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LOW COST HIGH PERFORMANCE GENERATOR TECHNOLOGY PROGRAM

LCHPG Mission Application Study
July 1975

PREPARED FOR NUCLEAR RESEARCH AND APPLICATIONS DIVISION, ENERGY RESEARCH AND DEVELOPMENT
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LOW COST HIGH PERFORMANCE
GENERATOR TECHNOLOGY PROGRAM

MISSION APPLICATION STUDY

VOLUME 4

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Prepared for
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TABLE OF CONTENTS

	<u>Page</u>
Notice	ii
Contents	iii
List of Figures	v
List of Tables	vi
I. Summary	I-1
II. Introduction	II-1
III. Advanced Mission Application	III-1
A. DoD Mission	III-1
1. SURVSATCOM	III-1
2. Advance Defense Support Program (ADSP)	III-5
3. Laser Communications Experimental Satellite	III-6
4. Satellite Data System (SDS)	III-9
5. Global Positioning Satellite (GPS)	III-9
6. Deep Space Surveillance Satellite (DSSS)	III-14
7. Unmanned Free Swimming Submersible (UFSS)	III-19
B. NASA Missions	III-24
1. Pioneer H	III-24
2. Pioneer Jupiter Orbiter (PJO)	III-25
3. Mariner Jupiter Uranus (MJU)	III-25
4. Advanced Viking	III-27
IV. Nuclear Safety Considerations	IV-1
A. Defining Safety	IV-1
B. Designing Safety into a System	IV-3

TABLE OF CONTENTS (CONTINUED)

	<u>Page</u>
C. Influence of the Mission	IV-8
V. Specific LCHPG Applications	V-1
A. Pioneer H	V-1
1. Heat Source	V-8
2. Isolation Hot Frame	V-9
3. Thermoelectric Modules	V-10
4. Cold End Hardware	V-11
5. Insulation	V-11
6. Housing and Fins	V-12
B. Advanced Viking/Mobile Lander	V-13
C. Recommended 500 W(e) LCHPG Design	V-23

LIST OF FIGURES

<u>Figure Number</u>	<u>Title</u>	<u>Page</u>
III-1	Schematic of Liquid Loop System-Option One	III-4
III-2	LCHPG Installation on Spacecraft Configuration "B" (ADSP)	III-7
III-3	LCHPG Installation on Spacecraft Configuration "A" (ADSP)	III-8
III-4	LCHPG Installation on GPS II Spacecraft	III-11
III-5	225 W(t) Heat Source Module	III-15
III-6	Heat Pipe Thermal Control with Swing-Out Doors - VM Configuration	III-16
III-7	Conceptual Designs-UFSS Integration Study	III-21
III-8	LCHPG Installation on MJS Spacecraft	III-26
III-9	Advanced Viking Mobile Lander Power	III-28
V-1	Conceptual Design - 80 Watt LCHPG	V-3
V-2	Conceptual Design - 80 Watt LCHPG/Pioneer H Interface	V-7
V-3	Conceptual LCHPG for Advanced Viking Spacecraft	V-16
V-4	Cutaway of Conceptual LCHPG for Advanced Viking Spacecraft	V-17
V-5	Viking Wind Screen	V-18
V-6	Conceptual Design 500 Watt LCHPG	V-24

LIST OF TABLES

<u>Table Number</u>	<u>Title</u>	<u>Page</u>
III-1	Potential RTG Mission Integration Programs	III-2
III-2	Operating Temperature for VM Installation	III-18
III-3	VM-Capsule Heat Source Weight Summary	III-19
III-4	UFSS Radioisotopic Thermoelectric Generator Performance	III-23
IV-1	Accident Environments/Failure Mechanisms	IV-4
IV-2	Significant Heat Source Design Parameters	IV-6
V-1	RTG Performance Summary, Pioneer H	V-4
V-2	RTG Size/Weight Summary, Pioneer H	V-5
V-3	Safety Related Data, Pioneer H	V-6
V-4	RTG Performance Summary, Viking Mobile Lander	V-19
V-5	RTG Size/Weight Summary, Viking Mobile Lander	V-20
V-6	Congifuration/Material Summary	V-25
V-7	Performance Summary	V-27
V-8	Size/Weight Summary	V-28
V-9	Heat Source Safety Summary	V-29

I. SUMMARY

This report contains the result of initial efforts to investigate application of selenide thermoelectric RTG's to specific missions as well as an indication of development requirements to enable satisfaction of emerging RTG performance criteria.

Potential mission applications in DoD such as SURVSATCOM, Advance Defense Support Program, Laser Communication Satellite, Satellite Data System, Global Positioning Satellite, Deep Space Surveillance Satellite, and Unmanned Free Swimming Submersible illustrate power requirements in the range of 500 to 1000 watts. In contrast, the NASA applications require lower power ranging from 50 watts for outer planetary atmospheric probes to about 200 watts for spacecraft flights to Jupiter and other outer planets. The launch dates for most of these prospective missions is circa 1980, a requirement roughly compatible with selenide thermoelectric and heat source technology development.

A discussion of safety criteria is included to give emphasis to the requirements for heat source design. In addition, the observation is made that the potential accident environments of all launch vehicles are similar so that a reasonable composite set of design specifications may be derived to satisfy almost all applications.

Details of the LCHPG application potential is afforded by three designs: an 80 watt RTG using improved selenide thermoelectric material, a 55 to 65 watt LCHPG using current and improved selenide materials, and the final 500 watt LCHPG as reported in Volume 2.

The final results of the LCHPG design study has shown that in general, all missions can expect an LCHPG design which yields 10% efficiency at 3 watts/lb with the current standard selenide thermoelectric materials, with growth potential to 14% at greater than 4 watts/lb in the mid 1980's time frame.

II. INTRODUCTION

In order to identify the potential utility of the LCHPG, a Mission Application Study was initiated in May of 1974. Periodically from May 1974 to June 1975, various missions were identified and analyzed for application of the LCHPG design. However, during this same period, the LCHPG went through various design iterations aimed at optimizing the electrical performance and overall efficiency of the converter and heat source. Thus, as the predicted performance of the LCHPG design improved, based on a combination of converter design and increased ΔT between the hot and cold junction temperatures, so too did the utilization characteristics improve for the various mission applications. For example, the Advanced Viking/Lander LCHPG was predicted, in June 1974, to have an efficiency of seven percent. By the end of the optimization study, a ten percent system was feasible using the current standard thermoelectric materials.

A major part of this study effort was the technical liaison with major spacecraft planners in government agencies such as DoD and NASA and their respective industrial contractors. This liaison was to enable specification of selenide RTG characteristics to be identified which would satisfy a variety of mission requirements over a broad power range. To this end, design studies of heat source and thermopile were performed to define what the existing technology offers as well as the developments required for satisfying future safety and performance goals.

III. ADVANCED MISSION APPLICATIONS

The LCHP RTG was considered for primary application to a variety of applications in the early 1980's. User agency sources examined for this study were NASA, DoD and commercial.

In the case of the NASA missions, all are to outer planets for which solar energy conversion techniques are not practical. The DoD and commercial missions, all earth orbital, can potentially use other sources than RTG's; however, the RTG offers low degradation, long life, small size, no dependence on orientation or eclipse, and has radiation hardening capability. These advantages must be traded off, of course, against the potentially higher cost and the radioisotope safety considerations.

Eleven specific missions that have the potential for RTG use are summarized in Table III-1. These missions are discussed in more detail below.

A. DoD MISSIONS

1. SURVSATCOM

This program is officially listed as one of the United States Air Force Electronic Systems Division (ESD) major projects and is identified as System #2029. For the SURVSATCOM mission, three launch vehicles, Titan III-C, Atlas/Centaur and Space Shuttle are under consideration by the Air Force. Experience with RTG integration is available for Titan III-C (LES 8/9) and Atlas/Centaur (Pioneer 10/11). Pertinent available information for Space Shuttle payload bay accommodations is as follows:

