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SC-RR-71 0533

**THERMAL DESIGN AND ANALYSIS OF
THE APOLLO LUNAR RADIOISOTOPIC
HEATER**

Robert D. Klett
Reentry and Space Sciences Division

AEROSPACE NUCLEAR SAFETY DEPARTMENT - 9510

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OF
THE APOLLO LUNAR RADIOISOTOPIC HEATER

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September 1971

ABSTRACT

This report presents the reentry thermal analysis of the Apollo Lunar Radioisotopic Heater for a random tumbling and spinning motion with a surface averaged heat flux. Orbital decay and lunar returns with reentry angles ranging from -6.25° to -90° were analyzed. The temperatures of the heater in a launch pad abort fire were also predicted. The study of the original design led to several recommendations for modifications which were incorporated in the final heater configuration.

The cutoff date for information appearing in this report is February 1969.

Key Words: (1) Optimization (2) Thermal Insulation (3) Aeroheating

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Summary

As part of the Aerospace Nuclear Safety program, safety analyses were conducted to determine the behavior of nuclear power supplies during launch aborts and reentries from space. In addition to the safety analyses conducted on the Apollo Lunar Radioisotopic Heater, Sandia was requested by the AEC to recommend changes that would increase the safety of the ALRH.

The aerothermal study consisted of preliminary analyses to locate any potential safety hazards followed by a series of parametric studies to determine the best method of solving each problem. All changes, as well as the basic design, were restricted by the tight schedule and limited budget of this program. The environments investigated were launch pad aborts, orbital decay, and lunar return reentries with reentry angles ranging from -6.25° to -90° .

The results of the study indicate that:

- (1) A thermal resistance between the ablator and the interior heat sink greatly reduced internal reentry temperatures, particularly during short duration reentries.
- (2) The most efficient thermal resistor was 7 lb/ft^3 fibrous carbon insulation.
- (3) The temperatures of all internal components were predicted to remain well below melt temperature for all reentries provided the heat shield remained intact.
- (4) The ALRH will contain the fuel inventory when the bare titanium capsule is exposed to the URS, Sandia standard, and Sandia extended launch abort fire models.

Acknowledgment

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THERMAL DESIGN AND ANALYSIS OF THE APOLLO LUNAR RADIOISOTOPIC HEATER

Introduction

As part of the first NASA Apollo mission, the Early Apollo Scientific Experiments Package (EASEP) gathered data on the characteristics and environment of the lunar surface and subsurface. One part of the EASEP shown in Figure 1 is the Passive Seismic Experiment Package (PSEP) used to monitor seismic activity. It detects meteoroid impact and free oscillations in lunar rotation. It may also detect surface tilt tidal deformations partially resulting from periodic variations in the strength and direction of external gravitational fields acting on the moon. The PSEP, which is powered by solar cells, transmits data only during the lunar day.

The temperature of the PSEP electronic package must be maintained between -65°F and $+140^{\circ}\text{F}$ during the lunar day and night to remain operational (Ref. 1). Because of the -300°F lunar night temperatures, an unheated PSEP would be limited to a maximum operating life of 14 days. To maintain the proper temperature levels in the electronics package, two 15-watt Apollo Lunar Radioisotopic Heaters (ALRH's) were attached to the PSEP (Figures 1 and 2). Thermal energy from these heaters is produced by the radioactive decay of plutonium dioxide microspheres. The ALRH was the first nuclear heat source to be used in a superorbital mission.

The development of the ALRH was under the direction of the Atomic Energy Commissions' Space Electric Power Office (AEC/SEPO). Mound Laboratory was given the overall design and manufacturing responsibility. Sandia Laboratories provided the aerothermodynamic analyses and safety test programs to evaluate the design and provide input data for the Safety Assessment Report which was written by AEC Headquarters personnel. Sandia was also responsible for recommending heat shield design changes that would improve the efficiency and safety of the ALRH. After a preliminary analysis of the ALRH, Sandia conducted a design study to select the best method of thermal protection and to optimize the materials, sizes, and location of each heat shield component. An exploded view of the final ALRH design, incorporating the recommended design changes, is shown in Figure 3. The heater is a cylinder approximately 3 inches long and 3 inches in diameter with a mounting flange on one end. Each heater weighs 2.13 pounds. The plutonium-238 dioxide microsphere fuel is encapsulated in dual 0.020-inch thick tantalum - 10 percent tungsten liners. The Ta-10W is protected from oxidation by a 0.010-inch thick platinum - 10 percent rhodium clad. A 0.250-inch thick titanium fragment shield protects the clad and liners in case of an explosive abort and during earth impact after reentry. The 0.150 inch of 7 lb/ft^3 fibrous carbon insulation inside the ablation shield keeps the titanium and Pt-10Rh from melting during lunar return reentries. The 0.60-inch thick AXF-5Q POCO graphite ablating heat shield has plugs on both ends for assembly and thermal stress relief. The outer container, which is 0.030-inch thick 304 stainless steel, protects the graphite from oxidation below 3010°R and supports the heat source. The 6061 aluminum base plate distributes the heat to the PSEP. All metal containers have welded closures, and the length to diameter ratio of all containers is approximately one.

The safety philosophy of the ALRH is to provide containment of the fuel particles through reentry and earth impact, through a launch pad abort fire, or through an explosive abort.

The aerothermodynamic analyses and tests of the ALRH were conducted by the Reentry and Space Sciences Division of Sandia Laboratories between December 15, 1968 and March 1, 1969 and were reported in Ref. 2. This report documents the aeroheating, preliminary thermal studies, heat shield design and optimization, selected reentry studies, and launch pad abort fire analyses that were performed as part of the entire aerothermodynamic analysis. Details of the ablation and steady-state and reentry temperature predictions not included in this report can be found in Ref. 3.

