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APOLLO LUNAR RADIOISOTOPIC HEATER
AEROTHERMODYNAMIC SUMMARY REPORT

Reentry and Space Sciences Division
Aerospace Nuclear Safety Department 9510

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APOLLO LUNAR RADIOISOTOPIC HEATER AEROTHERMODYNAMIC SUMMARY REPORT

Reentry and Space Sciences Division
Sandia Laboratories, Albuquerque

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ABSTRACT

This report summarizes results of the aerodynamic, trajectory, ablation, thermal, and stress analyses completed on the Apollo Lunar Radioisotopic Heater (ALRH) between December 1968 and February 1969. Results of the experimental program conducted during the same time period in the Sandia High Enthalpy Arc Tunnel and the NASA/MSFC 1.5 megawatt facilities are included.

March 1969

The cutoff date for information in this report is March 1969.

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SUMMARY

Aerothermodynamic studies were conducted to determine the ability of the Apollo Lunar Radioisotopic Heater (ALRH) to withstand environments resulting from aborts during various phases of the Apollo mission. The analyses included: (1) definition of aerodynamic models for the ALRH, (2) trajectory calculations for Earth orbital decay reentry and lunar return reentries at reentry angles from 0° to 90° , (3) calculation of multi-orbit reentries, (4) prediction of maximum graphite removal by ablation for all trajectories considered, (5) construction of one- and three-dimensional ALRH thermal models, (6) ALRH internal temperature predictions during reentry and launch abort environments, and (7) thermal stress calculations. The experimental program, which paralleled the analytical studies, was conducted in the Sandia High Enthalpy Arc Tunnel and the NASA/MSC 1.5 megawatt facilities. In these tests, the ALRH was successfully subjected to approximate earth orbital decay and -6.25 degree lunar return reentry environments.

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APOLLO LUNAR RADIOISOTOPIC HEATER AEROTHERMODYNAMIC SUMMARY REPORT

Introduction

The Apollo Lunar Radioisotopic Heater (ALRH) is designed to provide thermal energy during the lunar night to protect an instrument package intended to be left on the lunar surface. The heater provides fifteen watts of thermal energy by utilizing the isotopic decay of radioactive material. Two heaters are used in the instrument package.

An aerothermodynamic analysis of the ALRH was conducted by the Reentry and Space Sciences Division of Sandia Laboratories between December 1968 and February 1969. The work was completed to determine the ability of the heat source to contain its fuel inventory in reentry and launch environments resulting from possible aborts during various phases of the Apollo Mission. Mound Laboratory was charged with ALRH design responsibility, while Sandia Laboratories provided the aerothermodynamic and safety test programs to evaluate the design and to provide input to the Safety Assessment Summary report to be written by AEC Headquarters. As the Sandia evaluation and test programs progressed, design changes leading to improved heat source capabilities were recommended to Mound Laboratory. These recommendations included definition of protective materials and their sizes and locations outside of the basic fuel container. The final ALRH design, incorporating the recommended changes, is shown in Figure 1.

The aerothermodynamic analysis was conducted within the framework established during the SNAP-27 program. Information from the SNAP-27 program was used directly in the ALRH program where applicable. The ALRH was evaluated against the reentry environments identified as severe by the SNAP-27 Reentry Working Group using methods considered appropriate by the group. The ALRH was subjected to test environments approximating earth orbital decay reentry and -6.25 degree lunar return reentry in the Sandia High Enthalpy Arc Facility and the NASA/MSC 1.5 megawatt arcjet facility respectively.

This report contains a description of each component part of the aerothermodynamic analysis and summarizes the results and conclusions drawn from the study.

Aerodynamic Analysis

The ALRH configuration analyzed was a right circular cylinder 3.02 inches long and 3.0 inches in diameter. The heater weighed 2.05 pounds without the mounting flange. The heater was assumed to be released free and unencumbered from the LM vehicle at 300,000 feet altitude. The LM disassembly sequence has been studied by General Electric, NASA/MSC, and Sandia in the SNAP-27 program. The NASA/MSC data (Reference 1) describing LM aerodynamics and disassembly were used in the ALRH study.

