young technology with potential system efficiency perhaps as high as 20%, which is three times the system efficiency of the SOA SiGe RTG. AMTEC ARPS has the potential for doubling the specific power of the SiGe RTG to 9 watts/kg.

The AMTEC-ARPS major technical issues are:

1) A BASE to metal ceramic seal,
2) Converter containment material fabrication process,
3) A reproducible wick-evaporator fabrication process, and
4) Electrical feed-through fabrication process.

Segmented-Thermoelectric (STE)-ARPS contains thermoelectric materials that produce a current and a voltage when placed in a temperature gradient. Each thermoelectric material, whether n-type or p-type, exhibits a maximum figure-of-merit at some temperature. If a single material is used in each leg of the unicouple, the effective efficiency will be an average over the entire temperature range, which is less than the maximum. If each leg of the unicouple is segmented, the temperature gradient over each segment will be small. Thermoelectric materials developed with a high efficiency over the temperature range for each segment will achieve a higher overall efficiency over the entire temperature range. This may provide an ARPS with double the efficiency of the SiGe RTG.

The major technical issues are:

1) Developing a compatible high temperature (973 K to 1273 K) thermoelectric material
2) Developing joints between the segments with very small thermal and electric resistance
3) Developing barriers that prevent inter-diffusion between segments
4) Developing joints between the high-temperature thermoelectric materials and a hot shoe.

The team assessments of these advanced converter technologies for ARPS are as follows.

The ASEC-ARPS is the most mature of the various technologies and has the highest conversion efficiency. The near term SEC-ARPS option (Version 1.0) has about the same specific power as a SOA SiGe-RTG. Advanced ASEC-ARPS systems offer improvement in specific power. The ASEC-ARPS option may be appropriate for larger Mars surface missions, but may not meet the lifetime, EMI and vibration requirements for some outer planet and SEC missions. Thermoacoustic Stirling engine technology may offer less vibration and longer lifetime over conventional Stirling engines but it is at an early stage of development.

AMTEC-ARPS has the potential for higher specific power and efficiency than SOA SiGe RTG. The lifetime of AMTEC ARPS basic conversion components (BASE, electrodes and current collectors) have demonstrated to be greater than 20 years. It is planned that accelerated testing will validate the lifetime performance of components and the converter. There are no EMI or vibration problems. This technology is attractive for missions where low mass and long life are requirements.

STE-ARPS has the potential for considerably higher specific power and efficiency than SOA SiGe RTG. STE-ARPS converters (unicouple) are amenable to accelerated lifetime testing as they are being developed. The STE-ARPS lifetime is well known once the converter lifetimes have been validated and there are no EMI or vibration problems. For missions where low mass and long life are of paramount importance, this technology is attractive.

Low-temperature thermionics and thermal photovoltaics technologies are at NASA TRL 1 and 2. Therefore, the team could not assess the efficiency, specific mass or lifetime for an ARPS. The team recommends NASA's crosscutting technology program, the DOE's Provisional Research and Development Agreement (PRDA) program and Small Business Innovative Research (SBIRs) or Small Business Technology Transfers (STTRs) fund these two technologies until technical feasibility is demonstrated.

6.0 Results and Recommendations

The relationship between the performance and readiness of the three strongest ARPS technologies is illustrated in Figure ES-2. RTG technology is at the lower left of the diagram with low specific power and low efficiency. It is desired to move toward the upper right of the diagram representing high efficiency and high specific power. However, the readiness levels are lowest in this region.

The assessment team recommends that ASEC, AMTEC, and STE technologies be funded and developed by NASA in accordance with a technology plan that includes technology readiness gates. The progress towards meeting these technologies readiness gates for each technology should be reviewed yearly by the same independent review board. One or two of these technologies should be selected in about two years on the basis that it meets the technology gates and the requirements for the greatest number of future NASA ESS, SEC, and MEP missions. The selected technology should then be developed to TRL 5 under a technology program. When the technology reaches NASA TRL 5, a NASA flight project would presumably develop the technology to TRL 6 to meet the project specific requirements in parallel with the spacecraft preliminary design.

Table ES-2 provides data for the selected candidate technologies: ASEC, AMTEC and STE. In each case, a GPHS module was assumed to deliver 240 watts thermal at BOM and the number of modules was chosen to make the BOM power 100 watts electric or greater. Thermo-acoustic technology is considered as part of the Stirling 2.0 technology in Table ES-2.

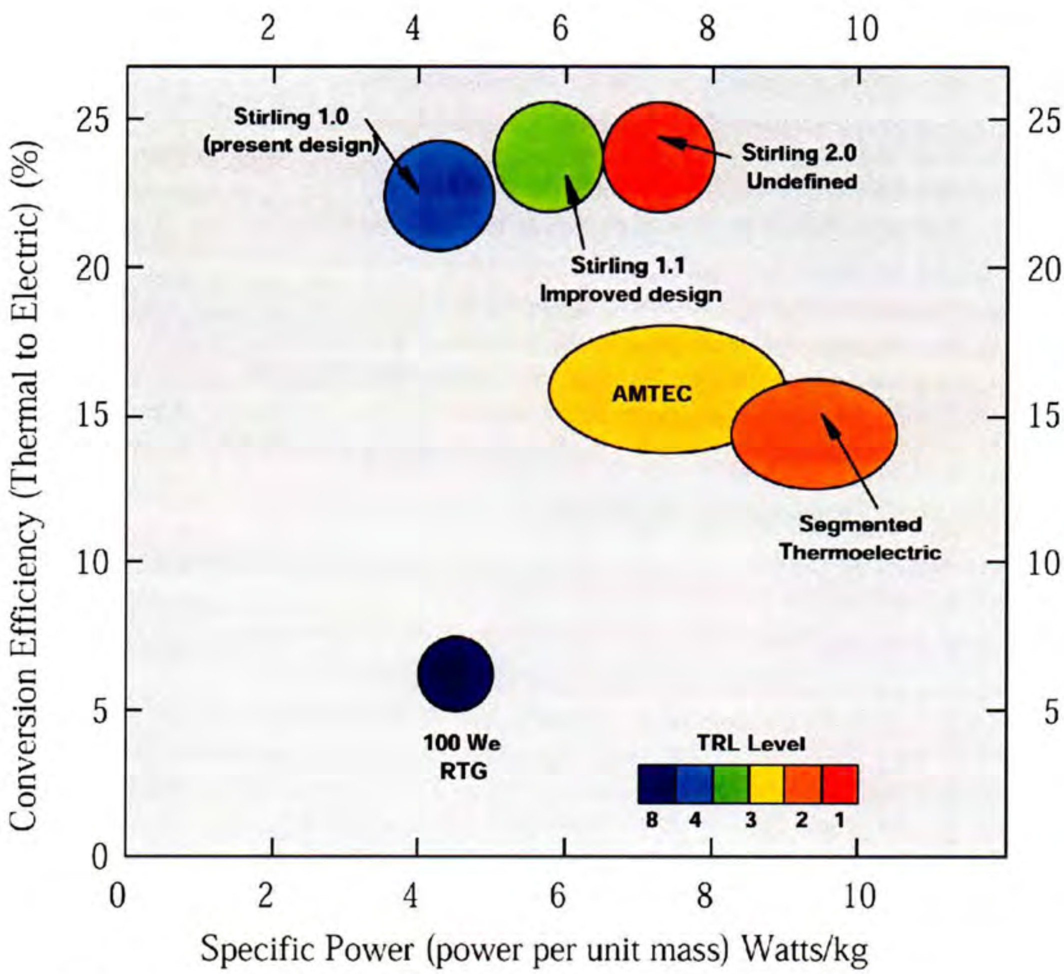


Figure ES-2. Comparison of Conversion Efficiency and Specific Power

6.1 Road Map and Technology Plan for Promising Technologies

A Technology Plan was developed for the most promising technologies based on the starting NASA TRL and the major technical issues remaining for each technology.

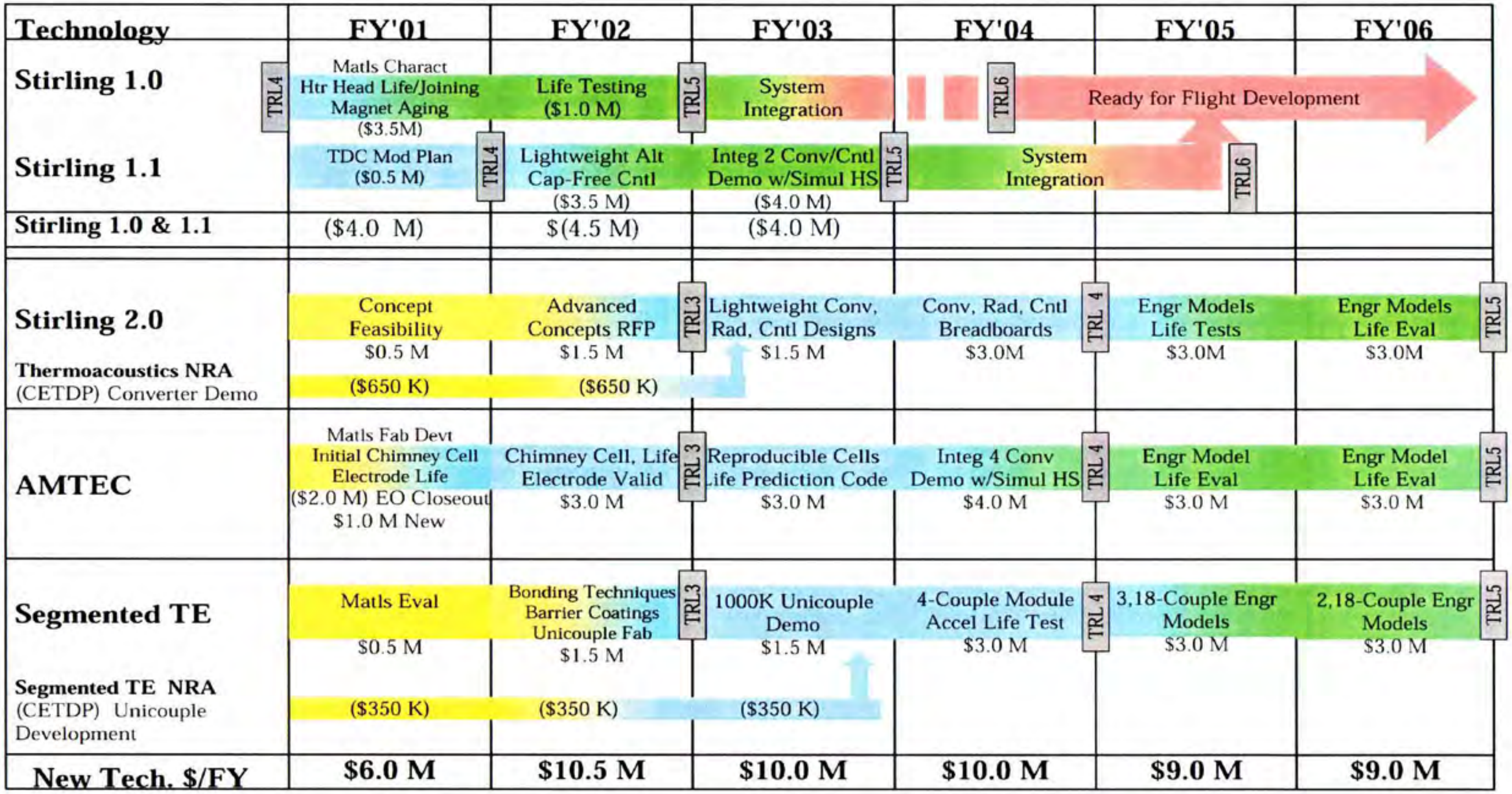
The recommended ARPS technology roadmap (Figure ES-1) allows options to either develop Stirling 1.0 or 1.1 to flight readiness on an accelerated schedule, or to develop Stirling 2.0 roughly in parallel with AMTEC and Segmented Thermoelectrics for missions that launch beyond about 2011. This roadmap is not a rigid plan to be followed regardless of further developments. It is a framework that defines our best estimate at this time of what is appropriate in the future. As progress is made in some areas, and disappointments are found in others, funds can be transferred between technologies to develop the most promising technologies at the fastest possible rate. A systems engineering team led by the DOE with support from JPL and GRC should be funded to prepare ARPS concept designs for each conversion technology. These ARPS system concept designs would be prepared to satisfy NASA mission requirements. The concept designs would also be used to help establish the technology readiness gates and direct the conversion technology development program.

An alternative roadmap that provides the same rate of progress for Stirling 2.0, AMTEC, and STE technologies is shown in Figure ES-3.

Table ES-2. Characteristics of Candidate ARPS Technologies

Technology	NASA TRL	BOM Watts	System Mass kg	Spec Pwr W/kg	Sys Eff.%	GPHS Mod-ules (d)	Reqts for TRL5	Dev't Risk	Life Issues	S/C IF Issues (e)	Resiliency to Partial Failure
Small SiGe RTG (a)	8	139	31.2	4.5	6.5%	9	None	None	None	None	Highly modular
Stirling 1.0 (b)	4	110	27	4.1	23%	2	$4.5M-2yr	Low	Engine & Control Electronics Helium leakage	AC/DC Control electronics Radiator Vib, EMI	Failure of one converter may lead to generator failure
Stirling 1.1	4	120	20	6.0	25%	2	$8M-3yr	Med			
Stirling 2.0	2	120	16	7.5	25%	2	$12M-5yr	High			
AMTEC (LMA) (c)	3	139	25	5.6	14.5%	4		High	Seals, wick Evaporator	Launch vehicle	Failure of converter results in generator partial power loss
AMTEC Chimney	3	120	13.6	8.8	16.7%	3	$15M-5yr	High	Containment, matls, fab Process	acceleration	
Low T Segmented TE	2	125	14	8.9	13%	4		High	Joint bonding, barriers	None	Highly modular
High T Segmented TE	2	144	14	10.2	15%	4	$15M-6yr	Higher	New Material Joint bonding Barriers	None	

(a) LMA 9 GPHS Vacuum RTG concept for Europa Orbiter '04 or '05 launch.
(b) LMA Stirling RPS study concept for Europa Orbiter.
(c) LMA AMTEC preliminary design for Europa Orbiter.
(d) Each GPHS module assumed at 240 thermal watts at BOM.
(e) Potential spacecraft interface issues with some missions.



Notes: Stirling 1.0 and 1.1, System Integration and Flight Development costs, and CETDP costs not included in totals. Independent Review Board evaluates technology progress TRL 3, 4, and 5 to determine to proceed or not.

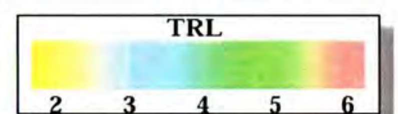


Figure ES-3. Alternate Roadmap with Parallel Development of Major Technologies

Advanced Radioisotope Power System Team
2001Technology Assessment and Recommended Roadmap
for Potential NASA Code S Missions Beyond 2007

Part II - Final Report

1.0 Introduction

NASA's Office of Space Science requested JPL to lead an assessment of advanced power source and energy storage technologies that will enable future (beyond 2006) NASA Space Science missions and prepare technology road maps and investment strategies. The power source technologies to be reviewed are advanced radioisotope power sources (ARPS), solar cells, and fuel cells. The energy storage technologies are batteries, regenerative fuel cells and flywheels. This summary report is the result of reviewing the power requirements for future NASA science missions and providing a technical assessment of the radioisotope power conversion technologies for these future NASA missions. Reports on solar cells and fuel cells are also being developed.

1.1 Spacecraft Power Technology

All spacecraft require electrical power in order to accomplish their mission. Power is provided either by a photovoltaic (PV) array with batteries or by radioisotope thermoelectric generators (RTG). Over the years the efficiency, specific power and lifetime of PV arrays with batteries have steadily improved. PV arrays with batteries are the power source of choice for most space missions within 2 AU of the sun because of their high specific power, efficiency and reliability.

However, there are missions for which PV arrays with batteries are unsatisfactory. This includes missions where the solar flux is too low or variable due to eclipse, or where distances from the sun are large or changing. Examples include missions on Mars that require (a) operation in shadows, (b) extended lifetimes where seasonal variations and settling of dust on PV arrays would be deleterious, or (c) where power throughout the 24.66-hour diurnal cycle is essential.

Missions such as Solar Probe would present unique challenges to PV arrays due to the 75,000/1-ratio change in solar flux from 5 AU at Jupiter to 4 solar radii from the center of the sun. RTGs have performed as designed for Mars missions and beyond and for Ulysses, a Solar Polar mission.

Thermoelectric converters that convert heat directly to electrical power have played an important role in NASA's solar system exploration missions. Since 1961, the United States has launched 42 RTGs on 25 spacecraft for various NASA and the Department of Defense (DOD) missions. These missions include Earth orbits, the surfaces of the Moon and Mars, missions to the Sun, fly-by missions to all the Solar System outer planets except Pluto, and orbiter missions of Jupiter and Saturn.

An RTG consists of a radioisotope heat source, a converter that converts heat to electricity, and a radiator to reject waste heat to space. The DOE has developed a modular general purpose heat source (GPHS) as the building block for supplying heat for radioisotope power systems (RPS)s. The GPHS module is a rectangular block approximately 9.7 cm x 9.3 cm x 5.3 cm. Each module's mass is 1.445 kg and includes 440 g of Plutonium 238 Dioxide (PuO_2). Each module produces ~240 watts of thermal power at beginning of life (BOL) and the Pu-238 decays with a 87.8 year half-life. The GPHS module is designed to operate in a vacuum or in an inert atmosphere. The space vacuum design is used for outer planet missions where the helium produced by the Pu-238 decay is vented to space. For use on Mars or in other planetary atmospheres the GPHS modules and converter would have to be in a sealed vacuum or an inert gas environment. Disposing of the helium released from the Pu-238 in a sealed environment would require a different thermal insulation and helium gas vent design than the space vacuum design. Further description of the GPHS is given in Appendix E.

Proven RTG converter technology uses SiGe or PbTe-TAGS thermoelectric unicouples: Both are space proven. The PbTe-TAGS converter has lower specific power (~3 watts/kg) lower heat rejection temperature, (i.e., larger radiator area) and requires an inert gas pressure over the PbTe-TAGS to inhibit sublimation. The PbTe-TAGS converter design with the GPHS modules sealed and no venting of the helium gas required is applicable to Mars. The SiGe RTG converter has higher specific power (5.1 W/kg at 285 watts), smaller radiator area, slightly higher conversion efficiency (6.5% vs. 6.2%), and the helium gas is vented directly to space vacuum. The Viking Landers on Mars used the PbTe-TAGS converter where venting of helium was not required. Since Viking, all NASA Code S missions have used the SiGe RTGs for deep space missions on launches from 1977 to 2000. The SiGe RTG technology is the current SOA technology because of the extensive experience with SiGe RTGs from 1976 to 2000. However, PbTe-TAGS technology was very successful on Viking and Pioneer and is an option for future NASA space missions.

1.2 Reasons for Advanced Radioisotope Power System (ARPS) Technology

The system efficiency of the state-of-art (SOA) SiGe RTG is 6.5% and a 100 watt-class SiGe RTG specific power is ~4.5 watts/kg. The SOA SiGe RTG has a proven lifetime greater than 20 years.

The reasons to develop ARPS technologies are:

- To increase the specific power by about a factor of 2 (9 to 10 watts/kg),
- To increase the system efficiency by a factor of 2 to 4 (13% to 25%),
- To reduce the RPS recurring cost, and
- To reduce the RPS fabrication time from project start through delivery to the launch site.

ARPS technologies that increase both the system specific power and efficiency is the highest priority. A more efficient ARPS converter reduces the amount of Pu-238 fuel and the cost for any given power level. The team used a scaled-down 100-watt design of the Cassini 285-watt SiGe RTG as a baseline against which ARPS technologies were compared. Based on the potential set of future deep space science and Mars missions, the power requirements could be met with an ARPS module that is sized to deliver 100 watts electric at the beginning of mission (BOM). Deep space and Mars missions may require several such ARPS modules. The assessment team assumed that an ARPS uses a specific number of GPHS modules at an assumed BOM thermal power of 240 watts/module. The number of GPHS modules was chosen so that each ARPS technology BOM power output is 100 watts or greater.

2.0 Study Overview and Description

2.1 Objectives

The purpose of this study is to recommend to NASA investment strategies for ARPS technology to assure that radioisotope power systems will be available for future NASA solar system exploration missions. While photovoltaic/battery power systems are used where feasible, there are many proposed missions where an ARPS will be required. Advanced radioisotope power system (ARPS) technologies have the potential to increase the specific mass by a factor of 2 and increase the efficiency by factors of 2 to 4 over the SOA SiGe RTG.

A study team was selected to assess the performance and development risks and estimate resources required to develop the most promising advanced thermal-to-electrical conversion technologies for ARPS. The evaluation of candidate technologies was made on the basis of the performance potential of the technology, the present state of readiness, how well its technical challenges are defined and how well it appears to fulfill future mission needs. The assessment goal is to provide NASA with insight and understanding on each advanced technology so NASA can make good decisions on which ARPS technologies to develop for its future missions.

2.2 Approach

The study approach was to select a technical assessment team of knowledgeable space power experts to gather information, discuss in detail the technical data, draw conclusions, make recommendations, and document the results in a report. The team conducted three two-day meetings to obtain mission requirements for ARPS, assess the technical status of each potential converter technology, select the promising advanced technologies, and prepare roadmaps including estimated resources for developing the most promising technologies. NASA mission planners presented future mission plans and the requirements for ARPS for the NASA Solar System Exploration, Sun Earth Connection and Mars Exploration programs. Experts from each technology area presented technical information to the assessment team.

After reviewing NASA future mission plans, a 100 watt-class ARPS module was selected as the power level best suited to meet most future mission requirements. Therefore, the 100 watt-class ARPS was selected by the team for evaluating and comparing future potential ARPS technologies. There are some potential planet surface missions that may require 30 to 40 milliwatts ARPS. The team assessed this power level, but to a much lesser extent.

The assessment team examined each ARPS technology to answer the following questions:

- What is the conversion process and how does it work?
- What is the present status of the technology?
- What programs are presently funded?
- What is the potential of the technology in terms of specific power, efficiency, and lifetime?
- What technical challenges remain, and are they well defined?
- What resources are needed to advance the technology to NASA TRL 5?

2.3 Schedule

The assessment team conducted three two-day meetings. The first meeting was held at JPL in September 2000, where the assessment team received input on power requirements for NASA's future space science missions. The team reviewed each mission for power level, mass, lifetime, vibration, EMI, etc. requirements. Also at the first meeting, the team received input from the technologists developing AMTEC, Thermoelectric, Thermionic and Thermo-Acoustic technologies as applied to Advanced Radioisotope Power Systems (ARPS)s.

The second meeting was held at GRC in October 2000, where the team received input on Stirling and Thermal Photovoltaics technology and more detailed information on the Thermoacoustic technology. The third meeting was held at DOE Headquarters in Germantown to obtain a consensus on the present NASA Technology Readiness Level (TRL) of each technology reviewed. The promising ARPS technologies were also agreed to, and the top level road maps with estimated resources were prepared to bring the promising technologies to NASA TRL 5.

2.4 Assessment Team

The assessment team members were:

Rao Surampudi, NASA JPL, Chairperson; Bob Carpenter, OSC (DOE support contractor);

Mohamed El-Genk, UNM; Lisa Herrera, DOE; Lee Mason, NASA GRC;

Jack Mondt, NASA JPL; Bill Nesmith, NASA JPL; Donald Rapp, NASA JPL; and

Robert Wiley, BA&H (DOE support contractor).

2.5 Criteria

The parameters used to evaluate the advanced technologies were:

- Safety
- Lifetime and Fault Tolerance
- Specific Power
- Conversion Efficiency
- Applicability to meet a wide range of mission requirements
- Development Risk
- Spacecraft Interface Issues
- Converter/GPHS Interfaces Issues
- Feasibility of Validating the Lifetime Performance for 15 year missions

The criteria that the assessment team used for the ARPS technologies are described below.

Safety—Any ARPS must meet all NASA and DOE nuclear safety requirements.

Lifetime and Fault Tolerance—Future NASA ESS and SEC missions require lifetimes of 6 to 30 years. Lifetime performance prediction codes should be validated by experimental tests based on reliable and accurate accelerated component, subsystem, and system tests. The system must be able to survive a single point failure and still meet the EOM mission power requirements.

Specific Power—High specific power of the ARPS reduces the mass of the power system. Mass is always a premium for space missions, and is especially so for outer planet missions.

Applicability to meet a wide range of mission requirements—The cost to develop any ARPS technology to flight readiness is high. Therefore the developed ARPS technology must be applicable to as many missions as possible. ARPS technologies can be developed at different power levels, but at each power level the system must go through a lengthy and costly design-test-qualification procedure. Missions require different power levels. An economical approach is to develop the 100 We ARPS module for low power missions and use multiple ARPS modules for high power missions.

Conversion Efficiency—High conversion efficiency reduces the amount of radioisotope materials needed for a given electric power. This reduces system mass and cost. It also reduces the risk that radioisotopes may not be available.

Development Risk—The assessment of technology development risk depends on the technical challenges and how well they are known and defined. The team used a qualitative risk assessment of low, medium or high based on how well the technology challenges are defined, and whether the suggested approaches appear feasible.

Spacecraft Interface Issues—These include but are not limited to the following:

a. Mechanical Interface: Volume and radiator area of ARPS determines ease of integrating with the spacecraft. Vibration introduced by dynamic machinery must be controlled to meet instrument requirements. Field of view required by the power system radiator can be in direct competition with science instruments.

b. Thermal Interface: Waste heat provided by the ARPS to the spacecraft is a major benefit for all planetary missions.

c. Electrical Interface: Power conditioning and control is required to match the converter output (ac or dc) to the user's needs. EMI must be controlled to meet instrument requirements.

Converter/GPHS Interfaces Issues—Efficient thermal coupling is required from the radioisotope heat source to the converter and from the converter to the heat rejection radiator to reduce mass and radiator area.

Feasibility of Validating the Lifetime Performance for Long-Duration Missions

Determining applicable acceleration factors for technologies are difficult. Some technologies can be accelerated at the critical component level to provide confidence that the technology has long life. Other technologies must be accelerated at the subsystem or system level only and confidence in the lifetime of the technology is not reached until late in the development.

2.6 NASA TRL Description

The team's description of the NASA TRL used to evaluate ARPS technologies is given in Figure 2.6-1. The assessment team used the NASA Technology Readiness Level (TRL) to categorize the technical maturity of each technology. The technology development funded by NASA Space Science Office involves advancing an ARPS technology from TRL 2/3 to TRL 5. When an ARPS technology reaches TRL 5, enough progress has been demonstrated that it can be turned over to a system integrator for further development by a flight project for a specific mission. The team estimated the time and dollars needed to advance promising ARPS technologies from TRL 2, 3, or 4 to TRL 5. The team qualitatively assigned low, medium, or high development risk as the probability that the technology might not reach TRL 5 with the estimated resources.

JPL projects require that a technology reach TRL 6 before it can be used on a flight system. The transition from TRL 5 to TRL 6 can be done by the system integrator for a project but must be completed by preliminary design review (PDR).

TRL	Accomplishment
1-2	Concept and application are formulated. Basic phenomena are observed in a laboratory environment.
3	Critical functions are tested in a laboratory environment of breadboard configuration to validate proof-of-concept's potential performance and lifetime.
4	A breadboard system (or at least all the major components of the system) are tested in the laboratory and verify that components will work together effectively in a system. At this level, preliminary analytical and experimentally data is available to calculate lifetime performance of critical components.
5	A realistic breadboard portion of the system is thoroughly tested in a relevant environment that demonstrates the flight system design. Lifetime performance predictions of critical components are validated with accelerated tests.
6	System engineering model with approximate "form, fit, and function" of a flight systems or prototype demonstration tested in a relevant environment on ground or in space. System lifetime performance prediction validated based on accelerated life testing of components and subsystems.

Figure 2.6-1. The NASA Technology Readiness Scale

3.0 Current Technologies Used on Previous Missions

3.1 Thermoelectric Materials

Three classes of thermoelectric materials have been used successfully in RTGs, which are suitable for low, medium, and high temperature operation. These temperature limits are based primarily on the figure-of-merit and vapor pressure (or melting point) of the thermoelectric materials. Mechanical strength, dopant migration, coefficient of thermal expansion (CTE), and compatibility with hot/cold shoe materials must also be considered.

3.1.1 Bismuth Telluride

Bismuth Telluride (BiTe) is limited to a hot-junction temperature of ~250°C (523 K). To maximize its efficiency, the cold-junction must be as low as possible ~0°C (273 K). Because of its relatively high figure-of-merit over this temperature range, the thermoelectric efficiency is ~5%. The radiator area would be large and massive to radiate the waste heat in space at such a low cold-side temperature for power levels of 10 watts or greater. Therefore, BiTe RTGs for space use are of most interest in the low watt power range or on the surface of outer planets where the temperature is very low.

3.1.2 Lead Telluride/Tellurides of Antimony, Germanium, Silver

Tellurides of Antimony, Germanium, Silver (TAGS), a p-type material, must be used with an n-type Lead Telluride (PbTe), either alone, or segmented with PbSnTe on the hot end. The PbTe-TAGS material has an intermediate figure-of-merit. This telluride couple is limited to a hot-junction temperature of ~550°C (823 K). Thermoelectric efficiencies of up to 7 to 8% can be achieved with a hot-junction temperature of 550°C (823 K) and a cold-junction temperature of 155°C (423 K). A 423 K heat rejection temperature allows for reasonable radiator size and mass, as long as the space RTG power output is in the range of 100 to 150 watts.

3.1.3 Silicon Germanium

Silicon Germanium (SiGe) thermoelectric material has a relatively lower figure-of-merit, but can operate over a large temperature range. With a hot-junction temperature of ~1,100°C (1,373 K) and a cold-side temperature of ~300°C (573 K) the thermoelectric efficiency is ~7%. With a 573 K cold side the radiator is small, and the waste heat can be used very effectively for spacecraft heating.

3.1.4 Segmented Thermoelectrics

More efficient RTGs can be made by segmenting thermoelectric elements to operate over the temperature ranges where the figures-of-merit are high. PbTe-TAGS thermocouples with cold segments of BiTe are used for terrestrial RTGs where a low heat sink temperature is available. For space use, the added power must be traded-off against added radiator size and mass to make the use of the BiTe segment worthwhile.

