A. 2430

VC33
(10)

OK
T

TRW-2617-3

TRANSIT RTG

Final Report. Program Report Number 4

THIS DOCUMENT CONFIRMED AS
UNCLASSIFIED
DIVISION OF CLASSIFICATION
BY JHKahn/asmh
DATE 4/17/73

December 15, 1971

TRW Systems Group
Redondo Beach, California

MASTER

DISTRIBUTION OF THIS DOCUMENT IS UNLIMITED

RS310

NOTICE

This report was prepared as an account of work sponsored by the United States Government. Neither the United States nor the United States Atomic Energy Commission, nor any of their employees, nor any of their contractors, subcontractors, or their employees, makes any warranty, express or implied, or assumes any legal liability or responsibility for the accuracy, completeness or usefulness of any information, apparatus, product or process disclosed, or represents that its use would not infringe privately owned rights.

This report has been reproduced directly from the best available copy.

Available from the National Technical Information Service, U. S. Department of Commerce, Springfield, Virginia 22151.

Price: Paper Copy \$3.00

Microfiche \$0.95.

TRANSIT RTG
FINAL REPORT
(Program Report Number 4)

prepared for
United States Atomic Energy Commission
Albuquerque Operations
Albuquerque, New Mexico

Contract AT(29-2)-2617

15 December 1971

NOTICE

This report was prepared as an account of work sponsored by the United States Government. Neither the United States nor the United States Atomic Energy Commission, nor any of their employees, nor any of their contractors, subcontractors, or their employees, makes any warranty, express or implied, or assumes any legal liability or responsibility for the accuracy, completeness or usefulness of any information, apparatus, product or process disclosed, or represents that its use would not infringe privately owned rights.

TRW
SYSTEMS GROUP

ONE SPACE PARK • REDONDO BEACH, CALIFORNIA

MASTER

DISTRIBUTION OF THIS DOCUMENT IS UNLIMITED

CONTENTS

	<u>Page</u>
1. INTRODUCTION	1-1
2. RTG SYSTEM	2-1
2.1 Design Requirements	2-1
2.2 Design Summary	2-3
2.3 RTG/Spacecraft Interfaces	2-12
2.4 Reliability	2-35
2.5 Transit RTG Qualification Test	2-55
2.6 Flight Model Acceptance Test	2-60
3. CONVERTER	3-1
3.1 Converter Configuration	3-1
3.2 Analyses	3-5
3.3 Testing	3-7
3.4 Development	3-17
4. HEAT SOURCE	4-1
4.1 Design Requirements	4-1
4.2 Baseline Design	4-1
4.3 Supporting Analyses	4-9
4.4 Testing	4-38
5. HEAT SOURCE SUPPORT STRUCTURE AND CAGE ASSEMBLY	5-1
5.1 HS/SSCA Functional Requirements	5-1
5.2 HS/SSCA Configuration	5-4
5.3 Design Analysis of HS/SSCA	5-12
5.4 Vibration Tests	
6. NUCLEAR SAFETY	6-1
6.1 Summary	6-1
6.2 Nuclear Safety Criteria	6-2
6.3 Heat Source Evaluation Relative to Safety Criteria	6-6
6.4 Risk Evaluation	6-15

CONTENTS (Continued)

	<u>Page</u>
7. GROUND SUPPORT EQUIPMENT	7-1
7.1 Heat Source Handling Tool	7-1
7.2 RTG Handling Tool	7-4
7.3 Hoist Sling	7-8
7.4 RTG Transport Dolly	7-10
7.5 Converter Shipping Container	7-10
7.6 Heat Source Shipping Cask	7-15
7.7 Electrical Test Equipment	7-20
7.8 Heat Source Capillary Tools	7-23
7.9 Special Hand Tools	7-26

1. INTRODUCTION

This is the final report required as part of the Transit RTG Phase II Contract AT(29-2)-2617. It summarizes the Transit RTG performance capabilities, operational and safety characteristics, major component configurations, supporting analyses and test data and provides basic applications information. The report is intended to give the reader an overview of the Transit RTG flight design and characteristics at the conclusion of the development and flight qualification program. Historical information is not included except where necessary to enhance understanding. However, a view of program progress and design evolution may be obtained by reading three previously issued Transit RTG Program Reports:

Transit RTG Program Report Number 1 - 11464-6010-T0-00
dated 15 January 1970

Transit RTG Program Report Number 2 - 11464-6010-T0-01
dated 30 June 1970

Transit RTG Program Report Number 3 - 11464-6010-T0-02
dated 15 January 1971

The following sections describe the overall RTG system and each major component including the Thermoelectric Converter, the Heat Source and the Supporting Structure. Nuclear Safety considerations are then summarized followed by a final section dealing with the Ground Support Equipment necessary to handle and check out the Flight RTG prior to launch.

2. RTG SYSTEM

2.1 DESIGN REQUIREMENTS

System design requirements set forth in TRW SY11-1, "Transit RTG System Specification - Performance and Design Requirements," are summarized below.

Performance. The direct current power from the RTG shall be at least 30 watts, and the load voltage at which this power is deliverable shall be at least 3.0 volts. It is a design goal that these output minimums be satisfied over an orbital operating lifetime of 5 years, with a 0.96 overall reliability objective for achieving this lifetime.

Figure 2-1. Allowable RTG Dynamic Envelope

Weight and Envelope. The total weight of the fueled RTG shall not exceed 30 pounds. The RTG shall fit within the dynamic envelope shown in Figure 2-1. This envelope is based on utilization of the -40 Scout heat shield, with RTG/spacecraft field joint at station + 1.95.

Storage and Air Exposure.

The RTG shall be storable for at least one year, or shall be configured to permit its disassembly into com-

ponents which are storable for that minimum period. The fully-assembled RTG shall withstand the effects of air exposure for 1000 cumulative hours. Of this total, 200 hours may be under the Scout heat shield in the launch-ready configuration.

Launch Environment. The RTG shall survive vibration, acceleration, shock, acoustic, and other environments at qualification levels defined in Table 2-1. Qualification levels are 1.5 times expected environments during launch and ascent. Launch is from WTR using the Scout vehicle with FW-4 fourth stage.

Table 2-1. Qualification Environments

	AXIS	FREQ-HZ	PEAK G*	DA-INCHES	DURATION
SINUSOIDAL VIBRATION	THRUST (Z)	10- 62 62- 95 95- 135 135- 240 240- 350 350-2000	10.0 33.0 4.5 7.0 4.5	0.05	2 OCTAVES/MIN ONE SWEEP
	LATERAL (X, Y)	15- 30 30- 45 45- 150 150- 300 300-2000	14.0 3.0 4.5 1.5	0.31	2 OCTAVES/MIN ONE SWEEP

*OVERRIDE TO LIMIT RESPONSE OF HEAT SOURCE TO 56G OR CONVERTER TO 35G

	AXIS	FREQ-HZ	RMS G	G ² /HZ	DURATION
RANDOM VIBRATION	X,Y,Z	20-2000	9.4	0.045	4 MIN/AXIS

	OCTAVE-HZ	PRESSURE-DB*	DURATION
ACOUSTICAL NOISE	37- 75	121	2 MIN
	75- 150	118	
	150- 300	116	
	300- 600	113	
	600-1200	110	
	1200-2400	107	
	2400-4800	103	
	4800-9600	98	
	BROADBAND	124	

*REF
0.0002 DYNE/CM²

	AXIS	G	DURATION	REMARKS
ACCELERATION	Z	24	3 MIN	
	X,Y	3	1 MIN	
SHOCK	Z	30	10 ± 3 MS	3 HALF-SINE PULSES

SPIN	240 RPM FOR 20 MIN, DESPIN IN 1 SECOND
DEPRESSURIZATION	SEA LEVEL TO 50 TORR FOLLOWING PRESSURE - TIME PROFILE UNDER SCOUT HEAT SHIELD ON TRANSIT FLIGHT

2.2 DESIGN SUMMARY

The assembled RTG is shown in Figure 2-2. It is comprised of two separately-stored subassemblies: (1) the heat source mounted in the heat source cage, and (2) the converter with heat source support structure installed. The top cover of the converter is removable, providing access to the interior for installation and removal of the caged heat source.

Figure 2-2. RTG Assembly

2.2.1 Heat Source

The heat source is shown in Figure 2-3. It weighs a maximum of 14 pounds and generates 850 watts when initially fueled with PuO_2 in cermet (PMC) form. The fuel is encapsulated in a T-111 strength member with inner Ta-10W liner and outer Pt-20Rh clad. The capsule is sheathed in a composite graphite reentry heat shield, and the total assembly is sealed in an outer Inconel can to prevent prelaunch oxidation of the heat shield. A ceramic pressure relief device (PRD) in the liner wall releases helium while containing fuel particles. The released helium passes through two 4-mil diameter holes in one end of the strength member and another hole in the opposite end of the clad, accumulating in the heat shield voids and in the gap between heat shield and outer can. The can vents through a 15-foot, 4-mil ID coiled capillary tube which is opened shortly before launch. The capillary bleeds helium off at a slow enough rate to preclude admission of air during the prelaunch period and to maintain sufficient helium back-pressure for good internal heat transfer in orbit. The capillary is resealed if a prolonged launch hold occurs.

Figure 2-3. Heat Source

2.2.2 Converter

The converter is a 12-sided prism with one 14 3 x 5 7-inch Isotec panel on each face and inert top and bottom covers. Each Isotec panel has 36 PbTe thermocouples in 9 series-connected sets of 4 parallel couples. The 12 panels are in series. Thus, in the generator, there are 108 quads in series and a total of 432 thermocouples. Open failure of all four thermocouples in one quad is necessary to totally fail the generator

Viewed externally, as in Figure 2-4, each panel presents an array of second-surface mirrors. The mirrors, with emissivity of 0.74 and solar absorptivity of 0.05, overlay silver radiator strips backed by a phenolic honeycomb structure and aluminum multifoil superinsulation. Thermoelements cantilevered inward from the radiators terminate in silver collector plates facing the heat source. Molybdenum-opacified quartz washers stacked around each thermoelement prevent electrical shorts to the foils and inhibit PbTe sublimation.

Figure 2-4. Converter

The panels are supported by MgTh corner posts, with top and bottom shear rings grooved to accept the panel edges. The shear rings are screwed to the panels and the panels to the corner posts. Top and bottom covers, with lightweight honeycomb structures and superinsulation, are attached to the posts by means of studs at the top and threaded inserts at the bottom. The bottom cover is reinforced with an inner ring which protrudes from the underside and is flanged to fit a mating ring on the spacecraft. Heat source loads are transmitted to the ring via mounting studs attached to it and projecting upward through the insulation blanket. The panels are interconnected by silver straps joined by means of a silver-tin solder and mechanically held by two bolts.

The miniature Isotec thermoelements and thin-foil radiators with honeycomb structure achieve both lightweight and a large radiation surface. This enables efficient conversion at low enough temperature ($T_h \approx 400^\circ\text{C}$) to permit unsealed operation without sublimation of the lead telluride. Since the converter is not sensitive to the shape of the heat source and is physically decoupled from it, the converter has readily accommodated heat source configurational changes and retains that flexibility for future improvements.

2.2.3 Heat Source Cage and Support Structure

The heat source is held in a titanium and Inconel cage and the caged heat source is held by a titanium support structure (Figure 2-5). The heat source and cage are factory-preassembled, as are the heat source support structure and converter.

The cage consists of two Inconel 718 end caps bearing upon each end of the heat source and joined by a Ti-6Al-4V frame structure. The end caps are bolted to the three longitudinal bars of the frame, and the bars are preloaded to insure positive clamping of the heat source during launch. This prevents movement during boost vibration of a sliding plug in the graphite heat shield, the sliding plug being necessary to avoid overstressing the graphite during post-reentry cooldown shrinkage.

The heat source support structure is a tripod comprised of six Ti-6Al-4V tubular members tied together in pairs terminating in three

HEAT SOURCE SUPPORT STRUCTURE AND CAGE ASSEMBLY

SUPPORT STRUCTURE

CAGE

**SUPPORT STRUCTURE
AND CAGE ASSEMBLY**

**CAGED HEAT SOURCE
ON SUPPORT STRUCTURE**

Figure 2-5. Heat Source Cage and Support Structure

forward and three aft points. The forward points attach to fittings with threaded studs and alignment pins for mating with the cage truss. The aft points attach to mounting studs which are integral parts of the converter aft cover. The mounting studs are designed to minimize thermal conduction losses, and thermal reflectors covering the support structure feet reduce radiative losses.

2.2.4 Instrumentation

Individual panel voltages as well as total RTG output voltage are measured via leads from the top and bottom of each panel. These leads are routed to a header on the RTG base where they connect to the voltage instrumentation cable. Thermocouples are emplaced at the hot and cold caps of single, centrally-located thermoelements on every other panel. Leads from the hot caps are brought to one header and leads from the cold caps to another. These two headers then connect to a single temperature instrumentation cable. The reference junction for these measurements is provided in the spacecraft. RTG output current is measured in the spacecraft. RTG internal resistance can be determined from the change in output current and voltage induced by momentarily switching an additional load in parallel with normal spacecraft loads.

2.2.5 Handling Provisions

The converter is shipped and stored in a pressurized hemispherical container which mates to a transport dolly as shown in Figure 2-6(A). The caged heat source is shipped and stored in a Universal Shipping Container (B). A special heat source handling tool (C) is used to transfer it to the generator. When installed in the generator, it should be oriented so that index marks on the cage and support structure are aligned. A generator handling tool (D) is used to move the assembled RTG. This tool consists of three latching segments with upper and lower handles. The segments fasten to the bottom and brace against the top of three corner posts spaced 120 degrees apart.

(A) Converter Container
On Dolly

(B) Heat Source Container

(C) Heat Source
Handling Tool

(D) Generator
Handling Tool

Figure 2-6. Handling Equipment

2.2.6 Weight Summary

Component weights for the flight model are summarized in Table 2-2. Since the actual flight heat source will not be fabricated until March 1972, maximum weights are given at various levels of assembly as specified on the flight heat source production drawings. The actual weight of the complete fueled qualification heat source was 13.9 pounds. Converter, heat source support structure, cage and instrumentation weights represent actual measured values.

Table 2-2. RTG Weight Summary

<u>Heat Source</u>		<u>14.00</u>
Fuel	5.91	
Fueled capsule	9.29	
Fueled capsule with graphite heat shield	13.44	
<u>Converter</u>		<u>13.18</u>
Thermoelectric panels (12)	6.48	
Top support structure and insulation	1.89	
Bottom support assembly and insulation	3.52	
Structural frame	1.02	
<u>Heat Source Support Structure and Cage</u>		<u>2.40</u>
Cage end caps (2)	0.98	
Cage tubes and fittings	0.62	
Support structure	0.62	
Miscellaneous (bolts, washers, etc.)	0.18	
<u>Instrumentation</u>		<u>0.24</u>
<u>Total</u>		<u>29.82</u>

2.2.7 Performance

The predicted BOL performance of the flight unit (Figure 2-7) is based on acceptance tests of the F-1 converter, and assumes an 850-watt heat source. The best estimate for BOL power output in space is 36.8 ± 0.3 watts at a load ratio of 1.2. The expected maximum hot-cap temperature is $400 \pm 2^{\circ}\text{C}$.

Figure 2-7. RTG BOL Performance in Space

Figure 2-8. RTG Power Output Versus Time

The nominal predicted Transit flight generator performance vs time (Figure 2-8) is derived from thermoelectric module tests instrumented to monitor individual element Seebeck coefficient and resistance. A constant load-matching factor, linear changes in n-element properties with time, and log-log changes in p-element properties are assumed. Additional discussion of the performance as a function of time is given in the Section 2.4, Reliability.

2.3 RTG/SPACECRAFT INTERFACES

2.3.1 Mechanical Interface

The Transit RTG is mounted to the spacecraft by means of two flanged rings, one on the spacecraft and one on the RTG, secured by a clamp strap. The clamp itself is a spacecraft component, and the flanged ring on the spacecraft is part of its central structural cylinder. The flanged ring on the RTG side is an integral part of the RTG base-plate. Details of the RTG flanged ring are shown in Figure 2-9.

RTG azimuth is fixed by an indexing pin and hole on the flanged rings of the spacecraft and RTG as shown in Figure 2-10. The index pin is on the spacecraft ring. It is for alignment only and does not carry shear loads. Rotation of the RTG with respect to the spacecraft during launch vibration and fourth stage spin-despin is prevented by the clamp strap.

During ascent the RTG must be vented to prevent a large pressure differential across the thermoelectric panels. There is considerable venting through RTG mechanical joints. In addition, venting occurs through the single instrumentation feedthrough hole outboard of the RTG flanged ring and through the three inboard tubular studs to which the heat source support structure attaches. Simulated ascent depressurization tests have shown the adequacy of these features, when the equivalent of one 0.25-inch diameter vent hole is provided in the spacecraft structural cylinder on which the RTG is mounted and into which the tubular studs vent.

Clearances between the RTG and -40 Scout heat shield are shown in Figure 2-11, based on an RTG/spacecraft mating plane location at vehicle station 1.95. The mating plane is defined as the interface between the flanges of the two mating rings.

2.3.2 RTG/Spacecraft Thermal Interface

RTG temperature and RTG heat input to the spacecraft have been calculated for both ground and in-orbit environments. Thermal properties assumed for this purpose are shown in Figure 2-12. The spacecraft was taken to be an octagonal, isothermal body. The indicated Scout heat shield thermal conductance was obtained from LTV.

Figure 2-9. RTG Interface Data

BLANK

Figure 2-10. RTG Interface Data

BLANK

Figure 2-11. RTG/Scout Heat Shield Envelope

THERMAL CONSTANTS:

Solar Flux	442 Btu/hr-ft ²
Earth Flux	68 Btu/hr-ft ²
RTG Fuel Inventory	827 watts
Effective Thermal Conductance Across Scout Heat Shield	0.67 Btu/hr-ft ² -°F

Figure 2-12. RTG Spacecraft Thermal Interface

Where solar radiation is indicated, the sun vector was taken normal to the RTG/spacecraft longitudinal axis. Where ground air conditioning is indicated, the cooling air was assumed to be ducted from an entrance in the upper cylindrical section of the Scout heat shield to the nose section above the RTG. The air was then assumed to flow downward past the RTG and to exit through vents in the lower cylindrical section. The air inlet temperature and flow rate were assumed to be 70°F and 800 cfm.

2.3.2.1 Prelaunch Air Environment

Operational considerations require that the fully-assembled RTG be capable of standing 800 hours in air without the vehicle heat shield installed plus an additional 200 hours with the heat shield in place. The maximum allowable RTG hot cap temperature is 200°C during the 800-hour period and 250°C during the 200-hour period. The allowable cold cap temperature is limited to 180°C during the entire 1000-hour period. Estimated hot cap and cold cap temperatures under various prelaunch conditions are shown in Table 2-3.

Table 2-3. Prelaunch Hot Cap and Cold Cap Temperatures (°C)

	Without Scout Heat Shield		With Scout Heat Shield		
	Vertical	Horizontal	Without Air Conditioning		With Air Conditioning
			Vertical	Horizontal	Vertical
Hot Cap - Average	160	171	234	245	130
Hot Cap - Maximum	166	195	240	269	136
Cold Cap - Average	103	105	181	186	78
Cold Cap - Maximum	110	115	187	196	85
Heat Source Surface	400	406	428	434	389
Assumed Ambient Air	32	32	26	26	26

Spacecraft Emissivity ----- .05

RTG Thermal Power ----- 850 Watts

Spacecraft Temperature ----- 38°C

RTG Load ----- Open Circuit

Prelaunch heat inputs from the RTG to the spacecraft are shown in Figure 2-13 without the Scout heat shield, and in Figure 2-14 with an air-conditioned Scout heat shield. Radiative, conductive, and total heat inputs are given as a function of spacecraft temperature. The radiative component is seen to be small.

Condition: Without Scout Heat Shield and Without Air Conditioning

Figure 2-13. On-Stand Heat Transfer Modes to Spacecraft Versus Spacecraft Temperature

Figure 2-14. On-Stand Heat Transfer Modes to Spacecraft Versus Spacecraft Temperature

External RTG temperatures were calculated for a fixed 100°F spacecraft. The external RTG temperature distribution is shown in Figure 2-15 without the Scout heat shield, and in Figure 2-16 with air-conditioned and non-air-conditioned Scout heat shields.

Figure 2-15. RTG On-Stand Temperatures (°F) Without Scout Heat Shield

Figure 2-16. RTG On-Stand Temperatures (°F) Under Scout Heat Shield

2.3.2.2 Orbital Environment

For long life and minimum degradation, thermoelement hot-cap temperatures should not significantly exceed 400°C during vacuum operation. With the nominal 850-watt heat source, flight-unit tests indicate the maximum hot cap temperature to be 392°C in a radiation-less environment. In the presence of earth and solar radiation, the maximum hot cap temperature is 405°C on the panel in direct sunlight, and the maximum hot cap temperature averaged over all the panels is 402°C .

The in-orbit heat flow from RTG to spacecraft as a function of spacecraft temperature is given in Figures 2-17 and 2-18. External RTG temperatures with and without incident radiation are shown in Figure 2-19 for a spacecraft temperature of 100°F , and the thermal balance is given in Table 2-4.

Figure 2-17. In-Orbit Heat Transfer Modes to Spacecraft Versus Spacecraft Temperature

Condition: With Earth and Solar Radiation

Figure 2-18. In-Orbit Heat Transfer Modes to Spacecraft Versus Spacecraft Temperature

Figure 2-19. RTG In-Orbit Temperatures, °F

Table 2-4. RTG Heat Balance

	Watts
Thermoelectric Panels	705
Top Cover	55
Bottom Cover	62
Heat Source Support Studs	14
Extraneous Losses	4
	<hr/> 850

2.3.3 Electrical Power Interface

The RTG furnishes unregulated low-voltage DC power to the spacecraft. Power conditioning is a spacecraft function. In a vacuum environment, the load connected across the RTG output terminals must draw sufficient current to assure adequate Peltier cooling of the thermoelectric hot junctions. The presence of a suitable load is assured during space operation by a shunt regulator in the spacecraft power conditioning equipment. In air, RTG temperatures are not greatly affected by the electrical load.

The presence of air in the superinsulation reduces its thermal impedance. As a result, generator efficiency is very low and usable power is not available from the RTG during prelaunch and ascent phases. However, on the ground, output voltage and current serve as rough indicators of the RTG's condition.

The RTG power output tap provides two 8-32 studs to which spacecraft power leads attach. The polarity of each stud is marked as shown in Figure 2-10. The studs offer a free length of 3/8-inch to mount the lugs, an AN960C8L washer, and an MS21043-08 nut. The inner diameter of the lugs on the spacecraft power leads should clear the 8-32 studs (0.164-inch). The outer diameter should be no greater than 0.375-inch, since a larger diameter does not increase mating area. The current path is directly from the RTG silver terminal strap to the lug.

Immediately below the output power tap is a tiedown point for the power cable. The tiedown is an extension of the bolt that attaches the converter bottom shear ring and bottom cover to the corner post. The power cable can be tied to the bolt extension with the cable tie-cord fitted into the groove around the bolt extension. Details are shown in Figure 2-9.

2.3.4 Instrumentation Interface

Flight instrumentation provides 12 panel voltage measurements and 12 temperature measurements, with special provisions for measuring RTG internal resistance. An instrumentation schematic is shown in Figure 2-20. RTG output current and regulator shunt current are not shown since they are measured in the spacecraft.

The flight instrumentation sensor signals are routed to the spacecraft by means of two cables with connectors, one for panel voltages and one for temperature measurements. Cable and connector locations on the RTG can be seen in Figures 2-9 and 2-10. Temperature and voltage instrumentation parameters are given in Figure 2-21.

2.3.4.1 Temperature Measurement

The temperature sensors are Type J iron-constantan thermocouples. One thermocouple is located on the hot cap and one on the cold cap of a single thermoelectric element on every other panel (six elements total). The selected element is in the fifth row of couples on a panel and is, alternately, a "p" or "n" element.

The sensor leads are terminated at two headers bonded onto the converter bottom support structure. The headers (Figure 2-22) are Kovar with an electroless nickel finish. Electrical isolation between the header terminals and base is provided by glass beads. The headers facilitate transition from the 2-mil wire used for the sensor to No. 28 AWG Type J wire used in the temperature instrumentation cable. Strain relief for the welds at the header terminals is provided by a cable clamp. The Type J leads to the connector on the RTG side of the interface must be continued from the connector to the reference junction on the spacecraft side. The base of the clamp is bonded to the converter bottom support structure.

The temperature instrumentation cable (Figure 2-23) runs from the headers to a UR46-14-44S connector. At the connector each thermocouple lead terminates in a metal film resistor rated at 422Ω , ± 1 percent, 0.05 watt. The resistors are encapsulated in a semi-rigid epoxy compound within the connector backshell. The resistors in each temperature instrumentation lead isolate the RTG from spacecraft circuitry. The source impedance (at the connector) for each sensor circuit is made up primarily of the two resistors (844Ω , total).

6. RESISTORS R14 THRU R41 ARE 422 Ω .
5. RESISTORS R1 THRU R13 ARE 500 Ω .
4. UNUSED CONNECTOR CONTACTS NOT SHOWN FOR CLARITY.
3. THIS DWG NOT TO BE USED FOR FABRICATION OR PROCUREMENT.
2. SEE TRW DWG 407028 FOR ELECTRICAL INTERFACE CONTROL.
1. FOR TOP ASSY DWG, SEE 405480 RTG ASSY & 405520 CONVERTER.

NOTES: UNLESS OTHERWISE SPECIFIED

Figure 2-20. Temperature and Voltage Instrumentation Schematic

BLANK

POWER OUTPUT SCHEMATIC WITH PANEL IDENTIFICATION

5 IN ADDITION TO THESE TEMPERATURE AND VOLTAGE MEASUREMENTS, THE SPACECRAFT CONTRACTOR WILL MAKE PROVISIONS FOR MEASURING TOTAL RTG VOLTAGE AND CURRENT, AND WILL IMPLEMENT A MEANS OF VARYING THE RTG LOAD TO PERMIT DETERMINATION OF RTG INTERNAL IMPEDANCE

4 DURING VACUUM OPERATION THE RTG MUST BE LOADED SUCH THAT THE VOLTAGE AT THE POWER TAP DOES NOT EXCEED 6.0 VOLTS FOR MORE THAN 30 SECONDS.

3 THIS DRAWING ESTABLISHES INTERFACE REQUIREMENTS WHICH AFFECT THE RTG ASSEMBLY DRAWING 405480.

2 435 DIMENSION DOES NOT APPLY IN AREA OF DOUBLER.

1 VENTS TRANSIT RTG DURING LAUNCH.

NOTES: UNLESS OTHERWISE SPECIFIED

NOTE FOR OFFICIAL MECHANICAL AND ELECTRICAL INTERFACE DATA SEE TAW DRAWING 40508.

Figure 2-21. RTG Instrumentation Data

BLANK

Figure 2-22. Kovar Header

Figure 2-23. Instrumentation Cable

2.3.4.2 Panel Voltage Measurement

The voltage across each of the 12 RTG thermoelectric panels is measured by providing leads from the terminal straps on each panel to a header mounted on the converter bottom support structure and then from the header to a connector which mates to a receptacle on the spacecraft. With the exception of the panel terminal straps on either side of the power tap, the terminal straps are overlapped and connected alternately at the top and bottom of the panels.

The isolation resistors for the panel voltage instrumentation leads are mounted at the panel terminal straps where the straps of adjacent panels overlap. The terminal straps at either side of the output power tap do not overlap but are incorporated into the tap structure. For these two straps, the resistors are mounted at the strap on either side of the corner post. The resistors are 500Ω , ± 1 percent, 1-watt wire wound units usable at the temperature encountered at the mounting locations ($\leq 180^{\circ}\text{C}$). One resistor lead is soldered directly to the terminal strap, the other is resistance-welded to the panel voltage instrumentation lead.

The constantan leads are terminated at a header identical to those used for the temperature instrumentation. The leads are resistance-welded to the header terminals. The signal leads are continued to the spacecraft receptacle by the panel voltage instrumentation cable. This cable is comprised of a UR46-10-19S connector, backshell, and 13 Kapton-insulated, No. 28 AWG constantan wires. The constantan wires are crimped to the contacts which are then inserted into the connector, the backshell is attached to the connector, and the volume within is encapsulated with a semi-rigid epoxy compound. Strain relief on the header terminal welds is provided by a cable clamp similar to that used for the temperature instrumentation cable.

The resistors in each panel voltage instrumentation lead isolate the signal source and RTG ground and spacecraft circuitry. The source impedance for each pair of wires is made up of the two isolation resistors. Other contributors to the impedance (panel internal resistance and lead resistance) sum to less than the tolerance on the resistors.

2.3.4.3 Resistance Measurement

The RTG parameter most useful for predicting performance is the internal resistance of the converter. Changes in the internal resistance can be used to assess thermoelectric material degradation and open thermoelectric elements, particularly when analyzed in conjunction with measured panel voltages.

On the ground, internal resistance is easily measured by passing AC current through the RTG and measuring voltage drop. In space, resistance can be measured by switching the RTG load enough to produce a distinct voltage change. A change in load voltage of 0.3 to 0.5 volt should be sufficient. By differencing the load voltage and current of the two operating points, the internal resistance can be calculated.

Figure 2-24. Typical Internal Resistance Error Vs Time Since Load Shift

The change in voltage and current must be measured before Peltier effects change junction temperatures appreciably. Figure 2-24 shows the expected error as a function of time after load switching. The measurement of current and voltage should be synchronized so that the delay after load switching is always the same. In this way, while a constant bias may occur in the absolute resistance, the change in resistance between measurements will be unaffected.

2.3.5 Nuclear Radiation Interface

Neutron and gamma radiation dose rates measured as a function of axial and radial distance from the heat source are given in Figure 2-25. The 850-watt Transit qualification heat source (TF-01) was used for these measurements. The dosages must be regarded only as "typical". Dosages from the flight heat source, when fabricated, can be expected

Figure 2-25. RTG Radiation Levels

to deviate somewhat due to changes in fabrication processes. Deviations will also occur with thermal power, fuel age, plutonium species, oxygen content, and impurities.

The factor for converting neutron fluence to dose equivalent is 2.86×10^{-5} mrem per n/cm². The initial neutron yield from TF-01 was 5.44×10^7 n/sec ± 5 percent. This corresponds to a specific yield of 3.63×10^4 n/sec/g Pu238. The indicated yield is from all sources, including spontaneous fission, (α -n) reactions in oxygen and impurities, and multiplication from neutron-induced fissions in the total assembly.

The following features should be noted from the dose rate curves:

- The neutron dose is about 50 times greater than the gamma dose and, therefore, may be taken as the total dose.
- The neutron dose is reduced by less than 20 percent when the heat source is within its shipping container.
- The radial dose is almost twice the axial dose at the same distance from the heat source center.
- The total radial dose is approximately 0.9 rem/hour at 1-foot and decreases as the square of the distance.

In evaluating the dose rates given, it is useful to note that the maximum permissible weekly dose for radiation workers is 100 mrem per week. Thus, without exceeding permissible levels, a worker can remain 4 hours per week at 6 feet from an unshielded source. Of course, the work period can be extended by proper placement of neutron shielding.

2.3.6 Procedural Interface

Detailed RTG handling and checkout procedures are given in the Transit RTG Operations Manual, and launch pad monitoring of RTG functions is defined in the RTG Diagnostic Information Plan. Three RTG characteristics affecting ground operations should be noted:

- 1) Usable power to the spacecraft will not be available except when operating in a vacuum, since the RTG insulation does not function effectively in air.
- 2) Cumulative in-air exposure of the fueled RTG is restricted to 1000 hours. Of this total, 800 hours are for unshrouded exposure to ambient air, and 200 hours are under the Scout vehicle heat shield.
- 3) The heat source capillary tube must be opened prior to launch. Extensive launch delays will require reclosing and reopening the capillary, since the total allowable time the capillary may remain open in air is 250 hours.

RTG handling and checkout sequences for spacecraft acceptance tests at APL and for prelaunch spacecraft operations at WTR are shown in Figures 2-26 and 2-27. The APL acceptance tests are performed on the all-up flight configuration, with fueled RTG installed. At WTR, when the spacecraft system is spin-balanced, the nuclear heat source is temporarily replaced by an inert dummy simulating its mass and center of gravity.

2.4 RELIABILITY

2.4.1 Reliability Model

The Transit RTG is to provide at least 30 watts of power to the spacecraft for five years with a reliability objective of 0.96. Degradation of output power to levels below 30 watts as well as total failure are included in the reliability model (Figure 2-28). The failure and degradation modes considered are:

- 1) Opening of four parallel couples (total failure)
- 2) Openings and shorts in wiring, connectors, and instrumentation
- 3) Structural deformation or collapse
- 4) Gradual thermoelement degradation

Figure 2-26. RTG Handling Sequence at APL

Figure 2-27. RTG Handling Sequence at WTR

Figure 2-28. Reliability Math Model Block Diagram

2.4.2 Converter Open-Circuit (Failure Model No. 1)

Catastrophic (total) failure probability is diminished by incorporating quadruple series-parallel element redundancy. Such failures can occur due to open elements only if all four elements open in any one of the 108 quads. The probability that this does not happen is given by the first term of the math model in Figure 2-28, where t is mission time in hours and λ is failure rate in reciprocal hours.

The failure rate for a given confidence level equals the expectancy (Table 2-5) divided by the number of couple hours. For example, if one open-couple failure occurs in one million couple-hours of testing, the indicated failure rate is 1.69×10^{-6} at 50 percent confidence and 3.89×10^{-6} at 90 percent confidence. Failure rates at 50 and 90 percent confidence limits were substituted into the model for Failure Model No. 1. Reliabilities obtained are shown in Figure 2-29.