TABLE III-1
POTENTIAL RTG MISSION INTEGRATION PROGRAMS

<u>MISSION</u>	<u>USER AGENCY</u>	<u>SYSTEM POWER REQUIREMENT</u> Watts(e)	<u>LAUNCH VEHICLE</u>	<u>MISSION PARAMETERS</u>	<u>TARGET LAUNCH DATE</u>	<u>SPACECRAFT CONFIGURATION</u>	<u>POTENTIAL SPACECRAFT CONTRACTORS</u>
<u>DoD:</u>							
SURVSATCOM	a. USAF (ESD)	a. 1200 - 1600	a. Titan III C	22,300 n. mi; semi-synch.	a.	Conventional (Attitude stabilized)	a. TRW, Rockwell
	b. USAF (SAMSO)	b. 600 - 1000	b. Atlas/Centaur		b.		b. TBD
	c. NASA (Houston)	c. TBD	c. Shuttle		c. 1980		c. TBD
Laser Com. Exp.	USAF (Hqtrs., STP)	150 - 400	Titan III C	22,300 n. mi; semi-synch.		Conventional (Attitude stabilized)	Hughes and McDonnell Douglas
Advanced Defense Support Program	USAF (SAMSO)	600 - 800	Atlas/Agena	Orbital (800/ 21,000 n. mi @ 63 1/2 °)		Conventional (Attitude stabilized)	TRW, Grumman and others
Satellite Data System	USAF (SAMSO)	800-1000	Titan III B/Agena	22,000 x 220/ 400 n. mi		Spin stabilized (1 RPM)	Hughes
Global Positioning System (NAVSTAR)	USAF (SAMSO)	400	Atlas F.	Orbital (10,400 n. mi; 12 hr.)		Conventional (3-axis stabilized)	Rockwell
Deep Space Surveillance Satellite (DSSS)	USAF (SAMSO)	1100 W(t)*	Atlas F.	400 n. mi. elliptical		Attitude Stabilized	Hughes, Rockwell
Unmanned Free Swimming Submersible (UFSS)	Navy (NRL)	1000	Submersible	N/A	N/A	N/A	N/A
<u>NASA:</u>							
Pioneer II	AMES/ESRO	160 - 200	Atlas/Centaur/TE- M-364-4/AKM	Jupiter Orbiter	1981	Boom Mounted (Spin stabilized)	TRW/MBB
PJO (with Probe)	AMES	160 - 200/50	Atlas/Centaur/TE- M-364-4/AKM	Jupiter Orbiter	1981	Boom Mounted (Spin Stabilized)	TRW, Hughes, McDonnell Douglas, AVCO
MJU (with Probe)	JPL	400/50	Titan III E	Jupiter/Uranus	1979	Boom Mounted (3-axis stabilized)	JPL, McDonnell Douglas, AVCO
Advanced Viking	Langley	65	Titan III E	Mars Mobile Lander	1981	Enclosed Module (like V'75)	Martin Marietta Aerospace

*Radioisotope heat source for cooling system.

- (a) The orbiter vehicle will provide mounting and venting capability for an auxiliary RTG heat removal KIT having the capacity to remove 48,000 Btu/hour. Specific request for this KIT was made by the Air Force since the Shuttle Orbiter Active Thermal Control (ATC) subsystem could not accommodate the additional heat dissipation requirements for RTG's after liftoff and during the time payload bay doors are closed.
- (b) Jet Propulsion Laboratory has conducted preliminary studies to investigate design of the KIT. An MJS'77 type spacecraft with four (4) MHW RTG's was examined and a schematic of one approach to heat removal is shown in Figure III-1.
- (c) It is believed that Rockwell/Downey will design the heat removal KIT and that JPL may provide the technical direction.
- (d) Primary contacts for this activity are:
 - Jack Fisher, NAR/Downey - 213-922-2111, Ext. 1976
 - Carl Fritz, NASA Marshall - 205-453-4263
 - Dr. Ewald Heer, JPL - 213-354-3060
 - Larry Dumas, JPL - 213-354-6504
 - Capt. Janson, JSC-SAMSO - 713-483-3541
 - Harold Lambert, NASA Houston - 713-483-5226
- (e) NASA MSFC personnel felt that the major obstacle to overcome is launch approval. It is possible that need for a fail-safe RTG may arise to assure adequate safety if the removal KIT fails and the payload bay doors cannot be opened.

FIGURE III-1

SCHEMATIC OF LIQUID LOOP SYSTEM - OPTION ONE

A primary source of information on Space Shuttle system payload accommodations can be obtained from the NASA document JSC-077000, Volume XIV entitled, "Payload Accommodations, Program Definition and Requirement."

2. Advance Defense Support Program (ADSP)

It has been determined that the selenide RTG will be very beneficial for a desired 63-1/2° inclination, 12-hour elliptical orbit. The major benefits are a 200 to 300 pound weight advantage over an equivalent solar-battery system and improved reliability relating to reduced complexity associated with an attitude controlled spacecraft. The 63-1/2° inclination orbit is one of three orbits under consideration. The other two, geo-stationary equatorial and synchronous polar, are more favorable to a solar-battery system but do not provide the highest degree of global coverage.

The following technical details have been obtained:

- (a) The power requirement is not fixed but is in the range of 600 to 800 watts(e) at 28 volts dc.
- (b) Mission life is 5 years; mean mission duration for subsystem design life is 7 years.
- (c) The RTG mission application involves 4 satellites in 63-1/2° inclined, 12-hour orbits with 20,990 n. mi. apogee and 800 n. mi. perigee. This enables 6-hour satellite dwell time over the north polar region during each orbit.
- (d) There is no requirement for thermal coupling to the spacecraft.
- (e) An Apogee Kick Motor (AKM) is a high probability requirement for the mission. The AKM would be mounted on the opposite end of the spacecraft from the primary earth pointing sensor. In this configuration, 2 RTG's may integrate more readily into the 10-foot diameter shroud. Since this

configuration was somewhat favored, it was further evaluated in design studies. The results of these studies are shown in Figures III-2 and III-3.

- (f) Meteoroid protection has not been specified, but will probably be the same as the GPS program.
- (g) The minimum flight flux constraint relates to the spacecraft memory. Limits are 1×10^{14} n/cm² (1 Mev equivalent) and 1×10^{10} rad (γ)/sec.
- (h) The launch vehicle is Atlas SLV-3A/Agena with 10 feet diameter shroud.
- (i) Testing is specified to MIL-STD-1540A, dated 1974.

3. Laser Communications Experimental Satellite

SAMSO is expected to prepare a wideband laser communications system for a flight experiment in 1979. The experiment will contain two packages - a solid-state neodymium-doped yttrium aluminum garnet (YAG) laser transmitter, developed by the Air Force Avionics Laboratory, and a carbon dioxide transceiver, to be supplied by NASA's Goddard Space Flight Center. An engineering model of the YAG transmitter is due to be completed by the Avionics Laboratory next summer. DoD and NASA have been defining ground rules for the joint experiment which would demonstrate the relative merits of the two different laser technologies for space communications. Initial flight tests will exercise the technology as a ground-to-satellite link although the Air Force ultimate goal is a wideband laser communications crosslink between satellites.

If the laser is electrically energized by a xenon lamp, total system power is on the order of 400 W(e). The laser is also amenable to being thermally energized by a solar pump. In this case, spacecraft electric power would be reduced by about 250 W(e).

FIG. III-2 LCHPG INSTALLATION ON SPACECRAFT CONFIGURATION "B"
(ADSP)

FIGURE III-3

LCHPG INSTALLATION ON SPACECRAFT CONFIGURATION "A"
(ADSP)

4. Satellite Data System

The Satellite Data System (SDS) is an on-going program managed by SAMSO. The mission duration is 7-1/2 years at a reliability of 0.935. The satellite is launched by a Titan III-B/Agenda with an apogee of 22,000 n. mi. and a perigee of 200 to 400 n. mi. Since the orbit is subject to Van Allen radiation, the current solar cell-battery system has a power degradation of about 200 watts to an EOM level of 618 watts. Cover glass radiation protection of the solar paddles is 60 mils and the total power system weight is 396 lbs. Of this total weight, approximately 50 lbs is attributable to power control system electronics which would be common to either nuclear or solar power systems. Four (4) spacecraft systems are currently being procured from Hughes of which the electric power system cost is estimated at \$4.6M or \$1.15M each. The spacecraft, which spins at 1 rpm, is integrated with the launch vehicle by Lockheed, Sunnyvale.

If solar panel degradation exceeds 200 watts, the RTG becomes a viable option, particularly if no other launch vehicle than Titan III-B/Agenda is available. Future EOM power growth may go to the 800 to 1000 watt range. To achieve the associated weight growth required by solar power, an increased launch capability in the form of Titan III-C/Agenda may be needed at a cost increment of \$7M.

5. Global Positioning Satellite (GPS)

a. System Requirements

The Global Positioning Satellite (GPS), called NAVSTAR, will permit any suitably equipped user to obtain continuous, three-dimensional position fixes to accuracies on the order of 30 feet. The anticipated accuracy of velocity information, particularly important for weapon delivery missions, will be of corresponding quality. To achieve

its full, initial operational capability a decade from now, the system will employ a complex of 24 satellites. The spacecrafts will be grouped equally into three rings situated in 10,400 n. mi. 12-hour circular orbits inclined by 73 degrees.

The spacecraft contractor, Rockwell International's Space Division, will initially provide three prototype satellites, each about 800 lbs with at least 300 W of available end-of-life dc power. These, plus an experimental navigational satellite, built by the Naval Research Laboratory, will be placed in two orbital planes by Atlas F boosters. In accordance with an over-riding philosophy of encouraging competition to get technical and cost benefits for the services, the program office wants to reopen the satellite segment to a fresh industry competition at the end of Phase I. The potential number of vehicles for the system is large for a satellite program. The initial 24 satellites would be supplemented by 4 or 5 satellites annually for a 30-year period. Anticipated mean-time-to-failure is about 5 years.

A major constraint of this program is an 800 lb payload throw weight which is imposed by specified use of the stockpiled Atlas F boosters.

b. RTG Integration

The GPS II spacecraft system power requirements of 400 watts(e) can be satisfied with two 200 watt(e) LCHPG's using a modified heat rejection system. Integration of two LCHPG's into the Global Positioning Satellite spacecraft general arrangement base-line configuration is shown in Figure III-4. Use of LCHPG's would eliminate the solar array panels including their deployment and orientation mechanisms, batteries, battery chargers, solar array control electronics unit, solar cell power conditioning unit, yaw steering and the thermal control louvers located in the shear panels.