Due to the difference in coefficient of thermal expansion between SiGe and the telluride materials, attempts to cascade SiGe with PbTe-TAGS have added more disadvantages in converter fabrication and RTG size and mass than the advantages gained in the higher conversion efficiency.

3.2 RTG Experience

Small BiTe RTGs have been designed for a number of terrestrial applications. The 40 mW RTG under development for space by High-Z Inc. is a good example of the use of BiTe RTGs (with a generator efficiency of 3 to 4%).

Small milliwatt SiGe RTGs have been used extensively in terrestrial applications (with a generator efficiency of ~0.5%). Sixteen SiGe RTGs have been used in space over the past 24 years. Multi-

Hundred Watt RTGs (160 watt BOM) were flown on DOD LES 8 and 9 and NASA Voyager 1 and 2 missions. GPHS-RTGs (285 watt BOM) were flown on NASA Galileo, Ulysses, and Cassini missions.

Prior to SiGe RTG technology, 28 RTGs launched on U.S. space missions used PbTe or PbTe-TAGS thermoelectric converters. PbTe-TAGS RTGs ranged in power from 2.7 We (SNAP-3) to 73 We (SNAP-27), and weighed 2.3 kg and 29.5 kg, respectively. The specific power of those early RTGs ranged from 1.2 to 2.9 W/kg at launch. SNAP-19 RTG had PbTe-TAGS unicouples and powered the Pioneer 10 and 11 spacecraft and the Viking 1 and 2 Mars Landers. Each Pioneer RTG produced ~40 watts (BOM) and weighed 13.6 kg for a specific power of 2.9 W/kg. Each Pioneer RTG was fueled with 645 watts of Pu-238 for a BOM generator efficiency of 6.2%. Each Pioneer spacecraft was powered by four SNAP-19 RTGs and operated for 25 years. Each Viking RTG produced ~42.5 watts (BOM) and weighed 15.4 kg for a specific power of 2.8 W/kg. Each Viking RTG was fueled with 682 watts of Pu-238 for a BOM generator efficiency of 6.2%. Each Viking Lander was powered by two SNAP-19 RTGs and operated for 4 and 6 years before signals were lost due to transmission difficulties.

The SNAP-19 RTGs used Min-K insulation with a cover gas mixture of helium and argon to control hot-junction temperatures over the life of the mission. The Pioneer RTGs were 20 inches in diameter (fin tip to fin tip) and 11.2 inches high. The Viking RTGs were 23 inches in diameter (fin tip to fin tip) by 15.6 inches high.

The GPHS RTGs used on the Galileo, Ulysses, and Cassini missions produce 285 watts (BOM). The RTGs weighed 56.2 kg, were 42.7 cm in overall diameter (fin tip to fin tip) by 113 cm in length. The general purpose heat source consisted of 18 GPHS modules stacked along the axis of the cylinder, surrounded by 572 SiGe thermoelectric unicouples. The temperature drop across the unicouples is nominally 700 K (1,273 K hot junction and 573 K cold junction) and the outer shell temperature of the RTG was nominally 473 K at BOM.

3.3 Small RTG

Lockheed Martin Astronautics (LMA) prepared a design for a small, 100-watt SiGe RTG in which the heat source support system is the same as the existing SiGe RTG technology. The individual component designs of the heat source support structure were lightened to be compatible with the lower required pre-load for less GPHS modules. The pressure relief device (PRD) design was downsized to be compatible with the lower internal volume of the small RTG. The quartz cloth separator design between the molybdenum foil in the multi foil insulation system was replaced with zirconia powder design to provide mass savings. The development for the zirconia powder as spacers between foils is a technical risk. Zirconia powder has been used between metal foils in terrestrial applications. The design has fewer unicouples that increases the hot side temperature by 70 K. This increases the thermal gradient across the unicouples, which increase the generator efficiency. The calculated BOM power output for this design is 108 watts with 6 GPHS Modules and 85 watts after 12 years with an estimated mass of 21 kg and an estimated mass BOM specific power of 5.1 watts/kg.

4.0 Missions Requiring RPSs

4.1 Future NASA Missions

4.1.1 Solar System Exploration Missions

NASA's exploration of the solar system has planned a series of missions to explore outer planets and comets. Table 4-1 summarizes NASA's mission plans to explore the outer planets. The potential near term outer planet missions are the Europa Orbiter, Pluto/Kuiper Express, and Comet Nucleus Sample Return. Europa, Jupiter's fourth largest moon at a distance of ~5.2 AU from the sun, has attracted immense interest because of indications that a liquid ocean lies underneath its icy crust. If an ocean exists on Europa, that increases the chance that some form of life exists or existed there.

Table 4-1. SSE Missions that Could Require or Benefit from ARPS

Mission	Science	Special Condition	Proposed Power System	Comments
Europa Orbiter	- Is there an ocean of water beneath the ice? - Are there places where the ice is thin or where water reaches the surface? - Could the environment support pre-biotic chemical processes?	Strong radiation fields	Cassini spare GPHS-RTG (F-5) and new fuel GPHS-RTG (F-8) or Stirling ARPS	RTG or Stirling ARPS are being considered.
PlutoKuiper Express	- Determine origins of Pluto, Charon, and Kuiper Belt. - What is the surface composition and atmospheric structure of Pluto and Charon? - What is the organic inventory of the far outer solar system?	15 year mission life	Cassini spare GPHS-RTG (F-5) or new fuel GPHS-RTG (F-8)	May require RTG for long life and reliability for a one S/C flyby of Pluto and Charon.
Europa Lander	- Age and composition of the Europa surface? - What organic chemical processes take place? - Is there any potential access to liquid water? - Any indications of biological activity?	6-8 year mission life	AMTEC-ARPS	Considering power system based on low mass AMTEC and non-nuclear alternates.
Comet Nucleus Sample Return	- What is the chemical composition of comet nucleus material? What does it tell us about the primordial solar system? - How have comets evolved since their formation? How does their composition vary with depth and location on the nucleus? - What can we learn about the likely effects and mitigation of commentary impacts?	6-10 year mission, surface power to drill 1 to 3 meter holes for samples	Solar for SEP and Lander	Planners not counting on ARPS, but it is worth investigating how ARPS could improve the mission.
Neptune Orbiter	- What is atmospheric structure and chemistry? - What is the structure and behavior of its magnetosphere? - What are Triton's physical properties? Is it a captured Kuiper object? What can it tell us about the formation and evolution of the far outer solar system? - What are the dynamics of the rings and satellites?	10 year flight to Neptune, 2-4 year orbital tour	AMTEC- ARPS	Considering power system based on low mass AMTEC and non-nuclear alternates.

Table 4-1. SSE Missions that Could Require or Benefit from ARPS (Continued)

Mission	Science	Special Condition	Proposed Power System	Comments
Titan Explorer	- What prebiotic chemistry is taking place at Titan and what can it tell us about the primordial Earth and the origin of life? - What is the composition of Titan's surface and how does it interact with the atmosphere? - How has Titan evolved over its history?	-	AMTEC-ARPS	Considering power system based on low mass AMTEC and non-nuclear alternates.
Saturn Ring Observer	- What are the physical properties of the icy particles comprising Saturn's rings? - What do the detailed, time-varying interactions tell us about the evolution of Saturn's rings? - What does the detailed study of Saturn's rings tell us about the early stages of planet formation and the present-day dynamics of extra-solar disks, accretion disks, and spiral galaxies?	-	AMTEC-ARPS	Considering power system based on low mass AMTEC and non-nuclear alternates.
Venus Surface Sample Return	- What is the age and chemical composition of Venus's surface? What is its atmospheric composition? - Why did Venus and Earth take such different evolutionary pathways? - Was there ever liquid water on Venus? Where did it go? - What can Venus tell us about the future of planet Earth?	-	Solar and batteries for landed operations	Planners not counting on ARPS, but it would be worth investigating how ARPS could improve the mission.
Jupiter Polar Orbiter	- Study the Jupiter aurora - Study the high latitude atmosphere including polar hood and circulation - Gravity and dynamo magnetic fields - Middle and outer magnetosphere - Other studies best performed at high latitudes	-	AMTEC-ARPS (180 W BOL) (option 2)	Considering power system based on low mass AMTEC and non-nuclear alternates.

Pluto is the only planet in our Solar system not yet explored by a spacecraft. Pre-project planning for a Europa Orbiter launch in 2003 and a Pluto/Kuiper Express launch in 2004 was conducted from FY'97 through FY'99. Estimated project costs and delays in launch vehicle and advanced development have postponed these launch dates by at least 5 years. After these two missions, a number of other potential Solar System Exploration missions are planned.

The Solar Probe mission is the nearest term mission planned by the Sun-Earth Connection Enterprise. The mass requirements for the Solar Probe and several Outer Planet missions as of August 1999 were based on an estimated AMTEC-ARPS module with 21% efficiency, 10 kg mass and 96 watts output power at EOM six years after launch. Near-term ARPS technology does not meet these requirements.

The outer planet missions could require low-mass, long-life ARPS. The outer planets program is enabled by radioisotope power. SOA RTG could provide the necessary power, but the outer planet missions' power system requirements for low mass are much less than the SOA RTG mass. Tables 4-2 and 4-3 summarize preliminary estimates of power needs for planned outer planet missions as of late CY'99. These tables represent a "snapshot" at this time of mission plans that are undergoing continual changes. Additional information is provided in Appendix A for some of the missions.

4.1.2 Mars Exploration Program

The MEP plan as of November 2000 proposes a series of missions on two-year centers, alternating between landers and orbiters. Landers could be launched in 2003, 2007, 2011 and beyond. The 2011 opportunity is baselined as a sample return mission. Beyond 2011 some form of outposts might be started as a transition to the eventual human exploration of Mars. The orbiters would be used as telecommunication relay stations after they carry out their primary science missions. The orbiters would use solar photovoltaic/batteries power systems.

The 2003 mission would utilize two Rovers with much more capability than the Pathfinder Rover. The 2003 Rovers could use a solar photovoltaic/battery power system with Radioisotope Heater Units (RHUs) as heaters. The plan is to arrive at a semi-equatorial zone location and operate the primary mission until the photovoltaic/battery power system degrades seriously, as dust builds on the arrays.

The 2007 Lander may use a number of new technologies not available to the 2003 Mars Rovers. At the present time, a considerable effort is going into planning and developing technology for the Mars 2007 Lander. A large rover is being considered as the Mars 2007 payload.

The required lifetime of the Mars 2007 Rover has not been finalized. With new technology for entry, descent, and landing, this Lander will be placed in a location where a long-range rover can carry out significant exploration of the site. The 2007 mission might cache a sample for a later sample return mission.

Also planned for 2007 is a new line of "scout" missions that will be selected by the science community. These missions could be airborne vehicles, balloons, or small landers with a cap of $300 M for each mission. Power requirements for scout missions cannot be specified until the missions are defined, but are likely to be much less than a 100 watts.

Table 4-2. Power Mode Summary for Baselined Outer Planet Cruise Stage/Orbiters†

Cruise Stage/Orbiter	Power Mode (watts electric)					RPS Comments
	Launch	Cruise	Encounter	Science	Other *	
Europa Orbiter	160.3	204.0	304.2	228.9	-	(b)
Pluto/Kuiper Express	120.9	132.2	182.3	154.4	-	(c)
Europa Lander (Carrier)	-	78.5	205.1	-	-	ARPS on lander
Neptune Orbiter	123.7	114.1	80.7	101.1	147.6 #	(d)
Saturn Ring Observer	121.2	44.6	52.0	158.1	146.5 #	(d)
Titan Explorer (Carrier)	183.6	152.3	49.4	162.5	-	(d)

(a) Li-Ion battery to provide additional power for these modes.
(b) Original plan called for ARPS but project prefers RTGs.
(c) Original plan had called for ARPS but project prefers existing Cassini spare RTG.
(d) Plan assumes AMTEC-ARPS with 212 watts BOM.
(e) Plan assumes AMTEC-ARPS with 106 watts BOM.

* Li-Ion battery to provide additional power for these modes.
\# Includes data transmission.
† These data are taken from documents prepared in early 2000 and may have changed since then.

Table 4-3. Power Mode Summary for Baselined Outer Planet Landers†

Landed Vehicles	Power Mode (watts electric)					RPS Comments
	Entry	Mobility	Sample Collection	Sample Analysis	Data TX	
Europa Lander	237.3	-	56.0	109.8	205.1(a)	(d)
Titan Explorer (Aerobot)	52.3	30.1	57.8	58.0	58.4	(e)
Titan Explorer (Rover)	49.4	76.4	70.5	76.4	75.8	(e)
Titan Explorer (Aerover)	9.0	75.5	56.3	62.2	62.8	(e)

(a) Li-Ion battery to provide additional power for these modes.
(b) Original plan called for ARPS but project prefers RTGs.
(c) Original plan had called for ARPS but project prefers existing Cassini spare RTG.
(d) Plan assumes AMTEC-ARPS with 212 watts BOM.
(e) Plan assumes AMTEC-ARPS with 106 watts BOM.

† These data are taken from documents prepared in early 2000 and may have changed since then.

An ARPS with a specific power of 5 watt/kg or greater would be a significant enhancement for larger Mars surface missions. An ARPS would allow 24-hour operation, long life, imperviousness to dust storms, and the ability to operate in shadows and canyons.

A variety of future Mars missions may occur after a Mars sample return. Of particular interest are the missions that would involve would deep drilling—some to 200 m and some to 1 to 3 km. These missions would require extended stays on Mars in polar areas, where radioisotope power sources are required.

4.1.2.1 Mars Power Options

RTGs on the Viking Landers operated as predicted on the Mars surface for six years. The production facilities to manufacture this hardware are not readily available. RTG production capabilities could be resurrected to the point of readiness for Mars 2007 if a decision to use an RTG is made by October 2002. A Stirling-ARPS might possibly be developed in time for Mars 2007. Solar power is regarded as the baseline against which RTG and Stirling ARPS will be compared. Solar cell/battery power systems have the advantages and disadvantages listed in Table 4-4.

The characteristics of solar powered systems on Mars are discussed in Appendix B.

The ARPS assessment team decided that ARPS technology could not be ready for the Mars 2007 missions. Therefore, this team made no recommendation on the power source for Mars 2007 or the Europa Orbiter 2008 missions. The recommendation for a power source for any NASA mission before 2011 is to come from the Casani assessment team that was commissioned by NASA and DOE headquarters after this assessment was nearly finished.

4.1.2.2 Requirements for ARPS on Mars

ARPS enables a number of missions that cannot be accomplished with solar/battery power systems. The Viking Lander I was productive for 6.4 years. The Viking landers use of RTG power enabled Mars meteorology to be enriched by observing seasonal changes in atmospheric pressure, frost in winter, two dust storm cycles, and diurnal changes in the weather over an extended period. The tentative requirements for future Mars surface missions are summarized below.

Pre-launch sterilization: The Mars 2007 mission requires a 4A-sterilization that can be met by alcohol wipe-down as was used on Mars Pathfinder. The Mars Sample Return mission would require a 4B terminal sterilization of the ARPS. That sterilization is currently presumed to be a hydrogen peroxide gaseous bath for the entire spacecraft after encapsulation in the launch vehicle shroud or a bio-shield.

Launch Site Processing: The ARPS would be located on the Lander, inside an entry aeroshell. As such, it may require early installation and be inaccessible on the launch pad. The ARPS would have to provide its own cooling in an enclosed ambient environment on the launch pad. It also must be compatible with all spacecraft encapsulation and stacking operations.

Table 4-4. Advantages and Disadvantages of Solar Power on Mars Surface

Advantages	Disadvantages
• Proven technology • No special heat rejection system needed for cruise and entry • High specific power based on peak power	• Intermittent operation • Extended lifetime requires dust mitigation • Battery cycle life • Seasonal variations in solar incidence • Subject to interruption by dust storms • Inoperable in canyon shadows

Cruise Environment: The cruise from Earth to Mars ranges from 7-13 months depending on trajectory type. During this time, ARPS would be enclosed in the entry aeroshell and would have no direct view to space. The ARPS would have to provide radiative and/or conductive heat transfer to aeroshell hardware that would radiate the heat to space. Electrical power from the ARPS is not required, but it could be used during cruise if available. The spacecraft would likely have a cruise phase photovoltaic array.

Entry, Descent, and Landing (EDL): ARPS dynamic loads during EDL could exceed previous RTG loads. Preliminary estimates of entry and landing loads are: 10 g Z-axis, 5 g in X or Y-axis with static parachute; 6 g for supersonic drogue chute (Z-axis); 10 g (Z-axis) for subsonic chute; landing shocks, 20 g limit static (any axis). Load models are under development and more detailed dynamic load information will be available in the future.

Surface Dust Environment: Martian dust would accumulate on all radiator surfaces. While it has been observed that accumulation on angled or vertical surfaces is substantially less than that on horizontal surfaces, the differences have not been quantified. It is not clear what the effect would be on radiator performance.

Mars Surface Power: An early 2007 mission reference design requires 200 watts minimum on the surface of Mars at landing. The surface environment is a 6-10 torr CO_2 atmosphere with daily temperatures ranging from 170-270 K with varying amounts of suspended and deposited dust. A variety of mission scenarios (summarized in Appendix C) that could not be done with a solar/battery power system would be enabled with a 200 watt RPS on a Mars rover.

4.1.3 Sun-Earth Connection

The Solar Probe is the nearest term SEC mission. The Solar Probe would determine the mechanisms and sources for coronal heating and solar wind acceleration, and will explore the dynamics of interior convection in the Polar Regions. A limited one solar pass mission may be possible with solar battery power system. However, the scientists prefer a two pass solar mission to exam the sun characteristics at solar maximum and solar minimum activities. The two pass solar mission requires an ARPS similar to the very successful almost three pass Ulysses mission.

In addition to the Solar Probe, the NASA Sun-Earth Connection Program may include three missions that could require ARPS: Interstellar Probe, Interstellar Trailblazer, and Outer Heliosphere Radio Imager.

All three have baselined an ARPS in their plans. These missions are described briefly in Appendix D.

4.2 Fuel Availability

Radioisotope heat source availability is limited by the quantity of fuel that can be produced and processed or that is already in the inventory. Pu-238 had been produced in U.S. reactors and processing facilities at DOE's Savannah River Site. With the shutdown of the last production reactor at the SRS, the U.S. no longer has a Pu-238 production capability.

There is sufficient Pu-238 on hand for about 18 GPHS modules, or the equivalence of one GPHS RTG. The DOE has a contract with Russia to procure up to 5 kg of Pu-238 every 6 months through 2002. Therefore, the DOE can purchase up to 10 kg in 2001 and another 10 kg in 2002 assuming a purchase is made every 6 months starting in January 2001. However, the DOE believes that it can purchase additional Pu-238 from Russia.

The DOE recently completed a programmatic environmental impact statement (EIS) which includes the evaluation of alternatives for re-establishing a domestic Pu-238 production and processing capability. In its Record of Decision, the DOE announced its decision to produce Pu-238 at the Advanced Test Reactor in Idaho and at the High Flux Isotope Reactor in Tennessee.

The future availability of Pu-238 will have a significant effect on prioritization of ARPS technologies. If it is anticipated that beyond 2002 we can still purchase Pu-238 from Russia or re-establish domestic production facilities, then specific power and lifetime will be the most vital governing characteristics of ARPS technologies. If, on the other hand, supplies beyond 2002 will be difficult to obtain, conversion efficiency may be the most important characteristic.

5.0 Advanced Technologies Evaluated

Technologies evaluated in this study are listed in Table 5-1 and described after the table.

Table 5-1. Technologies Evaluated

Technology	Specific Technology	Comments
Thermoelectric	SiGe RTGs	Used on Voyager, Ulysses, Galileo and Cassini missions
	PbTe-TAGS	Used on Viking and Pioneer Missions
Stirling Engine Converter	Version 1.0	Present design; efficiency is very good but mass is high; lifetime is not certain; most mature of the ARPS technologies; only one with a reasonable chance of being made ready for Mars 2007
	Version 1.1	Make several near-term improvements to reduce mass of Version 1.0.
Advanced Stirling Engine Converter	Version 2.0 or Thermoacoustic	Low mass Stirling engine, alternator, radiator and controller Potential for high efficiency long life Stirling Engine. Considered as an advanced form of Stirling technology.
AMTEC	Refractory Metal Chimney	Potential for low mass and medium efficiency. Being developed under existing DOE contract
Segmented Thermoelectric	Advanced Materials/ Segmented Unicouple	Potential for low mass and medium efficiency. Solid state device configuration, operations, and handling are similar to SiGe RTG
Thermionic	Cesiated triode	Low temperature 1300 K thermionic. Early stage of technology development
	Micro-miniature	Early stage of research
Thermo-photovoltaic	Thermal PhotoVoltaic Cell	Early stage of research

Advanced Stirling Engine Converter (ASEC)-ARPS. ASEC-ARPS version 2.0 technology consists of a reciprocating free-piston Stirling heat engine with a linear alternator that is a low-mass version of the Stirling engine alternator now under development by DOE and NASA. The ASEC-ARPS has the principal advantage of increased conversion efficiency of ~25%. This is almost four times the system efficiency of the SOA SiGe RTG. Near-term versions 1.0 and 1.1 of the ASEC-ARPS technology have specific power that is nearly the same as the SiGe RTG. Version 2.0 would improve this somewhat. The ASEC-ARPS major technical issues are:

1) Validating the system lifetime for 6 to 15 year missions,
2) Developing a low-mass, long life, highly reliable Stirling engine alternator,
3) Developing a high efficiency, low-mass, long life, highly reliable controller,
4) Reducing the EMI for space missions that measure very small magnetic fields, and

5) Reducing the Stirling engine alternator vibration for very sensitive seismic instruments.

Another ASEC-ARPS engine technology consists of a thermoacoustic heat engine with no moving parts that drives two linear alternators. The thermoacoustic engine offers the possibility of lower mass and longer life than the reciprocating free-piston Stirling engine.

Alkali Metal Thermal to Electric Converter (AMTEC)-ARPS. The AMTEC-ARPS produces electric power by pressure-induced flow of sodium ions through a beta-alumina solid electrolyte (BASE) which produces dc current and voltage. AMTEC delivers dc power with no vibration and very small EMI. AMTEC has a potential system efficiency (~20%) nearly three times the system efficiency of the SOA SiGe RTG. The present AMTEC converters being developed have a predicted efficiency of 14%. AMTEC ARPS has the potential for doubling the specific power of the SiGe RTG (from 4.5 to 9 watts/kg). A three-year $10 M/year AMTEC-ARPS technology development program was conducted with the goal to advance AMTEC ARPS technology to NASA TRL 6 for the NASA JPL Europa Orbiter (EO) mission. Detailed road map and technology gates were established for AMTEC-ARPS. The AMTEC-ARPS technology did not meet the technology gates by September 1999 as required for an EO November 2003 launched mission. Therefore, the AMTEC development was scaled back in FY'00 to a $3M/year technology development effort.

The technical preliminary design of a 100-watt class AMTEC-ARPS generator and the detailed design of AMTEC converters were completed. Five AMTEC converters were fabricated and tested. As a result, the technical issues for AMTEC-ARPS converter appear to be defined. Solutions have been proposed for these technical issues and are being pursued. The AMTEC lifetime performance of four critical components, BASE, anode, cathode, and current collectors, have been operated under accelerated temperatures and the predicted lifetimes of these components are greater than 20 years. The wick/evaporator and electrical feed-through components are under development and need validated lifetime prediction codes. The AMTEC converter needs accelerated lifetime tests to validate its lifetime performance prediction codes.

The AMTEC-ARPS major technical issues are:

1) A BASE to metal ceramic seal,
2) Converter containment material fabrication process,
3) A reproducible wick-evaporator fabrication process,
4) Electrical feed-through fabrication process, and
5) A reproducible converter fabrication process with performance as predicted.

Segmented-Thermoelectric STE-ARPS. STE-ARPS consist of a solid material that produces a dc current and a voltage when placed in a temperature gradient. Each thermoelectric material, whether n-type or p-type, exhibits a maximum figure-of-merit at some temperature. If a single material is used in each leg of the unicouple, the effective efficiency will be an average over the temperature range, which is less than the maximum. If each leg of the unicouple is segmented, the temperature range over each segment will be smaller than the total temperature gradient. Thermoelectric materials developed with a high efficiency over this smaller temperature range for each segment will achieve a higher efficiency over the entire temperature difference from the hot side to the cold side. This has the potential to produce a STE-ARPS with double the efficiency of the SiGe RTG.

The major technical issues are:

1) Developing a compatible high temperature (973 K to 1273 K) thermoelectric material,
2) Developing joints between the segments with very small thermal and electric resistance,
3) Developing barriers that prevent inter-diffusion between segments, and
4) Developing joints between the high temperature thermoelectric materials and a hot shoe with very small thermal and electric resistance.

Low Temperature Thermionics and Thermal Photovoltaics. Low-temperature thermionics and thermal photovoltaics technologies are at NASA TRL 1-2. Therefore, the team could not assess the efficiency, specific mass, or lifetime for an ARPS. The team recommends NASA's crosscutting technology program, the DOE PRDA program, and SBIR fund these two technologies until technical feasibility is demonstrated.

The team assessment of these advanced converter technologies follows.

5.1 Alkali Metal Thermal to Electric Converter Technology

5.1.1 Summary

A considerable amount of work has been done on AMTEC. AMTEC is currently at NASA TRL 2/3. The potential benefits of this technology are low-mass, medium efficiency, low EMI, dc power output, and no moving parts. The potential system efficiency of AMTEC-ARPS is 20% (nearly 3X the present SOA RTG) although present designs are less efficient. Accelerated testing can validate the lifetime of the AMTEC-ARPS. Analytical and experimental results of accelerated test of AMTEC BASE, electrodes, and current collectors predict greater than 20-year lifetime with less than 5% performance degradation. Eight flight prototype design AMTEC converters have been built and tested as of January 2001. The AMTEC technology development team identified the AMTEC technical issues and development efforts are underway to resolve some of these issues. There is a reasonable probability that a well-funded technology development program could advance AMTEC to TRL5 by FY'05.

5.1.2 Introduction

In 1962, during work on the sodium-sulfur battery, J.T. Kummer at the Ford Motor Company Scientific Laboratory conceived the thermally powered sodium concentration cell, based on the unique electrical properties of beta-alumina solid electrolyte. Kummer and his colleague Neil Weber experimentally demonstrated the feasibility of such a device and obtained a patent in 1968. Shortly thereafter, Weber began research on the new thermal-to-electric converter in collaboration with T.K. Hunt and T. Cole. This new converter is called the alkali metal thermal to electric converter (AMTEC). JPL began investigating this technology in 1983. The DOE joined the AMTEC technology development program for ARPS in 1997.

Many organizations have been involved over the years in development of AMTEC technology. This report focuses on the work that has been conducted over the last 3 years through NASA/JPL and DOE. JPL conducted in-house technology development on AMTEC BASE, electrodes and current collectors for many years at a very low level of funding and for the past 4 years at an average level of $600K/year. In March 1998, the DOE selected and awarded a system integration contract to Lockheed Martin Astronautics (LMA) with Advanced Modular Power Systems, Inc. (AMPS) as a subcontractor. This contract was originally based on the goal of achieving readiness for the Europa Orbiter (EO) 2003 launch and the Pluto Kuiper Express (PKE) 2004 launch missions as planned in 1998. A number of converter redesigns were made as the spacecraft and EO mission evolved. Over the 3-year period (FY'97-FY'99) AMPS spent approximately $16 M developing the advanced technology for an AMTEC converter for the EO and PKE missions. There were very specific technology readiness gates that AMTEC Technology had to meet by September 1999 to be ready for the planned November 2003 EO launch. In August 1999, DOE and NASA jointly reviewed the technology status against the technology readiness gates and agreed that the technology could not be ready for a November 2003 launch. Therefore, in FY'00 the AMTEC technology effort was scaled back from a flight technology development to a technology development program at $3M per year ($2 M at AMPS and $1 M at JPL).

5.1.3 Description (How it Works)

An AMTEC converts heat to electricity by using the unique characteristics of BASE. BASE conducts sodium ions but not electrons. An electrochemical potential is generated when sodium is present at two different pressures separated by the BASE.

Typical operating conditions for AMTEC are a high-pressure sodium vapor region at 900 - 1200 K, and a low pressure liquid sodium region at a condenser temperature of 400 - 650 K. It is desirable to operate the high-pressure region at as high a temperature as possible, in order to increase the conversion efficiency. However, high temperatures (>1200K) introduce degradation mechanisms of the BASE that limit lifetime. The BASE ceramic is coated on both sides with porous metal electrodes. Vapor phase sodium atoms are ionized at the high-pressure sodium/electrode/BASE interface and sodium ions flow through the BASE to the low-pressure side under the pressure gradient. Electrons are collected in the porous electrode on the high-pressure side and travel through an external load and recombine with sodium ions at the low-pressure side BASE/electrode interface. Figure 5.1-1 shows a schematic of the BASE, electrodes, and current collectors to illustrate how the various processes occur.

After recombination of electrons and sodium ions at the low-pressure side, sodium vapor travels through the porous electrode, leaves the electrode as vapor, and is collected as a liquid on a cold condenser. The sodium is re-circulated through the cell using a porous molybdenum wicking system that uses capillary forces to transport the sodium from the low-pressure region to the high-pressure region. The sodium in the high-pressure region evaporates from the wick and flows as a vapor to the surface of the electrolyte where it again ionizes and enters the BASE. A simplified schematic of an AMTEC cell is shown in Figure 5.1-2.

For the purpose of this report the following definitions are being used. The **cell** converts thermal energy to electrical energy (i.e., electrolyte, electrodes, current collectors, insulator, and metal-to-ceramic seals). The **converter** is an assembly of cells that convert heat to electric power (i.e., 8 AMTEC cells connected in series with evaporator/wick and cylindrical outer containment). A **generator** contains converters in a series parallel electrical arrangement heated by a radioisotope heat source. The generator can be integrated with a spacecraft as the power source. The generator includes multiple converters, thermal insulation, GPHS modules, a support structure housing, dc-to-dc converter if required, and a controller if necessary.