Following release of the ALRH from the LM vehicle at 300,000 feet, free molecule flow theory was used to compute the free molecule hypersonic

ballistic coefficient of the heater for random tumble, crosswise, and end-on orientations. These calculations were based on a molecular speed ratio of 14 and a surface temperature of 730°R. Modified Newtonian flow theory, with corrections for three-dimensional effects, was used to compute the continuum hypersonic ballistic coefficient of the heater for the same three heater orientations. Computer Code HRS015 was then used with the Sandia QUIKTRAN remote terminal and the IBM 7040/44 in Los Angeles to calculate the transition flow hypersonic ballistic coefficient of the heater for these three orientations. The results of these calculations are presented in Figure 2.

Trajectory Analysis

The ALRH aerodynamic data shown in Figure 2 were used with existing LM aerodynamic data as input to trajectory calculations. The LM aerodynamic models utilized the NASA/MSFC constant ballistic coefficients of 79.6 pounds per square foot for the combined ascent and descent stages and 111.8 pounds per square foot for the descent stage only for crosswise orientations.

The above aerodynamic models were used with the Sandia TTA trajectory code and the Sandia CDC 3600 computer to calculate a number of trajectories. The pertinent LM configuration aerodynamic model was used down to an altitude of 300,000 feet, with the heater aerodynamic model used down to impact. The pertinent orbital decay LM configuration is descent stage only, while the pertinent supercircular velocity reentry LM configuration is the ascent and descent stages.

Computer code HRS004-7 was used with the Sandia QUIKTRAN remote terminal and the IBM 7040/44 computer in Los Angeles to compute the initial conditions at 400,000 feet altitude for the orbital decay trajectory. The TTA trajectory code was then 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.

The computer outputs for these trajectory calculations list pertinent aerodynamic and heating parameters as a function of reentry time from 400,000 feet altitude. The computer listings are available for reference; due to their bulk, they are not included in this report.

Multiple Revolution Orbital Decay Analysis

Entry into the Earth's atmosphere at supercircular velocities at an angle too shallow to cause prompt reentry results in highly elliptical trajectories with initial apogees at near lunar distances. The subsequent decay and circularization of these orbits results in many passes of the ALRH through the Earth's atmosphere prior to its final reentry. Computer code HRS017-3 was written to provide an approximate solution to the multiple orbit decay problem. The constant perigee solution predicts a higher number of revolutions prior to final reentry than would be obtained by "exact" integration of the equations of motion. The approximate method, however, requires approximately 1/100 of the computer time and provides punched card input for use in ablation analyses.

Computer code HRS017-3 was used with the Sandia CDC 6600 and the UNIVAC 1108 to compute the trajectory and reentry environment experienced by the heater reentering crosswise (side-on-stable)* during multiple revolution orbital decay. Typical heating rate pulses expected during the 280,000 foot perigee case are shown in Figure 3. Perigee altitudes and the resultant number of revolutions for which the calculations were completed are as follows:

<u>Perigee Altitude</u>	<u>Number of Revolutions</u>
300,000 feet	62
290,000 feet	39
280,000 feet	25
270,000 feet	15
260,000 feet	7

As previously mentioned, in analyzing the multiple revolution problem the ALRH was considered to be in the side-on-stable orientation at all times during the trajectory. This allows the ablation analysis to consider material removal from the same stagnation line of the ALRH for each pass through the Earth's atmosphere, thereby maximizing the amount of material removed. While the ALRH is not expected to rigorously adhere to this flight attitude, this technique does provide a conservative ablation analysis.

Ablation Analysis

The ablation which occurs on the ALRH was determined by using Sandia's Charring Material Ablation (CMA) code (Reference 2). The CMA code uses an implicit, finite-difference computational procedure for computing the one-dimensional transient transport of thermal energy in a three dimensional isotropic material which can ablate from the front surface and decompose in depth.