The reference LCHPG design configuration described in Volume 2 (LCHPG Design Study) was scaled down to a 200 watt(e) unit. A preliminary system integration study

FIGURE III-4

LCHPG INSTALLATION ON GPS II SPACECRAFT

resulted in a system unit weight of 190 pounds. Significant system parameters are presented below:

GPS-II LCHPG SYSTEM PARAMETERS

Thermoelectric efficiency	13.75%
Thermal efficiency	81.4%
System efficiency	11.19%
Hot junction temperature	900 °C
Cold junction temperature	80 °C
Radiator temperature	50 °C
Power output (per unit)	200 watts
Thermal inventory (per unit)	1787 watts
Assumed temperature of space	194 °K
Fin efficiency of radiator	95%
Radiator area (per unit)	37.5 ft ²
Radiator web thickness	.020"
Radiator heat pipe spacing	2.8"
Number of radiator heat pipes (per unit)	44
Radiator heat pipe diameter	0.30"
Housing "spreader" heat pipe number (per unit)	8
Housing heat pipe diameter	0.30"
Ground service conditioning air flow (total)	1860 cfm
Conditioning air temperature (inlet)	37 1/2 °F
Total weight (per unit)	95 lbs
Radiator weight	0.66 lbs/ft ²

c. Component Optimization

The system parameters (listed on the previous page) were based on the use of axially grooved copper/water heat pipes. Aluminum/methanol, aluminum/ammonia and other pipe/fluid combinations must be evaluated to provide minimum weight and maximum heat transport capability in the chosen operational temperature range. Additional weight could be saved by further integration of the radiator and spacecraft: use of the radiator as a structural part of the spacecraft and extension of the heat pipes around the spacecraft in order to provide uniform surface temperature. The design integration study was based on the assumption of partial solar load on each radiator. If spacecraft orientation requirements result in one radiator constantly facing the Sun, and the other in shade, cross connection of heat pipes or other techniques can be used to spread the heat effectively over both heat rejection surfaces. Excessive radiator temperature fluctuations and minimum shaded temperatures may require the use of variable conductance heat pipes in order to provide thermal control of the spacecraft. Additional radiator surface areas may be required to dissipate heat from the electronic packages. Further weight saving might be accomplished by the use of second surface mirrors to improve radiator emissivity. The above system parameters were obtained using silicon stabilized zirconium oxide coatings as used on production SNAP 19 units. However, use of second surface mirrors, although expensive, may be cost/weight effective for a low temperature integrated design.

d. Alternate Configuration

Although only an integrated approach has been evaluated herein, an independent RTG/Radiator system (perhaps deployable boom mounted) may offer weight and integration advantages, particularly if the electronic package heat loads are high.

6. Deep Space Surveillance Satellite (DSSS)

a. Mission

The Deep Space Infra Red Surveillance Satellite has been considered for an outer space detection system having a 400 n. mile or higher polar circular orbit and utilizing an Atlas F launch vehicle. December 1978 has been targeted for launch of an experimental flight.

Cryogenic refrigerators for cooling infrared detectors require thermal inputs of approximately 1000 watts. As contrasted with a solar energy system, radioisotopic power advantages include increased hardness (nuclear burst survivability) and lighter weight. The latter is important in order not to exceed the payload capability of the Atlas F booster.

The heat source module, developed as part of the LCHPG program, was evaluated and integrated with the Vuilleumier (VM) refrigerator.

b. Basic Heat Source Module

The building block for the VM engine heat source is the 225 watt module presented in Figure III-5. As shown, this module contains nine fuel capsules of 25 watts each. The radioisotope is pure plutonium oxide (PPO) pressed into a cylindrical pellet .898 in. diameter x .586 in. long, and it is contained in a platinum-rhodium-tungsten alloy (Pt-3008) of 20 mils thickness. Each of the three groups of fuel capsules is surrounded by 0.100 inch pyrolytic graphite and the three capsules in a group are separated by compliant graphite felt. As indicated, the basic container and end plugs are made of 3D graphite. However, the depicted threaded plugs are under investigation and an alternate closure may be devised if necessary.

c. VM Engine Heat Source Assembly

The arrangement of five heat source modules around the VM engine cylinder shown in Figure III-6 yields the required thermal input (~1100 watts) in a compact, easily

FIGURE III-5

225 W(t) HEAT SOURCE MODULE

FIGURE III-6

HEAT PIPE THERMAL CONTROL WITH SWING-OUT DOORS VM CONFIGURATION

installed configuration. In this design, fueling is accomplished by positioning the pentagonal array of heat source modules which weighs about 22 pounds on top of a ring of fiberfrax insulation. Subsequently, the covering assembly of insulation, heat pipes and emergency heat dump doors are lowered around the modules and bolted in place. This geometrical arrangement and assembly sequence allows pre-installation checkout of the thermal control systems (heat pipes and emergency dump) as well as checkout of the VM engine with electrical heaters in place of the radioisotope fuel.

The central features of this heat source concern the thermal control by means of variable conductance heat pipes and the backup function of the emergency doors. During normal functioning of the VM radiator subsystem (Coolanol loop) the sodium filled heat pipes maintain the VM cylinder temperature at the specified operating temperature (1275°F), by transferring excess heat directly to the radiator. This control is effected regardless of the operating state of the VM engine, as long as the radiator loop is functioning. The emergency dump doors will be activated only if either the radiator loop fails, or the heat pipes fail (as would be expected after a very long time after the spacecraft has ceased functioning). In this event, the opened doors allow direct radiation of heat to space or through an intermediate barrier/radiator suitably sized for acceptable overall thermal conductance to space.

The temperatures of the cylinder wall, graphite and maximum fuel capsule are given in Table III-2 for the normal VM engine operating mode (cylinder wall at 690°C). Also presented are temperatures for the VM engine turned off and the heat pipes effecting thermal control, and for the emergency dump mode where no heat is assumed transferred to the engine or heat pipes.

TABLE III-2
OPERATING TEMPERATURES (°C) FOR VM INSTALLATION

	<u>Normal VM Engine Operation</u>	<u>VM Engine Off Heat Pipe Control</u>	<u>Emergency Dump Doors Open</u>
Cylinder wall	690	710	510
Average graphite	860	880	500
Maximum capsule	900	920	550

It is seen that during normal engine operation, the maximum capsule temperature is maintained at 900°C. When the engine is not operating, the variable conductance heat pipes maintain the cylinder wall and capsule temperatures (with small increase of 20°C) by dumping heat to the radiator coolant loop. In case of failure of the coolant and/or heat pipes, the emergency doors open. If the view to space is relatively unobstructed, the temperatures drop to modest values. However, even with an intermediate barrier or partial obstruction of view the temperatures can be maintained well below normal operating values.

The weights of all components shown in Figure III-6 are contained in Table III-3. It should be noted that the emergency doors only add about two pounds since the side insulation is required to limit heat losses to less than 50 watts during normal operation. Also, the heat pipes contribute only about one pound so that a doubling of the number of pipes for redundancy and/or temperature smoothing (if required) would add very little weight.

The preceding comments are a first look at the adaptability of the LCHPG modular heat source to the VM engine. Other approaches are possible, one being the multi-foil insulation thermal control system devised by Fairchild Industries. In order to provide a more thorough examination to include reliability consideration, additional efforts are

required. In particular, detailed integration studies between the spacecraft, VM engine and heat source contractors are required to achieve an attractive, workable design.

TABLE III-3
CAPSULE HEAT SOURCE WEIGHT SUMMARY

<u>Component</u>	<u>Weight (lbs.)</u>
Heat source assembly (modules, enclosure, support)	22
Top and bottom insulation and structure	3
Emergency dump doors (includes insulation and mechanism)	5
Heat pipe assembly (heat pipes, heat exchanger, graphite)	4
Total	34

7. Unmanned Free Swimming Submersible (UFSS)

(A more detailed discussion of RTG integration configurations and performance analyses are presented in "Unmanned Free Swimming Submersible/Low Cost High Performance Generator-Integration Study," TES-SNSO-3-31, July 1975.)

a. Mission Requirements

The Unmanned Free Swimming Submersible (UFSS) of the Naval Research Laboratory (NRL) is a vehicle envisioned for many search, survey and reconnaissance missions requiring long range and long on-station capability. The basic near-term mission calls for the UFSS being deployed overboard from a surface vessel, traveling near the water surface at a speed of 2 to 5 knots over a range of 500 miles, and then descending to the ocean bottom, where it will remain for about 60 days. A reciprocal operation returns the UFSS to the surface vessel. Total elapsed time for the mission is expected to be about 90 days.