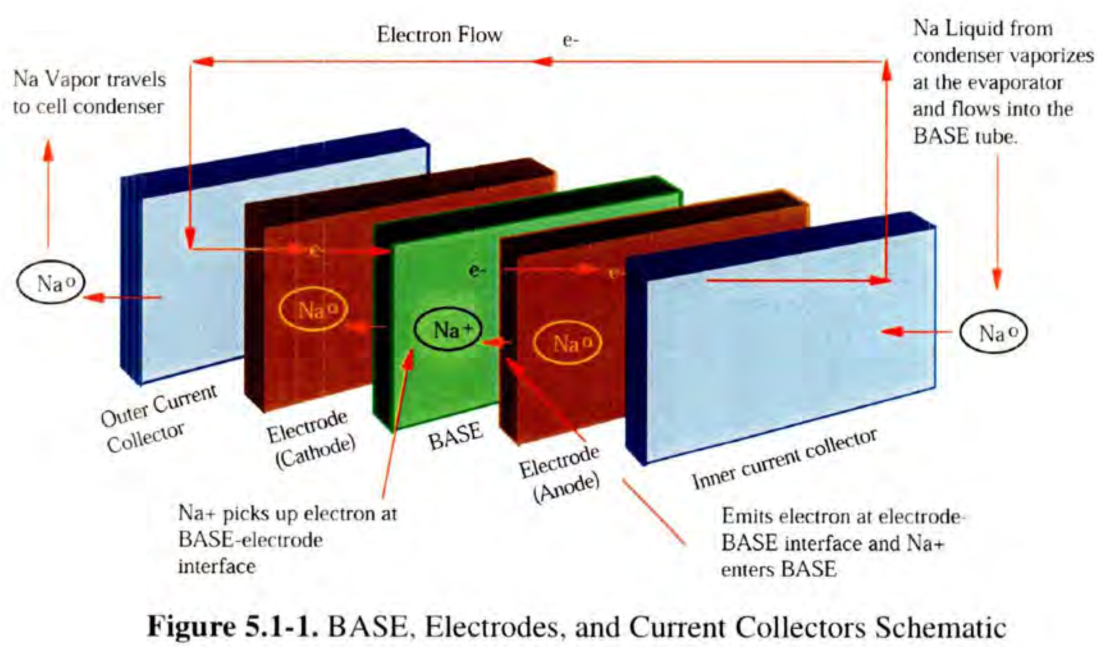


Figure 5.1-1. BASE, Electrodes, and Current Collectors Schematic

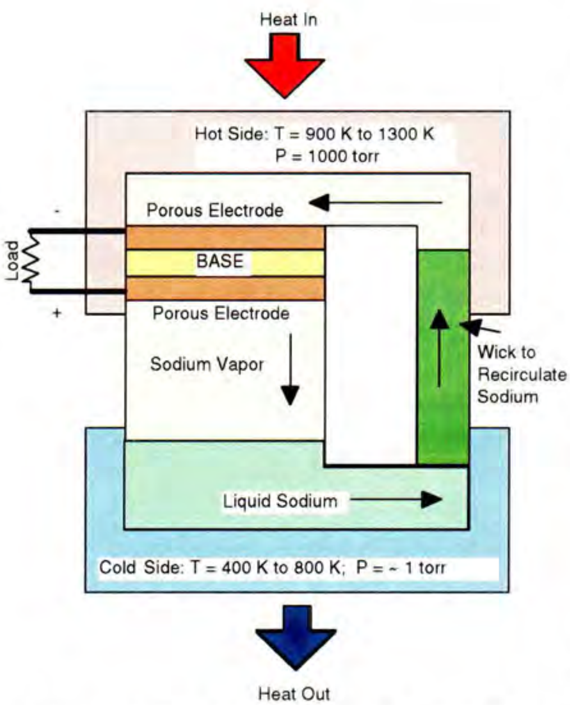


Figure 5.1-2. Schematic of AMTEC Cell

Thermal losses from the hot-side to the cold-side, pressure drops between the cathode and the condenser, and internal electrical contact resistances and leakage currents lower the efficiency below the Carnot level. AMTEC is a low-voltage, high-current converter, but useful voltages can be achieved by connecting multiple cells in series. An assembly of cells sharing a sodium reservoir is called a converter. The original converter design used a cylinder with a uniform diameter as illustrated in Figure 5.1-3. This design was the basis for most of the work done on AMTEC in the past. More recently, OSC and AMPS developed an alternate design using a smaller diameter between the AMTEC cells and the condenser called a chimney converter.

The chimney converter increases the efficiency by minimizing heat losses from the AMTEC cells to the condenser. Figure 5.1-4 illustrates this chimney design concept which is the design being developed as agreed to by DOE, JPL, LMA, and AMPS under the technology development effort beginning in FY'01.

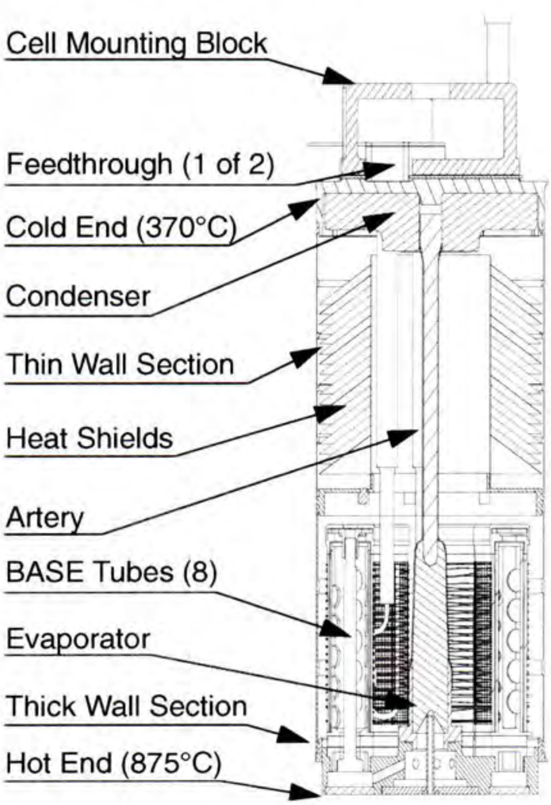


Figure 5.1-3. Current Cylindrical Converter Design

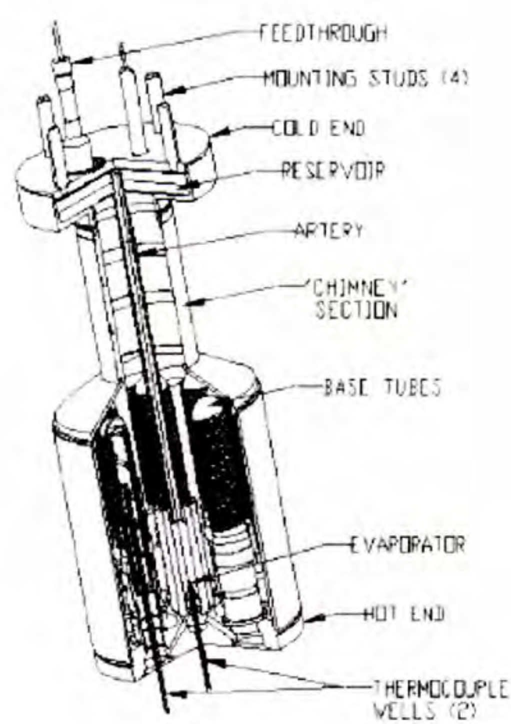


Figure 5.1-4. Chimney Converter Concept

5.1.4 AMTEC-ARPS Design

LMA developed a preliminary design for a AMTEC-ARPS generator from 1997 to 1999, as shown in Figure 5.1-5. Sixteen AMTEC converters are arranged around four GPHS modules. The heat source support system uses a single stud at each end to pre-load the GPHS modules. Molybdenum foil and quartz cloth thermal insulation are used along the length of the AMTEC converter walls and along the inboard and outboard dome. The converters are cantilevered from the generator housing outer circumference. The converters are electrically connected in a series/parallel configuration to obtain 28 volt dc power. A pressure-relief and a gas valve are located 90° from each other. The LMA design is 13.7 inches long by 15 inches in diameter.

Orbital Science Corporation (OSC) developed an alternate generator concept design that is illustrated in Figure 5.1-6. Eight chimney converters are arranged at the top and bottom sides of two GPHS modules. The AMTEC chimney converters are insulated with Min-K thermal insulation and surrounded by multi-foil insulation between the converters and the generator housing. Min-K was used on the SNAP-19 RTGs. The converters are electrically connected in series parallel to obtain 28 volts dc. The conceptual generator design is roughly 6 inches square by about 14 inches long.

The OSC design uses two GPHS modules whereas the LMA design utilizes four GPHS modules. Thus, two OSC generators are equivalent to one LMA generator. The OSC design is predicted to be more efficient. The system efficiency of a generator is less than that of a converter due to thermal and electrical losses. Thermal losses have been estimated to be 6 to 10% of the heat available, depending on the design and thermal insulation. Electrical losses and converter performance degradation are much more difficult to predict.

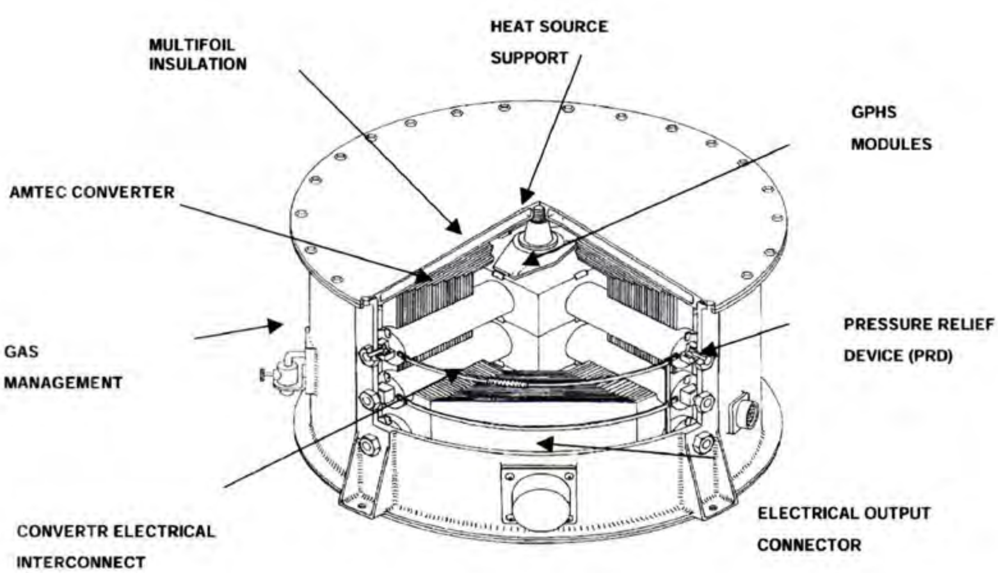


Figure 5.1-5. LMA 134 Watt BOM Generator Preliminary Design

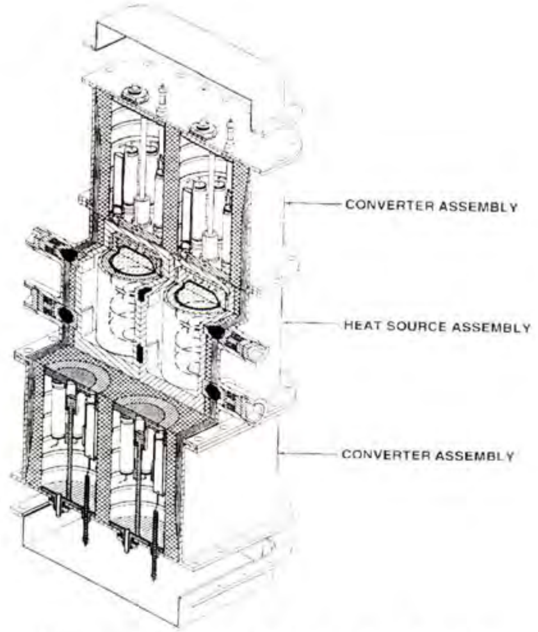


Figure 5.1-6. OSC 73.5 Watt BOM Conceptual Generator Design

OSC estimated a potential 21% conversion efficiency for their generator design if all seals were tight and there is no leakage. The actual achievable conversion efficiency of an AMTEC generator is difficult to estimate because of lack of reproducible experimental converter performance data. Early experiments produced measures experimental converter efficiencies of 13 to 14%. LMA estimated generator efficiencies in the 15-16% range. These LMA efficiencies were based on converter experimental data produced in 1998-1999. Future AMTEC converter technology development may improve on these efficiencies.

Table 5.1-2 provides estimated operating performance for the LMA and OSC concepts using conservative estimates for leakage and degradation based on preliminary laboratory measurements. In the ideal case, if these deleterious effects can be eliminated, the overall system efficiencies would rise by as much as 5-6%.

Accelerated experimental tests on AMTEC cell components (BASE, electrodes, and current collectors) at JPL validated long life performance prediction codes for these components with less than 5% degradation in 40 years. The lifetime performance of converter seals, wick, evaporator, and surface emissivities is being developed.

Table 5.1-2. Estimated Operating Performance Based on Available Test Data

Component	LMA Concept	OSC Concept
No. of AMTEC Converters	16	8
No. of GPHS Modules	4	2
System Mass (kg) AMTEC Converters Housing and other components GPHS Modules	25 6.7 12.5 5.8	9.3 3.1 3.3 2.9
Hot Side Temperature (°C)	850	877
Cold Side Temperature (°C)	382	368
Converter Efficiency (%)	15.5	17.0
Generator Efficiency (%) (thermal, interconnect, etc.)	90	90
Thermal Power of Heat Source at BOM (Watts)	960	480
Electrical Power at BOM (Watts)	134	73.5
Generator Overall Conversion Efficiency (%)	14.0	15.3
BOM Specific Power (Watts/kg)	5.4	7.9

The AMTEC radioisotope power system can be scaled based on the number of converters and number of GPHS modules. This is possible if the interconnects and housing are designed in to allow stacking the GPHS module and AMTEC converter in layers. This assumes that adding additional layers does not require complete redesign and re-optimization. However, it is more likely that several 100-watt AMTEC-ARPS modules would be used to power missions that require more than 100 W.

5.1.5 Technical Status

In January 1997 AMTEC was chosen as the high-risk baseline technology to be developed for the potential Europa Orbiter and Pluto/Kuiper Express outer planet missions. This technology development plan was based on a very short schedule and a high cost profile. During this program, considerable work was completed in converter design, fabrication, and testing as well a preliminary design of an AMTEC-ARPS generator. In the course of this work a number of AMTEC converter technical issues were identified and well defined. Although AMPS, ORNL, Mound Laboratory, OSC, TAMU and JPL are attempting to resolve several of the technical issues under a DOE/NASA converter technology development task, AMTEC is no longer a baseline technology for any of these missions.

The technical status of AMTEC cells, converters, and generators is described below. At the cell level, TiN electrodes with short lifetimes have been replaced with longer-life WRh electrodes. At the converter level, the inner and outer current collectors and interconnect between cells have been improved. The BASE tube manufacturing process has been improved to increase the mechanical integrity. The converter has been optimized through thermal analysis, resulting in an outer wall with integral ribs for thermal and mechanical benefits. The wick artery/evaporator assembly design has been optimized for high efficiency and mechanical strength. Custom fixtures and tooling have been developed for ease of fabrication. The sodium fill process, test chambers, and data acquisition systems have been standardized. At the generator level, detailed analysis, designs, and specifications have been prepared for the integration of the AMTEC converter technology into a system-level generator. The technical challenges that remain are discussed in the next section.

A number of AMTEC cells and converter have been built and tested experimentally. In parallel with the cell and converter technology development, BASE, electrodes, current collectors, containment materials and seals have been fabricated and tested experimentally. However, the latest cell technologies cannot always be included in the converters being fabricated and tested. The performance data of the components is the basis for the performance data given in Table 5.1-2. The lifetimes of the high temperature critical components (BASE, electrodes, and current collectors) have been experimentally demonstrated to be greater than 20 years. The technical lifetime performance challenges are the metal-ceramic seals, the wick/evaporator performance degradation, and any incompatibilities between materials within the converter and the generator over a 15-year lifetime. The consensus is that lifetimes in excess of five years with acceptable degradation rates should be attainable. The AMTEC-ARPS interfaces with the GPHS and spacecraft quite easily. An emergency heat rejection system will likely have to be developed in the unlikely event of loss of sodium from two AMTEC converters.

5.1.6 Technical Issues

An extensive AMTEC technology development program was conducted over the last few years. Other conversion technologies included in this report were not evaluated to the same technical depth as AMTEC. The very comprehensive list of technical issues for AMTEC is partly attributable to the effort that has been invested in attempting to ready this technology for potential outer planet missions. Other technologies are likely to discover problems as they are developed. Table 5.1-3 summarizes the technical challenges. Table 5.1-4 summarizes work that still needs to be done to assure reliability and performance of AMTEC-ARPS.

Table 5.1-3. AMTEC Technical Issues

Technology	Issue	Description
Cell	Cell material incompatibilities	Potential issues remain for MoRe: (1) low CTE for match with BASE and (2) high thermal conductivity
Cell	Base Tube Assembly Seal	Reliable rugged seals for BASE to metal ceramic that meets leakage, compatibility, and lifetime requirements
Cell	Wick pore size and fabrication	Produce reproducible small pore size wicks; Resolve crooked artery fabrication issue; leakage; CVD coating sleeve design. Verify performance of wick in sodium
Cell/ Converter	Braze	Braze for metal to ceramic feedthrough; stability over time; no reaction with other materials or impurities
Cell/ Converter	Fabrication/ Repeatability	Build multiple cells and converters within tight tolerances and reproducible performance
Cell/ Converter/ Generator	Emergency heat rejection	Loss of Na causing temperature increase in the fueled clads resulting in grain growth – potentially increasing the probability of fuel release (safety consequences)
Converter	Open circuit voltage is low at high temperatures for multiple cells in a converter	Open circuit voltage of 0.5 V instead of ~1.5 V This implies that cells are leaking sodium, or that the sodium pressure on the hot side is about 1/1000 of what it ought to be for the stated hot side temperature; or that electronic shorts exist, or some combination of the three.
Converter	Plasma discharge in Na vapor	Plasma discharges in the sodium vapor at voltages above 5 V needs to be experimentally verified or not
Converter	Current collector	Need data that proves contact resistance is not an issue

Table 5.1-4. Work to be done to Assure Reliability and Performance of AMTEC

Technology	Issue	Mitigation
BASE	Micro-cracks	Implement pre-treatment of BASE to remove impurities
Cell	MoRe electrodes	Obtain long term endurance data – understand interaction of MoRe with BASE. Consider grading:(BASE to sapphire; Sapphire to MoRe)
Cell/ Converter	Long Lifetime Demonstration	Effects of impurity over time, reactions involving all materials; empirical degradation data; component life models; electrode life model; Converter life model
Cell/ Converter	Reliability	Need data to determine reliability if power loss from single converter (AFRL tests not applicable due to Na leakage)
Converter	Zero-g Operation	Prove flow of Na in zero-g is same as or better than one-g
Converter	SiO_2 in Insulation may lose O_2 and O_2 may react	Conduct long-term accelerated tests to determine maximum allowable thermal insulation temperature. Explore alternatives for other insulators.
Converter	Component understanding	Predict how component design changes affect converter
Converter	Actively cooled	Verify performance of AMTEC with passive radiator
Converter	Design/ Fab of Chimney Converter	Develop design and fabrication details and issue drawing package
Generator	Structural Properties	Vibration loads at launch temperatures; thermal shock during rapid heat-up from fuel loading; random vibration

5.1.7 Budgets and Schedules

Approximately $16 million was spent by AMPS and $1.8 million by JPL during the 3-year period from FY'97-FY'99 developing a flight AMTEC converter. The AMTEC technology development program that is now in place funded AMPS $2 million in FY'00 and plans to fund AMPS $2 million each year in FY'01 and FY'02. Planned funding for JPL is $1 million per year in FY'01 and FY'02. The goal is to resolve the technical issues and test three prototype chimney converters by the end of FY'02 to demonstrate predicted performance and long life. The team estimates an additional year at $3 million is needed to fabricate and test four reproducible AMTEC converters. The team estimated an additional 2 years at $4 million per year to fabricate and test: 1) an electrically heated 4-converter generator, 2) a multiple-converter prototype with thermal insulation and a simulated electrical heat source, and 3) additional converters for life testing. The planned funding per year and each activity required to reach NASA TRL 5 by the end of FY'05 is shown in Table 5.1-5.

5.1.8 Recommendations

AMTEC has the potential for high specific power and conversion efficiency. AMTEC directly produces 28 volt dc electrical power, has low EMI, and no vibrations. These attributes make it attractive to continue development of AMTEC. Previous AMTEC technology development identified technical issues and approaches to resolve these issues. The goal of the ongoing technology program should be an operating, high performance converter with reasonable lifetime by the end of FY'03. This is a technology readiness gate where progress towards the goal will be reviewed to determine whether to fund further work. If technical progress through FY'03 demonstrates that a high-performance, reproducible converter can be made, that would justify spending of $8 M over the following two years to develop and test a multiple-converter prototype system with simulated heat source and sink. Successful AMTEC multiple-converter prototype test and life tests on individual AMTEC converters would demonstrate NASA TRL 5 by the end of FY'05.

5.2 Stirling Converter Technology

5.2.1 Summary

Stirling cycle technology has been under investigation by NASA and DOE for heat engine and cooler applications since the 1970's. Most of the early work focused on kinematic designs for automotive and terrestrial power applications. The free-piston design was introduced to eliminate seals and lubrication in the power linkage, thus making it useful for space (vacuum) applications. Recent Stirling efforts have focused on a 55-watt, free-piston Stirling converter, known as the Technology Demonstration Converter (TDC). The TDC has demonstrated both design power output and high conversion efficiency (>25%) making it a candidate for future ARPS for NASA's outer planet missions and Mars landers. Future prospects for Stirling technology include a lower mass version of the TDC, innovative clean-sheet system designs, and high temperature heater heads. A variant of the Stirling cycle using thermo-acoustic transducers may provide benefits for an advanced Stirling.

Table 5.1-5. Projected Program for AMTEC Development to TRL 5

Activity	Year				
	2001	2002	2003	2004	2005
AMPS prototype converters	$2M	$2M			
JPL technical support	$1M	$1M			
AMPS 4-converter generator			$3M		
AMPS engineering model multiple converter system and life test AMTEC converters				$4M	$4M

5.2.2 Introduction

Free-piston Stirling technology for space applications underwent extensive development during the 1980's SP-100 Space Reactor Program. The objective was to develop a high performance conversion technology for 100-kw class nuclear reactor power systems. The Space Power Demonstrator Engine (SPDE) was a 25-kw Stirling engine built in 1984 by Mechanical Technology Incorporated (MTI), which included two opposed 12.5-kw converters connected at the hot-end with a common expansion space. After initial demonstration, the two converters were separated and accumulated over 400 hours of operation. That effort was followed in the late 1980's with the development of the single cylinder, 12.5-kw Component Test Power Converter (CTPC) which incorporated new materials and a heat pipe heater head configuration to increase hot-end temperature. The CTPC underwent successful testing and compiled 1500 hours of total operation. Stirling technology experienced a temporary hiatus until the late 1990's with the development of the 55-watt TDC by the Stirling Technology Company (STC). The TDC, shown in Figure 5.2-1 is a compact, high efficiency converter designed to be used in an opposed-pair configuration for GPHS radio-isotope power applications. Extensive performance, vibration, and electromagnetic interference (EMI) testing were conducted on the TDC as preparation for application to an outer planet mission.

NASA and DOE have combined to support the TDC development with $4.4 M between FY'97 and FY'00. A $2 M conceptual design study managed by DOE is presently underway that will down-select to one system integration contractor for a Stirling RPS in FY'01 from three candidates. That effort was originally aimed at providing an ARPS for a potential Europa Orbiter 2006 mission, but is now likely to be redirected toward future Mars surface missions. NASA GRC is supporting completion of the TDC technology development with a proposed FY'01 $3.5 M in-house project of which $1.5 M is funded to date.

5.2.3 Description (How it Works)

A schematic showing the operation of a free-piston Stirling converter is provided in Figure 5.2-2. There are four steps in the thermodynamic cycle. Heat is added to the working fluid during the high temperature, isothermal expansion (step 1) and rejected during the low temperature, isothermal compression (step 3). A regenerator removes heat from the working fluid during the first constant volume displacement (step 2) and returns it during a second constant volume displacement (step 4). A displacer is used to transfer heat through the regenerator during the constant volume processes. The pressure forces produced during the expansion and compression steps are acted on a power piston connected to a mover that reciprocates in a linear alternator (typically between 60 and 100 Hz). Gas or flexure bearings are used to maintain non-contacting linear motion within the cylinder. The alternator produces single phase, ac power (typically between 50 and 70 Vrms). Most Stirling power systems include power electronics to convert the ac into the usable dc for the spacecraft bus.

5.2.4 Modular Conceptual Design

A Stirling converter consists of a Stirling heat engine and a linear alternator. An ARPS consist of one or more converters, a heat source containing GPHS modules, a waste heat radiator, an ac-to-dc power controller and the structure to support the system. A secondary heat removal system is also anticipated (not yet designed) to dissipate GPHS heat during certain mission phases or in the event of a converter failure.

5.2.4.1 Configuration and Materials

A 100 watt-class ARPS based on Stirling can span a wide range of designs and performance based on a phased technology evolution. It is useful to refer to these different configurations in terms of version number. Version 1.0 represents the present baseline ARPS design using the TDC with minimal changes required for space flight qualification. A design layout of the TDC is provided in Figure 5.2-3. The hot-end is composed of Inconel, Nickel, and 316 Stainless Steel. The pressure vessel and power piston are composed of 304 Stainless Steel. The alternator uses FeNdB magnets and Hyperco laminations. A total of

10 TDCs have been or are in the process of being fabricated by STC with only minor variations in the design and construction materials. The TDCs are supplied with a nichrome electrical heater surrounding the Stirling heater head and liquid ethylene glycol coolant lines for the cooler. The flight version will require modifications to the heater head to facilitate radiation heat transfer from the GPHS and replacement of the coolant lines with a conductive or heat pipe radiator interface. Replacement of bolted flanges with weld seals on the Stirling pressure vessel is required. The radiator and controller (not shown) could use conventional materials and standard fabrication processes.

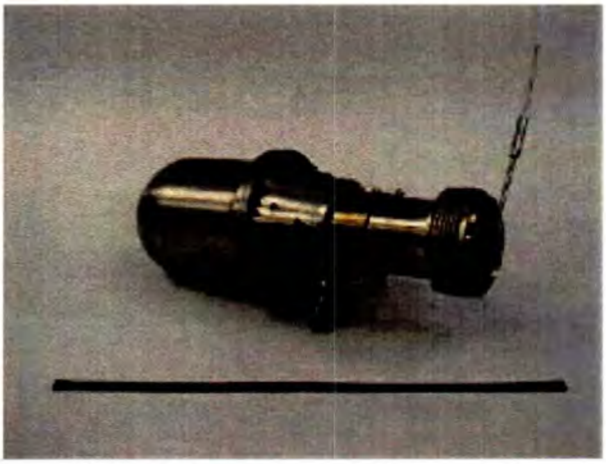


Figure 5.2-1. 55-Watt Stirling TDC

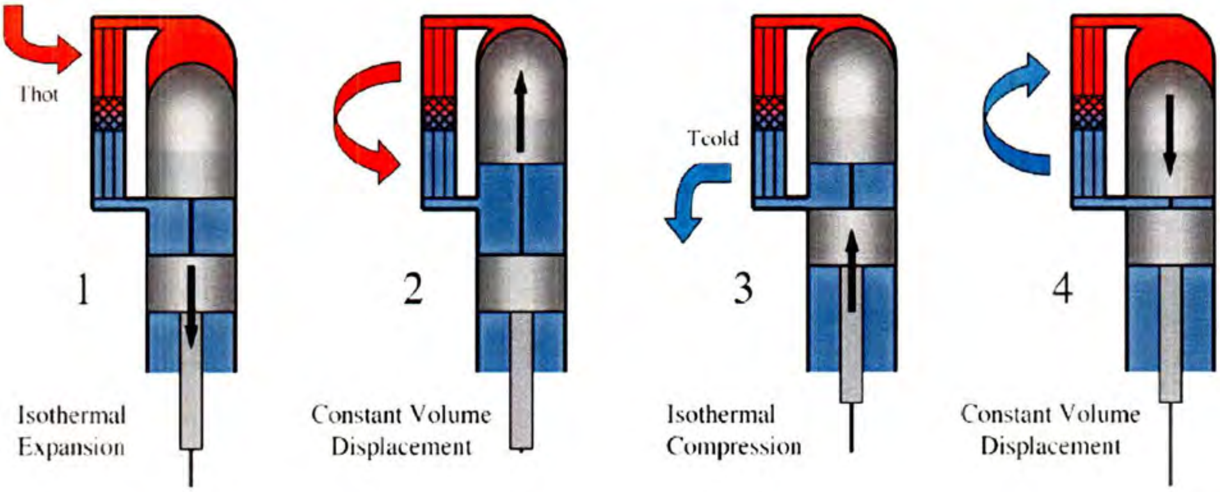


Figure 5.2-2. Stirling Cycle Thermodynamic Process

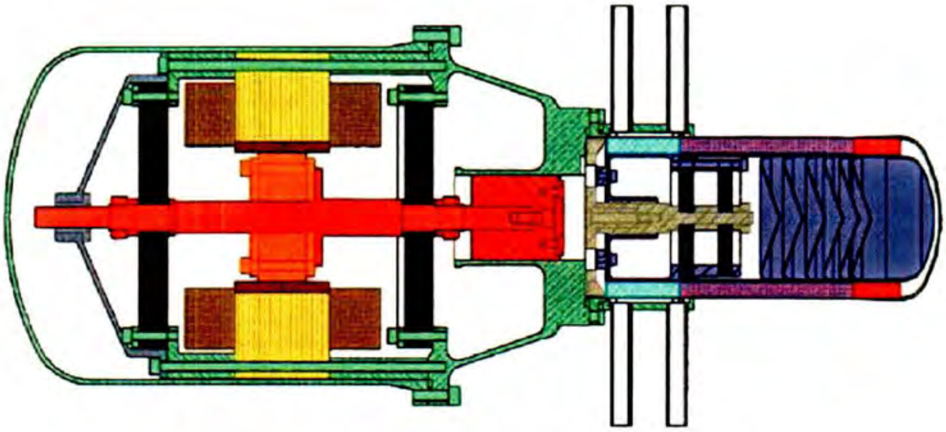


Figure 5.2-3. TDC Conceptual Design Layout

Version 1.1 represents the next generation TDC, with mass and volume reductions in the linear alternator leading to a smaller, lighter pressure vessel and the potential for an overall decrease in converter mass of up to 45%. The advanced alternator would use a flux concentrator magnet design rather than the conventional magnet geometry in Version 1.0. The modification is expected to reduce the outer diameter of the alternator pressure vessel from 4 inches to 3 inches. Other potential changes in Version 1.1 include capacitor-free control electronics, resulting in a 75% mass savings in the controller and improved ac-to-dc conversion efficiency.

