NEW FRONTIERS AO
RADIOISOTOPE POWER SYSTEM (RPS) INFORMATION SUMMARY
October 2003

All Radioisotope Power System (RPS) units used in missions proposed for this AO, including the services associated with their provisioning on space missions (e.g., National Environmental Policy Act (NEPA) Compliance, Nuclear Safety Launch Approval, Emergency Preparedness and Planning), will be provided by NASA and the Department of Energy (DOE) as Government Furnished Equipment (GFE) and Services (GFS). Funding for these units and services will be provided directly by the New Frontiers Program, and budgeted within the cost cap for each selected mission.

This document describes the RPS units that would be made available to proposed missions, and presents the costs that should be assumed in developing proposals. During Phase A study of the selected mission concepts, the teams that have proposed use of RPS in their missions will be provided a NASA Point of Contact (POC) who will support them in developing more refined approaches and cost estimates for accommodation of RPS.

1.0 Introduction

Radioisotope Power Systems (RPS's) generate electrical power by converting heat released from the nuclear decay of a radioisotope into electricity. First used in space by the United States in 1961, these devices have consistently demonstrated unique capabilities over other types of space power. A key advantage is their ability to operate continuously, independent of orientation and distance from the Sun. The systems are also long-lived, rugged, compact, highly reliable, and relatively insensitive to radiation and other environmental effects. As such, they are ideally suited for missions involving long-lived, autonomous operations in the extreme environments of space and planetary surfaces outside of Earth orbit.

Most RPS concepts consist of two principal elements: a heat source and a power conversion system. The heat source includes the radioisotope fuel encapsulated within a series of barriers and protective shells that prevents its release into the environment. The heat produced from this thermal source flows to a power conversion system, which transforms a portion of the heat into electricity. The remaining unconverted heat is removed, and rejected to space via radiators.

The RPS units employed by NASA have used plutonium-238 (Pu-238) fuel and thermoelectric devices to convert decay heat into electricity. Forty-four Radioisotope Thermoelectric Generators (RTG's) have been launched on 25 U.S. missions over the last four decades. The Apollo missions to the Moon, the Viking missions to Mars, and the Pioneer, Voyager, Ulysses, Galileo, and Cassini missions to the outer Solar System have all used RTG's. In each instance, the units met or exceeded their operational requirements. As a testament to their reliability and longevity, the RTG's on the Pioneer 10 spacecraft, which was launched in 1972, have operated reliably now for over three decades and continue to generate power well beyond the orbit of Pluto.

The Galileo, Ulysses, and Cassini, missions were each powered by one or more General Purpose Heat Source RTG's (GPHS-RTG's) shown in Fig. 1-1. Each unit was designed to deliver a nominal power level of 285 We (watts electric) upon final assembly from a stack of 16 General Purpose Heat Source (GPHS) modules. The next use of this unit is on the upcoming New Horizons Pluto mission, which is currently planned for launch in the 2006-2007 timeframe. This mission will use the last GPHS-RTG remaining in the U.S. inventory.

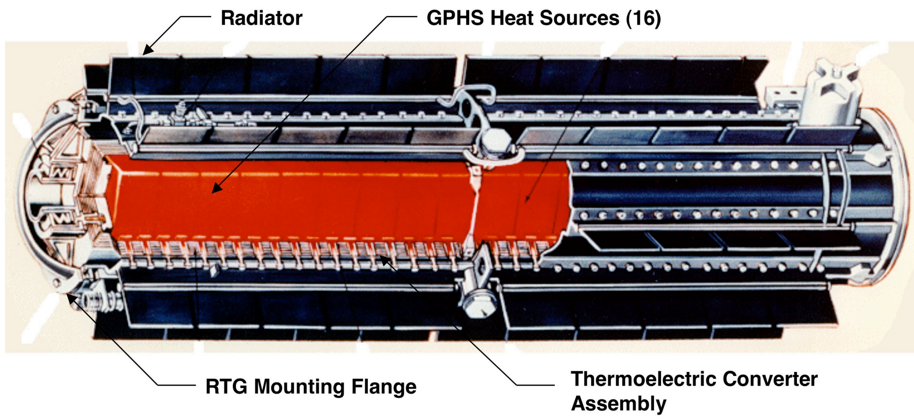

Figure 1-1: GPHS-RTG

The GPHS-RTG was designed to operate solely in space, but NASA's requirements for RPS on future missions have been expanded to include operation on planetary bodies, such as Mars. To accommodate this broadened range of capabilities, DOE and NASA have initiated development of two new RPS options: the Multi-Mission RTG (MMRTG) and the Stirling Radioisotope Generator (SRG). These units both use GPHS modules and operate at a power level greater than 110 watts electric (We) at beginning of mission (BOM). As with previous RTG's, the output power decreases over time due to the decay of the Pu-238 fuel, and the gradual degradation of power conversion components. The predicted power output for each unit over a 14-year mission is shown in Fig. 1-2.

The MMRTG and SRG differ primarily in their method of thermal-to-electric power conversion. The principal design requirements, capabilities and costs associated with these two units are shown in Table 1-1.