The RTG electrical power requirement is 1000 watts, approximately half for propulsion and half for electronics. Three conceptual designs incorporating two types of thermoelectric materials and using both pod and hull mounted generator integration were evaluated.

b. Conceptual Designs

The three integration concepts investigated are shown in Figure III-7:

- i. Concept A is the enclosure of thermoelectric generators within an in-line, detachable pod. The prime configuration uses four 250 W(e), TPM-217 (selenide) generators. Use of TAGS/2N thermoelectrics in both a modified LCHPG shape and an HPG was also evaluated.
- ii. Concept B is the generators located in a bay of a single structure UFSS pressure vessel.
- iii. Concept C is a low center of gravity, slender pod approach for the power system.

The principle considerations for development of each design were to provide generators with acceptable thermal environments for maximum performance and safety. The configurations were designed to withstand stresses of depths to 3000 feet without yield (room temperature material properties, 4500 foot hydrostatic proof pressure). Power system weights varied from 614 pounds for Concept C (LCHPG) to 1056 pounds for Concept A (HPG).

c. Thermal Evaluation

The thermal evaluation consisted of predicting significant temperatures of the generator systems operating in water and investigating methods of cooling the generators at other times. While out of water, the primary concern is to protect the hardware from excessive temperatures. The limiting temperature is imposed by either the 85% lead-

DETACHABLE POD (CONCEPT A)

BAY (CONCEPT B)

SLENDER POD (CONCEPT C)

FIGURE III-7

CONCEPTUAL DESIGNS - UFSS INTEGRATION STUDY

15% iridium solder used in the cold end hardware (which has a melt temperature of approximately 550 °F) or by fuel capsule maximum temperatures of about 1000 °C.

The results of the thermal study indicated that during normal operation, any of the designs evaluated can be made to have cold junction temperatures sufficiently low to assure good operating efficiencies. Selection of the generator housing and mounting structure thicknesses determine the cold junction temperature. Converter design and thermal inventory provide the temperature differential that yields the hot junction temperature. While pod and hull thicknesses do not affect housing temperatures significantly, the thermal conductivity and thickness of the housing and attachment structure do influence housing temperatures greatly. Designs in which the housing directly contacts the water provide lighter weight and more efficient systems.

Auxiliary means for cooling the generators will be required when they are not completely submerged, i.e., aboard ship, in transit, etc. The various methods investigated included evaporative cooling, water cooling, convective heat rejection plates, and convective cooling in air. The optimum method depends upon the design configuration and support equipment available.

d. Thermoelectric Performance

Performance calculations for six combinations of generator and hull attachment (Concept A) and two designs using the generator housing as a pressure vessel (Concept C) are presented in Table III-4. Each design is based on a nominal 1000 watt electrical output at beginning of life (BOL). The LCHPG with current standard selenide and the HPG with TAGS/2N thermoelectrics were included in the performance evaluation. The thermoelectric efficiencies given in Table III-4 were obtained from data supplied by the 3M Co. for the selenides (without extraneous resistance or low temperature phase change effects) and by Teledyne computer code analyses for the TAGS/2N material.

TABLE III-4
UFSS RADIOISOTOPIC THERMOELECTRIC GENERATOR PERFORMANCE

Concept Design	Generator Configuration	Housing Thickness (in.)	Junction Temperatures		BOL T/E Efficiency (%)	BOL Thermal Inventory (watts)	BOL System Efficiency 1000 watts (%)
			Hot (°C)	Cold (°C)			
A	Two 500 W(e) LCHPG's, selenides	.75	900	162	11.85	9970	10.0
		1.00	900	141	12.30	9690	10.3
	Four 250 W(e) LCHPG's, selenides	.75	900	162	11.85	9970	10.0
		.90	900	150	12.10	9813	10.2
		1.00	900	141	12.30	9690	10.3
	Four 250 W(e) LCHPG's, TAGS/2N	1.00	535	162	7.90	13610*	7.4
		1.20	535	150	8.17	13175*	7.6
		1.33	535	141	8.35	12877*	7.9
	Six 167 W(e) HPG's, TAGS/2N	.50	535	141	8.35	12821	7.8
		.75	535	125	8.60	12068	8.3
	Four 250 W(e) LCHPG's, selenides	1.00	900	169	11.75	10060	9.9
	Two 500 W(e) LCHPG's, selenides	.75	900	138	12.30	9662	10.4
C	Two 500 W(e) LCHPG's, selenides	1.40	900	80	13.50	9145	10.9
	Six 167 W(e) HPG's TAGS/2N	1.40	535	80	9.0	11891	8.4

*Thermal efficiency of LCHPG (TAGS/2N) system has been extrapolated to lower temperature and adjusted for having one module on each of four sides.

Beginning-of-life thermal inventories listed in Table III-4 are dependent upon system efficiencies calculated for specific thermoelectric conversion material and junction temperatures. System efficiencies ranged from 9.9% to 10.9% for the LCHPG selenide thermoelectrics and from 7.4% to 8.4% for the HPG TAGS/2N material.

B. NASA MISSIONS

1. Pioneer H

Major European interest has been expressed in a potential Pioneer spacecraft Jupiter orbiter mission. The mission would involve placement of the backup Pioneer/Jupiter F/G spacecraft in a highly eccentric orbit around Jupiter for particle and field measurements, perhaps as far out from the planet as where the bow wave could be encountered. Technical feasibility studies have already been conducted by NASA/TRW and ESRO/MBB teams aimed at a December 1980 launch opportunity.

In addition, Pioneer H also has been considered by NASA for an out-of-the-ecliptic mission into a polar orbit of the sun.

2. Pioneer Jupiter Orbiter (PJO) with Probe

More recent interest has been expressed in applying the backup Pioneer F/G spacecraft to a Jupiter orbiter mission including an atmospheric entry probe of Jupiter. Jupiter atmospheric probe studies are now being undertaken in the wake of Pioneer F's findings that the Jupiter atmosphere is warmer and less dense than ground observations from Earth had predicted. The result was to move such a probe into the priority mission category recommended by NASA's science advisors.

3. Mariner Jupiter Uranus (MJU)

Mariner missions to the outer planets during the next decade will probably use spacecraft which are similar to the MJS scheduled to be launched in 1977. A preliminary investigation of a 500W(e) LCHP RTG integration with the Mariner spacecraft using advanced selenide thermoelectrics in place of the MHW RTG was made and is depicted in Figure III-8.

The major observations of this preliminary study were as follows:

- (a) The LCHP RTG can be readily integrated into the MJS spacecraft.
- (b) Assuming that three 150W(e) MHW RTG units will weigh 234 lbs., a direct weight saving of 103 lbs can be expected by use of a single 500 W(e) LCHPG RTG, which weighs 131 lbs.

- (c) The LCHP RTG with a BOL system efficiency of 11% will require appreciably less plutonium radioisotope fuel, a reduction on the order of 2650 W(t). This results in a substantial fuel cost saving on the order of \$1M to \$1-1/2M.

4. Advanced Viking

The advanced Viking studies which have been underway by NASA/LRC and their systems contractor, Martin Marietta Aerospace (MMA), have been actively supported by the Energy Research and Development Administration (ERDA) and Teledyne Isotopes over the past 2 years. During this evolutionary design evaluation and program planning activity, the selenide RTG has become a strong candidate to provide power for a mobile version of the backup Viking '75 Lander which is in the planning stage for launch during the early 1980's. Since the mobility of this Lander is directly proportional to individual RTG power in excess of the present SNAP 19 capability of 35 to 40 watts(e), and since it is desired to retain the present thermal management technique, the higher efficiency selenide thermoelectrics are very attractive. Figure III-9 shows that a 50 to 65 watt(e) selenide RTG can provide a range of greater than 50 km even with hazard avoidance and uneven terrain maneuvering. Moreover, an integrated system weight reduction of 36 lbs is achievable.

FIGURE III-9
ADVANCED VIKING MOBILE LANDER POWER

IV. NUCLEAR SAFETY CONSIDERATIONS

Adequate design of a nuclear power supply for a space mission implies that nuclear safety requirements are met. What is safety and what are the safety requirements? The following sections attempt to answer these questions. Also considered are: the systematics of designing safety into a nuclear power supply and the extent to which differing missions affect the design.

A. DEFINING SAFETY

In the process of defining safety, it is expedient to consider its antithesis-risk. For nuclear systems, the risk we are concerned with derives from accident sequences: occurrence of the accident, release of nuclear material so as to expose people (externally or internally) and the incidence of injury engendered wholly or partly by that exposure.

The steps in this process are random ones. The best system we can design will never preclude the risk sequence; the worst will never guarantee it. Therefore, risk is measured, in part, by a probability. The other notion involved is the consequence - the number of injuries or exposures. Qualitatively, then, the risk may be defined in terms of the net probability of exceeding a given number of injuries or exposures.

For practical use in system design, a more direct approach is needed. Guidance in this respect is provided by Ref. IV-1. Section 5.2 of that document implies an overall requirement for nuclear systems in the form:

Ref. IV-1: "Heat Source Safety Design Specification for Space Application," approved by Dobry, T. J., Dix, G. P., and Gabriel, D. S., July 8, 1974 (AEC/SNS-issued document).

$$PQ = \text{Constant}^*$$

where P is the estimated net probability of releasing radioactive material to the atmosphere and Q is the average of curies released as an airborne source term.

Thus, according to this criterion, any accident event likely to produce a large source term compared with other possible events must have a correspondingly smaller occurrence probability. This requirement must be met by design. How one attains this goal is described in the next section.