Aerodynamic Heating

Trajectories were computed for orbital decay and several superorbital reentries covering the range of possible reentry angles. It was assumed that the ALRH was attached to, and protected by, the LM down to an altitude of 300,000 feet. Below 300,000 feet, ALRH orientations of cross flow stable and random tumbling were analyzed. The LM aerodynamic models utilized constant ballistic coefficients of 79.6 pounds per square foot for the ascent and descent stages and 111.8 pounds per square foot for the descent stage only for crosswise orientations. The ballistic coefficients for the ALRH are given in Figure 4 as a function of altitude. The Sandia TTA computer code (Ref. 4) was used to compute the following trajectories:

- (1) Orbital decay, heater random tumbling;
- (2) Orbital decay, heater crosswise;
- (3) Supercircular velocity, -6.25° reentry, heater random tumbling;
- (4) Supercircular velocity, -6.25° reentry, heater crosswise;
- (5) Supercircular velocity, -20.0° reentry, heater random tumbling;
- (6) Supercircular velocity, -20.0° reentry, heater crosswise;
- (7) Supercircular velocity, -38.05° reentry, heater random tumbling;
- (8) Supercircular velocity, -38.05° reentry, heater crosswise;
- (9) Supercircular velocity, -90.0° reentry, heater random tumbling;
and
- (10) Supercircular velocity, -90.0° reentry, heater crosswise.

Multiple orbit trajectories of the heater in a crosswise orientation were also computed. These highly elliptical orbits result from supercircular velocity lunar return trajectories which reenter the earth's atmosphere at too shallow a reentry angle. The multiple orbit trajectories were computed for perigee altitudes of 300,000 feet, 280,000 feet, and 260,000 feet. Thermal analyses of the multi-orbit trajectories are described in Reference 3.

Aerodynamic heating computations for the ALRH included free molecule and continuum convection, and gas cap radiation.

The convective reference heat rate to the ALRH in the free molecule flow regime is determined by

$$\dot{q}_{FM} = \frac{a \rho_{\infty} V_{\infty}^3}{50063} \quad (\text{Ref. 5}) \quad (1)$$

where

$\dot{q}_{FM}$ = free-molecular flow stagnation point heating rate (Btu/ft²-sec)

a = thermal accommodation coefficient

ρ_{∞} = free stream density (lbs/ft³)

V_{∞} = free stream velocity (ft/sec).

In the continuum flow regime, the convective reference heating rate to a one-foot radius sphere is determined by

$$\dot{q}_c = 17600 \left(\frac{\rho_{\infty}}{\rho_s} \right)^{0.5} \left(\frac{V_{\infty}}{V_c} \right)^{3.15} \quad (\text{Ref. 4}) \quad (2)$$

where

$\dot{q}_c$ = continuum convective stagnation point heating rate (Btu/ft²-sec)

ρ_{∞} = local air density

ρ_s = sea level air density

V_{∞} = free stream velocity

V_c = circular orbital velocity.

The convective heating which occurs in the transition flow regime was obtained by extending the free molecule and continuum heating curves to their intersection and using whichever heat rate was lower. This method gives results close to various transition theories (Ref. 5) as well as flight test data (Ref. 6).

Local free molecule convective heating rates were found by multiplying the reference heating rate [Eq. (1)] by the following free molecule heating ratios from Ref. 7.

$$\left. \begin{array}{l} F_{FM} = \cos \theta + \frac{0.337}{M_{\infty}} \sin \theta \quad \text{for } 0^{\circ} \leq \theta \leq 90^{\circ} \\ \text{or} \\ F_{FM} = 0 \quad \text{for } 90^{\circ} < \theta \leq 180^{\circ} \end{array} \right\} \quad (3)$$

θ = The angle between a normal to the surface and the velocity vector.

M_{∞} = Free stream Mach No.

Local or average continuum convective heating rates were found by multiplying the reference rate from Eq. (2) by $1/\sqrt{R}$ and the continuum heating ratio (F_c). R is the radius of the ALRH. The ALRH continuum heating ratios were based upon wind tunnel tests conducted at Cornell Aeronautical Laboratory and Rhodes and Bloxson by Sandia (Refs. 8 and 9). The heating ratios for a cross flow stable orientation are given in Figures 5 and 6. The heating ratios given in Table I are the surface averages of the angle of attack test data. These data were averaged for spinning and tumbling motions using techniques given in Reference 5 to yield the average heating ratios ($\bar{F}_c$) in Table II.

TABLE I

Average Continuum Flow Heating Ratios for the ALRH vs. Angle of Attack
($\alpha = 0$ For End-On Orientation)

α	Sides	Front End	Back End	Entire Surface
0	0.068	0.625	0.050	0.158
15	0.099	0.580	0.050	0.171
30	0.150	0.565	0.050	0.203
45	0.210	0.530	0.050	0.237
60	0.234	0.410	0.050	0.233
75	0.252	0.180	0.050	0.206
90	0.255	0.072	0.072	0.194

TABLE II

Average Heating Ratios for a Tumbling ALRH
in Continuum Flow

Location	Motion	Heating Ratio
Ends	Random-tumble-spinning	0.206
Sides	Random-tumble-spinning	0.216
Entire surface	Random-tumble-spinning	0.212
Ends	Side-on-spinning	0.072
Sides	Side-on-spinning	0.255
Entire Surface	Side-on-spinning	0.194

The reference radiant heating to the ALRH during superorbital velocity reentry was determined by

$$\dot{q}_R = \left(3.748 \times 10^{-6}\right) \rho_{\infty}^{1.16} R^{0.5} V_{\infty}^3 \left(V_{\infty}/10^4 - 3.25\right)^{1.337} \quad (4)$$

(Ref. (10))

where

$\dot{q}_R$ = radiative stagnation point heating (Btu/ft²-sec)

ρ_{∞} = free-stream density (lbs/ft³)

R = nose radius (ft)

V_{∞} = free-stream velocity (ft/sec).

