N73-10745

GULF GENERAL ATOMIC

GULF-GA-A12242

STUDY OF RADIOISOTOPE SAFETY DEVICES FOR ELECTRIC PROPULSION SYSTEM

VOLUME I - SUMMARY REPORT

by

G. B. Bradshaw, W. G. Homeyer, F. D. Postula,
and E. J. Steeger

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Prepared under Contract No. NAS 2-5891 by
GULF GENERAL ATOMIC COMPANY
San Diego, California

for

AMES RESEARCH CENTER

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

October 6, 1972

GULF GENERAL ATOMIC COMPANY
P.O. BOX 608, SAN DIEGO, CALIFORNIA 92112

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1. SUMMARY

Continued studies of a radioisotope-thermionic power supply for electric propulsion to the outer planets focused on the safety equipment to protect against dispersal of the isotopic fuel. The safety equipment is designed to be separated from the power supply and jettisoned early in the heliocentric phase of the mission, as soon as Earth escape is verified. Because of this, the mass of the safety equipment has little effect on the mass of the payload delivered to the target planet. The objective of this study was to improve the reliability of the safety equipment and increase safety margins at the expense of some increase in mass.

A new reference design was prepared for the 5 kW(e) thermionic power supply fueled with 44.2 kW(t) of $^{244}Cm_2O_3$. The safety equipment in this design is a passive containment system which does not rely on the operation of any mechanisms such as a launch escape rocket or deployment of parachutes. It includes: (1) a blast shield to protect against the explosion of the launch vehicle; (2) a combination of refractory thermal insulation and heat storage material to protect against a sustained launch pad fire; (3) a reentry body with a spherical nose and a large (2.44 m diameter) conical flare at the aft end to stabilize the reentry attitude and lower the terminal velocity in air; (4) composite graphite thermal protection to sustain the reentry heat pulse; (5) crushable honeycomb behind the nose to limit (to 200 Gs) the deceleration of the radioisotope source due to impact on land at terminal velocity; (6) a double-walled secondary containment vessel surrounding the isotopic capsules; (7) neutron shielding to reduce external dose rates; and (8) an auxiliary cooling system employing redundant heat pipes to remove the radioactive decay heat from the heat source and reject it to the surroundings or to a forced convection loop. Items 1, 2, 3, and 5 were added or modified during this phase of the study to replace the escape rockets and parachutes considered in the previous phase. The mass of the power supply is 724 kg at launch, 575 kg of which is jettisoned

after the Titan III-D/Centaur launch vehicle has boosted the power supply to an escape trajectory.

The potential for achieving the 36,000-hour lifetime required for some missions was also studied. The Cm_2O_3 capsules (at 2030°K surface temperature) and emitter heat pipes (at 1900°K) are expected to be the limiting components. The effect of oxygen released from the capsules by diffusion through the walls and through the helium vent plug was studied. It was shown that oxygen-induced sublimation of tungsten could remove as much as 0.2 mm from the surface of the fuel capsules in 36,000 hours, but that the loss rate would probably be much less. It was estimated that oxygen permeating the walls of the capsules and heat pipes could oxidize as much as 70 mg of lithium in each heat pipe. The calculations showed that there are large uncertainties in estimating oxygen effects from existing data, but that there is considerable flexibility in the reference design to minimize the effects of oxygen by use of gettering and other techniques.

This study showed the feasibility of protecting a 44.2 kW (1.3×10^6 Curie) source of ^{244}Cm from dispersal during launch accidents. Further work is necessary to establish the technology for long lived radioisotope capsules and emitter heat pipes. Once the technology is established and lifetime and performance limits are defined, it will be possible, with the help of detailed development plans and cost estimates, to make a more well-founded decision whether this system should be developed for some electric propulsion missions to the outer planets or the deep space environment.

2. INTRODUCTION

A recent study (Ref. 1) examined in preliminary fashion the suitability of an isotope-thermionic power supply for electric propulsion to the outer planets. The main feature was a separable entry safety container which was discarded early in the sun-centered part of the flight. The reference design case was a 5 kW(e) system using a Curium-244 heat source. Because of the preliminary nature of the study, various power levels and system configurations were examined in limited detail.

As the first phase of this study came to a close, frequent mission analyses were carried out using the latest values for estimated power and propulsion system masses. These calculations indicated that in spite of the weight growth during the first phase of the study, the system was still of interest for propulsive application. Furthermore, some of the mission parameters, such as payload, were rather insensitive to some of the propulsion system mass values such as the mass of the separable safety equipment. This meant that continued work could take advantage of these insensitivities to make the design generally more convincing from an engineering and safety viewpoint. Also, a more detailed look at a specific design concept with fewer alternatives could now be justified. As a result of changes in cost and subsystem mass that evolved during the first phase of the contract, such items as the nominal mission target and mission duration were changed for this second phase.