To see how the quantity PQ is related to risk, consider an exposure situation in which radioactive material is released in a populated region. The dose received by any person in this distribution (plume) of radioactivity is directly proportional to the number of curies released. The probability of injury per individual (p_{injury}) assuming a linear injury response is directly proportional to dose and, therefore, to Q. The expected number of injuries is merely the product p_{injury} and the number of people exposed in the plume.

If the expected number of injuries is small compared to unity (normally the case) it is numerically equal to the probability of one or more injuries.** Since as shown above, the expected number of injuries is directly related to Q, the probability of one or more injuries is directly related to Q, given fuel release, and PQ is likewise linearly related to risk associated with one or more injuries.

Thus we have defined risk and indicated that the product of fuel release probability and associated airborne source term is a direct measure of risk. Defined in this way, the risk may be controlled, in the design process, in accordance with Ref. IV-1.

A final consideration in the quantitative specification of risk or safety is that probabilities and source terms are generally subject to uncertainties of factors of three

*The constant is specified absolutely in Ref. IV-1 but will undoubtedly be revised, in time, from its present value.

**Assuming Poisson Statistics and with X = expected number of injuries, the probability of exactly 0 injuries is e^{-X} , hence the probability of one or more is $1 - e^{-X}$. This is approximately x, for $x \ll 1$.

or more, and further, significance would not be attached to differences much smaller than an order of magnitude. It is therefore appropriate to use the logarithm of PQ as a risk index, R,

$$R = \log PQ,$$

rather than PQ directly. Since safety is the antithesis of risk, we may also define a safety index, S,

$$S = -\log PQ.$$

If Q is expressed in curies, S will, in general, be a positive number. As S increases by one unit, therefore, the system is safer by an order of magnitude as a result of a lower failure probability, smaller source term, or both.

B. DESIGNING SAFETY INTO A SYSTEM

This discussion pertains primarily to radioisotope thermoelectric generators (RTG's). These systems comprise an integral heat source, whose operational function is to supply heat, and a thermoelectric converter which converts a portion of this heat to electricity and rejects the rest. For the most part safety requirements are met in the design of the heat source, although this is not a necessary restriction and need not provide an optimal system.

Consider first the performance requirements which dictate the nature of the heat source. The heat source must supply heat to the converter at interface temperatures in the neighborhood of 1000°C. This must be accomplished in a way such that the temperature profile within the heat source permits satisfactory operation during the mission and does not degrade its capacity for fuel containment under near-earth accident conditions.

Accidents to be considered for aerospace missions originate in malfunctions of the launch vehicle, spacecraft, or RTG. Accident consequences which must be considered are presented in Table IV-1. These are differentiated according to those occurring within the launch complex and later, in-flight accidents.

It is seen from Table IV-1 that a broad spectrum of thermal, thermochemical, and mechanical environments may be experienced by a heat source subsequent to a

TABLE IV-1
ACCIDENT ENVIRONMENTS/FAILURE MECHANISMS

<u>Launch Complex</u>	<u>In-flight</u>
Blast overpressure	Thermal stress
Fragments	Reentry heating
Fireball	Impact
Impact	Post-impact environments
Solid propellant	

vehicular accident. The sequential nature of the environments must also be considered. Thus, e.g., impact response in the launch complex and after reentry will depend on the effect of the preceding thermal environments.

To survive accident environments with reliability required by Ref. IV-1, typical heat source configurations (like that shown later in this report) have evolved. The unit heat source is a rectangular parallelepiped made of a graphitic material which provides protection against the thermal environments of reentry and launch pad mishaps. The rectangular shape provides a relatively large drag coefficient which, when combined with planar (large aspect ratio) mass distribution, limits reentry heating and reduces impact velocity.

The fuel capsules are comprised of fuel pellets or powder enclosed in metal cladding. This cladding provides impact resistance under the appropriate thermal and target conditions and resistance against loss of containment in post impact thermo-chemical (e.g., oxidizing) and mechanical environments.

Given the conceptual configuration for the heat source, we must determine the combination of individual design parameters so as to meet the safety specification (Ref. IV-1) while maximizing system performance as measured by electrical power divided by system weight. Note that, in view of the number of environments shown in Table IV-1, the safety index described above becomes:

$$S = -\log \sum_{K=1}^N P_K Q_K,$$

where P_K is the total probability of fuel release associated with the K^{th} nominal environment (see Ref. IV-1) and Q_K is the associated nominal source term.

The more significant design parameters with regard to their effect on safety are listed in Table IV-2. It is seen that many of these parameters have an effect on survival of the heat source for two or more failure mechanisms. Furthermore, it is possible that features improving survival chances in one environment may be detrimental in another. For example, use of a clad made of refractory metal generally increases its capability in a thermal environment but also decreases its resistance to oxidation.

TABLE IV-2

SIGNIFICANT HEAT SOURCE DESIGN PARAMETERS

<u>Component</u>	<u>Parameter</u>	<u>Related Failure Mechanism/Environment</u>	<u>Impact on Source Term (given failure)</u>
Capsule	Wall thickness	Impact, post impact, fragment.	Minor,
	Thermal inventory (size)	Impact, post impact.	~Linear dependence.
	Material selection	All.	
	L/D	Impact.	Unknown.
Heat shield	Surface thickness	Thermal stress, impact, solid propellant.	Primary dependence is on impact kinetic energy and its relation to the source term.
	Shape	Reentry heating, thermal stress and impact.	
	Intercapsule thickness	Impact.	
	Lateral dimensions	Reentry heating, impact.	
	Heat source volume		
	Material selection	All.	
Fuel	Physical state.	Impact, (pellet/powder).	Unknown for impact. Significant for solid propellant environment.

Source term effects are also listed in Table IV-2. The primary design parameters here are those which affect impact speed, physical size of the capsule, and parameters affecting containment time in a solid propellant fire. Larger capsules will produce larger source terms if they fail. For a fixed fuel inventory and impact speed, however, reducing capsule size will increase the total number of capsules, and therefore, the probability of failing at least one.

To design a heat source which meets the goal established in Ref. IV-1 requires modeling of the failure probabilities and source terms for each failure mechanism identified for the heat source. This must be done explicitly as a function of the design parameters. An example of how this is done in the case of the impact failure probability is provided in Ref. IV-2.

Thus, both system weight and safety index are functions of the design parameters for the heat source. By modeling each quantity explicitly, it is possible to select candidate configurations systematically, calculate associated weights and safety indices, and select the minimum weight system from the subset of candidates meeting the design specification. A simplified version of this process is presented in Ref. IV-2.

Ref. IV-2. "LCHPG Technology Program, Volume 2 Design Study," TES-SNSO-3-25.

C. INFLUENCE OF THE MISSION

From the preceding discussion one may conclude that that design requirements for a heat source depend upon the reliability of the launch vehicle. Employment of different vehicles implies that there might be sufficient difference in design requirements to warrant development of generators tailored to each application. Instead, Ref. IV-1 sets up accident environment/probabilities, to be used in deriving failure probabilities, without regard to specific launch vehicles and missions. We will consider in the following why this is justified.

First consider differences in missions and launch vehicles which could lead to custom heat source design. The most likely missions listed in Figure III-1 are either synchronous orbital or planetary missions.* The primary differences among these missions safetywise are in the spectrum of reentry conditions for misorientation accidents.

For synchronous missions the most severe reentry state at an altitude of 400,000 feet is a speed of about 36,000 feet per second and a vertical flight path. In contrast, initial reentry conditions for the planetary missions may exceed 40,000 feet per second at flight path angles in the neighborhood of 15° below the horizontal. While the spectrum of initial conditions is different, the consequences in terms of heat source response, primarily thermal stress, are about the same.

* A low-altitude, Earth-orbital mission is not considered in Table III-1. For this type of mission, the orbital lifetime may be only a few half-lives. Evidently special consideration must be given this kind of mission since reentry and impact survival and post-impact recovery requirements are significantly greater than for other missions.

Differences in design requirements resulting from launch vehicle considerations are also relatively small. The three launch vehicles to be used in the most likely missions between now and 1980 include Titan III-C, Titan III-E and Atlas/Centaur/TE-364-4. Titan III-C has less severe overpressures for launch pad explosions by virtue of having solely hypergolic liquid fuel. However, overpressure magnitudes have little contribution to the overall risk, based on Pioneer and Viking experience.

Both Titan vehicles have extensive solid propellant inventories in Step 0. Atlas/Centaur/TE-364-4 has a solid motor only in the third stage with approximately 2300 pounds of fuel compared with Titan's 850,000 pounds of solids. However, owing to the location of the delta stage near the payload, the exposure of nuclear fuel to burning solid propellant chunks was adjudged to be about as likely for the Atlas/Centaur vehicle as for Titan (comparing Pioneer and Viking safety analyses).

These and other detailed comparisons support the conclusion that neither mission type nor launch vehicle imply substantially different design requirements for nuclear heat sources. Accordingly, it appears to be advantageous to develop a standardized heat source, because of significant savings in R & D costs, compared with tailoring the nuclear system design to the individual application. It is implicit in this conclusion that varying power requirements among missions can be met by use of modular RTG or heat source concepts without introducing design penalties.