This equation correlates well with available experimental data in Reference 11. Equation 4 was solved for the -38° and -90° lunar return trajectories (Ref. 3). Tables III and IV compare the stagnation point convective and radiative heating for these two trajectories. Since the stagnation radiation heating was always less than 5 percent of the convective heating, the radiation heating was omitted from the ALRH thermal analyses.

The aerodynamic heating rates discussed thus far in this section are cold wall heating rates. These cold wall rates are used as input to the CMA computer code (Ref. 12). The CMA code calculates a corrected heating rate based upon the more realistic conditions of a hot wall, with mass injection into the boundary layer and oxidation of the graphite surface. Averaged surface cold wall heating rates and the CMA corrected rates are compared in Figures 7 through 11 for random tumbling and spinning motions. Figures 12 through 15 compare the cold wall and CMA corrected

stagnation line heating rates for a cross flow stable orientation. The CMA corrected heating rates were used as inputs to thermal models to predict temperatures.

TABLE III
Stagnation Point Heating for a -38° Lunar Return Reentry
of a 0.125 ft. Radius Sphere

<u>Altitude (feet)</u>	<u>Radiant Heating (Btu/sq-ft-sec)</u>	<u>Convective Heating (Btu/sq-ft-sec)</u>
300000.0	0.223	172.27
299307.0	0.234	175.84
288129.0	0.495	242.71
276955.0	1.024	332.00
265785.0	2.412	467.30
254621.0	4.104	604.36
243466.0	7.318	776.86
232323.0	12.578	984.51
221199.0	20.554	1223.90
210105.0	32.352	1503.10
199057.0	48.445	1818.17
188062.0	70.073	2189.14
177163.0	93.756	2595.58
166395.0	111.080	3022.28
155818.0	105.526	3462.64
145511.0	49.221	3899.86
135591.0	0.	4194.71
126207.0	0.	4209.06
117535.0	0.	3870.99
109741.0	0.	3189.35
102937.0	0.	2364.33
97143.0	0.	1598.68
92292.0	0.	1017.41
88262.0	0.	629.65
84916.0	0.	383.92

TABLE IV

Stagnation Point Heating for a -90° Lunar Return Reentry
of a 0.125 ft. Radius Sphere

<u>Altitude (feet)</u>	<u>Radiant Heating (Btu/sq-ft-sec)</u>	<u>Convective Heating (Btu/sq-ft-sec)</u>
300000.0	0.223	172.29
290862.0	0.418	225.66
272647.0	1.376	376.91
254429.0	4.141	605.90
236215.0	10.595	909.87
218017.0	24.374	1310.00
199864.0	50.691	1819.00
181803.0	97.198	2475.98
163295.0	163.898	3303.98
146394.0	205.309	4309.43
129532.0	24.685	5274.06
113907.0	0.	5474.56
100305.0	0.	4368.16
89348.0	0.	2581.89
81069.0	0.	1244.83
74993.0	0.	551.56
70514.0	0.	247.74

One-Dimensional Thermal Model

The size, shape, and materials of the outer stainless steel container, Pt-10 Rh clad, both Ta-10W liners, and the fuel specified in the original Mound Laboratory design remained unchanged throughout this analysis. However, changes were made in the Ti fragmentation shield and the heat shield based on Sandia design studies and tests. The original design:

- (1) Had dual heat shields and dual fragmentation shields; starting from the outside, there were layers of graphite, titanium, graphite, and titanium;
- (2) Had no insulation;
- (3) Used Stagpole 6014 for the heat shields; and
- (4) Had openings on only one end of the graphite shields.

Based on heat transfer theory and previous experience with the SNAP-27, Sandia recommended a change to single heat and fragmentation shields prior to any thermal analyses. It was also recommended that ATJ graphite be used for the heat shield based on material properties and availability. These changes were made so the reference design used for the beginning of this study included a 0.75-inch ATJ graphite heat shield over a 0.25-inch thick titanium fragmentation shield.

During the heat shield design phase of the ALRH program, there was a need for a thermal model that could be used to perform preliminary safety studies, to conduct parametric studies, and to investigate alternate designs. This need established the requirement for a model that was easily changed, used little computer time, and was an accurate representation of the physical model. The modified one-dimensional model shown in Figure 16 met these requirements.

The relative surface areas and volumes of concentric cylinders with length to diameter ratios of one can be duplicated by a conical model with a 90° apex angle. For the ALRH configuration, the conical model with the surface area proportional to the radius squared is more accurate than either a cylindrical model (area proportional to r) or a cartesian coordinate model (area=constant). A more detailed explanation of this type of modified one-dimensional thermal modeling is given in Appendix A.

With the surface averaged heat flux used for tumbling motion, the conical model is nearly as accurate as a three-dimensional model. Surface averaged heating rates can also be used to accurately predict temperatures inside the high resistance insulation for stable nonspinning reentry. This was made possible by the relatively large conduction cross section and the small conduction length in the circumferential direction in the graphite ablation shield. As an example, the predictions of peak interior temperatures made with the one-dimensional conical model were within 5°R of the temperatures predicted with the 426 node three-dimensional model described in Reference 3.

External nodes were removed from the model as melting or ablation occurred. Internal confined nodes included the heat of fusion, but they were not removed if melting occurred. All material properties (References 13 through 23) were programmed as functions of temperature.

The conical one-dimensional model, used with the HERMAN heat transfer computer program (Ref. 24), was used to compute operating temperatures, reentry temperatures, and temperatures in a launch pad abort fire for heat shield design and safety assessments of the ALRH.

Heat Shield Design Concepts

A preliminary study of the reference ALRH design indicated that the titanium and Pt-10 Rh would melt during a -6.25° lunar return reentry. This study also showed that the -6.25° lunar return produced the highest internal temperatures for the reference design, followed by orbital decay and the -38° and -90° lunar returns.

Modifications to the reference design were then investigated in an attempt to reduce internal temperatures during reentry. The following restrictions were placed on the modifications because of system requirements and schedule limitations.