In the continued study effort reported here, the goal was to consider methods by which the separable safety equipment defined in the previous study could be modified to improve the aerospace nuclear safety aspects of the concept. Because these subsystems have low or negligible effect on the mission payload, considerable mass increases could be tolerated. Also, a more detailed examination was made of some of the lifetime and material compatibility problems as well as unique orbital and superorbital abort possibilities.

As a result of these studies, the reference design reported previously (Ref. 1) was modified substantially. The principal modification was to eliminate the launch escape rocket and parachute systems and replace them with thermal insulation and heat storage materials, a blast shield, an enlarged aerodynamic flare, and a thickened crushable honeycomb section in the nose of the reentry body. The new design is described in Section 3. It is based on the following mission and system constraints:

Radioisotope	^{244}Cm
Power	5 kW(e)
Mission Duration	36,000 hours
Launch Vehicle	Titan III D-Centaur

The full power level of 5 kW(e) is required during the first 10,000 hours and the last 10,000 hours of the mission, with low power required during the intervening 16,000 hours. The launch vehicle will inject the radioisotope electric propulsion system directly into a heliocentric trajectory with a hyperbolic speed on the order of 7.5 km/sec.

3. REFERENCE DESIGN

The reference design of the radioisotope power supply is shown in Fig. 1. The launch configuration shown here consists of three subassemblies. These are the radioisotope heat source, the thermionic converter assembly, and the safety equipment. The safety equipment is jettisoned after the power supply is launched into a hyperbolic trajectory relative to the earth. This leaves the flight configuration which consists of the heat source and converter assembly. The separation of the safety equipment from the flight configuration is illustrated in Fig. 2, and the flight configuration is shown in Fig. 3.

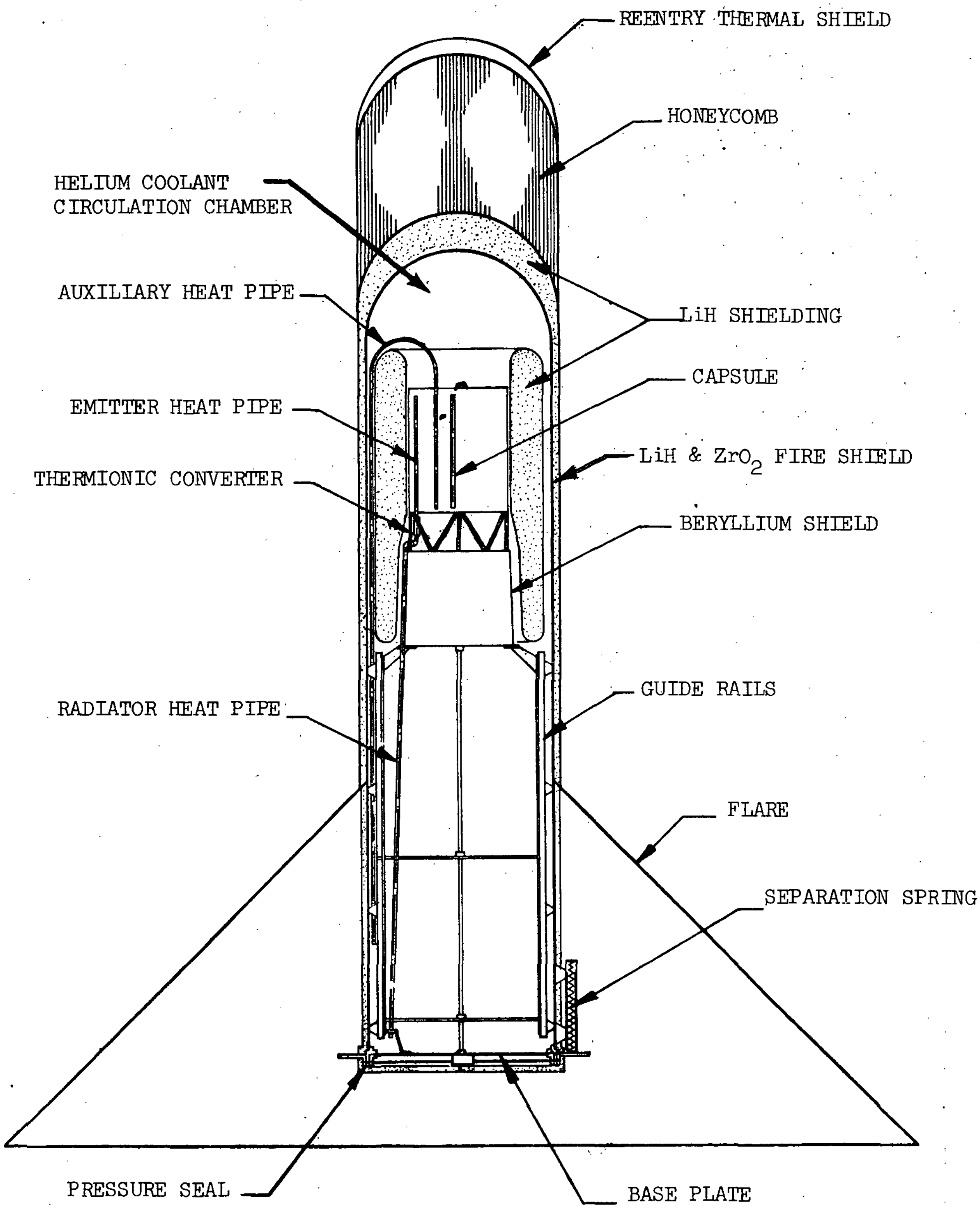
As shown in Fig. 1, the radioisotope heat source consists of the radioisotope capsules and the capsule holder. Heat is transferred from the capsules to the emitter heat pipes in the converter assembly by thermal radiation. The capsule holder is covered with thermal insulation to reduce radiant heat leakage. The thermionic converter assembly consists of:

1. The 69 emitter heat pipes (lithium in tungsten) which remove heat from the radioisotope capsules and concentrate it in the thermionic converters

2. The 69 thermionic converters, which convert a portion of the isotopic heat to electrical power

3. The electrical transmission lines which transmit the electrical power from the thermionic converters to the power conditioning equipment

4. The 69 radiator heat pipes (potassium in niobium 1% zirconium) which reject the waste heat, not converted to electricity, to space



a) OVERALL VIEW

Fig. 1. Radioisotope thermionic power supply - launch configuration

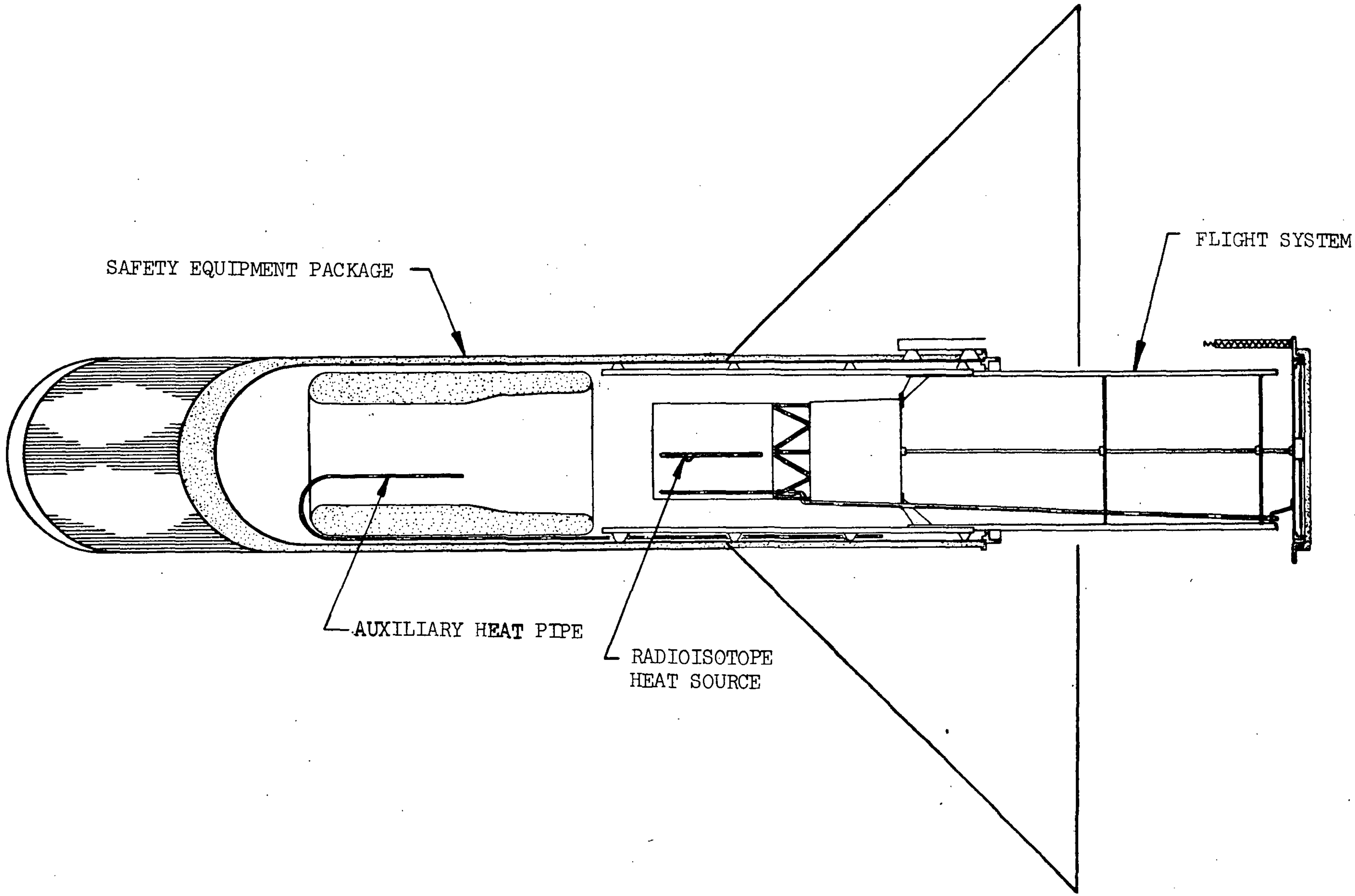


Fig. 4.2. Separation of safety equipment from flight system

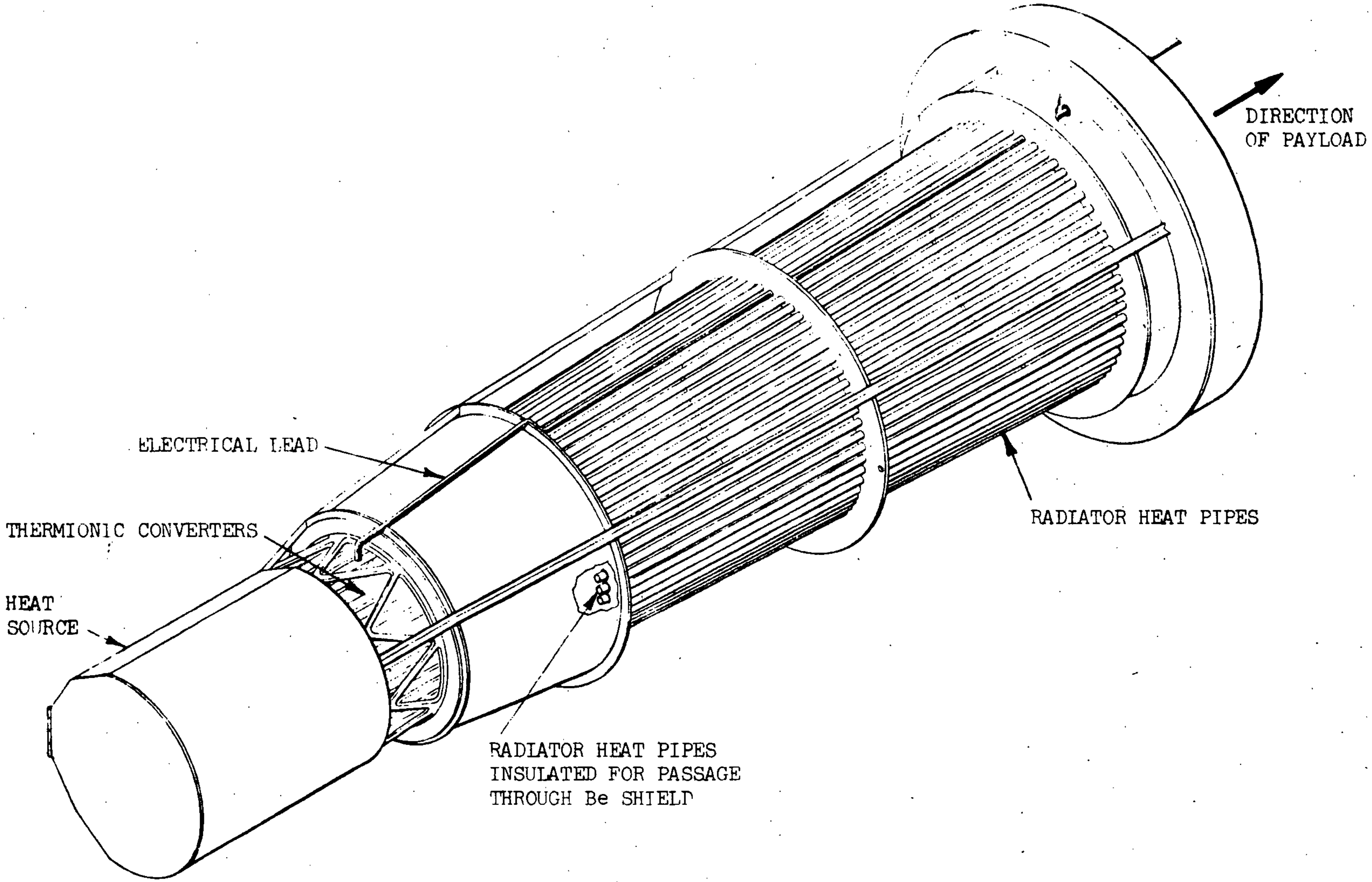


Fig. 3. Flight configuration

5. The beryllium neutron shield which protects sensitive components in the payload (2 meter dose plane diameter) and power conditioning equipment from radiation damage

6. The extendible boom (not shown) which moves the radioisotope power source away from the payload and power conditioning after launch

7. The power conditioning equipment (not shown) which converts the 15 V output of the thermionic power source to the levels required by the electric thrusters.

The safety equipment includes:

1. The 408 auxiliary cooling heat pipes (potassium in stainless steel) which remove heat from the radioisotope capsules prior to and during launch

2. The containment shell which provides secondary containment of the radioisotope and protects refractory metal components from oxidation in the earth's atmosphere

3. The helium container and circulator to remove isotopic heat prior to launch