It is worthwhile noting that these remarks are (and must be) based on the limited information regarding launch vehicle failure modes and effects which are currently at our disposal. It is totally in keeping with the philosophy of developing a standardized heat source to investigate thoroughly the characteristics of the few launch vehicles to be used in missions planned in the next decade or so in order to confirm and quantify the preliminary conclusions above.

V. SPECIFIC LCHPG APPLICATIONS

Three specific classes of RTG's were conceptually designed during this study: an 80 watt Pioneer H unit using improved selenides, a 55-65 watt series of Viking Mobile Lander RTG's using both current standard and improved selenides and a 500 watt LCHPG RTG using current standard selenides. These designs are presented in the following sections.

A. PIONEER H

A conceptual design for a Pioneer H RTG was generated for the following assumptions:

BOL power output	80 watts
Hot junction temperature	800°C
Heat rejection temperature	163°C
Sink temperature	-56°C
T/E conversion material	improved selenides
Module insulation	Fiberfrax, vacuum

The resultant design is shown in Figure V-1. The generator is of a two-fin heat pipe augmented configuration with a system efficiency of 12%. This attractive efficiency results primarily from the use of the improved selenides. An examination of recent events in the selenide development program shows that achievement of the improved thermoelectric properties for a 1980 Pioneer launch may be optimistic. Nevertheless, the performance statistics for this 80 watt RTG are useful in showing the potential attractiveness of an advanced RTG, even at the conservative hot junction level of 800°C⁽¹⁾.

A performance summary on this RTG is given in Table V-1; a detailed weight statement in Table V-2, along with overall dimensions; and safety related data in Table V-3. This 80 watt RTG would weigh 21.5 pounds and occupy an envelope of 10.3 x 23.4 inches. Output power after three years is estimated at about 75 watts.

Figure V-2 is a depiction of two of these 80 watt RTG's installed on the same Pioneer spacecraft used for the Jupiter flyby flights.

(1) At the preparation of this section, additional study effort on Contract SNSO-3 had been initiated to examine RTG performance for hot junction higher than 800°C and rejection temperatures lower than 163°C to enhance efficiency. The work was performed on a 500 watt RTG using current standard selenides and is reported in Section V-C.

It may be noted, also, that the rejection temperature of 163°C, selected for this Pioneer conceptual design, is a compromise between RTG size, weight and efficiency. At some sacrifice in size and weight, efficiency can be improved by designing to a lower rejection temperature. This type of tradeoff would be performed in more detail and discussed with mission personnel prior to any RTG design selection. The generator design illustrated here is merely meant to show a representative case.

LEGEND

1. Fuel Capsule (24)
2. Heat Shield (6)
3. Housing Side/Fin with Heat Pipes
4. Thermoelectric Module & Thermal Insulation (2)
5. Cold End Compression Assembly
6. Thermal Insulation, Load Bearing, Typ
7. Lower End Cover
8. Upper End Cover, Fueling Port
9. Electrical Output Receptacle
10. Generator Mounting Bolt Holes (4 ea. end)
11. Heat Source Isolation Hot Frame Seal Bellows
12. Fin Stiffening Rib

FIGURE V-1

CONCEPTUAL DESIGN - 80 WATT LCHPG

TABLE V-1
RTG PERFORMANCE SUMMARY, PIONEER H

Output power, BOL (watts)	80
Load voltage, series parallel	4
Specific power, BOL (watts/lb)	3.7
Steady state temperatures, BOL (°C)	
Sink	-56
Radiator	163
Cold junction	177
Hot junction	800
Isolation hot frame	830°C
Heat shield, outer surface	905°C
Capsule	920°C
Efficiencies, BOL	
Overall	12.0
Thermal	90.0
Thermoelectric	13.4
Fuel inventory (watts)	666

TABLE V-2
RTG SIZE/WEIGHT SUMMARY, PIONEER H

<u>Component</u>	<u>Weight (pounds)</u>
Fuel	3.7
Encapsulating member	3.6
Heat shield	3.3
Isolation hot frame and bellows	1.8
T/E modules	1.0
Bulk thermal insulation	1.8
Housing/fins/ribs/saddles	4.5
Heat pipes	9.0
Cold end hardware	0.6
Connector and wiring	0.3
	<hr/>
Total	21.5
RTG height (inches)	10.3
Fin tip-to-tip, diagonally (inches)	23.4

TABLE V-3
SAFETY RELATED DATA, PIONEER H

Hypersonic ballistic coefficient, broad face, (lbs/ft ²)	21.7
Heating factor	1.7
Impact velocity, (ft/sec)	159
Mean failure velocity on steel, (ft/sec)	198
Conditional impact failure probability (p_f)	.017
Peak reentry capsule temperature (°C)	
Orbital decay (25,700 fps, -0.1°)	1515
Prompt reentry (44,000 fps, -10°)	1832
Capsule impact temperature (°C)	
Orbital decay (25,700 fps, -0.1°)	750
Prompt reentry (44,000 fps, -10°)	800
Heat shield recession, (in.)	
Orbital decay (25,700 fps, -0.1°)	.070
Prompt reentry (44,000 fps, -10°)	.090

FIG. V-2 CONCEPTUAL DESIGN - 80 WATT LCHPG/PIONEER H INTERFACE

The RTG mounting and deployment structure is composed of three parallel tubes equally spaced in an equilateral triangular pattern and attached to a machined triangular truss structure to which the RTG's are mounted. The tubular portion of this structure is retracted during the launch phase of the spacecraft mission and extended after trajectory injection by centrifugal force of the spinning spacecraft after an explosive bolt release of the retracted position restraint. Two of these structures are used on the spacecraft, oriented 120° apart in a plane perpendicular to the spacecraft spin axis.

In the installation shown in Figure V-2, the center of the RTG length is located approximately where the joining plane center between the two original RTG's was located, thus shortening the three boom legs several inches. Since these new RTG's would be lighter than the sum of the two that they replace, on each boom, the magnetometer boom will have to be shortened to achieve dynamic balance. This will require a reexamination of the magnetic field around the spacecraft with these new type RTG's which has not been done.

The delta truss frame on the end of the boom is essentially the same as the original Pioneer design though the thickness of the web and flanges of these members may be reduced as the RTG mass that they support is lower. The delta frame mounting plate to which the RTG is directly attached has been redesigned to fit this new RTG.

A detailed design description of this 80 watt RTG follows:

1. Heat Source

The heat source consists of six modules each 2.654 x 2.654 x 1.686 inches thick and each containing four fuel capsules surrounded by 3D graphite. This developmental heat shield material was selected for its potentially greater resistance to thermal stress cracking and its improved impact energy absorption characteristic when compared to

homogeneous graphites. Each fuel capsule has the following features:

Encapsulation material	Pt, 30%Rh, 8%W
Fuel	PPO
Fuel inventory, watts	27.75
Encapsulant wall thickness, inch	0.060
Fuel capsule dimensions, inches	0.877 dia. x .886 lg.
Helium vent location	end

The fuel capsules are positioned in machined recesses of the heat shield with the diagonal pairs installed from opposite faces of the heat shield block to accommodate the threaded closure plugs while allowing minimal capsule spacing. A minimum of 0.40 inch of graphite is provided for capsule reentry protection.

2. Isolation Hot Frame

The six heat source modules are assembled into a rectangular metal box called the isolation hot frame, made of TZM. This enclosure serves to isolate the heat source and converter gas environments (helium and vacuum, respectively) and also serves as a structure for reacting the thermopile spring loads.

The criterion applied in designing the frame is that creep be not greater than 0.010 inch at the center of each of the two sides on which the thermoelectric modules are located, after 24 hours of normal RTG operation. The selected design approach is to machine inwardly oriented stiffening ribs into the two larger faces of the frame. These ribs are centered under each row of couples in the thermoelectric module. The other two faces, as well as the lower end of the frame, are made from sheet metal gage flat pieces. Fusion welding along the edges of the ribbed and flat faces provides a gas tight unit.

The fueling port end of the frame is brazed to a metal seal bellows made of very thin Haynes 188 alloy. It is 3.76 in. inside diameter with nine convolutes per inch that are .125 in. high. The upper end of the bellows is formed with a short cylindrical flange that is seal welded at the upper cover to provide final converter section closure. The bellows construction provides a long thermal path and thus reduces the conductive heat losses as well as allowing a high degree of flexibility for accommodating the tolerance variations at generator assembly and dimensional changes in the generator assembly induced by differential thermal expansion.

3. Thermoelectric Modules

The generator design contains two thermoelectric modules with the following characteristics:

Thermoelectric material	improved selenides
Spring pressure, psi	300, N-leg 150, P-leg
Element diameter, inch	0.423 N-leg 0.371 P-leg
Element length, inch	0.3 inch
Number of couples	48
Load voltage, volts	4
Circuit	series parallel, 2 strings

The hot shoes are made of molybdenum and are sprayed with Al_2O_3 to electrically isolate them from the isolation hot frame. A rectangular piece of Fiberfrax 3.45 x 4.18 x 0.33 inches thick and provided with holes for the thermoelectric elements is used for thermal insulation.