- (1) The weight and exterior dimensions could not be increased.
- (2) The titanium thickness could not be reduced. In addition to the impact protection provided by the titanium, it was also a necessary heat sink for the short duration lunar return reentries.
- (3) No changes could be made to either the Pt-10 Rh or Ta-10W containers.
- (4) No materials or techniques requiring any development time could be considered.
- (5) All materials had to be readily available, easily fabricated, and well characterized.

Since the most severe trajectory had a relatively high heating rate and was of relatively short duration, a thermal resistance between the high temperature ablation shield and the interior heat sink was the least complicated modification that could be made. Because of the large ratio of graphite thickness to circumference, a thermal resistance in this location would also reduce hot spots at the front of the fragment shield if the ALRH reentered with a stable orientation. Little of the heat passing from the front to the back of the ablation shield would go through the high resistance path and the small amount of heat that reaches the titanium could be distributed in the thick fragment shield.

Three modifications were considered to obtain the desired thermal resistance. These were as follows:

- (1) Placing 0.05-inch radiation gaps on each side of the titanium. All surfaces were assumed to be oxidized for a high emissivity. Radiation gaps are not an efficient method of insulating heat sources because of the high resistance at low (operating) temperatures and the low resistance at high (reentry) temperatures. However, this method might have been adequate and it was the fastest and cheapest method of adding thermal resistance.
- (2) The same as (1) but with low emissivity unoxidized surfaces. If oxidation could not be prevented, low emissivity coatings could have been used.
- (3) Replacing the inner 0.15 inch of graphite with a high temperature insulation, such as fibrous carbon, carbon foam, carbon felt, a refractory felt, or a ceramic foam. Anisotropic insulation, such as pyrolytic graphite, was not necessary because of the

large amount of heat that could be conducted circumferentially in the ablation shield. It would have been desirable to use an insulation that decreased in conductivity with increasing temperature, but the conductivity of all of the high temperature, low conductivity insulators available increased with temperature. A 12 lb/ft³ fibrous carbon felt was selected for this phase of the study as being representative of this group of insulators.

The -6.25° lunar return trajectory with the ALRH in a random tumbling and spinning mode was selected for the thermal resistance comparative study. The radiation equilibrium temperature (3400°R, Ref. 7) of the ALRH during orbital decay was lower than the melting temperature of titanium (3500°R). Therefore, orbital decay was not a critical trajectory. Table V lists the important thermal parameters for each part of the heat source for the four configurations. Impact temperatures are not given if the component melts during reentry. The results of the study can be summarized as follows:

- (1) Reference Design: This design has the lowest interior operating temperatures, the highest interior reentry temperatures, and the highest graphite temperature gradients during reentry. The titanium reaches a temperature 680°R above melt, and the Pt-10Rh reaches a temperature 340°R above melt.
- (2) Radiation Gaps: These modifications result in the highest interior operating temperatures. Interior temperatures and graphite temperature gradients are lower than for the reference design. With oxidized surfaces, the titanium reaches a temperature 290°R above melt and the Pt-10Rh does not melt. With unoxidized surfaces, the Pt-10Rh does not melt and the peak titanium temperature reaches the melt temperature. Although this is an improvement over the reference design, peak interior reentry temperatures are still too high and would be slightly higher for a stable reentry.
- (3) Insulation Layer: The insulated heater has the lowest temperature gradients in the graphite and the lowest interior temperatures during reentry. The titanium remains 1820°R below melt and the Pt-10Rh 2120°R below melt.

Figures 17 through 21 show the large differences in reentry temperatures for the reference and insulated heat shield designs. The small difference in ablation shield temperatures (Figure 17) is caused by the differences in the heat conducted into and out of the fragmentation shield assembly. When parts of the reference design melt, heats of fusion affect the temperature profiles of other components, as indicated in Figures 18 through 20. The large thermal lag in the insulated ALRH delays the internal temperature rise until after peak heating, when the temperature gradient from the ablation shield has decreased.

TABLE V

Maximum Temperatures, Temperature Gradients, and
Ablation of the ALRH
for Random Tumble and Spin in
a -6.25° Lunar Return Reentry

Parameter	Reference Design	2 Gaps Oxidized Surfaces	2 Gaps Unoxidized Surfaces	0.15" 12# Fibrous Carbon Insulation
<u>Graphite Heat Shield</u>				
Temp. in Space, °R	734	734	734	734
Peak Reentry Temp., °R	4635	4671	4685	4774
Impact Temp., °R	1663	1658	1654	1511
Temp. Gradient, Nodes G2 to G7, °R	876	824	814	700
Total Ablation, In.	0.032	0.032	0.032	0.032
<u>Titanium Strength Member</u>				
Temp. in Space, °R	740	843	854	900
Peak Reentry Temp., °R	4181	3786	3503	1679
Impact Temp., °R	--	--	--	1675
<u>Pt-10Rh Liner</u>				
Temp. in Space, °R	742	1348	1370	902
Peak Reentry Temp., °R	4139	3566	3091	1678
Impact Temp., °R	--	2214	2358	1677
<u>Ta-10W Capsules</u>				
Temp. in Space, °R	743	1349	1371	903
Peak Reentry Temp., °R	4137	3563	3090	1678
Impact Temp., °R	1714	2215	2359	1678
<u>Fuel</u>				
Temp. in Space, °R	904	1474	1494	1053
Peak Reentry Temp., °R	4069	3538	3108	1790
Impact Temp., °R	1892	2330	2461	1790

The insulated design appeared to be the most promising and was compared to the reference design for orbital decay and -38.05° lunar return reentries. As shown in Table VI, the insulated design is superior in all categories during reentry. The ablator/insulator concept of reentry protection was selected because it prevents melting of the titanium, reduces impact temperatures, reduces the temperature gradients across the graphite, and reduces weight with only a slight increase in internal operating temperatures.