4. The lithium hydride neutron shield to protect persons from neutron radiation prior to launch or following launch aborts

5. The graphite reentry shield to protect against aerodynamic heating following a high altitude launch abort

6. The aerodynamic flare which reduces the ballistic coefficient of the power source and assures a stable reentry attitude

7. The zirconia felt thermal insulation and lithium hydride heat

absorber which protect the secondary containment shell and auxiliary cooling heat pipes from overheating during a launch pad fire

8. The impact energy absorber which protects the secondary containment shell and other safety equipment from damage during impact on earth's surface.

The separation of the safety equipment from the heat source and converter assembly is illustrated in Fig. 2. This separation is performed after the power source has reached a hyperbolic trajectory and there is no longer danger of reentry into the earth's atmosphere. Actuation of separation of the safety shell from the flight configuration is by retraction of three pins at the periphery of the aft bulkhead. When the pins are retracted, a separation force is supplied by three coil springs which are stored within the flare volume. During separation, the flight system is guided along three rails so no side force is applied at the moment of separation. The auxiliary cooling heat pipes slide out from between the radioisotope capsules and the emitter heat pipes as the heat source is withdrawn from the safety equipment. When the heat source is fully removed, a hinged, insulated door closes over the end from which the auxiliary heat pipes have been withdrawn, and the temperatures of the heat source and converter assembly rise to their normal operating conditions.