The following sections provide more information relevant to the provisioning of RPS for the New Frontiers AO. Sections 2.0 and 3.0 describe the MMRTG and SRG, including their requirements, design and capabilities. Section 4.0 provides an overview of the GPHS, the basic building block for MMRTG and SRG thermal sources. Finally, Section 5.0 summarizes all of the activities, processes and costs that AO respondents should assume in proposing an RPS-powered mission.

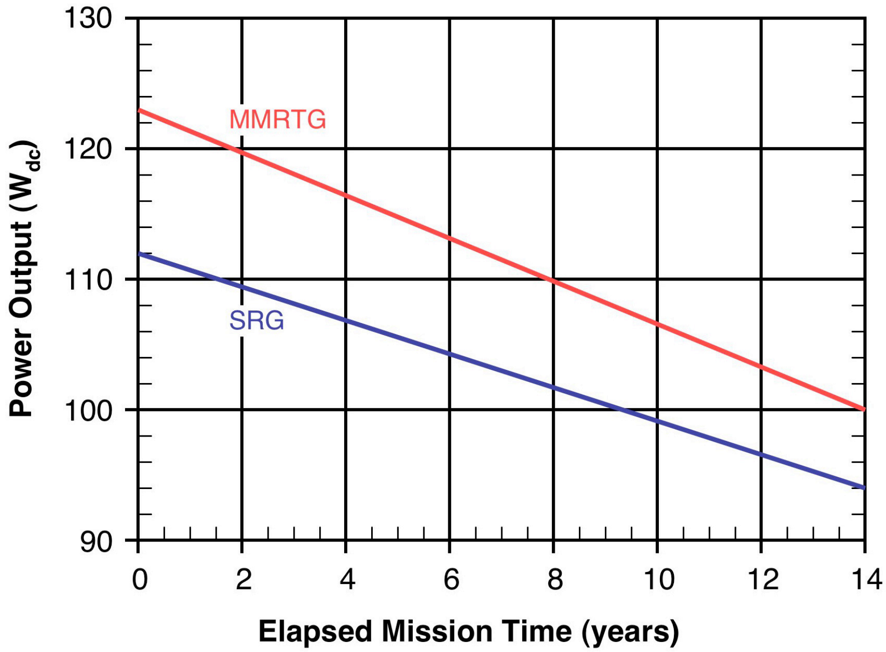

Figure 1-2: RPS Power Output vs. Time for Deep Space Applications

Table 1-1: Summary of RPS Capabilities and Costs

Power Source	**MMRTG**	**SRG**
Power (We)	> 110 BOM (nominal 123) ~100 @ 14 yrs	> 110 BOM (nominal 112) ~94 @ 14 yrs
Mass (kg)	40	34
Envelope (length x fin-fin width)	65.0 cm x 63.0 cm	88.9 cm x 26.7 cm
Fuel Load	8 GPHS modules (~4 kg Pu-238)	2 GPHS modules (~1 kg Pu-238)
Voltage (Vdc)	28 +/- 0.2	
Operational Environments	Space & Atmosphere	
Design Lifetime (yrs)	≥14	
Design Vibration Load (g^2/Hz)	0.2 (example for new ELV)	
Design Acceleration Load (g)	40 (example for new ELV)	
EMI/EMC (nT @ 1 meter)	25 (mission-specific)	
Sterilization (Mars only)	NASA 4A or 4B	
Availability	July 2009	See Section 3.2
Delivered Hardware Cost per Unit ($M)*	20	5, 15**

* Does not include additional costs for NEPA/EIS, Launch Approval, Emergency Preparedness and Planning, Integration, etc. See Section 5.0 for information on these elements.

** $5M per unit for first three units (fabricated under current SRG development program). $15M per unit for fourth and additional units.

2.0 Multi-Mission RTG (MMRTG)

The MMRTG differs from the current GPHS-RTG in two important aspects: it is roughly half the size of the GPHS-RTG (i.e., MMRTG uses eight GPHS modules and delivers at least 110 watts of electric power at BOM), and it is being designed to operate on planetary bodies as well as in deep space. The smaller modular design provides more flexibility in meeting the needs of a wider variety of missions.

Much of the MMRTG converter design is based on the SNAP-19 RTG, which was used successfully on the Viking 1 and 2 Mars landers and the Pioneer 10 and 11 spacecraft. The SNAP-19 RTG performed successfully in multiple environments ranging from the oxidizing atmosphere of Mars to the vacuum environment of deep space. The power conversion technology for the MMRTG uses SNAP-19 type thermoelectric elements configured in a similar series/parallel arrangement that provides redundancy. This approach minimizes development risk through use of flight-proven technology.

2.1 MMRTG Design

Figure 2-1 illustrates the MMRTG design, which consists of three basic assemblies: the heat source, the converter, and the outer case/radiator. The heat source consists of eight GPHS modules with a passive approach for collecting and venting helium gas generated by fuel decay. The converter employs Lead-Telluride/Tellurium-Antimony-Germanium-Silver (PbTe/TAGS) thermocouples similar to the SNAP-19 converter. The outer case and integrated conduction fin radiator is made of an aluminum-beryllium alloy (AlBeMet).