TABLE VI

Maximum Temperatures, Temperature Gradients, and
Ablation of Apollo Heat Source
Random Tumbling and Spinning

Parameter	-38.05° Lunar Return		Orbital Decay	
	Reference Design	0.15" 12# Fibrous Carbon Insulation	Reference Design	0.15" 12# Fibrous Carbon Insulation
<u>Graphite Heat Shield</u>				
Temp. in Space, °R	734	734	734	734
Peak Reentry Temp., °R	4283	4345	3272	3303
Impact Temp., °R	1823	1707	1735	1621
Temp. Gradient, Nodes G2 to G7, °R	1923	1799	70	48
Total Ablation, In.	0.01	0.01	0.036	0.036
<u>Titanium Strength Member</u>				
Temp. in Space, °R	740	900	740	900
Peak Reentry Temp., °R	2824	1388	3241	1983
Impact Temp., °R	1866	1388	1769	1957
<u>Pt-10Rh Liner</u>				
Temp. in Space, °R	742	902	742	902
Peak Reentry Temp., °R	2806	1387	3240	1983
Impact Temp., °R	1866	1387	1769	1957
<u>Ta-10W Capsules</u>				
Temp. in Space, °R	743	903	743	903
Peak Reentry Temp., °R	2805	1387	3240	1983
Impact Temp., °R	1867	1387	1770	1957
<u>Fuel</u>				
Temp. in Space, °R	939	1053	939	1053
Peak Reentry Temp., °R	2625	1472	3269	2070
Impact Temp., °R	2037	1472	1940	2057

Insulation Selection and Optimization

The selection of the insulation material was based on the following requirements.

- (1) Low thermal conductivity.
- (2) Minimum increase in conductivity with temperature over the expected temperature range.
- (3) Resilient to compensate for differential thermal expansion of the ablator and fragment shield.
- (4) Strong enough to support the fragmentation assembly during shock, vibration, and reentry loading.
- (5) Chemically compatible with the adjacent materials.
- (6) Could be manufactured to the desired shape.

The materials considered for the insulator included 7 to 12 lb/ft³ fibrous carbon, 3 to 30 lb/ft³ carbon foam, carbon felt, Al₂O₃ foam, ZrO₂ foam, BeO foam, MgO foam, and tantalum felt. The best insulator for this application was found to be the 7 lb/ft³ fibrous carbon developed by ORNL, Y-12. Since this material had been characterized by Sandia (Ref. 22) and Y-12 could meet the ALRH schedule, the 7 lb/ft³ fibrous carbon insulation was incorporated into the ALRH design. The conductivity of this insulation is given in Figure 22.

A parametric study was conducted with the one-dimensional thermal model to determine the optimum thicknesses of the ablator and insulator. Minimum reentry temperatures were obtained with 0.60 inch of ablator and 0.15 inch of insulation. Operation temperatures were also satisfactory.

A second parametric study was conducted to determine the operating and reentry temperatures of the ALRH for orbital decay and a -6.25° lunar return with insulation conductivities ranging from 0.1 to 100 times the measured value for 7 lb/ft³ fibrous carbon (Figure 22). The fibrous carbon was a relatively new material and, hence, rigid quality control had not yet been achieved, and the effects of configuration and all environments were not precisely known. Therefore, it was desirable to know the effects if the overall resistance of the insulation was not exactly as predicted based on the experimental conductivity data. This study showed the safety factor in the insulated design, and it provided a rapid means of finding the critical temperatures of the ALRH after the thermal resistance of the insulator was found experimentally. If it had been necessary to select an alternate insulation for any reason, this study also showed the range of conductivities that would have been acceptable.

Figure 23 shows there is little change in the operating temperatures of the titanium and fuel if the conductivity is higher than predicted. Operating temperatures rise more rapidly with decreasing conductivity. However, it is unlikely that the conductivity multiplier would be less than 0.8 (equivalent to an argon filled gap). Figure 24 shows the peak reentry temperatures of the titanium. The conductivity could be increased by a factor of 15 without causing the titanium to melt during reentry. Figure 25 shows the titanium impact temperatures. When conductivity

multipliers are less than 2 to 3, the titanium temperature does not peak before impact so maximum and impact temperatures are the same. With higher conductivity insulation, the thermal lag is reduced and the internal parts of the heater begin to cool before impact.

A simulated orbital decay plasma jet test of an inert ALRH was conducted in the Sandia HEAT facility (Ref. 2). Figure 26 shows that the predicted exterior temperatures for the test were a close approximation to orbital decay temperatures. The predicted interior temperatures for the test were lower than orbital decay because there was no internal heat generation during the test. Prior to the test, the outside of the titanium was painted with seven temperature sensitive paints. These paints melt at given temperatures ranging from 1360 to 2960°R. Posttest observations indicated that peak stagnation line temperatures ranged from approximately 2100 to 2400°R; the aft side of the model, 180° from the stagnation line, ranged from approximately 1300 to 1400°R.

The nonuniform axial temperature distribution could have been caused by either nonuniform heating or by the holes and grooves cut in the model for instrumentation leads and attachments. The average peak test temperature was approximately 1860°R compared to a predicted temperature of 1440°R. The higher than expected average peak temperature could have been caused by low thermal resistance in the insulator.