Based on GE&ES analysis, a risk-allocated failure rate of 1 percent is used to account for p-element and n-element failures during processing, manufacturing, and launch. This gives 9 open elements, since there are a total of 864. The n-element reliability risk excludes element failures due to porosity accumulation on the assumption by GE&ES that design and process improvements now incorporated will exclude this

Table 2-5. Expectancy Vs Confidence

No of Failures	Confidence Level												
	50%	55%	60%	65%	70%	75%	80%	85%	90%	95%	97-1/2%	99%	99-1/2%
0	0 693	0 799	0 917	1 051	1 204	1 387	1 619	1 897	2 303	2 995	3 689	4 605	5 298
1	1 689	1 844	2 023	2 219	2 440	2 693	2 994	3 373	3 890	4 745	5 572	6 638	7 431
2	2 674	2 883	3 105	3 348	3 616	3 921	4 279	4 724	5 323	6 296	7 225	8 406	9 275
3	3 672	3 916	4 176	4 455	4 763	5 110	5 515	6 014	6 681	7 755	8 767	10 045	10 975
4	4 671	4 946	5 237	5 549	5 890	6 274	6 721	7 267	7 993	9 153	10 241	11 604	12 594
5	5 670	5 973	6 292	6 633	7 005	7 422	7 906	8 495	9 274	10 513	11 668	13 108	14 150
6	6 669	6 998	7 343	7 711	8 111	8 558	9 075	9 703	10 532	11 842	13 059	14 570	15 659
7	7 669	8 021	8 390	8 782	9 209	9 684	10 232	10 897	11 771	13 148	14 422	16 000	17 133
8	8 669	9 043	9 434	9 850	10,300	10 802	11 380	12 078	12 999	14 431	15 763	17 402	18 578
9	9 668	10,064	10 476	10 913	11 387	11 914	12 519	13 249	14 206	15 705	17 085	18 783	19 998
10	10 669	11 083	11 515	11 974	12 470	13 020	13 651	14 411	15 407	16 962	18 391	20 144	21 398
11	11 669	12 102	12 533	13 031	13 548	14 121	14 777	15 602	16 598	18 208	19 682	21 490	22 779
12	12 668	13 120	13 589	14 087	14 623	15 217	15 898	16 742	17 782	19 443	20 962	22 021	24 145
13	13.668	14 137	14 624	15 149	15 696	16 310	17 014	17 873	18 958	20 669	22 231	24 139	25 497
14	14 668	15 160	15 669	16 202	16 765	17 400	18 125	18 996	20 128	21 887	23 490	25 446	26 836
15	15 668	16 176	16 700	17 253	17 843	18 510	19 254	20 143	21 324	23 114	24 776	26 783	28 198

phenomenon. For the 9 open-circuit thermocouples being considered, converter reliabilities are 0.99998 and 0.9999 at the 50 and 90 percent confidence levels, respectively.

Figure 2-29. Probability of No Open-Circuit Converter Vs Number of Open-Circuit Elements

2.4.3 Internal Short Circuits (Failure Mode No. 2)

Analyses were performed to determine the magnitude of power losses due to shorting modes. The effects of internal shorts were studied with the RTG and spacecraft frames electrically isolated as well as grounded.

Grounded Frame

With the converter frame grounded to the spacecraft frame through negligible resistance, loss of power can occur for either of the following shorting modes:

- Two or more element-to-foil shorts bridging one or more couples in the same panel
- One or more element-to-foil-to-frame shorts

From limited data, the indicated probability that a given foil will short to an element is 0.0176, and the probability that a given foil will short to the frame is also 0.0176. In order to lose power, however, two element-to-foil shorts must occur on the same foil (or on electrically-connected foils) or a foil shorted to a frame must also be shorted to an element. It is conservatively assumed that when more than one short occurs in a panel, it involves the same foil.

In fixing the power loss, the number of couples bridged by the combination of shorts and the resistance in the shorted path must be introduced. The element-to-foil resistance is typically 3 ohms, and the foil-to-frame resistance 2 ohms. This gives shorting resistances of 6 ohms and 5 ohms, respectively for element-to-foil-to-element and element-to-foil-to-frame paths.

The percent of total power remaining in the load as a function of bridge resistance and number of thermocouples bridged to ground, is plotted in Figure 2-30 for a full generator. This figure can also be used to obtain percent power remaining in a single panel with either element-to-foil-to-element or element-to-foil-to-frame shorts.

Figure 2-30. Effect of Short Circuits Occurring Between RTG Thermocouples and Frame

Proceeding on the basis of the foregoing assumptions and data, the probability of having two or more element-to-foil shorts in the same panel is determined by the binomial distribution to be:

$$Q_{2fo} = 1 - \left[\binom{17}{0} (1-0.0176)^{17} + \binom{17}{1} (0.0176) (1-0.0176)^{16} \right] = 0.04$$

Similarly, the probability of having one or more combined element-to-foil-to-frame shorts in a panel is 0.0625. This is the product of individual probabilities for element-to-foil-to-frame shorts in the same panel (0.25) and one or more foil-to-frame shorts in the same panel (also 0.25).

With the probability of two or more element-to-foil shorts occurring in one panel equal to 0.04, the probabilities of such failures occurring in a number of panels in a 12-panel generator are:

$$\text{For exactly 0 panels: } Q_{2fo}(0) = \binom{12}{0} (1-0.04)^{12} - (0.96)^{12} = 0.630$$

$$\text{For exactly 1 panel: } Q_{2fo}(1) = \binom{12}{1} (0.04) (1-0.04)^{11} = 0.290$$

$$\text{For exactly 2 panels: } Q_{2fo}(2) = \binom{12}{2} (0.04)^2 (1-0.04)^{10} = 0.0670$$

$$\text{For exactly 3 panels: } Q_{2fo}(3) = \binom{12}{3} (0.04)^3 (1-0.04)^9 = 0.0115$$

$$\text{For exactly 4 panels: } Q_{2fo}(4) = \binom{12}{4} (0.04)^4 (1-0.04)^8 = 0.0014$$

$$\text{For exactly 5 panels: } Q_{2fo}(5) = \binom{12}{5} (0.04)^5 (1-0.04)^7 = 0.00012$$

The average expected power loss due to element-to-foil-to-element shorts in 12-panel generators is:

$$\begin{aligned} \% \text{ Loss} &= 0.63(0) + 0.29(0.03) + 2 \times 0.067(0.03) + 3 \times 0.0115(0.03) + \dots \\ &= 0.03(0.5633) = 0.0169 = 1.69 \end{aligned}$$

With the probability of one or more combination element-to-foil-to-frame shorts occurring in one panel equal to 0.0625, the probabilities of such occurrences in a number of panels in a 12-panel generator are:

$$\text{For exactly 0 panels: } Q_{\text{fo-fr}}(0) = \binom{12}{0} (1-0.0625)^{12} = 0.49$$

$$\text{For exactly 1 panel: } Q_{\text{fo-fr}}(1) = \binom{12}{1} (0.0625) (1-0.0625)^{11} = 0.34$$

$$\text{For exactly 2 panels: } Q_{\text{fo-fr}}(2) = \binom{12}{2} (0.0625)^2 (1-0.0625)^{10} = 0.13$$

$$\text{For exactly 3 panels: } Q_{\text{fo-fr}}(3) = \binom{12}{3} (0.0625)^3 (1-0.0625)^9 = 0.033$$

$$\text{For exactly 4 panels: } Q_{\text{fo-fr}}(4) = \binom{12}{4} (0.0625)^4 (1-0.0625)^8 = 0.006$$

Again, the average number of couples bridged for one combination of element-to-foil-to-frame shorts may be assumed to be one-half the total number of couples in a converter. It also may be assumed the average loss of two combination shorts is two-thirds of the converter couples and the average loss for three such shorts is three-fourths of the couples, etc. The total shorting resistance for this mode, as previously stated, is 5 ohms.

From Figure 2-30 the percent losses for a 5-ohm shorting resistance bridging various fractions of the total number of couples are found to be as tabulated below:

<u>Fraction of Total Number of Couples Short-Circuited</u>	<u>Percent Power Loss</u>
1/2	3.8
2/3	6.3
3/4	8.0
4/5	9.0

The average expected power loss in a generator due to element-to-foil-to-frame shorts is:

$$\begin{aligned} \% \text{ Loss} &= 0.49(0) + 0.34(3.8) + 0.13(6.3) + 0.033(8) + 0.006(9) \\ &= 0 + 1.29 + 0.82 + 0.26 + 0.054 \\ &= 2.42 \end{aligned}$$

Isolated Frame

With the Transit RTG frame electrically isolated from the spacecraft ground there is no change in power losses due to element-to-foil-to-

element shorts in the same panel. There is an effect, however, on losses due to element-to-foil-to-frame shorts. With the RTG electrically isolated, two rather than one element-to-foil-to-frame shorts are required before a power loss occurs. The average expected power loss for this shorting mode becomes 1.13 percent.

Probable power losses for the two shorting modes and for isolated and grounded frames are summarized in Table 2-6. Shorting mode losses for the isolated frame condition are small compared to degradation power losses discussed below, and they are of less concern with the incorporation of mica post insulation foil restraints in the flight configuration. The power loss due to shorts for the isolated frame condition is included in the power degradation analysis (Failure Mode No. 4).

Table 2-6. Power Loss Vs Shorting Mode

Shorting Mode	<u>Percent Power Loss</u>	
	Frame Grounded	Frame Isolated
Element-to-foil-to-element	1.69	1.69
Element-to-foil-to-frame	2.42	1.13

2.4.4 Instrumentation Short Circuits (Failure Mode No. 2 Continued)

The RTG instrumentation system is comprised of 12 thermocouple channels and 13 voltage taps. The thermocouples are connected to the hot and cold caps of one element on every other panel of the 12 converter panels. The voltage taps are connected to the 11 electrical bonds between panels plus the two output terminals.

Each thermocouple channel contains two isolation resistors, two terminal board pins, iron and constantan thermocouple wires, and two active pins in a connector. There are seven welded and two crimped connections within each thermocouple channel.

Similarly, each voltage tap circuit has an isolation resistor, a terminal board pin for junction of light and heavier gauge constantan lead wires, and a single connector pin. Three welded, one silver-soldered, and one crimped connection are involved in each tap.

Impact of Instrumentation on RTG Reliability

Possible failure modes in which instrumentation failures can lead to loss of output power are as follows:

- a) One T/C wire insulation abraded through touching electrically active elements.
- b) Two T/C wires with insulation abraded through touching the frame or touching each other.
- c) One T/C wire with insulation abraded through touching the frame, with one panel foil touching both an electrically active element and the frame.
- d) One voltage tap isolation resistor shorted with the voltage tap wire also shorted to the frame, and one T/C wire insulation abraded through and touching the frame.
- e) Two voltage tap isolation resistors shorted with short circuit of both wires to the frame or each other.
- f) One voltage tap isolation resistor shorted with short circuit of the wire to the frame and one panel foil touching electrically active element and frame.

Only condition (a) of the above failure modes has a significant impact on RTG reliability. The 2-mil iron thermocouple wire has a 15.3 ohm-per-foot impedance. The iron wire length for bridging a significant number of thermoelectric elements in a panel is, at a minimum, 3 inches. The impedance of this length is 3.8 ohms, to which a contact resistance of 2 to 3 ohms must be added. For a total shorting resistance of 6 ohms, the power loss is 1.2 percent or 0.045 watt. If it is assumed that one out of every 10 T/C insulated wires touches an electrically active element and that one out of 100 abrades away its insulation, then the average power loss per converter is:

$$0.45 \text{ watt} \times \frac{1}{10} \times \frac{1}{100} \times 12 \text{ TC wires} \\ = 0.45 \times 0.012 = 0.0054 \text{ watt}$$

For the voltage tap isolation resistors with a failure rate of 6.0×10^{-9} failures/hours, the resistor short circuit failure rate becomes 6.0×10^{-10} failures/hours. The reliability for all 13 voltage tap isolation resistors is:

$$e^{-13(6.0 \times 10^{-10} \times 43,800 \text{ hours})} = 0.99966$$

Thus, the probability of observing a power loss due to shorting isolation resistors is negligible since not only must at least one resistor fail short circuit, but the wire insulation must be abraded through on the wires and the wire be touching active electrical elements. Likewise, the other voltage tap failure modes are considered insignificant.

Instrumentation System Reliability in Yielding Desired Measurements

Reliabilities calculated for all thermocouple and voltage channels under conditions of 0, 1, and 2 single channel failures are as follows:

<u>Instrumentation</u>	<u>Channels Failing</u>	<u>Channels Operating</u>	<u>Five-Year Reliability</u>
Thermocouples	0	12	0.9798
	1	11	0.9998
	2	10	>0.9999
Voltage Taps	0	13	0.9951
	1	12	>0.9999
	2	11	>0.9999

2.4.5 Structural Limitations (Failure Mode No. 3)

The reliability analysis of the converter structure was limited to the component exhibiting the smallest margin of safety. This was the top-cover honeycomb core when subjected to stress. The analysis yielded a reliability greater than 0.99999 for a core shear stress of 18 psi plus a decompression stress of 2.6 psi (at decompression ΔP of 0.1 psi). The same analysis technique applied to the HSSS and cage yielded a reliability higher than 0.999999.

2.4.6 Power Degradation (Failure Mode No. 4)

For all practical purposes, the reliability of the RTG depends only upon Failure Mode No. 4, since data substituted into the reliability model show a reliability of 0.9997 considering Failures Modes 1, 2, and 3. Failure Mode No. 4 involves the degradation of power to an unsatisfactory level. The power output drops off with increasing time due to decay of

the isotope fuel, short-circuit and open-circuit failure of thermoelectric elements, and changes in the material properties of the elements. The material property changes concerned are electrical resistance, thermal conductance, and Seebeck voltage.

In addition to average expected degradation, parameter variations must be included in the degradation analyses. The variance associated with the given power level must be used in combination with the mean expected degradation to determine the probability of meeting the specified power level.

The degradation curve shape is shown schematically in Figure 2-31. It follows the general form $P = Ae^{-Bt}$. The probability of exceeding 30 watts at end of mission is the fraction of the area of the statistical distribution curve above the 30-watt power level.

Figure 2-31. EOL Power Probability Schematic

2.4.6.1 Monte Carlo Modeling of EOM Power

A parameter variation technique was applied to the reliability modeling of the converter in order to include not only the mean expected

degradation but also the variance of the degradation. Changes in thermoelectric properties and converter insulation characteristics were determined by extrapolating the statistical best fit curves to five years and adjusting the data for environmental effects. Then, in order to reflect the variance about the best fit curves, values of the parameters were inserted into the end-of-mission power prediction computer program using random numbers for the deviates of the statistical distributions of the parameters, based on observed data where possible. In doing so, it was recognized that correlation may exist among the parameters and, therefore, a random choice of numerical value for one parameter automatically determined the numerical values of the correlated parameters.

The process of randomly picking a set of parameter values and computing end-of-mission power was repeated 20 times each for mission operating times of 2, 3, 4, and 5 years. The resulting Monte Carlo synthesis of the end-of-mission power was used to determine the probability of achieving an end-of-mission power greater than 30 watts by noting the proportion of the probability density function lying above 30 watts. In the Monte Carlo synthesis, physical properties were varied as a function of time and, where appropriate, as a function of fabrication, air operation, and other environmental variables. The parameters varied were:

- Thermal power
- Number of open-circuit and short-circuit elements
- Element thermal conductivity
- Element Seebeck coefficient
- Element resistance
- Insulation conductivity

The number of open elements used in the Monte Carlo simulation was established as follows:

- An average of 9 open-circuited elements (1 percent risk allocation for 864 elements) was assumed after 5 years of orbital life, based upon best estimates from GE&ES.
- The cumulative probability curves of the Poisson distribution were used to determine the probability of observing 0 to 19 open-element failures for the assumed average of 9 open-circuit elements.

- Random number intervals covering the range from 0 to 1000 were established corresponding to each of the 19 failure probabilities (Figure 2-32).
- For each individual computer run, a number from 0 to 1000 was picked from a table of random numbers and the corresponding number of open-couple failures was used for that run.

Figure 2-32. Random Number Assignments for 0-19 Open Elements

Thermoelectric property changes with time were evaluated by regression analysis of element experimental data. Results are presented in Figures 2-33, 2-34, and 2-35. Power output data on couples sampled from the QM panels were analyzed to predict converter output variability from within panel and panel-to-panel variabilities. Results, summarized in Table 2-7 are small compared to the variability in output power due to thermal input variability, as calculated from the limited data obtained from the Qualification Heat Source, and possibly also in comparison with insulation variability. The modeling was based on an average BOL power of 37.0 ± 0.24 (1σ) watts, with changes in the assumed average BOL power ranging from 34.0 to 37.0 watts.

Figure 2-33. Five-Year Seebeck Coefficient Changes

Figure 2-34. Five-Year Resistance Changes

7

Figure 2-35. Five-Year Thermal Changes

Table 2-7. Electrical Power Output Variation (milliwatts)

	Within Unit σ	Unit-to-Unit σ	Total σ
Couples	-	2.15	2.15
Panels	12.9	0.85	12.9
Converter	44.6	Unknown	4.6
Heat Source *	-	240	240

Figure 2-36 presents the expected percentage power output and $\pm 3\sigma$ variations as a function of time. Figures 2-37 through 2-40 show the probabilities of meeting various EOM power levels as a function of BOL powers of 34, 35, 36, and 37 watts and mission durations of 2, 3, 4, and 5 years.

*Included for comparison only.

Figure 2-36. RTG Power Vs Time

Figure 2-37. EOM Power for 34-Watts BOL

Figure 2-38. EOM Power for 35-Watts BOL

Figure 2-39. EOM Power for 36-Watts BOL

Figure 2-40. EOM Power for 37-Watts BOL

2.4.6.2 Confidence for EOL Power Reliabilities

Confidence statements on the probability of achieving an EOM power greater than a given percentage of BOL power have been determined, using the Monte Carlo synthesis of EOM Power. Confidence limits were based on the size of the data base having the greatest effect on EOM power. The most significant data in this respect came from seven modules with improved hot cap bonds used to determine element physical properties at EOM. With a sample size of 7, the confidence levels for achieving 0.96 reliability for different EOM power levels (given as a percentage of BOL power) were calculated using a computer program developed for this purpose.* Results are shown in Figure 2-41.

* From techniques presented in "Table of Factors for One-Sided Tolerance Limits for a Normal Distribution," Sandia Corporation Monograph SCR-12, April 1958, and presented in "Applications of the Non-Central t-Distribution," Biometrika, Volume 21, 1940, pp. 362-384.

Figure 2-41. RTG Power Output and Mission Life

2.5 TRANSIT RTG QUALIFICATION TEST

The objective of the Transit Generator Qualification Test Program was to demonstrate the capability of the Transit generator to satisfy flight performance requirements after being subjected to environments equivalent to or more severe than actual prelaunch and launch conditions.

2.5.1 Qualification Test Sequence and Results

The test sequence making up the qualification test program is summarized in Table 2-8. The qualification sinusoidal and random vibration levels used are given in Table 2-1. All testing through vacuum performance Test V, with the nuclear heat source, was accomplished with the Qualification Model configured as shown in Figure 3-1. Because of electrical shorts between the thermal insulation foils and the power circuit, noted after vacuum performance Test V, the qualification program was interrupted and a decision made to replace a number of shorted panels with a panel containing a modification to the adhesive-dot insulation restraint and panels using the mica-post insulation restraint technique. The replacement panels were independently subjected to those tests necessary to impose the same environments which the Qualification

Model had received and then used to replace panels in the Qualification Model to form the reconfigured Qualification Model. Additional testing was accomplished as indicated in the last portion of Table 2-8.

All structural components of the converter and heat source support structure and the heat source assembly successfully completed the qualification program described above and were considered satisfactory for use in the flight unit. However, the converter insulation support system in the panels and top cover failed the test and required redesign and retest.

Table 2-8. Qualification Program Test Sequence

Configuration	Test No.	Test Description
Original Qualification Model (QM) Panels 78, 79, 80, 81, 82, 83, 84, 85, 86, 87, 89, 90	1	Receiving Inspection
	2	Vacuum Outgassing
	3	Vacuum Performance Test I
	4	Vacuum Performance Test II
	5	200°C/300 Hour Air Test
	6	250°C/200 Hour Air Test
	7	Vacuum Performance Test III
	8	Depressurization
	9	Acoustics
	10	Vacuum Performance Test IV
	11	Vacuum Performance Test V (with nuclear heat source)
Panel Conditioning	{ 12 13 14 15 16 17	Vacuum Performance Test I
Panels 91, 97, 100		Acceptance Level Vibration
		Vacuum Performance Test II
		200°C/300 Hour Air Test
		250°C/200 Hour Air Test
	17	Vacuum Performance Test III
Panels 91, 92, 94, 97, 100	{ 18 19	Depressurization
		Acoustics
Reconfigured Qualification Model (Q or R) Panels 79, 80, 82, 83, 84, 86, 87, 90, 91, 94, 97, 100	20	Vacuum Performance Test VI
	21	Qualification Level Vibration (with nuclear heat source)
	22	Vacuum Performance Test III
	23	100-Hour Vacuum Test

The major areas of concern revealed by the qualification program are summarized below.

Initial Power Output. The power output of the as-received thermoelectric converter with a thermal input equivalent to the nuclear heat source (850 watts) was 32.4 watts compared to 34.2 watts, the minimum value required to meet the mission objectives. Changes made subsequent to the fabrication of the QM and incorporated in the flight converter significantly improved the performance.

Mirror Blisters. Following the air test, the thermal control mirrors were found in a blistered condition. Subsequent vacuum performance testing significantly reduced the magnitude of the blisters and demonstrated that the condition did not affect performance of the converter. Because of reduced environmental temperatures, this condition will not occur in the actual prelaunch environment.

Failed Thermoelectric Elements. Following qualification-level vibration, five p-elements and two n-elements were found to be broken. The number of open elements was not inconsistent with that predicted from the reliability model which assumed an average of nine during the mission lifetime. However, the qualification test vibration environment was changed to more realistic levels by limiting the converter response rather than the previously used limitation on the heat source response as indicated in Table 2-1.

Panel Insulation Restraint Failure. Panel insulation foil-to-power circuit shorts as low as 0.1 ohm were found on the original QM panels with adhesive-dot restraint following vacuum-performance Test V, and on two of the three reconfigured QM panels using the mica-post insulation restraint following qualification vibration. The initial shorts were apparently of the single-point type and no power loss due to the shorts were noted during vacuum performance Test V. However, during the 100-hour performance test (No. 23 in Table 3-8), indications were that the shorts contributed to approximately a 2-watt power loss. Thus redesign and requalification of the insulation support system were required

Top Cover Insulation Restraint Failure. Following the lateral axis qualification sine vibration the top cover insulation blanket was found to have separated from the top cover support and draped over the heat source, causing electrical shorting by touching the thermoelectric elements. Thus redesign and requalification were required.

2.5.2 Requalification Program for Panel and Top Cover Insulation Restraint

The requalification program included redesign of both panel and top cover insulation restraint techniques to allow modification of existing flight hardware, design verification testing, and qualification testing at the component level.

The panel insulation restraint redesign consisted of strengthening the existing mica post by adhesive bonding of the post to the panel honeycomb structure. The top cover restraint redesign consisted of the following changes to the design initially tested on the Qualification Model:

- The addition of a mica post at the center of the insulation blanket
- The use of a 2-inch diameter aluminum disk at each of the total of five mica posts to distribute the shear loads
- Replacement of the fiberglass threads holding the mica posts in place with Inconel wire fastened to the outside of the top cover structure

A detailed discussion of the redesigned panel and top cover insulation restraint is given in Section 3.4.

The environment to which the test articles were subjected was modified somewhat from that used during the RTG qualification test series to take into account the use of air-conditioning when the RTG is under the Scout heat shield and the results of a vibration analysis which indicated that the vibration input should be response limited at the converter during Z-axis vibration to more realistically simulate the launch condition.

The revised qualification test sequence is summarized in Table 2-9.

Table 2-9. Requalification Test Sequence

Vacuum Performance Mapping (I)	Obtain performance data at $\bar{T}_H = 300, 350,$ and 392°C to establish baseline performance.
Vibration	Perform sine and random vibration twice in succession at acceptance levels. Maintain power input at 885 ± 5 watts during test.
Air Operation	Subject test unit to 250 hours of operation in room ambient air with 885 ± 5 watts power input.
Depressurization	Subject test unit to depressurization, at a rate simulating Scout launch.
Vibration	Perform sine and random vibration at qualification levels. Maintain power input at 885 ± 5 watts during test.
Vacuum Performance Mapping (II)	Obtain performance data at $\bar{T}_H = 300, 350,$ and 392°C to determine effect of test environment.
Post-Test Inspection	Perform post-test examination to determine physical effects of test environment.

Note: A power input of 885 watts was used to represent the maximum expected value for the Flight Nuclear Heat Source.

The major changes in the test program were:

- The use of a response limiting accelerometer located on the top cover of the converter rather than on the heat source during vibration in the Z-axis
- The addition of a random input as part of the acceptance vibration
- An increase in the input level during acceptance vibration to correspond to the expected flight levels
- Reduction of the duration and operating temperature of the air test
- Elimination of the acoustic test

The design verification testing was carried out using panel P-98 and top cover DV-2 mounted on the Structural Test Model converter (STM) to better simulate the response expected in an RTG. Qualification testing was carried out in the same manner using panels P-99 and P-117 and top cover DV-3. The results of the test program demonstrated that:

- The added insulation restraints did not adversely affect performance
- The modified panel restraint effectively eliminated any relative motion between the insulation foils and the thermo-electric elements thus eliminating the possibility of electrical shorts

The successful completion of the requalification test program together with the data obtained during the initial qualification and testing, demonstrated that the Transit RTG containing the modified insulation restraint is qualified for flight.

2.6 FLIGHT MODEL ACCEPTANCE TEST

The Transit RTG Flight Model (FM) acceptance test program consisted of the following test sequence:

- Vacuum Performance Mapping (VPM-I)
- Acceptance Level Vibration
- Vacuum Performance Mapping (VPM-II)
- Earth IR and Solar Radiation Simulation
- Air Operation Simulation

Because the actual flight nuclear heat source will not be fabricated until March 1972, all testing was carried out using an electrical heat source. Due to modification of the panel and top cover insulation restraint, as described in Section 2.5, and made after the initial vacuum performance mapping, it was necessary to repeat that portion of the test.

A comparison of the results of the acceptance test program with the acceptance criteria is given in Table 2-10.

As indicated in the table, all of the requirements were satisfied or exceeded; at an input thermal power of 850 watts, the RTG power output of 36.8 ± 0.3 watts exceeds the minimum requirement by 7.6 percent.

Table 2-10. Comparison of Flight Model Performance with Acceptance Criteria

	Requirement	Actual
Performance		
Power output (watts)	34.2 min.	36.8 $\pm$ 0.3
Average hot cap temperature ($^{\circ}$ C)	392 max.	392 $\pm$ 2
Output voltage (volts)	5.0 min.	5.45
Weight (lb.)	30 max.	29.82
Electrical Isolation (ohms)	200,000 min.	>1 megohm
Mechanical and Electrical Interfaces	per drawing 407028	Requirements satisfied
Panel Resistance Change During Acceptance Test	$\pm$ 2% max.	<2%

Notes:

1. All requirements to be met following acceptance level vibration.
2. Performance to be achieved under simulated space conditions without solar and earth IR radiation and 850 watts thermal input.
3. Weight is based on the assumption that the weight of the caged flight heat source does not exceed 15.9 pounds.

2.6.1 Vacuum Performance Mapping (VPM-I)

The initial vacuum performance mapping was accomplished by the converter subcontractor (GE&ES) prior to and following the modification to the panel and top cover insulation restraint. The testing was conducted in a space simulation chamber without solar or earth IR radiation and included open-circuit load and resistance measurements at hot cap temperatures ranging between 300 and 410 $^{\circ}$ C.

During the first test (prior to insulation restraint modification), at a power input of 850 watts, the flight unit produced 37.4 watts compared to 33.2 watts for the qualification model under similar conditions. The major portion of this improvement in performance was due to reduced

thermal losses attributed to:

- The use of mica post insulation restraint rather than adhesive dots
- Lower thermal conductivity of warm-pressed elements as compared to cold-pressed elements
- Improved panel and end cover joints to reduce thermal losses

An overall comparison of the FM data, before and after the insulation restraint modification, can be obtained from the data in Tables 2-11 and 2-12. At a power input of 850 watts, the loss in electrical power output due to the modification was approximately 0.5 watt or 1.3 percent. A more accurate comparison between the two tests was complicated by the fact that the differences were small with respect to the accuracy of the measured electrical data and the limited temperature data acquired with only 12 flight thermocouples.

Table 2-11. F-1 Performance Summary
(before insulation rework)

Parameters	Data Set				
	1	2	3	4	5
$\bar{T}_H$ ($^{\circ}\text{C}$)	400.2	400.1	393.6	401.8	347.8
$\bar{T}_C$ ($^{\circ}\text{C}$)	132.3	134.1	130.3	119.3	120.0
P_{OUT} (watts)	38.9	38.8	37.7	—	29.3
P_{INPUT} (watts)	864	877	853	746	765
η (%)	4.5	4.43	4.42	—	3.83
V (volts)	5.61	5.12	5.47	10.72	4.13
R ($\text{m}\Omega$)	670	674	660	—	596
m	1.2	1.0	1.2	o. c. *	1.0

* o. c. = Open Circuit

Table 2-12. F-1 Converter Performance Summary
(after insulation rework)

Parameter	Data Set				
	1	2	3	4	5
$\bar{T}_H$ ($^{\circ}\text{C}$)	401.1	399.8	393.6	400.6	348.4
$\bar{T}_C$ ($^{\circ}\text{C}$)	132.5	133.4	130.4	118.4	119.1
P_{OUT} (watts)	38.8	38.6	37.2	—	23.4
P_{INPUT} (watts)	870	880	851	748	769
η (%)	4.46	4.39	4.37	—	3.69
V (volts)	5.63	5.11	5.46	10.68	4.09
R (m Ω)	678	679	664	—	590
m	1.2	1.0	1.2	o. c. *	1.0

* o. c. = Open Circuit

2.6.2 Acceptance Test Vibration

The flight model was subjected to flight level vibration with an electrical heat source supplying a thermal input of 885 watts. Visual inspection of the unit following vibration revealed that with the exception of one cold cap instrumentation thermocouple, which became detached, no changes took place during the test.

2.6.3 Vacuum Performance Test

The vacuum performance testing initiated after the vibration test was accomplished at TRW in a 7 x 12 foot space simulation chamber equipped with provisions for simulating the earth IR and solar radiation.

The following tests were conducted in the chamber:

- Vacuum Performance Mapping (VPM-II)
- Power output as a function of load ratio
- Earth and solar radiation simulation

The results of these tests compared to pre-vibration performance are discussed in the following sections.

2.6.3.1 Vacuum Performance Mapping (VPM-II)

During the vacuum performance test, data were obtained at hot cap temperatures of approximately 300, 350, 375, 392, and 400°C with the load ratio held constant at approximately 1.2.

Curves of nominal power output and hot cap temperature as a function of power input are presented in Figures 2-42 and 2-43. Data obtained during testing at GE&ES prior to the vibration test were included for comparison. The difference in power output at an input power of ~850 watts was 0.5 watt and the maximum difference in hot cap temperature was ~3°C. These differences are not attributed to the effects of vibration; they can be accounted for by the measurement differences between the two facilities. In the region of 850 watts a change of 0.5 watt in power output corresponds to a shift of ~5 watts in power input or 0.58 percent. The best estimate for the measurement system accuracy is about 0.75 percent.

Figure 2-42. F-1 Converter Power Output vs Power Input

Figure 2-43. F-1 Converter Hot Cap Temperature vs Power Input

Figure 2-44 is a plot of power output as a function of the converter load ratio for a constant power input (850 watts). The curve shows that peak power occurs at a load ratio of 1.45. The effect of load ratio on hot cap temperature is shown in Figure 2-45. From the curve of Figure 2-45 it can be seen that for load ratios above 1.2 the design hot cap temperature of 392°C without solar or earth radiation is exceeded, but only by a few degrees.

At the maximum power point (a load ratio of 1.45) the power output exceeded the value at a load ratio of 1.2 by 0.4 watt while the temperature increased by over 4°C. The same data are plotted in the form of I-V curves in Figure 2-46. For a power input of 850 watts and a load ratio of 1.2 the voltage is 5.45 volts and the nominal power output is 36.8 watts.

Figure 2-44. F-1 Converter Power Output vs Load Ratio

Figure 2-45. F-1 Converter Hot Cap Temperature vs Load Ratio

Figure 2-46. F-1 Converter Characteristics in Vacuum

2.6.3.2 Solar and Earth IR Radiation

Table 2-13 summarizes the effects of solar and earth radiation on the generator temperature and power output. Both sets of data were obtained at the same thermal power input of ~850 watts.

The change in temperature was consistent with that noted during the Qualification Model testing, however, the power output did not increase during the Qualification Model test. The increase in power output can be accounted for by the increase in converter hot cap to radiator temperature difference of 1.4°C . This increase was apparently due to the change in thermoelectric properties at the higher temperature.

Table 2-13. Effects of Solar and Earth IR Radiation

	Without Radiation	With Radiation	Change
Average hot cap temperature ($^{\circ}\text{C}$)	392.8	402.8	+ 10
Average cold cap temperature ($^{\circ}\text{C}$)	130.8	139.4	+ 8.6
Temperature difference	262.0	263.4	+ 1.4
Power output	36.8	37.1	+ 0.3

Figure 2-47 shows the measured panel hot and cold cap temperatures before and during solar and earth IR simulation. It can be seen that all panels experienced both hot and cold cap increases due to solar simulation and that the hot cap increases were more uniform than the cold caps. This was due to thermal reradiation which occurred within the converter effectively smoothing out the hot cap temperature increase.