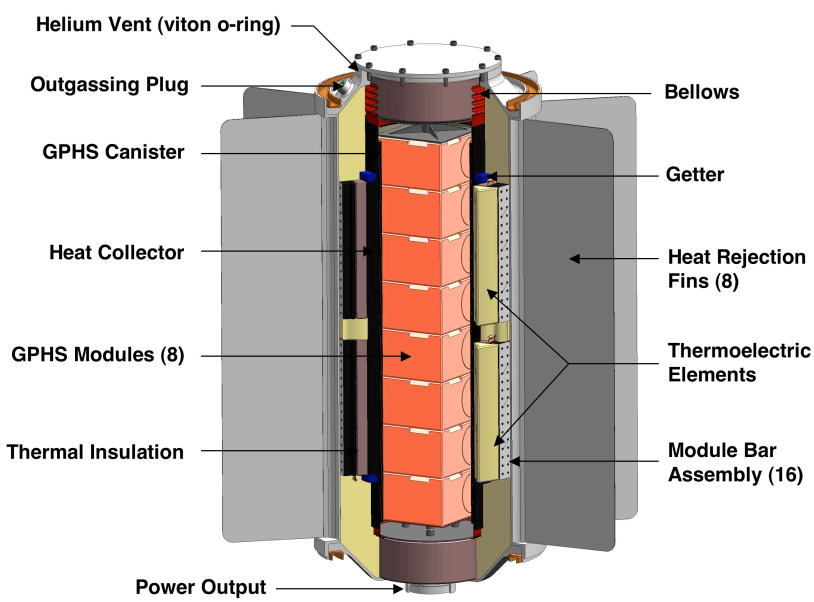

Figure 2-1: MMRTG Design Concept

The GPHS modules making up the heat source assembly are mated face-to-face to take advantage of their interlocking features. An external preload is applied to the stack of modules to prevent separation and misalignment during launch and application of mission loads. This preload is maintained by partially compressing the Min-K thermal insulation material on either end of the stack.

The stack of GPHS modules is contained within a thin superalloy liner that separates the GPHS assembly from the thermoelectric converter housing and its insulation. This liner provides separate gas-filled spaces between the GPHS assembly and the converter assembly. The decay-product helium diffuses out through Viton o-rings at the ends of the MMRTG. In the event of inadvertent reentry and ground impact due to a launch failure, this liner weakens and deforms sufficiently to release the GPHS modules, which are designed to withstand individually the aerodynamic heating and loads. In order to maintain the iridium clad temperatures between 960˚ C and 1300˚ C for launch safety, a few layers of multifoil insulation are sandwiched between the GPHS modules and liner.

A zirconium alloy "getter" material is enclosed in both the heat source cavity and thermoelectric converter housing to prevent oxygen from deteriorating system performance over the long 14+ year design lifetime. Potential sources of oxygen contamination include back-diffusion through the venting o-ring material and residual outgassing from the bulk thermal insulation package.

The thermoelectric converter assembly resides between the heat source liner and the outer housing wall. This cavity is hermetically sealed and filled with a cover gas consisting of 95% argon and 5% helium (to allow for leak checking). The heat is converted to electricity by 768 thermocouples configured in 16 series/parallel modules. The thermoelectric materials are PbTe for the n-leg and TAGS-85/PbSnTe for the p-leg, both standard materials currently in production. Nominal hot and cold junction temperatures are approximately 540˚ C and 210˚ C, respectively.

Figure 1-2 shows the predicted power profile over 14 years in deep space. At 250 thermal watts per GPHS, nominal MMRTG output power is 123 We at BOM and 100 We at 14 years. The system is designed to operate over a range of 23-36 Vdc, but is optimized to provide near maximum power to the spacecraft bus at 28 Vdc ± 0.2 over the design lifetime.

Waste heat is rejected from the converter housing via eight conductive fins located radially around the housing. Each fin is approximately 12.7 cm in length and extends along the entire 65-cm axial length of the converter. The fin tip-to-tip dimension is 63 cm. The housing and fins are constructed of AlBeMet (a lightweight, conductive, 62% beryllium/38% aluminum alloy) and are coated with a high emissivity material. The nominal fin root temperature is approximately 200˚ C. An auxiliary cooling tube runs axially along the base of each fin. This can be mated with an auxiliary heat removal system for mission phases in which active cooling is required.

2.2 MMRTG Development Plan

In June 2003, DOE awarded the contract for development of the MMRTG to the team of Boeing-Rocketdyne (Canoga Park, CA) and Teledyne Energy Systems (Hunt Valley, MD). The MMRTG project is targeting to make available three units (two flight and one spare) for the Mars Science Laboratory (MSL) mission, which is planned for launch in October-2009.