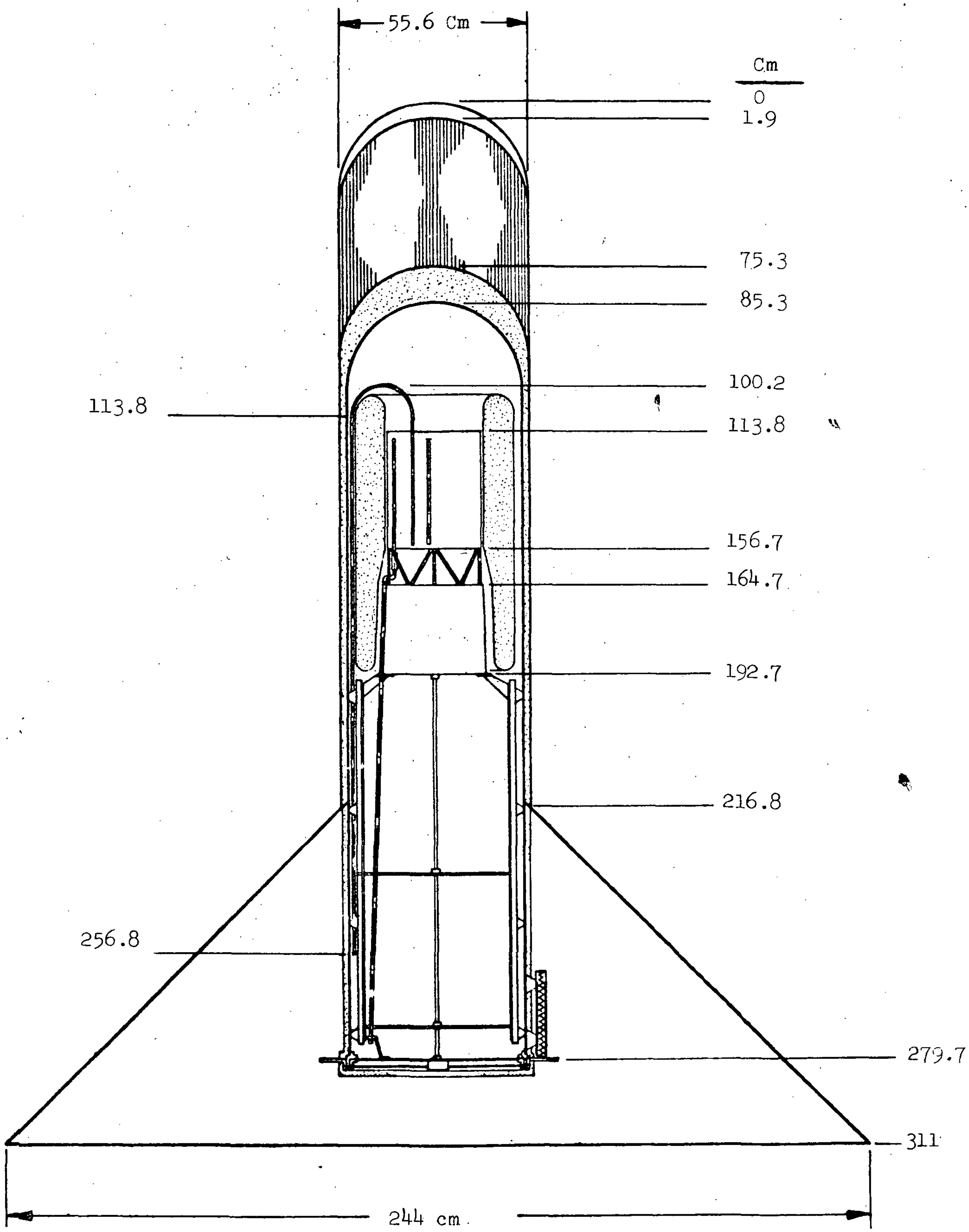


Fig. 4. Overall dimensions of launch configuration

4. SYSTEM MASS AND POWER SUMMARY

Figure 4 illustrates the overall dimensions of the launch system and the major components of the radioisotope thermionic power supply at the reference design power level.

A summary of the masses of the major components of the system is given in Table 1. From this table it is seen that the total mass at launch is 723.8 kg while the mass of the extended mission flight system is 148.5 kg.

A summary of the electrical power output from the system at beginning-of-life (BOL) and end-of-life (EOL), after 36,000 hours, is shown in Table 2.

TABLE I

SYSTEM COMPONENT MASSES

A. Extended Mission System

Curium isotope capsules	29.2 kg
Emitter heat pipes	9.0 kg
Thermionic diode components	6.4 kg
Space radiator heat pipes	6.3 kg
Structure for space radiator	3.2 kg
Beryllium shield	24.1 kg
Heat source structure and insulation	15.5 kg
Transmission lines and boom	18.9 kg
Power conditioning	30.0 kg
Blast shield	5.9 kg
Flight System Total	148.5 kg

B. Disposable Safety System

Graphite and insulation on nose	11.3 kg
Graphite on cylindrical body	17.0 kg
Graphite and insulation on flare	123.7 kg
Aluminum honeycomb	25.1 kg
Zirconia/LiH fire shield	67.5 kg
Aerodynamic flare structure	35.7 kg
LiH shielding	78.6 kg
Auxiliary heat pipes	32.5 kg
Launch and impact structure	128.4 kg
Jettison mechanisms (EST)	12.8 kg
Helium baffle	2.3 kg