Figure 2-47. T-1 Converter Hot and Cold Cap Temperatures Before and During Solar Simulation ($^{\circ}\text{C}$)

Figure 6-7 is a plot of the temperature increase experienced by a panel as a function of the panel's orientation relative to the sun's rays. The figure may be used to predict the effect of solar and earth IR simulation on uninstrumented panels; for example, the uninstrumented panels with an orientation of 90 degrees from the sun's rays (panels 104 and 110) are predicted to have experienced a 10°C increase in both hot and cold cap temperatures.

Figure 6-7. Panel ΔT During Solar Plus IR Simulation vs Angle Between Sun's Rays and Normal Vector to Panel

6.3.4 Air Operation Simulation

The performance of the Flight Model under ambient air conditions was obtained by operating the unit in a nitrogen atmosphere with a power input of 850 watts. The characteristic performance curve for this condition is shown in Figure 6-9. The maximum power output at a load ratio of 1.2 was 1.92 watts at 0.94 volt with an average hot cap temperature of 165°C and an average cold cap temperature of 104°C . Typical

hot cap and cold cap temperatures under open conditions were 168°C and 104°C , respectively.

Figure 6-8. F-1 Converter Characteristics in GN_2

6.3.5 Performance Summary

Table 6-5 summarizes the expected performance of the Transit Flight RTG under initial in-orbit conditions, assuming a nuclear heat source with a thermal power of 850 watts. The nominal values are based on an analysis of the data obtained during the acceptance test; the tolerances reflect an estimate of the uncertainties in the measured values due to instrument inaccuracies and extraneous heat losses arising from

the use of an electrical heat source in the acceptance testing rather than a nuclear source.

Table 2-14. Predicted Transit RTG Initial In-Orbit Performance (850-watt NHS)

Average hot cap temperature ($^{\circ}\text{C}$)	400 ± 2
Average cold cap temperature ($^{\circ}\text{C}$)	139 ± 1
Power output (watts)	36.8 ± 0.3
Load voltage (volts)	5.45 ± 0.01
Load current (amps)	6.77 ± 0.02

3. CONVERTER

3.1 CONVERTER CONFIGURATION

The Transit thermoelectric converter utilizes the Isotec thermoelectric technology developed by GE&ES for the AEC. The basic design requirements were discussed in conjunction with the RTG System (Section 2.1).

The converter assembly, shown in Figure 3-1, consists of a frame structure, 12 thermoelectric panels, top and bottom support structure or covers, top and bottom shear rings, 12 corner posts and associated attachment hardware. Changes to the baseline converter design were necessary in order to arrive at a final flight configuration, including the method of panel and top and bottom cover thermal insulation restraint and the development of high density, low porosity n-elements fabricated by vacuum warm-pressed techniques.

The major converter components are described below.

3.1.1 Corner Posts

The 12 corner posts, fabricated from magnesium-thorium, are attached to the bottom support structure and each of the thermoelectric panel frames to provide vertical support for the converter.

3.1.2 Shear Rings

Two magnesium-thorium shear rings, containing grooves to accept the tongue at the top and bottom of each thermoelectric panel are attached at the corner posts.

3.1.3 Coatings

All the exterior surfaces of the converter, with the exception of the thermoelectric panel radiators which contain second-surface mirrors, are coated with a high emissivity (0.88), low absorptivity (0.20), paint, Z-93 to enhance the heat rejection capability of the assembly. The collectors on the interior of the assembly are coated with a high temperature, high emissivity (0.85) paint, Pyromark, to minimize the temperature of the heat source needed to transfer the required amount of heat.

3.1.4 Thermoelectric Panels

Figure 3-2 is a section of a typical thermoelectric panel. Each panel is constructed of a magnesium thorium frame and a phenolic honeycomb structure to which 36 n-p thermoelectric couples are attached. The thermoelectric materials are PbSnMnTe for p-element and PbTe doped with iodine for the n-element. P-element end caps are iron and n-element end caps are nickel. The end caps are bonded to the collector at the hot end by spot welding and to the radiator at the cold end by laser welding. Multifoil insulation, consisting of alternate layers of aluminum foil and aluminum opacified quartz paper is used to minimize heat losses. Molybdenum-opacified quartz washers are stacked around each thermoelement to prevent electrical shorts to the foils and inhibit PbTe sublimation.

Figure 3-2. Typical Thermoelectric Panel Cross Section

3.1.5 Top Support Structure

The top support or cover is a laminated structure comprised of two 0.010-inch magnesium-thorium sheets with 0.0007-inch by 3/8-inch cell aluminum honeycomb. A multifoil thermal insulation blanket is attached, which interfaces with the thermoelectric panel insulation to form the insulation system.

3.1.6 Bottom Support Structure

The bottom support is a laminated structure comprised of 0.010-inch magnesium-thorium sheet with 0.0007-inch by 3/8-inch cell and

BLANK

0.0007-inch by 3/16-inch cell aluminum honeycomb plus an inner ring of 2024-T851 aluminum which incorporates the structure support attachment of the heat source support structure. As with the top assembly a multilayer thermal blanket is attached and interfaced with the panel insulation.

3.2 ANALYSES

Converter analyses were performed to establish the configuration of the converter and to analytically verify its capability to satisfy the design requirements. In the following sections, the major analyses performed during the program are summarized.

3.2.1 Degradation Mechanisms

Causes of electrical power output degradation include:

- Chemical reactions at the thermoelectric element bonds
- Thermoelectric material instability
- Vaporization of the thermoelectric material during operation in vacuum
- Thermal cycling of the thermoelectric elements
- Oxidation of the thermoelectric material during operation in air
- Changes in properties of the second-surface mirrors
- Changes in properties of the insulation materials

The major contributing cause to degradation is associated with the p- and n-element bond and material instability. Details of the mechanisms involved are given in "Bonding of Lead Telluride Thermoelectric Elements," GA-8997 (Confidential). In the p-element the effects of manganese-telluride precipitation and gold-tin migration appear to level off after about 5000 to 7000 hours. The almost linear decrease in power of the n-element is apparently due to porosity buildup.

Investigation of the porosity buildup indicates that the effect may be minimized by changes in the fabrication process so as to produce higher density elements. The fabrication of high density n-elements has been accomplished by the use of warm pressing techniques.

Vaporization effects are very minor because of the baffling effect of the molybdenum opacified insulation around the thermoelements and the low operating temperature. Test results indicate that thermal cycling under the expected operating conditions should not be a problem.

Data on the behavior of the converter after operation in air for 300 hours at 200°C and 300 hours at 250°C indicate that there is a small amount of degradation but that power is recovered after an additional short operating period in vacuum.

The second-surface mirrors used on the radiator have been subjected to ultraviolet and charged particle bombardments at levels simulating Transit orbital conditions. No significant degradation in solar absorptivity or emissivity was noted.

Examination of modules operated for 1000 hours in air at 200°C and 5000 hours in vacuum at 450°C has revealed no change in the properties of the insulation. The air operation results in the formation of an aluminum oxide layer about 1.4×10^{-3} microns thick which is well below the thickness (0.5 micron) at which infrared reflectance changes occur.

Combining the degradation mechanisms described above, a converter power versus time profile was computed using the techniques described in Section 2.4. Figure 2-36 presents the results. Note that the converter is expected to produce approximately 82 percent of the beginning-of-life power after 5 years in orbit at a nominal hot cap temperature of 400°C.

3.2.2 Stress

A structural analysis similar to that described in detail in Section 5.3 and based on a dynamic lumped model of the complete converter system was performed to allow an evaluation of the margin of safety for the converter components. The analysis used peak amplification factors which were experimentally measured on the Structural Test Model (STM) converter during qualification-level tests in the X, Y, and Z axis. Results of the structural analysis are presented in Table 3-1. The margin of safety is positive for all components and is believed adequate considering that minimum allowable strength properties were used in all cases.

Table 3-1. Summary of Transit Converter Stresses and Safety Margins Based on Experimental Amplification Factors

	Calculated Stress (psi)	Allowable Stress (psi)	Margin of Safety
<u>Honeycomb Panel</u>			
Honeycomb shear stress	113	180	0.59
Face stress	17,260	25,000	0.45
Face wrinkling stress	17,260	19,330	0.12
Bond shear stress	526	1,600	2.04
<u>Corner Posts</u>			
Compression	2,710	19,000	6.0
Buckling	73	2,332	30.9
Screw fastener, shear	12,200	13,400	0.1
<u>Bottom Cover</u>			
Center plate face stress	4,000	20,000	4.0
Outer plate face stress	6,220	20,000	2.22
Core shear stress	29.3	46	0.57
Face wrinkling stress	4,000	8,500	1.13
<u>Top Cover</u>			
Face stress	2,870	20,000	5.97
Core shear stress	20.6	40	0.94
Face wrinkling stress	2,870	4,330	0.51

3.3 TESTING

The converter test program was comprised of component-level tests (i.e., 2-couple modules and 36-couple panels) and converter assembly-level tests.

3.3.1 Component Test Description

The component testing fell into three major categories: (1) thermo-electric performance, (2) structural performance, and (3) fabrication

process support.

The thermoelectric performance tests were conducted under environmental conditions simulating the ground test, launch, and space operation of the converter at design and off-design values. The tests included thermal cycling, air operation, and long-term operation in vacuum.

The structural performance tests utilized panels subjected to shock and vibration when mounted to a structure simulating the converter. The panels were then subjected to performance and life testing at the design operating temperature.

Fabrication process support tests evaluated thermoelectric materials, bonding processes, and the method of attachment of the second-surface mirrors to the converter radiator surface.

A summary of the component tests, typical of those carried out during the program, is presented in Table 3-2.

3.3.2 Converter Test Description

Testing at the converter level was designed to provide engineering data necessary to evaluate the design adequacy of the converter and provide the information to determine the design changes necessary to achieve the required performance.

Two engineering tests, designated as the Structural Test Model (STM) test and the Engineering Test Model (ETM), were carried out during the program. These tests, together with the RTG Qualification Test discussed in Section 2.5, provided the information required to finalize the flight converter design.

3.3.3 Component Test Results

3.3.3.1 Thermal Cycling Tests

A total of six modules were subjected to ten thermal cycles in air to hot cap temperatures of 200°C , at a rate of approximately 23°C per second, in order to simulate temperature transients caused by insertion and removal of the heat source. The results of these tests indicated a small increase in resistance, approximately 2.8 percent, but no

Table 3-2. Summary of Modules and Panels on Test

Test Location	Test Times	Item	Description	Test Sequence	Objectives
GE&ES	12,040	QC-47	Two-couple module	P-A-P(L)	Evaluate effect of air operation on bond life at 400°C average hot cap temperature
	3,234	QC-85	Two-couple module	P-A-P(L)	Evaluate effect of air operation on bond life at 400°C average hot cap temperature
	8,865	QC-61	Two-couple module	P-TC-P(L)	Evaluate effect of thermal cycling on bond life at 400°C
	11,437	QC-56, -65	Two-couple module	P(L)	Bond life at 390°C average hot cap temperature
	13,059	MSC-150	Two-couple module	P(L)	QC 3M thermoelectric material and p- and n-elements at 410°C average hot cap temperature
	10,404	QC-60, -62	Two-couple module	P(L)	Life performance test at 420°C
	8,579	M2C-170, -171	Two-couple module (warm-pressed elements)	P(L)	Evaluate long term performance at 400°C
	7,011	M2C-175	Two-couple module (warm-pressed elements)	P(L)	Evaluate long term performance at 400°C
	5,915	M2C-176	Two-couple module (warm-pressed elements)	P(L)	Evaluate long term performance at 400°C
	3,503	MSC-183, -184	Two-couple module (warm-pressed elements)	P-A-P(L)-E	Evaluate long term performance at 400°C after air exposure
	1,709	MQC-101, -104	Two-couple module (warm-pressed elements)	P(L)	Evaluate long term performance at 400°C
	5,332	M2C-181, -182	Two-couple module (warm-pressed elements)	P(L)	Evaluate long term performance at 400°C
	2,017	MQC-10	Two-couple module (warm-pressed elements)	P(L)	Evaluate long term performance at 400°C
	6,456	P-75, -76	36-couple panel	P-V-P-A-P-V-P(L)	Evaluate long term performance at 400°C hot cap temperature after environmental test
TRW	7,979	DVT-7, -8	Two-couple module	P-A-P-L(E)	Evaluate effect of air operation on performance at 400°C hot cap temperature
	13,594	QC-37	Two-couple module	P-L(E)	Evaluate long term performance at 400°C hot cap temperature
	5,844	QC-53, -59	Two-couple module	P-L-P(E)	Evaluate long term performance at 400°C hot cap temperature
	13,594	QC-40	Two-couple module	P-L(E)	Evaluate long term performance at 390°C hot cap temperature
	4,122	QC-93, -95	Two-couple module	P-L(E)	Evaluate long term performance at 375°C hot cap temperature
	4,435	PE-77	36-couple panel	P-L(E)	Evaluate long term performance at 400°C hot cap temperature

* (L) indicates final status

detrimental effects to the physical integrity of the couples. Additional testing was done, using two modules, to simulate the worst thermal cycle condition caused by depressurization during launch. The thermal cycle used has a rate of rise of approximately 65°C per minute from 250°C to 450°C . Both modules were subjected to five thermal cycles.

No significant changes were noted at the end of the cycling. One module was then subjected to 20 additional cycles from room temperature to 250°C and back to room temperature at approximately 100°C per minute. Again, no significant change in resistance was noted, leading to a high degree of confidence in the ability of the elements to withstand thermal cycling.

3.3.3.2 Air Tests

Four modules were subjected to 1000 hours of air operation at a hot cap temperature of 200 to 250°C followed by over 1000 hours of operation in vacuum at a hot cap temperature of 400°C . The results shown in Figures 3-3 and 3-4 are typical. The resistance of the modules increased and the power output decreased rapidly during the first 100 hours of air operation and then became stable. During subsequent operation in vacuum the resistance decreased and the power output increased approaching the original value in about 1000 hours.

Module resistance changes were found to be due to an increase in the resistance of the n-elements; however, the mechanism of the change was not identified. It should be noted that these tests were carried out with n-elements fabricated using the cold pressing process; subsequent testing using warm-pressed n-elements resulted in a significant reduction in the resistance change.

3.3.3.3 Structural Tests

In addition to the air and vacuum performance tests, two modules were subjected to tensile testing and metallurgical examination after the air operation.

Figure 3-3. Module Resistance vs Hours of Operation in Air

Figure 3-4. Module Power vs Vacuum Life Test Hours After 1050 Hours of Air Operation

The strength of the n-elements tested was 3160 gm and 4000 gm and that of the p-elements was 1185 and 1400 gms. These values exceed the minimum required value of 1800 gm and 1000 gm for the n- and p-elements, respectively. However, the values for the p-elements were below the average for unoperated elements. Unfortunately, the sample size was insufficient to determine if the deviation from the average was due to the air operation.

3.3.3.4 Life Testing

Figure 3-5 gives the performance as a function of time for panels P-75 and P-76. These panels used cold-pressed elements but were fabricated with full process control. The panels were subjected to short-term air operation and qualification-level vibration prior to life testing. The degradation rates for the panels were consistent with the reliability model discussed in Section 2.4.

Table 3-3 summarizes the performance of vacuum warm-pressed modules and vacuum warm-pressed panel P-92. The changes in power output are again consistent with the Transit reliability model. Panel P-92 was subjected to short-term air operation and qualification-level vibration prior to the life test.

3.3.4 Converter Test Results

3.3.4.1 Structural Test Model (STM)

The basic objective of the STM test was to evaluate the capability of the converter to meet the structural requirements for vibration, mechanical shock, and acoustic field environments.

The STM is a typical 12-sided polygonal structure consisting of the following major components: converter frame structure, heat source support structure, electrical heat source, 10 inactive (simulated) thermoelectrical panels, 2 active thermoelectric panels, and a bottom structural support plate. The unit was subjected to the following sequence of tests:

Figure 3-5. Sequential Test Panels Power Output

- 1) Acceptance level vibration (X and Z axes) — 2/3 of qualification level
- 2) Qualification level vibration (X, Y, and Z axes)
- 3) Shock test (Z axis)
- 4) Reverberant acoustic vibration
- 5) Qualification random vibration (X, Y, and Z axes)

No thermoelectric elements on the active panels (PE-51, PE-52) were damaged during the test sequence and all structural components with the exception of the panel and top and bottom cover thermal insulation blanket restraints were demonstrated to be adequate.

Table 3-3. Performance Changes of Vacuum Warm-Pressed Test Items

Module No.	Test Time (hours)	Total Change in Normalized Power (percent)	Estimated Contribution of Element Type to Total Change	
			n-Element (percent)	p-Element (percent)
M2C-170	7905	- 6.4	- 1	- 5
M2C-171	7905	- 4.5	- 1	- 3
M2C-174	5600 ^(a)	- 3.9	d	d
MSC-175	6340	- 5.2	- 1	- 4
MSC-176	5244	- 5.2	d	d
MSC-181	4660	- 5.1	- 0.5	- 4
MSC-182	4660	- 3.9	~0	- 4
MSC-183	2830 ^(b)	- 2.5	~0	- 2
MSC-184	2830 ^(b)	- 2.1	~0	- 2
P-92	1870 ^(c)	- 0.5	d	d

(a) Removed for metallographic inspection

(b) Preconditioned with 500-hour air operation at 200 to 250°C

(c) Transit qualification test panel removed for installation in QM

(d) Individual element data not available

During the initial vibration testing, it was found the method of stitching the insulation blanket to the top and bottom covers using metal wires was not adequate to hold the blanket in place. The top cover was reworked by increasing the number of stitches and replacing the metal wire with fiberglass thread. The modified insulation assembly survived qualification level vibration.

Electrical checks following qualification level vibration indicated shorts between the insulation foils and the power circuit in both active panels, PE-51 and PE-52. Visual examination revealed that the insulation

blanket had shifted so that it was resting against some of the elements. The opacified washers normally surrounding the elements were badly deformed and clustered in the opening that remained between the element and the insulation blanket.

3.3.4.2 Engineering Test Model (ETM)

The Engineering Test Model represented the reference design configuration but did not have the modifications necessary to prevent slippage of the insulation blanket and subsequent electrical shorting of the thermoelectric panels. Therefore, the test sequence did not include mechanical testing. The ETM test sequence included the following:

- 1) Performance test
- 2) Performance test with solar input
- 3) Rapid decompression at Scout ascent rate
- 4) Air operation test (500 hours)
- 5) Performance test

Following acceptance testing at GE&ES the ETM was installed in a 7 by 12 foot chamber at TRW which contained a liquid nitrogen-cooled shroud and provision for solar and earth radiation simulation. The chamber was sufficiently large so that no corrections for the view factor as compared to space were necessary. Figure 3-6 is a photograph of the ETM installed in the test chamber.

Subsequent depressurization, air and thermal efficiency tests were carried out in a 30-inch chamber.

The results of the ETM Test Program lead to the following conclusions:

- Of the environments to which the converter was subjected, mechanical shock due to shipment and handling, thermal cycling, air operation, vacuum operation, depressurization and solar and earth radiation; only air operation caused any significant changes in performance.
- Air operation resulted in a 10 percent increase in the electrical resistance and a compensating decrease in the thermal conductance. Therefore, no significant change in power output occurred. These results verified analytical predictions regarding the relationship between electrical resistance and thermal conductivity which could not be demonstrated by earlier panel and module tests.

- The effects of solar and earth IR radiation obtained during the test were consistent with design calculations and indicated that operation at hot cap temperature of 392°C in a cold wall vacuum chamber without solar simulation or earth IR will result in an average hot cap temperature of 400°C during space operation. No significant change in power output occurred because of the solar and earth radiation effects
- Although the 39.1-watt power output of the converter, at an average hot cap temperature of 400°C , exceeded the expected value of 37.1 watts, the thermal power required to achieve 400°C was in excess of the expected value by approximately 14 percent because of high heat losses through the top and bottom covers. Subsequent thermal efficiency experiments have demonstrated that these losses can be reduced to acceptable levels.

Figure 3-6. ETM Converter in 7 x 12 Test Chamber

3.3.4.3 Qualification and Flight Model

The Qualification and Flight Acceptance Tests and Test Results are contained in Sections 2.5 and 2.6 and will not be repeated here. However, the significant conclusion from these tests is that the Flight Converter incorporating the improvements discussed in detail in the next section satisfies all Converter Design Requirements.

3.4 DEVELOPMENT

3.4.1 Vacuum Warm-Pressed N-Elements

Examination of Isotec n-elements produced by the cold pressing technique and operated for long periods in a vacuum environment revealed extensive porosity at a point just below the element effective hot junction. Analytical considerations indicated that the porosity was due to the migration of entrapped gases. The use of a warm pressing technique in which the elements were pressed in a vacuum at elevated temperatures ($\sim 600^{\circ}\text{C}$) resulted in a high-density element with a low gas content even after being subjected to the gas environment associated with the hot and cold cap bonding operation.

Table 3-4 summarizes the results of density measurements of a group of warm-pressed elements compared to conventionally produced cold-pressed elements. It shows a reduction to nearly one-tenth in the density change caused by a simulated bonding cycle of the vacuum, warm-pressed elements.

Table 3-4. Density Changes of PbTe as a Function of Processing and Atmosphere Environment (density in grams/inch^3)

Sample Description	Density as Pressed	Density After Heating in Argon ($685^{\circ}\text{C} - 10 \text{ min}$)	Density After Heating in H_2 ($685^{\circ}\text{C} - 10 \text{ min}$)	Total Change in Density
Four elements conventional cold-pressed	133.6	131.8	130.6	2.2%
Vacuum hot-pressed in developmental apparatus	134.5	134.1	134.0	0.4%
22 elements from four different runs done in production vacuum warm-pressing station	134.37	134.24	134.02	0.26%

The photomicrographs of Figure 3-7 compare the porosity in an n-element fabricated by cold pressing with an element fabricated by the vacuum, warm pressing technique. Both elements were operated at $\sim 400^{\circ}\text{C}$ for about 5000 hours. The amount of porosity in the vacuum, warm-pressed element is negligible compared to the cold-pressed element.

Cold Pressed Element

Warm Pressed Element

Figure 3-7. Comparison of Cold and Warm Pressed Elements

Warm-pressed elements also appear to have a higher strength than the cold-pressed elements; a group of 12 elements subjected to tensile tests broke at loads ranging from 3875 to 6740 grams compared to the specification value of 2000 grams.

Based on the data discussed above and the performance of modules and panels using vacuum, warm-pressed elements discussed in Section 3.3.3, the vacuum, warm-pressed process was selected for the flight design converter.

3.4.2 Panel and Top Cover Restraint

As a result of the slipping of the panel multifoil thermal insulation and the structural and thermal problems associated with holding the top and bottom thermal insulation blanket in place, observed during the STM vibration testing and the ETM test, a program was undertaken to develop improved techniques for achieving a satisfactory method of restraining the multifoil insulation blanket.

3.4.2.1 Panel Restraint

Two primary panel insulation restraints were investigated; one method used small adhesive dots between each insulation layer and the other used glass thread passing through mica buttons placed between each insulation layer to hold the insulation to the honeycomb panel structure.

Because of initial success with the adhesive dot technique as compared to the stitched or mica post design, the adhesive dot restraint was subjected to a more extensive test program. Based on the successful completion of the test program described in Table 3-5 this technique was selected for use in the Qualification Model converter and the mica post development was continued as a backing technique.

The Qualification Model panel restraint utilized 15 (3 columns of 5 rows each) adhesive dots applied between each insulation layer. A 0.129-inch diameter hole was used in the paper to control the adhesive dot size and a 0.062-inch diameter hole was used in the foil to provide a complete adhesive column through the insulation. Figure 3-8 illustrates a typical adhesive column.

Table 3-5. Panel Insulation Support Qualification Program

	Glue Dot			Mica Post		
	P-75	P-76	QM	DP-03	P-92	P-94
Performance	X	X	X	X	X	X
Vibration - Acceptance	X	X	X	X	X	X
Vibration - Qualification	X	X	N/A	X	X	X
Performance	X	X	X	X	X	X
Thermal Cycle	X	X	X		X	X
Air	X ⁽¹⁾	X ⁽¹⁾	X ⁽¹⁾		X ⁽²⁾	X ⁽³⁾
Performance	X	X	X		X	X
Decompression			X			
Acoustic			X			
Performance (NHS)			X			
Vibration - Qualification	X	X			X	
Performance	X	X			X	
Life	X	X			X	

(1) 300 hours at 200°C T_H
 200 hours at 250°C T_H

(2) 200 hours at 250°C T_H

(3) 335 hours at 250°C T_H (180°C = T_C)

Despite the success of the two panel test, when the Qualification Model was subjected to essentially the same test sequence, foil-to-power shorts were noted on six of the twelve panels following the performance test with the nuclear heat source.

To allow continued testing of the Qualification Model a modification to the adhesive dot restraint was developed. It consisted of reinforcing

Figure 3-8. Adhesive Dot Restraint

the adhesive dot at four locations on the panel with a stitch which passed through the column of adhesive and was tied to the panel frame as shown in Figure 3-9. However, a subsequent investigation of the failure of the adhesive dot restraint lead to the conclusion that the adhesive did not have sufficient temperature capability to provide the required design margin needed to attain high reliability. As a result the mica post restraint design was specified for the flight converter.

In the mica post configuration shown in Figure 3-10, the insulation foils are held in place by the compressive load applied to the mica buttons by a fiberglass thread which is loaded to approximately 7 pounds prior to tying. A vycor rod is placed to react the shear loads on the mica post and minimize the loads applied to the fiberglass thread during vibration. As discussed in Section 2.5 the mica post design required modification in order to successfully pass the qualification. During qualification testing the insulation restraint on two panels failed. A third panel survived the test sequence. The most significant difference between the third panel and the others noted during post-test examination was that the insulation

Figure 3-9. Stitch/Adhesive Dot Restraint

Figure 3-10. Panel Mica Post Insulation Restraint

blanket was locally bonded to the honeycomb directly under the mica stack in a manner which would make the stack, vycor rod, and thread act as an integral column. It was postulated that without this bond the dynamic load applied to the vycor rod caused it to fail and transfer the load to the fiber-glass threads resulting in a shear failure in the threads at the interface of the blanket and honeycomb structure.

It was concluded that the restraint provided by the mica post could be made adequate to withstand the environment if the insulation paper at the base of the honeycomb post could be glued to the honeycomb panel structure in a manner similar to that observed in the panel which successfully passed the earlier testing.

A development program to evaluate various methods of obtaining a bond between the honeycomb and the mica post which could be applied to the existing flight panels without damage to the panels resulted in the design shown in Figure 3-11. This design allows adhesive to be injected with a hyperdermic needle through two holes drilled in the radiator side of the panel frame to the vicinity of the mica post. In addition, adhesive is injected along side the existing mica post threads.

Results of developmental tests of six mica posts assembled to simulate the flight design showed:

- A very strong bond is developed at the mica post panel interface.
- The threads when impregnated with the adhesive form a continuous sized structure.
- With the use of a drill jig the holes can be repeatedly located at the required position relative to the mica post.

The adequacy of the design was verified through panels tests and the requalification testing described in Section 2.5.

3.4.2.2 Top Cover Insulation Restraint

As a result of the ETM testing it became apparent that excessive heat was being lost through the top and bottom insulation blanket. To reduce this heat loss a development program using the ETM converter as a test bed was conducted in order to evaluate a variety of top cover insulation blanket configurations.

Figure 3-11. Panel Assembly Showing Insulation Restraint Design

Since the basic construction of the top and bottom blankets is identical, the results of the top cover tests apply to the bottom cover as well.

The results of the program are summarized in Table 3-6. As a result of the test program, blanket No. 5 was chosen for use with the Qualification Model converter. The top cover insulation restraint arrangement is shown in Figure 3-12. Both top and bottom insulation blanket restraint made use of 36 stitches around the periphery of the cover and 4 stitches arranged around the center of the blanket. The stitches at the periphery of the blanket were installed loosely in a manner to maintain the desired spacing between layers of foil. At the four center stitches thin mica discs were placed between each layer of insulation to prevent thermal shorting between foils and to provide mechanical reinforcement. Each mica disc was provided with two holes to allow for passage of the quartz thread. The four stitches at the center of the blanket were pulled more tightly to act as rigid posts.

As discussed in Section 2.5, the top cover insulation blanket failed the initial qualification level vibration necessitating a further redesign and component qualification test. This was successfully accomplished and the final flight design incorporated the following changes to the configuration shown in Figure 3-12.

- 1) The fiberglass thread through the 1/4-inch diameter mica post was replaced by a 10-mil diameter Inconel X wire preloaded to maintain a compressive force on the post at all times.
- 2) Shear loads and tearing forces on the hot insulation layers were counteracted by a 2-inch diameter, 8-mil thick aluminum disc formed into a rigid shape.
- 3) A 3/4-inch diameter, 0.02-inch thick stainless washer with a 1/8-inch half-round cylinder in the center was used to evenly distribute loads from the wire to the post and aluminum disc.
- 4) Two 1-inch diameter, 1/16-inch thick aluminum washers were used to distribute the load across the 0.57-inch thick aluminum honeycomb top structure, which has a crushing strength of 20-50 psi. The phenolic honeycomb spacer was hollowed out to accommodate the washer beneath the insulation, and epoxy 8751 was used to hold the washer in place.

Table 3-6. Summary of Top Cover Insulation Blanket Tests

Top Cover Insulation Blanket	Outside Top Cover (°C)				Inside Top Cover (°C)		Heat Flow (watts)		
	Edge	9 in. from Center	6 in. from Center	Center	Middle	Center	Radiated from Top	Conducted Side to Top	Through Insulation
Blanket No. 1									
Open circuit	11	--	-13	-23	436	446	81	14	67
Closed circuit	18	--	-9	-19	445	456	88	19	69
Low emissivity top cover (closed circuit)	118	--	122	124	440	451	44	0	44
Blanket No. 2									
Open circuit	-1	-22	-31	-37	440	445	63	23	40
Closed circuit (2nd run)	6	--	-27	-36	451	457	69	25	44
Closed circuit (4th run)	5	-17	-27	-34	449	455	68	25	43
Closed circuit (5th run)	9	--	-25	-33	462	463	71	22	49
Blanket No. 3									
Open circuit	--	--	-30	-40	449	--	70	20	50
Closed circuit (1st run)	--	--	-26	-37	457	--	74	20	54
Closed circuit (2nd run)	12	-1	-22	-34	454	458	78	20	58
Blanket No. 4									
Open circuit	10	-11	-23	-37	449	465	73	23	50
Closed circuit	14	-8	-21	-36	456	472	76	23	53
0.023 gap (open circuit)	-36	-46	-57	-65	446	462	39	0	39
0.100 gap (open circuit)	-28	-38	-51	-60	447	459	48	0	48
0.025 aluminum ring (open circuit)	10	-7	-20	-38	450	464	76	23	53
Low emissivity top cover (open circuit)	100	98	95	95	444	456	34	0	34
Blanket No. 5									
Open circuit (1st run)	11	-1	-17	-30	450	472	78	17	61
Open circuit (2nd run)	12	-1	-18	-32	449	471	78	18	60

Blanket Description:

- No. 1 34 layers of aluminum foil and opacified paper with 126 stitches.
 No. 2 34 layers of aluminum foil and opacified paper with 42 stitches and 48 glue posts.
 No. 3 50 layers of aluminum foil and opacified paper with 36 stitches and 4 mica posts.
 No. 4 16 layers of aluminum foil and 25 layers opacified paper and 8 layers double-aluminized Kapton with 37 stitches.
 No. 5 34 layers of aluminum foil and opacified paper with 36 stitches and 4 mica posts.

- 5) A 3/4-inch diameter wave washer with a free height of 0.09-inch was used on the cold side to assure a preload force on the wires and counteract differential thermal expansion, elastic deformation in the wire and honeycomb, and creep in the mica post assembly and wire.
- 6) The preload wires were secured to two posts which were part of a 3/4-inch diameter, 0.04-inch thick stainless washer. They were secured after obtaining the desired preload by soldering, bending around the tabs and redundantly spot welded.

A total of five reinforced posts are used with the fifth post located at the center of the blanket to prevent contact between the insulation blanket and the heat source structure during vibration. A crosssection of an individual post is shown in Figure 3-13.

Figure 3-12. Baseline Top Cover Restraint

NEW ITEMS:

- 119 - 3/4" diameter, .040" thick stainless steel terminal
- 120 - BeCu retaining plug, 5° taper.
- 122 - .010" diameter Inconel X-750 wire.
- 123 - #0-80 screw, stainless steel.
- 124 - top support structure with five 1.0" diameter x 0.17 high aluminum cups.

Figure 13. Cross Section of Top Cover Insulation Restraint

4. HEAT SOURCE

4.1 DESIGN REQUIREMENTS

4.1.1 Operational

The principal operational design requirements for the Transit heat source are as follows:

- Operate for 5 years in orbit
- Contain sufficient fuel to generate 30 watts electrical power at end of life in conjunction with an Isotec converter compatible with a maximum system weight of 30 pounds
- Withstand ground handling and storage for periods of up to 1 year

4.1.2 Nuclear Safety

The nuclear safety criteria imposed on the Transit heat source are discussed in Section 6.2.

4.2 BASELINE DESIGN

The baseline Transit heat source design is shown in Figure 4-1 and consists of the following three subassemblies: (1) a three-layer refractory metal capsule which contains the fuel, (2) a two-layer graphite heat shield which resists reentry environments and reduces the impact velocity of the heat source, and (3) an oxidation-resistant outer can which provides the capability of 1-year storage and prelaunch fueled operation in air for up to 1000 hours.