This section describes the milestones and products relevant to the New Frontiers mission, including the availability of MMRTG flight units. Because the MMRTG project is still in the early stages of development, the current development plan and schedule, which is outlined below, should be considered preliminary and could change as the project progresses.

The MMRTG project will complete fabrication and testing of an Engineering Unit (EU) by the end of 2004. It will have the same design as the flight unit MMRTG, but instead of using Pu-238-fueled GPHS modules, it will rely on heat produced from an electrical resistance heater that simulates the thermal characteristics of a GPHS stack. After 2004 the EU may be available for integrated tests of the MMRTG with other spacecraft systems. In addition, data from planned tests could be used to develop simulations and characterizations of MMRTG interfaces for integrated spacecraft modeling.

Final design of the MMRTG will begin in 2004, and Critical Design Review (CDR) of the units used for qualification tests and eventual flight is scheduled to take place in late-2004. At this time, detailed interface descriptions would be available for use on New Frontiers and other flight programs. It is reasonable to expect that further documentation describing MMRTG/spacecraft interfaces, and launch vehicle and supporting processing/launch range infrastructures could be provided by late-2004 to coincide with the midpoint of New Frontiers Phase B.

Fueling and testing of the qualification unit will take place in 2006, and will be completed by mid-2006. Models simulating MMRTG mass and thermal characteristics will also be built during this time and available by mid-2006. Current plans call for acceptance testing of the three flight units planned for the MSL mission by February 2008, August 2008, and February 2009. These units could be shipped individually once their acceptance tests are completed, but they will likely be delivered to KSC closer to the October 2009 launch date.

For New Frontiers, DOE could provide up to two more MMRTG's, beyond the three planned for MSL. However, the infrastructure for Pu-238 processing, flight unit fabrication and assembly restricts the maximum production rate that can be attained without significantly impacting program costs. Accounting for the three MMRTG's being considered for MSL, it is reasonable to assume that two units for New Frontiers could complete acceptance testing by April and June 2009, and be delivered to KSC by July 2009.

The New Frontiers mission should assume that it would be provided access to a spare if a problem develops with a flight unit at the launch site. However, this scenario is highly unlikely, and has never occurred before on any NASA mission. The New Frontiers program would not be charged for access to a spare.

3.0 Stirling Radioisotope Generator (SRG)

The SRG is similar to the MMRTG in terms of power output and multi-mission operational capability. However unlike MMRTG and the GPHS-RTG, it employs a dynamic Stirling cycle for heat-to-electric power conversion. This process is roughly four times more efficient than thermoelectric conversion. As a result, the SRG produces comparable power to the MMRTG, but employs only a quarter of the Pu-238.

Although dynamic systems are inherently more complex than their static counterparts, RPS technology programs and Stirling cryocooler flight applications over the last several decades have proven that these systems can be designed to meet RPS reliability and lifetime goals. The SRG is being designed to power and operational requirements similar to the MMRTG, as shown in Table 1-1.

3.1 SRG Design

The SRG design, shown in Fig. 3-1, consists of a beryllium housing, two Stirling convertors, an electronic controller located between the convertors, two GPHS modules located at either end surrounded by bulk thermal insulation, and auxiliary components mounted on the housing exterior. Thermal-to-electric power conversion is performed by the free-piston Stirling engines, which are integrated with a linear alternator in a common pressure vessel. Each closed-cycle Stirling engine converts the heat from one GPHS module into reciprocating motion with a linear alternator, thus producing an AC electrical power output of approximately 65 watts. The AC power is then converted to a DC power level of approximately 56 watts. The two convertors in each SRG deliver at least 110 watts of DC power to the spacecraft. The SRG has an anticipated efficiency of over 23% and specific power close to 3.3 We/kg at beginning-of-mission (BOM).

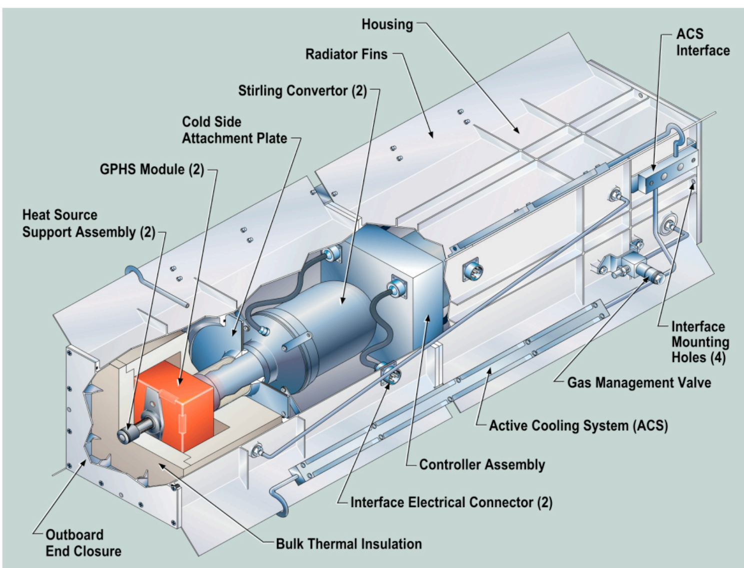

Figure 3-1: Stirling Radioisotope Generator (SRG)

The use of synchronous opposed Stirling convertor pairs reduces the natural vibrations of the dynamic system under normal operating conditions. The design of the 55 watt convertor is based on previous development work through NASA Small Business for Innovative Research (SBIR) contracts, a DOE development contract, and technology work at NASA GRC. Testing of a single 10-We electrically-heated Stirling convertor has over 70,000 hours of accumulated test time with no maintenance or degradation in performance. These data suggest that the SRG can meet the lifetime and reliability goals of NASA's deep space missions.