The internal heat generation was removed from the modified one-dimensional model, and the averaged surface heating rate of the plasma jet test was programmed into the model. This program was run with insulation conductivities ranging from nominal to three times nominal. It was found that the model predicted approximately the same peak titanium temperature that was observed during the test when the conductivity was multiplied by 1.8 (Figure 27). Since the apparent conductivity of the insulation was approximately 1.8 times that used in the optimization studies, the operating, reentry, and impact temperatures can be found from Figures 23 through 25 by reading the temperatures for a conductivity multiplier of 1.8. As an example, the peak titanium reentry temperatures would be 1970°R during orbital decay and 1680°R during a -6.25° lunar return. The deviation in the apparent conductivity of the insulation was thought to be caused by the cylindrical shape of the insulation sleeve, variations in manufacturing, and/or environmental degradation of the insulation. During tests conducted by Sandia to find a better insulator for the Pioneer heat source, it was discovered that fibrous carbon insulators would partially graphitize during the short reentry heat pulses (Ref. 25). Figure 28 shows a comparison of the conductivities of a carbonized and fully graphitized fibrous carbon insulator similar to that used in the ALRH. The increase in low temperature conductivity caused by partial graphitization could have easily caused the higher than predicted titanium temperatures during the HEAT test.

Ablation Shield Temperature Gradients and Final Heat Shield Design

Temperature gradients in the ATJ graphite ablation shield at various times during a -38.05° lunar return reentry are shown in Figure 29. A thermal stress analysis (Ref. 2) using these gradients indicated that the ablation shield would fail in this environment. Thermal stresses were highest at the end opposite the end

plug, so the design was changed to have plugs in both ends of the ablation shield. Thermal stress failures were further reduced by finding an ablation material with a higher merit index $\left(\frac{KF}{\alpha E}\right)$ where:

K = thermal conductivity

F = ultimate tensile strength

α = thermal expansion

E = tensile modulus of elasticity .

Thermal stresses based on the temperature gradients for pyrolytic graphite (Figure 30) and POCO AXF-5Q (Figure 31) were compared to the stresses in the ATJ ablation shield. The POCO AXF-5Q was found to be superior to the other two graphites. Figures 32 and 33 show the temperature gradients in a POCO AXF-5Q ablation shield during -6.25° and -20° lunar return reentries. Cracking due to thermal stresses was predicted for -20° and steeper lunar return reentries. Simulation of supercircular reentry trajectories at reentry angles steeper than -6.25° was beyond the capability of ground test facilities. Therefore, it was not possible to demonstrate experimentally which reentries would produce a severe enough thermal stress failure to cause heat shield disassembly.

The addition of a second end plug and the change to a POCO AXF-5Q ablation shield were the final two modifications to the heat shield. The heat shield components are shown in Figure 34. A cross section view of the final ALRH design showing materials and thicknesses is shown in Figure 35. Table VII gives the weight breakdown for the ALRH components.

TABLE VII

Weights of the ALRH Components

<u>Component</u>	<u>Average Weight (lbs)</u>
Fuel (PuO_2 MCS)	0.0820
Ta- 10W Shim	0.0062
Ta- 10W Inner Liner	0.0384
Ta- 10W Outer Liner	0.0428
Pt- 10Rh Clad	0.0293
Ti Fragmentation Shield	0.2668
Graphite Insulator	0.0071
Graphite (POCO AXF-5Q)	0.9985
SS Container (304)	0.3977
Al Base Plate	0.2507
Rivets	0.0068
TOTAL	<u>2.1263</u>

Reentry Without the Heat Shield

Since the thermal stress analysis indicated that there could be thermal stress failures in the ablation shield for some of the supercircular reentries, the reentry temperatures of the heat source with the heat shield removed were computed. The following continuum flow ballistic coefficients were used for the various disassembly configurations.

Complete ALRH without the outer container, 28.0 lb/ft.²

Fragmentation assembly (without heat shield), 31.1 lb/ft.²

Ta-10W liner assembly (without Ti and Pt-10Rh), 31.7 lb/ft.²

The exact time of a possible heat shield failure was not known, so a bracketing technique was used with the ablator and insulator removed before reentry and at the time of peak heating. Figure 36 shows that, if the heat shield is removed prior to reentry during an orbital decay, the titanium and platinum-10 rhodium would melt but the melted fuel would be contained by the two tantalum-10 tungsten liners. Figures 37 and 38 show that, if the heat shield is removed before reentry or at peak heating during a -38° lunar return, the entire capsule would melt and the fuel would be released prior to earth impact.

Figure 39 shows that the fuel would also be released if the heat shield were lost prior to a -90° lunar return reentry. If the heat shield were lost at the time of peak heating during a -90° lunar return, only part of the fragment shield would melt and the fuel would not be released before earth impact (Figure 40). The fuel release conditions given in Figures 37 through 39 are in a velocity-altitude range that would cause the fuel particles to melt (Ref. 26).

This part of the study shows that fuel containment during reentry cannot be assured unless the heat shield remains intact.

Launch Pad Abort Fire

The ALRH was expected to have a substantial margin of safety in a launch pad abort fire because of the relatively large heat capacity and low internal generation per unit surface area. Therefore, whenever there was any doubt about the value of any parameter in the pad abort thermal analysis, the more severe or least desirable value was used. In some cases, the values selected may have been unrealistically severe so the actual margins of safety are larger than those indicated by this part of the analysis. The assumptions used in this study were as follows.

- (1) The stainless steel container, graphite heat shield, and insulation were removed prior to fuel ignition; this exposed the bare titanium for the entire duration of the fire.
- (2) The heat source was suspended in the fire ball with no protection of any kind. In the residual fire, the heater was assumed to be located three feet above the liquid surface where the fire temperature

is highest. It was also assumed the heater was not resting on or submerged in anything that could act as a heat sink.

- (3) All exterior surfaces were assumed to be oxidized for maximum heat transfer.
- (4) Three fire models were used--the standard URS, the standard Sandia (Ref. 27), and an extended Sandia model. The time-temperature plots of these fires are shown in Figure 41. The extended Sandia pulse is not believed to be possible, but it was included to dispel any doubts about the transition from the fire ball to the residual fire.
- (5) Nodes in the thermal model were removed when they reached the solidus temperatures of the materials.
- (6) The one-dimensional thermal model was used because heating and cooling were assumed to take place uniformly on all external surfaces.