The fuel capsule contains a nonselective continuous pressure relief device (PRD) which retains the fuel but permits helium gas generated in the fuel bed to pass out of the capsule into the heat shield cavity. The outer can, supported by the heat source support structure, is designed to contain this gas pressure for approximately 1.0 year from the time of fueling assuming 100 percent helium release.

A capillary vent assembly is attached to the outer can. The capillary is sealed during heat source storage and prelaunch operation and is opened just prior to launch to permit a controlled helium release while

Figure 4-1. Heat Source Design

maintaining a helium backpressure within the heat source, thus improving internal heat transfer.

The fuel capsule is coated with a special high emissivity coating which serves the dual purpose of improving the heat transfer between the fuel capsule and heat shield and also prevents any chemical reaction between the capsule clad and the graphite heat shield. In addition, an emissivity coating is applied to the outer surface of the can to minimize heat source temperatures.

The capsule is supported by the heat shield hemispherical ends as shown in Figure 4-1. The two graphite plugs, one of which is threaded into the outer heat shield and the other of which slides, permit a preload to be applied to the capsule through the heat shield to prevent rattling of the capsule within the heat shield during launch. In addition, the sliding plug accommodates the differential thermal expansion between the metallic capsule and the graphite heat shield during all phases of operation or abort. The threaded rings on the end with the sliding plug is configured to relieve thermal stress in certain critical areas during steep reentry after an early abort. The preload applied to the capsule is also transmitted through the clad to the strength member and eliminates any rattle between these two members. A finite gap does exist, however, between the inner fuel liner and the liner, and between the liner and strength member but is kept at a minimum to preclude excessive loads during launch. The heat shield contains internal cutouts in its cylindrical wall in order to minimize weight and thereby reentry heating and impact velocity. The thermal barrier is pyrolytic graphite, a low conductivity graphite to minimize internal temperatures during reentry. Poco graphite spacers restrain the thermal barrier sleeve from axial movement.

4.2.1 Heat Source Component Summary

A summary of the baseline heat source component function, materials, sizes, and weights is shown in Table 4-1. The data are based on the Transit Qualification Heat Source since the Flight Unit will not be assembled until March 1972.

Table 4-1. Heat Source Components (PMC Fuel)

Capsule Component	Principal Function	Material	Outside Diameter (in.)	Thickness (in.)	Outside Length (in.)	Nominal Weight (lb)
Fuel	~850-watt source	Plutonia molybdenum cermet	2.140	-	4.714	5.82*
Inner Liner	Provide fuel compatibility	Mo-46Re	2.139	0.005	-	0.09
Pressure relief device	Release helium, retain fuel	Ta-10W Al ₂ O ₃	-			0.02*
Liner	Provide fuel compatibility and containment	Ta-10W	2.204	0.020	5.367	0.56*
Strength member	Provide impact resistance	T-111	2.402	0.090	5.590	4.18*
Clad	Provide oxidation protection for strength member	Pt-20Rh	2.460	0.020	5.662	0.55*
Thermal barrier	Thermal protection of capsule during reentry	Pyrolytic graphite	2.839	0.150	5.812	0.53*
Spacers	Restrain thermal barrier from axial motion	POCO graphite	3.000	Variable	1.176	**
Heat shield	Reentry protection and aerodynamic profile to limit impact velocity	POCO AXF-5Q	5.000	0.250	7.125	3.68*
Can	Oxidation protection for heat shield	Inconel 718	5.045	0.010	8.119	0.51*
Capillary assembly	Release helium from the can	Stainless steel	0.032 OD	0.004 ID	180	0.05*
Total						13.99

* Measured weights (average) from Qualification Heat Source

** Weight included in thermal barrier

4.2.2 Components

4.2.2.1 Fuel

The nominal power level required in the Transit heat source is 850 watts. The fuel form was changed from the microspheres to solid solution cermet (SSC) and later to plutonia-molybdenum cermet (PMC) because of difficulties encountered in the fabrication of SSC discs. Molybdenum coating is applied to shards by vapor deposition in a fluidized bed and the fuel is hot-pressed to form a disc having a power density in excess of 3.4 watts/cc.

MRC Specifications 1-14531 and 1-14532 cover the requirements for fabricating PMC fuel discs. Specification 1-14531 is used to produce a constant geometry cylindrical disc and Specification 1-14532 is used to produce a variable geometry cone frustrum disc to meet the TRW fuel

envelope requirements. Twenty-two discs are required to produce the required thermal power level. The geometry of the 22 discs is as follows: 6 discs are cone frustra of three different sizes, 3 have diameters slightly less than standard discs, and 13 have a standard diameter of $2.145^{+0.000}_{-0.006}$ inches.

The power of the first fueled heat source for RTG qualification testing measured by calorimetry was 849.1 watts, resulting in an achieved fuel power density of 3.32 watts/cc.

4.2.2.2 Pressure Relief Device

Since plutonium 238 is an alpha-emitting isotope, it is necessary to either fully contain the helium generated by isotopic decay, or allow it to escape from the capsule in a controlled manner so that the fuel is completely contained. By the use of a pressure relief device in the capsule, creep and eventual rupture of the capsule, caused by the continual buildup of helium pressure, are eliminated.

The pressure relief device design, shown in Figure 4-2, utilizes Ta-10W in the flanged housing, end plug, seal plug, and core. The core is plasma-arc sprayed with Al_2O_3 and isostatically pressed in the flanged housing. Helium permeates the ceramic at a nominal flow rate of 3.3×10^{-5} scc/sec with a pressure differential of approximately 10^{-3} atmosphere.

Figure 4-2. Pressure Relief Device

Since the flanged housing is welded to the liner end cap, the PRD material was selected to be the same as the liner material to obtain optimum compatibility and weldability between the two components.

The seal plug on the external side of the PRD allows the liner to be decontaminated after fueling without admitting impurities to the capsule interior through the PRD. After fueling and decontamination, the seal plug is punctured before further assembly of the liner into the strength member.

4.2.2.3 Inner Liner/Liner

Early compatibility data between PuO_2 and Ta-10W at the expected capsule temperatures led to its selection as the primary liner material. However, more recent information from the Transit Qualification Capsule and several Pioneer capsules indicate that a replacement liner of Mo-50Re or an inner liner of Mo-50Re are required to improve compatibility.

The inner liner contains the fuel and serves as the primary fuel compatibility member of the heat source. It consists of a cylinder and two hemispherical end caps and is made of 0.005-inch thick Mo-46Re. The components are joined by slip, or lap joints, and serve only to prevent direct contact of the fuel with the primary Ta-10W liner. The basic Ta-10W liner has a cylindrical wall thickness of 0.020-inch and end caps of 0.030- and 0.040-inch thicknesses.

4.2.2.4 Strength Member

The strength member surrounds the liner and serves as the primary member to withstand structural loads. T-111 material was selected for the strength member due to its high strength at the required operating and abort temperatures and also its impact resistance. The wall and end cap thickness are 0.090 inch.

4.2.2.5 Clad

The clad protects the strength member from air oxidation during assembly, handling, and reentry and assists in providing fuel containment during launch pad aborts and in post-impact environments. Pt-20Rh was selected for the clad since it is extremely oxidation-resistant in the expected temperature range and because of considerable experience with this material in past applications. The clad wall thickness is $0.018 \pm$

0.002 inch in the cylindrical section and 0.020 ± 0.004 inch in the end caps.

4.2.2.6 Thermal Barrier

A pyrolytic graphite (PG) thermal barrier surrounds the fuel capsule and provides the necessary insulation to protect the capsule during reentry. Because of its anisotropic properties, it also diffuses the reentry heat so that temperature gradients will be minimized and hot spots will not occur in the heat source metallic components.

Pyrolytic graphite has a low thermal conductivity in the "c" direction (thickness) and two orders of magnitude higher conductivity in the "a-b" direction (hoop and axial). It is produced by chemical vapor deposition from a hydrocarbon gas on a graphite substrate mandrel.

The PG thermal barrier consists of a three-piece assembly: two body pieces with hemispherical ends and a sleeve. Each of the components is 0.075-inch thick yielding a thermal barrier which is 0.075-inch thick at the ends and 0.150-inch thick in the cylindrical portion.

4.2.2.7 Heat Shield

The heat shield is an aeroshell which reduces aerothermodynamic heating during reentry and subsequent impact velocities by presenting a large area at minimum weight. The heat shield also acts as a structural member and an ablator during reentry.

Although both polycrystalline bulk graphite ATJ-S and Poco AXF-5Q would meet the Transit design requirements, the Poco AXF-5Q was selected for the heat shield material because of the state of the art and availability of this material. Bulk graphites have the lowest recession rate of refractory materials under reentry conditions. Graphite also has the lowest density of the refractory materials capable of surviving reentry temperatures. The outer configuration of the heat shield is 5 inches in diameter and 7.125 inches long. The inside configuration is contoured leading to a 0.25-inch thick cylindrical section with 1-inch end plugs. The end plugs consist of a sliding unit on one end which accommodates the expansions of all the components and a threaded plug on the other end for assembly of the capsule into the heat shield.

4.2.2.8 Can

The Transit heat source may be exposed to air at temperatures ranging from 800 to 900°F for up to 1000 hours prior to launch.

Excessive oxidation of the graphite, if exposed, would occur under these conditions. Graphite has appreciable variability in oxidation resistance over the temperature range indicated, and variations in impurities and graphitization temperature will significantly affect the oxidation rate. Laboratory confirmation of excessive oxidation at this temperature has been obtained. Thus an oxidation barrier or "can" is required.

Several types of can designs were considered to protect the graphite heat shield from air oxidation during the 1000-hour prelaunch period and a final design employing a cylindrical body with convoluted end caps was selected (Figure 4-1). Inconel 718 was chosen for the can material. This selection was based on corrosion resistance, favorable mechanical properties at 1000^oF, good weldability, and the availability of this material.

4.2.2.9 Capillary Assembly

Helium is released from the can through a nominal 0.004-inch inside diameter stainless steel tube called a capillary tube. A 15-foot length of tube is coiled on a spool which is attached to the end of the can (Figure 4-1). The tubing has sufficient resistance to helium flow to provide a helium backpressure on the order of 3 psi within the can for 100 percent helium released from the fuel. The backpressure results in lower temperature differentials between internal capsule components and consequently lower internal operating temperatures. Fabricated tubes have been successfully tested for flow rate versus backpressure and assembly procedures have been demonstrated.

4.2.2.10 Coatings

High emittance, plasma-arc sprayed ceramic coatings were selected for use on components of the Transit heat source. High emittance alumina^{*} was chosen for the outside of the Inconel 718 can and the outside of the Pt-20Rh cladding. The choice of TRW-100M was dictated by the requirement that the high emittance coating be stable in intimate contact with graphite at elevated temperatures.

*The high emittance alumina is TRW-100M (platinized alumina).

4.3 SUPPORTING ANALYSES

The purpose of the analyses performed for the Transit heat source was to establish the configuration of the heat source and to analytically verify its capability to meet the design requirements. The analyses have, in general, used conservative assumptions. The final verification of the adequacy of both the analyses and the selected design came from the design verification tests (Section 4.4.1).

In the following sections more significant analyses performed during the course of the program are summarized.

4.3.1 Design Analyses

4.3.1.1 Initial Parametric Studies

Parametric investigations of heat source dimensions were initially performed for both the microsphere and solid solution cermet fuel forms. The results are summarized in Table 4-2. All heat sources shown in the table were designed for the material thicknesses of the baseline design.

Table 4-2. Heat Source Weight/Impact Velocity Parametric Analyses

Section	Fuel Form	Thermal Loading (watts)	Capsule				Heat Source				
			L (in)	D (in)	L/D	Fueled Weight (lb)	L (in)	D (in)	L/D	Fueled Weight (lb)	V (ft/sec)
1	Microspheres	850	6.70	2.46	2.72	8.62	8	5	1.60	13.68	334
	Microspheres	850	6.70	2.46	2.72	8.62	8	4	2.00	12.61	364
	Microspheres	850	6.70	2.46	2.72	8.62	8	3-1/2	2.28	12.13	382
2	Microspheres	850	6.35	2.54	2.50	8.54	8	5	1.60	13.75	336
	Microspheres	850	5.52	2.76	2.00	8.36	6.9	5	1.38	13.19	354
	Microspheres	850	3.73	3.73	1.00	8.16	5	5	1.00	12.87	405
3	Microspheres	850	3.73	3.73	1.00	8.16	8	5	1.60	16.53	368
4	SSC	836	5.66	2.46	2.30	9.37	7-1/8	5	1.42	14.00	350
	SSC	836	5.66	2.46	2.30	9.37	7-1/8	4-3/4	1.50	13.74	356
	SSC	836	5.66	2.46	2.30	9.37	7-1/8	4-1/2	1.58	13.50	363
5	SSC	836	5.66	2.46	2.30	9.37	8	4-1/2	1.78	14.07	350
	SSC	836	5.66	2.46	2.30	9.37	7-3/4	4-1/2	1.72	13.89	353
6	SSC	793	5.66	2.46	2.30	9.13	7-1/8	5	1.42	13.83	347
	Microspheres	793	6.33	2.46	2.57	8.15	8	5	1.60	13.26	320

Note: Subsonic drag coefficients used are conservative. See Table 4-17 for drop test results.

The first section of Table 4-2 shows the effect of decreasing the heat shield diameter for an 850-watt microsphere heat source design which was evaluated early in the program. Decreasing the outer heat shield diameter from 5 inches to 3.5 inches (minimum required graphite thickness for thermal protection and fabricability), leads to a heat source weight saving of up to 1.55 pounds. However, there is a concurrent increase of 48 ft/sec in impact velocity for side-on reentry. This is primarily due to the decrease in area exposed to reentry and also to a slight decrease in subsonic drag coefficient. The impact velocities* for heat shield diameters of 4 inches and 3.5 inches are unacceptably high.

The second and third sections of Table 4-2 show the effect of varying the L/D ratio of both the fuel capsule and heat shield for an 850-watt microsphere heat source. Again there is a decrease in heat source weight with decreasing L/D but a concurrent increase in impact velocity. The increase in impact velocity is due to the decrease in area exposed to reentry which overrides the effect of the small increase in the drag coefficient. The design in the third section of the table represents a spherical fuel capsule within a right circular cylindrical heat shield. It can be seen that the heat source weight is significantly higher than the heat sources with cylindrical fuel capsules. The reason for this is the increase in heat shield weight caused by the additional graphite thickness at the cylinder ends. The same trends of increasing impact velocity and decreasing heat source weight with decreasing L/D are shown in Sections 4 and 5 of the table for an 836-watt SSC heat source.

Section 6 shows the initial designs for the SSC and microsphere fuel forms. It can be seen that the SSC heat source is about 0.5 pound heavier with a higher impact velocity based on the previous analytical values of drag coefficients and not the Transit drop test data. This same trend is illustrated by Figure 4-3 which shows the variation in heat

*These values were calculated based on available data prior to the Transit subsonic drop tests.

source weight with fuel loading for a constant heat source diameter and nearly constant impact velocity for both the SSC and microsphere fuel forms. The solid fuel heat source is heavier than the microsphere heat source because the significantly higher physical density of the SSC fuel form is not compensated for by its slightly higher power density.

The subsonic drag coefficients used in all of the preceding analyses were later shown to be significantly conservative when compared to the results obtained from the drop tests (Section 4.4.1.7). However, the relative correlations between the various configurations are correct.

Figure 4-3. Variation in Heat Source Weight with Fuel Loading for a Constant Heat Source Diameter and Nearly Constant Impact Velocity

4.3.1.2 Helium Release Studies

The helium permeability of a typical Transit PRD covering the pressure range from 10^2 to 10^{-5} atmospheres was measured to provide information on the operation of the PRD under high flow conditions that might occur during reentry, due to retention of helium in the fuel and subsequent release on reentry. The results are presented in Figure 4-4. As can be seen, at low pressures the permeation rate was proportional to the pressure drop while at high pressures the permeation rate was proportional to the square of the pressure drop.

Figure 4-4. Experimental Permeability Data for Typical Transit Vent

The experimental results are compared with the theoretical predictions in Figure 4-5. The predicted performance was obtained by selecting a porosity and mean pore size of the ceramic ($\epsilon = 0.25$ and $C_m = 2.0\mu$) and solving for the remaining unknown (tortuosity) required to give a measured permeation rate at 1×10^{-4} atmosphere differential pressure. A computer program was then used to calculate the permeation rates as a function of pressure drop. As may be seen the correlation is excellent, particularly if the transition between laminar and turbulent flow is assumed to occur gradually.

Figure 4-5. Correlation Between Permeability Model and Experimental Permeability Data

These data were then applied to demonstrate survivability during reentry of the Transit SSC capsule with a pressure relief device. The reentry mode considered was side-on stable; 100 percent helium retention in the fuel during orbit operation and the percent of creep life method was used to predict the rupture time of the capsule strength member during reentry. Using Equations 1 and 2 to describe the gas flow capability of the PRD during reentry, it was shown that the SSC capsule would survive if reentry occurred within 7 years of encapsulation. For reentry times in excess of approximately 7 years after fuel encapsulation, or greater, the strength member would undergo creep failure if all helium were retained in the fuel matrix and then released during reentry. Complete helium retention for this long period is not expected and capsule failure during reentry does not appear to be a significant concern.

$$V = 6.44 \times 10^{-2} \Delta P \times \left(\frac{527.4}{T}\right)^{0.5} \times e^{-(4/\Delta P)} + 0.102 \Delta P^2 \left(\frac{527.4}{T}\right)^{1.63} \quad (1)$$

$$V = 5.54 \Delta P \left(\frac{527.4}{T}\right)^{0.5} \quad (2)$$

where:

V = flow rate in std cc/sec

ΔP = pressure drop across the pressure relief device in atmospheres

T = temperature in $^{\circ}R$

Equation 1 is valid for the viscous flow range and Equation 2 is valid for turbulent flow.

4.3.2 Aerodynamic Analyses

The aerodynamic analyses were concerned with possible reentry trajectories and the associated heating environments, which could be experienced by the Transit heat source. The reentry analyses have considered the Transit heat source to be reentering by itself from 400,000 feet altitude; breakup studies have been conducted which indicate that

breakup can be expected at a sufficiently high altitude so that the results obtained assuming a free heat source at 400,000 feet are valid.

The initial conditions for reentry were taken from safety studies which defined the possible reentry conditions for given abort situations. In addition an orbital decay trajectory was determined. All trajectory calculations assumed a point mass with a nonspherical, rotating earth, and the 1962 Standard Atmosphere. The trajectory program computed the free-molecular and convective heat transfer rates at the stagnation point of the vehicle. The trajectory calculations performed are tabulated in Tables 4-3 and 4-4.

Table 4-3. Transit Reentry Trajectory Cases

Abort No.	γ_I (deg)	V_I (ft/sec)	Azimuth (deg)	Latitude (deg)	Longitude (deg)	Altitude (ft)
1	26.48	28,342	179.9	-38.84*	34.50**	400,000
2	22.77	28,077	179.9	-30.11	36.12	
3	19.12	27,620	179.9	-24.56	38.35	
4	33.97	28,283	179.9	-66.43	34.09	
5	37.71	27,960	179.9	-81.69	33.23	
6	45.02	26,743	179.3	-66.83	-137.22	
7	40.47	27,500	179.55	-76.08	-42.45	
8	22.98	27,614	0.107	-40.89	38.56	
9	31.53	27,056	0.666	-75.24	38.89	
10	15.80	27,050	179.9	-29.00	41.20	
11	13.00	26,400	179.9	-40.70	44.40	
12	10.00	25,450	179.7	-42.50	42.00	
13	8.00	24,550	179.5	-37.70	16.00	
14	10.70	27,180	0.200	-4.00	61.70	
15	7.20	26,690	0.264	-5.00	48.00	
Orbital Decay	0.05332	25,678	180.00	45.00	0.00	
Grazing	2.00	26,932	179.81	-53.20	-125.56	

*Latitude = + North

**Longitude = + East

Table 4-4. Transit Skip Trajectory Conditions at 400,000 Feet

Identification No.	γ_i (deg)	V_i (ft/sec)	Azimuth (deg)	Latitude (deg)	Longitude (deg)
1	1.776	26946	179.75	-127.19	-68.65
2	1.500	26950	179.72	-127.31	-71.40
3	1.357	26952	179.70	-127.37	-72.82
4	1.580	26949	179.72	-127.27	-70.65
5	1.680	26947	179.73	-127.23	-69.71
6	1.400	26951	179.70	-127.35	-72.42
7	1.450	26950	179.71	-127.33	-71.90
8	1.300	26952	179.69	-127.39	-73.42
9	1.540	26949	179.72	-127.29	-71.02
10	1.560	26949	179.72	-127.28	-69.46
11	1.570	26949	179.72	-127.28	-69.40

The stagnation heating rate, stagnation enthalpy, and stagnation pressure history for a typical side-on stable orbital decay reentry is shown in Figure 4-6.

The stagnation heating rate, stagnation enthalpy, and stagnation pressure history for the heat source entering side-on stable in skip trajectory No. 1 ($\gamma_i = 1.776$ $V_i = 26946$, pass No. 2) are presented in Figure 4-7. The heating rate, enthalpy, and pressure history for terminal entry following a skip trajectory are similar to a random orbital decay.

4.3.3 Thermal Analyses

Thermal analyses were conducted of the 827-watt Transit heat source including appropriate modification to the thermal model based on the thermal verification test results. The following conditions were evaluated: air operation (with and without converter), orbital operation (with and without converter), and side-on reentry. The analyses were conducted using the TRW Systems Improved Numerical Differencing Analyzer program (SINDA).

Figure 4-6. Transit Orbital Decay Reentry

Figure 4-7. Transit Orbital Decay Reentry
 $(\gamma_i = 1.776, V_i = 26946, \text{pass No. 2})$

4.3.3.1 Steady-State Analyses

All clearances and voids inside the can were assumed to contain helium gas. Both radiation and conduction were considered as modes of heat transfer using nominal clearance sizes. The Poco graphite and PG were assumed to be in contact over the end caps. No contact was assumed between the can and graphite ends, however, because of the domed shape of the can. No allowance was made for support structure perturbation of the can temperature since previous analyses had indicated this to be negligible.

Air Operation. Steady-state temperatures for the heat source sitting in air at approximately 70°F are shown in Figure 4-8. Steady-state temperatures for the heat source in a converter at 350°F (RTG inside the launch vehicle shroud with air conditioning provided) are shown in Figure 4-9. Figure 4-10 shows the steady-state temperatures for the heat source in a converter at 483°F (RTG inside the launch vehicle shroud with no air conditioning provided).

Orbital Operation With Converter. Steady-state heat source temperatures were determined for two space operating conditions: closed-circuit normal converter operation ($T_{\text{conv}} = 752^{\circ}\text{F}$) and a failed open-circuit converter ($T_{\text{conv}} = 930^{\circ}\text{F}$). For these two conditions the heat source temperatures are shown in Figures 4-11 and 4-12.

Orbital Operation Without Converter. Steady-state temperatures were also determined for the heat source free in space. This condition could conceivably be achieved if the converter and cage assemblies were burned off during several passes through the atmosphere in a skip trajectory resulting from abort. Figure 4-13 shows the steady-state temperatures for the heat source contained in the can with helium gas. Figure 4-14 shows the steady-state temperatures after can burnoff with no helium gas present except in the liner-to-fuel gap. Fortunately, the latter condition only occurs in the unlikely event of a severe skip trajectory and is present only for a few minutes prior to terminal reentry.

2002

Figure 4-9. Steady State Temperatures with Heat Source in Converter and Shroud, Converter at 350°F on Launch Pad with Air Conditioning, Helium in All Gaps

Figure 4-10. Steady State Operating Temperatures with Heat Source in Converter and Shroud, Converter at 483°F on Launch Pad with No Air Conditioning, Helium in All Gaps

Figure 4-11. Steady State Operating Temperatures with Heat Source in Converter at 752°F (Closed Circuit) in Space, Helium in All Gaps

Figure 4-12. Steady State Temperatures with Heat Source in Converter at 930°F (Open Circuit) in Space, Helium in All Gaps

Figure 4-13. Steady State Temperatures with Heat Source in Space Without Converter, Helium in All Gaps

Figure 4-14. Steady State Temperatures with Heat Source in Space Without Can and Converter, Helium in Liner/Fuel Gap, Vacuum in All Other Gaps

4.3.3.2 Reentry Analyses

Reentry analyses were performed for the Transit heat source using the three-dimensional model shown in Figure 4-15. Several reentry trajectories were examined: orbital decay, hover, a maximum heating rate trajectory (Abort No. 5), and the several skip trajectories previously discussed.

The nominal computed heating rates for all trajectories were multiplied by a factor of uncertainty equal to 1.3, and the temperatures were computed assuming a CO exothermic chemical reaction on the Poco graphite surface during the subsonic portion of descent. Analysis of the Houston arc-jet tests led to the conclusion that CO rather than CO₂ is the reaction product which occurs at the graphite surface during subsonic flow. Helium gas was assumed contained in all gaps within the capsule during reentry. With the exception of the skip reentries, which are discussed below, air was assumed in all gaps and voids external to the capsule in the heat shield during reentry.

Table 4-5. Maximum Component Temperatures During Side-On Reentry, $F = 1.3$ CO Reaction Subsonic He/Air*

	Orbital Decay (°F)	Hover (°F)
Outer Poco**	4103	4125
Innter Poco**	3521	3582
Outer PG	3222	3303
Inner PG	2636	2741
Clad	2598	2707
Strength member	2591	2702
Liner	2588	2699
Fuel	2554	2684
Strength member at impact	2315	2427
Outer Poco** at impact	690	699
Minimum Poco thickness remaining	0.100 inch	0.054 inch

Table 4-6. Maximum Component Temperatures During Abort 5 Side-On Reentry, $F = 1.3$ CO Reaction Subsonic He/Air*

	Heat Source Alone from 400K Feet	Heat Source After Reentry Breakup
Outer Poco**	6514	5136
Inner Poco**	4790	3874
Outer PG	3560	2131
Inner PG	1811	1598
Clad	1763	1594
Strength member	1757	1595
Liner	1753	1597
Fuel	1775	1674
Strength member at impact	1718	1592
Outer Poco** at impact	1718	1592
Minimum Poco thickness remaining	0.190 inch	0.208 inch

*He/air = He inside the capsule/air external to the capsule

**Stagnation line, mid-plane

Skip Aborts. Aerodynamic analyses indicated that third and fourth stage vehicle malfunctions could result in skip reentries (Section 4.2.2). Figure 4-16 shows a band of total integrated heating prior to impact versus the number of skips possible. The circled numbers represent the cases analyzed in detail and correspond to the identified number previously given in Table 4-4. Of the 11 skips analyzed, skips ①, ⑤, and ⑪ resulted in the highest heating rates to the heat source. Of these, only ① and ⑤ resulted in heating rates sufficient to melt the outer can ($T_{\text{melt}} = 2350^{\circ}\text{F}$) which surrounds the heat shield. Based on these results, the region where can burnoff occurs was shaded in Figure 4-16. Helium conduction was assumed in all gaps prior to can burnoff. After can burnoff, but before terminal entry, vacuum was assumed external to the liner with helium inside the liner. During terminal entry the vacuum gaps were assumed to be filled with air.

Figure 4-16. Range of Skip Environments

The analyses showed that in all cases the heat source, following a skip into the atmosphere, would cool back to steady-state temperatures before its next pass. The steady-state exo-atmospheric temperatures were shown in Figures 4-13 and 4-14. The peak temperatures during skips but prior to terminal entry are shown in Columns 2 and 3 of Table 4-7. Columns 4 and 5 show the maximum and impact temperatures during terminal entry preceded by a skip in which the can has burned off. Terminal entry preceded by a skip where the can does not burn off results in temperatures similar to those of a random orbital decay. The ablation depths occurring in trajectory (1) and (5) were 176 and 185 mils, respectively; of this, only 29 and 28 mils occurred before terminal entry, since the can acts as an ablation shield during the early skips. The ablation depth for a skip in which the can does not burn off before terminal entry was found to be similar to that of a random orbital decay.

Table 4-7. Skip Temperatures During Side-On Reentry,
F = 1.3 CO Reaction Subsonic

Component	Peak Skip Temperature (°F)		Terminal Entry Temperature** (°F)	
	With Can	Without Can	Peak	Impact
Can	2330	-	-	-
Outer Poco*	2195	2600	4098	1379
Outer PG	1913	2101	3286	2325
Clad	1967	2207	2741	2387
Strength member	1992	2604	2736	2431
Liner	2021	2886	2734	2506
Fuel	2118	2981	2892	2658

* Stagnation line, mid-plane

** Without can initially

4.3.3.3 Ablation Analyses

Figures 4-17 and 4-18 present the ablation at the midplane of the heat source versus time for both trajectories analyzed assuming a CO_2 reaction occurred during subsonic flow. A refinement was made to the thermal model to accurately determine the corner rounding of the outer heat shield during reentry. A detailed three-dimensional ablative heat source model, shown in Figure 4-19, was used. This model described in detail the corner of the heat shield. The rest of the heat source was described by a less detailed model. The final heat source surface profile along the stagnation line at time of impact, caused by an orbital decay reentry, is shown in Figure 4-20. Figure 4-21 shows the final backside profile (180 degrees). Previous analyses had shown that the ablation caused by an orbital decay reentry was more severe than those associated with ascent abort; therefore, no corner rounding analyses were performed for the abort cases. During arc-jet testing, the subsonic reaction product was found to be CO rather than CO_2 . Using the CO reaction in the analytical model, the total recession was reduced from the values discussed above (see Section 4.4.1.14).

Figure 4-17. Transit Heat Source Ablation Depth Vs Time

Figure 4-18. Transit Heat Source Ablation Depth Vs Time

Figure 4-19. Three-Dimensional Transit Heat Source Corner Rounding Model

Figure 4-20. Transit Heat Source Surface Profile at Time of Impact, Orbital Decay Reentry, $F = 1.3$, CO_2 Product Stagnation Line

Figure 4-21. Transit Heat Source Surface Profile at Time of Impact,
Orbital Decay Reentry, $F = 1.3$, CO_2 Product, 180-Degree Backface

The hover trajectory results in the most severe ablation, with a minimum of 0.054 inch of graphite remaining at time of impact. The orbital decay results in approximately 0.01 inch of graphite remaining, whereas Abort 5 ablates very little leaving a minimum thickness of 0.19 inch. RTG/spacecraft breakup during Abort 5 results in the least ablation of those analyzed with 0.208-inch graphite left at impact. Again, Houston arc-jet reentry test data showed that the Poco heat shield does maintain sharp corners.

4.3.4 Structural Analyses

4.3.4.1 Inconel Can

A major portion of the structural analyses conducted for the heat source concern the Inconel can which provides oxidation protection prior to launch. This can was designed to provide helium containment for approximately 1 year after fueling. The helium pressure at the end of this period would be 72 psia, based on the assumptions of 100 percent helium release and space operation temperatures. During the early portion of the program, cans which could provide for helium containment for up to 6.5 years were also considered. The 6.5-year period includes 1 year of ground handling plus 5.0 years of space operation and a 6-month contingency factor. The pressure at the end of the 6.5-year period, again based on the previous assumptions, would be 312 psia.

A number of different stress analyses have been made for the convoluted end cap design which features a toroidal type outer section and

Figure 4-22. Convoluted End Cap Design

a central dome as shown in Figure 4-22. A large deflection shell program (Fidor) was used for these analyses. The results indicated the stress at the outer transition between the cylindrical shell and the end cap toroid was too high when the toroid height was approximately 0.2 inch. This was a result of both can and support structure geometry.

Additional analyses were then made to determine if a modified design would be acceptable. A small deflection shell program, AS 173A, commonly known as the NAA Shell Program, was used for these studies. The cross-section of the toroidal section of the end toroid to half of an ellipse. The depth of the ellipse was varied from 0.2 inch to a full semi-circular cross-section. These results indicated that only the semi-circular cross-section would have acceptable stresses for a 312 psia, 6.5-year life. The 0.707 elliptical section (ratio of minor-to-major axis) would have a buckling problem at this pressure as would all shallower cross-sections. For a 72-psia pressure, analyses indicated that an ellipse which is 0.4 inch in depth has acceptable stresses.

All of the above analyses were based on the nominal 0.010 inch thick Inconel 718 material. However, if a 0.010 nominal thickness material were used, there would be a thickness tolerance of ± 0.002 inch on the basic material and a possible reduction of 0.001 inch due to the forming operation. Thus, the material could be as thin as 0.007 inch. For a 72-psia pressure it was determined that 0.007-inch thick material was marginal from a buckling standpoint for a 0.4-inch depth ellipse; however, a 0.48-inch deep ellipse is acceptable.

The potential failure mechanism for the Inconel 718 can is creep rupture. A preliminary creep rupture analysis was accomplished considering the effects of linearly increasing stress at a temperature of 1000°F. This analysis indicates that a 1.5-year can is feasible, but a 6.5-year can made from 10-mil material would be marginal.

4.3.4.2 Effects of Helium Pressure on Heat Shield

The graphite heat shield was examined to determine its ability to resist sudden depressurization which could occur if helium is suddenly released in space or during ascent. In this event, the heat shield would temporarily become a pressure vessel. The pressurized time period would be brief because the helium would permeate through the porous graphite as well as escape through leak paths between the two end plugs of the heat shield.

Analyses were made which indicated that at pressure gradients of up to 255 psia, the graphite heat shield would not be damaged. The most critical area was the interface between the side wall and end.