The controller assembly is mounted inside the housing between the two Stirling convertors and regulates the power output of both. The housing also provides a heat sink and protection for the controller from natural radiation and solid particles. Conceptually, the controller is designed to function autonomously in order to simplify operational interfaces with the spacecraft. However, it also includes an externally commanded function that reduces the stroke distance at the nominal operating frequency. This feature enables better utilization of diminishing heat from the GPHS modules. Selective redundancy in the design of the controller includes the use of separate control channels for the two convertors. However, these channels are interconnected, and the unit can sense and disconnect one convertor in the event of a failure. Separating the two GPHS modules at opposite ends of the generator minimizes the impact of convertor failure by allowing the user to continue operation of the surviving convertor at its nominal temperatures.

The externally mounted gas management valve (GMV) is used to charge the cavity surrounding the GPHS module with inert gas (argon) during all ground operations, thus preventing oxidation of hot-side components. Just prior to launch, the helium accumulated from isotope decay and the argon fill gas are replaced by lower-conductance xenon, which provides more power during launch. During launch, the pressure relief device (PRD) is actuated barometrically to vent the cover gas and helium generated by isotope decay. For missions involving operation on Mars, atmospheric gases (CO_2) are allowed to fill the generator interior. An analysis of the chemical interaction between CO_2 and graphite indicates an insignificant loss of mass from the GPHS aeroshell, and no impact on SRG performance. This scheme eliminates the need for selective venting or getter material, and allows use of the same generator design in a vacuum environment.

The electrical power interface connector is on the same side of the housing as the PRD and GMV, and is accessible for mating with the spacecraft. The generator has two additional electrical interface connectors, one on each half-housing. One connector is for telemetry data from the Stirling convertors. The other is the power output, and interfaces with a resistive shunt during times when the user load is not connected. This design allows secure transfer of convertor loading once the generator has been fueled.

The SRG design incorporates an active cooling system (ACS) that is mechanically attached to four fins. This is used in the event that active cooling is required during the launch and cruise phases of a mission. This approach enables separate fabrication and testing of the ACS, and allows installation during the latter stages of SRG assembly. Furthermore, the ACS can be easily removed for missions involving launch aboard an Atlas V or Delta IV, where active cooling may not be required.

The approach of using a bolt-on ACS is similar to the adaptations made to the GPHS-RTG for launch of Galileo aboard the Shuttle. The ACS consists of four pads that attach to the SRG fins and interconnecting tubing that joins the four pads to an interface manifold. The interface manifold is located adjacent to the mounting interface at the inboard end and is similar to that used on the GPHS-RTG, a flat surface that can accommodate either an o-ring or flat gasket seal. Prior to deployment, the tubing connections to the generator are severed pyrotechnically.

The operating efficiency and electrical power output of the Stirling convertors are a function of hot side and cold side temperatures. The hot side temperature is limited to 650˚ C by the creep properties of Inconel 718 used in the construction of the heater head. Reducing the cold side temperature to increase efficiency is limited by the increasing mass of the heat rejection radiators. The SRG concept described here is based on a cold side temperature of roughly 46˚ C at BOM.

The SRG DC power output is shown in Fig. 1-2 for a 14-year deep-space mission. The power profile is based on a total fuel loading of approximately 500 watts at beginning of mission (BOM). This power output profile is based on the radiator fin size that produces a nominal convertor cold-end temperature of roughly 46˚ C at BOM. The power output of 112 watts at BOM is based on demonstrated convertor performance and assumes an 88% thermal efficiency for the generator insulation system, and an 86% controller efficiency. With the stroke control option, the hot side temperature is maintained at approximately 650˚ C. Thus even with isotope decay, the electrical power output is 94 watts after 14 years.

3.2 SRG Development Plan

DOE initiated development of the SRG in May 2002 under a contract with Lockheed-Martin (Valley Forge, PA) and its principal subcontractor, Stirling Technology Co. (Kennewick, WA). NASA Glenn Research Center (Cleveland, OH) is also providing technical support. The SRG project currently plans to build four units (three flight and one spare) as an alternate/backup RPS for the MSL mission in 2009.

This section describes the milestones and products relevant to the New Frontiers mission, including availability of SRG flight units. Because the SRG project is still in the early stages of development, the current development plan and schedule, which is outlined below, should be considered preliminary and could change as the project progresses.