The results of the study are shown in Figures 42, 43, and 44. No melting of any of the components was predicted for the URS fire. One-half of the titanium was predicted to melt in the standard Sandia fire, and all of the titanium was predicted to melt in the extended Sandia fire. The Pt-10Rh remained at least 300°R below the melt temperature in all three fires. The Ta-10W liners and fuel do not approach their melt temperatures.

Based on this conservative study, the ALRH is not expected to fail thermally in a pad abort fire.

Conclusions

The ALRH was subjected analytically to reentry and launch pad abort environments considered severe by the SNAP-27 Reentry Working Group. The following conclusions can be drawn regarding the heat shield design and safety of the ALRH.

- (1) The modified one-dimensional model was a useful tool for selecting a heat shield design concept and for optimizing the design. It also accurately predicted the interior reentry temperatures of insulated heat sources for all reentry orientations and motions and accurately predicted all temperatures for averaged surface heating, such as launch pad aborts and random tumbling and spinning reentries.
- (2) A thermal resistance between the ablator and interior heat sink greatly reduced interior reentry temperatures; the resistance is most effective during short duration reentries.
- (3) Although thermal resistance caused by radiation gaps reduced internal reentry temperatures, the most efficient resistor was 7 lb/ft³ fibrous carbon insulation.

- (4) Optimization curves for operating, peak reentry, and impact temperatures were helpful for selecting insulators, and insulator thicknesses, and for posttest analyses.
- (5) The temperatures of all internal components were predicted to remain well below melt temperature for all reentries provided the heat shield remained intact.
- (6) Release of the bare titanium capsule following thermal stress failure of the graphite heat shield leads to release of fuel in the melt zone.
- (7) The ALRH is predicted to contain the fuel inventory when the bare titanium capsule is exposed to the URS, Sandia standard, and Sandia extended launch abort fire models.

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Figure 1. Passive seismic experiment package in deployed configuration. One of the two Apollo Lunar radioisotopic heaters is circled.

Figure 2. Assembled Apollo Lunar radioisotopic heater.

Figure 3. Exploded view of the Apollo Lunar radioisotopic heater.

Figure 4. ALRH hypersonic ballistic coefficients for three orientations.

Figure 5(a). Heating ratios for the sides of a cross flow stable cylinder in continuum flow.

Figure 5(b). Heating ratios for the sides of a cross flow stable cylinder in continuum flow.

Figure 6. Heating ratios for the ends of a cross flow stable cylinder in continuum flow.

Figure 7. Surface averaged heating rate for a random tumbling and spinning Apollo heat source, orbital decay.

Figure 8. Surface averaged heating rate for a random tumbling and spinning Apollo heat source, -6.25° lunar return.

Figure 9.
Surface averaged heating
rate for a random tumbling
and spinning ALRH, -20°
lunar return.

Figure 10.
Surface averaged heating
rate for a random tumbling
and spinning Apollo heat
source, -38° lunar return.

Figure 11. Heating rate to ALRH (random-tumble and spin) for -90° lunar return reentry.

Figure 12. Heating rate to ALRH stagnation line (side-on-stable) for orbital decay.

Figure 13. Heating rate to ALRH stagnation line (side-on-stable) for -6.25° lunar return reentry.

Figure 14. Heating rate to ALRH stagnation line (side-on-stable) for -38.05° lunar return reentry.

Figure 15. Heating rate to ALRH stagnation line (side-on-stable) for -90° lunar return reentry.

Figure 16. Modified one dimensional model of Apollo heat source.

Figure 17. Outer temperature of Apollo heat source ATJ graphite heat shield, random tumbling and spinning -6.25° lunar return.

Figure 18. Outer temperature of Apollo heat source Ti strength member, random tumbling and spinning -6.25° lunar return.

Figure 19. Temperature of Apollo heat source PT-10Rh liner, random tumbling and spinning -6.25° lunar return.

Figure 20. Temperature of outer Apollo heat source Ta-10W capsule, random tumbling and spinning -6.25° lunar return.

Figure 21.
Outer temperature of Apollo
heat source fuel, random
tumbling and spinning -6.25°
lunar return.

Figure 22. Seven lb/ft³ fibrous carbon
in argon (Ref. 22).

Figure 23. Maximum temperatures of the fuel and titanium in space as functions of insulation conductivity multipliers, 7 lb/ft³ fibrous carbon as the reference.

Figure 24. Maximum reentry temperatures of the titanium during orbital decay and -6.25° lunar return as functions of insulation conductivity multipliers, 7 lbs/ft³ fibrous carbon as the reference.

Figure 25. Maximum earth impact temperatures of the titanium during orbital decay and -6.25° lunar return reentries as functions of insulation conductivity multipliers, 7 lbs/ft³ fibrous carbon as the reference.

Figure 26. Predicted ALRH orbital decay and heat facility test temperatures.

Figure 27. Comparison of observed and predicted peak temperatures of the titanium during the orbital decay test in the heat facility for various insulation conductivity multipliers.

Figure 28. Comparison of the conductivity of carbonized and graphitized CFG-4 fibrous insulation.

Figure 29.
Temperature distributions across
an ATJ graphite heat shield during
a random tumbling and spinning
-38° lunar return reentry.

Figure 30. Temperature distributions
across a pyrolytic graphite
ablation shield during a
-38.01° lunar return random
tumbling and spinning reentry.

Figure 31. Temperature distributions across a POCO AXF - 5Q graphite heat shield during a random tumbling and spinning - 38° lunar return reentry.

Figure 32. Temperature distributions across a POCO AXF-5Q graphite heat shield during a random tumbling and spinning - 6.25° lunar return reentry.

Figure 33.

Temperature distributions across a POCO AXF - 5Q graphite heat shield during a random tumbling and spinning -20° lunar return reentry.

Fragment Shield Assembly

Figure 34. Heat shield components.