4. Cold End Hardware

The cold end hardware supplies the required pressure load on the thermoelectric elements and a path for waste heat from the thermoelectric elements to the heat rejection surfaces of the generator housing. Helical compression springs made of 17-7 PH stainless steel wire are used to provide loading for the P and N legs. Though providing different loads, the springs are designed to have the same outside diameter and deflection to operating length. In order to provide a thermal path of low resistance, a flexible sleeve of aluminum foil surrounds each of the springs. The aluminum foil flexible sleeve assembly consists of a tube formed by wrapping the foil on a mandrel, slitting the tube walls for part of the sleeve length so that non slit end rings remain and then slightly bowing outward and flattening the resulting layered strips. This sleeve detail is then bonded at one end to a flat circular flanged end cap machined of aluminum alloy. The other end is solder bonded to the housing after the spring is installed. A thin disc of beryllia (BeO) is soldered to the capped face that rests against the thermoelectric module cold end conductor straps. The disc provides the necessary electrical circuit isolation. When the generator housing sides containing this cold end hardware are compressed during generator assembly the layered wall strips of the flexible sleeves buckle outward as the contained springs deflect. The buckling regions of adjacent sleeves are at different levels, so that a closely packed thermoelectric module may be utilized.

5. Insulation

The internal space between the isolation hot frame and the housing-fin assembly structure, not occupied by thermoelectric converter or cold end hardware equipment, is filled with Min-K like bulk thermal insulation of 20 lb/ft^3 density. On two sides and bottom, this insulation is 1.0 in. thick and around the thermoelectric module it is

approximately .90 in. thick. On the top, fueling end, it is 2.6 in. thick outside the bellows and 2.90 in. inside the bellows. This insulation was chosen for its load bearing qualities, required to support the isolation hot frame internal positive pressure on the non ribbed areas and to restrain and clamp the heat source assembly against dynamic excitation. It is recognized, however, that insulation development work must proceed to assure compatibility between the insulation material (Min-K or other) and the thermoelectric materials.

6. Housing and Fins

The generator housing and fins are made of 6061-T6 aluminum alloy plate welded together at the corners to form a rectangular box. Two sides with integral fins, two with integral mounting lugs (one with an electrical receptacle mounting boss) and two end covers make up the basic housing.

A bimetal seal ring and a circular fueling port closure disc are associated with the top end cover. The bimetal seal ring is machined from an explosively bonded bimetal disc that is aluminum alloy on one side and 300 series stainless steel on the other side. This seal ring is welded into the housing end cover and to the isolation seal bellows. The fueling port closure disc is welded on the cover flange after fueling.

The fin sides of the housing are machined with stiffening ribs and adjacent saddle grooves for heat pipe mounting. The web between ribs in the fin portion is .020 in. thick as are the ribs. In the housing region the web is .050 in. thick and the ribs are .100 in. thick. The inside face of the two sides over the thermoelectric module area is .100 in. thick and machined with circular recesses .050 in. deep that fit the cold end hardware sleeves. The other two sides are .050 in. thick over their entire area with the exception of the mounting lug and receptacle boss areas.

The fin is assumed nearly isothermal, a condition that is achieved through the use of heat pipes. Three heat pipes are solder bonded to copper coated saddles on each fin and extend from the fin tip across the thermoelectric converter area on the housing. The heat pipes used are the copper type of .25 in. inside diameter with closely spaced longitudinal capillary grooves. The total waste heat rejected by the heat pipe fins and housing surfaces is 586 watts which at 163°C T_{rad} requires 520 in.^2 of surface area. Each heat pipe is capable of transporting about 100 watts over a transport length of 6.7 in.

B. ADVANCED VIKING/MOBILE LANDER

Conceptual design studies were performed during June 1974 for a Viking mobile lander.

Conceptual RTG designs were initially created for the following ground rules:

Two-year power output	55 watts
Hot junction temperature	800°C
Maximum allowable radiator temperature	205°C (400°F)
Maximum sink temperature (2 hours, middle of day)	65°C (150°F)
Day conductive heat leak to structure	15 watts
Night ambient temperature	-126°C (-195°F)
Night convective coefficient	$1 \frac{\text{Btu}}{\text{ft}^2 \cdot \text{hr} \cdot ^{\circ}\text{F}}$
Night heat leak to lander switch	100 watts

Night conductive heat leak to structure	10 watts
Night radiator temperature	$\geq 93^{\circ}\text{C}$ (200°F)
Thermoelectric conversion material	current standard and improved selenides
Module insulation	Min-K, vacuum

Note that both near term and improved selenides were examined for this application to provide readiness for any conceivable advanced Viking launch. An accelerated selenide development program, coupled with propitious development results, could yield the improved material with its superior performance (about 3% better absolute efficiency) in less than 5 years. However, it is likely that the design would have to be based on current standard thermoelectric material if the mission occurs before 1980.

A large number of thermal requirements are apparent in the above test. First, the daytime sink temperature of 65°C is significantly higher than that encountered for high earth orbit or deep space missions (viz., -56°C). On top of this, a windscreen is required for the RTG, as on Viking '75, in order to prevent excessively low nighttime RTG temperatures that would result if the RTG were directly exposed to the Martian winds. Martin Marietta Corporation has estimated that, in order to ensure the transfer of 100 watts(t) into the Lander at night, radiator temperature at night must not drop below 93°C (200°F). The screen, however, creates a high effective background temperature for the RTG during the daytime, in addition to the already high day sink temperature of 65°C .

In order to satisfy the night radiator temperature requirement of $\geq 93^{\circ}\text{C}$ (200°F) and not produce an excessively large windscreen⁽¹⁾ and RTG radiator, day radiator temperature must be designed to a value approaching 205°C (400°F). This value, however, also represented an upper allowable limit because of the use of a conductive grease on the Lander thermal switch directly under the RTG.

The design selected to meet the Viking Mobile Lander requirements is shown in Figures V-3 and V-4. The windscreen is illustrated in Figure V-5. Performance characteristics and weights are summarized in Tables V-4 and V-5, respectively, for 55 watt versions of a current standard selenide RTG and an improved selenide RTG.

It may also be noted from Table V-4 that the current standard selenide RTG for 55 watts output power requires a fuel inventory of 805 watts. In addition to all the thermal requirements previously discussed for this application, concern was later expressed that the RTG fuel inventory not exceed 750 watts because of cruise mode spacecraft temperature limits that carried over from the existing Viking '75 design. To satisfy this constraint, power output would have to be reduced to about 51 watts. RTG weight would decrease about 2 pounds to about 26.

⁽¹⁾ Martin Marietta desired that the windscreen envelope not exceed 23 x 23 x 12 inches high, a requirement not quite satisfied with the current standard selenide RTG.

FIG. V-3 CONCEPTUAL LCHPG FOR ADVANCE VIKING SPACECRAFT

FIG. V-4 CUTAWAY OF CONCEPTUAL LCHPG FOR ADVANCE VIKING SPACECRAFT

FIG. V-5 VIKING WIND SCREEN

TABLE V-4
RTG PERFORMANCE SUMMARY, VIKING MOBILE LANDER

	<u>Current Standard Selenide RTG</u>	<u>Improved Selenide RTG</u>
Output power, EOM ⁽¹⁾ (watts(e))	55	55
Martian day temperatures, EOM (°C)		
Sink	65	65
Windscreen	132	132
Radiator	205	205
Cold junction	249	249
Hot junction	800	800
Martian night temperatures, EOM (°C)		
Sink	-126	-126
Windscreen	- 43	- 51
Radiator	123	113
Fuel inventory	805	579
Thermoelectric efficiency, EOM (%)	8.2	11.4
System efficiency, EOM (%)	7.0	9.7
Voltage, series parallel (volts)	6.0	5.8
Number of couples	82	72
Element sizes (inch)		
N-leg diameter	.393	.336
P-leg diameter	.344	.294
N or P-leg length	0.3	0.3
Specific Power ⁽²⁾ (watts/lb)	1.9	2.8

(1) EOM $\equiv$ 2 years after BOL.

(2) Including windscreen weight.

TABLE V-5RTG SIZE/WEIGHT SUMMARY, VIKING MOBILE LANDER

<u>Component</u>	<u>Weight (lbs) Current Standard</u>	<u>Weight(lbs) Improved</u>
Heat source	9.2	6.6
Thermoelectric modules	2.3	1.5
Bulk thermal insulation	1.1	1.0
Isolation hot frame and bellows	.4	.3
Cold end hardware	1.2	.6
Fins and heat pipes	7.0	4.0
Housing/end covers	3.5	3.3
Miscellaneous	1.7	1.4
Windscreen	1.9	1.0
	<hr/>	<hr/>
Total, including windscreen	28.3	19.7
RTG height, inches	9.8	9.0
RTG fin envelope, maximum, inches	34.5	26.6
Windscreen height, inches	10.3	9.5
Windscreen side, inches	25.1	19.8

Conversely, the improved selenide RTG requires only 579 watts to achieve 55 watts(e). Interest was expressed by Martin Marietta in obtaining up to 65 watts(e). At this level, fuel inventory would increase to 685 watts and weight would rise to about 24 pounds. Windscreen size for this 65 watt RTG would increase to 23 x 23 x 10 inches high.