The SRG project will complete fabrication of an Engineering Unit (EU) by the end of 2004. This unit will closely approximate the SRG flight configuration, but it will use heat generated from two GPHS simulators powered by electrical resistance heaters. After 2004, the EU may be available for integrated tests of the SRG with other spacecraft systems. In addition, experimental results would be used to develop simulations and characterizations of SRG interfaces for integrated spacecraft modeling.

Final design of the SRG will begin in 2004, and Critical Design Review (CDR) of the units used for qualification tests and flight applications will take place in mid-2005. At this time detailed interface descriptions would be available for use on New Frontiers and other flight programs. It is reasonable to expect that further documentation describing SRG/spacecraft interfaces, and launch vehicle and supporting processing/launch range infrastructures could be provided by late-2004 to coincide with the midpoint of New Frontiers Phase B.

Fueling and testing of the qualification unit will take place in 2006, and will be completed in November 2006. Models simulating SRG mass and thermal characteristics will also be built during this time and available by late-2006. Current plans call for acceptance testing of the four flight units planned for backup on the MSL mission by June 2008, August 2008, November 2008, and February 2009. For MSL, these units would be shipped together so that they all arrive at KSC in March 2009 to support a launch in October 2009.

Because acceptance tests are ordered sequentially, individual units could be made available at an earlier date, if these units are not used for the MSL mission. If only one flight unit is needed, then a flight unit and spare could complete acceptance tests by August 2008 and be delivered to NASA KSC by September 2008. Similarly, two flight units and a spare could be delivered by December 2008, and a complement of three flight units and one spare would be available at KSC by March 2009. As with MMRTG, the infrastructure for fabrication and assembly limits the maximum production rate. However, two additional SRG units beyond the four under development could also be provided, but at a later date of July 2009.

The New Frontiers mission should also assume that it would be provided access to a spare if a problem develops with one of the flight units at the launch site. Provision of a spare should be factored into the delivery date, but would not be charged to the New Frontiers program.

4.0 General Purpose Heat Source (GPHS)

The GPHS is the basic building block for the heat sources within the MMRTG and SRG. It includes the Pu-238 radioisotope fuel encapsulated within a protective clad that prevents its release into the environment, and a series of protective shells that prevent damage to the clad during inadvertent atmospheric re-entry and impact. The GPHS is designed as a module to allow development of RPS units with different thermal outputs and power levels, and to enhance safety during launch.

A cut-away view of a single GPHS module is shown in Fig. 4-1. Each module contains four plutonium dioxide (PuO_2) fuel pellets, with a thermal output of approximately 62.5 watts per pellet. Each pellet is encapsulated within a vented iridium capsule, which functions as the primary fuel containment. The encapsulated pellet is called a Fueled Clad (FC). Each GPHS module contains four FC's encapsulated within two cylindrical Fine Weave Pierced Fabric (FWPF) containers, known as Graphite Impact Shells (GIS's). Thermal insulators made from Carbon Bonded Carbon Fiber (CBCF) surround each GIS. These insulators are designed to provide acceptable iridium temperatures during possible reentry, descent and impact. Two GIS's with thermal insulator disks and sleeves are placed in a rectangular aeroshell to form a GPHS module. The aeroshell is the primary heat source structure that provides reentry thermal protection for the FC's.

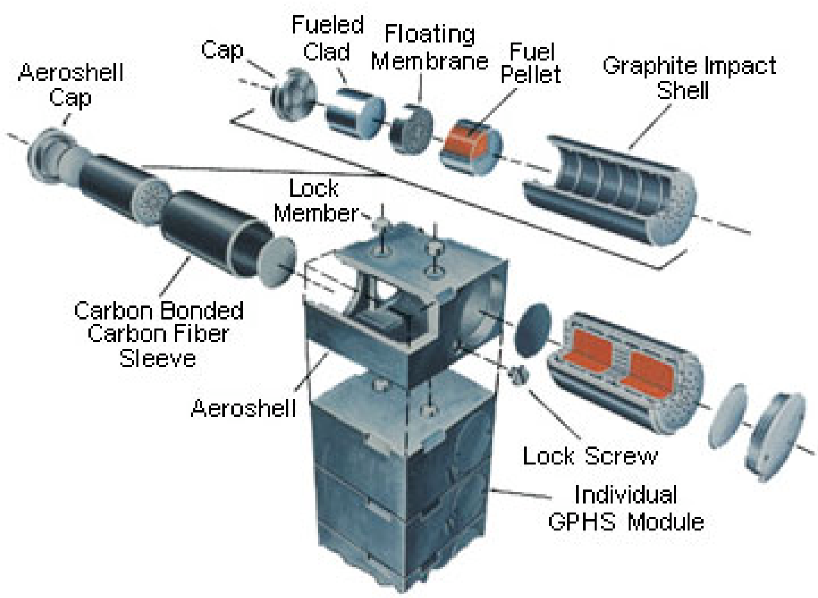

Figure 4-1: General Purpose Heat Source (GPHS)

The GPHS is designed to assure, to the maximum extent possible, containment and immobilization of the PuO_2 fuel. This applies to all mission phases, including ground handling, transportation, launch, ascent, temporary orbit, and during unplanned events such as reentry, impact and post-impact situations. The GPHS design requirements pertaining to fuel loading, mass, and dimensional envelope are summarized in Table 4-1.