Titanium blast shield	25.8 kg
Electronic recovery aids	14.6 kg
Safety System Total*	575.3 kg
TOTAL LAUNCH MASS	723.8 kg

*Mass of the safety system is uncertain by approximately 90 kg due to design tolerances in the aerodynamic flare diameter and graphite thickness (Vol. II, Section 4.4. and 6.3.3.2.).

TABLE 2

ELECTRICAL PERFORMANCE PARAMETERS

	BOL	EOL
Number of operating converters	69	59
Net output (kWe)	5.93	5.0
Overall efficiency (%)	13.4	13.2
Total thermal power (kWt)	44.2	37.77
Voltage to P.C. (volts)	15.2	14.9

5. AEROSPACE NUCLEAR SAFETY

5.1 SAFETY CRITERIA

The aerospace nuclear safety philosophy currently employed by the USAEC is complete containment of the radioisotope fuel during all normal and accident environments. Specifically, the major safety criteria considered in this study were:

1. Containment during a launch pad abort of a Titan IIID/Centaur vehicle, including survival during a 10-minute 2600°K solid propellant fire environment

2. Safety System protection during an ascent abort and resulting explosion schrapnel

3. Fuel and Safety System protection for worst-case atmospheric reentry trajectories

4. Fuel structural and thermal protection during terminal velocity impact onto land

5. Radiation shielding to minimize exposure of operating personnel and persons in the vicinity of an aborted system.

All of these design criteria were met using passive and redundant components, which are listed in Table 3.