Version 2.0 is a "place-holder" for an as-yet undefined complete redesign of the system with advanced converters, radiators, and electronics. Alternate Stirling engine configurations, including displacer-free, thermo-acoustic converters, would be considered. The Stirling hot-end would be designed to improve thermal integration with the GPHS. Radiators would use advanced materials such as beryllium, carbon-carbon, or pyrolitic graphite and could include heat pipes for heat transport to the radiator. Options for the advanced electronics include "controller-on-a-chip" designs and integration into the pressure vessel to simplify thermal control and spacecraft bus interfaces.

Thermo-acoustic devices using traveling wave tube technology may offer mass and reliability improvements over the free-piston Stirling design. Los Alamos National Laboratory (LANL) has developed and tested thermo-acoustic converters that have demonstrated high heat to (acoustic) power efficiencies. This approach is described in Sec. 5.2.8.

Version 3.0 is not defined, but might use high temperature, refractory materials in the hot-end to increase the heater head temperature and achieve higher conversion efficiency. Hot-end temperature would be increased from 923 K to 1200-1400 K resulting in a higher heat rejection temperature and smaller radiator (per unit power output). The system efficiency might be improved from 25% in Version 2.0 to 31-35% in Version 3.0, resulting in increased power output. The high temperature Stirling configuration might be particularly useful in a Venus surface mission where ambient temperatures exceed 600 K.

5.2.4.2 Operating Performance

A summary of performance estimates for four Stirling system configurations is given in Table 5.2-1. All the designs are assumed to include two GPHS modules and two converters, similar to the DOE/LMA design concept shown in Figure 5.2-5. The system power levels are based on 28 volt dc output from the controller at beginning-of-mission (BOM). System efficiency is calculated based on dc system power divided by GPHS heat at BOM (e.g., $109 \div (2 * 243) = 22.4\%$). The heat source mass is assumed constant for all four configurations. The heat source mass includes two GPHS modules, heat source supports, multi-foil insulation, housing, and covers. The Stirling mass includes two converters plus pressure relief devices, gas management valves, electrical connectors, seals, and fasteners. Contingency is calculated based on 20% of the system mass excluding the heat source. These estimates become increasingly uncertain as one moves from versions 1.0 to 3.0.

SiGe RTGs heat rejection temperatures is ~500 K. Stirling engines reject heat at ~350 K. The Stirling would require four times the radiator area to reject the same amount of heat to a 250K sink temperature. However, the Stirling engine is four times as efficient as the RTGs so only 1/4 as much heat is rejected, resulting in similar sized radiators if the Stirling radiator temperature is at 350 K.

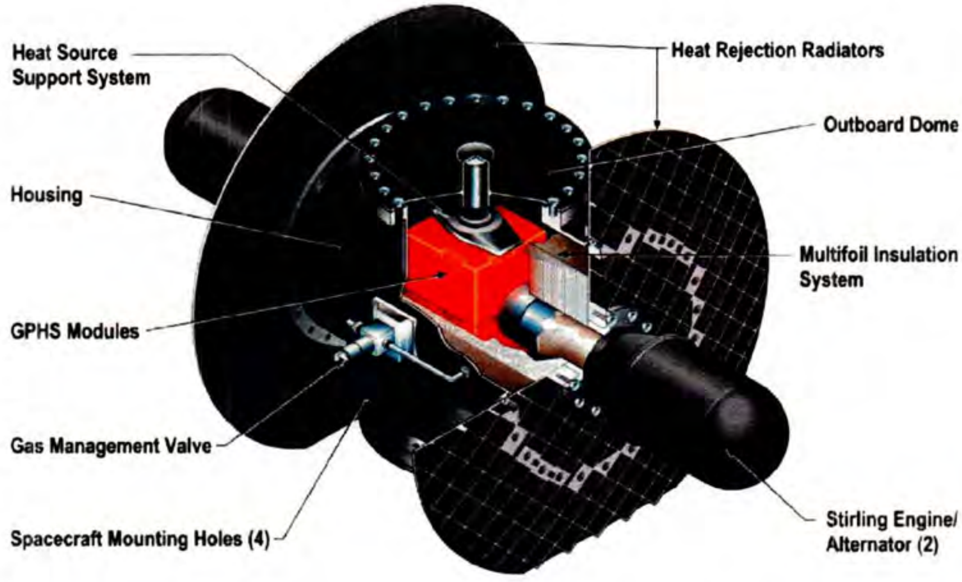


Figure 5.2-5. LMA 109 Watt BOM Stirling Generator Concept

Table 5.2-1. Performance Projections for 100 watt-class Stirling ARPS

Parameter	Version			
	1.0	1.1	2.0	3.0
Heater Head Temperature, K	923	923	923	1200-1400
BOM System Power, W (dc)	109	124	124	172
BOM Conversion Efficiency*	22.4%	25.5%	25.5%	31-35%
System Mass, kg	26.6	20.0	16.0	20.0
Heat Source Mass, kg (incl. 2 GPHS modules)	7.5	7.5	7.5	7.5
Stirling Mass, kg (incl. 2 Converters)	10.0	6.2	4.0	6.2
Radiator Mass, kg	2.6	2.6	2.3	2.6
Controller Mass, kg	4.0	2.0	1.0	2.0
Contingency, kg	2.5	1.7	1.2	1.7
System Specific Power, W/kg	4.1	6.2	7.8	8.6
* assuming 6% GPHS heat loss through thermal insulation				

5.2.4.3 Power Level Scaling

Stirling converters have proven to provide high efficiency over a wide range of output power as shown in Figure 5.2-6. The 10 watt Radioisotope Stirling Generator (RSG), shown in Figure 5.2-7 was developed for terrestrial applications under DOE contracts and has operated at 873 K hot side and 293 K cold side over 60,000 hours (6.8 years) without a regenerator with no maintenance and no performance degradation. Higher power terrestrial Stirling converters are operational. STC builds a 350-watt converter for remote generator applications. Other Stirling manufacturers have developed multi-hundred watt cryocoolers that use a motor driven Stirling cycle for cooling. System specific power should improve from the values indicated in Table 5.2-1 with higher power due to favorable scaling characteristics. The approximate limit for flexure bearing technology is in the 3 kW range. Above 3 kW, gas bearings or a combination of gas and flexure bearings would be required. Stirling converters developed during the 1980's for the SP-100 Space Reactor Program produced 12.5 kW per piston.

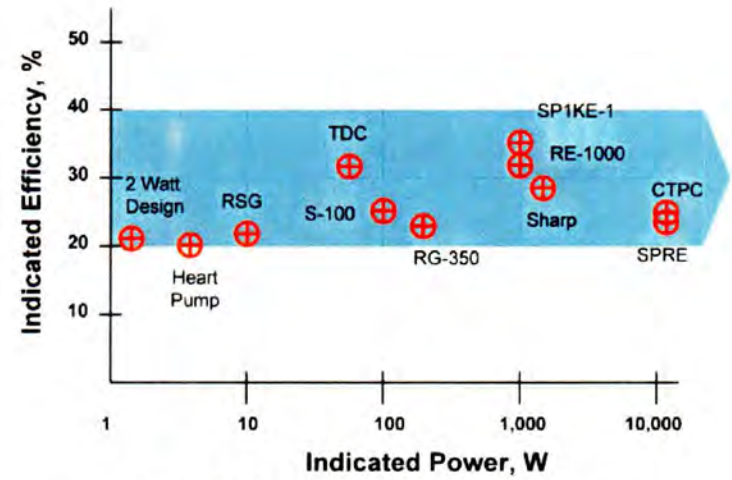


Figure 5.2-6. Stirling Efficiency Trends

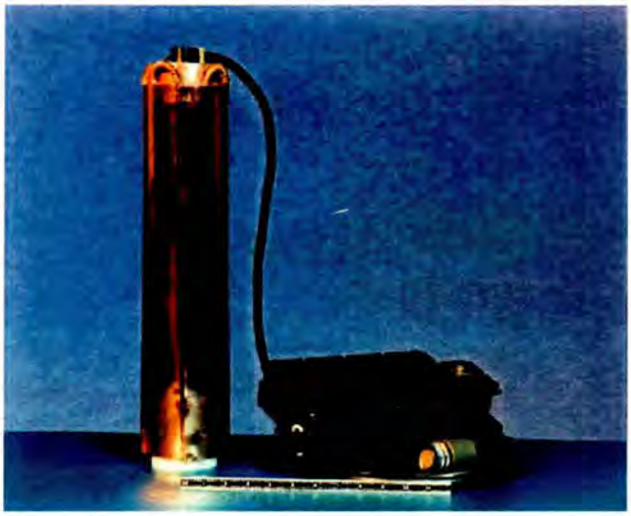


Figure 5.2-7. 10 Watt RSG

5.2.5 Technical Status

A summary of Stirling technology status is given in Table 5.2-2. DOE and NASA have supported the Stirling technology since 1997. DOE and NASA funded the design and development of the TDC through contracts with STC and OSC totaling about $1.3 M. The DOE NASA effort resulted in fabricating four TDC and demonstrating predicted power and efficiency. NASA GRC's contribution was two Small Business Innovative Research (SBIR) tasks. The first SBIR developed the technology to synchronize two thermodynamically independent converters. The second SBIR developed and demonstrated an adaptive vibration reduction system (AVRS). The AVRS motor reduces the balanced vibration from opposed converters by 500x using only 2 watts of power and the unbalanced vibration resulting from one of the two converter shutdown by 50x using only 7 watts of power. The vibration from a pair of opposed 55 watt converters is 20x below the NASA JPL Outerplanet/Solar Probe (OP/SP) jitter requirement for a Europa Orbiter spacecraft layout.

Table 5.2-2. Technology Status Summary

Characteristic	Achieved to Date	Probably Achievable
Converter Efficiency	29% at Th=923K and Tc=353K; 26% at Th=923K and Tc=393K	>30% at Th=923K and Tc<333K
Lifetime and Reliability	6 TDC units completed, 100's of hrs on TDCs; 6.8+ yrs on 10 W RSG	14 yrs based on independent life & reliability study
Specific Power	55 W TDC 5.2 kg; Estimated 4.1 W/kg System	Advanced TDC 2.5 kg; 6.2 W/kg System
Spacecraft Interface Issues	AC-DC power converter required; Low temp radiator required; Vibration 20x below required for EO; EMI/EMC acceptable for EO, PKE	EO radiation survivable organic replacements; EMI/EMC acceptable for Solar Probe
GPHS I/F Issues	TDC heater head requires radial radiation heat input from GPHS	Alternate Stirling designs may simplify heat input
Scalability	10 W for terrestrial isotope applications; 350 W terrestrial not space rated; 12.5 kW/piston SP100	10 W to 3 kW (flexures); 1 kW to 25 kW (gas bearings)
Verification	Electrical heater and liquid (ethylene glycol) coolant	GPHS heat input and space radiator simulation

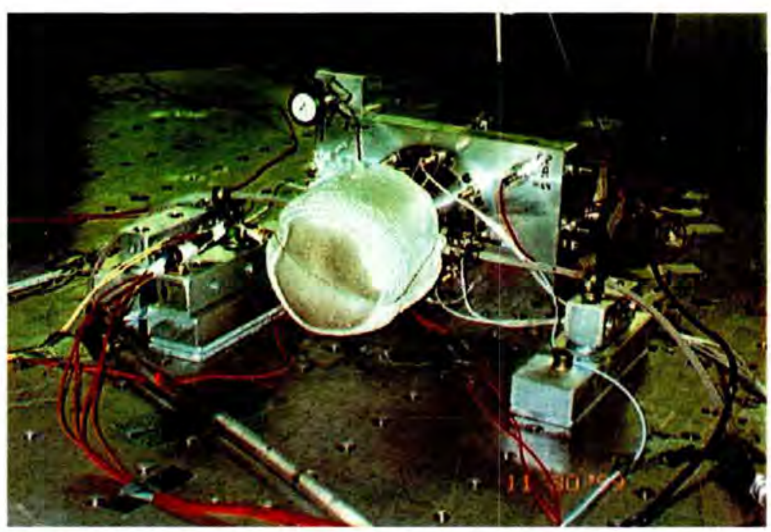


Figure 5.2-8. TDC in Launch Load Test Configuration

An assessment team from DOE, GRC, JPL, and industry was formed in August 1999 to review the Stirling technology status. The team recommended dynamic launch load evaluation, EMI/EMC characterization, and performance mapping of the 55-watt TDC. Launch vibration load testing was conducted at GRC in November 1999. A single TDC (S/N 001), shown in Figure 5.2-8, was successfully operated at full power and stroke in two different orientations through a series of random vibration tests representing maximum launch conditions. Following the launch load test, an EMI/EMC test was performed at GRC with two TDCs (S/N 001 and 002) in an opposed pair configuration as shown in Figure 5.2-9. The converters met both radiated and magnetic emission requirements as specified for the Europa Orbiter and Pluto Kuiper Express, but exceeded the magnetic field requirement for the Solar Probe. A post-test disassembly and inspection of TDC S/N 001 showed no apparent damage or change in the physical condition of the converter as a result of the launch load and EMI testing. Performance testing was conducted at STC using an electrical heater and liquid coolant for the heat sink. Two TDCs were tested (S/N 003 and 004) at a heater head temperature of 923 K (650°C) and 353 K (80°C) and 393 K (120°C)heat rejection temperatures. The 353 K rejection temperature test resulted in 62 watts ac power output and 29% converter efficiency. The 393 K rejection temperature test resulted in 56 watts ac power output and 26% converter efficiency.

The technology assessment team also requested additional analysis regarding radiation survivability, controller functionality, life and reliability, and fault tolerance. The radiation study identified several organic material replacements needed in the TDC to survive Europa's radiation environment. The controller study defined basic requirements and determined that the circuit could be designed with existing conversion and regulation techniques and implemented using the NASA X2000 electronic technology development program. Controller efficiency was estimated at 85 to 90% and mass was estimated at 3 to 4 kg, depending on redundancy requirements. The life and reliability assessment included a component level failure mode and effects analysis (FMEA) which addressed all phases of the TDC Stirling converter operational life from fuel loading to mission completion and identified a critical item list (CIL). The fault tolerance study reported that the loss of one converter in an opposed pair configuration would result in excessive temperatures leading to the failure of the second converter. That conclusion lead to the concept of three 100 watt generators per spacecraft to meet a 200 watt requirement, allowing one generator failure to occur without impacting mission success. Alternatively, a secondary GPHS heat removal system would be required for cooling of the heat source.

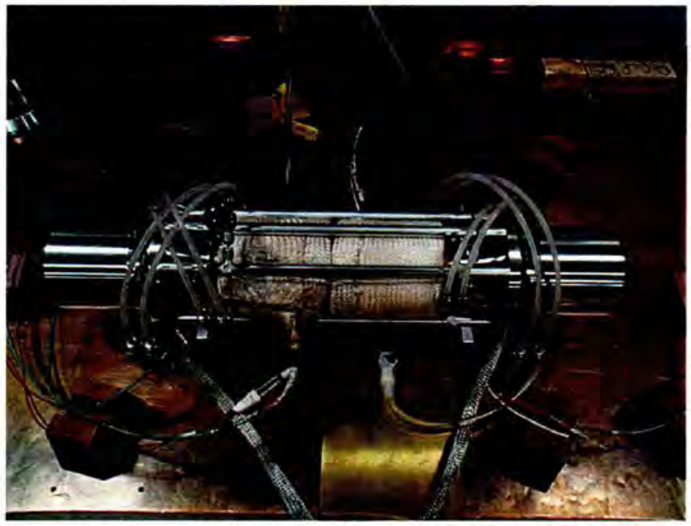


Figure 5.2-9. TDC Pair in EMI/EMC Test Configuration

With the successful completion of this initial technology assessment, DOE and NASA initiated the procurement of six new TDC converters for various testing and evaluation tasks. The new Stirling converters incorporate the organic material replacements identified during the radiation survivability study along with other modifications. Four converters (S/N 005, 006, 007, and 008) have been delivered to GRC and two are currently being tested. Two of the DOE units (S/N 009 and 010) are ultimately planned for extended life testing. The FY'00 cost to support technology assessment and fabricate four new TDC converters was $3.1 M.

5.2.6 Technical Issues

NASA GRC has initiated a technology development project to advance the flight readiness of Stirling converters. The initial objective is to bring the TDC converter to a NASA TRL 5 in support of the Version 1.0 flight design. The GRC project includes the following tasks:

- Independent performance verification and mapping of TDCs
- Heater head and joint accelerated lifetime performance technology development
- Aging characterization of NdFeB permanent magnets and thermal/electromagnetic FMEA of linear alternator
- Converter launch environment characterization
- EMI/EMC reduction and characterization
- Evaluation of converter organic materials for radiation survivability
- Support DOE reliability evaluation
- Two-dimensional CFD performance code

The GRC technology effort is complemented by a DOE Stirling RPS conceptual design integration activity. Three contractors (Lockheed Martin, Boeing/Rocketdyne, and Teledyne Brown) were selected to each prepare a conceptual Stirling RPS design and a proposal for developing a 100-watt class RPS for the Europa Orbiter mission. The plan was to evaluate the proposals and select one contractor to develop Stirling-ARPS for EO. Many of the Stirling-ARPS configuration, operational, and reliability issues must be addressed by a system integrator. The critical tasks include integration of the GPHS and Stirling converter, integration of the radiator within the spacecraft environment and life testing and/or accelerated life testing of Stirling converters. Since the design competition was originated, changes were made in NASA mission plans, and EO elected to baseline an existing RTG. As a result, it is likely that the system integrator may be asked to redesign the system for use on Mars.

The new technology tasks required to achieve the Stirling Version 1.1 goals would focus on a low-mass alternator and capacitor-free controller. The advanced alternator would use SmCo magnets, requiring similar aging tests and thermal/electromagnetic analysis as planned for the NdFeB magnets for Stirling version 1.0. Other modifications to the TDC may result from launch load tests with the pressure cavity exposed to identify areas for structural margin improvement. The advanced controller development requires electrical design and breadboard demonstration.

The details of the Version 2.0 technology development have not yet been selected. A competitive solicitation is anticipated to identify promising concepts for advanced converters, radiators, and electronics. This activity is expected to produce breadboard and engineering model prototypes that can be evaluated against potential flight system requirements. The thermo-acoustic approach (Sec. 5.2.8) is a strong candidate.

5.2.7 Budgets and Schedules

The total funding allocated to Stirling technology development between FY'97 and FY'00 was \$4.4 M. The three Stirling design versions are at various levels of maturity and would require different resources to achieve technology readiness. Table 5.2-3 presents a summary of the estimated budget and schedule required for each version to achieve TRL 5. Version 1.0 technology could be completed in two years for about \$4.5 M. The costs of the system integration contract and follow-on flight development system are not included in this technology budget. The costs and schedules for higher versions are less accurately known. It is estimated that an \$8 M three-year technology development effort would be required to achieve TRL 5 for Version 1.1. These costs include the design and fabrication of an advanced TDC and controller for performance validation and a fully-integrated system demonstration with electrically heated simulated GPHS modules and a space radiator operating in a vacuum.

Table 5.2-3. Stirling Budget and Schedule Summary

Stirling Version	NASA TRL	\$M	Yrs	Technical Challenges
1.0	3 or less	N/A	N/A	
	4	N/A	N/A	
	5	4.5	2	Space flight readiness, life testing
1.1	3 or less	N/A	N/A	
	4	4.0	1	Lightweight alternator, capacitor-free controller
	5	4.0	2	Integrated system test with GPHS simulation
2.0	3 or less	1.5	1	Radiator materials, alternate converter designs
	4	4.5	2	Breadboard units, performance validation
	5	6.0	2	Engineering models, life testing

It is estimated that Version 2.0 technology development would take approximately five years and $12 M to achieve TRL 5. The present Cross Enterprise New Research Announcement (NRA) project with TRW and LANL on thermo-acoustic conversion could contribute to this development. The Version 2.0 activity is expected to produce breadboard and engineering model prototypes that can be evaluated against potential flight system requirements.

5.2.8 Thermo-Acoustic Stirling Converter

A considerable amount of work has been done over the past 15 years in developing cryocoolers for cooling infrared sensors in space. The leading technology in this area has been based on the Stirling cycle. Over the past 5-6 years, the space cryocooler has changed emphasis from conventional Stirling cycle coolers to pulse tube coolers. It is possible that the advantages of the thermoacoustic approach for cryocoolers might also have similar implications for heat engines. Thermodynamically, the thermo-acoustic converter is identical to the Stirling converter. The thermo-acoustic system replaces the traditional Stirling displacer with a traveling pressure wave, eliminating the moving part in the hot-end. A power piston is required to produce mechanical work that is converted to electricity in a linear alternator. By eliminating the displacer, some inefficiency is removed including shuttle, appendix gap, and gas leakage losses. However, the thermo-acoustic driver introduces inefficiencies such as with buffer tube conduction and radiation losses.

Los Alamos National Laboratory constructed and operated a high power (kilowatts) steel thermal acoustic heat engine at a hot side temperature up to 725°C with water cooling the cold side at 15°C. The system is filled with helium gas at ~500 Psia. When heat is applied to the hot heat exchanger, an acoustic wave is created in the cavity at the resonance frequency of 80 Hz. Even though the gas forms a continuum through which acoustic power is transferred when heat is converted to an acoustic wave, we may think in terms of a hypothetical packet of gas that is driven back and forth through the regenerator by the acoustic pressure variations. A nearly stationary packet of gas at the hot heat exchanger absorbs heat and expands at constant temperature (T_h). The expanding gas passes through the regenerator starting at the hot end, and is cooled by the regenerator as it passes down the thermal gradient. As the packet passes through the regenerator, it cools and contracts. The nearly stationary gas at the cold end rejects heat to the cold heat exchanger at constant temperature (T_c), causing it to be compressed. The compressed gas now expands back through the regenerator up the thermal gradient and recoups the heat previously deposited. These cyclic changes take place at 80 hertz frequency. A voice coil mounted on the resonator is used to convert acoustic power to mechanical power to drive a linear alternator. S. Backhaus and G. W. Swift describe this technology in Nature 399, 335-338 (1999), and J. Acoust. Soc. Amer. 107, 3148-3166 (2000).

Thermo-acoustic engine alternator technology for space applications would operate at similar temperatures as the free-piston Stirling based on equivalent material constraints, resulting in comparable overall conversion efficiencies. Improved reliability may be possible by eliminating failure modes associated with the displacer. Another potential benefit for thermo-acoustic technology is reduced converter mass consistent with the projections for Version 2.0.

Recently, LANL and TRW won a 2-year, $1.3 M NASA Cross Enterprise Technology Development award in the FY'01 NASA NRA. They plan to develop and test a 100-watt class prototype using similar electronics to those developed by TRW for cryocooler applications.

5.2.9 Recommendations

Stirling converter technology is a proven high efficiency option for ARPS. Excellent converter technology progress was achieved from 1997 through 1999. The team recommends the advanced Stirling technology development be continued to keep the high efficiency and increase the specific power. Stirling Version 1.0 will demonstrate the technology readiness. Stirling Version 1.1 will develop a low mass alternator and controller for flight application that retains technology heritage from version 1.0. Stirling Version 2.0 will develop a lower-mass, long-life Stirling technology that may meet several future deep space and Mars missions requirements.

5.3 Thermoelectric Technology

5.3.1 Summary

Thermoelectric (TE) devices for converting heat to electrical power have played an important role in past NASA space exploration and planetary missions, and could play an important role in the next 10-20 years.

Current RTG technology encompasses two possible unicouples: SiGe and PbTe-TAGS. Both have been used in the past. The Viking Landers on Mars used the PbTe-TAGS approach where it has the advantage of not requiring venting of He. Since Viking, all NASA Code S missions that employed RTGs, used SiGe technology operating in a vacuum. Because of the extensive experience with SiGe RTGs from 1975 to 2000, SiGe technology is regarded as the current state of the art for RTG technology. Unicouples represent the one flight-qualified technology for RPS even though they have relatively low conversion efficiency.

This section reviews the present status and development of the advanced segmented thermoelectric (TE) technology being developed at JPL and Hi-Z, Inc. By segmenting the n- and p-legs of a unicouple into sections made from different materials, each with maximum figure-of-merit over a moderate temperature range, higher efficiencies are attainable than with single-material unicouples. Segmented TE technology can potentially be developed as a converter for GPHS with an overall system efficiency in excess of 10%.

The team also briefly reviewed current development in a more advanced technology, Quantum Well Thermoelectric (QW-TE), being developed at the University of California-Los Angeles (UCLA) and at Hi-Z technology, Inc. QW-TE technology has the potential for much higher efficiency (>30%), but is at a very early stage of research and development.

5.3.2 Segmented Thermoelectrics

5.3.2.1 Introduction

Radioisotope thermoelectric generators (RTGs) have converted heat from radioisotopes to electrical power for many NASA space exploration and planetary missions. Since 1961, the United States has launched 42 RTGs on 25 spacecraft for various NASA and Department of Defense missions. These missions include high and low Earth orbits, the surfaces of the Moon and Mars, Jupiter Orbit, the Sun, and fly-bys of all the solar system planets except Pluto. The RTG has performed as predicted for as long as 25 years. The SiGe and PbTe RTGs are proven technologies for long life space electrical power source aboard spacecraft. RTGs could be used over the next several decades for space power.

The DOE has developed a general purpose heat source (GPHS) that will be used as the building block for supplying radioisotope heat for future advanced converters. The GPHS module has the physical form of a rectangular block approximately 10 cm x 9 cm x 3 cm. Each such module has a mass of ~1.45 kg, including 440 g of Pu-238. The module produces 250 watts of thermal power at beginning of life (BOL) and the plutonium decays with an 87.8-year half-life. The GPHS module is designed to operate in a vacuum for outer planet missions where helium is vented to space. For use on Mars or in other planetary atmospheres the GPHS modules and converter would have to be sealed to maintain a vacuum or an inert gas inside. Disposing of the helium produced by radioactive decay in a sealed system poses a technical challenge.

SiGe and PbTe-TAGS RTGs have been used as space power sources. PbTe-TAGS RTG requires an inert gas pressure over the PbTe-TAGS to inhibit sublimation. It does not require venting the helium. Its disadvantage is that the hot side temperature is limited to ~800 K. Therefore, PbTe-TAGS RTGs have relatively low specific power. The SiGe-RTG has the advantages of higher hot side and cold side operating temperature (~1000 K) and low mass material, therefore higher specific power than PbTe-TAGS RTG. The SiGe RTG is designed to operate in a vacuum, requiring venting helium to space vacuum. The Viking Landers on Mars used the PbTe-TAGS RTG where its sealed design had distinct advantages. Since Viking, all NASA outer planet missions used SiGe RTGs from 1975 to 2000. Because of the extensive experience with SiGe RTGs it is regarded as the current state of the art RTG technology. Nevertheless, PbTe-TAGS technology may also be a viable option for missions operating in an atmosphere.

5.3.2.2 Description (How It Works)

A thermoelectric (TE) converter consists of a p-type and a n-type semiconductor elements (or legs), heated at one end, thermally in parallel, and connected electrically in series. The n- and p-legs are thermally insulated on the sides and are the same length.

The cross-sectional area of the n- and p-legs are different in order to compensate for the difference in their thermal conductivities. Heat flows from the source through the n- and p-legs to the heat sink (Figure 5.3.2-1). The electric power generated by the Seebeck effect in the TE converter depends on both the temperature of the source and temperature differential between the source and the sink, (T_H - T_C). The electrical potential developed in the n- and p-legs by the Seebeck effect, V_S, is directly proportional to the temperature difference, the proportionality constant being the net Seebeck coefficient, α. The net electric potential across the external load, V_L, equals V_S minus the internal electric losses in the p- and n-legs and the electric leads to the load. In addition to electrical losses due to the finite electrical conductance of the unicouple materials, there is a direct thermal short from T_H to T_C through these materials, producing a heat loss that reduces thermal efficiency. Thus, in order to maximize the performance of a TE converter, it is desirable to use TE materials that have high Seebeck coefficients, low electrical resistivity, ρ, and low thermal conductivity, k. The efficiency of a TE converter is the ratio of the electric power produced on the external load to the thermal power supplied by the heat source.

The conversion efficiency is primarily dependent on a figure-of-merit, Z, for the converter materials in the p- and n- legs at the operational temperatures involved. The TE material figure-of-merit has units of K^{-1} and is defined as:

$$Z = (\alpha^2/\rho k)$$

The figure of merit, Z, is quite temperature-dependent for any unicouple, showing a maximum at some temperature. Since the unicouple material spans the entire temperature drop from T_H to T_C, the efficiency of a device will depend on the variation of Z over the range from T_H to T_C.

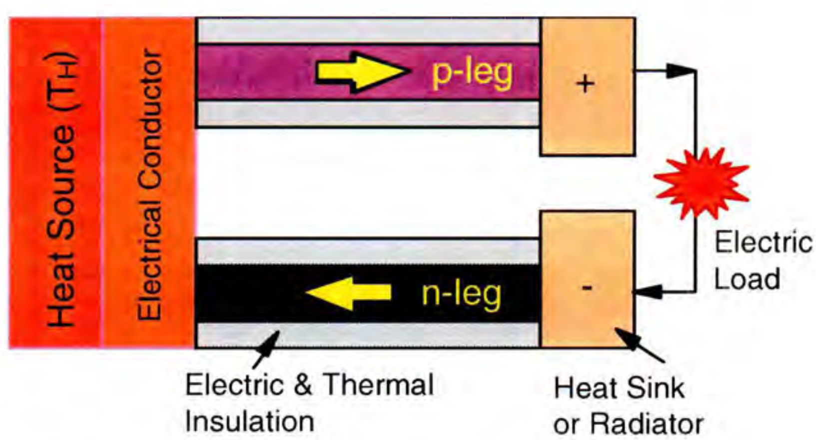


Figure 5.3.2-1. Schematic of a Thermoelectric Converter

The thermoelectric device can be designed to either maximize the conversion efficiency by minimizing (ρk), or to maximize the power output by matching the internal electrical resistance of the converter to be equal to the external load resistance. In the latter case, the power output is higher and the thermal efficiency is lower. In the former case, the thermal efficiency is higher and the power output is lower. The primary emphasis for RPS applications is on high efficiency to reduce the amount of radioisotope fuel needed. In the high efficiency mode, the expression for efficiency depends on two important quantities:

- The average of ZT over the temperature range involved, which determines what fraction of Carnot efficiency can be achieved.
- The Carnot efficiency $\{(T_H-T_C)/T_H\}$, which depends on the source and sink temperatures.