Table 4-1: GPHS Module Design Characteristics

Fuel	• Plutonium Dioxide (PuO_2) • Pu-238 content is a minimum of 82% weight or total Pu	
Fuel Loading • GPHS Module • Fueled Clad	PuO_2 (grams) 604 151	Thermal Power (watts) 250 (average BOM) 62.5 (nominal at cal measure)
Maximum Mass (kg)	1.6 kg	
Envelope (single GPHS)	5.82 cm (height) x 9.32 cm x 9.96 cm	

The cylindrical, solid ceramic fuel pellet contains approximately 151 grams of PuO_2 fuel and provides a thermal inventory of approximately 62.5 watts at BOL. Each fuel pellet is individually encapsulated in a welded iridium alloy (DOP-26) clad. The DOP-26 alloy is capable of resisting oxidation in a post-impact environment while also being chemically compatible with the fuel and graphitic components during high temperature operation and postulated accident environments. The fueled clad also has an iridium frit vent that permits release of helium gas while blocking fuel particulates. The average activity is about 7400 curies per module (12.3 curies/gram PuO_2).

5.0 Provisioning of RPS for Nuclear-powered Missions

Potential use of RPS and radioisotopes in space requires many special considerations that must be accounted for in the budgeting and scheduling of a space mission. Most of these elements, such as National Environmental Policy Act (NEPA) Compliance and Nuclear Safety Launch Approval (NSLA), are well-defined, multi-year processes involving development of specific documentation and coordination among several government agencies.

Many of these elements are delineated in NASA guidelines available through links in the New Frontiers Program Library, while some have evolved as accepted practices over the years. For the New Frontiers AO, the special considerations for use of RPS have been divided into the six elements described below.

5.1 NEPA Compliance/Environmental Impact Statement (EIS)

NEPA requires federal agencies to consider, before an action is taken, environmental values in the planning of activities that may have a significant impact on the quality of the human environment. NEPA accomplishes this by directing agencies to evaluate alternative courses of action that may mitigate the potential environmental impact of a planned activity, such as use of radioactive material on a space mission. NASA's implementing regulations for NEPA can be found at 14 CFR 1216.1 and 1216.3. These regulations specify actions that can be expected to have a significant effect on the quality of the human environment. Such actions, which include the development and operation of nuclear systems, require preparation of an EIS.

Development of the EIS commences as early as possible in the development program (at least 5 years before launch), with a target for completion by Critical Design Review (CDR) or earlier (at least 3 years before launch). NASA Headquarters is responsible for preparation of the EIS and has enlisted subcontractors to assist in its development. Development of the EIS also requires development of a nuclear risk assessment by the Department of Energy (DOE), and participation by NASA Kennedy Space Center (KSC) and the Jet Propulsion Laboratory (JPL), NASA's launch nuclear approval engineering technical representative.

5.2 Nuclear Safety Launch Approval (NSLA)

For any U.S. space mission involving use of nuclear energy for heating or electrical power, launch approval must be obtained from the Office of the President per Presidential Directive/National Security Council Memorandum #25 (PD/NSC-25) paragraph 9. The approval decision is based on an established and proven review process that includes an independent evaluation by an ad hoc Interagency Nuclear Safety Review Panel (INSRP). The NSLA begins with development of a launch vehicle databook (i.e., a compendium of information describing the mission, launch system, and potential accident scenarios). DOE uses the databook to prepare a preliminary safety analysis report (PSAR) for the space mission. In all, three safety analysis reports (SAR's) are typically produced and submitted to the INSRP – the PSAR, an updated

SAR (USAR) and a final SAR (FSAR). The DOE project office responsible for providing the nuclear power system develops these documents.

The ad hoc INSRP conducts its nuclear safety/risk evaluation in three sequential steps following the PSAR, USAR and FSAR. The results of the INSRP evaluation are documented in a nuclear Safety Evaluation Report (SER). The SER contains an independent evaluation of the mission radiological risk. The DOE uses the SER as its basis for accepting the SAR. If the DOE Secretary formally accepts the SAR-SER package, he/she forwards the package to the NASA Administrator for use in the launch approval process.

NASA distributes the SAR and SER to other cognizant government agencies, such as DOD and EPA, and solicits their assessment of the documents. After receiving responses from these agencies, NASA conducts internal management reviews to address the SAR and SER and any other nuclear safety information pertinent to the launch. If the NASA Administrator recommends proceeding with the launch, then a request for nuclear safety launch approval is sent to the Office of Science and Technology Policy (OSTP) within the Office of the President.

From a historical perspective, DOE has requested completion of a mission's launch vehicle databook at least three years prior to launch. Although this schedule has emerged as a convention, it is not a requirement. In fact, there are incentives to begin the databook preparation process earlier and complete it sooner, if possible.