A brief design description of this conceptual RTG follows:

This RTG consists of a heat source composed of a woven graphite matrix heat shield that houses eight (8) fuel capsules of noble metal construction containing three PPO fuel pellets of 28.5 watts thermal inventory each. This heat source is surrounded by a thin refractory metal sheath, connected to the fueling port opening by a bellows of the same material. The gas tight assembly thus formed isolates the thermoelectric converter section from the heat source section of the generator.

Four thermoelectric modules are located around the side surfaces of the heat source sheath, from which the thermoelectric couple hot shoes are insulated electrically by flame sprayed coating of alumina on the hot shoes. The elements in the module are loaded by springs in the cold end hardware. These springs are housed in flexible sleeves of aluminum that are solder bonded to the housing-fin side to facilitate conduction of waste heat to the radiating fins. Min-K type thermal insulation is used in the thermoelectric modules and at the generator end to minimize heat losses.

The square cross section housing is formed by welding together the four aluminum ribbed, heat pipe equipped fin-housing side assemblies around the thermoelectric module core. A clamping fixture is used to compress the cold end springs during welding. The pre-launch cooling tube assemblies around each end are then soldered into their prepared saddles and connected together to complete the cooling circuit. Generator assembly is

completed by welding attachment of the housing end covers, installing the eight (8) fuel capsulettes and welding of the fueling access cover in the upper end.

The wind screen is composed of thin aluminum bent-up angle frame welded together and covered with a resin bonded fiberglass sheet bonded or stapled to the frame.

C. RECOMMENDED 500 W(e) LCHPG DESIGN

The LCHPG design recommended for immediate development contains a heat source cavity of 800 in.³ enclosed in a membrane IHF. It employs fibrous insulation and operates at a hot junction temperature of 900 °C and a cold junction temperature of 135 °C. The fibrous system is chosen primarily because of low development risk and acceptable efficiency (>10%). Further, the fibrous system is 5% lighter than that with multi-foil and enables fuel capsule temperature 40 °C cooler through the membrane IHF application. Specific characteristics are contained in Tables V-6 through V-9, and a trimetric view is given in Figure V-6.

The system efficiency of 10.4% shown in Table V-7 assumes the use of Min-K type conductivity material (e.g., Min-K TE 1400) in the converter section of the generator. The use of Fiberfrax Hi Fi 660-H paper with currently measured thermal conductivity yields a slightly lower system efficiency of about 10.1%. With either insulation, system specific power at the 500 watt level is about 3 watts/lb. Complete weight details are given in Table V-8.

The highest capsule temperature is calculated to be 1025 °C a value believed well within the capability of noble metal clad materials. Other component temperatures summarized in Table V-7 likewise can be accommodated by the appropriate choice of materials.

The selection of the 800 in.³ heat source cavity volume is based on the safety response evaluation. A summary of that design from a safety viewpoint is given in Table V-9. Of particular significance is the low ballistic coefficient, the correspondingly low impact velocity and peak capsule temperatures and an overall safety measure superior to present technology heat sources.

FIGURE V-6

CONCEPTUAL DESIGN 500 WATT LCHPG

TABLE V-6
CONFIGURATION/MATERIAL SUMMARY

<u>Item</u>	<u>Material</u>	<u>Configuration</u>
Fuel	PPO or Shards	Right circular cylinder, 24.3 watts/pellet, L/D = 1
Encapsulating Member	Pt-3008	Right circular cylinder, 0.020 in. thick, 9 per heat shield
Heat Shield	Graphite, 3D Type	Parallelepiped, 22 per RTG
Reentry Insulator	Pyrolytic Graphite	Sleeve plus end caps; surrounds group of 3 capsules
Compliant Pad	Graphite Felt	Disc
Spacer for Capsules	POCO Graphite	Tube; axially separates capsules
Isolation Hot Frame	Pt-30 Rh	Square can open at ends
Bellows	Pt-30 Rh	Circular, 0.010 in. thick
Hot Frame Flanges	TZM at fueling end; Pt-30 Rh at other end	Transition pieces from square can to circular bellows
Seal Ring	Al /st Steel	Transition piece from bellows to housing; explosively bonded
Hot Side Dielectric	BeO	Either sprayed on straps or separate sheet
Hot Straps	Molybdenum	Sculptured; 0.060 in. thick
Thermoelectric P-Leg	Copper-Silver Selenide, Partitioned	Length = 0.3 inch Diameter = 0.227 inch No. = 634
Thermoelectric N- Leg	Current Standard Gadolinium Selenide	Length = 0.3 inch Diameter = 0.297 inch No. = 634
Cold Straps	Copper	0.010 in. thick; 4 string parallel circuit
Cold End Hardware	Aluminum, BeO Dielectric, 17-7 PH Steel Spring	Flexible sleeve type; 20 one-mil wraps; helical, round wire compression type spring
Converter Insulation	Min-K TE 1400 or Fiberfrax HiFi 660-II Paper	Min-K 20 lbs/ft ³ ; Fiberfrax 12 lbs/ft ³

TABLE V-6 (Continued)

<u>Item</u>	<u>Material</u>	<u>Configuration</u>
End Plug Insulation	Min-K TE 1400 or Min-K 2000	2 inches thick
Housing	Aluminum	Rectangular section, 0.050 in. , min.
Fins	Aluminum	0.020 in. , 27.8" x 39.5", 4 per RTG
Heat Pipes	Copper/Water	Internal axial grooves; 44 per RTG; 0.297 in. O. D.
Housing Rib	Magnesium	2.3 inches high; 0.080" thick
Fin Rib	Magnesium	Tapered from root-to-tip; 0.035" thick
Electrical Receptacle (power output)	TBD	Hermetically sealed to housing
Emissive Coating	ZnO + K ₂ SiO ₄ binder (IITRI Z-93)	0.005" thick
Controlled Leak Device	TBD	Located in one end cover
Heat Source Cavity Gas Fill	Helium	Few psi up to 1/2 atm.
Converter Gas Fill	Vacuum	Leak-tight

TABLE V-7
PERFORMANCE SUMMARY

OUTPUT POWER, BOL (watts(c))	500
LOAD VOLTAGE (volts)	28
SPECIFIC POWER, BOL (watts/lb)	3.0
STEADY STATE TEMPERATURES, BOL (°C)	
Sink	-79
Radiator	105
Cold Junction	135
Hot Junction	900
Isolation Hot Frame	930
Heat Shield, Outer Surface	935
Heat Shield, Maximum	970
Pyrolytic Sleeve, Maximum	1015
Outboard Capsules	1015
Inboard Capsules	1025
EFFICIENCIES, BOL (%)	
Overall	10.4
Thermoelectric	12.4
Thermal	83.9
FUEL INVENTORY (watts(t))	4807

TABLE V-8
SIZE/WEIGHT SUMMARY

<u>Component</u>	<u>Weight (pounds)</u>
Isolation Hot Frame	10.4
T/E Elements	5.2
Thermopile Hardware (hot straps, cold straps, dielectric, cold end hardware)	12.9
Insulation	13.1
Heat Pipes	8.4
Heat Pipe Saddles	3.0
Fin (Web)	8.5
Housing/Covers	11.9
Housing Ribs	5.0
Fin Ribs	7.8
Emissive Coating	3.8
Heat Shield & Reentry Insulators, PPO	40.8
Encapsulating Members, PPO	7.7
Fuel, PPO	26.7
	<hr/>
Total, PPO	165.2
RTG Height (inches)	45.5
Fin Tip-to-Tip, diagonally (inches)	63.7
Housing Cross Section (inches)	7.0 x 7.6

TABLE V-9
HEAT SOURCE SAFETY SUMMARY
REFERENCE DESIGN

Heat Source Volume (in. ³)	800
Fuel Form	PPO
Module Dimensions (in.)	4.5 x 4.5 x 1.796
Capsules/Module	9
Number of Modules	22
Module Weight (lb)	3.42
Total Inventory (curies)	1.46 x 10 ⁵

1. Reentry

Face	Ballistic Coefficient (lb/ft ²)	Heating Factor	Orbital Decay		Prompt	
			Maximum Capsule Temperature (°C)	Graphite Recession (in.)	Maximum Capsule Temperature (°C)	Graphite Recession (in.)
Broad	15	1.30	1310	0.031	1540	0.037
Edge	37	1.87	1580	0.086	1790	0.155

2. Impact

	Impact Temperature (°C)	Mean Impact Velocity (ft/sec)	Mean Failure Velocity (ft/sec)	Capsule Failure Probability	Accident Probability	Fractional Released Fines (curies/curie)	Σ/I (curie ⁻¹)
Orbital Decay Reentry	690	130	285	4 x 10 ⁻³	3 x 10 ⁻⁴	4.8 x 10 ⁻⁶	5.8 x 10 ⁻¹²
Prompt Reentry	740	130	275	5 x 10 ⁻³	3 x 10 ⁻⁵	4.7 x 10 ⁻⁶	0.7 x 10 ⁻¹²

$$\text{Safety Index} = \left[-\log_{10} \Sigma \right] = 6.0$$