Figure 5.3.2-2 shows the figures of merit for several TE materials as a function of the temperature.

The problem for high efficiency thermoelectric conversion is that whereas one desires as high a Carnot efficiency as possible, thereby requiring a large temperature difference from T_H to T_C, the average ZT over this large temperature difference is not very large. Therefore, the overall efficiency is low because ZT is not large.

Segmented Thermoelectrics

Each thermoelectric material, whether n-type or p-type, exhibits a maximum in ZT at some temperature. If a single material is used in each leg of the unicouple, the effective value of ZT will be an average over the temperature range. The average ZT will be considerably less than the maximum ZT. If each leg of the unicouple is segmented so that a large thermal gradient from T_H to T_C is established down the leg, the temperature range over each segment will be considerably less. If materials can be found with high ZT over the small temperature range of each segment, it may be possible to achieve high ZT over the entire temperature range from T_H to T_C. This will increase the thermoelectric conversion efficiency by a considerable amount. For a unicouple operating between 1200 K and 473 K, the dependence of efficiency on average ZT is given in Table 5.3.2-1.

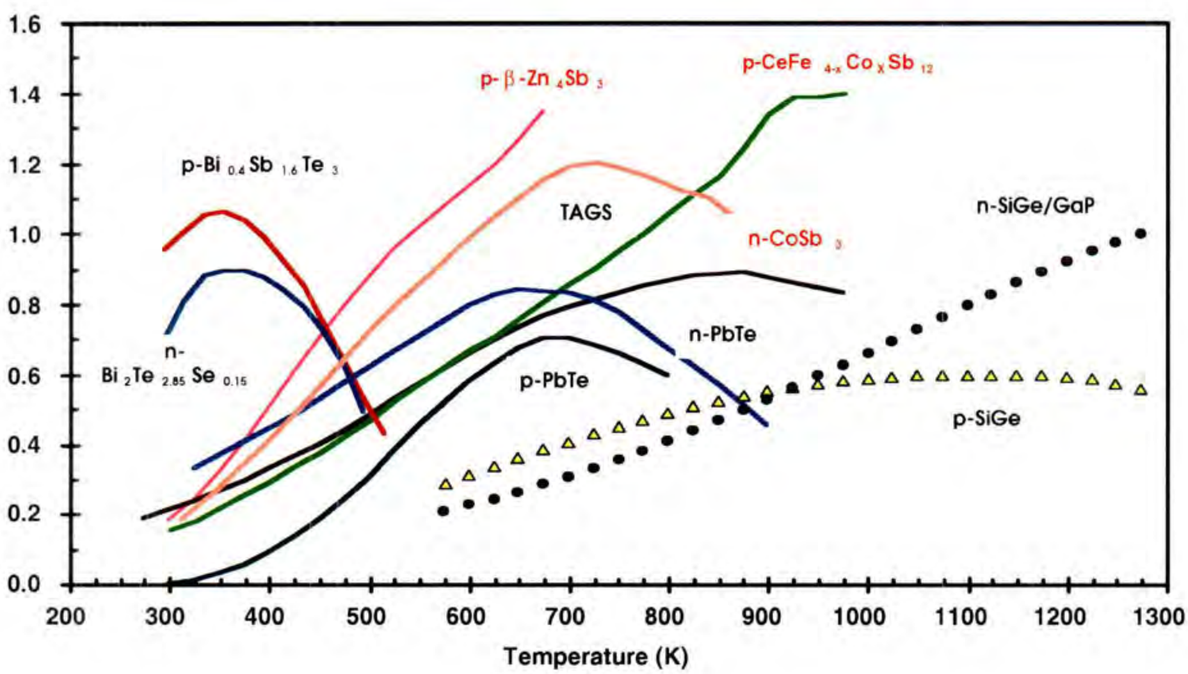


Figure 5.3.2-2. Thermoelectric figure-of-merit, ZT, as a function of temperature for state-of-the-art SiGe and Bi_2Te_3-based alloys and the new Zn_4Sb_3, $CoSb_3$ and $CeFe_4Sb_{12}$ compounds.

Table 5.3.2-1. Thermoelectric efficiency on average ZT for $T_H = 1200$ K and $T_C = 473$ K

ZT (avg.)	Efficiency	ZT (avg.)	Efficiency
0.2	0.039	1.3	0.164
0.3	0.055	1.4	0.171
0.4	0.070	1.5	0.178
0.5	0.084	1.6	0.185
0.6	0.097	1.7	0.191
0.7	0.108	1.8	0.197
0.8	0.119	1.9	0.203
0.9	0.129	2	0.209
1	0.139	3	0.215
1.1	0.148	4	0.221
1.2	0.156	5	0.227
		6	0.233

The technical challenges in developing segmented thermoelectrics include the following:

1) Finding materials with high ZT over various temperature ranges that are compatible with one another (e.g., similar coefficients of thermal expansion),
2) Developing methods for joining segments with very low electrical resistivity, while preventing inter-diffusion of atoms from one segment to another,
3) Optimizing the relative lengths and areas of the segments, and
4) Matching load resistance.

An illustration of a first-generation advanced segmented thermoelectric unicouple incorporating new materials developed at JPL is shown in Figure 5.3.2-3. The actual device is shown in Figure 5.3.2-4.

5.3.2.3 Configuration and Materials of a Segmented Thermoelectric Unicouple

A semi-empirical model has been developed at JPL to optimize the geometry of the legs and calculate the efficiency of a segmented converter for any combination of materials for which properties are known. However, even a modest contact resistance of interconnects between the thermoelectric segments in the n- and p-legs can dramatically reduce the efficiency of the converter. Calculations showed that a low contact resistance, less than about 20 $\mu\Omega cm^2$, is required to achieve high conversion efficiency.

The concept of a segmented TE generator has been under development at JPL and at High-Z Technology of San Diego. The segmented converter shown in Figure 5.3.2-4 is being developed at JPL. In this converter, each section has the same current and heat flow rate as the other segments in the same leg. Thus, in order to maintain the desired temperature profile (i.e., keeping the interface temperatures at their desired level) the geometry (length and cross section area) of the n- and p-legs must be optimized. Specifically, the relative lengths of each segment in a leg must be adjusted, primarily due to differences in thermal conductivity, to achieve the desired temperature gradient across each material.

Preliminary fabrication and extensive materials development have been conducted at JPL under the sponsorship of Defense Advanced Research Projects Agency (DARPA). Several segmented converters have been fabricated using a combination of powder metallurgy and brazing techniques. One of these converters is shown in Figure 5.3.2-4. Further research is required, however, to develop low contact resistance bonds for some of the junctions, before the predicted high thermal and electrical performance of the converter can be tested.

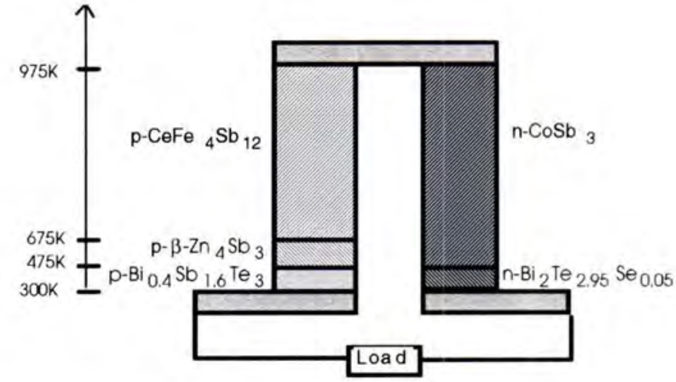


Figure 5.3.2-3. Schematic of First Generation Advanced Segmented unicouple at JPL

Table 5.3.2-2 lists a summary of the experimentally measured thermoelectric properties and the dimensions of the n- and p-legs of the segmented converter developed at JPL. Table 5.3.2-3 lists the predicted performance of an optimized, segmented converter shown in Figure 5.3.2-4.

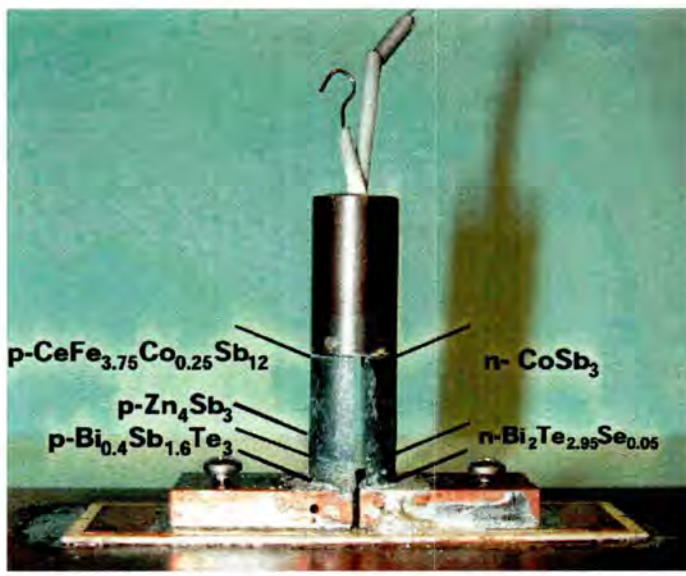


Figure 5.3.2-4. First Generation Advanced Segmented Unicouple Tested at JPL

As indicated earlier, in order to achieve high converter efficiency, not only must the electric contact resistance be lower than 20 μΩcm^2, but in addition, the bond between various segments must be mechanically stable and function as an effective diffusion barrier at operating temperatures. The latter is required to prevent material diffusion across the junction between any two bonded materials, which can potentially deteriorate the thermoelectric properties of these materials, and, hence of the converter with time. Tests have been conducted at JPL with and without a diffusion barrier. The barriers tested are ~100-micron thick Ni, Ta, Pd, and $Pd_{70}Ag_{30}$ foils between the two lower segments of the p-leg in Figure 5.3.2-4 (Zn_4Sb_3 and $Bi_{0.4}Sb_{1.6}Te_3$). The Ni, Ta, and Pa have coefficients of thermal expansion (CTE) of 13.3, 6.5, and 11.6, respectively. These values are much smaller than those of Zn_4Sb_3 (19) and $Bi_{0.25}Sb_{1.75}Te_3$ (19), which could cause cracking, as has been confirmed with test results. With a Ni foil, the electrical contact resistance at room temperature was nil; however, a crack developed in the center of the joint when heated up to 423 K for a few hours. In addition, significant diffusion of materials occurred across the Ni foil. When a Ta foil was used, the contact resistance was very high ~200 μΩcm^{-2}, but no cross material diffusion was found after operating for 7 days at 423 K; however, post-test examination of the junction microstructure revealed a small crack. Test results identified the $Pd_{70}Ag_{30}$ alloy as a promising bonding material because of its low contact resistance (~0.5 μΩcm^2) at room temperature and the absence of any cross diffusion after operating for 9 days at 423 K. In addition, the $Pd_{70}Ag_{30}$ alloy has a CTE (16) that is close to those of Zn_4Sb_3 and $Bi_{0.25}Sb_{1.75}Te_3$ of 19 and 16, respectively.

Table 5.3.2-2. Experimental thermoelectric properties and dimensions of the n- and p-legs of the segmented converter developed at JPL

TE Material	Electrical Resistivity (mΩ.cm)	Seebeck Coefficient (μV/K)	Thermal Conductivity (mw/cm K)	Relative length
p-Ce filled Skutterudite	0.88	186	29.0	0.740
p-Zn_4Sb_3	2.60	175	6.5	0.126
p-$(Bi/Sb)_2Te_3$	1.02	196	13.8	0.134
n-$CoSb_3$	1.42	-243	5.39.7	0.854
n-Bi_2 $(Te/Se)_3$	1.29	-195	15.6	0.146

Table 5.3.2-3. Predicted optimum design and performance parameters by JPL of a segmented thermoelectric converter

Parameter	Value	Parameter	Value
Overall length (mm)	10	Load Resistance (mΩ)	4.31
Cross-section area of p-leg (mm^2)	10	Electric current (A)	5.39.1
Cross section area of n-leg (mm^2)	83.8	Load electric power (W)	6.58
Device resistance (mΩ)	4.31	Conversion efficiency (%)	15.06

In very recent work at JPL, experiments were carried out on a "skutterudite-only" unicouple. This material is one of three segments on the p-side for a segmented design employing two segments on the n-side. This design is expected to produce a conversion efficiency of 16%. Using just a p-skudderite matched to n-type $CoSb_3$, a conversion efficiency of nearly 10% was attained between 590°C and 90°C. This is very encouraging since the experimental numbers very closely match the expected behavior from the performance model. This implies that the contact resistance is low, and the thermoelectric properties of the individual materials were measured accurately. The results are shown in Figure 5.3.2-5.

High-Z Technology (Hi-Z) has also been developing segmented thermoelectric converters using skutterudite materials that are both similar and different from those currently under investigation at JPL. They are also developing different fabrication techniques. Hi-Z uses thermal spraying for contact-metallization and fine steel mesh at joints. Figure 5.3.2-6 shows a schematic of the segmented converter being developed at Hi-Z. The work performed at Hi-Z is also sponsored by DARPA. Because of its high resistivity in contact with the skutterudite and the rapid sublimation at high temperatures (~798 K), the zinc antimonide (Zn_4Sb_3) was eliminated from the p-leg. Fe powder was found to give very low contact resistance between $Ce(FeCo)_4Sb_{12}$ and p-Bi_2Te_3. Hi-Z is in the process of developing a segmented bi-couple converter having two p- and n-legs in a cross section of 5x5 mm and initial performance and life testing is planned. In these tests, they will measure the electric power output across a matched load resistance, as a function of hot side temperature of 873 K and for a cold temperature of 298 K. Other work planned under existing funding includes developing a commercial module. This effort includes fabrication of n- and p-legs by co-pressing powders and the assembly and initial testing of the modules.

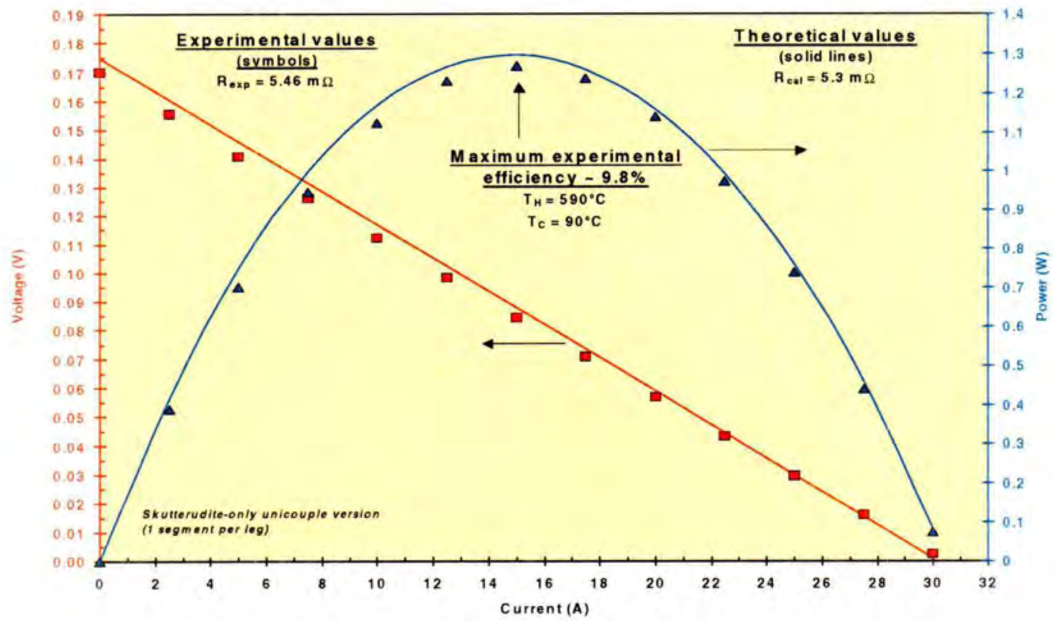


Figure 5.3.2-5. Recent data on a unicouple using a single skudderite for the p-leg and $CoSb_3$ for the n-leg

5.3.2.4 Technical Status of Segmented Thermoelectrics

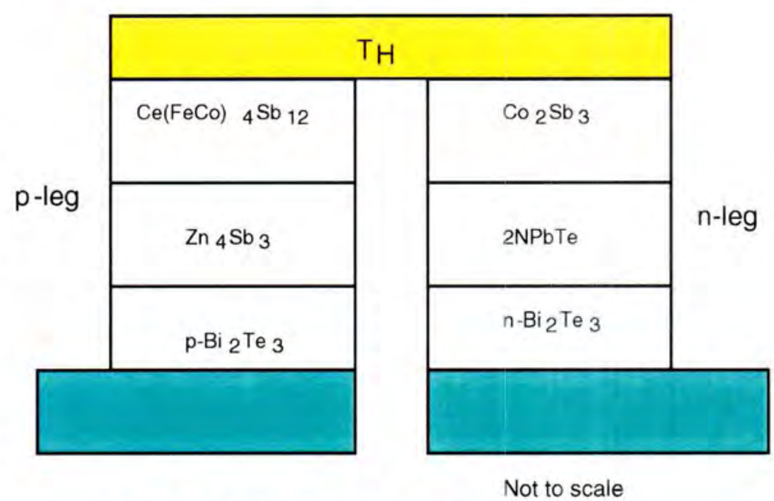


Figure 5.3.2-6. Segmented Thermoelectric Converter Being Developed at Hi-Z, Inc.

The segmented TE converter technology has been under development at JPL and Hi-Z with DOD funding at modest levels for many years. NASA has not supported this activity. Because DOD needs are different from those of ARPS, the segmented TE work has concentrated on segments for a lower temperature range (950 K to 350 K) than is desired for ARPS (1200 K to 500 K). Although considerable progress was made, there still remain the challenges of finding compatible materials for 950 K to 1200K, and suitable joining techniques. The assessment team is of the opinion that it is worth investing in this technology with the hope that these problems might be solved in several years. Based on the current performance predictions of the segmented thermoelectric converter being developed at JPL, a conversion efficiency of ~15% is possible when operated between 1200 K and 500 K. This efficiency would reduce the mass of plutonium fuel by more than a factor of two compared to conventional SiGe RTGs. In addition to the high efficiency, TE converters have the potential of significant reductions in total RPS mass compared with the SiGe RTG (Table 5.3.2-4).

Additional advantages of the segmented converter technology are its modularity and possible miniaturization to very low power levels of tens and hundreds of milliwatts electric.

The next milestone in the development of segmented thermoelectric converters would be to perform accelerated life tests and investigate integration issues of the converter with the GPHS for high power RPS (Table 5.3-5).

Should these technology goals be achieved—high performance, high efficiency, low mass, and long-life—an engineering model radioisotope power system with segmented thermoelectric converters could be built by year 2006-2008. Such RPSs are required for several NASA missions planned this decade, as described in Section 2. A number of these missions call for electrical power requirements in the 50 to 200 watts range and 6 to 10 years mission duration. The estimates of the electric power requirement for the Cryobot Europa mission are from 25 to 50 watts. Considering the cold environment of Europa (~183 K), the cold-side of the segmented TE converters could conceivably be maintained at or below room temperature, which is feasible when using Bi_2Te_3-based alloys as the lower segments. The resulting increase in the temperature gradient across the converter could increase the system's overall efficiency by 1 to 2% to ~15-16%.

Table 5.3.2-4. A Comparison of Performance and Mass Estimates of Segmented TE Converters Technology for BOM 100 We RPS with Existing Flight (SiGe) RTGs

Parameter	SiGe-RTG	S-TE RPS (Projected - intermediate term)	S-TE RPS[@] (Projected -far term)
PERFORMANCE			
Hot temperature (K)	1273	973	1200
Cold temperature (K)	573	373	573
Thermal power at BOM (W)[#]	2000	1000	1000
Effective efficiency (%)	6.5	13.0	15.0
Electric power output (We)	107	107	124
Number of GPHS modules or bricks	8.0	4.0	4.0
MASS ESTIMATES			
$^{238}PuO_2$ Mass (kg)	5.02	2.51	2.51
GPHS modules (kg)	11.4	5.7	5.7
Housing (kg)	3.1	1.65	1.55
Radiator fins (kg)	0.45	1.17	0.215
Converter (kg)	5.65	2.83	2.8
Other structure (kg)	2.5	1.44	1.4
Total System Mass (kg)	23.1	12.8	11.7
System Specific Power (We/kg)	**4.6**	**8.4**	**10.6**

[@] Estimates for Segmented Thermoelectric Converters, with 200 K higher radiator temperature, 227 K higher hot-side temperature, and 27 K larger ΔT.

Table 5.3.2-5. Technical Status of Segmented Thermoelectric Converters

Performance Characteristics	Achieved to Date	Probably Achievable	Possibly Achievable
Conversion Efficiency	• 13% (or 21% of Carnot) is predicted based on measurements of material properties and interfacial resistance. • Identified potential high ZT materials and performed tests of potential alloys for use in bonds of segments. • Developed a semi-empirical performance model. • Developed a fabrication methodology and fabricated a segmented converter. • Began testing converter	• Complete the converter tests and confirm the current predictions for hot and cold junction temperatures of 973 and 373 K, respectively. Demonstrate 13% conversion. • Identify and measure the performance of high ZT material up to 1200 K.	• Demonstrate a converter performance up to 1200 K hot side temperature and cold side temperature up to 500 K, at conversion efficiency > 15%. • Develop and test multi-converter generator. • Identify performance degradation mechanisms. • Develop reliable models. • Resolve integration with GPHS
Lifetime and Reliability	• Confirmed performance of promising materials for bonds and fabrication techniques for a short term (months).	• Conduct accelerated life tests to confirm performance of bonds for long term (> 15yrs). • Conduct accelerated performance tests of the converter and generator.	• Demonstrate and verify acceptable life (>15 years) performance and reliability for planned space missions.
Specific Power	• No data. However, based on RTG experience and at the currently predicted level of performance (efficiency of 15% for the converter and 13% for RPS) ~40 We/kg for the converter and ~8 We/kg for a 100 We RPS are achievable.	• Demonstrate 15% efficiency for the converter and 13% for the RPS and 8.4 We/kg specific power for the system.	• Demonstrate 10.6 We/kg for the power system.
Spacecraft and GPHS Interface Issues	• Operating at relatively hot (973 K) side and cold (373 K) side temperatures would lower the temperature of the fuel pellets and cladding in GPHS, and the thermal efficiency of generator and increase the mass of the structure and radiator. • The low temperature heat rejection is not compatible with waste heat utilization on spacecraft	• Demonstrate effectiveness of radiative coupling of the converters to the GPHS; It should not be different from current RPSs and RTGs. • Increase the heat rejection temperature to be more compatible with waste heat use on board of spacecraft.	• Resolve the integration issues, including those caused by venting He gas and packaging with GPHS modules, particularly for use in low electric power missions (<10 We).

Table 5.3.2-5. Technical Status of Segmented Thermoelectric Converters(Continued)

Performance Characteristics	Achieved to Date	Probably Achievable	Possibly Achievable
Scalability	• In general, the inherent modularity of TE converters makes them easily scalable in the electric power range from a few to tens, or even hundreds of We. • However, for low power levels, in the tens to a few hundred mWe, segmented TE converters may not be suitable, but miniaturized ones using similar high ZT materials could be used, and currently are under development at JPL.	• Develop and demonstrate modular TE converters with low specific mass for use in low power generators. • Segmented TE converters technology is likely be limited to a conversion efficiency below 20%, unless new fabrication techniques such as laminated n- and p- legs are developed and implemented successfully.	• Can also be used to match new heat source module sizes if developed.
Verification	• No Data	• Conduct converter and prototype tests involving an artificial heat source, several converters, simulated space environment, and radiator. • Perform accelerated life tests and develop and benchmark performance models.	
Safety	• Very safe. Device has no fluids or hazards materials.	• Very safe. Device has no fluids or hazards materials	• Very safe. Device has no fluids or hazards materials

5.3.2.5 Technical Issues for Segmented Thermoelectrics

JPL has constructed a segmented converter and will begin testing and evaluating its performance. This converter is similar to that shown in Figure 5.3.2-4. Hi-Z is also in the process of fabricating segmented n- and p-legs. Future work at Hi-Z involves fabrication and testing of a segmented converter under present funding from DARPA and other sources. The development and verification effort on this promising converter technology would greatly benefit from a sustained funded program over the next several years to resolve the following technical issues:

- Identify high temperature materials with high ZT values and develop reliable fabrication techniques for integrating them into the n- and p-legs of a converter.
- Develop reproducible fabrication and integration methods and techniques of the various segments.
- Develop and confirm performance of reliable bonds with very low contact resistance, good mechanical strength, very low vapor pressure, comparable CTE, and which do not permit diffusion of the segment's materials across segment boundaries.
- Perform converter performance and accelerated tests and verify lifetime performance prediction models.

- Perform performance and life tests of a generator with simulated heat source in simulated space environments and radiative waste heat rejection.
- Address and investigate integration issues with the GPHS and spacecraft.

These issues could probably be addressed satisfactorily within 3-5 years, given an appropriate level of sustained funding and effective technical collaboration between JPL, industry, DOE, and academia.

5.3.2.6 Budget and Schedule for Segmented Thermoelectrics

The estimated budget and schedule required to address and resolve the technical issues (Sec. 5.3.2.5) to reach the different NASA TRLs is shown in Table 5.3.2-6. The NASA TRLs 3 and 4 represent a technology readiness gate where the progress will be reviewed in detail to decide to proceed or not to NASA TRL 5.

5.3.2.7 Recommendations

Segmented thermoelectric technology is a follow-on to RTGs that has the potential to significantly increase the efficiency and specific power of RTGs. The geometry, configuration, safety, long-life, and heritage of RTGs would be maintained. If RTGs are used on near-term missions, it would be desirable to upgrade from RTGs to segmented-TE ARPS in follow-on missions. The team recommends NASA fund segmented TE technology as given in Table 5.3.2-6 to develop the segmented thermoelectric ARPS for future NASA deep space missions.

5.3.3 Quantum-Well Thermoelectrics

5.3.3.1 Description of Quantum Well Thermoelectric device (how it works)

Quantum Well Thermoelectric (QW-TE) technology is currently being developed at Hi-Z in San Diego and the University of California-Los Angeles (UCLA). The Department of Energy under an SBIR sponsors the technology development effort at Hi-Z. In addition, the Department of Defense (DOD) sponsors the development of the QW-TE technology at UCLA through the MURI program, in which Hi-Z is a subcontractor to UCLA.

Table 5.3.2-6. Estimated Budget and Schedule for the Development of the Segmented TE Converter Technology.

Activity	Year					
	2001	2002	2003	2004	2005	2006
Low Temperature TE Materials Evaluations	$0.5M	$0.5M				
Low Temperature Unicouple Fab & Demonstration		$0.5M	$1.0M			
High Temperature TE Materials Evaluations		$1.0M	$1.5M			
High Temperature Unicouple Fab & Demonstration				$3.0M		
Fab and Test 4-Couple Converter Module					$3.0M	
Fab and test three 18-Couple Converters						$4.0M
NASA TRL	2		3	4		5

The quantum well theory on which these devices are based utilizes the fact that the electronic and thermal conductivity of the bulk material can be altered when it is made of a two-dimensional, nano-structure with periodic quantum wells. The electron transmission coefficient from cell to cell is very high, reducing the resistivity by a considerable margin. A typical QW-TE device consists of a series of two-dimensional, nano-structure quantum well layers, sandwiched between two barrier layers. The effective Z can be increased by a significant amount over bulk thermoelectric materials if the product of electronic resistivity and thermal conductivity in the denominator of Z can be decreased sufficiently. A QW-TE material with a ZT of 6 may be possible, and as Table 5.3.2-1 shows, this leads to a significant improvement in efficiency. Typically, the QW material has a very narrow band gap, while the barrier material has a relatively large band gap. The predicted enhancement in the figure-of-merit of the QW nano-structure is attributed to the confined motion of the charge carriers and phonons in the two-dimensional QW "tunnels" separating them from the ion scattering centers.

5.3.3.2 Configuration and Materials of QW Thermoelectrics

The values of α and κ for several QW-TE thin films have been determined over a broad range of temperatures, from 4.2 K to 1200 K. The measured α^2/k values for the p-type B-C and n-type SiGe films are more than a factor of 10 to 30 times higher than the bulk p-type B-C and the n-type SiGe. Experimental devices have been fabricated using molecular beam epitaxy (MBE) and sputtering techniques. Two-dimensional QW devices have been synthesized using alternating thin films of B_4C and B_9C in one device, and Si and $Si_{0.8}Ge_{0.2}$ in another device. These films have been deposited on thin, single crystal, silicon substrates to minimize the bypass thermal losses.

Several one- and two-couple devices have been fabricated with p-type B_4C/B_9C QW films and n-type bulk Bi_2Te_3. One of these couples produced 0.182 milliwatt at a ΔT of a 50 K. This device produced ten times more power than the bulk Bi_2Te_3 commercial material of the same dimensions at the same ΔT. Hi-Z is also producing thicker B_4C/B_9C films (>10 μm) on thinner Si substrates (<1 μm) to minimize thermal bypass heat losses. Successful scale up of these films for the p-leg is expected to yield a 1.0 cm^2 device that could produce ~5 We at a ΔT of 200 K. Assuming minimum heat losses, the efficiency of this QW-TE device could approach 20%.

5.3.3.3 Technical Status of QW Thermoelectric

Several B_4C/B_9C-Bi_2Te_3 and B_4C/B_9C-Si/SiGe p-n couples with low contact resistance were fabricated and the results appear very promising. Each leg in a couple consists of a square of 1000 0.5-mm thick multilayer of B_4C/B_9C- (p-type) and Si/SiGe (n-type) films. The films were deposited on 0.5-mm thick silicon substrate that is approximately 1cm x 1cm. At a ΔT of 50 K (T_{cold} = 313 K and T_{hot} = 363 K), the voltage measured on this couple was ~0.1 V. The contact resistance was a few ohms, which is very low compared to the total resistance of the couple, which was approximately 20 kΩ. This is the resistance of the films and does not include the Si substrate.