4.3.4.3 Structural Analyses During Reentry

A stress analysis of the outer heat shield when subjected to aerodynamic and inertial loading during heat source reentry was conducted. This analysis indicated that the stresses and strains due to aerodynamic loading when combined with the thermal stresses and strains will not cause failure of the heat shield.

4.3.5 Thermal Stress (480 n mi Nominal Trajectory)

The Transit heat source is protected from the reentry thermal environment by an outer heat shield consisting of four components, and an inner heat shield consisting of three components. This configuration was shown in Figure 4-1. The outer heat shield material is AXF-5Q graphite, and the inner heat shield material is pyrolytic graphite. Graphite spacers serve to limit axial movement of the inner PG sleeve.

The heat shields were analyzed for two side-on reentry conditions, Abort No. 5 and Orbital Decay for a 480 n mi nominal trajectory. The Abort 5 reentry is a steep angle reentry and produces the highest surface temperatures and maximum thermal gradients in the components. The orbital decay reentry is a shallow angle reentry which produces higher heat source internal temperatures than Abort 5.

Thermal stresses for all heat shield components were critical for Abort No. 5, and the thermal stress analysis indicated that all components will survive this condition. Reverse thermal gradients which occur during the cooldown phase of both reentry conditions were also reviewed. The thermal stresses in the heat shield components were found to be insignificant for these reverse gradients.

The thermal stress analysis was performed by two static, finite element computer programs. A description of the two programs is as follows:

Stress Analysis of Axisymmetric Solids (SAAS). This is a solids of revolution program which will compute stresses and strains resulting from thermal loading. It requires both axisymmetric geometry and loading. It can account for material property (E, G, α, ν) variations with temperature and has a bilinear stress-strain option (plasticity effects).

Stress Analysis of Axisymmetric Solids Subjected to Unsymmetric Loading (SAAS-U). This program is very similar to SAAS except: (1) will accept non-axisymmetric loadings through use of Fourier coefficient expansion techniques; (2) material property variations with temperature can be accommodated in only a very crude manner; (3) has no bilinear stress-strain option, therefore only linear effects can be considered.

The geometry of the thermal stress models are symmetric about the longitudinal Z-axis. The outer heat shield body and the inner heat shield sleeve are also symmetric about a plane perpendicular to the longitudinal axis and passing through the midpoint of the cylinder.

All components are assumed to act independently with no structural interaction. The radial gap and longitudinal slip joint preclude interaction between the inner and outer heat shields. The relative thermal expansion of the components will always provide a radial gap between the adjacent components of the heat shields.

The material properties for the analyses were the most conservative of those which are available. Early thermal stress analysis of the outer heat shield body indicated that the conservative properties might be too restrictive. However, the final analysis verified that the body would survive with the assumption of conservative properties, thereby eliminating the requirement of selective procurement of material for flight articles.

The outer heat shield body was analyzed for the Abort No. 5 re-entry at $t = 15$ seconds and at $t = 17$ seconds. The steepest thermal gradient occurs at $t = 17$ seconds. At $t = 15$ seconds the gradient is 10 percent smaller; however, the temperature in the critical area is 800°F cooler, resulting in a substantially reduced allowable strain to failure. The minimum margins of safety are 7 percent at $t = 15$ seconds and 12 percent at $t = 17$ seconds. For combined aerodynamics and thermal loading these minimum margins occur on the inner surface of the body.

The SAAS program was used for the analysis of the outer heat shield body. In the course of the analysis, it was found that the assumption of elastic material properties of the SAAS-U program was too conservative for AXF-5Q graphite at the temperatures involved. In addition, it was found that although the thermal loading is unsymmetric, the temperature gradients and geometry of this particular component made possible an accurate prediction of stresses and strains with a computer program which assumes axisymmetric loading.

The criterion for failure of the outer heat shield body was a comparison of the principal thermally-induced strain with the uniaxial strain to failure. This was necessary since, for the nonlinear analysis, the predicted strains were more accurate than the stresses. For the outer heat shield graphite, strain-to-failure data are available only up to 4350°F. At temperatures above 4000°F, AXF-5Q graphite is very ductile, and the failure strain above 4350°F is greater than 2.0 percent. Tensile failure strains were conservatively assumed to apply for compression.

The analysis for all other components was performed using the SAAS-U program. The failure criterion was a comparison of the principal thermally-induced stress to the uniaxial tensile failure stress or the uniaxial compressive yield stress, when compression allowables were available.

The outer heat shield threaded ring and screw plug were analyzed for the Abort No. 5 reentry at $t = 24$ seconds. The minimum margin was 0.69. The minimum tensile margin of safety for the screw plug was 9 percent.

A thermal stress analysis of the outer heat shield plug was not performed. A comparison of the geometries and thermal loads of the plug and screw plug indicated that the plug would also be acceptable.

The inner heat shield body and sleeve were analyzed for the Abort No. 5 reentry at $t = 30$ seconds. For these analyses, tensile allowable stresses were assumed to apply also for compression. The minimum margin of safety in the sleeve is greater than 100 percent. At $t = 24$ seconds the temperature gradient in the sleeve is approximately 50 percent greater than at $t = 30$ seconds. The thermal stresses are approximately

proportional to the temperature gradient. Since the stresses at $t = 30$ seconds were small, another analysis at $t = 24$ seconds was not felt to be necessary. The margin of safety would be greater than 50 percent at $t = 24$ seconds, and the sleeve was judged acceptable for Abort No. 5.

4.3.6 Thermal Stress (570 n mi Nominal Trajectory)

An analysis was performed to determine the ability of the heat source heat shield to survive reentry thermal stresses and/or strains for the 570 n mi trajectory. Based upon a similar analysis discussed in Section 4.3.5 for the 480 n mi nominal trajectory, it was determined that the maximum stress occurs in the heat shield body during the peak heating rate reentry trajectory (Abort No. 5), which for the 570 n mi nominal trajectory is referred to as Abort No. 5A.

The Abort No. 5A reentry, with a steeper reentry angle and a higher peak heating rate, results in a higher temperature gradient across the heat shield than the previous Abort No. 5 reentry. The outer graphite surface temperature increased approximately 18 percent and the minimum margin of safety at the most critical element, including deceleration loads, decreased. Table 4-8 summarizes the temperature, strains, and probability of heat shield failure for the Abort No. 5A reentry.

Table 4-8. Abort No. 5A Outer Heat Shield Body Data

Temperature ($^{\circ}\text{F}$)	2685
Strain (in/in)	0.00855
Strain including deceleration loads (in/in)	0.00866
Strain including deceleration loads (percent)	0.866
Probability of failure	8×10^{-2}

The analysis used the same analytical model and procedures as used for the 480 n mi trajectory. New node temperatures were computed for Abort No. 5A using the appropriate material properties, which vary with temperature.

The critical time for the Abort No. 5A reentry was determined as 13.75 seconds after initial reentry and was based on the fact that the ratio of temperature gradient to allowable strain was maximum at this time.

If the predicted strain is compared to the design curve allowable, a negative margin exists (compared to the 7 percent margin which existed for the Abort 5 analysis for the 480 n mi orbit). However, the design curve is deliberately conservative. When the mean value of the allowable strain, as determined by destructive and nondestructive data from the flight hardware are used for comparison, a positive margin of 28 percent exists.

4.4 TESTING

The Transit heat source was subjected to extensive testing in order to qualify it for flight. Its capability to successfully perform as required and withstand all normal and credible abort situations was evaluated. Both design verification and sequential testing were performed.

4.4.1 Design Verification Testing

Design verification testing on the Transit heat source, performed both by TRW Systems and Atomics International (AI), has been completed, with the exception of long-term PRD flow tests which are still underway. Tests performed by TRW included heat source thermal verification, thermal cycling, vibration, PRD flow, can verification, capillary flow, material compatibility, post-impact fuel containment, heat shield material qualification, simulated reentry (arc-jet), and drop tests. Tests performed by AI included capsule thermal verification, impact, overpressure, shrapnel and fire tests. Table 4-9 presents a summary of the test and the number and types of test units. One of the heat sources employed in the fragmentation and fire tests was previously exposed in the overpressure tests.

4.4.1.1 Capsule Thermal Verification

The purpose of this test series was to determine the actual capsule component temperatures, temperature drops, and thermal expansions during a variety of conditions.

Table 4-9. Design Verification Tests

NO.	DESIGN VERIFICATION TESTS	COMPANY PERFORMING TESTS	NUMBER AND TYPE OF TEST SPECIMENS	SSC FUEL SIMULANT REQUIRED
1	Capsule Thermal	AI	1 Capsule Plus Heater	No
2	Heat Source Thermal Verification	TRW	1 Heater Plus Heat Shield and Can	No
3	Thermal Cycling and Vibration	TRW	1 Heat Source	Yes
4	PRD Flow & Particle Retention	TRW	16 PRD's	No
5	Can Design Verification	TRW	2 Cans	No
6	Capillary Flow	TRW	2 Assemblies	No
7	Compatibility	TRW	28 Stacked Disk Assemblies	No
8	Drop Test	TRW	32 Models	No
9	Impact	AI	12 Heat Sources	Yes
10	Overpressure	AI	2 Heat Sources	Yes
11	Fragmentation	AI	3 Heat Sources	Yes
12	Fire	AI	1 Capsule 2 Heat Sources	Yes
13	Post Impact	TRW	2 Impacted Capsules	Yes
14	Heat Shield Verification	TRW	Material Specimens	No
15	Reentry Verification	TRW	5 Heat Sources	Yes

A test capsule was assembled from actual capsule components, typical in materials, dimensional tolerances, coatings, and treatments to the production capsule components. The capsule was modified to contain an internal electrical heater to simulate the radioisotope fuel and instrumentation to determine capsule component temperatures.

A schematic of the test capsule is shown in Figure 4-23. Component temperatures were measured both in vacuum and in helium.

Figure 4-23. Capsule Thermal Verification Test

The capsule thermal verification test results were compared with corresponding analytical values in order to verify the thermal model. It was found that when the thermal model was adjusted to reflect actual test conditions, very good correlation existed between analytical and test data. These adjustments included allowance for (1) uneven distribution of capsule gaps, (2) low thermal conductivity of heater core (as opposed to actual fuel), and (3) cooling of large diameter thermocouple joints by conduction along the wires and by helium convection. The analytical and test results are summarized in Table 4-10.

One additional test was performed to determine the effect of helium pressure on the conductance in the gaps between capsule components. Results of this test showed that at pressures above 1 psia the gap conductances remained nearly constant, as shown in Figure 4-24.

Table 4-10. Comparison of Capsule Thermal Test and Analysis -
801 Watts Internal Helium Pressure - 1 Atmosphere

Component	Analytical		Test As Measured (°F)
	Unadjusted (°F)	Adjusted (°F)	
Liner	1092	1162	1170
Strength Member	1015	1058	1040
Clad	941	941	940

Figure 4-24. Variation of Conductance with Pressure in Helium Gaps

4.4.1.2 Heat Source Thermal Verification

The heat source thermal verification test was conducted to verify and upgrade the thermal model.

The test unit consisted of an electrically-heated Inconel 600 capsule placed inside actual graphite heat shield parts and a modified can.

The heat source was mounted in a simulated cage and placed in a vacuum chamber as shown in Figure 4-25. A temperature-controlled simulated converter was

used for the tests requiring a converter. The helium backpressure was varied within the heat source to represent a range of possible helium release rates from the fuel.

Three steady-state environments were of primary interest:

- (1) heat source in 70°F with no converter, representing the heat source sitting in ambient air, (2) heat source in air with $T_{\text{conv}} = 483^\circ\text{F}$, representing the system in its prelaunch environment under the Scout shroud without air conditioning, and (3) heat source in vacuum with $T_{\text{conv}} = 752^\circ\text{F}$, representing the operating system in orbit.

Figure 4-25. Thermal Verification Test Setup

Test temperatures are shown in Figures 4-26 through 4-28 for an 800-watt heat source. This fuel loading was later changed to 827 watts and the temperature increased correspondingly. Calculated temperatures are also shown in parentheses for comparison. The maximum difference between the experimental and analytical values was 70°F. By making minor changes to the thermal model, the maximum difference was reduced to 50°F, resulting in a maximum deviation of 5 percent between theory and experiment.

4.4.1.3 Thermal Cycling and Vibration

The Transit heat source experiences different temperatures during assembly, storage, acceptance testing, launch pad operation, steady-state operation, reentry, and post-impact. During launch, ascent, and orbit injection, the Transit heat source will be subjected to sinusoidal and random vibration. The thermal cycle test program was performed to evaluate the behavior and interaction of the heat source components under conditions leading to launch. The vibration test program was performed to verify the structural adequacy of the heat source design to

Figure 4-26. Heat Source in Air
Power = 800 Watts, No Converter

Figure 4-27. Heat Source in Air
Power = 800 Watts, $T_{\text{Converter}} = 500^{\circ}\text{F}$

Figure 4-28. Heat Source in Vacuum
Power = 800 Watts, $T_{\text{Converter}} = 750^{\circ}\text{F}$

withstand the vibrational loads. A secondary objective was to determine the response, resonant frequencies, maximum stresses in the heat shield and can, and inertial load input from the heat source support structure and cage.

The test specimen consisted of a Transit heat source (S/N T-20) identical to the flight unit except that simulant fuel and porous carbon spacers were used. Also small holes were drilled in the can and heat shield body to permit instrumentation.

Thermal Cycling. Two separate thermal cycles, shown in Figure 4-29 were performed: (1) the heat shield subassembly, including the capsule, was thermal-cycled at typical conditions the cladding will experience prior to launch, and (2) the complete heat source assembly was thermal-cycled at the conditions which the can will experience under the same sequence of events. The tests were performed in the NRC vacuum furnace using a pre-programmed Data Trak curve follower.

The graphite and carbon parts did not crack and the emissivity coatings remained integral with no spallation or degradation

Figure 4-29. Thermal Cycling Test Schedules

Vibration Tests. Vibration testing was performed at room temperature. The test unit, in a heat source support structure/cage assembly on a converter baseplate, was mounted on an electrodynamic vibration table. It was subjected to a sequence of lateral-axis and thrust-axis acceptance sinusoidal, qualification sinusoidal, and qualification random vibration tests. The heat source design (with the exception of porous carbon spacers) survived vibration at levels nearly six times those of launch (200-g response measured versus 37-g maximum flight response expected). The platinized alumina coatings on the capsule, can, and indentations in the graphite end plugs did not spall, and the flow rates of the capillary tube assembly and PRD were unaffected. The can did not

plastically deform as indicated by measurements, visual examination, and strain gage data. The cladding, strength member, and liner did not deform and remained helium leak tight in all weld areas. The loads imposed on the heat shield body, end plugs, and thermal barrier bodies and sleeve were small and the parts did not crack, spall, or abrade. However, one of the porous carbon spacers did crack during qualification level testing. The design was subsequently modified to replace these carbon spacers with Poco graphite ones. Testing of the six sequential test units which included the spacers indicated that the graphite spacers did survive all vibration tests.

4.4.1.4 Pressure Relief Device Tests

Design verification tests were performed to verify that the pressure relief device will release the helium generated by radioisotope decay in a controlled fashion and will retain fuel particles for an extended period of time. Fourteen of the sixteen PRD verification tests were completed, and long-term (8000-hour) testing is still in progress with the remaining two units. These latter two assemblies have been successfully operating for approximately 7500 hours under constant pressure differentials and flow rates.

Eight of the PRD assemblies were tested to determine the fine retention characteristics. Absolute filters were utilized in these tests both upstream and downstream of the PRD's. After testing, the filters were subjected to neutron activation analyses. No thorium fuel simulant was detected in any of the downstream filters. A PRD fine retention test using powered $^{238}\text{PuO}_2$ performed at MRC indicated that the PRD released helium and retained live fuel particles well within required specifications.

The following conclusions were reached:

- The helium permeation rates of all PRD assemblies tested remained stable not only through short-term, but also during long-term testing.
- At high pressure differentials, relative to the normal operating conditions, the permeation mechanism of the nonselective PRD's changed from slip flow to viscous laminar flow. During this change, the effect of the pressure gradient on the permeation rate changed from a first order dependency (proportional

to ΔP) to a second order dependency (proportional to $(\Delta P)^2$). Thus, the PRD's have, in effect, a built-in safety valve. If the helium pressure in the capsule should suddenly build to a very high value during reentry, due to sudden (nonsteady-state) release of excess helium from the fuel, the flow increases to relieve the pressure.

- Experimental testing during this program has demonstrated the ability of the Transit PRD to retain both submicron thorium fuel simulant and live PuO_2 while passing helium at the flow rates expected during steady-state and nonsteady-state (reentry) conditions.
- There was no detectable chemical-metallurgical interaction between the tantalum and alumina utilized in fabricating the PRD in any of the assemblies which have undergone post-test examination.
- Post-test visual examination of specimens revealed no major change in appearance attributable to exposure environments.
- There is good correlation between the experimental helium permeation data and the theoretical equations for both slip and viscous flow in a porous medium.
- The greatest spread in helium permeation rates, at a constant pressure, occurred at the highest test temperatures of 2700°F . It is suspected that slight differential expansion between the metal and the alumina may have resulted in an increase of the flow area at the higher temperatures. At the low permeation rates experienced by these vents, a very slight area change would be sufficient to affect the flow rate. However, the flow rates taken at room temperature after the simulated reentry test series were unchanged from those measured prior to simulated reentry testing.
- The helium permeation rate through the nonselective Transit PRD's was relatively independent of temperature compared to the permeation rate through a selective device.

4.4.1.5 Can and Capillary Design Verification Tests

In the Transit heat source, helium, generated by decay of the fuel, flows through the pressure relief device then through the holes in the strength member and cladding end cap into the cavities of the graphite heat shield and the heat source can. The can serves several purposes: (1) it protects the graphite heat shield from oxidation during storage and ground operation, (2) it contains all helium and radon during ground operation, (3) it retains a backpressure of helium during space operation

to improve heat transfer and minimize heat source temperatures; and (4) it minimizes ablation during early reentry. Helium is retained during space operation by means of a capillary tube attached to the can. The capillary tube is hermetically sealed at the time of heat source assembly but is opened prior to launch. During space operation the capillary tube assembly allows helium to permeate to space thus avoiding a pressure buildup in the can, but at the same time retains sufficient helium backpressure in the heat source to reduce internal temperatures.

Since the capillary tube is opened before launch, the can was designed to contain the initial assembly helium pressure plus at least six months of helium generated by the fuel, assuming storage of the heat source during that period of time. If storage of the heat source is required for longer time periods, the fill tube on the can will be opened at Mound Laboratory to allow the stored helium to be released. The fill tube will then be rewelded to seal the can. This operation may be repeated, if necessary, several times.

Can Pressure Tests. In order to prove the validity of the can design, tests were conducted on two full-scale flight-type cans utilizing reference design weld joints. Emittance coatings were applied on the surface of the cans. To simulate the actual quantity of gas in the cans, a graphite cylinder equal in void volume to the actual flight assembly was inserted in each can.

Each can assembly was temperature-cycled under pressure to simulate events occurring during assembly, storage, helium venting, and launch pad operation. Each can was then heated to the maximum ground prelaunch temperature (990°F) and further pressurized for 4 hours to pressures greater than 10 percent above the actual helium pressure expected after six months of storage (68 psia, assuming 100 percent helium release from the fuel). In addition, one can assembly was pressurized to 200 psia at 70 and 990°F to determine the minimum safety factor available in the can design. This same can was also pressurized without a cage to an initial assembly pressure of 16 psia and heated to 650°F to simulate conditions which could occur after the can is welded closed but prior to assembly in the cage.

No helium leakage was found in either can assembly after thermal cycling and pressure testing. The high emissivity coating remained intact during the nominal pressure tests (68 psia), but some local peeling and cracking of the coating in the end cap gage bearing area occurred during the 200 psia test.

Capillary Flow Tests. Tests were performed on two capillary tube assemblies to verify experimentally the calculated helium flow rates through the capillary. Air was also used to obtain data on a different molecular weight gas. The flow tests were performed at the temperature and differential pressure ranges shown in Table 4-11. Figures 4-30 and 4-31 show the results of these flow tests. Agreement between test results of both capillary assemblies was excellent. A comparison of the test results with theoretical flow calculations was considered good.

Table 4-11. Capillary Tube Test Matrix

Type	Temperature ± 5°F	Differential Pressure (psia)	Backpressure
Helium	900	0.1, 0.5, 1, 2, 10, 35, 70	10 ⁻⁵ torr or 14.7 psi
	1000	2, 10, 35, 70	10 ⁻⁵ torr or 14.7 psi
	1100	2, 10, 35, 70	10 ⁻⁵ torr or 14.7 psi
Air	700	2, 10, 25	14.7 psi
	800	2, 10, 25	14.7 psi
	900	0.5, 1, 2, 10, 25	14.7 psi

4.4.1.6 Material Compatibility Tests

The contact between dissimilar materials of construction of the Transit radioisotope heat source initially raised questions of compatibility between adjacent components under operating and reentry conditions. Interdiffusion between the T-111 strength member and Pt-20Rh cladding, for example, could be detrimental if the extent were sufficient to seriously degrade the impact resistance of the strength member.

Figure 4-30. Experimental and Theoretical Helium Flow Rate Vs Differential Pressure

Figure 4-31. Experimental and Theoretical Air Flow Vs Differential Pressure

An investigation was conducted to verify that the Transit heat source would maintain its chemical and metallurgical integrity under operating and reentry conditions. Compatibility tests were performed with 28 sets of stacked discs of the capsule and heat shield materials arranged in an order to contain all the dissimilar material interfaces in the Transit heat source with the exception of the fuel. Fuel form compatibility tests were accomplished by LASL and MRC. The stacks were subjected to thermal environments simulating operational and reentry conditions. The test matrix consisted of three temperatures (1200, 1500, and 1900°F) and three durations (3, 30, and 271 days) to simulate operating conditions. Several supplemental tests at 2200 to 2800°F for durations of 60 to 120 seconds were also run to provide additional data. The extent of interaction at each interface was determined by metallographic and microprobe analyses.

Results of these analyses showed that no detectable interactions occurred at the pyrolytic graphite/platinized alumina/Pt-20Rh, the AXF-5Q/platinized alumina, or the Inconel 718/platinized alumina interfaces at temperatures to 1900°F and durations of 3, 30, and 271 days. Very slight interactions occurred at the Pt-20Rh/platinized alumina/T-111 and the T-111/Ta-10W interfaces at the highest test temperature (1900°F) only.

Based on the results of the 3-, 30-, and 271-day simulated operational compatibility tests, the following conclusions can be drawn:

- No measurable interactions between dissimilar materials occur at the PG/platinized alumina/Pt-20Rh, the AXF-5Q/platinized alumina, or the Inconel 718/platinized alumina interfaces.
- The maximum amount of interaction at the Pt-20Rh/T-111 interface for the anticipated operating temperature of less than 1500°F is sufficiently low to be acceptable. However, the interaction at the anticipated operating temperature can be completely eliminated or considerably reduced by use of a 0.005-inch thick platinized alumina barrier between the materials.
- The maximum amount of interaction at the T-111/Ta-10W interface during the heat source operating lifetime is sufficiently low to be acceptable.

- Carburization of the Inconel 718 in pressure contact with AXF-5Q graphite occurs rapidly at temperatures of 1200°F and above in a static environment. Evaluation of additional specimens currently being tested at 1000°F will determine whether a diffusion barrier is desirable between the Inconel 718 can and AXF-5Q graphite heat shield. Based on these test results, the localized areas of the sliding and screw plugs of the AXF-5Q heat shield, which are held under pressure contact with the Inconel 718 can, were coated with platinized alumina to provide diffusion protection.

Based on the results of the simulated reentry compatibility tests, the following conclusions can be drawn: The extent of interaction occurring at the Pt-20Rh/T-111 interface is sufficiently low to be acceptable at 2800°F. This interaction is completely eliminated by use of a 0.005-inch thick platinized alumina barrier between the materials. No interactions at any of the other material interfaces are of concern since they are negligible below 2800°F.

4.4.1.7 Drop Tests

Subsonic drop tests of the heat source were performed in order to ascertain the magnitude of the impact velocities. A total of 32 full-scale drop models, configured to achieve the proper weight, center of gravity, and moment of inertia of the actual heat source was tested.

Eighteen test models were similar in external geometry to the SSC heat source design. Nine additional models had a profile smaller than the nominal heat source design to determine the variation of impact velocity with the ballistic coefficient. The remaining seven test models simulated the basic design of the heat source containing the microsphere fuel form. Selected models had various degrees of corner rounding to determine the effects of ablation on impact velocity.

The drop tests were performed at the Sandia Range in Tonopah, Nevada, where radar, geophones, and contraves cameras were used to provide model trajectory information. The velocity, deceleration, altitude, range, dynamic pressure, Reynolds number, Mach number, ballistic coefficient, and drag coefficient were determined from these tests.

The results of the drop tests are presented in Table 4-12. The terminal velocities presented therein are for a Tonopah altitude of approximately 6000 feet. Because of the atmospheric conditions existing at

Table 4-12. Model Specifications and Terminal Velocity

Model Type	Model No	Weight (lb)	Diameter (in)	Length (in)	Corner Radius (in)	Terminal Velocity [*] , ft/sec		
						Radar	Time-of-Fall	Contraves
Dummy	D1	13 68	5	7-1/8	0	310	-	-
	D2	13 68	5	7-1/8	0	302	300	-
	D3	13 69	5	7-1/8	0	305	-	-
	1	13 55	5	7-1/8	0	311	306	295
	2	13 49	5	7-1/8	0	-	301	-
	3	13 54	5	7-1/8	0	291	291	291
	10	13 55	5	7-1/8	0	322	308	319
	11	13 56	5	7-1/8	0	309	298	-
	12	13 55	5	7-1/8	0	305	298	-
	4		5	7-1/8	1/8	Model came apart		
	5	13 47	5	7-1/8	1/8	325	317	-
	13	13 49	5	7-1/8	1/8	331	315	330
	14	13 48	5	7-1/8	1/8	313	304	-
	6	13 45	5	7-1/8	1/4	339	337	324
	7	13 47	5	7-1/8	1/4	345	339	345
	15	13 48	5	7-1/8	1/4	306	-	-
	16	13 46	5	7-1/8	1/4	340	336	343
	8	13 51	5	7-1/8	0 - 1/4 (Variable)	314	303	304
	9	13 50	5	7-1/8	0 - 1/4 (Variable)	324	316	-
	17	13 52	5	7-1/8	0 - 1/4 (Variable)	313	-	-
	18	13 53	5	7-1/8	0 - 1/4 (Variable)	311	316	-
	2	12 72	4	7-1/8	0	378	363	370
	20	12 75	4	7-1/8	0	348	350	-
	21	12 76	4	7-1/8	0	350	348	352
	22	12 70	4	7-1/8	1/8	376	363	380
	23	12 73	4	7-1/8	1/8	-	361	375
	24	12 72	4	7-1/8	1/4	364	349	-
	25	12 72	4	7-1/8	1/4	433	408	-
	26	12 71	4	7-1/8	0 - 1/4 (Variable)	420	391	427
	27	12 73	4	7-1/8	0 - 1/4 (Variable)	359	350	348
	3	12 98	5	8	0	334	323	-
	29	12 97	5	8	0	312	298	313
	30	12 90	5	8	0	326	305	316
	33	12 92	5	8	1/4	377	328	367
	34	12 97	5	8	1/4	372	321	-
	35	12 98	5	8	0 - 1/4 (Variable)	357	327	-
	36	12 92	5	8	0 - 1/4 (Variable)	Model came apart		

^{*} Corresponds to an altitude of approximately 6000 feet

Tonopah on the date of the drop tests, these values also represent 110 percent of sea level terminal velocity. The radar data were determined to be accurate to within 3 percent. The contraves film data are accurate to 5 percent and are probably low due to drift of the model which cannot be accounted for by a single contraves camera. Geophone (time-of-fall) data have an overall accuracy of 7 to 8 percent.

These data were used to select impact velocities for the impact test series, assuming a 1/8-inch corner radius as predicted by early

ablation analyses. The impact velocities defined by the radar data, which were the most accurate (and also the highest), were 331 ft/sec for the solid fuel design, 5 inches in diameter by 7-1/8 inches long, and 340 ft/sec for the microsphere design, 5 inches in diameter by 8 inches long. Since the weight of the heat source is less than that of the model (13.34 pounds, without can, versus 13.54 pounds), the maximum predicted sea level impact velocity, based on the final heat source design, weight, and the drop test results, is 297 ft/sec. This results in an impact test velocity of 327 feet which is 110 percent of the sea level terminal velocity. The impact tests were conducted at this velocity.

4.4.1.8 Impact Tests

The objective of the impact test series was to determine the impact response of 12 Transit heat sources impacted into granite at a velocity 10 percent in excess of the sea level terminal velocity as determined from the drop tests (Section 4.4.1.7). Test parameters were varied to investigate the most critical angle of impact and the effect of the maximum and minimum calculated impact temperature on the ability of the capsule to contain simulant fuel (SSCS). A second objective of this test series was to provide two impacted capsules for the post-impact oxidation tests.

All test capsules were fabricated using the same techniques and the same materials as those to be employed in the flight hardware, except that fuel simulant was used in place of the PMC fuel. Ten of the capsules were encased in heat shields and thermal barrier of the correct dimensional characteristics but of relatively inexpensive ATJ graphite. Two heat sources were preconditioned in an arc-jet simulating orbital decay prior to impact testing. These units employed a flight-type reentry heat shield. No Inconel 718 cans were employed on any of the units since the can would melt off during reentry prior to impact.

The impact tests were conducted at the Sandia Corporation Rocket Sled Facility. Each heat source was preheated; when the proper temperature was reached, the furnace was raised and the sled rockets fired, accelerating a granite block with attached catcher box down the track toward the stationary heat source. The catcher box, lined with aluminum

honeycomb covered with asbestos sheet, was used to prevent post-impact damage to the capsule.

The only variations among the 12 tests were the impact angle and impact temperature. The test parameters, both planned and actually measured, are found in Table 4-13.

A summary of test results is shown in Table 4-14. The Group I impact test series consisted of five capsules heated to 2400°F and tested at five different angles. All of the tests with the exception of the end-on test were successful since no SSCS release was detected. Some SSCS was released in the end-on test (T-33). In this test a section of a SSCS puck apparently pierced through the liner, strength member and clad, and was exposed on the impact surface. Direct survey detected 100 to 150 counts/minute and wipes showed activity on the impact face.

The Group II impact test series consisted of three capsules. All capsules were heated to approximately 2000°F. No thoria was detected external to the capsules after any of the tests. The test at 45 degrees (T-26) resulted in a tear in the clad, strength member, and liner about 0.80 inch long by 0.040 inch wide. Swipes of the surface indicated no activity, though some direct readings were obtained near the crack. The Group III impact tests were performed at the conditions previously shown to be most severe during the Group I and II tests. Two of the capsules (T-25, T-28) had been previously exposed to an orbital decay reentry heating cycle in an arc-jet. To obtain data on the effect of impact on the strength member weld, two of the units were to be impacted on the non-PRD end. One unit (T-24), however, impacted at too low a velocity due to a failure of a sled rocket. A third unit was therefore impacted on the non-PRD end. None of the capsules released SSCS. One capsule (T-25) tested at 45 and 2000°F had a hairline crack in the clad on the impact surface. Pinhole leaks in the clad weld area and a hairline crack through all three components were detected in T-28, tested end-on.

In summary, three heat sources sustained breaches in all three capsule shells (T-26, T-33, and T-28). A detectable quantity of SSCS escaped from capsule T-33 but none was detected from either of the other

Table 4-13. Transit Impact Test Matrix

	Test No.	Capsule No.	Impact Angle		Impact Velocity (ft/sec)		Impact Temperature (°F)		Impact End
			Planned	Measured	Planned	Measured	Planned	Measured	
Group 1	1	T-36	Side-On (0°)	Side-On (0°)	327 ⁽¹⁾	325	2400 ⁽²⁾	2400	PRD
	2	T-35	22.5°	22.5°	327	324	2400	2380	PRD
	3	T-34	45°	45°	327	330	2400	2400	PRD
	4	T-37	67.5°	67.5°	327	327	2400	2410	PRD
	5	T-33	End-On (90°)	End-On (90°)	327	325	2400	2435	PRD
Group 2	6	T-31	Side-On (0°)	Side-On (0°)	327	338	2000 ⁽³⁾	1987	PRD
	7	T-26	45°	45°	327	331	2000	1998	PRD
	8	T-9	End-On (90°)	End-On (90°)	327	329	2000	1995	PRD
Group 3	9	T-25 ⁽⁴⁾	45°	45°	327	338	2000	1990	PRD
	10	T-21	45°	45°	327	340	2000	2070	Non-PRD
	11	T-28 ⁽⁴⁾	End-On (90°)	End-On (90°)	327	332	2400	2410	Non-PRD
	12	T-24	End-On (90°)	End-On (90°)	327	287 ⁽⁵⁾	2400	2405	Non-PRD

(1) 110 percent of the maximum sea-level impact velocity as determined from drop tests.

(2) Maximum ground impact temperature of strength member predicted by reentry thermal analyses.

(3) Minimum ground impact temperature of strength member predicted by reentry thermal analyses.

(4) Preconditioned by arc-jet testing.

(5) This value low due to misfire of one rocket on the sled.