The time varying inputs (heat rate, enthalpy, and pressure) to the CMA code were obtained as punched cards directly from computer code HRS017-3 for the multiple orbit reentry cases. For the prompt reentry conditions, computer code HEP (Heating, Enthalpy, Pressure), uses the output from the TTA trajectory code to prepare the necessary punched cards for input to the CMA code.

*In the crosswise orientation, the heater may be either stable (side-on-stable) or rotating.

The results of the CMA code computation are summarized in Figures 4, 5, and 6. Figure 4 indicates the ablation which would occur on the ALRH if a mission abort occurred following translunar injection and resulted in a multiple revolution reentry. For multiple passes through the Earth's atmosphere in a side-on-stable orientation, the amount of graphite removed is shown as a function of the number of orbits prior to a final reentry. As shown, the maximum ablation would occur if the ALRH experienced approximately 25 orbits before final reentry. This corresponds to an initial perigee of 280,000 feet. Increasing the initial perigee beyond 280,000 feet results in more orbits but less ablation since the ablation per orbit at higher perigee altitudes drops off sharply.

Figure 5 shows the ablation results for a side-on-stable prompt reentry as a function of reentry angle. As shown, the maximum ablation occurs for a -6.25° lunar return reentry. It should be noted that the predicted ablation for the side-on-stable reentry orientation is conservative since the ablation model allows heat transfer in only the radial direction; in reality some three-dimensional thermal relief would occur. This thermal relief would reduce the stagnation line temperature and consequently reduce the ablation.

Figure 6 shows the ablation results for a random tumble and spin prompt reentry as a function of reentry angle. For this case the maximum ablation occurs during an orbital decay reentry.

Thermal Analysis

Aerodynamic Heating

Heating distributions over the surface of the ALRH and the ratio of average heating rates to a reference heating rate for continuum flow were based on wind tunnel tests conducted at Cornell Aeronautical Laboratory and at Rhodes and Bloxson. The tests measured the heating rates to cylinders at angles of attack from 0° to 90° in hypersonic flow (References 3 and 4). The ratio of average heat rates to the ALRH to the stagnation point heating rate of a sphere of the same radius as the ALRH are given in Table I for various angles of attack. Using the methods derived in Reference 5, the heating ratios in Table I were averaged for random tumbling and spinning and for side-on-spinning motion. The results are given in Table II.

The reference heating rates for each trajectory from the TTA trajectory program were multiplied by $1/\sqrt{R}$ and the above ratios to obtain cold wall heating rates for the ALRH. The cold wall heating rates were used as the input to the CMA ablation program which computed the surface recession and the net aeroheating rates corrected for hot wall, mass injection into the boundary layer, and heat of combustion. The cold wall and corrected surface averaged heating rates for a random tumbling and spinning ALRH during orbital decay and -6.25° , -20° , and -38° lunar return reentries are given in Figures 7 through 10. The corrected heating rates were used in the modified one-dimensional thermal model to predict temperatures and any disassembly of the ALRH during reentries and launch pad abort fires.

Table I
Average Heating Ratios to the ALRH
($\alpha = 0$ 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	0.193

Table II
Heating Ratios for a Tumbling ALRH

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

One-Dimensional Thermal Model

During the 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 11 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). 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. As an example, the predictions of interior temperatures made with the conical model were within 5°R of the temperatures predicted with the 426 node three-dimensional model.

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 (Reference 6 through 16) were programmed as function of temperature.

The conical model, used with the HERMAN heat transfer computer program, was used to compute operating temperatures, reentry temperatures, and temperatures in a launch pad abort fire (Reference 17).

Alternate Design Study

The first Mound design analyzed employed an ATJ graphite heat shield and contained no insulation. A preliminary study of this reference design indicated that the titanium and Pt-10Rh would melt during a -6.25° lunar return reentry. This study also showed that the -6.25° lunar return was the most severe for the reference design.

Modifications to the reference design were investigated in an attempt to reduce the internal temperatures during reentry. The following restrictions were placed on the modifications:

1. The weight and exterior dimensions could not be increased,
2. The titanium thickness could not be reduced,
3. No changes could be made to the Pt-10Rh or Ta-10W containers,
4. New materials had to be readily available and well characterized, and
5. No materials or techniques requiring any development could be considered.