As noted in Ref. 1, these criteria were established considering that the mission profile calls for direct insertion of the radioisotope power supply into a heliocentric orbit with no requirement for reentry and retrieval, except in the case of launch aborts. Although no suitable

TABLE 3

RTPS SAFETY PACKAGE SUBSYSTEMS

1. Passive orientated reentry aeroshell with spherical nose and aerodynamic flare
2. Pyro-Carb ablator with zirconia felt insulation backing for reentry protection
3. Auxiliary heat pipe radiator
4. Auxiliary radiator solid propellant fire protection shield
5. Honeycomb
6. Launch pad helium circulation cooling chamber
7. Blast shield for fragment protection
8. Location aids for land and water impact

guidelines exist concerning acceptable radiation exposure to persons in the vicinity of an impact on land, it has been estimated that recovery could be accomplished in 12 hours, which would reduce the maximum exposure to persons within 3 meters to less than 25 rem - the maximum whole body dose for reactor siting studies.

5.2 LAUNCH PAD EXPLOSION AND FIRE

Launch abort environments for the RTPS were modeled after those presently employed for Space Nuclear Systems, such as the TRANSIT and Pioneer RTGs and the Isotope Brayton Systems.

For the present study the worst-case thermal environment, in terms of total heat input, was assumed for the heat shield design - $2600^{\circ}K$ for 10 minutes. Concepts considered to protect the safety package in this environment included:

1. High C_p heat shield materials
2. Insulation blankets
3. Thermal switch materials
4. Concentric radiation shields
5. Composite insulation/heat storage shields.

The latter approach was the only method found acceptable from both a thermal protection and total mass standpoint. Table 4 tabulates the design characteristics of the fire shield.

The approach taken to protect against the blast fragment environment has been to place a thin titanium fragment shield at the aft end of the system to attenuate high energy fragments and provide partial protection to the flare.

TABLE 4

CHARACTERISTICS OF THE LiH/ZIRCONIA LAUNCH PAD ABORT FIRE SHIELD (2600°K FIRE, 10 MINUTES)

LiH thickness	1.9 cm (0.75 in.)
Stainless steel structure thickness	0.152 cm (0.06 in.)
Zirconia thickness	0.31 cm (0.12 in.)
Shield length	142 cm (361 in.)
LiH density	0.75 gm/cc
Stainless steel density	8.0 gm/cc
Zirconia density	0.9 gm/cc
LiH mass	32.6 kg
Stainless steel mass	28.4 kg
Zirconia mass	6.5 kg
Total shield mass	67.5 kg

The fire shield and outer structure are designed to withstand the maximum predicted overpressure of 150 psi from an explosion on the launch pad. Lower pressures will be experienced during a high altitude abort.

5.3 REENTRY AND AEROTHERMODYNAMICS

Reentry of the radioisotope thermionic power system (RTPS) occurs only in the event of failure of the launch vehicle to place the payload in an escape trajectory. All possible abort conditions were considered. The most likely abort cases cause the RTPS to reenter as a free body.

The most severe reentry cases are: (1) steep reentry (-90^{o} flight path angle) at escape velocity which resulted in the highest heating rates and external surface temperatures, and (2) orbital decay reentry with its attendant long heating pulse which resulted in the highest internal structural temperatures.

An aerodynamically oriented reentry configuration was adopted to cause the vehicle to impact on the energy absorbing nose. An oriented configuration also minimizes reentry protection requirements since the most severe heating is generally localized over the small nose region.

Oriented reentry was provided by a flare located just aft of the active radiator area. In addition to providing aerodynamic stability, the flare also reduces the ballistic coefficient and, thus, reduces reentry heating. The flare was sized to provide the maximum drag for the minimum weight.

The worst aerodynamic heating occurs during the -90^{o} reentry on the flare due to turbulent flow conditions and results in a maximum heat flux of 2500 B/ft^2 sec. The worst total heating occurs during orbital decay reentry at the stagnation point and results in a total heat load of 6200 B/ft^2. The worst ablation occurs during reentry at -10^{o} at escape velocity due to the relatively long heat pulse and high surface temperatures.

A carbon/graphite fibrous composite, Pyro-Carb 406, was selected for the ablative heat shield. A layer of zirconia felt backs up the Pyro-Carb 406 and insulates the underlying structure on the nose and flare, and structure-safety system on the cylindrical body.