NASA Headquarters is responsible for implementing the NSLA process for NASA missions. It has traditionally enlisted JPL to assist in this activity. DOE supports the process by analyzing the response of RPS hardware to the different accident scenarios identified in the databook, and prepares a probabilistic risk assessment of the potential radiological consequences and risks to the public and the environment for the mission. NASA KSC is responsible for overseeing development of databooks, and traditionally uses JPL to characterize accident environments. KSC subcontractors are also under contract to provide information relevant to launch vehicle accident probability analysis, and other contractors assist in performing impact assessments and analyses. The development team ultimately selected for this New Frontiers mission would be responsible for providing payload descriptions, describing how the nuclear hardware integrates into the spacecraft, describing the mission, and supporting NASA KSC and JPL in their development of the databooks for the EIS and NSLA processes.

5.3 Emergency Preparedness and Planning

Any launch involving significant amounts of radioisotope materials requires special accommodations at the launch site to ensure mitigation of associated hazards arising from an unlikely launch anomaly. This activity involves deployment of emergency response team assets at the launch site and preparations to respond to any launch anomaly with radioisotope materials onboard. It also includes the detailed planning that must be conducted prior to deployment of these assets, including formulation of procedures for handling different accident scenarios. The deployed assets range in capability and size – from a small contingent (from one of DOE's Radiological Assistance Program (RAP) Regions) to larger resources (which could form the basis of a Federal Radiological Monitoring and Assessment Center (FRMAC)). The radiological emergency preparedness and planning requirements are tailored for each launch based on the understood risk (documented in the FSAR) and experience/lessons learned from previous missions using radioisotope materials.

As the Lead Federal Agency (LFA), per the Federal Radiological Emergency Response Plan (FRERP), NASA has responsibility for overall emergency preparedness and planning. As part of that effort, DOE performs the planning and preparedness functions, both on and off-site, associated with any response to launch anomalies possibly involving the release of radiological materials. DOE would provide the initial radiological response team, including command and control, for resources off-site under provisions of the FRERP. The funding for these activities would be provided to DOE directly by NASA as part of the overall project cost.

5.4 RPS/Spacecraft Accommodations, Processing and Integration

Use of RPS requires special provisions for accommodations and processing at the launch site. There are also unique aspects that have to be accounted for when integrating the unit(s) with the launch vehicle. RPS also requires special security to protect the units and the radioisotope fuel. This element begins early in the design process and culminates in activities directly supporting processing and integration at the launch site. The total cost associated with these activities should be estimated based on figures provided in Table 5-1.

5.5 Risk Communication

The unique issues associated with using nuclear materials on missions require extra measures to ensure communication of risk throughout all activities in the program. The design of nuclear-powered spacecraft depends on how technical decisions impact safety and development risk of the entire system. Most importantly, these impacts dictate how risks to the populace and environment are communicated to the public and key stakeholders. This activity ultimately supports all other nuclear-unique activities, such as NEPA Compliance, NSLA, and Emergency Preparedness, in addition to the activities usually conducted for any space science mission, such as education and public outreach.

5.6 Delivered Hardware

DOE provides RPS units for NASA missions per a 1991 interagency Memorandum of Understanding (MOU). Details relevant to specific flight missions and development programs are detailed in Supplements to the MOU. The provision and delivery of RPS units for this New Frontiers mission will be covered in a new MOU Supplement that will reflect the budget allocation for RPS in the eventual development contract. Flight unit delivery costs are expressed on a per unit basis, and include the purchase and processing of Pu-238, fabrication and integration of the RPS assembly, delivery, and acceptance tests.

5.7 Summary

Table 5-1 summarizes the time-phased costs associated with these different elements. Proposers should treat the cost estimate for each element and the data in Table 5-1 as a reference in developing the total cost of provisioning RPS units in their mission concept.

Table 5-1: RPS Provisioning Element Costs ($M)

Activity/Element	**FY04**	**FY05**	**FY06**	**FY07**	**FY08**	**FY09**	**TOTAL**
NEPA Compliance/EIS	0.4	1.0	0.6				2.0
Nuclear Launch Safety Approval*	0.2	0.8	1.5	2.5	2.0	1.0	8.0
Emergency Preparedness		0.1	0.1	0.1	0.1	1.6	2.0
Spacecraft Accommodations, Processing & Integration*	0.2	0.2	0.5	1.1	4.0	4.0	10.0
Risk Communication	0.1	0.2	0.2	0.2	0.5	0.8	2.0
Delivered Hardware Costs (for N flight units)**	0.1•T	0.2•T	0.3•T	0.3•T	0.1•T		T = N•C

Notes:
- • All costs (except Flight Unit Delivery) are independent of number of RPS units.
- * Does not include NASA KSC costs (e.g., launch vehicle data book, RPS accommodations). See ELV Launch Services Information Summary in New Frontiers Program Library for appropriate NASA KSC cost assumptions.
- ** Expressed as fraction of total Delivered Hardware Costs (T), where T = N (number of units) • C (cost per unit). See Table 1-1 for appropriate values for unit cost (C).