- (1) Insulator, 7 lb/ft³ fibrous carbon, 0.15 inch thick
- (2) Ablation shield, POCO AXF-5Q graphite, 0.60 inch thick
- (3) Outer container, 304 stainless steel, 0.03 inch thick
- (4) Base plate, 6061-T651 aluminum, 0.18 inch thick
- (5) Rivet, solid universal head
- (6) Fragment shield, titanium, 0.25 inch thick
- (7) Clad, platinum-10 rhodium, 0.01 inch thick
- (8) Outer liner, tantalum-10 tungsten, 0.02 inch thick
- (9) Inner liner, tantalum-10 tungsten, 0.02 inch thick
- (10) Fuel, $^{238}\text{PuO}_2$ microspheres

Figure 35. Cross section view of final ALRH design.

Figure 36. Reentry temperatures of an ALRH without the heat shield during an orbital decay reentry.

Figure 37. Reentry temperatures of an ALRH without the heat shield during a -38° lunar return reentry. Fuel release conditions $V = 34,000$ ft/sec, Alt. = 160,000 ft.

Figure 38. Reentry temperatures of an ALRH assuming heat shield removal at peak heating during a -38° lunar return reentry. Fuel release conditions $V = 17,100$ ft/sec, Alt = 97,800 ft.

Figure 39. Reentry temperatures of an ALRH without the heat shield during a -90° lunar return reentry. Fuel release conditions $V = 34,300$ ft/sec, Alt. = 146,000 ft.

Figure 40.
Reentry temperatures of an ALRH assuming heat shield removal at peak heating during a -90° lunar return reentry.

Figure 41. Apollo launch pad abort fire temperatures.

Figure 42. ALRH temperatures in the standard URS Apollo pad abort fire.

Figure 43. ALRH temperatures in the standard Sandia Apollo pad abort fire.

Figure 44. ALRH temperatures in the extended Sandia Apollo pad abort fire.

APPENDIX A

One-Dimensional Thermal Models of Uniformly Heated Bodies with Layers of Uniform Thickness

The thermal response of many objects can be predicted accurately with a one-dimensional thermal model if the surface heating or cooling rates are uniform. Examples of uniform surface heat transfer are: (1) Operation of a symmetric heat source in space, (2) reentry aerodynamic heating of a random tumbling and spinning body, and (3) heating in a launch pad abort fire. Averaged surface heating rates and a one-dimensional model can be used to accurately predict transient temperatures inside a high resistance insulation for configurations such as the ALRH, even if the heating rates actually vary greatly over the surface of the vehicle. The type of one-dimensional thermal model discussed in this appendix is also useful during the early phases of a design when parametric studies and investigations of alternate designs require a model that can be easily changed, uses little computer time, and is a reasonably accurate representation of the physical model.

The conventional method of one-dimensional thermal modeling is to use a small section of the physical model. Radial heat transfer in uniformly heated spheres can be accurately predicted using spherical conduction equations and appropriate radiation area and emissivity factors, but this is the only finite configuration that can accurately be represented with a one-dimensional physical model. One-dimensional cylindrical coordinate heat transfer equations applied to a cylindrical physical model and one-dimensional cartesian coordinate equations applied to finite plates or discs do not accurately predict temperatures because corner and end or edge heat transfer is not included. The magnitude of the errors that are introduced by using a section of a physical model and one-dimensional cartesian or cylindrical coordinates can be demonstrated by comparing the actual areas perpendicular to heat transfer of concentric cylinders to the areas that are assumed using one-dimensional physical models. Figure A-1 shows cross section views of the concentric cylinders that will be used for this comparison. The dashed lines represent the surface areas of the concentric layer as functions of radius (R) and longitudinal distance from the center (X). Figure A-2 gives the normalized surface areas of concentric cylinders with length to diameter ratios greater than or equal to one, and Figure A-3 gives the normalized surface areas of concentric cylinders with L/D ratios less than or equal to one.

Surface areas of infinitely long concentric cylinders (Figure A-2) are proportional to the radius of the cylinders the same as are the surface areas in one-dimensional cylindrical coordinates. However, the surface areas of concentric cylinders with L/D ratios of one are proportional to the radius squared. Figures A-1 and A-2 show that the areas of the longer cylindrical heat sources used in most space nuclear power supplies cannot be accurately represented by a cross section of the cylinder. Therefore, one-dimensional cylindrical coordinate models cannot be used to accurately predict temperatures of uniformly heated finite cylinders.

Surface areas in one-dimensional cartesian coordinates are constant and are applicable only for thin discs. Figures A-1 and A-3 show that a constant area assumption is not justified even for a uniformly heated cylinder with an L/D ratio as low as 0.25.

Although the ALRH thermal model was depicted as a cone (area proportional to X^2 or R^2), it is not necessary to have a drawing of the thermal model. However, a plot of surface area versus the distance from the center similar to Figures A-2 and A-3 is helpful for selecting node sizes and computing radiation area factors, conductances, convection coefficients, internal generation, and heat capacities. Figure A-4 shows such an area plot for the one-dimensional ALRH thermal model. The area under the curve represents the volume of the node. The curves will be smooth for concentric configurations, but there will be discontinuities across the gaps in nonconcentric bodies.

After a table of thicknesses, volumes, and surface areas for each node is completed based on the dimensions of the entire layer, the modeling continues the same as for any one-dimensional physical model. This technique can be applied to most uniformly heated bodies with layers of uniform thickness. Large differences in the shapes of the different layers may preclude the use of any one-dimensional thermal model.

Figure A-1. Surfaces of closed concentric cylinders as functions of R or X .

Figure A-2. Normalized surface areas of enclosed concentric cylinders as functions of the normalized radii for L/D ratios ≤ 1 showing end effects.

Figure A-3. Normalized surface areas of enclosed concentric cylinders as functions of the normalized longitudinal distance from the center of the cylinder for L/D ratios ≥ 1 showing circumferential edge effects.

Figure A-4. Area plot of the one-dimensional ALRH thermal model.

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