Three modifications were considered.

1. Placing 0.05-inch radiation gaps on each side of the titanium with oxidized metal surfaces.
2. The same as (1) but with unoxidized surfaces.
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. A 12 lb/ft³ fibrous carbon felt was selected for this 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 parametric study. Table III 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 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: Radiation gaps are not an efficient method of insulation because of the high resistance at low (operating) temperatures and the lower resistance at high (reentry) temperatures. This modification results 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.

3. Insulation Layer: The insulated heater has the lowest temperature gradients in the graphite and lowest interior temperatures during reentry. The titanium remains 1820°R below melt and the Pt-10Rh 2120°R below melt.

Table III

Maximum Temperatures, Temperature Gradients, and
Ablation of the Apollo Heat Source
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

Figures 12 through 16 show the large difference in the temperatures of the reference and insulated designs during reentry. The insulated configuration 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 IV, the insulated design is superior in all categories during reentry.

Table IV

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

Fibrous carbon with a density of 7-lb/ft³ was found to have lower thermal conductivity and be more resilient than the 12-lb/ft³ fibrous carbon. Since the lower density insulation was characterized up to 5220°R and procurement was not a problem, it was selected as the best insulation for this application. This elastic insulation also acts as a compliance member between the graphite and titanium. Since the

insulation reduces interior hot spots, the insulated heater is not as orientation-dependent as the other configurations. The insulated heater 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. Because of these advantages, the 7 lb/ft³ fibrous carbon insulation was incorporated in the design.

Conductivity Parametric Study

A parametric study was conducted to determine the operating and reentry temperatures of the ALRH for orbital decay and -6.25° lunar returns with insulation conductivities ranging from 0.1 to 100 times that recommended for the 7-lb/ft³ fibrous carbon in one atmosphere of argon. The fibrous carbon is a relatively new material and hence rigid quality control has not yet been achieved, and the effects of configuration and all environments are 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 shows the safety factor in the insulated design, and it provides a rapid means of finding the critical temperatures of the ALRH after the thermal resistance of the insulation is found experimentally. If it was necessary to select an alternate insulation for any reason, this study also shows the range of conductivities that would be acceptable.

Figure 17 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 18 shows the peak reentry temperatures of the titanium. The insulation thickness is near optimum. The conductivity could be increased by a factor of 15 without causing the titanium to melt during reentry. Figure 19 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 (reported in a later section of this report) was conducted in the Sandia HEAT facility. Heat sensitive paint was applied to the titanium before the test. Posttest reading of the paint showed that the average peak temperature of the titanium was between 1800°R and 2200°R.

The internal heat generation was removed from the modified one-dimensional model, and the surface averaged heating rate of the plasma jet test was programmed into the model. This program was run with insulation conductivities ranging from nominal to 3 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 2 (see Figure 20). Since the apparent conductivity of the insulation is approximately twice that predicted, the operating, reentry, and impact temperatures can be found from Figures 17 through 19 by reading the temperature for a conductivity multiplier of 2. As an example, peak titanium temperatures would be 2020°R during orbital decay and 1710°R during -6.25° lunar return.

Temperature Gradients and Reentry Without the Heat Shield

Temperature gradients in an ATJ graphite heat shield at various times during a -38.05° lunar return reentry are shown in Figure 21. A thermal stress analysis (reported in a later section of this report) using these gradients indicated that the heat shield would fail in this environment. The heat shield material was then changed to the stronger POCO AXF-5Q graphite. Temperature gradients in the AXF-5Q graphite are shown in Figures 22, 23, and 24 for -6.25° , -20° , and -38.05° lunar return reentries.

Since the thermal stress analysis predicted the AXF-5Q graphite would also fail during a -38.05° lunar return, the reentry temperatures of the heat source with the graphite heat shield removed were computed. The exact time of graphite failure was not known, so a bracketing technique was used with removal assumed graphite before reentry and at the time of peak heating. Figures 25 and 26 show that the Ta-10W melts prior to peak heating and releases fuel in the melt zone for both -38.05° and -90° lunar returns if the heat shield is removed prior to reentry. Figure 27 shows the Ta-10W just reaches melt temperature during a -38.05° lunar return if the heat shield fails at the time of peak heating, but the fuel is still released in the melt zone. If the heat shield fails at the time of peak heating during a -90° lunar return, the fuel would not be released before earth impact (Figure 28).