Table 4-14. Summary of Impact Test Results

Test No.	Capsule No.	Results of Radiological Survey After Test		Post-Test Capsule Volume (cm ³) (2)	Helium Leak Test Results			Remarks
		Direct	Smear		Clad	Strength Member	Liner	
1	T-36	0	0	(3)	Breach	(4)	(4)	Post-impact oxidation specimen
2	T-35	0	0	437 $\pm$ 5	Leak at star	(1)	Breach	Disassembled at LASL
3	T-34	0	0	480 $\pm$ 5	Leak at star	(1)	(1)	Disassembled at LASL
4	T-37	0	0	400 $\pm$ 5	Breach	No leak	Breach	Disassembled at AI
5	T-33	<1000 cpm ⁽²⁾	13 cpm ⁽²⁾	480 $\pm$ 5	Breach	Breach	Breach	Disassembled at AI
6	T-31	0	0	440 $\pm$ 5	Breach	No leak	No leak	Disassembled at AI
7	T-26	<150 cpm ⁽²⁾	0.8 cpm ⁽²⁾	364 $\pm$ 5	Breach	Breach	Breach	Disassembled at LASL
8	T-9	0	0	500 $\pm$ 5	No leak	No leak	No leak	Disassembled at AI
9	T-25	0	0	(3)	Breach	(4)	(4)	Post-impact oxidation specimen
10	T-21	0	0	370 $\pm$ 5	Leak at star	No leak	Breach	Disassembled at AI
11	T-28	0	0	366 $\pm$ 5	Breach	Breach	Breach	Disassembled at LASL
12	T-24	0	0	388 $\pm$ 5	No leak	No leak	No leak	Disassembled at AI

(1) Helium leak test not available at LASL, but no apparent breaches.

(2) See text for explanation of observations.

(3) Not available.

(4) Helium leak tests could not be performed because the capsules will be included in the post-impact oxidation tests.

X-rays revealed no breaches.

(5) Pretest capsule volume (calculated), including coating = 378 $\pm$ 2 cm³.

two capsules. It was concluded that the capsule design was marginal with respect to the safety criteria for granite impact.

4.4.1.9 Overpressure Test

The purpose of the overpressure test was to evaluate the capability of the Transit heat source to withstand the blast environment resulting from a Scout motor explosion during a launch abort. Two Transit heat sources containing SSC fuel simulant were preheated and tested simultaneously using 301 pounds of TNT to provide the required overpressures. The two different overpressure values were obtained by varying the capsule separation distance from the explosion center. The high yield specimen (20 percent TNT equivalent ground burst) was exposed to overpressure only. The low yield specimen (2 percent TNT equivalent ground burst) was subsequently tested in both fragmentation and fire environments.

The two test capsules (T-16 and T-17) were fabricated using the same techniques and the same materials as those to be employed in producing the flight hardware, except that fuel simulant was used in place of PMC fuel discs. Capsule T-16 was encased in a simulated thermal barrier and ATJ heat shield. The abort sequential unit, T-17, was encased in a pyrolytic graphite thermal barrier and a simulated outer heat shield. Both heat sources were sealed in Inconel 718 cans and were installed in cages and support structures. In addition, Unit T-16 was mounted on a simulated converter mounting structure of aluminum honeycomb.

The test was conducted in a test field at the Sandia Laboratories in Albuquerque, New Mexico. The T-16 heat source and simulated spacecraft were on one post 5.5 feet from blast center, and the T-17 heat source and another simulated spacecraft were on another post 11.8 feet from blast center, both 12 feet above ground level. Each heat source was heated using a clamshell-type heater which surrounded the support structure and cage prior to detonation of the TNT. Just prior to detonation the heaters were removed. The temperatures of T-16 (20 percent yield) and T-17 (2 percent yield) at the time of TNT detonation were 655 and 576°F, respectively.

No detectable damage to any of the capsule shells occurred for the 2 and 20 percent TNT equivalent yields at ground level. In the close-in unit (5.5 feet from center-to-center), the cage, can and heat shield were stripped from the capsule. However, the capsule was essentially undamaged. The sequential abort unit (11.8 feet center-to-center) retained the cage, can, and heat shield.

4.4.1.10 Fragmentation Tests

The purpose of the fragmentation test series was to demonstrate the ability of the Transit capsule to withstand the high velocity impact of typical fragments which could be generated by the launch abort of a Scout vehicle. The test series consisted of three tests, each employing a different type of fragment traveling at a velocity consistent with the source of the fragment.

The three test capsules, Serial No.'s T-10, T-17, and T-23 were fabricated using the same techniques and same materials as those employed in producing flight hardware, except that fuel simulant was used in place of the PMC fuel discs. T-10 and T-23 were encased in simulated heat shields and all-up Inconel 718 cans. T-17 was the launch abort sequential unit which had previously been included in the 2 percent ground level overpressure test (Section 4.4.1.9). It was included in the fragmentation test series in the "as tested" conditions. Thus, it was surrounded by the slightly damaged heat shield, can and cage.

The tests were conducted at the Sandia Laboratories' Rocket Sled Facility in Albuquerque, New Mexico. The heat source was in each test placed beside the rocket sled track. The fragment attached to the moving sled passed through a precut hole in a steel plate and impacted the heat source. The fragment was freed from the holder by the steel plate before impacting the heat source, thereby carrying none of the rocket sled momentum to the heat source. A 10-inch diameter pipe lined with styrofoam was used to catch the heat source after fragment impact in order to minimize or prevent further unrepresentative damage to the heat source.

In the first test Unit T-10 was impacted end-on by a titanium fragment at a velocity of 740 ft/sec. The fragment was 2.5 x 2.5 x 0.088 inch 6Al-4V titanium, weighing 0.088 pound. The nominal heat source pretest temperature was 1200°F. The fragment penetrated through the can and graphite and impacted the capsule. The cladding was breached but the strength member and liner were intact. The sequential abort heat source, T-17, was impacted edge-on by a simulated electronics package. The package was 4.2 x 4.4 x 5.1 inches, weighed 3.5 pounds, and was made up of printed circuit boards and potted transformers inside a 6061 aluminum box. The nominal test impact velocity was 310 ft/sec, and the nominal heat source temperature was 1200°F. There was no visible damage to the capsule. Test No. 3 (Unit T-23) was conducted using an actual spent igniter from the Scout fourth stage. The igniter weighed 0.86 pound and was made of aluminum. The nominal heat source preheat temperature and nominal impact velocity were 1200°F and 1190 ft/sec, respectively. There was no breach in either the clad or the strength member. Post-test disassembly revealed a slight crack in the liner on the impact surface.

4.4.1.11 Fire Tests

Three test conditions were selected to represent a variety of possible fire environments which could occur as a result of a launch abort of the Scout vehicle. The objective of the tests was to evaluate the ability of the heat source to survive these environments.

The test capsules were fabricated using the same techniques and the same materials as those employed in producing the flight hardware, except that fuel simulant was used in place of the PMC fuel. Heat source T-15 was encased in the PG thermal barrier, Poco heat shield, and the Inconel 718 can. The unit was preconditioned prior to the fire tests by dropping it from 75 feet onto a concrete surface to simulate the damage which might occur from a free-fall from the Scout vehicle. T-22 was tested as a bare capsule with no graphite or Inconel can. Finally, Unit T-17 was tested in the last of the sequential abort environments. This unit had previously been subjected to overpressure and fragmentation tests. It was included in the fire test in the "as tested" condition resulting from the previous tests.

The tests were conducted at the Sandia Corporation Coyote Canyon Test Facility. Heat source T-15, after the 75-foot drop, was preheated to 1200°F and placed between two blocks of propellant, each 10 x 12 x 24 inches prior to ignition of the propellant. The two propellant blocks were from the first and fourth stages of the Scout vehicle, respectively. Following a burn of approximately 4.5 and 3 minutes, respectively for each of the two propellant blocks, the capsule was completely intact except for the loss of some of the platinized alumina coating. Capsule T-22 was preheated to 1200°F and placed on its side on top of a block of Albol IIB (Scout first stage) propellant which was 10 x 12 x 24 inches. The propellant was ignited nearly instantaneously by the hot capsule contact. No instrumentation was included on the capsule. About 60 percent of the cladding remained on the strength member. The bottom part of the cladding had melted off. There was some corrosion of the outer surface of the strength member; the maximum penetration was about 0.04 inch. However, the remainder of the strength member and liner were integral. No fuel simulant was released. Finally, heat source T-17, in the condition resulting from the fragmentation test, was preheated to 1200°F and placed next to a propellant block with the heat source surface 6 inches from the uninhibited, vertical surface of propellant. The propellant block was 10 x 12 x 24 inches. The capsule remained completely intact.

Residual fires even as severe as direct contact between a bare capsule and the Scout first stage propellant of full-web thickness left at least one completely intact capsule shell and no fuel simulant was released in any test. The test environments appeared to present no real threat to fuel containment.

4.4.1.12 Post-Impact Fuel Containment Tests

The objective of this test was to determine the ability of the impacted heat source capsule to contain the radioisotope fuel in a typical post-impact environment.

Two Transit heat source capsules (T-25 and T-26), previously impacted with a resulting breach in the clad, were selected for this test.

Both capsules were placed in a representative soil, heated, and exposed to a stream of air. The soil used in these tests consisted of 5 percent limestone added to a blended mixture of one-third weight each of bentonite, feldspar, and Ottawa sand. Each capsule was placed in or on the soil contained in an Inconel boat, which in turn was placed in a 5-inch diameter closed tube heated externally by a clamshell heater.

A sample collector placed on the downstream end of the superalloy tube was removed daily to determine if any fuel simulant had been released. The sample was placed in a vacuum on a semiconductor alpha particle detector which was connected to a 400-channel pulse height analyzer. Decay products of Th^{232} emit their own characteristic α rays at various discrete energies. The pulse height analyzer separates these energies and an accurate determination can be made in identifying specific nuclides if sufficient quantities are present.

The sample collectors evaluated did not contain any detectable thorium. The two radioactive species identified in most samples were Bi^{212} and Po^{212} which are decay products of gaseous Rn^{220} . The Rn^{220} permeated through the oxidized components of the capsule and, together with its decay products, was partially absorbed by the collector sample.

In addition to radiation counting, samples were also periodically analyzed on the electron microprobe. Again, no thoria was detected. Additional samples were collected for analysis by titration and neutron activation techniques.

Transit capsule T-25 had been impacted at approximately 35 degrees resulting in a small crack in the clad on the impact surface. The standard 4-mil vent hole was incorporated into the clad before initiation of the test. The capsule was tested partially buried in soil at a temperature of 1450°F and an air velocity across its surface of 0.8 to 1 ft/sec. The test was terminated after 73 hours. The following key observations were made during the disassembly of this unit.

- 1) Both ends of the cladding were split open and a longitudinal crack ran from each end toward the center. A 1/4-inch region at the center prevented the clad from completely opening up.

- 2) The cladding interior surface was dark-colored; however, the cladding was ductile.
- 3) In areas where the cladding had split open early during the test, the liner and strength member had oxidized completely.
- 4) In other areas an estimated 0.050 inch of the strength member was oxidized to a light-yellow oxide, consisting of many thin layers parallel to the original metal surface. The other 0.040-inch thick layer of strength member was intact and only partially oxidized.
- 5) The liner was brittle under the partially oxidized strength member.
- 6) No thoria contamination was detected on the soil near the downstream end of the capsule.
- 7) The presence of the soil appeared to have had no effect on the ability of the capsule to contain fuel.

Transit capsule T-36 had been impacted side-on resulting in longitudinal tears in the cladding. It was tested at a temperature of 1150°F and an air velocity of 2 to 10 ft/sec while lying on top of the soil. The temperature of 1150°F corresponds to that expected for an actual capsule lying on the earth's surface. The test was terminated after 34 days. In approximately 3 days the first signs of oxidation were noted. By the fourth day extensive oxidation was observed which caused the cladding to split. The following observations were made during the disassembly of capsule T-36.

- 1) Both ends of the cladding were split open. A 3-inch long section of the clad, at the middle of the capsule, was split longitudinally.
- 2) Volumetric expansion, resulting from oxidation of the capsule components, particularly the fuel simulant, resulted in the capsule diameter increasing to approximately 5 inches. The length of the capsule, including the split ends, increased to approximately 7 inches.
- 3) Pieces of broken fuel simulant pucks extended approximately 1 inch from one clad end-cap region (downstream side). Fuel simulant pucks on the opposite end of the capsule were contained within the clad end-cap region.
- 4) The cladding interior and exterior surfaces were dark-colored; however, the cladding was ductile.

- 5) The exposed portion of the capsule consisted of a layer of light-yellow oxide identified as tantalum oxide. The upper 0.010- to 0.020-inch thick portion of this layer was a very fine powder. The remaining 0.070- to 0.080-inch thick layer consisted of many layers parallel to the original metal surface.
- 6) The fuel simulant had lost structural integrity and crumbled when lightly pressed.

In conclusion, volumetric expansion, resulting from oxidation of the refractory capsule components and especially the fuel simulant, caused the cladding to expand and eventually split open. Rapid and complete oxidation of the capsule components, except for the cladding, can therefore be expected within a few days (weeks at best) after impact. The appearance of MoO_3 downstream of the capsule indicated that fuel simulant oxidation proceeded within a few days. The presence of the soil in these tests appeared to have no effect on fuel containment.

4.4.1.13 Heat Shield Material

The scope of the verification tests of the Poco AXF-5Q graphite was expanded for the qualification and flight hardware compared to the design verification test units to increase the confidence level for material integrity during reentry. The expansion consisted of increasing the number of modes employed in the nondestructive ultrasonic "c" scan tests and conducting destructive tests on each billet and plate from which hardware piece parts were to be excised. The number of indications allowed in a particular heat shield component were minimized with some indications allowed in low stress areas only.

Some difficulties were encountered in obtaining acceptable Poco AXF-5Q billets and plates for the outer heat shield qualification test and flight hardware. Forty-eight billets were processed by Poco graphite with densities ranging from 1.67 to 1.78 gm/cc. The original density requirement was 1.80 gm/cc. However, since the tensile data obtained from AXF-5Q billets cores which had densities ranging from 1.760 to 1.795 gm/cc, used for the design verification and sequential test programs, exceeded the TRW design allowables at room temperature, 2000 and 3000°F, the minimum density requirement for subsequent AXF-5Q graphite was lowered from 1.80 to 1.76 gm/cc minimum.

Nondestructive Tests. The initial testing revealed the difficulty in detecting discontinuities on round billets due to orientation effects. Orientation effects did not affect detectability of discontinuities in plate material because the plates were ultrasonically scanned from at least three sides. An investigation into employing shear waves to overcome the orientation effects in the round billets was successful. Figure 4-32 is a photomicrograph of a typical defect detected in a billet core. This defect is sufficiently shallow that it would not have been detected by longitudinal wave scanning only, which was the only mode used for the design verification and sequential test billet materials. Thus a higher confidence level exists that all indications having a cross-sectional area greater than that produced by a 20-mil flat-bottom hole standard were detected in all billets and plates using echo "c" scan technique. Table 4-15 summarizes the indications predicted to be present in the qualification model heat shield components. The sound velocity variability of 4.9 percent in the qual backup body occurred at the body end closest to where the destructive test specimens were obtained. The destructive test results indicate that this billet, even with its variability, satisfies the Transit requirements.

Destructive Tests. Destructive tests were conducted on each piece of as-received material to ensure that the tensile and thermophysical properties meet design requirements. Tensile tests were conducted at room temperature, 2000 to 3000^oF; thermal conductivity tests were conducted from room temperature to 1800^oF. Where a low tensile property value which would not meet the design requirements was reported for a particular plate or billet, a retest was performed. Retests in all cases confirmed that the material would meet or exceed the design requirements. The design curves for stress analyses were taken as the lower limit for strain-to-failure and thermal conductivity and the upper limit for E, and coefficient of thermal expansion, whichever would yield a conservative design, and were obtained from a literature survey. The test results qualified the material for Transit use.

Figure 4-32. Defect Detectable by Only Longitudinal Shear Wave Mode (Mag 30X)

Table 4-15. Ultrasonic Nondestructive Test Summary

Assembly	Body P/N 405414-1		Screw Plug P/N 405415-1		Sliding Plug P/N 405413-1		Threaded Ring P/N 405412-1	
	C-Scan No. of Defects	Density Variability (%)	C-Scan No. of Defects	Density Variability (%)	C-Scan No. of Defects	Density Variability (%)	C-Scan No. of Defects	Density Variability (%)
Qualification	6	2.0	5	0.8	10	1.5	6	0.5
Qualification Backup	0	4.9	4	0.5	6	0.2	2	1.0

4.4.1.14 Reentry Verification Tests

As part of the Transit Design Verification Test Program, a series of arc-jet tests was performed to determine the adequacy of the Transit heat shield and thermal model. Existing arc-jet facilities could not duplicate all the possible environments which may be experienced by a Transit heat source because of its size. The tests discussed herein were, therefore, conducted to approximate a nominal orbital decay environment and a step heat pulse only, and were conducted in order to verify the analytical thermal model using these environments, and obtain some insight into heat shield response. The thermal model, once verified, was then used to predict the heat source behavior under actual expected reentry conditions.

The arc-jet tests were conducted at the NASA/Houston 10 mw arc-jet facility. Five full-scale heat sources were tested: four under simulated orbital decay and one under a maximum attainable thermal flux condition. Two of the orbital decay units included the Inconel 718 can; the other two did not and were tested as a preconditioning step prior to impact testing. The thermal stress unit was not canned either, in order to expose the graphite to the maximum step heat pulse possible.

All the models were instrumented with thermocouples on the interior components. The resulting temperature-time traces were then compared to values calculated using the Transit thermal model in order to improve its prediction capabilities. Post-test measurements of heat shield ablation permitted a check of the ablation analysis and were used to define the

amount of corner rounding which had occurred. This latter quantity is important in calculating the terminal impact velocity of the Transit heat source.

The test results coupled with subsequent analyses showed that the Transit heat source will survive an orbital decay trajectory under the worst conditions of side-on stable reentry. The fuel capsule will not be affected by the total heat input, and the heat shield thickness is adequate for the expected ablation. The heat shield, along the stagnation line, showed some increased ablation approaching the ends, but the corners remained sharp. Neither the dummy ATJ models used for facility check-out nor the full-scale Transit heat sources had any corner rounding on the stagnation line. Since the side-on stable position is the most severe, relative to heat flux in the corner region, the results indicate that the Transit heat source will not experience any corner rounding during reentry.

The Transit heat source thermal model results correlated well with the internal temperatures of the heat source during a reentry simulation after making some minor modifications. Using the model, the temperatures of the capsule clad were computed to be well below the clad melt temperature for the simulated orbital decay heat pulse.

Ablation of the outer graphite heat shield was predicted by the ablation model for the two orbital decay simulation units tested without the Inconel can. The ablation results, coupled with observations during the test and clad thermocouple response, all indicated that a CO reaction was obtained during subsonic flow tests achieved in the NASA/Houston arc-jet facility. Tests with the Inconel can indicated that the can retards ablation.

4.4.2 Sequential Testing

Sequential testing was performed using six simulant fueled heat sources to verify the containment capability of the heat source when exposed to the cumulative effects of various operational and abort environments. The sequential tests consisted of thermal cycling, capillary tube permeation, vibration, reentry arc-jet, drop/impact, and post-impact fuel containment tests. The sequence of the tests is shown in Figure 4-33.

Figure 4-33. Sequential Tests

4.4.2.1 Thermal Cycle Testing

The sequential thermal cycle test program was performed to evaluate the behavior and interaction of the heat source components under conditions leading to launch and to precondition six heat sources for subsequent testing. The test specimens consisted of six Transit heat sources (T-45 to T-50) which were identical to the flight unit except that fuel simulant was used instead of PMC fuel. Flight-type heat shields were employed in all units.

The heat sources were thermal-cycled under the conditions and temperatures which the can will experience prior to launch. All heat sources were held in a simulated cage and five of the tests were performed in an air furnace using a preprogrammed Data Trak curve follower. All fill tubes and capillary assemblies were sealed. Unit No. 3 (T-47) was thermal-cycled in the NRC vacuum furnace because a leak developed in the can fill tube during an attempted repair welding operation at MRC.

X-ray and visual examination indicated no adverse effects of the thermal cycles. None of the units were disassembled at this stage; all were readied for vibration testing.

4.4.2.2 Vibration Testing

The sequential vibration test program was performed to verify the adequacy of the heat source design to withstand qualification level vibration loads and to precondition four of the heat sources for subsequent testing.

The test specimens were those previously subjected to sequential thermal cycle testing. Unit No. 6 (T-49) had also previously undergone capillary permeation testing (Section 4.4.2.3).

Vibration testing was performed at room temperature in a manner similar to the Design Verification Vibration Test (Section 4.4.1.3). All test units, mounted in a heat source support structure/cage assembly which in turn was mounted on a converter baseplate, were subjected to a sequence of lateral and thrust axis vibration tests. Unit No. 5 (T-50) was subjected to X-, Y-, and Z-axis qualification level sinusoidal and random vibration. The others were all subjected to Y- and Z-axis flight level sinusoidal and random vibration.

Post-test evaluation and disassembly of Unit No. 5 (T-50) indicated no degradation of any heat source component except minor chipping of several SSCS fuel pucks. Visual and X-ray examination of the other units indicated no damage. Post-test evaluation of Unit No. 6 (T-49) is described in Section 4.4.2.3 below. All other units (Nos. 1 to 4) were prepared for further sequential testing.

4.4.2.3 Capillary Permeation Test

The capillary permeation test was designed to simulate the pressures and temperatures that the heat source could be subjected to on the launch pad in order to determine if some oxygen could back diffuse through the capillary after opening and adversely affect the T-111 strength member. Sequential Unit No. 6, which had been sealed at Mound Laboratory with a helium pressure of 16 psia at 70°F, was thermal-cycled in an air furnace in a sealed condition as outlined in Section 4.4.2.1. After completion of this test, the unit was checked for helium leakage; none was found. It was placed in an air furnace and the capillary permeation test performed. This test consisted of opening the capillary tube and thermal

cycling the unit over a 250-hour period as shown in Table 4-16 to simulate possible launch pad conditions.

Table 4-16. Capillary Permeation Test Matrix

Corresponding Actual Operation	Temperature $\pm 25^{\circ}\text{F}$	Rate ($^{\circ}\text{F}/\text{Min.}$)	Cycle Duration* (hours)
Prior to test	68		
Capillary opened	600	+ 6.5	1
S/C heat shield installed (no cooling)	914	+ 4.3	48
A/C** on	850	- 2.0	120
A/C off	914	+ 3.0	24
A/C on	850	- 2.0	54

* Does not include temperature transition time

** Air conditioning

Helium was detected flowing out of the capillary tube at the end of the 250-hour test. The heat source at this time was at the low temperature portion of the cycle (air conditioning on). The capillary tube was then resealed before reducing the temperature of the heat source to room temperature.

Since helium was detected at the end of the test, the pressure in the heat source can did not decrease below atmospheric during the temperature cycling, and no air back flowed into the unit through the capillary tube due to a pressure differential. This was substantiated by calculating, as discussed in Section 4.4.1.3, the pressure decrease in the can for the 250-hour thermal cycle performed. Theoretically, the pressure at the end of the test at 850°F would have been above 15.0 psia. Subsequent evaluation of this unit showed no detrimental oxidation of the heat source components due to this test.

4.4.2.4 Reentry Arc-Jet Tests

Arc-jet tests were performed on the four remaining sequential units in order to verify the adequacy of the heat source under simulated orbital decay reentry and in order to precondition the heat sources for the drop/impact tests. The test specimens were those previously exposed to thermal cycling and vibration environments.

Three of the units were preheated to approximately 1000°F prior to exposure to the simulated orbital reentry in the NASA/Houston 10 mw arc-jet facility. A simultaneous malfunction of the control thermocouple and clamshell heater caused the fourth unit to be preheated to temperatures in excess of the melting point of the can (2350°F). Thus, certain parts of the can were melted prior to the arc-jet test.

The four units survived the arc-jet tests without any deleterious effects. Since only one thermocouple was used in each unit to monitor preheat temperatures, no direct temperature data were obtained during the tests. Ablation data were obtained, however, and agreed well with both the predicted values and those obtained during the design verification tests.

The unit which experienced the overheat condition during the preheat phase of the test was disassembled and subjected to detailed post-test evaluation. Portions of the platinized alumina coating on the noble metal clad were found to spall upon handling. This was attributed to the long excessive overheat condition to which this unit was subjected. All other components were in good condition. The other three units were subsequently delivered for drop/impact testing.

4.4.2.5 Impact/Drop Test

Sequential Units T-45, T-46, and T-47 were hot drop-tested at the Sandia Corporation, Tonopah test range. The heat sources, which had previously undergone thermal cycle, vibration, and reentry testing were heated to a temperature of 2420°F $\pm$ 25°F aboard a C-54 aircraft just prior to being dropped. Because of weather-caused delays, one of the units (T-47) underwent three such heating cycles prior to being dropped. The oven was continuously purged with argon gas during the heating cycle

to inhibit oxidation of the graphite heat shields. However, sufficient oxygen apparently leaked in to cause some reaction with graphite. During the first heating cycle of Unit T-47, the unit was above 800°F for approximately 18 hours. In addition, it was in a lowered and ready-to-drop position for the final 10 hours. In this position air leakage into the furnace was sufficient to cause a hole to develop completely through the outer heat shield. This heat shield was subsequently replaced with a previously arc-jet tested heat shield. During the next heating cycle about 0.060 inch of shield surface was oxidized in the region 180 degrees from the stagnation line. The unit was rotated 180 degrees for the final heating cycle.

All of the units were dropped onto a concrete pad from an altitude of 15,000 feet above sea level, or about 9,500 feet above the ground. Impact velocity and data for the sequential test units are presented in Table 4-17 along with the previous drop test results. All of the units impacted in a near side-on attitude. Unit T-45 had a deep gouge in the clad resulting from secondary impact. A helium leak check showed that this gouge penetrated the clad. The clad on Unit T-46 had a hairline crack and the obsolete star area in the clad opened. Unit T-47 had a small longitudinal crack in the clad. Based on the results of previous impact tests the appearance of all three capsules indicated that the strength member was not breached.

The measured impact velocities were lower than nominal test for the Transit heat source (327 ft/sec). This was due to the lighter weight of the simulant fueled heat sources, roughening of the heat shield during arc-jet reentry testing, and the fact that the Transit nominal test velocity includes a factor for corner rounding, which did not occur during the arc-jet reentry test preconditioning of the units tested. Correcting the highest observed impact velocity among T-45 through T-57 (which is 270 ft/sec for T-46) for the difference in weight between an actual, fueled heat source after reentry (13.34 pounds - can removed during reentry) and the test heat source (12.54 minus ~0.25 pound for pretest oxidation) yields an expected impact velocity for an actual heat source at 6000 feet (the Tonopah test altitude) of 280 ft/sec. At sea level the corresponding velocity would be 253 ft/sec (correction factor of 0.9).

Table 4-17. Summary of Applicable Drop Test Velocities

Unit No.	Corner Radius (in.)	Weight (lb)	Release Altitude (ft)	Velocity ⁽¹⁾ (ft/sec)	Temperature (°F)
<u>Models</u>					
D1	0	13.68	15,080	311	R. T.
D2	0	13.68	15,010	300	
D3	0	13.69	15,050	306	
01	0	13.55	15,170	310	
2	0	13.49	15,080	301 ⁽²⁾	
3	0	13.54	15,070	297 ⁽³⁾	
10	0	13.55	14,960	323	
11	0	13.56	14,920	308	
12	0	13.55	15,100	315 ⁽⁴⁾	
05	1/8	13.47	14,990	324	
13	1/8	13.49	15,070	330	
14	1/8	13.48	15,070	317	
<u>Sequential Units</u>					
T-45	As is	12.41 ⁽⁵⁾	15,030	267 ⁽⁷⁾	2450
T-46	As is	12.54 ⁽⁵⁾	15,000	270-290 ⁽⁶⁾ , 263 ⁽⁷⁾	2440
T-47	As is	12.49 ⁽⁵⁾	14,935	275-290 ⁽⁶⁾ , 268 ⁽⁷⁾	2390

(1) Based on radar at 6000 feet

(2) Based on time of fall

(3) At 6400 feet

(4) At 7400 feet

(5) Ablation in furnace may have reduced weight by additional 1/4 pound

(6) Based on quick-look radar at 5335 feet

(7) Based on preliminary CZR data

Therefore, the impact tests on granite which were conducted at 326 ft/sec nominal velocity were actually tests at about 130 percent of expected sea level velocity rather than the planned 110 percent of expected sea level terminal velocity.

Although all of the units had minor breaches in the clad, the strength members remained intact and there was no evidence of thorium fuel simulant release.

4.4.2.6 Post-Impact Fuel Retention Testing

The purpose of the post-impact fuel retention test was to evaluate the fuel containment capability of the impacted capsules in a typical post-impact air environment. Two of the three sequential units (heat sources T-46 and T-47) were originally scheduled for testing at TRW. However, sequential Unit No. 3 (T-47) was delivered to LASL for testing using an internal heater whereas Unit No. 2 (T-46) was tested at TRW in a furnace.

The cladding of capsule T-46 had been breached in two areas by drop/impact testing. One breach, 0.062 inch long and 0.015 inch wide, occurred in the star area. The second breach, 0.066 inch long and 0.003 inch wide, occurred opposite the impact surface. The breach in the star area was sealed with Pt filler material, utilizing the TIG welding technique, because actual flight hardware will not contain the star cut in the clad. A 0.004-inch diameter helium release hole was then drilled through the cladding end cap to duplicate the current capsule design. The second breach was left in the as-impacted condition. The capsule was instrumented with three Pt-13Rh thermocouples attached to the external surface of the clad.

The facility utilized for this test was modified from that used for the post-impact fuel retention design verification testing. A 7-inch diameter flanged Inconel tube was used to contain the heat source. A glass view port was installed at the upstream side of the Inconel tube to facilitate sighting of the capsule. The view port was fitted with a movable shutter which was normally closed to eliminate condensation of volatile species on the glass. The soil was eliminated from this test because the previous design verification tests indicated that it did not affect the oxidation of the capsule. Therefore, the capsule was placed on 1-inch alumina supports positioned on the bottom of the Inconel tube.

After placing the capsule in the center of the Inconel tube, the assembly was placed within a clamshell-type heating unit which in turn was covered with insulating fire brick. The inlet and outlet air lines, thermocouple extensions, and heater connections were then attached.

The temperature of the capsule was raised to 1150°F and the air velocity was adjusted to approximately 1 ft/sec. Daily readings were recorded of the capsule temperature, heater voltage and current, air flow velocity, air temperature, and air humidity. In addition, a collector sample was removed daily from the Inconel tube to check for any evidence of thorium fuel simulant carried downwind.

Capsule T-47 was prepared for testing at LASL. A hole approximately 0.75 inch in diameter was core-drilled lengthwise into the capsule to a depth of two-thirds the overall length. The core drill contained sufficient molybdenum to replace that removed with the SSCS core and the core drill was left in place. A closed-end platinum sleeve was inserted into the core and welded to the capsule clad. A firerod heater was then installed in the sleeve for internal heating.

The test conditions at LASL were open field testing with the capsule resting on natural soil. The soil side capsule temperature was maintained at approximately 1150°F with a heater input of 433 watts. This was imposed by the limitation of the heater installation for this test. For the case of the live fueled capsule with 827 watts, the soil side temperature is predicted to be 1400 to 1500°F for this particular capsule with substantial portions of the emissivity coating removed. The LASL test setup was limited to visual observation of the results.

After 10 days of testing at TRW a slight bulge was noticed on top of the capsule in the area of the cladding breach. One day later the first visible signs of oxidation were detected in this area as the result of volumetric expansion of the oxides within the capsule. After 12 days, the original breach in the cladding had propagated longitudinally along the visible top portion of the cladding to a point approximately $3/8$ inch from the 0.004-inch hole. The crack at this time had opened to approximately $5/16$ inch in width.

After 18 days of testing, the first radioactivity was detected in the air exhaust filter connected to the hood. The two radioactive species identified at this time were Bi^{212} and Po^{212} which are decay products of the noble gas Rn^{220} . This indicated that portions of the refractory strength member and liner had oxidized sufficiently to allow the radioactive

gas to permeate through these porous oxides. At this time oxidation and resulting volumetric expansion appeared to accelerate more rapidly as an increasing amount of the internal capsule components were exposed to air flow.

On the 20th day of testing the cladding had split approximately 1.5 inches from the original 0.003-inch wide and 0.066-inch long post-impact breach towards the end of the capsule visible from the view port. The oxide at this time protruded approximately 1.5 inches above the top of the capsule. Three days later fuel simulant puck outlines could be seen as they began to displace the loose tantalum oxides on the top portion of the capsule.

After 27 days of testing the cladding section in the area of the 0.004-inch diameter machined hole split open exposing portions of the first ten pucks on that end. Each puck appeared to have expanded radially, to a progressively greater degree from the front to the rear as viewed through the sight glass.

Between the 30th and 40th day of testing the capsule continued to split apart longitudinally until the cladding had opened up exposing the top half of the capsule. The visible end of the capsule cladding during this period had split open in four sections. During the 47th day of testing approximately five pucks were displaced from the upstream end of the capsule and came to rest between the capsule and tube wall. This was apparently due to the earthquake in the Los Angeles area. Each of these pucks appeared to have broken into two pieces of equal size.

After 47 days of testing much of the tantalum oxide had fallen off or was displaced by the air stream and the upper half of the fuel simulant pucks were exposed; no further visible change in the capsule configuration appeared to occur through the remaining test duration of 13 days.

In the TRW test a sample collector placed on the downstream end of the Inconel container was removed daily to determine if any fuel simulant had been released into the air stream. The sample was placed in an evacuated container containing a semiconductor alpha particle detector which was connected to a 400-channel pulse height analyzer. Decay products of Th^{232} emit their own characteristic α and β particles at

various discrete energies. When an alpha-emitting sample is analyzed, the analyzer displays the alpha energy spectrum of the sample, i. e., a plot of the number of alpha particles detected versus channel number, with each channel corresponding to a certain alpha energy.