5.4 IMPACT ON WATER

The most likely abort impact mode is immersion in sea water. Provisions have been incorporated in the design of the RTPS for water flotation and recovery. Additional protection is provided by the tungsten alloy capsules which have excellent sea water corrosion resistance with less than 0.3 mils per year surface recession rate expected.

5.5 LAND IMPACT

The safety package is designed to survive terminal velocity impact onto a hard unyielding surface by a combination of energy absorption honeycomb and structural support of the heat source within the safety package. Various energy absorption and aerodynamic drag augmentation systems were considered. The aerodynamic flare selected was designed to reduce the terminal velocity to 45 m/sec. Crushing of the aluminum honeycomb energy absorber limits the deceleration on impact to 200 gs.

5.6 LOCATION AND RECOVERY AIDS

Recovery aids are incorporated in the safety package for locating the RTPS following an accidental abort impact onto land or water. In addition to trajectory and tracking data on the spacecraft, a number of techniques and devices will be employed to aid in rapid location of the power supply. These include: (1) radio beacons such as the SARAH system (Search and Rescue and Homing), (2) radar chaff and reflective coatings on the flare base and blast shield, (3) sound fixing and ranging (SOFAR) devices, (4) sea water dye markers, (5) flashing lights, (6) underwater pingers, and (7) flotation devices.

8. CONCLUDING SECTION

This study showed the feasibility of a passive containment system for the ^{244}Cm isotope. The passive containment system has inherent reliability and safety advantages over the original design (Ref. 1), in which mechanical systems were required to operate in certain launch abort conditions. In the present design, the launch escape rocket system is replaced by a blast shield and a fire shield. The blast shield protects against a booster explosion on the launch pad or during ascent to earth escape. The fire shield, which consists of refractory thermal insulation (zirconia felt) and a heat storage material (lithium hydride), protects the isotopic fuel and its containment vessel from a sustained propellant fire on the launch pad. The aerodynamic flare, which stabilizes the orientation during reentry from an abort trajectory, was enlarged to reduce the ballistic coefficient and the terminal velocity. At the lower velocity, the impact energy is absorbed by a crushable honeycomb structure in the nose of the reentry body, with no parachutes required to reduce the velocity.

The design changes, incorporating a passive containment system and eliminating the need for a launch escape rocket and parachutes, resulted in an increase in the mass of the safety equipment. The mass of the safety equipment, which is discarded once Earth escape has been verified, was increased by 270 kg to 575 kg. Since the safety equipment is jettisoned early in the heliocentric portion of the mission, its mass has a relatively small effect on mission performance. The reduction in payload mass due to the 270 kg increase in the safety equipment is only about 55 kg for a typical mission, such as an orbiter of Saturn.

Calculations of the effects of oxygen released from the Cm_2O_3 fuel showed that oxygen-induced sublimation of the capsule and emitter heat pipe or oxygen contamination of the lithium working fluid in the heat pipe could limit the lifetime of the system. More experimental data on oxygen release rates and effects of oxygen are required before radioisotope power systems of this type can be designed with reasonable assurance of meeting a specified lifetime objective. It appears that lowering the temperature of the isotopic fuel and using oxygen getters in the heat source will both be effective in reducing oxygen effects on the heat source.

The next step in the development of the radioisotope thermionic power supply for electric propulsion applications should be to establish the materials and fabrication technology for the isotope capsule and emitter heat pipe. Capsule vents must be developed to release helium but minimize the loss of fuel and resist plugging by condensed fuel vapors. The bulk of this development can be performed using a non-radioactive fuel simulant. A few confirmatory tests should be performed with $^{244}Cm_2O_3$-fueled capsules. Fabrication methods for emitter heat pipes should be developed and heat pipes tested over a range in temperature, heat flux, and oxygen pressure so that the results can be extrapolated to the long lifetimes required for electric propulsion missions.

In addition to establishing the feasibility of a 36,000-hour lifetime, the technology development efforts will determine the maximum thermionic emitter temperature and, hence, the maximum performance level for the power system. Coupled with the technology development effort to establish performance levels should be a more thorough and detailed planning effort for the total development program through the first flight. A detailed program plan will permit a more accurate estimate to be made of the development schedule and cost of the system. The decision whether to develop the system would then be made on the basis of more accurate mission performance, schedule, and cost data.

REFERENCES

1. Homeyer, W. G., C. A. Heath, and A. J. Gietzen, "Preliminary Study of a Radioisotope Electric Propulsion System" Vol II, "Final Report," Gulf General Atomic Report GA-10339, 1970.