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

Three-Dimensional Thermal Model

As mentioned previously, there is a possibility that the ALRH will reenter the Earth's atmosphere in a side-on-stable orientation. This reentry mode produces a large temperature variation over the ALRH surface as well as through the material thickness. In order to determine what this temperature variation will be and what effect it will have on the system as it reenters, it was necessary to construct a three-dimensional thermal model. The required model was constructed for use on the Chrysler Improved Numerical Differencing Analyzer (CINDA) computer code. CINDA is a multi-option systems compiler computer program, operational on the Sandia UNIVAC 1108. It analyzes a mathematical model of any arbitrary one, two, or three-dimensional lumped parameter representation of a physical system governed by a set of diffusion equations. To utilize the code, a thermal analog network representing the physical system must be constructed. Nonlinear material properties and boundary conditions may be input as functions of one or more independent variables.

Figure 29 shows a sketch of the ALRH thermal model in two planes. The model consists of 426 nodes as follows:

<u>Material</u>	<u>Number of Nodes</u>
Graphite	300
Insulation	36
Titanium	30
Platinum -10 Rhodium	12
Tantalum -10 Tungsten	24
Fuel ($^{238}\text{Pu O}_2$ Microspheres)	24
Total	426

The stainless steel container was not included in the thermal model since it will have little effect upon the thermal response of the system during reentry.

The thermal model allows for heat transfer by both conduction and radiation. All radiating surfaces are assumed to be grey with temperature varying emissivities. All material properties are allowed to vary with temperature, and the internal energy generation is included in the fuel nodes. The reentry aerodynamic heating is accounted for by applying a time-varying heat generation to the nodes on the external surface. Each surface node has a different aerodynamic heating rate applied to it to account for the heating distribution which occurs over the ALRH as it reenters the atmosphere in a side-on-stable orientation. For steep lunar return reentries, the effects of ablation on the heating rate (from the CMA computer code) are taken into account; however, for orbital decay and -6.25° lunar return reentries, the ablation effects on the heating rate are not accounted for.

The results of the three-dimensional thermal analysis are available for reference as computer output, where the temperature as a function of time is printed for each node for the following trajectories:

1. Orbital decay reentry.
2. -6.25° lunar return reentry.
3. -38° lunar return reentry.
 - (a) cold wall heat input
 - (b) CMA heat input
4. -90° lunar return reentry
 - (a) cold wall heat input
 - (b) CMA heat input

Due to the bulk of these computer printouts, they are not included in this report; however, a representative sample of the results is shown in Figures 30 through 37. Figures 30 and 31 show the temperature distribution near the center plane for orbital decay in the circumferential and radial directions, respectively. The maximum surface temperature, the maximum circumferential temperature gradient, and the maximum radial temperature gradient can be obtained from these figures. Figures 32 through 37 show the same information for -6.25° , -38.05° , and -90° lunar return reentries. The time in each of the figures is referenced from 400,000 feet altitude.

Since the maximum internal temperatures of the ALRH are reached for random tumble and spin reentries rather than for side-on-stable reentries, and since these temperatures were readily obtained through the modified one-dimensional thermal model, the maximum internal temperatures reached are not shown in this section of the report.

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 selection may have been unrealistically severe. 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, and an extended Sandia model. The time-temperature plots of these fires are shown in Figure 38. The extended Sandia pulse is not believed to be possible, but it was included to dispell 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 39, 40, and 41. 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.

Thermal Stress Analysis

Thermal stress analyses were completed for the ALRH in random tumbling and side-on-stable orientations during lunar return reentries at angles of -6.25°, -20°, and -38°. In addition, the heater was analyzed for test condition A, the rectangular pulse at 533 BTU/ft^{3/2}-sec for 55 seconds which is used in the NASA/MSFC 1.5 megawatt facility. The thermal stresses were calculated by the use of a finite element program (ARBAS 2) on the CDC 6600 computer.