The sample collectors analyzed did not contain any detectable thorium. However, subsequent to the 18th day of testing, radioactivity was detected daily in these samples and in the filter which sampled the air exhaust into the hood. The radioactive species in each sample were identified as decay products of Rn^{220} . This indicates that the radon gas permeated through the oxidized components of the capsule and, together with its decay products was partially absorbed by the filter paper. No sampling was accomplished in the LASL test.

During the LASL test the clad split open after 3.5 days and molybdenum oxide crystals were visible. The heater burned out shortly thereafter and the test was discontinued. Direct alpha readings were recorded immediately after the test.

The initial rate of volumetric expansion of the oxides within both capsules was dependent on the size of the original post-impact breach. Gross oxidation occurred after the original breached areas were split open and an increasing amount of the internal components were exposed to air flow. This occurred after a few days in both TRW and LASL tests.

The initial splitting of the capsules occurred in a longitudinal direction which was the orientation of the post-impact breach.

Under the TRW test conditions only small amounts of thoria migrated from the capsule. The only radioactivity detected during this test was identified as daughter products of Rn^{220} , a noble gas which can permeate through the oxidized components of the capsule.

4.4.2.7 Summary and Conclusions of Sequential Testing

None of the prereentry conditions caused any structural change in the heat shield or capsule which could jeopardize orbital decay reentry and impact. Thermal cycle testing caused no degradation which was apparent after testing or which became manifest in subsequent testing.

The capillary tube permeation testing did not produce an influx of air into the heat source which might have caused degradation of the heat shield and strength member. Vibration testing caused no mechanical damage to the heat source components and only slight chipping of the simulant fuel pucks. Arc-jet reentry testing roughened the heat shield surface by a can/heat shield reaction but caused no cracking or corner rounding and ablation was well within acceptable limits. Drop/impact testing resulted in minor breaches in the cladding, as expected, but no strength member breaches or fuel simulant release occurred. Structural damage was less severe than previously observed from impacts on granite. Post-impact fuel containment testing demonstrated that the helium release hole in the cladding coupled with even slight tears in the clad is sufficient to limit the post-impact fuel containment capability to several days.

Minor discolorations which apparently occurred during welding were found on all capsule strength members and liners. Additional discolorations were found on the capsule members of Units T-49 and T-50 (which had the release hole in the clad). This could have occurred during thermal cycle testing due to outgassing of impurities from the heat shield. The preassembly outgassing procedure was therefore modified for all flight units to eliminate any possibility of this occurrence. In any event, the degree of surface oxidation observed (which resulted in slight bulk interstitial element content increase) is not sufficient to adversely affect the elevated temperature impact properties of the materials.

Generally, the platinized alumina coating on the claddings remained intact, but the coatings on Units T-47 and T-48 were found to be blistered and locally spalled. This was probably the result of unusually high (2400°F) long-term heating cycles caused by test anomalies which will not be encountered during Transit mission abort or normal operation. Spallation appears to be due to the differential thermal expansion between the coating and the cladding.

In summary, the complete Transit heat source design successfully satisfied all the test requirements of the prelaunch, launch pad hold, launch, and orbital decay reentry phases of the Transit mission. As

expected, minor breaching of the noble metal clad did occur upon impact on concrete. Post-impact fuel containment was limited to several days by the oxidation of the tantalum-based strength member and liner and the formation of volatile MoO_3 by oxidation of the cermet simulant fuel matrix. Such oxidation is, of course, enhanced by breaches in the noble metal clad, but would proceed even in an unbreached capsule since air can be admitted to the interior of the capsule through the small 0.004-inch venting hole in the clad. The duration of thorium containment in the post-impact oxidation environment has not been conclusively determined but it appears to be on the order of weeks at a temperature of 1150°F and on the order of days at a temperature of 1450°F . It is emphasized that a confirming experiment using actual PuO_2 fuel must be performed before drawing final conclusions about the immobilization of the actual fuel subsequent to post-impact oxidation.

5. HEAT SOURCE SUPPORT STRUCTURE AND CAGE ASSEMBLY

The Heat Source Support Structure and Cage Assembly (HS/SSCA) supports the Transit RTG Heat Source (HS) within the converter assembly. It was designed to accept a cylindrical HS weighing 14 pounds with external dimensions as follows:

	<u>Minimum</u>	<u>Maximum</u>
Diameter (inches)	4.5	5.0
Length (inches)	7.0	8.0

The components of the HS/SSCA are the Heat Source Support Structure (HSSS), the Heat Source Cage (HSC), and associated fastening devices.

5.1 HS/SSCA FUNCTIONAL REQUIREMENTS

Requirements imposed by thermal and structural considerations as well as the need to easily assembly and preload the HS, are given below.

5.1.1 Structural Requirements

5.1.1.1 Heat Source Positioning

The center of gravity of the combined HS and HS/SSCA must be within ± 0.050 -inch of the RTG axis.

5.1.1.2 Support During Launch

The HS/SSCA must support the HS in the Scout launch environment at the temperature experienced within the Scout heat shield.

5.1.1.3 Vibration

HS/SSCA resonant frequencies must occur outside the range of spacecraft resonance. The originally predicted spacecraft lateral resonant frequency was in the range of 50 to 100 Hz, and the axial resonant frequency was in the range of 100 to 175 Hz.

However, revised vibration requirements which were developed as the program progressed specify that the natural frequencies of the support structure and cage assembly be as given below. The natural frequencies cited are for the HS/SSCA with heat source installed and the total assembly mounted on the converter baseplate.

$$\text{Lateral} \quad 50 \text{ Hz} < f_{nl} < 90 \text{ Hz}$$

$$\text{Axial} \quad f_{na} > 150 \text{ Hz}$$

5.1.2 Thermal Requirements

The HS/SSCA must function during and after exposure to the following sequential thermal environments.

- a) One year of storage (HS and HSC assembly only) with the HS surface maintained at a maximum temperature of 600°F.
- b) Eight-hundred hours of RTG ground operation in air with a maximum heat source surface temperature of 860°F.
- c) Two-hundred hours of RTG ground operation under the Scout heat shield with a maximum heat source surface temperature of 910°F.
- d) Five years of RTG operation in space with a maximum heat source surface temperature of 1070°F and with up to 4 psia helium pressure contained within the heat source.

5.1.3 HS Preload Requirements

5.1.3.1 Relative Movement of HS Components

The HSC must provide a preload to the HS to prevent relative movement of the HS components. The preload applied must be sufficient to force the HS can into intimate contact with the graphite plugs (both sliding and threaded plugs) and force the graphite plugs into intimate contact with the capsule.

5.1.3.2 Helium Pressure Buildup

The HSC must maintain the required preload during any buildup of helium pressure in the HS can. This is accomplished by increasing the preload to compensate for the force due to the anticipated helium pressure. In the original design, the can was fitted with a burst disc which vented helium after 1.5 years of space operation. However, as the heat source design developed, the burst disc was replaced with a capillary tube opened prior to launch, relieving the requirement for the cage to counteract significant helium pressure during launch and space operation.

5.1.3.3 Thermal Expansion

The HSC must compensate for differences in thermal expansion between itself and the HS and maintain the required preload at launch. Preload is applied at a temperature lower than that experienced at launch and must be modified to account for the anticipated temperature rise.

5.1.4 Operational Requirements

The heat source is assembled into the HSC at Mound Laboratory. The assembly is performed at ambient room temperature. The HS/HSC assembly is then stored at Mound in the universal shipping container for a period of up to 1 year with a maximum temperature on the HSC preload bars not to exceed 600°F.

During storage the preload is not fully applied because the HSC is required to react only the force exerted by helium pressure within the HS can. The helium is vented every six months to hold the pressure below approximately 50 psia. The capillary tube venting device is opened prior to launch, dropping internal pressure below 3 psi during launch and space operation.

Over a total period of six months at APL and subsequently at WTR, the HS-HS/SSCA will be subjected to the following environments:

- 600 hours in vacuum within converter
- 800 hours in air within converter
- 200 hours in air within converter under Scout heat shield
- Remainder of six-month period in HS shipping cask

The can will be vented prior to shipment to APL. It is desirable to perform the 600-hour vacuum test as soon as possible after venting.

This will minimize helium-induced stresses in the titanium preload bars when high temperatures due to vacuum operation are experienced. Creep deformation of the preload members should be limited to less than 0.1 percent during that test. If the test begins more than three months after venting, it may be advisable to clamp a special device around the HSC, relieving the load in the preload bars during the vacuum test. The preload required for flight will be applied to the HSC at WTR no more than 1000 hours before actual liftoff.

5.2 HS/SSCA CONFIGURATION

5.2.1 HSSS Configuration

The HSSS is fabricated of Titanium 6Al-4V. It consists of six tubes and the six fittings which serve to interface between the HSSS and the HSC and between the HSSS and the converter. A top view of this structure, seen from directly above, has the shape of an equilateral triangle. The midpoints of the sides of the triangle are the locations for the fittings which interface with the HSC. The vertices of the triangle accept the fittings which attach to the mounting studs provided in the converter baseplate. Details of the original HSSS design are presented in Figures 5-1 and 5-2.

5.2.2 HSC Configuration

The HSC consists of the following primary elements:

- a) Two end caps of Inconel 718, one on each end of the heat source
- b) Three vertical tie rods of Titanium, 6Al-4V, spaced 120 degrees apart around the periphery of the heat source.
- c) Six end fittings attached to the tie rods and three center fittings, all of Titanium, 6Al-4V
- d) Twelve diagonally canted struts of Titanium, 6Al-4V, which join the tie rod end fittings to the center fittings located at the medial plane of the heat source cage

The tie rods serve to clamp the two end caps together and thereby provide a preload to the HS. This prevents rattle of HS internal components during launch. The tie rods also accept any stresses generated by the

Figure 5-1. Heat Source Support Structure

Figure 5-2. Heat Source Support Structure

buildup of pressure within the HS due to release of helium from the fuel. The diagonal struts interconnect the three vertical tie rods and serve as load paths between them and the center fittings. The diagonal struts receive the major portion of the HSC induced stresses due to the Scout launch environment. The center fittings are the interface points with the HSSS and receive the bolts for mounting the caging structure. Details of the HSC design are presented in Figures 5-3 and 5-4. Photographs of fabricated components are presented in Figures 5-5 and 5-6.

5.2.3 Mated Configuration

The mating of the HSC to the HS requires a preloading operation. This operation consists of tightening the bolts which hold the upper end cap to the cage assembly with the HS held between the two end caps.

Figure 5-3. Baseline Heat Source Cage

Figure 5-5. Developmental Heat Source Cage

Figure 5-6. Developmental End Cap Assembly

The mounting bolts for the lower end cap are previously bottomed-out. The HSSS is fastened to the mounting studs in the converter baseplate by means of standard nuts. The mating of the HSC to the HSSS, within the converter, is performed by torquing the HSC mounting bolts into the self-locking inserts provided as a part of the HSSS apex fittings.

5.2.4 Design Changes to the HS/SSCA

As a result of design analyses described in Section 5.3 and the test results described in Section 5.4, it was necessary to make some changes to the original HS/SSCA design. The final design HS/SSCA is shown in Figure 5-7 without a heat source. It incorporates the following changes:

Support Structure

- Tube wall thickness increased from 0.016 to 0.035 inch.
- Upper fitting has larger beam section for added stiffness.

Figure 5-7. Flight HS/SSCA

Cage (Figure 5-8)

- Preload tubes changed from 0.250 to 0.375 inch diameter and from 0.016 to 0.035 inch wall thickness.
- End fittings redesigned to accept 0.375-inch diameter tube and tapped for 1/4-inch, 28 UNF bolt instead of 10-32-UNF bolt.
- Center fitting split to provide damping. Footprint at SS interface increased and overall height decreased to stiffen the joint. Bolt hole diameter increased to accept 1/4-inch bolt.

End Caps (Figure 5-9)

- Cap redesigned from I-beam structure to truss structure for greater stiffness.

Other

- Bolt locking accomplished with tab washers instead of self-locking helicoil inserts.
- Stress-relief cycle introduced after welding in fabrication sequence.

In addition, thermal radiation shields were added around the support structure feet as shown in Figure 5-10. They consist of 0.005-inch aluminum sheeting folded into a reflective shroud and held in place by the support attachment to the baseplate. Molybdenum washers separate the aluminum shroud from the titanium structure. By lowering stud temperatures, the boots reduce conductive heat losses. Prior to fabrication of the final flight articles, the Ti-6Al-4V cage preload rods were changed from tubes (0.375-inch O. D., 0.035-inch wall) to solid bars (0.250-inch diameter). The change, made because of delays in delivery of the tubular material, increased the rod cross-sectional area 30 percent. Although there was a slight weight increase (0.04-pound for three rods), the accompanying decrease in stress reduces creep strain in the preload bars during vacuum chamber testing.

Figure 5-10. Heat Source Support Structure
Foot Radiation Shield

5.3 DESIGN ANALYSIS OF HS/SSCA

5.3.1 Preliminary Analysis

A preliminary analysis was performed to size structural members, to verify that stiffness and load requirements were met, and to approximate a minimum weight design. The HS/SSCA was considered as a space frame, pin connected at all joints, with a top and bottom beam system grasping the heat source and applying a preload. The failure criterion for the 3/8-inch diameter HSSS titanium tubes and the 1/4-inch diameter HSC titanium tubes is buckling under compressive load; the 1/4-inch diameter preload tubes are most sensitive to tension and the top preloader

Figure 5-8. Final Design Cage Assembly

BLANK

Figure 5-9. Final Design End Cap Assembly

BLANK

beams are sensitive to bending. Resonance frequencies of the system, considering it as a single lumped mass on an assemblage of springs, were predicted to be approximately 200 Hz in the thrust direction and 83 Hz in the lateral direction. Structurally, using a vibration amplification "Q" factor of 10 at resonance, all members were found to have positive margins of safety. High localized stresses were computed, however, at the connection between the HSSS and HSC. Thermal stresses, due to heating from the temperature at initial assembly to the temperature at the time of flight, were found to be of secondary importance.

5.3.2 Analytical Structural Models

5.3.2.1 AS 147 Program

The design was subsequently modeled, using the AS 147 dynamics program, to obtain a more valid description of the mode shapes and natural frequencies of the structure. Moments of inertia of the heat source were accounted for as well as the masses associated with the structure itself; end fixities were modeled for each member. The results of this analysis indicated a first lateral mode at 96 Hz and a first axial mode at 176 Hz with no significant coupling occurring between the torsional modes and either the axial or lateral modes. The most significant finding was that the apex points of the HSSS tended to "breathe" radially when excited in the thrust axis; thus the entire structure had a lower natural frequency than was predicted by the preliminary analysis. Subsequent testing of the STM/HS/SSCA verified that this was the case.* Consequently, a circumferential tie was introduced at the top of the HSSS to constrain the breathing motion, increasing the natural frequency of the HSSS in both thrust and lateral directions.

*The HS/SSCA subsystem was tested at TRW as part of the STM test sequence. Natural frequencies were in good agreement with the analysis; however, the amplification factor (Q) in the axial direction was found to be 25, two and one-half times greater than assumed in the analysis.

5.3.2.2 STRUDL Program

The computer model used for the dynamics analysis was expanded to more node points for a stress analysis using the STRUDL program. This more sophisticated analysis was basically in agreement with previous analyses; i.e., the structure was adequate for an amplification factor of 10 at qualification level thrust-axis vibration inputs above 175 Hz. High localized stresses were again indicated at the HSSS to HSC interface.

5.3.3 Preload

The preload is required to provide positive contact between the preloading system of the cage, the can around the heat source, the sliding graphite plug in the heat shield, the graphite body shield, and the internal radioisotope capsule. "Rattling" of the components during launch is then prevented. The amount of preload required is basically that required to press the can against the graphite plug, deform the Pt-20Rh clad through its initial gap so that it is in intimate contact with the T-111 strength member, and compensate for relative thermal expansion between the preload mechanism and the heat source. Thus, the magnitude of the preload is determined by:

- The pressure in the can at the time of launch (essentially a linear function of the time from initial heat source assembly or from last pressure bleedoff)
- The temperature change between the time of preload application and the time vibration environments are imposed
- The coefficients of thermal expansion of the materials
- The force necessary to deform the Pt-20Rh clad

The total preload therefore consists of three components to counter the effects of helium pressure, differential thermal expansion, and dynamic loads during launch vibration.

5.3.3.1 Helium Pressure Component

The initial helium pressure is based upon the closure conditions at assembly. The closure weld is completed with the heat source in a chill block. The average temperature of the internal helium is 200°F

and its pressure is 16 psia. When the chill block is removed, the average helium temperature and the pressure increase to 800°F and 31 psia with the heat source alone in air; to 950°F and 32 psia with the heat source in the RTG and the RTG under the spacecraft shroud; and to 1100°F and 38 psia in vacuum. For conservatism, the highest value (38 psia) was taken as the initial internal pressure.

It was then assumed, also conservatively, that 100 percent of the helium generated by fuel decay is released from the fuel matrix, increasing the pressure to 58 and 78 psia after 6 and 12 months, respectively. These increases are based on a helium generation rate of 3.3×10^{-5} cc/sec for an 850-watt heat source and a 1300 cc total void volume available for the helium.

The total force exerted by helium on the end cap of the heat source can was calculated using an end cap area of 20 square inches. Not all of this force needs to be reacted by the external cage, since it is partially reacted by the cylindrical body of the can. The cage reactive component, estimated at 55 percent of the total force, is 630 pounds at the 6-month design point.

5.3.3.2 Thermal Expansion Component

Between the time the preload is applied on the ground and the time it is needed to restrain the heat source at lift-off and during ascent, the magnitude of the cage-to-heat-source load is diminished by differential thermal expansion. The initial preload must be large enough to compensate for this relaxation, the magnitude of which is calculated using the extreme values of possible temperatures. On the ground the preload is assumed to be applied at temperatures characterizing the caged heat source without converter; in flight it is assumed that vacuum temperatures are reached. On this basis, the resultant preload component is calculated to be 705 pounds.

5.3.3.3 Dynamic Component

The dynamic load which must be reacted by the cage during launch vibration is one-half the product of the heat source mass times its maximum acceleration. The maximum heat source response during vibration at qualification levels (1.5 times flight levels) is 56 g. A 15-pound

heat source and a 75-g response are assumed, again conservatively. This requires a 560-pound reaction.

5.3.3.4 Total Preload

Summing the above components, the total preload is 1900 pounds or, distributed among three preload bars, 633 pounds per bar. The bolt torque required for this load is 63 inch-pounds, based on the observed torque/load relationship for this configuration. A preload of 70 inch-pounds was specified to compensate for uncertainties. This is well below the torque at which any component of the caged heat source will be overstressed. Helical inserts in the preload bars would be the first component to fail, and this will not occur until the loading reaches approximately 1700 pounds per bar (torque of 170 inch-pounds).

Although it had been planned originally to measure the applied preload by means of strain gages, a simple torque-wrench reading was eventually substituted. The torque-wrench accuracy was found to be sufficient for the allowable range of preloads.

5.3.4 Thermal Analysis

The calculated HS/SSCA operating temperatures are shown in Figure 5-11. The set of four temperatures at each point corresponds, from top to bottom, to the following conditions:

- In-air
- In-air within converter
- In-air within converter under Scout heat shield (no air conditioning)
- In vacuum

5.3.5 Creep in Cage Preload Bars

Creep in the preload bars of the heat source cage was investigated to determine under what conditions preload adjustments should be made to prevent excessive strain. Conservative assumptions, made to bound the possible creep strain, were (1) 100 percent helium release from the fuel, and (2) no stress relief as strain proceeds.

Figure 5-11. Temperature (°F) Vs Environment

Low-stress creep data were obtained for Ti-6Al-4V by extrapolating Larson-Miller parameter plots given in a December 1964 Martin Company report, "Astro Rocket Structure and Heat Shield Design Criteria." Temperature-time curves for fixed creep strain were then plotted for different stresses (Figure 5-12). Calculated preload bar temperatures (°F) under various conditions are as follows:

	<u>Ends</u>	<u>Center</u>	<u>Average</u>
In Air	346	223	285
In Air, Converter	533	459	496
In Air, Converter, Shroud	640	590	615

Figure 5-12. Creep Strain in Preload Bars

It is clear from Figure 5-12 that creep can be ignored at all of the above temperatures except those in vacuum. The in-orbit vacuum environment is of no concern, because strain in the preload bars can be tolerated after launch. However, strain during ground testing in the space chamber vacuum environment should be controlled. The shaded band of temperatures in Figure 5-12 covers a 40°F arbitrary increment above the calculated in-vacuum, average temperature. The 900°F upper limit is used for conservatism. Callouts along the 900°F line are for full and partial preload conditions. The full preload includes components for differential thermal expansion, helium pressure, and inertia forces during launch vibration.

The thermal expansion component disappears when in-vacuum temperatures are reached. Helium components are shown for 6- and 12-month storage periods, assuming 100 percent release in both cases. The inertia component is not required in vacuum chamber testing. In order to minimize creep, the preload should be decreased by the amount of this component before installation in the chamber.

If the preload is adjusted to take out the inertia component but is set to counteract 12 months of helium, 0.1 percent strain occurs in 4 days and 0.5 percent in 4 weeks. The latter strain corresponds to a preload bar elongation of 40 mils, about one turn of the preload bolt. At the lower temperature limit, 0.1 and 0.5 percent strains occur in 3 weeks and 21 weeks, respectively.

5.4 VIBRATION TESTS

As part of the STM tests, the HS/SSCA was subjected to low-level (1-g peak) resonance sweeps, and acceptance and qualification-level sinusoidal vibrations. The various sinusoidal sweeps were applied to the HS/SSCA in two configurations: hard-mounted to the exciter and mounted onto the converter baseplate. In each case an electrical heat source, simulating the weight and dimensions of the heat source, was utilized to ensure proper loading of the HS/SSCA. The HS/SSCA was instrumented with strain gauges (Figure 5-13) to monitor reactions of the various structural members. A tri-axial accelerometer was mounted

on the top of the electrical heat source to determine the amplification factor. The heat source was heated with an input of 800 watts bringing its temperature to 700^oF. The test series conducted for the initial HS/SSCA design is described in Sections 5.4.1 and 5.4.2, and the results are summarized in Table 5-1.

Figure 5-13. Instrumented Heat Source Support Structure and Cage Assembly with Electrical Heat Source

5.4.1 HS/SSCA Hard-Mount Vibration Test

The definition of axes for the hard-mount is as follows:

- X-Axis: The X-axis is a horizontal line passing through the projections of a cage tie rod and the HS/SSCA longitudinal axis.
- Y-Axis: The Y-axis is a horizontal line perpendicular to the axis at the longitudinal axis of the HS/SSCA.
- Z-Axis: The longitudinal axis of the HS/SSCA.

Table 5-1. Vibration Response of Initial HS/SSCA

		Original Design Requirement	Modified Design Requirement	Analytical Model Prediction	Test Results			
					Hard Mount	On Converter Base	STM (Hot)	
							Resonance Sweep	Modified Qualification*
Lateral Axes	Natural Frequency (Hz)	>125	> 50 < 90	96	85	80	-	-
	Amplification Factor	≤ 10	≤ 10	-	10	10	-	-
Thrust Axes	Natural Frequency (Hz)	>200	>150	176	175	165	158	135
	Amplification Factor	≤ 10	≤ 10	-	40	25	18	10

*33g, 100-175 Hz, 7g, 125-2000 Hz

5.4.1.1 X-Axis Vibration

The HS/SSCA was subjected to a resonance sweep, acceptance level vibration, and qualification level vibration in the X-axis. Assembly resonance changed from approximately 85 Hz to 78 Hz as the magnitude of impressed vibration was increased. The magnitudes of induced strains (and corresponding stresses) were within the capabilities of the structural members. The amplification factor (Q) of the structure ranged as high as 17 at the 1-g level, a value higher than anticipated.

5.4.1.2 Y-Axis Vibration

The HS/SSCA response in the Y-axis for resonance sweep, acceptance level vibration, and qualification level vibration was essentially equal to the X-axis response.

5.4.1.3 Z-Axis Vibration

The HS/SSCA was designed to meet a vibration specification which required an input of 17 g's in the frequency range of 100-175 Hz. Initial testing of the HS/SSCA revealed a very high amplification factor (Q=40) at the 180 Hz resonant frequency. A subsequent review of the spacecraft resonances resulted in a revision to the acceptance test to allow notching the input such that the level is reduced from 17 to 3.5 g over the frequency range 100-200 Hz. At the revised input level the resonant frequency of the HS/SSCA was found to be 175 Hz. The amplification factor remained at 40.

5.4.2 HS/SSCA - Converter Baseplate Vibration Tests

The lateral axes for these tests were displaced 15 degrees with respect to the X- and Y-axes defined for the Hard-Mount tests.

5.4.2.1 Lateral Axis Vibration

The HS/SSCA responded essentially as described above for the Hard-Mount X-axis configuration during lateral vibration except that resonant frequencies were slightly lower and amplification factors did not exceed 15.

5.4.2.2 Z-Axis Vibration

Z-axis vibration inputs were limited to a resonance sweep at 1-g and the modified acceptance level vibration as described for the Hard-Mount configuration. The resonant frequency was approximately 165 Hz.

The amplification factor was a maximum of 25 which again precluded testing at full acceptance and qualification levels.

5.4.3 Final HS/SSCA Vibration Testing

The HS/SSCA was redesigned as discussed in Section 5.2.4 and again subjected to vibration tests. The vibration specification was modified in the interim (see Table 5-1), reducing the input level between 150 and 175 Hz to the value specified for frequencies above 175 Hz. Results of the final tests are presented in Table 5-2.

During Z-axis tests with the STM baseplate, it was noticed that frequencies were lower than anticipated. This was attributed to the test history of the baseplate which had been subjected to a great number of stress cycles both during the STM tests at TRW and other tests at GE&ES. Preliminary Hard-Mount tests with the HS/SSCA mounted directly to its vibration fixture were then run.

A second baseplate was obtained and the Z-axis vibration tests rerun. The system natural frequencies were raised 10 Hz with the new baseplate with no appreciable change in amplification factor. The tests were taken to qualification levels. The first test at qualification levels was run at a sweep rate of 5 octaves per minute, double the specified rate. Upon rerun at the proper sweep rate, three cage diagonal struts failed at a frequency of 140 Hz. The test continued to 2000 Hz with no overt system breakup. Subsequent investigation revealed that the failed struts were pure titanium rather than the 6Al-4V titanium alloy specified. At the test temperature, the ultimate strength of pure titanium is less than one-half that of the alloy.

No further testing was performed on Engineering Model HS/SSCA's alone. However, in the course of the RTG flight qualification program, a properly fabricated HS/SSCA was tested to qualification vibration levels without failure.

5.4.4 Heat Source Cap Deflection Test

A heat source cap deflection test was performed to determine the magnitude of the heat source cap spring rate. The spring rate is of interest because it affects the cage preload. The measured spring rate

was 1500 pounds/inch. Deflection of the cap through 0.012 inch is required to ensure contact with the graphite plug. For the measured spring rate, 18 pounds will produce the required deflection. This is insignificant compared to the preload.

Table 5-2. Vibration Response of Redesigned HS/SSCA

	Mounting	Longitudinal Axis Input						Lateral Axis Input			
		Resonance Survey		Acceptance Test		Qualification Test		Resonance Survey		Qualification Test	
		Cold	Hot	Cold	Hot	Cold	Hot	Cold	Hot	Cold	Hot
Resonant Frequency (Hz)	Hard Mount	225	-	-	-	-	-	-	-	-	-
	On STM Baseplate	173	170	155	155	-	-	100	90	-	85
	On New Baseplate	183	183	-	172	-	150	-	-	-	-
Amplification Factor	Hard Mount	71	-	-	-	-	-	-	-	-	-
	On STM Baseplate	32	33	10	13	-	-	11	12	-	5
	On New Baseplate	31	33	-	20	-	11	-	-	-	-

6. NUCLEAR SAFETY

The safety philosophy for the Transit RTG is to maximize the probability for plutonium containment under all normal and accident environments. Specifically, containment shall be assured to the maximum extent possible during all ground operations and handling, transportation, storage, launch, ascent, space operation, reentry, and impact events. These events have been grouped into four chronological mission phases: prelaunch, launch, ascent, and orbit and post-orbit. Although complete fuel containment for the active life of the plutonium (approximately 1000 years) may not be realistic, the fuel form and heat source configuration and materials were selected within the current state of the art to maximize containment and plutonium immobilization times.

6.1 SUMMARY

The major nuclear safety tasks which were accomplished are presented in Table 6-1. Final Accident Evaluation, Probability Determination, Accident Environment Definition, Heat Source Response Evaluation, and Risk Evaluation have been performed for each of the Transit mission phases.

Table 6-1. Nuclear Safety Tasks - Completed

Mission Phase	Accident Evaluation	Probability Determination	Accident Environment Definition	Heat Source Response Evaluation	Risk Evaluation
Prelaunch	X	X	X	X	-
Launch	X	X	X	X	X
Ascent	X	X	X	X	X
Orbit and Post-Orbit	X	X	X	X	X

The method used in the Accident Evaluation was to follow through the normal sequence of events in each mission phase and identify the possible accidents and sequences of environments which could result from each accident. Probability Determination consisted of defining the occurrence probability associated with each of the event sequences identified in the Accident Evaluation. Accident Environment Definition included defining the specific accident environment conditions, such as impact velocity, fire temperature and duration, etc., that could be generated by the sequence of events identified in the Accident Evaluation. In the Heat Source Response Evaluation, the results of analyses and/or tests were used to evaluate the ability of the heat source to withstand the credible accidents and their environments. The final Risk Evaluation assessed the potential consequences of postulated fuel releases. All safety analyses, tests, and results were thoroughly reported in the Transit Final Safety Analysis Report (FSAR) in accordance with established AEC format.

6.2 NUCLEAR SAFETY CRITERIA

Specific safety criteria which are applicable to the Transit mission are identified below. Based upon these safety criteria, safety design standards were set forth for each phase of the Transit mission. These were based upon the potential environments resulting from the major accident situations (discussed later in Section 6.3.2).

Advancements in heat source technology have considerably extended predictable containment lifetimes. However, there are still a number of environments in which the probability of complete, long-term fuel containment cannot be accurately assessed. This is especially true in certain terrestrial and space environments which in themselves are not well understood and where the heat source can be exposed for time periods considerably in excess of the operational lifetime (5 years). In such environments, an assessment of the ability of the heat source to contain the fuel must be based almost exclusively on extrapolations and/or analyses rather than experimental data.

Specific terminology is used in the safety criteria listed below to designate the conditions under which fuel containment can be reasonably demonstrated by a combination of analyses and experiments as opposed to conditions where containment can only be predicted by analytical methods.

Where containment can be reasonably demonstrated using current analytical and experimental methods, the following phraseology is used:

"The heat source shall contain the fuel " or

"The heat source shall preclude inadvertent release "

Where containment can only be predicted by analytical methods, the following phraseology is employed:

"The heat source shall be designed to contain the fuel "

The latter terminology also implies that every effort be made in the selection of the heat source configuration and materials to obtain the longest term fuel containment possible.

6.2.1 Prelaunch Phase

6.2.1.1 Ground Handling

The heat source shall preclude inadvertent release of, or contact with, the fuel during all ground handling operations. Ground handling procedures shall be compatible with radiation exposure limits set forth in AEC-MC-0524.

6.2.1.2 Ground Operation

The heat source shall be capable of continuous operation in the RTG under sea level ambient conditions for up to 1000 hours. There shall be no degradation in any component which will affect heat source integrity in subsequent operational and abort environments.

6.2.1.3 Transportation

The heat source, in conjunction with the shipping container, shall preclude release of, or contact with, the fuel in both the normal and potential accident environments associated with ground transportation.

The shipping container shall also satisfy the requirements of the AEC as set forth in AEC-MC 0529 and the Department of Transportation (DOT) regulations, as applicable, for any potential shipping mode.

6.2.1.4 Storage

Fuel containment shall be assured in all normal and accident environments associated with extended storage of the capsule or heat source. Storage shall be in accordance with AEC-MC 0526. Direct radiation levels shall not exceed the limits specified in AEC-MC 0524. In addition, the heat source shall withstand the effects of 10 days of storage while in the RTG launch configuration. There shall be no degradation in any component which will affect heat source integrity in subsequent operational and abort environments. The capsule design, storage, and handling procedures shall prevent an accumulation of radon and its daughters in excess of the levels specified in AEC-MC 0524.

6.2.2 Launch Phase

The heat source shall contain the fuel during launch phase aborts of the Scout vehicle and for a reasonable time thereafter so that recovery can be effected.

6.2.3 Ascent Phase

Fuel containment shall be assured during normal ascent of the Scout vehicle. There shall be no degradation in any component which will affect heat source integrity in subsequent operational and abort environments. Also, fuel containment shall be assured during ascent phase aborts, i. e., failure to achieve any orbit, and throughout the subsequent reentry and impact environments specified below.

6.2.4 Orbit and Post-Orbit Phase

6.2.4.1 Orbit

The heat source shall be designed to prevent in-orbit release of fuel.

6.2.4.2 Operation

The heat source shall withstand the effects of normal operation in the RTG under space vacuum conditions for a period of at least 5 years.

There shall be no fuel release or degradation in any component during this time period which would compromise the subsequent intact reentry and intact impact capability of the heat source.

The fuel shall exhibit mechanical, thermal, and chemical stability within itself and with containment materials at system operating temperature for at least a 5-year period.

6.2.4.3 Reentry

The heat source shall contain the fuel during ballistic reentry and early orbital decay reentry, i. e., from orbits with lifetimes of up to 5 years. Also, the heat source shall be designed to contain the fuel during late orbital decay reentry, i. e., from orbits with lifetimes in excess of 5 years.