The efficiency was obtained from the electric power data and the measured values of α and κ at T_{hot} = 363 K and T_{cold} = 313 K. The values of Z were calculated over the temperature range from 313 to 363 K using bulk material, thermal conductivity data. The measured values of voltage and resistance gave a load power of ~0.125 μWe. At these same temperatures and dimensions, a bulk Bi_2Te_3 couple produced only 0.01 μWe, a bulk B_9C-SiGe couple produced only 0.004 μWe, and a bulk SiGe couple produced 0.02 μWe. Therefore, the B_4C/B_9C-Si/SiGe P-N couple produced about ten times more power than the bulk Bi_2Te_3 and about thirty times more power than bulk B_4C/B_9C-Si/SiGe.

Although this QW-TE couple was fabricated with thin films, Hi-Z hopes to duplicate the obtained results with much thicker films on a thinner insulating substrate. Silicon substrates with thickness of 5.0 μm and 10.0 μm and insulating substrates like Kapton are available commercially. If the fabrication of thick films on these substrates is successful, a 1.0 cm x 1.0 cm couple, like the one described above, could produce

1250 μWe at ΔT of 50 K. The final goal is to fabricate and measure the properties of the thicker P-N couples on very thin or insulating substrates

Total electrical and thermal losses of 5 μm thick Si substrate with a film thickness of 10 μm will be about 20%. The thermal losses from a 30-μm thick Si substrate are only 10%. A 10-μm thick film on 0.5-mm thick Si substrates could routinely be fabricated. Similar films could also be fabricated on 5 μm Si substrate.

5.3.3.4 Technical Issues for Quantum Well Thermoelectrics

For electrical power applications, the concern is that the B_4C/B_9C-Bi_2Te_3 and B_4C/B_9C-Si/SiG materials will inter-diffuse at some elevated temperature and lose their two-dimensional nano-structure and associated quantum well properties. For power generation applications, the B-C and SiGe alloys appear to be potentially good selections for the following reasons:

a. B and C have very low diffusion coefficients into one another.

b. Si and Ge have very low diffusion coefficients in one another.

The dopants boron and phosphorous, however, can diffuse much more quickly. Therefore, high temperature, aging studies are necessary to determine how long these films will remain stable at the anticipated operating temperatures. The B-C as well as Si-Ge alloys do not have to be deposited in an exact stoichiometry to be useful thermoelectric materials. Thus, the deposition process can be conducted with less critical controls.

5.3.3.5 Recommendations

Quantum well devices have sufficient potential that they need to be studied and developed by NASA. This technology is in an early stage of R&D and it is appropriate that funding should come from NASA and DOE research funds and SBIRs.

5.4 Thermionic Technology

5.4.1 Summary

Two very different technical approaches to thermionic conversion systems can conceivably be adapted to an advanced RPS. One is the Hybrid Cs/O Triode that incorporates innovations to allow a cesiated triode to operate at emitter temperatures of 1300-1400 K. The other approach is the Sandia Microminiature Thermionic Converter (MTC). This design is a vacuum diode with a μm-scale inter-electrode gap and inherently low work function coatings on the electrodes. The Defense Threat Reduction Agency (DTRA) is currently funding development of the MTC, in addition to work relevant to the triode but at elevated temperatures. The Department of Energy has also provided funds for MTC development.

A principal challenge for thermionic devices is achieving adequate efficiencies at the comparatively low temperatures available from a GPHS (~1200 K). The Hybrid Cs/O Triode employs a series of steps to achieve high efficiency at reduced emitter temperature. The first step introduces very low-pressure oxygen (~10^{-8} torr) into the inter-electrode space to reduce electrode work functions and allow lower cesium pressure. The feasibility of oxygenation has recently been demonstrated using two different methods, but the emitter temperature in these tests was ~1800 K. It is predicted that oxygenation can raise the electrode efficiency by two percentage points and lower the emitter temperature by about 200 K, but this is not enough for GPHS applications. The second step reduces the electron reflection at the collector by using a surface of platinum black. DTRA is funding experiments on this method, but again at 1800 K. Suppression of reflection is predicted to further improve electrode efficiency to 17% at an emitter temperature of 1530 K, but again, this is probably still too high for the GPHS. The final proposed innovation locally injects Cs^+ ions into the inter-electrode space from a separate anode rather than igniting an arc between the emitter and collector. The introduction of this additional electrode is the reason for designating the device as a "triode." This method is called the "hybrid mode" and has not yet been proved experimentally. However, hybrid mode operation may now be possible due to the high emission afforded by the Cs/O. The predicted electrode efficiency is 23% at an emitter temperature of 1,310 K.

As with other thermionic devices, the system efficiency is expected to be appreciably lower than electrode efficiency, but no conceptual RPS design has been developed for the Hybrid Cs/O Triode. An attractive design may be feasible if the theoretical triode performance parameters can be achieved. However, Cs/O development is at TRL 2, and triode development is at TRL 1. At these early stages, the Hybrid Cs/O Triode technology appears to be a much higher risk than some other options. It is recommended that NASA monitor the progress of the DTRA program, and that the experiments be expanded to include measurements at low emitter temperatures.

The MTC employs state-of-the-art electronic manufacturing techniques to apply electrode coatings and etch the μm size gap. The emitter coating is based on dispenser cathode technology utilizing a porous tungsten-Sc_2O_3 matrix containing barium compounds. Barium and oxygen migrate through the bulk material and disperse over the emitter surface to create the low work function. They also transfer to the collector to lower its work function. An initial conceptual design of a RPS-MTC system predicts 14% system efficiency and 6.6 watts/kg. However, these predictions assume significant technical advances that have not yet been demonstrated.

Microelectronic techniques for assembling MTCs have been developed and demonstrated to some extent, and gap sizes in the 1-10 micron range may be feasible. However, the electrode coating technology is unproven. Coatings sputtered so far have been extremely patchy, i. e., the areas that appear to have low work function are limited and surrounded by much larger areas of significantly higher work function. Although net current has been produced at T_E = 1173 K and lower, the apparent Richard-Dushman coefficient for electron emission derived from the test data is only 0.002 A/cm^2-K^2 vs. a design value of 10 A/cm^2-K^2. In addition to the coating, the complete MTC cell must be designed and optimized to limit thermal and electrical losses at the cell level. Some prior work on vacuum thermionic diodes with barium and metal electrodes indicates that the barium that deposits on the collector rapidly forms an insulating or reflective layer, resulting in significant loss of performance. It must be determined whether this phenomenon will occur in the MTC, and if so, how it can be mitigated. Finally, lifetime issues must be identified and resolved, the most obvious one being whether a gap of only 1-5 μm can be maintained over time.

MTC development is at TRL 1-2. It appears likely that a relatively large investment will be needed to advance this technology. For now, it is recommended that NASA merely monitor the progress of the DTRA program.

5.4.2 Introduction

In its simplest form, a thermionic diode consists of a heated emitter (cathode) separated by a small gap from a collector (anode). The emitter is heated to the point that the electrons in the high energy tail of the Fermi-energy distribution have sufficient energy to overcome the potential (work function) binding them to the material so they literally "boil off" the emitter surface. With an external load applied, the electrons traverse the gap and enter the collector. Since the electrons must have enough energy to overcome the work function of the emitter, choosing the emitter surface to minimize this work function is an important aspect of thermionic design.

The basic relationship governing thermionic emission from a hot surface is the Richard–Dushman equation:

$$J = AT^2 exp(-\phi / kT)$$

where J is the current density in amps/cm^2

A is the Richard-Dushman constant = 120 amps/cm^{2-}K^2

T is the temperature in K

ϕ is the material work function in eV

k is Boltzmann's constant = 8.62 x 10^{-5} eV/K

In a thermionic diode, several phenomena act to limit the current flow. A major factor is that the electron cloud in the space between the emitter and collector produces a space charge potential that acts as a barrier to electrons traversing the gap. In principle, the space charge effect can be largely negated in two general ways. For a vacuum diode, the gap spacing can be reduced to the point that it is comparable to the mean free path of the electrons. Alternatively, a plasma of positive ions can be produced in the gap to neutralize the negative space charge of the electrons. This plasma is typically created by a low voltage arc between the electrodes that ionizes cesium gas introduced into the gap from an external source. The cesium provides a second benefit in that it forms a partial film on the emitter that reduces the bare work function. Historically, the plasma or ignited mode thermionic diode has been the most successful approach, even though some energy is expended in creating the arc.

As with most energy conversion systems, efficiency must be carefully defined. For thermionics, diode efficiency is the electrical power produced by a single diode divided by the heat entering the emitter. This definition accounts for losses due to electron cooling of the electrodes and thermal radiation from the emitter to the collector. It is often called electrode efficiency, although some investigators also add in electrical losses in the electrical leads typical of laboratory test devices using a single diode. A series of interconnected diodes (converter) introduces additional thermal and electrical resistance losses in the interconnecting leads. External to the thermionic converter in a RPS generator, additional heat will be lost by thermal bypass through the insulation and GPHS supports. Therefore, the overall system efficiency is the useful electrical power at the external leads of the generator divided by the total heat generated by the radioisotope in the GPHSs.

Thermionic conversion has been investigated for space applications since the early 1960s. Both solar and nuclear energy sources were considered. Much of the recent work on thermionics was conducted for space reactor applications, nearly all of which was terminated in 1993. The work that continued fell into two areas. As a Congressional interest item, the Department of Defense received plus-up funds to continue thermionics R&D. This money went to the Air Force initially, but shifted to the Defense Nuclear Agency (DNA) beginning in 1996. DNA is now part of the Defense Threat Reduction Agency that is continuing the program. For naval applications, thermionics R&D was sponsored by the DOE Office of Naval Reactors and administered through Bettis Atomic Power Laboratory. The Bettis work on thermionics was terminated in 1995.

Most of the thermionics work done in the past was at emitter temperatures too high to be compatible with a GPHS. The Assessment Team received briefings on both concepts for thermionic diodes that operate at emitter temperatures compatible with the GPHS. The Cs/O hybrid mode triode expands the capabilities of the standard cesiated diode. The MTC is a vacuum diode with the goal of employing inherently low work function coatings on the electrodes. Because the technical approaches are very different, each concept will be described separately.

5.4.3 Cs/O Hybrid Mode Triode

5.4.3.1 Description (How it works)

The performance space of a cesium plasma diode is shown in Figure 5.4.3-1. This carpet plot illustrates the relationship between emitter temperature and electrode efficiency for combinations of power density and back voltage (V_B). V_B is the total potential resisting current flow within the diode, and is equal to the sum of collector work function (ϕ_c), the arc drop (V_d) necessary to reduce space charge effects, and plasma attenuation (V_a). Neglecting back emission from the collector, the general expression for current density in the ignited mode is:

$$J = AT_E^2 \, exp[-(V + V_B)/kT_E]$$

where T_E is the emitter temperature

ϕ_E is the emitter work function

V is the voltage produced by the diode

The present design for the Cs/O triode is the culmination of a series of steps attempted over a period of many years to reduce V_B. Most of the prior work sought to increase power density rather than lower emitter temperature. The steps to lower V_B (and lower emitter temperature) are summarized in Table 5.4.3-1. The projected performance associated with each stage is also given, although these levels have yet to be fully demonstrated in laboratory tests. In particular, the hybrid mode has yet to be demonstrated successfully in any regime.

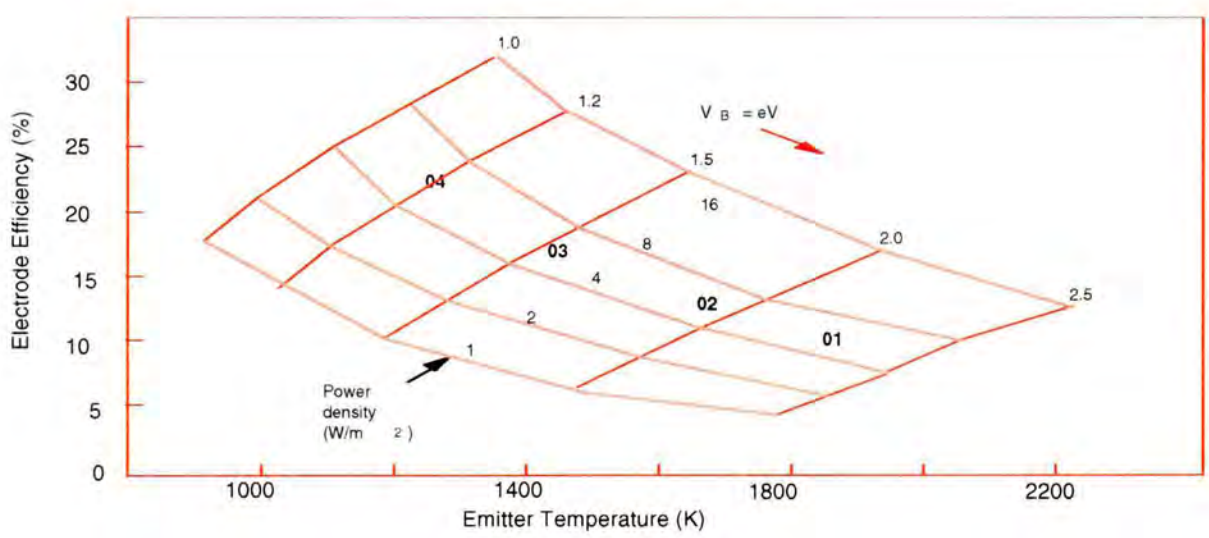


Figure 5.4.3-1. Thermionic Performance Map

Table 5.4.3-1. Cs/O Hybrid Mode Triode Basic Development Stages

Basic Development	Vapor Source	Emitter Surface	Collector Surface	Plasma	V_B (eV)
1. Circa 1970	Liquid Cs reservoir	Adsorbed Cs	Adsorbed Cs	Hot ignited	2.2
2. Demonstrated DECOR Cs/O	DECOR	Co-adsorbed Cs/O	Co-adsorbed Cs/O	Hot ignited	1.9
3. Cs/O +suppression of e- reflection	DECOR	Co-adsorbed Cs/O	Structured + Co-adsorbed Cs/O	Hot ignited	1.6
4. Cs/O - hybrid mode	DECOR	Co-adsorbed Cs/O	Structured + Co-adsorbed Cs/O	Injected ions into cold plasma	1.2

at Collector in Plasma Attenuation
$V_B = (\phi_c) + (V_d) + (V_a)$ = Total Energy Loss
Stage 1: $V_B = 1.6 + 0.4 + 0.2 = 2.2$ eV (11% 1860K) | *Efficiencies and temperatures*
Stage 2: $V_B = 1.4 + 0.4 + 0.1 = 1.9$ eV (13% 1665K) | *at 6 W/cm²*
Stage 3: $V_B = 1.1 + 0.4 + 0.1 = 1.6$ eV (17% 1530K) |
Stage 4: $V_B = 1.0 + 0.1 + 0.2 = 1.2$ eV (23% 1310K) |

The first row describes the state-of-the-art cesium plasma diode conditions. Row 2 introduces very low-pressure oxygen (~10^{-8} torr) into the cesium plasma inter-electrode space, long recognized as a method of reducing electrode work functions and allowing the cesium pressure to be lowered. The particular delivery system briefed to the Assessment Team was named the Dynamic Equilibrium Cesium-Oxygen Reservoir (DECOR), but other methods are possible. Use of DECOR raises the estimated electrode efficiency by two percentage points and lowers the temperature by about 200 K; not enough for GPHS applications. Row 3 combines the Cs/O system with changes to the collector surface (use of platinum black) to reduce the degree of reflection of electrons that reach the collector. This further improves the theoretical efficiency to 17% at an emitter temperature of 1530 K, which again is probably still too high for the GPHS. Row 4 nearly eliminates the arc drop by utilizing an innovative method for locally injecting Cs^+ ions into the inter-electrode space from a separate anode rather than igniting an arc between the emitter and collector. In this technique, the injected ions come from "bright ignited plasma tufts" that are produced at various anode sites along the collector. These tufts are generated from an outside voltage source that applies a small but sufficient bias between the emitter and anode to cause local arcs. Cesium atoms in these tufts are ionized and "injected" into the main body of the unignited plasma by the negative charge from the trapped electrons. This method is called the "hybrid mode" and the system is called a "triode" because of the additional electrode. Earlier attempts at hybrid mode operation were unsuccessful because the combination of low work functions and low Cs pressure could not be achieved. However, it may now be possible due to the high emission afforded by the Cs/O.

These combined steps are projected to reduce V_B from 2.2 eV to 1.2 eV. As a result, the theoretical electrode efficiency (at 6 W/cm^2) is projected to increase from 11% to 23% while the emitter temperature is reduced from 1860 K to 1310 K. This temperature is compatible with GPHS operation. However, it must be emphasized that these estimates of performance are based on models and not yet substantiated by data.

5.4.3.2 Modular Conceptual Design

No conceptual design using the GPHS was presented to the committee. A 1971 vintage concept developed by Douglas Laboratories was described, but it is not relevant to this assessment. However, it should be possible to develop a practical RPS design utilizing multiple Cs/O triodes if they can achieve the theoretical performance.

The emitter material would probably be tungsten, tungsten-coated molybdenum, or rhenium. Candidate collector materials are niobium and molybdenum with a layer of platinum black or some other treatment to reduce electron reflection.

The electrode parameters are described above. However, converter efficiency will be significantly less than electrode efficiency of a single triode. Additional thermal and electrical losses occur in the inter-cell electrical connectors and in the system leads. Furthermore, the collector surface treatment to reduce electron reflection will probably increase surface emissivity, resulting in some increase in thermal radiation losses across the inter-electrode gap. External to the diodes, some heat will be lost by thermal bypass through the insulation and GPHS supports. For a string of standard cesiated diodes, lead efficiency is typically 70-80% of electrode efficiency, and the bypass losses will reduce system efficiency another 2-3 percentage points. Hence, an RPS with a diode efficiency of 23% can be expected to have a system efficiency in the range of 10-15%.

A Cs/O triode RPS should be scalable over the range of a few watts to 1 kilowatt by merely changing the number of cells, much like the GPHS-RTG configuration can be scaled. Milliwatt sizes may also be feasible, but a very different miniature cell would have to be developed.

5.4.3.3 Technical Status

Table 5.4.3-2 summarizes the technical status of the Cs/O triode. In recent laboratory-level experiments, oxygenated performance was demonstrated on small planar diodes. Under the sponsorship of the Bettis Atomic Power Laboratory, stable operation was obtained for 1000 hours using the Rasor DECOR delivery system at the following conditions: T_E = 1800 K, T_C = 800 K, gap spacing = 20 mils, Cs pressure = 1 torr. The effective V_B was ~1.9 eV. Bettis repeated the experiment using porous nickel in place of platinum in the DECOR and achieved stable performance for hundreds of hours with V_B ~1.8 eV.

Under DTRA sponsorship, General Atomics recently achieved stable oxygenated performance using a nearly identical laboratory device without DECOR. In these experiments, the source of oxygen was emission from oxygenated collectors supplied by both ThermoElectron and the Russian institute SIA LUTCH. The best results were obtained with one of the LUTCH collectors. At T_E = 1800 K and a spacing of 0.5 mm, the diode was stable for 190 hours after which the test was intentionally ended. The point of maximum electrode efficiency was 14.5%, and a maximum power density of 5.5 W/cm^2. With other collectors, electrode efficiencies in excess of 17% were achieved, but the performance was unstable, apparently from hydrogen contamination.

5.4.3.4 Technical Issues

Several issues are yet to be resolved at the laboratory level. Although stable Cs/O performance at conventional high emitter temperatures has been shown, no recent tests have investigated reduced T_E. Furthermore, neither of the two methods of oxygen delivery has been optimized. Independent thermal control of the DECOR must be established. For oxygenated collectors, design parameters and manufacturing techniques are being developed with DTRA funding. Triode development and operation (the critical factor needed for reasonable efficiency at GPHS temperatures) remains to be demonstrated.

Adapting these designs and techniques to practical ARPS systems is another technical challenge. Scale-up and operation outside of a carefully controlled laboratory environment must be addressed.

Table 5.4.3-2. Cs/O Triode Technology Status Summary

Characteristic	Achieved to Date	Probably Achievable	Possible
Diode Conversion Efficiency	13-17% diode efficiency in separate tests @ ~1800 K emitter. No data at temperatures compatible with GPHS	~15% diode efficiency @ 1400 K emitter temperature	20-25% diode efficiency @ 1300-1400 K emitter temperature (Corresponds to system efficiency of 10-15%)
Lifetime and Reliability	Up to 1000 hours stable operation	5-10 years if stable oxygen supply is developed (basis for this is not clear)	10+ years
Specific Power	No ARPS design	TBD	
Spacecraft Interface Issues	No ARPS design	No problems anticipated	
Converter-GPHS Interface Issues	No ARPS design	No problems anticipated	
Scalability	No analysis	Watts to 1 kilowatt	New triode design needed for milliwatts
Verification	No analysis (Potential for accelerated life test?)	Full prototype with electric heaters	
Safety	No analysis	No issues	

5.4.3.5 Budgets and Schedules

A huge amount of work has been done on thermionics over the past 40 years, but most of this was on systems working at higher temperatures than can be provided by a GPHS. The major accomplishment to date on technology relevant to operation at GPHS temperatures is the successful demonstration of stable performance of oxygenated diode at the laboratory scale. This was accomplished in both the Bettis and DTRA programs, but using different methods of oxygen delivery and at ~1800 K emitter temperatures. Although the temperature was high, a similar approach can potentially be used at GPHS temperatures. Roughly $2 M has been spent on these two programs.

Bettis is no longer sponsoring thermionic work. For FY'01, the DTRA program includes a modest effort at General Atomics to investigate oxygenated collectors with both tungsten and rhenium emitters. Auburn University is developing oxygenated collectors for that program. In addition, Dr. Rasor is examining reduced electron reflection at the collector.

Although the DTRA program continues relevant work, the objective is increased efficiency and power density, not lower emitter temperature. No work is planned to investigate hybrid-mode triode operation.

A new program would be needed if the hybrid mode triode is to be developed. Conceptual design of an ARPS system utilizing the GPHS is also necessary to estimate system level performance based on electrode performance. However, this new program could utilize existing facilities because the DTRA experiments on Cs/O diodes and reflection repression could easily be modified to include testing at lower emitter temperatures.

Cs/O development is at TRL 2, and triode development is at TRL 1. Since development is a high-risk undertaking, specific cost and schedule estimates are nebulous. The University of New Mexico Engineering Research Institute (NMERI) is a participant in the DTRA program, and has provided the Assessment Team with an estimate of ~$300K to demonstrate hybrid triode feasibility. This amount may be sufficient to conduct the initial experiments, but it is doubtful that full feasibility can be established within that scope of work. In addition, NMERI proposes a conceptual design study of an ARPS using the triodes for ~$1 M.

5.4.3.6 Recommendations

Cs/O development is at TRL 2, and triode development is at TRL 1. At these early stages, it is difficult to answer the questions What is the probability that the technology could be raised to TRL 4 or 5 in a few years? and How much will it cost? The proponents seem to think that it will not take much time to prove the concept. The assessment team recommends not funding thermionic technology and use NASA funds to develop higher NASA TRL technologies.

5.4.4 Microminiature Thermionic Converter

5.4.4.1 Description (How it Works)

The second approach to reducing the emitter temperature is to use a vacuum diode with extremely close spacing between electrodes having very low bare work functions. The electrode spacing is limited to a few microns, on the order of the mean free path of the emitted electrons, so that the voltage loss due the electron space charge in the gap is minimized without use of a plasma. Thus, cesium ions are not needed to neutralize this space charge. Low work functions are produced with special oxide coatings that introduce barium into the inter-electrode gap.

Sandia National Laboratories (Albuquerque) has been pursuing MTC technology with funding from DOE and DTRA. Figure 5.4.4-1 shows a schematic of the Sandia MTC concept. State-of-the-art electronic manufacturing techniques are employed to apply the coatings and etch the spacing. In the Sandia approach, the coatings are applied by RF sputtering. The emitter coating is currently based on dispenser cathode technology utilizing a porous tungsten-Sc_2O_3 matrix containing barium compounds. Barium and oxygen migrate through the bulk material to the surface and disperse over the emitter surface to create the low work function. These elements also evaporate from the emitter and deposit on the collector coating to reduce its work function.

For vacuum diodes, the effects of electron reflection at the electrodes are very important and have recently been reexamined by Marshall (Marshall, Albert C., "An Equation for Thermionic Currents in Vacuum Energy Conversion Diodes," App Phys Lett, 73, No. 20, November 16, 1998.). The physics also applies to plasma diodes. His expression for net current density is:

$$J = \tau(J_E - \Gamma J_C)$$

where J is the current density in amps/cm^2

J_E is the ideal emitter emission current

J_C is the ideal back emission from the collector

τ is the effective transmission coefficient

Γ is the symmetry index

τ and Γ are functions of the average spectral electron transmissions (1 - reflectance) incident on the electrodes. They account for situations where the apparent Richard-Dushman coefficients, A*, for the electrodes are not equal and not = 120 amps/cm^2–K^2. This is the case for the MTC. A low A* (and corresponding τ) reduces current density, but reasonable efficiency may still be possible because the heat loss due to electron cooling is also reduced proportionately.

5.4.4.2 RPS Concept Point Design at 100 Watts

Very little work has been done as yet on the design of a complete MTC cell. Under DTRA funding, a very conceptual model was developed for RPS with MTCs replacing the thermoelectrics of an RTG. The MTC configuration is shown in Figure 5.4.4-2. Since the MTC substrate materials can be similar to those contained in a SiGe unicouple stack, the model assumes that similar bonding techniques can be employed to create a completely bonded MTC stack. The alternate foil/separator multilayer insulation (MLI) used in the GPHS-RTG is replaced with the thinner dimpled or ZrO_2 coated MLI to accommodate the MTC stack thickness of ~0.5 cm. No structural analysis has been done to verify the integrity of this configuration.

Sandia's goal is to achieve the following performance parameters: (1) work functions ($\phi_E = \phi_C = 1.0$ eV), (2) apparent Richard-Dushman constant ($A^* = 10$ A/cm^2-K^2), and (3) thermal emissivity ($\varepsilon = 0.1$). With these assumed values, Sandia's model projects that for $T_E = 1200$ K and $T_C = 600$ K, the maximum electrode efficiency can be as high as 22% for 1 micron spacing, 18% for 5 micron spacing, and 12% for 10 micron spacing. For identical emitter and collector coatings with $A^* = 10$ A/cm^2-K^2, the appropriate transmission parameters are τ ~0.043 and $\Gamma = 1.0$.

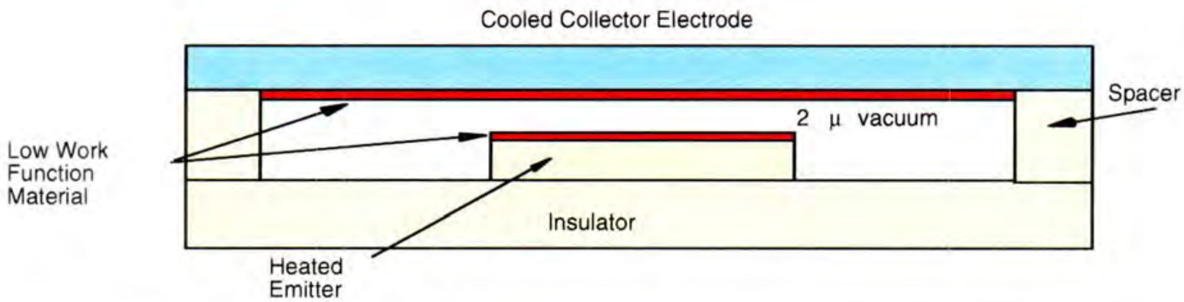


Figure 5.4.4-1. Sandia Microminiature Thermionic Converter (MTC)

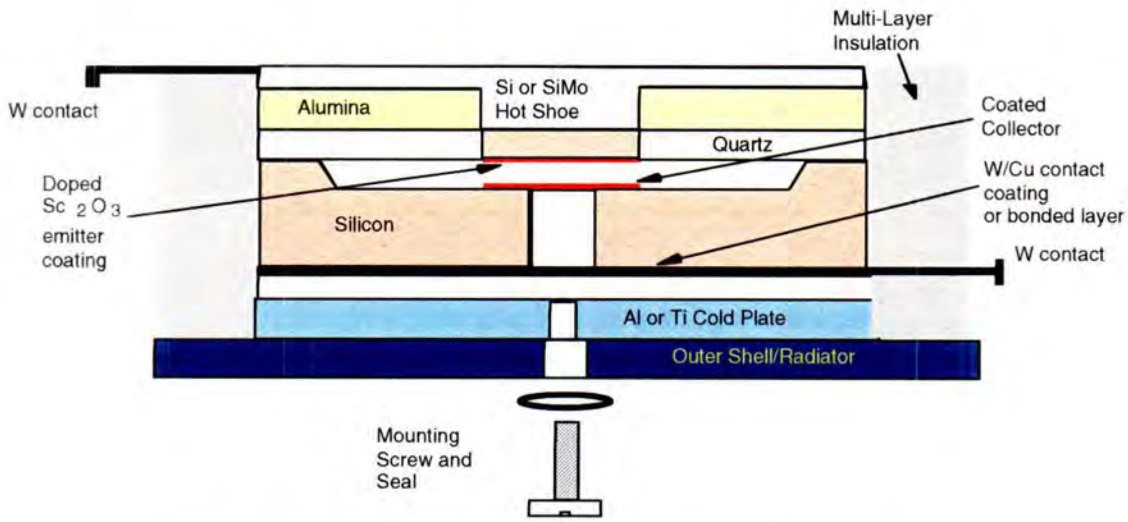


Figure 5.4.4-2. ARPS-MTC Configuration (not to scale)

Using Sandia's projected values, the GPHS-MTC model yields the beginning of life (based on BOL = 250 watts thermal per GPHS) results summarized in Table 5.4.4-1. The efficiency loss between the electrode and system levels is due primarily to: 1) thermal conductive losses in the cell, and 2) electrical and thermal losses through the cell interconnections. Thermal conductance loss is approximated from a finite element analysis done by Sandia. Lacking a definitive cell design, interconnect losses are merely placeholder values of 10% thermal and 10% electrical. No analysis has been done on the effect of reduction of heat due to radioisotope decay, but thermionic performance is a strong function of emitter temperature. Hence, some type of active thermal control is probably required.

It should be emphasized that the projected performance given in Table 5.4.4-1 has not been achieved in the laboratory. Indeed, actual results achieved so far are orders of magnitude worse than the projected values.

Although the MTC is best suited for low power systems, it should be scalable to a few hundred watts for ARPS applications. A conceptual model has been developed for MTCs mated to a Radioisotope Heater Unit (RHU) for milliwatt power.