ATJ graphite was analyzed for the random tumble, -38° reentry case early in the program. However, the published properties of POCO AXF-5Q graphite suggested potential gains in the use of that material with little risk of performance lower than that of ATJ. POCO AXF-5Q was, therefore, selected as the ALRH heat shield material.

For the random tumble and spin case, the model used was a cylinder; it was assumed that during heating the plugs in each end became free and did not add to the stress state of the cylinder. The stresses in the

plugs were obtained and are lower than those in the cylinder. The maximum stress in the cylinder is a hoop stress which is located at or near the center of the cylinder. The results of the random tumbling analysis are shown in Figures 42 through 49. An average margin of safety based on a factor of safety of 1.0 on ultimate was used to compare the two materials. Also, a margin of safety based on a factor of safety of 1.0 on the average strength minus three times the standard deviation of the strength variation was used. Table V summarizes the results.

Table V
Results of Thermal Stress Analysis of Random
Tumble and Spin Reentry Mode

<u>Reentry Angle</u>	<u>Material</u>	<u>Time During Reentry (seconds)</u>	<u>Max. Stress (psi)</u>	<u>Msu Average</u>	<u>Msu 3σ</u>
-6.25°	POCO	65	6,300	+0.236	-0.11
-20°	POCO	22	11,000	-0.33	-0.91
-38°	POCO	14	12,200	-0.48	-1.10
-38°	ATJ	14	5,000	-0.33	-0.90

The Msu values for POCO were calculated from the material properties in Reference 19. The values of Young's modulus, Poisson's ratio, and the coefficient of thermal expansion as a function of temperature were supplied by the manufacturer. The properties for ATJ were taken from Reference 20.

The side-on-stable reentry mode was analyzed by modeling a ring under the conditions of plane strain with finite elements. The temperature profile at the center of the cylinder was used. Figures 50 and 51 show the maximum thermal stress for the -6.25° and -38.05° lunar return reentry angles and for test condition A. Table VI lists the maximum tension stresses and margins of safety.

Table VI
Results of Thermal Analysis of
Side-on-Stable Reentry Mode

<u>Condition</u>	<u>Material</u>	<u>Time During Reentry (seconds)</u>	<u>Max. Stress (psi)</u>	<u>Msu Average</u>	<u>Msu 3σ</u>
-6.25°	POCO	75	7600	+0.078	-0.32
-38.05°	POCO	12	16700	-1.02	-1.89
Test A	POCO	7	11600	-0.41	-1.02

Because the cylinder is finite in length, the effect of end conditions on the hoop stress was also examined. A ring was modeled and stressed by an axisymmetric (i.e. random tumble and spin) temperature profile at the plane where the maximum stress occurs in the cylinder. The conditions of plane stress and plane strain were used. The stresses in the finite cylinder were 145 percent higher than the plane strain condition, and the plane strain condition produced stresses which were 129 percent higher than the plane stress condition. The values in

Table VI do not reflect the additional stress due to the end conditions of the cylinder. The amount of increase for the side-on-stable condition is a matter of judgment because the maximum thermal moment does not exist around the entire circumference as it does in the case of the random tumble and spin condition. The magnification factor would be between 1.0 and 1.45.

Experimental Program

As part of the overall evaluation and demonstration of the safety of the ALRH, an experimental test program was conducted to provide an estimate of the ALRH response to selected reentry environments. This section of the report describes three series of arc jet tests which were run to supplement the reentry aerothermodynamic analysis of the ALRH.

Reentry Environments

The conducting of a meaningful test program presupposes a knowledge of the environments of concern. Preceding sections of this report have described the aerodynamic and trajectory characteristics of the ALRH which provided the required reentry environment.