6.2.4.4 Impact

The heat source shall contain the fuel after impact at any impact angle upon a flat, non-yielding surface with a compressive strength equivalent to granite. It shall be capable of surviving this impact at a velocity of 10 percent in excess of the sea level impact velocity which the heat source can achieve following either abort or early orbital decay reentry as defined in heat source drop tests, Section 4.4. Also, the heat source shall be designed to survive impact at the velocity specified above following late orbital decay reentry.

6.2.4.5 Post-Impact

Following the impact associated with abort or early orbital decay reentry, the heat source shall be designed to provide the maximum long-term containment possible with a design objective of 1 year following impact. The fuel capsule shall be recoverable intact from the earth's surface or shallow water (300 feet or less).

However, in the event of ultimate fuel release, the resultant long-term concentration shall not exceed the levels specified in AEC-MC 0524 for the general population in uncontrolled areas.

Long-term fuel containment is not required in the event of deep sea water immersion.

6.2.4.6 Burial

In the case of earth burial, the heat source shall be designed to contain the fuel for the longest time possible. However, in the event of ultimate fuel release, the resultant long-term concentrations above the surface of the earth shall not exceed the levels specified in AEC-MC 0524.

6.3 HEAT SOURCE EVALUATION RELATIVE TO SAFETY CRITERIA

6.3.1 Normal Mission Evaluation

The normal mission was evaluated to identify the precautionary procedures required to ensure that no planned activities could directly or indirectly degrade the heat source ability to withstand accident environments should they occur. A summary of the reference operational procedures and related equipment is found in Section 7. Included below is a general discussion of the safety considerations which are applicable to the reference operational procedures for prelaunch activities and also a discussion of the heat source capabilities in relation to the normal mission.

6.3.1.1 Normal Prelaunch Activities

The major areas of interest during the prelaunch period, which include transportation, handling, storage, and testing are (1) radiation protection, (2) fire protection, (3) impact prevention, and (4) criticality control.

For transportation the heat source shipping cask has been designed such that total radiation levels in the environs of the cask and the truck, when the cask is in transit, are well within the levels which are applicable for the mode and type of carrier. Handling and testing procedures have been evaluated for operator time at expected dose levels to assist in limiting the accumulated dose to extremities and critical organs to below the allowable for each operator.

Preventative action is taken at all times to minimize the probability of the RTG being exposed to a fire. The normal procedures of keeping flammable materials away and suitable fire extinguishers present during testing and storage periods are followed. Marked escorts during transportation coupled with the use of extra driving care minimizes transportation accident fire probabilities.

Criticality control does not influence the prelaunch activities at all due to the low quantity of fuel compared to its critical mass. It is assumed here that the Transit heat source will not be exposed to any other radioactive materials during the mission.

6.3.1.2 Mission Safety

The heat source has been tested in a variety of conditions representing all phases of the normal mission, namely prelaunch, launch, and orbit operation. Descriptions of all heat source test results are found in Section 4.4. They include tests of individual components (i. e., pressure relief device (PRD), can, and capillary tube), tests investigating interactions between components (i. e., materials compatibility), and tests of assembled heat sources (i. e., thermal verification, thermal cycle, and vibration). Tests were also conducted using sequential environments of a mission (i. e., thermal cycle, vibration, arc-jet reentry, air-drop concrete impact, and then post-impact oxidation all on the same heat sources). The first two of these environments are applicable to a normal mission.

Prelaunch operations include handling, transportation, and tests. The operational tests of the PRD and heat source can and capillary, as well as the thermal cycle, vibration, and intra-capsule materials compatibility tests all demonstrated that normal environments constitute no threat to fuel containment and that the heat source components will not degrade during the prelaunch phase so as to effect the performance of the heat source during the launch or subsequent mission phases. A summary of the test procedures and results is found in Section 4.4.

The only significant environment during a normal launch and ascent consists primarily of vibration (launch phase aborts are covered in Section 6.3.2). The vibration tests verified that no degradation occurred in the flight heat source design even with the qualification level vibration spectrum which was 1.5 times the levels expected during actual flight.

In-orbit design considerations include long-term operation of the PRD and capillary tube and chemical compatibility among heat source components. Long-term PRD data have been obtained and no change in flow rate at a constant operating pressure differential has been observed.

It is concluded that the PRD will perform as designed during the mission life, both releasing the generated helium from the liner and retaining any fuel fines which may be present.

The conclusion is the same for the performance of the capillary tube in releasing helium from the heat source can and maintaining a sufficient backpressure of helium to insure adequate internal thermal conduction, provided that at least 10 percent of the helium generated within the fuel is released from the fuel matrix at operating temperature. A review was made of analytical and test data on the thermal conductivity of helium versus pressure and its effect on liner temperature versus helium release rate. The results indicated that when at least 10 percent of the helium generated in the fuel is released, the liner temperature will be less than 100°F above the normal operating temperature associated with 100 percent helium release from the fuel. Data available to date indicate that even though the release rate dips to slightly above 10 percent shortly after reaching operation temperature, a short time (days) later, the helium release rate starts to increase still further above the required level and it appears that it will eventually be asymptotic toward 100 percent of the generation rate. There is also evidence indicating that helium release rate increases with temperature so that there is a temperature self-compensating capability within the heat source, i. e., if a low helium release rate should occur, the conduction between capsule shells would be reduced causing an increase in fuel temperature which would result in a high helium release rate, etc.

The verification of chemical compatibility between capsule components, exclusive of the PMC fuel, was simulated in long-term tests (~1 year). The compatibility tests of heat source materials (Section 4.4.1) show that long-term operation at operating temperatures will result in little or no change in the chemical makeup of any capsule parts and virtually no change in mechanical strength during the mission lifetime. Initial short-term (2500 - 3500 hours) test data from LASL indicated that fuel/liner compatibility would be acceptable at the normal operating temperatures in orbit. However, recent test results obtained from the Transit Qualification Heat Source and several Pioneer capsules have shown the desirability of incorporations of Mo-50Re liner or an inner foil of Mo-50Re

determine what changes should be made in the fuel/liner interface to improve compatibility. A Mo-50Re liner or an inner foil of Mo-50Re have been suggested and are being more thoroughly evaluated by the AEC National Laboratories prior to incorporation. The mechanical strength of the capsule shells as a function of time are also under continued evaluation by LASL and MRC. The long-term effects of helium retention by the fuel (and observed migration to the grain boundaries), with respect to the creation of fines prior to or at impact, are also under study.

6.3.1.3 Ultimate Disposition

The nominal mission duration is 5 years; however, the orbital lifetime of the nominal orbit (570 n mi) is between 1000 and 2500 years. Mechanical forces on the heat source are negligible in orbit, since it is vented. If, as currently expected, materials degradation is not excessive at the nominal temperature, meteoroid puncture is the only mechanism for release of fuel in orbit. The probability of fuel release in the nominal orbit due to meteoroid puncture within 5 years is 3×10^{-6} .

6.3.2 Accident Evaluation

In order to evaluate the heat source response to various accident environments, a Failure Mode Analysis was undertaken to identify all credible accidents, to determine the probabilities of radiological risk resulting from system response to the accident environments, and to estimate the magnitude of risk.

The Failure Mode Analysis was initiated by establishing the sequence of events which can occur between encapsulation and the complete decay of the plutonium fuel. These events were separated into the four mission phases previously enumerated at the beginning of Section 6. The potential accidents which could occur within each phase were identified and accident networks were developed showing the sequence of events which result in exposure to an accident environment. Typical networks for the prelaunch phase of the mission are shown in Figures 6-1 and 6-2 for transportation and handling, storage, and test. (The number in each event box except the first on the networks is the conditional event probability and the probability of any sequence of events is the mathematical product of the conditional event probabilities

for the events in sequence.) These networks are discussed in great detail in the FSAR.

Several basic conclusions can be made regarding heat source response to the various accident environments. These conclusions are summarized below for each mission phase using test results coupled with information from the Failure Mode Analysis.

Figure 6-1. Transportation Network

6.3.2.1 Prelaunch Phase

Nearly all accident environments which could be generated as a result of prelaunch accidents are insignificant when compared to those of other mission phases. The possible impact velocities are low (up to 190 ft/sec) compared to terminal velocity at sea level after reentry (297 ft/sec). The possible fire temperatures (ranging up to about 2000°F)

are low compared to those possible during a launch phase abort residual fire (effective temperatures over 4000°F). Even though the prelaunch fire duration could be much longer, the equilibrium heat source temperatures are low compared to those reached in a launch phase abort.

An inherent characteristic of prelaunch environments is that, if they are adverse, their durations are limited and their locations are usually well known. In addition, the substantial protection of a specially designed shipping container is available during the mission phase.

6.3.2.2 Launch Phase

The major accident environments of launch aborts include overpressure, shrapnel, impact (covered in Section 6.3.2.3), and fire.

Overpressure created by the explosive effects of a 20 percent TNT equivalent ground burst of the Scout fourth stage have been shown to be the maximum expected from a launch phase abort of the Scout vehicle. Tests conducted in the Transit program (Section 4.4) have shown that the capsule will survive this environment with considerable margin of safety. It has been concluded that the probability of having a fuel release as a result of overpressure is exceedingly small.

Shrapnel velocities as high as 1100 ft/sec at the heat source location (after having passed through the entire spacecraft) have been predicted for the 20 percent TNT equivalence of the Scout fourth stage. The Transit fragmentation tests included a test with the Scout fourth stage igniter at 1175 ft/sec. The damage to the capsule included only a small crack in the liner. Other capsule members were intact and no fuel simulant was released. Complete capsule integrity was preserved in another fragmentation test employing an electronics box and only a slight breach occurred in the clad of a third test unit using a tank fragment traveling at a velocity of 730 ft/sec. It has been concluded that the probability of fuel release from a fragment impact is also extremely small.

The most severe residual fire environments were determined to be close proximity to or direct contact with fragments of the Scout first stage motor solid propellant which were full-web thickness. Tests were conducted which included heat sources next to and between fragments of the Scout first stage propellant which had the thickest web. In addition,

a bare capsule was tested resting on a chunk of first stage propellant of the full-web thickness and this most severe exposure to fire left the capsule with an intact liner and only a somewhat reduced thickness strength member. All simulant fuel was contained. It has been concluded, as was the case for the two previous launch phase abort environments, that the probability of capsule failure in the fire environment is vanishingly small.

It is finally concluded that no fuel release will occur as a result of any credible launch phase abort for the Transit mission utilizing a Scout vehicle provided adequate fuel/liner compatibility can be satisfactorily demonstrated at LASL and MRC prior to launch.

6.3.2.3 Ascent Phase

The three major environments which could result from an ascent phase abort are reentry aerothermodynamic heating, rigid-media impact, and post-impact oxidation. All ascent malfunctions of the Scout vehicle were investigated to define the most severe reentry conditions. Scout vehicle breakup, heat source thermal, ablative, thermal stress, and dynamic loading analyses in key areas were conducted. Arc-jet tests were conducted to verify the heat source thermal and thermal stress analyses techniques. The conclusions from the tests and all analytical results indicate that the heat source will survive all identified environments associated with reentry. Based on this condition, the probability of a high-altitude fuel release during reentry is considered to be vanishingly small.

One of the safety criteria deals with the heat source ability to survive impact. The 12 granite impact tests (described in Section 4.4.1) resulted in three heat sources with at least one crack in all three capsule shells. Two had been tested at impact temperatures in the end-on impact attitude, which is an unlikely impact angle for a flat-ended cylinder. Impacts essentially side-on are expected and so even a 45-degree impact angle (the impact angle of the third capsule with cracks) is not considered likely. Three heat sources were impacted on concrete after exposure to the complete sequence of environments including thermal cycle (pre-launch), vibration (launch), and arc-jet (reentry). All impacted essentially side-on (Section 4.4.2) and no cracks were observed in any capsule

shell other than the clad. Ascent aborts result in prompt reentry so that long-term aging effects on the fuel and/or capsule, if any, are not applicable to the ascent phase. The results of the safety effort indicated that the capsule is marginal with respect to the safety criteria for granite impact. However, the strength member when protected by a suitable liner is capable of completely containing the fuel for concrete impacts at terminal velocity based on the test data available.

The post-impact criterion indicates that the duration of containment after impact shall be maximized with a design objective of containment for 1 year. A temperature of 1150°F is the expected temperature for a capsule resting on the surface of the ground with the clad coating intact and 1450°F is typical of a capsule partially buried with most of the clad coating removed. The results of the post-impact oxidation tests (Section 4.4) indicate that containment durations on the order of weeks at a temperature of 1150°F and days at a temperature of 1450°F can be expected. It was concluded that although the design requirement was satisfied, the 1-year containment objective cannot be achieved at present.

6.3.2.4 Orbit and Post-Orbit Phase

The nominal orbit life at an altitude of 570 n mi is between 1000 and 2500 years. However, a short orbit could result in reentry in 5 years or less. Analyses and tests applicable to the ascent phase are also considered applicable to the short-orbit phase. Arc-jet tests verified orbital decay heat source thermal and ablative analyses techniques. The conclusion is that the heat source will survive the reentry environments of decay from short orbit.

The same impact and post-impact considerations that were outlined for the ascent phase are applicable to impact after reentry from a short orbit. Tests performed to identify any deleterious effects to the heat source materials during the nominal 5-year mission indicate that heat source material degradation is insignificant with the possible exception of the fuel liner as previously stated. Additional tests are in progress to resolve this question at LASL and MRC.

6.4 RISK EVALUATION

The study of the radiobiological risks associated with postulated fuel release events identified in the Failure Mode Analysis has been completed. The Risk Evaluation is discussed in detail in the FSAR.

7. GROUND SUPPORT EQUIPMENT

Ground Support Equipment (GSE) includes a heat source handling tool, RTG (or converter) handling tool, hoist sling, transport dolly, shipping and storage containers for the heat source and converter, and special hand tools. Also included as ground support equipment is the RTG operational checkout console, referred to as "Electrical Test Equipment" or "ETE".

7.1 HEAT SOURCE HANDLING TOOL

The heat source handling tool is used to remove the caged heat source from its shipping and storage container and to insert it into the thermoelectric converter. The tool is also used for the reverse operation. As shown in Figure 7-1, it consists of a stationary outer shaft with an attached finger support frame, and a rotatable inner shaft to which the engagement fingers are attached. The outer shaft has sling engagement eyes and holding grips. The inner shaft is pinned to the installation grip and release grip. The release grip controls a trigger which locks the inner shaft to the outer shaft when the engagement fingers are in their closed, or installed position. The tool is fabricated of stainless steel with the exception of the two fiberglass-laminate hand grips.

To install the heat source handling tool onto the caged heat source, the engagement fingers are rotated to the open position, as shown in Figure 7-1. The tool is fitted over the cage end cap with the tool's center pilot inserted into the end cap center hole and each leg of the tool's finger support frame straddling one of the end cap bolts. The engagement fingers are closed by rotating the installation grip in a counter-clockwise direction while the tool is supported by the holding grip. The engagement fingers then engage the end cap with one finger above and one below the end cap lower beam. When engagement is complete, there will be a definite "click" as the locking trigger falls into position, locking the outer and inner shafts together. To disengage the tool from the heat source, the release grip is pulled back and turned in a clockwise direction until the engagement fingers are in the open

Figure 7-1. Heat Source Handling Tool

position. Figure 7-2 shows the heat source handling tool installed on the caged heat source in the inner container of the heat source shipping cask, and shows the heat source handling tool attached to a sling.

Figure 7-2. Heat Source Handling Tool
and RTG Handling Tool Installed

A more detailed presentation of the sling and the heat source handling tool attachment is shown in Figure 7-3. Caution must be exercised to supplement the sling support with careful manual guiding of the tool in order to prevent inadvertent contact of the heat source with surrounding objects. In addition, when the tool is used to remove the heat source from a horizontally mounted converter, manual support is essential to counterbalance the cantilevered weight of the caged heat source.

Figure 7-3. Heat Source Handling Tool
Attachment Details

7.2 RTG HANDLING TOOL

The RTG handling tool is used to facilitate handling of the thermoelectric converter, with or without a fueled heat source. The tool is designed both as an aid to manual handling and as an interfacing mechanism between the converter and hoist sling.

The RTG handling tool consists of three like segments, each of which fits into converter bottom cover tiedown holes and a corner post hole, and which engage each other via latches to form a concentric ring

Figure 7-4. Heat Source RTG Handling Tool

around the upper circumference of the converter. Figure 7-7 shows the RTG handling tool attached to the converter while Figure 7-5 gives greater detail of the construction of a tool segment. The tool is fabricated of stainless steel, with teflon as a cushioning material on the standoffs and laminated phenolic grips on the hand handles.

To install the RTG handling tool (a two-man operation), the toggle mechanisms on two tool segments are

Figure 7-5. RTG Handling Tool Detail

opened until the upper and lower handles define planes which are mutually perpendicular. The cross bar locator pins are then inserted into RTG corner post bottom tiedown holes. Holes to be utilized are indicated by the three guide extensions on the top cover tiedown studs. The main bars are swung upward so that the toggle mechanism pins engage the 0.166-inch corner post holes and the upper hand-keyed standoff engages the top of the corner post (Figure 7-6). The two segments are latched together and the latch pin inserted. The emplaced segments are supported while the third segment is installed. The two remaining latches are closed and pinned, completing the installation. Once the tool is installed, it is positively locked to the converter.

Figure 7-6. RTG Handling Tool and Converter Contact Points

RTG handling tool removal is accomplished by essentially reversing the above operations. Caution must be exercised while installing or removing the handling tool from the converter. The tool must be engaged carefully to prevent contacting the converter at locations other than those specified in the procedure. Caution notes in the procedure flag the critical operations during tool installation and removal.

With the tool installed, the generator may be lifted and moved manually by two men. It may be carried in either a horizontal or vertical orientation or may be reoriented from one to the other. The generator may also be lifted and moved with a hoist and sling. Figure 7-7 shows an RTG model being lifted while in the vertical orientation, while Figure 7-8 depicts lifting in the horizontal orientation.

Figure 7-7. Lifting Converter with RTG Handling Tool

Figure 7-8. Lifting Converter with RTG Handling Tool and Sling (Horizontal Orientation)

7.3 HOIST SLING

The hoist sling is used for a number of RTG and heat source lifting operations. It is also used to lift the converter shipping container cover and the inner container of the heat source shipping cask.

The sling assembly consists of three parts: a three-legged flexible sling, a spreader bar for three-legged sling applications, and a spreader bar for two-legged sling applications. Figure 7-9 depicts the various sling configurations and indicates operations for which each configuration will be used. The sling is also shown in use in Figures 7-2, 7-7, 7-15, and 7-17.

The sling is used in a one-legged configuration with the heat source handling tool in a horizontal orientation, and in a two-legged configuration for lifting the inner container of the heat source shipping cask or the converter when horizontally oriented. During these operations the unused leg or legs must be attached to the hoist ring to prevent interference with the operation or damage to the RTG. The sling is attached to the object to be lifted (or to the spreader bars or hoist ring) by means of

CONFIGURATION C-1

USE: A) LIFTING TOP OF CONVERTER SHIPPING CONTAINER
B) LIFTING HEAT SOURCE HANDLING TOOL

CONFIGURATION C-2

USE: LIFTING HEAT SOURCE INNER CONTAINER

CONFIGURATION A

USE: LIFTING OF THE CONVERTER ASSEMBLY -
CONVERTER AXIS HORIZONTAL

CONFIGURATION B

USE: LIFTING OF CONVERTER ASSEMBLY -
CONVERTER AXIS VERTICAL

Figure 7-9. RTG Hoist Sling Configuration

clevises and self-locking clevis pins. The clevis pins have safety cables to insure they remain in the vicinity of the clevises when removed therefrom. The clevis pin is to be either installed in the clevis or in the holes provided in the spreader bar.

7.4 RTG TRANSPORT DOLLY

The RTG transport dolly is used for intrafacility movements of the heat source while installed in the inner container of the heat source shipping cask, and the RTG or converter while installed on the base of the converter shipping container.

The dolly (Figure 7-10) consists of a structural frame with four caster wheels. The two casters on the end opposite the handle are rigid, the other two are swiveled. The swiveled casters are equipped with wheel locks. The dolly holds the converter shipping container between the brackets located at the corners of the dolly base opening. These brackets contain holes for retention pins to lock the converter shipping container to the dolly. On the solid forward portion of the dolly's deck is a support fixture which holds the inner container of the heat source shipping cask. The inner container is also held in place with retention pins. The dolly, with the converter shipping container and the inner container of the heat source shipping cask both installed, is shown in Figure 7-11.

In addition to its use as a vehicle for intrafacility movement of the RTG or its components, the RTG transport dolly also serves as a work platform. It provides a holding fixture for the inner container of the heat source shipping cask during heat source loading and unloading operations, and it also accommodates the converter shipping container base.

7.5 CONVERTER SHIPPING CONTAINER

The converter shipping container is used to contain and protect the thermoelectric converter during storage, transport, and intrafacility movement. In conjunction with the transport dolly, as noted in the preceding section, the shipping container base may also be used as a work platform.

Figure 7-10. RTG Transport Dolly

Figure 7-11. RTG Transport Dolly with Converter Shipping Container and USC Inner Container

The base of the container (Figure 7-12) has a flanged ring which mates with the RTG mounting ring and clamp strap. The flanged ring structure is attached to the container base by isolators which reduce shock and vibration levels. Shock indicators record the maximum shock levels experienced by the converter in three mutually perpendicular axes.

Figure 7-12. Converter Shipping Container Mounting Ring Detail

Box channels on the container base accept fork lift tines. The box channels contain holes for installation of bars which guide the container cover when it is being removed or emplaced. Additional holes in the box channels receive the retention pins which secure the container to the transport dolly.

The container cover is a domed cylindrical structure. The cover has three lifting eyes near the outside edge of the dome which interface with the hoist sling. The bottom of the cover is flanged to interface with the mating flange on the base. The base flange is grooved to accept an "O" ring. When the cover is seated on the base and secured with the Marman clamp, as shown in Figure 7-13, a hermetic seal exists between the cover and base. The base has a pressure gauge and a connector fitting, with a shut-off valve, feeding into the container's enclosed volume. The fitting is used during evacuation and pressurization of the container. The container is first evacuated to 2 torr, or less, and then pressurized with argon to 4.5 ± 0.2 psig. The pressure gauge on the container base is used as an aid in this operation and, during storage or shipment, as a means of monitoring container pressure.

Installation and removal of the cover from the base must be performed with caution. When raised or lowered, the cover must not be allowed to contact the converter. Four cover guide bars, which fit into holes provided on the base, are attached to the base with clips. The bars are removed from their attachments and inserted into the holes prior to operations with the cover. The bars extend above the converter and are placed so that a misaligned cover will strike a guide bar prior to the opposite side striking the converter. Use of the guide bars is predicated on careful raising and lowering of the cover. The bars should serve as a visual indication of alignment and should not be depended upon to restrain a badly misaligned cover. Figure 7-14 shows the container cover being installed onto the base.

The container cover will not clear a converter with the RTG/converter handling tool installed. This handling tool must always be removed prior to container cover installation. The container cover must never be placed over a converter with the heat source installed, since maximum allowable converter in-air temperatures may be exceeded.

Figure 7-13. Converter Shipping Container

Figure 7-14. Installation of Converter Shipping Container Cover

7.6 HEAT SOURCE SHIPPING CASK

The heat source shipping cask, also referred to as the Universal Source Container (USC), is used for storage and transport of the radio-isotope heat source. It consists of three assemblies: an inner container, an outer container, and a carrier.

7.6.1 Inner Container

The inner container is a stainless steel cylinder divided into a top and bottom section as shown in Figure 7-15. The bottom has an exterior flange which mates with the top section and three interior flanges which simulate the heat source support structure engagement to the cage center fittings. The interior flanges have captive nuts which mate to the cage center fitting captive bolts. On each interior flange, a 6-inch rod is provided for guiding the top section onto the bottom section. The bottom section external flange contains a groove for a metal "O" ring, 12 one-half inch thru-holes, and 12 threaded thru-holes.

Figure 7-15. Top and Bottom Sections of Inner Container

The top section of the inner container has an external flange with 24 thru-holes, which mate with those on the bottom flange. Twelve are to accommodate bolts which attach the two sections together, and the other 12 continue the thru-holes on the bottom section flange to allow for circulation of coolant when the inner container is installed in the outer container.

The top section flange extends beyond that of the bottom section and mates with the USC outer container. There are eight captive bolts in this section of the flange and these mate to threaded holes in the step of the outer container. On the top of the inner container top section there are two lifting lugs, a plug and a valved connection fitting. The lifting lugs are provided as attach points for a hoist sling. The plug is not used for normal operation and is lockwired. The valved connection fitting is used to purge the inner container and to replace the air with helium. It is also used to sample the contents of the inner container for possible release of radioactive material. The valve handle will also be lock-wired during shipping and storage to prevent inadvertent release of the gas in the inner container. The inner container, with the top and bottom sections mated, was shown previously in Figure 7-11.

7.6.2 Outer Container

The outer container is an aluminum-finned casting with a removable top cover. The rings are 1/8-inch aluminum sheet welded to the casting. Twenty-four fins are spaced equally around the external surface of the container. The body of the outer container is 17 inches in diameter with a minimum wall thickness of 1.5 inches. The inner profile of the body has a step-down in diameter. This step provides the surface to which the inner container is attached. Below the step the wall thickness is 2.5 inches. The bottom is a 1-inch aluminum plate welded directly to the casting. The top is a machined 1-inch plate which mates to the machined surface of the top of the body casting. The top of the body casting contains an "O" ring groove and tapped holes to receive the top plate mounting bolts. When an "O" ring is placed in the groove and the top plate bolted to the body, the outer container is

hermetically sealed. The top plate has two lifting lugs to facilitate removal, and a valve and pressure gauge. The body is secured to the carrier baseplate by six bolts. The body is not removed from the carrier baseplate after once being attached except for repair purposes if it or the carrier baseplate has suffered damage.

The annular space between the inner and outer containers is pressurized with helium at 10 psig as a heat transfer medium.

Figure 7-16 presents a view of a prototype outer container. The holes shown on the top plate are for test purposes only and will not be included on user items. Figure 7-17 shows the inner container within the outer container. The inner container mounting bolts have been backed out and the hoist sling clevises attached to the lifting lugs to effect removal of the inner container.

Figure 7-16. USC Outer Container

Figure 7-17. Inner Container Removal

7.6.3 Carrier

The carrier provides a means of handling the USC and also provides radiation and thermal protection. The carrier's major strength member is a 38-inch square hot rolled three-quarter inch steel plate. Built up from the baseplate is an angle iron frame with an expanded metal cage and removable top of the same material. The baseplate is reinforced with two 7-inch box channels. In addition to stiffening the baseplate, the box channel arrangement provides an interface with fork lift tines for USC handling operations. Handles, swivel and fixed casters (the later with brakes) and a set of jacks are provided on the carrier

to facilitate manual handling operations. Warning and identification signs are permanently attached to the expanded metal cage. Figure 7-18 shows the carrier details. The outer container can be seen in place behind the expanded metal cage.

Figure 7-18. Universal Source Container

7.6.4 USC Usage

The caged heat source is installed into the USC with the aid of the heat source handling tool. A metal "O" ring is installed in the groove in the bottom section flange of the inner container and the top and bottom sections are mated. The inner container is evacuated and backfilled with helium, and is inserted into the outer container and bolted in place. A viton "O" ring is installed in the groove of the outer container flange and the top cover is installed. The outer container is filled with helium.

The USC is then closed by installation of the carrier cover. Removal of the caged heat source from the USC is essentially the reverse of the above. Radiation checks are performed at various steps during heat source installation and removal.

7.7 ELECTRICAL TEST EQUIPMENT

The Electrical Test Equipment (ETE) provides a complete instrumentation unit for functional checkout and performance monitoring of the RTG with either the radioisotopic or the electric heat source. Included in the ETE is a variable load for the thermoelectric generator, a convenient and fail-safe method of supplying power to the electrical heat source, a method of measuring the thermoelectric generator's internal resistance, an RTG simulator for self-check of the ETE, a thermocouple reference junction and a digital voltmeter for readout of temperature, voltage, and current sensor outputs. The ETE consists of two enclosures; one holds the RTG load resistor panel which is designed for use within 10 cable feet of the RTG, the other holds the remaining panels (including a second load resistor) and is designed for use within 40 cable feet of the RTG.

7.7.1 A10, Load Resistor Panel

The single panel cabinet contains an A10 Load Resistor Panel. This panel furnishes a resistive load for the RTG. The load consists of manually adjustable 1.5-ohm and 0.026-ohm resistors in series. The resistors permit load variations in the range of thousandths of an ohm to insure proper load matching. Panel A10 also includes a load relay, a taper load relay and a resistance measurement relay (all operated from Panel A1, described below), and a current shunt for measurement of total RTG load current in conjunction with a meter on Panel A1. Figure 7-19 shows Panel A10.

7.7.2 A1, RTG Output Panel

Panel A1, with the panels discussed below, is installed in a mobile rack which is connected with the RTG and the A10 cabinet. These panels are shown in Figure 7-20. Panel A1 provides a control switch for the load, taper load, and resistance measurement relays in Panel A10.

Figure 7-19. View of Load Resistor Panel (A10)

The taper load resistor, series-connected adjustable 10-ohm and 1-ohm resistors, is mounted on Panel A1. Meters for measuring RTG output voltage and current, in conjunction with the shunt in Panel A10, are included on Panel A1.

7.7.3 A2, Resistance Measurement Panel

Panel A2 furnishes a controlled AC voltage to the RTG through Panel A10, to measure the internal resistance of the thermoelectric converter. Regulated AC voltage is supplied through a variac to a step-down transformer. The transformer limits the available output voltage to approximately 7 volts AC. A magnetic circuit breaker is also included for safety purposes and limits the available output current to 5 amperes AC. The panel is decoupled for DC from the RTG by line capacitors. Panel meters indicate voltage and current imposed on the RTG, permitting calculation of RTG internal resistance. The current meter is magnetically coupled to the AC circuit. Provision for external current measurement is effected by a 10-milliohm shunt.

Figure 7-20. Front View of Rack Assembly

7.7.4 A3, Source Heater Input Power Control Panel

Panel A3 furnishes controlled excitation for the electrical heat source. Regulated AC voltage is supplied to the circuit through a 20-ampere variac. A 25-ampere magnetic circuit is used as an on-off switch as well as a safety mechanism. Another safety circuit consists of an adjustable meter relay which trips the circuit breaker if the electrical heat source temperature exceeds a set limit. Electrical heat source temperature is determined by a iron-constantan thermocouple with a single channel 0°C electrical reference junction (assigned the designation, Panel A14). An overtemperature condition is visually indicated by a panel lamp.

7.7.5 A4, AC Voltage Regulator Panel

Panel A4 is a GR-type 1581-ALR2 servo-driven, single-phase line regulator (± 0.5 percent) rated at 2.9 KVA, 117 vac, 60 Hz. Regulator power is supplied to Panels A1, A2, and A3.

7.7.6 A6, 26-Channel Reference Junction

Panel A6, a Consolidated Ohmic JR383-D- 0°C reference junction,

furnishes a 0°C electrical reference for the iron-constantan thermocouples used in the RTG flight instrumentation system. An additional 14 test thermocouples can be accommodated. The panel has a commutated output which may be stepped manually or cycled automatically. The channel being read is visually indicated on the panel.

7.7.7 A7, DVM Timeshare Selector Switch Panel

Panel A7 presents the output of Panel A6 and a manual 30-position, 3-pole switch (to which the output of A6 also goes) to the digital voltmeter, HP Model 3450A (Panel A8). Eighteen other measurements from Panels A2, A3, and A10, and from the panel voltage measurement flight instrumentation connector are monitored through the switch.

7.7.8 A13, RTG Simulator Assembly Panel

Panel A13, RTG Simulator Panel, is used during the calibration and verification of the other panels of the ETE and serves as an ETE troubleshooting tool. The panel has two 3.5-ohm resistors to simulate the electrical heat source, twelve 0.15-ohm resistors to simulate the RTG thermoelectric panels, internal resistances, and an adjustable power supply to simulate the power output of the RTG. Panel A13 is installed at the rear of the ETE mobile rack as shown in Figure 7-21.

The block diagram presented in Figure 7-22 depicts the connections between the ETE and the RTG, or ETG, for checkout purposes, the signal flow paths between panels and the connections to Panel A13 for self-check and calibration purposes.

7.8 HEAT SOURCE CAPILLARY TOOLS

Two tools are provided to open and close the heat source capillary tube. The capillary tube is opened prior to launch to permit controlled helium leakage. Extensive launch delays will require closure of the capillary tube to prevent excessive helium loss and possible back-flow of air into the heat source.

The tool used to open the capillary is a No. 55B Hunter hand tool modified for this operation. It is essentially a wire insulation stripping tool providing a uniform circumferential cutting force which results in a clean severance of the capillary tube. The closure tool is a No. 17-1

Figure 7-21. Rear View of Rack Assembly
Showing Panel and Simulator Only

Nicopress handtool which applies a circumferential force to reliably crush the tube near the end and effect a helium tight seal.

To open the capillary tube, its free end is carefully withdrawn through the hole in the heat source cage upper end cap with a pair of needle-nose pliers. The Hunter hand tool is used to score and break the tube one-half inch from its end. The exposed portion of the tube is then carefully pushed back into the end cap. For the closure operation, the

Figure 7-22. ETE Functional Block Diagram

procedure is similar except that the Nicopress tool is utilized to seal the capillary tube within one-half inch from its end.

7.9 SPECIAL HAND TOOLS

A set of special purpose hand tools are needed as an aid to RTG operations. These include torque wrenches, deep socket wrenches, and assorted pliers.