5.4.4.3 Technical Status

Microelectronic techniques for assembling MTCs have been developed and demonstrated to a considerable extent. It appears that gap sizes in the 1-10 micron range are feasible, although their longevity has not been demonstrated. Having developed this fabrication capability, the experimental work went on to focus on development of a suitable emitter coating and establishing reproducible results on a laboratory test device. Although much progress has been made on the test device, the coating technology is unproven. Coatings sputtered so far have been extremely patchy, i.e., the areas that appear to have low work function are limited and surrounded by much larger areas of significantly higher work function. Although net current has been produced at T_E = 1173 K and lower, the value of A* derived from the test data is only 0.002 A/cm^2-K^2. Table 5.4.4-2 summarizes the technical status of the MTC.

5.4.4.4 Technical Issues

Four significant issues must be resolved for the MTC to be successful. The major one is development of the emitter coating. Second, the complete MTC cell must be designed and optimized to limit thermal and electrical losses at the cell level. Third, some prior work on vacuum thermionic diodes with barium and metal electrodes indicates that the barium that deposits on the collector rapidly forms an insulating or reflective layer, resulting in significant loss of performance. It must be determined whether this phenomenon will occur in the MTC, and if so, how it can be mitigated. Finally, lifetime issues must be identified and resolved, the most obvious one being whether a gap of only 1-5 μm can be maintained over time.

5.4.4.5 Budgets and Schedules

To date, combined spending by DTRA and DOE totals up to about $2.5 M. Micro-fabrication techniques were developed to a considerable extent, but progress on cathode materials has been slow. Coatings have been deposited and tested, but patch effects greatly limit their emission capability.

DTRA has budgeted ~$500K for continued work in FY'01. Future funding depends somewhat on recommendations by the National Research Council panel that is reviewing the DTRA thermionics program. The DOE budget for FY'01 will probably be insufficient to continue support.

Table 5.4.4-1. Projected BOL Performance for a 100 We GPHS-MTC System

Power/ voltage	No. of GPHSs	T_E/T_C	Electrode Efficiency	System Efficiency	No./Diam of MTCs	System mass	Specific power
110 We 5 v	3	1200 K/600 K	19%	15%	28/5.7 cm	7.3 kg	15.2 We/kg

Table 5.4.4-2. MTC Technology Status Summary

Characteristic	Achieved to Date	Probably Achievable	Possible
Conversion Efficiency	No efficiencies measured	Unknown	~15% system efficiency @ 1200 K emitter temperature
Lifetime and Reliability	Coatings tested to a few hundred hours. Material stability and μm-scale gap are issues.	Unknown	10 years?
Specific Power	Not calculated	Unknown	~15 watts/kg
Spacecraft Interface Issues	No analysis, but shock & vibration loads may be an issue.		
Converter-GPHS Interface Issues	Only very conceptual design work	No major problems anticipated	
Scalability	No analysis	Milliwatts to 1 kilowatt	
Verification	No analysis	Full prototype with electric heaters	
Safety	No analysis	No issues	

5.4.4.6 Recommendations

There are still concerns that regardless of how much is spent, it might not be possible to find suitable emitter and collector materials while maintaining μm-scale inter-electrode spacing over long times of operation. Until a suitable emitter coating can be developed, an ARPS development program does not appear to be warranted. The MTC is at TRL 2, and the results of the current program indicate that development is a very high-risk undertaking. Sandia has proposed a three stage development program totaling about $5 M over about 4-5 years to establish feasibility at the laboratory level. Based on past expenditures, this cost estimate is probably lower than what will actually be required. The review team does not recommend NASA ARPS funding for this technology based on what is known at the present time.

5.5 ThermoPhotovoltaics

The assessment team reviewed the NASA GRC effort being conducted on developing the Thermophotovoltaics (TPV) technologies. This technology is primarily aimed at applications where temperatures are much higher than can be obtained with a GPHS. At GPHS temperatures, efficiencies of 1 to 2% are predicted with sink temperatures at 300 K. This is not attractive for ARPS for deep space missions. The Navy has a large program developing this technology for their applications that we were not able to review. The team recommends that NASA and DOE follow the DOD efforts and not fund the TPV technology development at this time.

6.0 Results and Recommendations

The following ARPS conversion technologies were reviewed and assessed for relevance to potential future NASA Deep Space Missions:

- Advanced Stirling Engine Converter (ASEC),
- Thermo-Acoustic (TA),
- Alkali Metal Thermal to Electric (AMTEC),
- Segmented Thermoelectric (STE),
- Low Temperature Thermionic (LTI), and
- Thermal PhotoVoltaics (TPV).

Estimated system masses and efficiencies were made for each technology and compared to a scaled-down design of a 100-watt SiGe RTG. The 100-watt RPS module was selected as a power system module size that meets the requirements of the majority of potential future NASA deep space missions.

The ASEC-ARPS has the highest conversion efficiency. The present version of SEC ARPS has about the same specific power as the SiGe RTG and has an unproven lifetime. ASEC-ARPS may not meet the EMI and vibration requirements for some deep space missions with sensitive instruments aboard. ASEC-ARPS appears to meet the requirements for Mars large rovers missions, but may not meet the lifetime requirements for SSE and SEC missions.

AMTEC-ARPS has the potential for higher specific power and efficiency than SiGe-RTG. The system lifetime of AMTEC-ARPS still has to be validated, but accelerated testing can probably be used. AMTEC-ARPS has no EMI or vibration problems. For missions where low mass is of paramount importance and the rejected heat is required to warm the spacecraft propulsion subsystem, this technology is attractive. Significant technical challenges remain but most of these have been identified and solutions are being developed.

STE-ARPS is an advanced thermoelectric converter. Therefore, long lifetimes can probably be validated with accelerated testing based on experience with other thermoelectrics. It has no EMI or vibration problems. The potential conversion efficiency is more than double that of SiGe RTGs. STE-ARPS technology is at low NASA TRL. However, it has the great advantage of using solid state material, no liquids or gases to leak, and minimal concerns with large G launch loads. Quantum well thermoelectrics, though attractive on paper, are at an early stage of R&D and are not sufficiently developed to be ready for advancement to higher TRL levels.

LTI-ARPS has been approached in two ways. In one approach, the use of a oxygen to lower the cesium plasma pressure in the gap reduces electron collision losses. The other approach uses micro-fabrication techniques to reduce the gap size and losses. Both approaches are at a low NASA TRL and their ultimate performance is no better than the other more mature technologies. Therefore, NASA support for LTI-ARPS is not recommended.

TPV-ARPS technology does not appear attractive at this time due to very low system conversion efficiencies, 1 to 2%, and the low heat rejection temperature. Therefore, NASA support for TPV-ARPS is not recommended.

The assessment team concluded that NASA should fund ASEC AMTEC and STE technologies for missions beyond 2011. The characteristics of these three technologies are summarized in Table 6 -1. In each case, a GPHS module was assumed to deliver 240 watts of heat at BOM and the number of modules was chosen to make the BOM electrical power 100 watts or slightly greater. Thermo-acoustic technology is considered by the team as a possible advanced Stirling engine for the ASEC-ARPS technology.

The relation between efficiency and specific power these three ARPS technologies is given by Figure 6-1. RTG technology is at the lower left of the diagram with low specific power and low efficiency. It is desired to move toward the upper right of the diagram representing high efficiency and high specific power. However, the readiness levels are lowest in this region.

Table 6-1. Characteristics of Candidate ARPS Technologies

Tech-neology	Yr. '00 TRL	BOM Watts	System Mass kg	SP Pwr W/kg	Syst Effic%	GPHS Mod-les	Reqts for TRL5	Dev't Risk	Life Issues	S/C IF Issues	Resiliency to Partial Failure
Small SiGe RTG	8	139	31.2	4.5	6.5%	9	None	None	None	None	Highly resilient
ASEC-ARPS	2-4	120	16	6.2*	25%	2	$12M-6yrs	Medium	Generator Control electronics helium leakage, Stirling System	ac-to-dc controller, radiator, vibration, EMI	Failure of converter may lead to generator failure
AMTEC-ARPS	2-3	120	13.6	8.8	16.7 %	3	$15M-5yrs	Medium	Seals, Wick/evap containment materials, fabrication	Launch vehicle aceleration	Generator partial power loss
STE-ARPS	2	144	14	10.2	15%	4	$15M-6yrs	High	High temp materials Bonding, diffusion barriers	None	Highly resilient

* potentially achievable

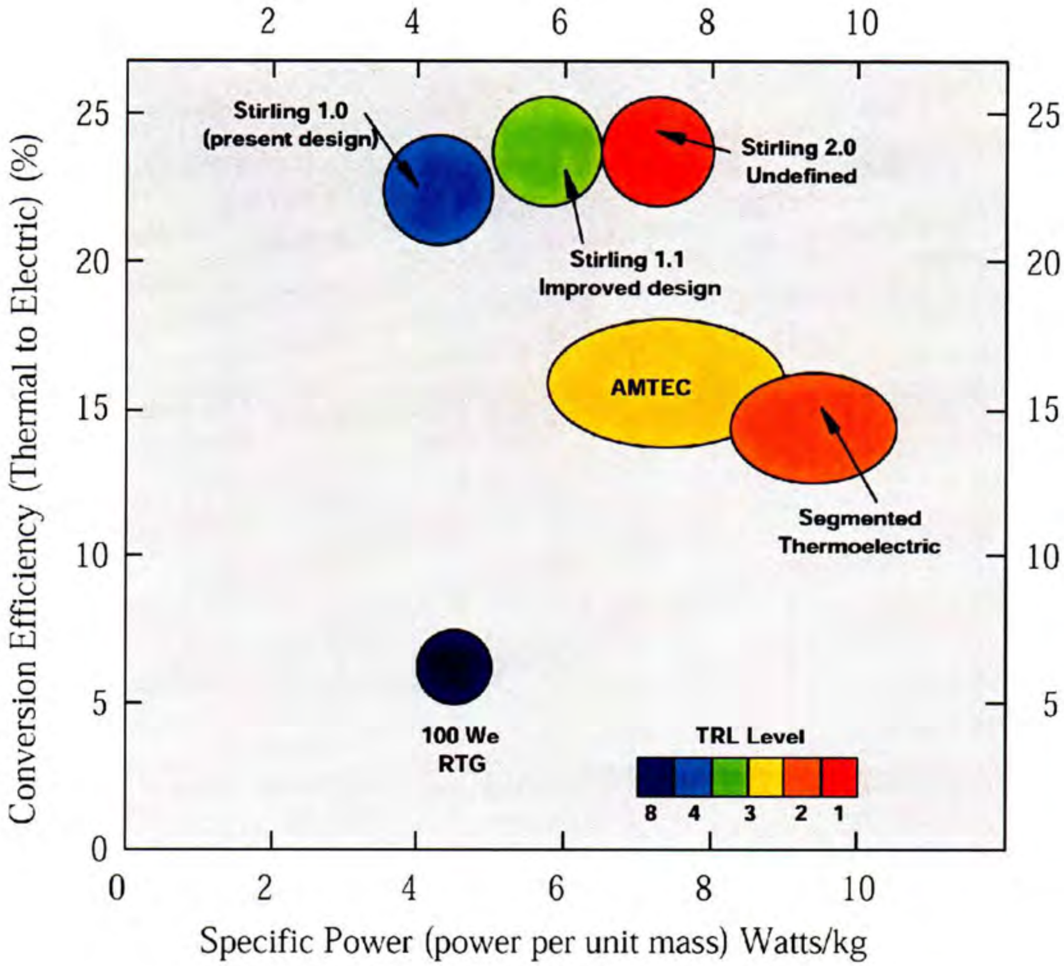


Figure 6-1. Comparison of Conversion Efficiency and Specific Power

Because of uncertainties in the performance of the ARPS conversion technologies, the assessment team decided that down-selecting to one ARPS conversion technology is not recommended at this time. The team recommends that future down-selects be based on demonstrated technical progress against target gates. The ASEC, AMTEC and STE technologies each have enough potential to warrant further development. Development of thermo-acoustic technology should be continued as part of ASEC.

The assessment team recommends that the ASEC, AMTEC and STE technologies be developed by NASA in accordance with a technology plan that includes funding per year and technology readiness gates as shown in Figure 6-2. The progress towards meeting the technology gates should be reviewed annually by a standing independent review board.

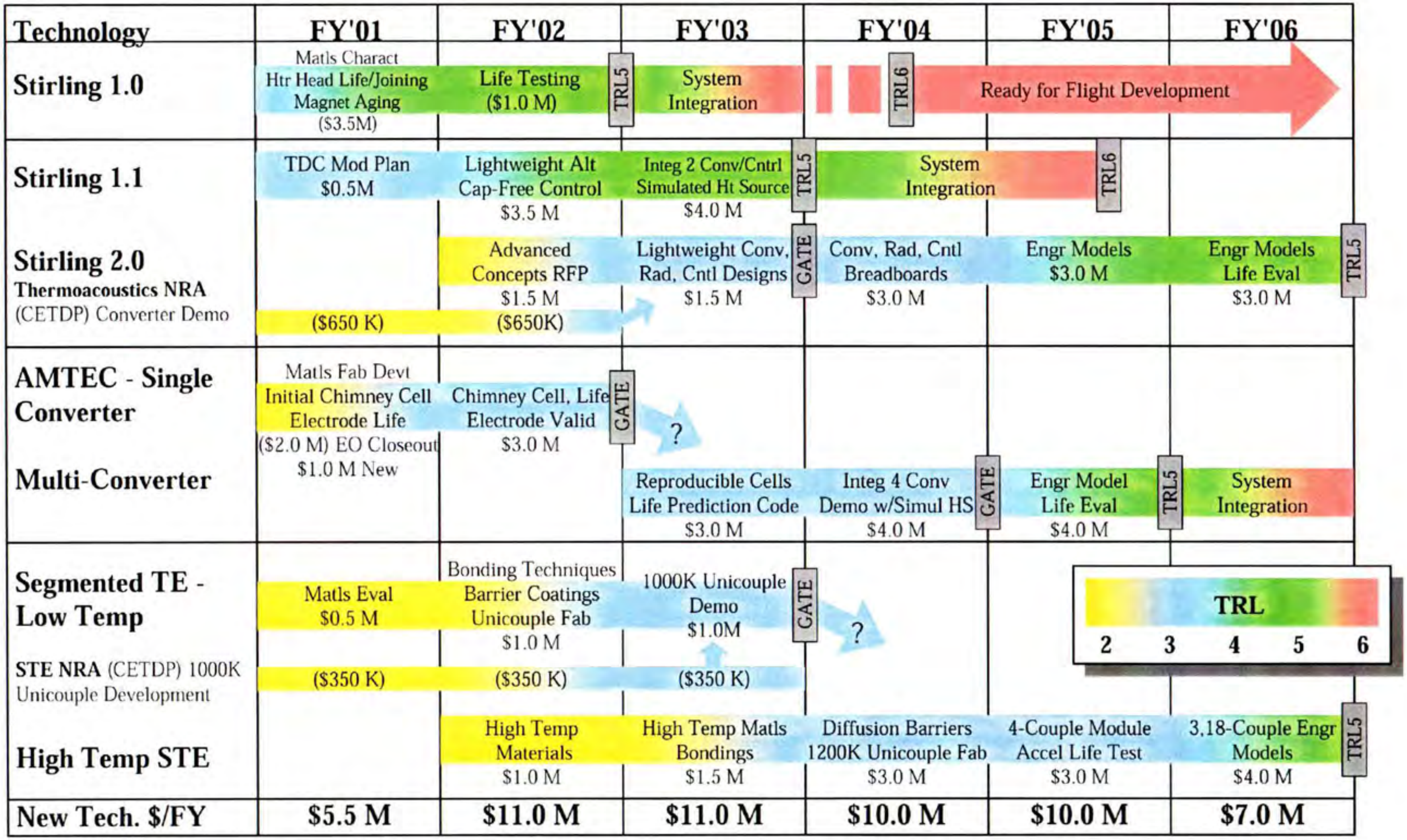
If any technology is not making adequate progress towards meeting the technology readiness gates, this provides a basis for discontinuing work or reducing the level of funding until technical issues are resolved. Following this plan should provide the basis for down-selecting by FY'05 to a single ARPS technology that meets the requirements for the greatest number of future NASA SSE, SEC, and MEP missions. The selected technology would then be developed to TRL 6 by FY'08 by a NASA flight project to meet its specific requirements.

Road Map and Technology Plan for Promising Technologies

A Technology Plan was developed for the most promising technologies based on the starting NASA TRL and the major technical issues remaining for each technology.

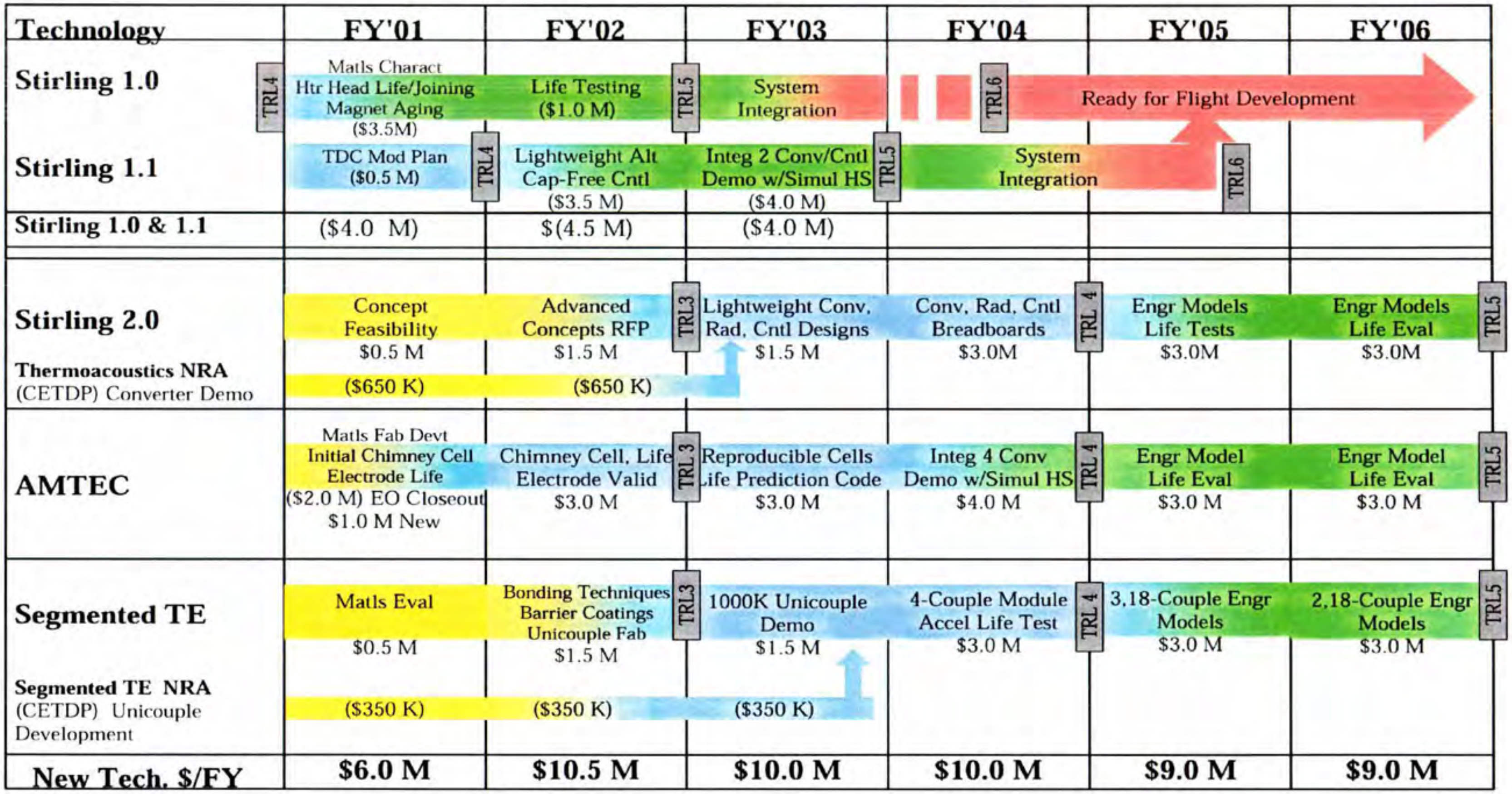
The recommended ARPS technology roadmap (Figure 6-2) allows options to either develop Stirling 1.0 or 1.1 to flight readiness on an accelerated schedule, or to develop Stirling 2.0 roughly in parallel with AMTEC and Segmented Thermoelectrics for missions that launch beyond about 2011. This roadmap is not a rigid plan to be followed regardless of further developments. It is a framework that defines our best estimate at this time of what is appropriate in the future. As progress is made in some areas, and disappointments are found in others, funds can be transferred between technologies to develop the most promising technologies at the fastest possible rate. A system engineering team led by DOE with support from JPL and GRC should be funded to prepare ARPS concept designs for each conversion technology. These ARPS system concept designs would be prepared to satisfy NASA mission requirements. The concept designs would also be used to help establish the technology readiness gates and direct the conversion technology development program.

An Alternate ARPS Technology Roadmap (Figure 6-3) was prepared with parallel development of Stirling 2.0, AMTEC, and STE for missions with launch dates beyond about 2011.



System integration and flight development costs, EO and Mars funds, and CETDP costs NOT included in totals. Independent Review Board evaluates technology progress each year to determine whether to proceed or not.

Figure 6-2. Recommended ARPS Technology Roadmap



Notes: Stirling 1.0 and 1.1, System Integration and Flight Development costs, and CETDP costs not included in totals. Independent Review Board evaluates technology progress TRL 3, 4, and 5 to determine to proceed or not.

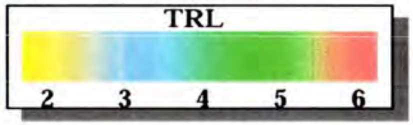


Figure 6-3. Alternate ARPS Technology Roadmap

Appendix A - Outer Planet Missions

A.1 Europa Orbiter

The first mission to Europa is the Europa Orbiter that will seek to confirm the existence of a subsurface ocean and study its characteristics. The other two are Pluto-Kuiper Express and Solar Probe. The Europa Orbiter (EO) mission is currently planned to launch in 2006. The spacecraft will take 3.25 years to reach Jupiter on a ballistic trajectory, where it will capture into Jovian orbit using chemical propulsion. In order to reduce energy necessary to get into Europa orbit, the spacecraft will spend two years performing multiple flybys of the Galilean satellites in what is know as the pump-down tour. After the tour, the spacecraft will capture into Europa orbit using chemical propulsion, then spend 30 days performing intensive remote sensing observations. Total mission duration is approximately 5.1 years. It is expected that this mission will consist of more than 300 orbits of Europa, with the objective of mapping the icy moon's entire surface. The Europa Orbiter craft must be designed to withstand severe radiation in Jupiter's system, the most intense radiation environment in the solar system. Spacecraft designers are developing creative configurations for self-shielding of the spacecraft. While the Galileo spacecraft was designed to withstand severe radiation, Europa Orbiter is being designed to withstand about seven times the radiation dose endured by Galileo. Europa Orbiter will receive a vastly higher radiation dose than Galileo because it will spend a long time at Europa's distance from Jupiter, within the high-radiation inner Jovian system. It will experience most of this radiation dose during the final three months before arrival and during its orbital mission around Europa. Galileo, on the other hand, was built with much heavier shielding and spent most of its time far outside the inner Jovian system. Although studies of solar panels are ongoing, a mission power requirement of more than 200 watts with the severe Jupiter and Europa eclipses will likely require the use of a radioisotope power source and a secondary battery system. The Europa mission was baselining ARPS-Stirling technology. However, the current baseline is the use of one flight spare RTG and one fabricated from spare parts of GPHS-RTGs from previous missions.

A.2 Pluto-Kuiper Express

The Pluto Kuiper Express (PKE) mission arrival was recently delayed to prior to 2020. This appears to imply a launch in the time period 2009-2011. It will utilize a ballistic trajectory to Jupiter. At Jupiter it will perform a gravity assist maneuver to gain energy and put it on the final trajectory to intercept Pluto. Total flight time to Pluto is on the order of 10 years (final mission architecture is still to be determined) with a post encounter data playback and Kuiper belt investigation extending the mission to approximately 12 years.

This mission originally baselined an alkali metal thermal-to-electric converter ARPS, but that is no longer the case. Power requirements are estimated roughly as ~200 We.

The power critical mode for the mission is encounter, when all instruments will be powered on and operating. In this mode the estimated total spacecraft power draw is 256 W. An auxiliary battery will be included and will draw down to approximately 70% depth of discharge during the encounter operations.

A.3 Europa Lander for Exploration

The future exploration of Europa (after Europa Orbiter) concentrates on surface and subsurface exploration. If Europa Orbiter finds compelling evidence of water on Europa, the next significant effort will be to place a lander upon the surface with an appropriate payload. The primary scientific goals of the proposed Europa Lander mission are to characterize the surface material from a recent outflow and look for evidence of pre-biotic and possibly biotic chemistry. The baseline mission concept involves landing a single spacecraft on the surface of Europa with the capability to acquire samples of material, perform detailed chemical analysis of the samples, and transmit the results to Earth.

The mission architecture is based upon a single vehicle consisting of a propulsion module, or carrier, and the lander. The Lander contains the bulk of the flight systems, including the power supply and electronics. The carrier contains the chemical propulsion elements necessary to inject into Jovian orbit. Europa Lander would perform the same pumpdown tour of the Galilean satellites to get into orbit of Europa. After the tour the carrier stage is discarded prior to entry and landing.

Surface operations are limited due to the thermal and radiation environment. A surface payload package and surface operations could be powered by an ARPS system. The primary science objectives are to analyze surface material from at least 0.5-m depths for prebiotic and biotic compounds, and to characterize the geophysical and geochemical environment. Integral to this mission will be some drilling and sample handling system for the extraction of samples and processing for analysis. Additional payload will provide analysis of the surface and monitoring of the seismic environment.

A.4 Titan Explorer

The Titan Explorer study is intended as a broad look at Titan exploration after Cassini/Huygens. Titan is an organic-rich satellite of Saturn. This mission investigates the surface environment with the intent to understand the distribution and composition of organics and the geological and geophysical processes that contributed to the evolution of Titan's prebiotic chemistry. The science objectives are to study distribution and composition of organics in the atmosphere, on the surface, and below the surface, to study the role of geological and geophysical processes and evolution in Titan's prebiotic chemistry and to investigate the dynamics, meteorology, cloud formation, and interactions of Titan's atmosphere with its diverse surface. Both aerobot and rover missions for Titan in-situ exploration have been considered. Additionally, an aerover mission (science return is a combination of the global aerobot science and the local rover science) is being considered.

The first mission studied was an aerobot mission. (As envisioned here, the term aerobot describes a balloon system that utilizes a condensable inflating fluid to control its altitude while drifting downwind). A strawman payload was selected based on current knowledge of Titan and instruments that are currently available or under development. Operationally, this is a global reconnaissance mission. Current knowledge of the winds leads one to expect that the aerobot could circle Titan every couple of weeks. The expected operational lifetime is about a month. The aerobot payload would continuously image and collect point spectra which it would use to decide where to sample. Radar would provide surface altimetry information. Samples would be collected by dropping a sampling device on a line and then reeling it in for analysis by a suite of instruments in the gondola. Wide- and narrow-angle imagery would be obtained of the sampling site. Both powered (steerable) and unpowered aerobots were considered.

The rover mission was assumed to have a surface range of several kilometers. The rover payload was a functional copy of the aerobot payload with seismometer and heat flow added. Sampling on the rover includes the capability to collect 10-cm cores for analysis by the instruments. The rover could operate largely autonomously or could be directed from the ground as in the Pathfinder mission. There would also be the possibility of re-sampling if an analysis proved to be of particular interest. The rover lifetime would be several months.

The Aerover mission is a combination of the aerobot and rover missions that initially performs the aerobot mission and then performs a rover mission. It was suggested that this could be accomplished with an inflatable-wheeled rover by over-inflating the wheels to provide aerobot buoyancy. This system would be robust for a variety of surface conditions. The mission would include a thirty-day aerobot phase followed by a thirty-day rover phase.

All of the point design mission concepts baseline a six -year indirect solar electric propulsion (SEP) transfer trajectory from Earth to Saturn launching in 2008. Upon arrival at Saturn the orbiter and lander enter together directly into the atmosphere of Titan, where a ballute (a hypersonic drag device combining characteristics of a balloon and a parachute) decelerates the spacecraft. At appropriate velocity, the orbiter separates and the lander continues to decelerate for entry. Power for the orbiter/carrier could be provided by two ARPS systems producing 212 We total at BOL. For surface operations, a single ARPS system is baselined for all options.

Additional surface mobility options have been proposed for Titan Exploration. In order to meet the science objectives of multiple samples while maintaining geographical diversity, various heavier than air options have been proposed. In all cases, the power required could potentially be provided by some form of ARPS technology.

A.5 Saturn Ring Observer

The Saturn Ring Observer is an ambitious concept to place a spacecraft in close proximity of Saturn's rings. This mission will study the rings and ring particles to better understand ring processes and evolution as a model for the origin of planetary systems. This will involve measurement of ring particle physical properties, dynamics, and spatial distribution. The science objectives are to make direct observations of the physical properties of the ring particles, kinematic processes in the rings, including velocity components in all three directions, scale height, coefficient of restitution in typical collisions, clumping/sliding/shearing behavior of particle agglomerations, and the spin states of ring particles.

A baseline mission concept has been developed. The primary arrival time at Saturn begins in 2014 and continues through 2020. This time interval corresponds to the maximum opening angle of the rings as seen from Earth. A point design trajectory has been defined that provides a 9-year flight time with a 2008 launch (2017 arrival). The initial trajectories examined were low thrust solar electric propulsion (SEP) trajectories, although ballistic trajectories and solar sail options have been investigated. Several options have been examined relative to the insertion scenario at Saturn. The implementation chosen uses a ballute single-pass aerocapture followed by a direct insertion above the Huygen's gap at apoapsis. This is a high technology scenario, but has the least required propulsive ΔV of all the scenarios (approximately 4590 m/s). The insertion is staged, using a separable chemical stage to deliver the ring observer to station. Once delivered, the ring observer spacecraft executes altitude maintenance above the ring plane and provides radial excursion capability. Science operations are scheduled for one month following insertion. The mission concept calls for placing the spacecraft in a ring-particle-like orbit with a very small inclination to the ring plane and then using four small plane change maneuvers per orbit to stay approximately 3 km above the plane. In effect, the spacecraft hovers above a particular point on the ring. Propellant is budgeted to make four changes in radial position after the initial orbit insertion.

System power for the spacecraft in this design would be provided by a combination of the ARPS and the solar arrays. The flight system would consist of three stages, the SEP stage, a chemical propulsion inter-stage, and the ring orbiter spacecraft. After launch, the SEP stage would provide the required ΔV to reach the planet. This stage completes its mission at 2.7 AU and is jettisoned. Prior to jettison, the solar arrays would provide power to the whole flight system. Following the jettison, power would be provided by two ARPS systems, providing 212 We at BOL. The maximum power mode for the ARPS system would be during science operations, where all the science instruments are powered.

A.6 Neptune Orbiter

The Neptune Orbiter mission is the next evolutionary step in the intensive study of the outer planets. Following in the heritage of Galileo and Cassini, the Neptune Orbiter mission will perform an intensive study of the Neptunian system from orbit. The overall science goals of the Neptune Orbiter mission are to study the rings, ring arcs, and shepherd satellites over a period of at least two years, perform intensive studies of Triton's surface and atmosphere, examine Neptune's atmosphere and magnetosphere, and study the satellites Larissa, Proteus, and Nereid.

A minimum energy transfer from Earth to Neptune would require more than 30 years, which is too long for a planetary mission. A number of alternatives to a direct transfer were examined. A Jupiter Gravity Assist (JGA) or flyby the most gain of the impulsive/ballistic trajectory alternatives; such a transfer is available for Earth departures in 2005, 2006, and 2007 and then not again until about 2017. A direct JGA transfer leaving in 2007 was selected as a baseline mission, with several solar electric propulsion options considered for launches later than 2007. The spacecraft would employ aerocapture at Neptune to capture into orbit and would spend at least 2 years in orbit.

The orbiter would employ two ARPS systems providing 212 We at BOL. The maximum power mode for the ARPS system would be during the data transmission modes from the spacecraft to Earth.

Appendix B - Mars Solar Power Characteristics

Mars surface solar energy systems are dependent on the level of solar irradiance at the surface. The pattern of extraterrestrial solar irradiance at any locality on Mars for any day of the year can easily be determined. The fraction that reaches the ground depends on the solar elevation and the optical depth of the dusty atmosphere. Under normal quiescent conditions, the optical depth is ~0.5 and the surface solar irradiance can be estimated at any locality on Mars for any day of the year. Under occasional dust storm conditions, the optical depth can increase to the 2-4 range resulting in significant diminution of solar irradiance.

The allowable duration of Mars surface missions depends upon several factors: array life, dust accumulation on photovoltaic arrays, battery life, and seasonal persistence of solar irradiance. If array life, dust accumulation, and battery cycling were not constraints, a solar-powered surface mission could operate on Mars over any interval for which the seasonal solar power availability is relatively constant (perhaps ±10%). Assuming that dust mitigation technology and long-life batteries and arrays can be developed, the seasonal variations in solar irradiance would limit surface mission duration as follows:

- In the north equatorial zone (5 to 20°N) there are no limits to mission life and solar availability is reasonably constant throughout the Martian year.
- As one moves northward, the interval of relatively constant solar availability becomes shorter and is centered on northern mid-summer (Ls ~70). At 30°N, 45°N, and 60°N, the allowable intervals are 330, 220 and 140 sols, respectively.
- As one moves southward, the interval of relatively constant solar availability becomes shorter and is centered on southern mid-summer (Ls ~250). At 0°, 15°S, 30°S, 45°S, and 60°S, the allowable intervals are 300, 260, 150, 110, and 75 sols, respectively.

If wider excursions are allowed in available power than the ±10% used here, mission life could be even longer than given above.

The solar irradiance on Mars is composed of the direct beam from the sun, plus the diffuse beam produced by scattering from the dusty atmosphere. Tilting photovoltaic arrays on Mars has less effect than on Earth because of the comparatively large diffuse component. If the diffuse component is isotropic, then tilting arrays to improve the solar elevation usually loses about as much diffuse component as it gains in the direct beam. On the other hand, at extreme polar latitudes, a vertical array that rotates to face the sun is the ideal collector for 24.6 hours of solar collection per sol in mid-summer.

The power system on a solar powered Lander or Rover is intimately coupled to a battery system to assure sufficient power for major daytime operations and survival overnight, including low power operations that occur at night. During the day, there are various power requirements for operations, typically including two significant peaks for transmission of data. At night, a variety of operations are carried out within the limited power available from a battery. These may include operation of instruments around the clock, observation of moons at night, and computation for data obtained during the day. Batteries are sized to produce an average power level overnight in the range of 10-20% of the noontime power. During the day, the batteries and solar array work together so that whenever more power is generated from the array than is needed by the Lander, the batteries are charging, and when the Lander requires more power than is generated by the solar array, the batteries are discharged. At night, the system works entirely on battery discharge power. Because of the overnight temperatures on Mars, some form of thermal control/management system is needed to keep the batteries from getting too cold overnight. One option is use RHUs to supply the needed heat at night, and this heat could be circulated via a loop heat pipe.

Horizontal solar arrays are expected to gather dust and decrease in performance at about 0.3%/sol initially, although the rate of diminution is likely to slow down with time. There is also evidence that the rate of degradation of performance may slow down with time, and the ultimate degradation might plateau out at perhaps less than 20%. However, this is quite uncertain.

Solar power systems can be modeled roughly for approximate purposes. The roughly estimated data are given in Table B-1 assuming no array cleaning. Table B-2 presents the data if array dust mitigation is available. The solar power baseline described in Tables B-1 and B-2 provides the state of the art of solar power against which a radioisotope power system must be compared. Table B-3 provides a comparison of radioisotope power with solar power on Mars for a generic radioisotope power system with an assumed specific power of 4.5 W/kg. It can be seen that the radioisotope power system weighs more than a solar system with same peak power, but the radioisotope power system:

- provides continuous power and heat for thermal control rather than intermittent,
- extends the allowable duration of the mission by a huge margin, and
- operates impervious to dust storms and shadows in canyons.

On the other hand, radioisotope power system requires a complex heat rejection system during cruise and in entry, descent, and landing.

Table B-1. Rough Estimates of PV System Masses Required for Various Mission Localities (Assuming No PV Array Cleaning)

Latitude	0-20° N	30-45°N	Northern Polar
Solar energy available per sol (W-hrs/m^2) sol 1	3500	3500	4400
Solar energy/sol after shadowing (W-hrs/m^2) sol 1	2800	2800	4200
Electrical energy per sol from solar (W-hrs/m^2) sol 1	620	620	920
Electrical energy per sol from solar (W-hrs/m^2) sol 90	510	510	920
Mission duration (sols)	90	90	90
Mission season	any season	mid-summer	mid-summer
Peak solar irradiance after shadowing (W/m^2) sol 1	450	400	150
Solar array size (m^2)	1.5	1.7	3.0
System conversion efficiency (solar to electrical) (%)	22	22	22
Peak Electric Power (W) sol 1	150	150	100
Peak Electric Power (W) sol 90	120	120	100
Solar array mass (kg)	3@	3.4@	14*
Average power level for 7 hour day (W) sol 1	96	96	100
Average power level for 7 hour day (W) sol 90	80	80	100
Minimum overnight power for 17.6 hrs (sol 1) (W)	15	15	N/A
Minimum overnight power for 17.6 hrs (sol 90) (W)	15	15	N/A
Battery voltage	28	28	N/A
Battery A-hrs rating	19	19	N/A
Battery W-hrs per 50% discharge	266	266	N/A
Battery continuous power for 17 hrs (W)	15.6	15.6	N/A
Battery mass (kg)	6	6	0

* Includes substrate and rotation mechanism.
@ Does not include substrate mass which is part of structure.

Table B-1. Rough Estimates of PV System Masses Required for Various Mission Localities (Assuming No PV Array Cleaning)

Latitude	0-20° N	30-45°N	Northern Polar
Thermal control mass (kg)	2	3	0
System mass (kg)	11	12.4	14

* Includes substrate and rotation mechanism.
@ Does not include substrate mass which is part of structure.

Table B-2. Rough Estimates of PV System Masses Required for Various Mission Localities (Assuming array dust mitigation is used and long-life batteries are available.)

Latitude	0-20° N	30-45°N	Northern Polar
Solar energy available per sol (W-hrs/m^2)	3500	3500	4400
Solar energy/sol after shadowing (W-hrs/m^2)	2800	2800	4200
Electrical energy per sol from solar (W-hrs/m^2)	620	620	920
Mission duration (sols)	any length	250	90
Mission season	any season	mid-summer	mid-summer
Peak solar irradiance after shadowing (W/m^2)	450	400	150
Solar array size (m^2)	1.3	1.5	3.0
System conversion efficiency (solar to electrical) (%)	22	22	22
Peak Electric Power (W)	130	130	100
Solar array mass (kg)	3.6@	4.0@	14*
Average power level for 7 hour day (W)	100	100	100
Minimum overnight power for 17.6 hrs (W)	15	15	N/A
Battery voltage	28	28	N/A
Battery A-hrs rating	19	19	N/A
Battery W-hrs per 50% discharge	266	266	N/A
Battery continuous power for 17 hrs (W)	15.6	15.6	N/A
Battery mass (kg)	6	6	0
Thermal control mass (kg)	2	3	0
System mass (kg)	11.5	13	14

* Includes substrate and rotation mechanism.
@ Does not include substrate mass which is part of structure; includes 1 kg for dust mitigation.

Table B-3. Comparison of Radioisotope Power with Solar Power
(Assuming the radioisotope power source has a specific power = 4.5 We/kg.)

Mission	Parameter	Radioisotope Power	Solar (No Dust Mitigation)	Solar (Dust Mitigation)
0-20°N	Duration	unlimited	90 sols	unlimited @
	Period of high activity	24.6 hrs/sol	6-7 hrs/sol	6-7 hrs/sol
	Peak active power (W)	100	100	100
	System Mass (kg)	22	10.5	11.5*
30-45°N	Duration	unlimited	90 sols	250 sols**
	Period of high activity	24.6 hrs/sol	7-8 hrs/sol	6-8 hrs/sol
	Peak active power (W)	100	100	100
	System Mass (kg)	22	12	13*
North polar	Duration	unlimited	90 sols	90 sols
	Period of high activity	24.6 hrs/sol	24.6 hrs/sol	24.6 hrs/sol
	Active power (W)	100	100	100
	System Mass (kg)	22	14	14

@ No solar limits but could be limited by battery or array life.
** Could be limited further by battery or array life.
* Includes 1 kg for dust mitigation.

Appendix C - Mars Science Enabled by RPS

C.1 Polar Missions

Polar missions that last longer than about 90 sols centered on mid-summer are only possible with a power source other than solar. In polar night there is no sun at all; in late summer and early spring sunlight falls at such a low angle of incidence that power from the solar cells is too weak to run a lander.

Scientific questions that could be addressed with a full Mars-year polar mission include:

a. Characterize the behavior of Mars' volatiles (CO_2 and H_2O) over a full year cycle.

   In the spring, CO_2 sublimates into the atmosphere, atmospheric pressure increasing as a result. What is the local behavior? How quickly does it evaporate in warm patches? How long does CO_2 frost remain in shaded sheltered locations? What are the resulting local weather patterns? What happens to the non-volatile residue left behind?

   In the northern summer, water in the north polar cap is exposed once the CO_2 has evaporated. What is the nature of the water cycle? Is water frozen into the soil? How much water is released in the summer?

   In the fall as temperatures drop, CO_2 begins to condense out of the atmosphere. How quickly does it return? Does it snow out of the atmosphere or simply condense on the ground? How much dust is entrained with it?

   In the winter, does the CO_2 anneal to form translucent ice or does it maintain a porous frosty nature? What exotic landforms are created in this winter wonderland?

   With a full cycle to study what can we learn about the formation and erosion of the polar layered terrain? Can we associate fine layering with yearly cycles? What does the accumulation of layers tell us about the climate history?

b. Study polar weather.

   Mars's polar caps control Martian meteorology in a much more direct way than, for example, Earth's poles. The Martian polar regions are the reservoir for a significant amount of its volatile inventory. Atmospheric pressure changes seasonally by ~30% as CO_2 condenses and sublimates at each pole. CO_2 transport from pole to pole drives the global circulation of the atmosphere. The ability to operate a polar mission for an entire Mars year will enable both global and local detailed studies of polar condensation / sublimation winds, which may cause local dust storms near the polar cap edge.

C.2 Equatorial - Mid-latitude Meteorology Missions

Although perhaps not as obvious as in the case for polar missions, any type of equatorial – mid-latitude mission could have significantly enhanced science return with a year-round, continuous source of power.

Solar-powered landers can have only minimal night-time operation, constrained by battery capacity. Night-time measurements of pressure, temperature, and wind velocity allow for complete diurnal observations of local weather.

Southern hemisphere meteorology has never been studied from the surface of Mars. A mission complementary to the Viking northern hemisphere missions should provide a data set that as a minimum, is equivalent to the rich Viking return, i.e., multiple Mars years. One example of a specific science investigation is the known lopsided transport of water between the two hemispheres. At the north pole, water is exposed and sublimates into the atmosphere in the summer. At the south pole, the temperature does not get above 140 K since the residual CO_2 cap doesn't completely sublime, thus the temperature does not get warm enough to evaporate water. The south pole is a cold trap, a sink for any water that comes in contact with it, while the north pole is a summertime source. Understanding the magnitude of the transport from north to south is important to the whole picture of Mars's water reservoirs and the water cycle.

C.3 Dust Storm Survivor

An interesting mission concept that could only be achieved by a non-solar powered lander is the "survivor" mission to study severe dust storms. A lander should be capable of surviving a severe dust storm if power were not a concern. Simple survival would be good, but even more interesting would be the case if the lander could make measurements during the storm. The Viking landers, in the northern hemisphere, were not in geographically favorable locations for severe dust storms. A lander targeted for example to Hellas or Noachis-Hellespontus or the Solis Planum-Argyre regions—the starting places of many severe storms—could study the onset, development and decay of a severe dust storm. This will certainly be important for any future human exploration missions in this region of Mars. The type of data such a mission could return might include:

- Conditions for storm onset,
- Development of high winds,
- Threshold for dust lifting,
- Mass and particle size distribution of lofted particles as a function of time,
- Witness samples on the lander to study erosive action on various types of materials,
- Correlation of features such as dust streaks observed remotely from orbit with *in situ* observations, and
- Storm decay back to normal conditions.

C.4 Geology Missions

Any sort of drilling operation will be much more efficient without the need for daily duty cycling due to power limitations.

A steady source of power would maximize the efficiency of a landed geology mission, while also simplifying the overall mission plan and minimizing daily adjustments for fluctuating power.

C.5 Mission Operations Benefits

The simplification of mission operations indirectly benefits science by freeing resources for science planning. Consider the '98 Mars Polar Lander (MPL) mission as a case study.

The MPL operations concept expected that every sol, depending on the opacity of the sky, a new prediction would be made for the power available to the solar-powered lander. Science plans had to be tailored to the predicted amount of power available on every sol. Significant margin was held for unpredictable changes in the weather. Spacecraft fault-protection routines that would have been triggered if the battery state of charge dropped below a tolerable level would have powered off the instruments and put the lander in safe mode. Return to nominal operations from safe mode could have cost several valuable days of the lander's limited mission life.

The planned duration of a working-day on the MPL ranged from 12 hours per day at the beginning of the mission to just 4 hours per day at the end of the mission. The remainder of the time the lander was in low power "sleep" mode. In order to acquire nighttime data, the lander had to be powered up from its low power state, the computer had to go through a reboot cycle, the solid state memory had to be configured properly to receive new data, and then after acquisition of the data the lander had to be powered down again. This complexity was driven solely by the necessity to manage power and energy.

Resources that might otherwise be available to a project for more science, on a solar-powered lander must be applied to the problem of power management. Software tools required include: prediction of power from solar arrays (including changes in time due to varying season and dust accumulation and sky opacity), use of power and energy by the spacecraft, predicted use of power by the instruments based on instrument states commanded, and a timeline tool to calculate predicted power balance based on the specific operational plan for the day.

Appendix D - Sun-Earth Connection Missions

D.1 Interstellar Probe

Science Objectives

- Explore the interstellar medium and determine directly the properties of the interstellar gas, the interstellar magnetic field, low-energy cosmic rays, and interstellar dust
- Determine the structure and dynamics of the heliosphere, as an example of the interaction of a star with its environment
- Study, in situ, the structure of the solar wind termination shock, and the acceleration of pickup ions and other species
- Investigate the origin and distribution of solar-system matter beyond the orbit of Neptune

Mission Description (Mid Term ~2010)

- Delta II 7425 Launch (719 kg Cap., $C_3 = 0$ km^2/s^2)
- Flight System Launch Mass: 564 kg
- Solar Sail Trajectory Targeted for Nose of Heliosphere
- 0.25 AU Solar Pass, 200 AU in 15 yrs.

Flight System Concept

- "Flying Antenna" Design Implementation (191 kg)
- Sized for 30-year Operations
- Payload: Fields & Particles + Imaging

Technology

- Solar Sail: < 1 g/m^2, 200 m radius
- DSN 70m Subnet w/ Ka-band Uplink
- Next Generation ARPS
- Next Generation System on a Chip
- Ka-band S/C Components and Phased Array
- Hot-Gas Propulsion
- Micro-S/C Technology
- Low Mass/Power Instrumentation

D.2 Interstellar Trailblazer

Science Objectives

- Explore the local interstellar cloud upstream of the heliosphere out to a distance of ~2000 AU
- Determine the scale size of structures in the local interstellar cloud
- Investigate variations in the elemental, isotopic, and ionic charge state composition of interstellar matter
- Investigate spatial variations in the distribution of energetic particles and dust
- Explore the limits of the Sun's influence on the local ISM
- Chart the Sun's environment for the coming 500 years

Mission Description (Far Term ~2020)

- Delta III Launch ($C_3 = 0$ km^2/s^2)
- Solar Sail Traj. Targeted for upstream direction
- 0.1 AU Solar Pass, 2000 AU in 30 yrs.
- Sail Jettison @ ~5 AU

Flight System Concept

- Spin-Stabilized Spacecraft (~225 kg)
- **Power: 600 W DC**

Measurement Strategy

- In-Situ Measurements of Elemental, Isotopic, and
- Molecular Composition of Interstellar Plasma and
- Neutrals, Low-Energy Cosmic Rays, and Interstellar
- Dust, via High-Resolution Spectrometry

Technology

- Solar Sail @ < 0.1 g/m^2, ~600 m Radius
- Thermal Control: = 0.1 AU Operations
- Communications @ 2000 AU
- **Advanced Power Generation**
- Miniaturized, durable instruments
- High-Temperature Materials

D.3 Outer Heliosphere Radio Imager

Science Objectives

- Determine the large scale structure of the heliosphere's outer boundary
- Provide a map of the 2-D shape of the outer heliospheric boundary, including the dynamic response to heliospheric disturbances and to the phase of the solar cycle

Mission Description (Far Term ~2020)

- Delta II Launch ($C_3 = 0\ km^2/s^2$)
- Solar Sail Traj. Targeted for Nose of Heliopause
- 0.25 AU Solar Pass, 20-40 AU in 2-3 yrs.
- Subsatellite Dispersal @ = 20 AU, 5-11 yr. Operations

Flight System Concept

- 1 "Mother" S/C and 16 "Subsatellites"
- Subsatellites Transmit Data to Mother S/C
- Mother Processes/Relays Data to Earth
- Maintains Interferometer Baseline Knowledge

Measurement Strategy

- Deploy Radio-Array @ 20 AU
- 16 Subsatellites Form Radio Interferometer
- Array Points Toward Heliopause (Ophiuchus)
- Process and Downlink 1 Sky Image/2 weeks

Technology

- Solar Sail @ 0.25 g/m^2, 300 m Radius
- **Next Generation ARPS**
- Micro-S/C Components and Subsystems
- Interferometer Multi-Baseline Acquisition and Measurement System

Appendix E - General Purpose Heat Source (GPHS)

E.1 Design Objectives

The 250-watt GPHS module was the first space radioisotope heat source which incorporated the following operational features:

a. Intact reentry and impact resistant design that prevents, or minimizes the release of the radioactive fuel during accident scenarios, and contains fuel during normal operations,

b. High operating temperature capability,

c. Modular design, and

d. Minimum mass.

E.2 Design Description

Each GPHS module is a rectangular parallelepiped shape, 9.72 cm x 9.32 cm x 2.65 cm, and weighs 1.445 kg. It produces heat from four pressed and sintered plutonium dioxide pellets, each weighing 151.5 g. A 250-Watt GPHS module would produce 62.5 Watts from each pellet due to the radioactive decay of 1 10 g of Pu-238. This heat output decreases over time as the Pu-238 decays (with a half-life of 87.75 years). A GPHS module that produces 250 Watts at launch will still produce 231 watts after 10 years (decays at ~0.79% per year).

Each fuel pellet is encapsulated in an iridium alloy especially developed for this application. The melting point of the Ir alloy is 2,454°C, but it forms a carbon-iridium eutectic at 2,256°C. To limit the grain growth in the iridium, its normal operating temperature is limited to 1,335°C. The iridium alloy capsule (or clad) is 0.559 to 0.635 mm thick and incorporates a vent to retain plutonium oxide particles and release the helium generated by the alpha decay of the fuel. Each Ir clad weighs about 50 g.

Two fueled clads are enclosed in a 4.24 mm thick composite fine weave pierced fabric (FWPF) graphite impact shell (GIS). Two impact shells are enclosed in a 4.7 mm thick FWPF graphite aeroshell. Between the impact shells and the aeroshell, a carbon-bonded carbon fiber (CBCF) graphite thermal insulator is used to limit the maximum clad temperature during reentry heating and to retain the clad's ductility during subsequent impact with the earth.

E.3 Heat Source Environment

The GPHS module was designed for use in a vacuum environment. An inert cover gas is used in the generator prior to launch to minimize oxidation of the graphite and other hot materials in the generator. Once the generator leaves the atmosphere, it is vented to the vacuum of space.

For use in a planetary atmosphere, such as on the surface of Mars, a sealed generator must be used to control the environment within the generator for similar reasons, not just because of the GPHS module materials.

E.4 GPHS Module Production

Pu-238 fuel pellets are produced and encapsulated at Los Alamos National Laboratory (LANL). LANL can normally produce 4 to 6 pellets per month. (As many as 13 pellets were produced in a month during Cassini.) LANL also produces purified oxide from scrap oxide within the inventory by removing impurities and decay products. There is a Bench Scale Scrap Recovery process in place to produce 300 g Pu-238/month and a full scale process (500 g Pu-238/month) is being qualified.

The iridium alloy material and clad vent set components are produced at Oak Ridge National Laboratory (ORNL). ORNL also produces the CBCF graphite insulators.

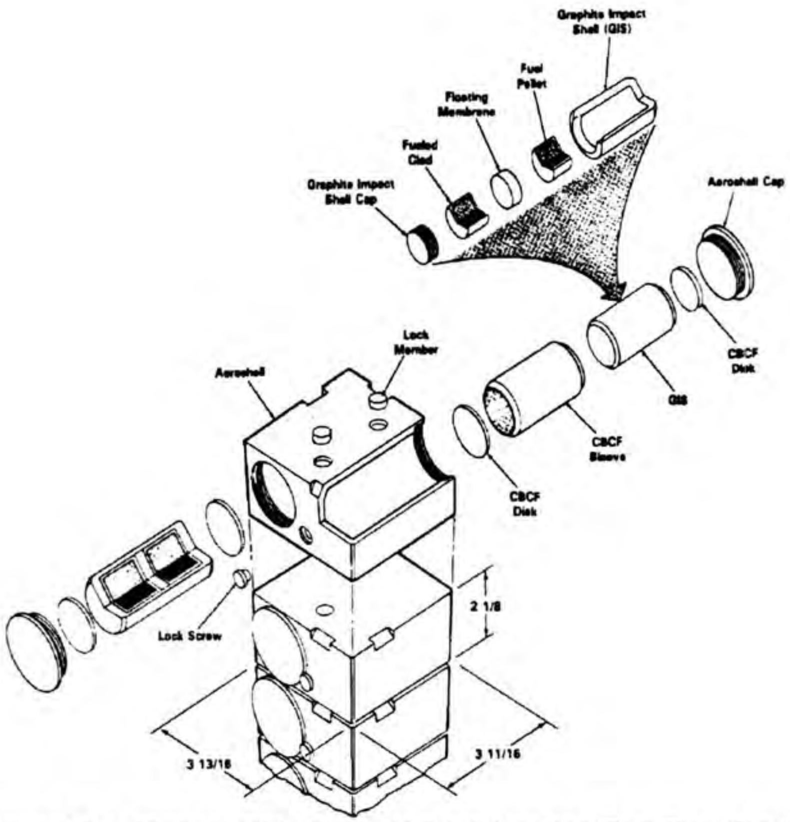


Figure E-1. Disassembled Impact Shell Holding Two Fuel Clads

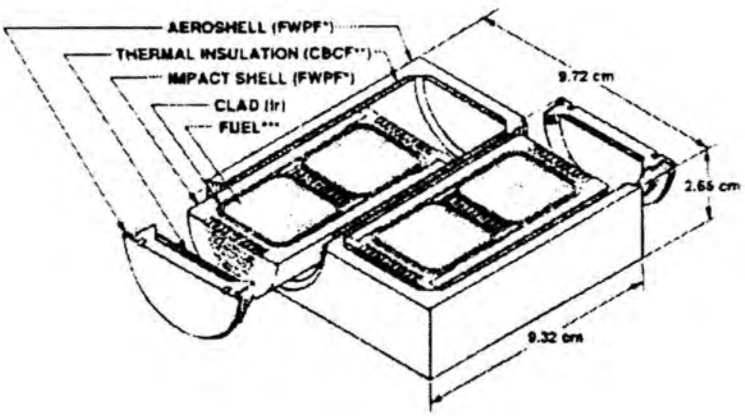


Figure E-2. Packaging of Two Impact Shells in a Graphite Aeroshell

Mound Laboratory procures the FWPF graphite billets from AVCO Textron and machines the GIS and aeroshell components. Mound assembles the GPHS modules (with fueled clads from LANL and CBCF insulators from ORNL) and outgasses them prior to loading them into generators supplied by the system contractor(s). GPHS modules and their individual components are produced under full nuclear quality control. Westinghouse provides Quality Assurance oversight support for DOE. Safety impact tests of fueled clads and GPHS modules are performed at LANL.

E.5 GPHS Module Availability

Radioisotope heat source availability is limited by the quantity of fuel which can be produced and processed or which is already in the inventory. In the U.S Pu-238 had been produced in reactors and processing facilities at DOE's Savannah River Site (SRS). With the shutdown of the last production reactor al the SRS, the U.S. no longer has a Pu-238 production capability.

DOE has a contract with the Russians to procure up to 10 kg of Pu-238 per year. Two purchases of Pu-238 were made in 1993 and 1995, totaling -9 kg at time of delivery. About 8 kg of Pu-238 from those two procurements remains in the DOE inventory for use in space power applications. This 8-kg is enough for about 18 GPHS modules, needed for one GPHS radioisotope thermoelectric generator. The current cost of the Russian Pu-238 is about $2,000 per gram. The contract with the Russians that permits additional purchases, expires at the end of 2002.

DOE is currently preparing a programmatic environmental impact statement which includes the evaluation of alternatives for re-establishing a domestic Pu-238 production and processing capability. Future Pu-238 sources and quantities will not be known until after the Record of Decision on the EIS is made.

Appendix F - Table of Acronyms

Term	Definition
ac	alternating current
AFRL	Air Force Research Labs
AMPS	Advanced Modular Power Systems, Inc.
AMTEC	alkali metal thermal to electric converter
ARPS	advanced radioisotope power system
ASEC	Advanced Stirling Engine Converter
ASTR	advanced Stirling
AU	Astronomical Unit
AVRS	adaptive vibration reduction system
BA&H	Booze-Allen and Hamilton, Inc.
BASE	beta-alumina solid electrolyte
BOL	beginning of life
BOM	beginning of mission
CBCF	carbon-bonded carbon fabric
CIL	critical item list
CTE	coefficient of thermal expansion
CTPC	component test power converter
DARPA	Defense Adv. Research Projects Agency
dc	direct current
DECOR	dynamic equilibrium cesium-oxygen reservoir
DNA	Defense Nuclear Agency
DOD	Department of Defense
DOE	Department of Energy
DTRA	Defense Threat Reduction Agency
EDL	entry, descent and landing
EIS	environmental impact statement
EMC	electromagnetic compatibility
EMI	electromagnetic interference
EO	Europa Orbiter
EOL	end of life
EOM	end of mission
ESS	Exploration of the Solar System
FMEA	Failure Mode and Effects Analyses
FWPF	fine-weave pierced fabric
SEP	solar electric propulsion
SOA	state of the art

Term	Definition
GIS	graphite impact shell
GPHS	general purpose heat source
GRC	Glenn Research Center
JGA	Jupiter gravity assist
JPL	Jet Propulsion Laboratory
LANL	Los Alamos National Labs
LMA	Lockheed Martin Astronautics
LTI	low temperature thermionic
MBE	molecular beam epitaxy
MEP	Mars Exploration Program
MHW	multi-hundred watt
MLI	multilayer insulation
MTC	miniature thermionic converter
MTI	Mechanical Technology, Inc.
NASA	National Aeronautics and Space Admin.
NMERI	New Mexico Engineering Research Institute
NRA	New Research Announcement
OP/SP	Outer Planet/Solar Probe
ORNL	Oak Ridge National Laboratory
OSC	Orbital Science Corporation
PDR	preliminary design review
PKE	Pluto Kuiper Express
PRD	pressure relief device
PRDA	Provisional R&D Agreement
PV	photovoltaic
QW	quantum well
R&D	research and development
RHU	radioisotope heater unit
RPS	radioisotope power system
RSG	Radioisotope Stirling Generator
RTG	radioisotope thermoelectric generator
S/N	serial number
SBIR	Small Business Innovative Research
SEC	Sun-Earth Connection
SNAP	Space Nuclear Auxiliary Power
SPDE	Space Power Demonstrator Engine

Term	Definition
SRS	Savannah River Site
SSE	Solar System Exploration
STC	Stirling Technology Corporation
STE	segmented thermoelectric
STTR	small business technology transfer
TA	thermoacoustic
TAMU	Texas A & M University
TAGS	Tellurides of Antimony, Germanium, & Silver
TDC	Technology Demonstrator Converter
TE	thermoelectric
TPV	thermal photovoltaics
TRL	technology readiness level
UCLA	University of California at Los Angeles