The aerothermal parameters which describe the reentry environment are illustrated in Figures 52 through 54. Figure 52 shows the variation of body stagnation pressure and total gas enthalpy along selected reentry trajectories following heater release at an altitude of 300,000 feet. The region of high aerodynamic heating is illustrated by indicating the point of maximum convective heating rate as well as the earlier and later times where the heating rate is one-half the peak value for each trajectory. Figures 53 and 54 illustrate the time history of the convective heating rate following release of the heater for orbital decay and supercircular reentry trajectories, respectively. The heating rate shown is a reference value equivalent to the rate experienced by a one-foot radius sphere flying the same trajectory. To obtain the actual rates to the ALRH, the values shown must be scaled for size and configuration. Inspection of the figures indicates that the reentry environment is essentially unaltered by the two modes of motion considered.

The -6.25° supercircular trajectory represents the nominal reentry following lunar return. However, a finite probability is attached to reentry at steeper angles. These conditions are represented by the reentry angles from -20° to -90° .

Test Facilities

Simulation of the supercircular environments utilizing a full scale ALRH is beyond the capability of existing ground test facilities for those reentry angles steeper than the nominal. Therefore, two test series were scheduled to provide simulation of the earth orbital decay trajectory and the nominal lunar return trajectory.

The Sandia High Enthalpy Arc Tunnel (HEAT) facility was selected to simulate the orbital decay reentry trajectory heating. This facility

is unique in that both the power and the gas flow rate customarily are programmed during a run to provide a variation in the heating rate and pressure analogous to that experienced along the trajectory. Simulation of the supercircular trajectories is beyond the present capability of this facility, however.

Arcjet facilities at the NASA Manned Spacecraft Center are capable of achieving heating rates to a full scale ALRH which are equivalent to the peak value experienced on the -6.25° supercircular trajectory. This heating rate is achieved at body pressures approximating those resulting from this trajectory. Therefore, the assistance of NASA was solicited in carrying out a test series in these facilities to provide simulation of the nominal lunar return trajectory.

Test Description

Four specimens for aerothermal tests were supplied by Mound Laboratories. The stainless steel cannister was removed from each test unit. The first of these specimens (FB-1) was fabricated with ATJ graphite as the heat shield material. The remaining three units (FB-2, FB-3, and FB-4) utilized POCO graphite as the heat shield material. All four units used ZrO_2 microspheres as a simulant for the active fuel material.

It was noted above that the reentry environment was not influenced by the two modes of motion considered. Since the side-on stable orientation results in greater surface recession during reentry, this orientation was chosen for test.

Test Series A

Specimen FB-1 was tested in the Sandia HEAT facility to a simulated orbital decay reentry trajectory. The unit was supported from the base region on a tantalum sting and was oriented in the crossflow mode. The reference heating rate pulse and body stagnation pressure history achieved in the test are shown in Figure 55. Also shown for comparison are the corresponding orbital decay trajectory values. It is apparent that a good simulation of the orbital decay trajectory was achieved. At the termination of the run, the test chamber was maintained in its evacuated state. The chamber pressure was then allowed to slowly increase to the ambient value over a 300-second time interval. This pressure buildup simulates the rate of pressure increase on the free-fall phase of the trajectory.

Unit FB-1 was not instrumented for this test. Data gathered consisted of the dimensional and weight changes resulting from the run and motion picture coverage of the run. The unit survived the test intact. Test results are discussed in a later section of this report. Heat shield ablation results for all test specimens are summarized in Table VII. This test unit was subjected to comprehensive posttest examination of the fragmentation shield and the Ta-10W fuel containers. The results of this examination are reported elsewhere.

Test Series B

This test series was conducted in the 1.5 megawatt arc jet facility at the NASA Manned Spacecraft Center. The series was intended to demonstrate the heater response to the -6.25° supercircular reentry

Table VII
Summary of Ablation Data

Test Specimen	FB-1	FB-2	FB-3(A)	FB-4	FB-3(B)
Heat Shield	ATJ	POCO	POCO	POCO	POCO
Test Series	A	B	B	B	C
Heating Pulse	Orbital Decay	B	A	B	Orbital Decay
Mass Loss, grams	28.1	21.55	8.06	19.50	40.2
Percent of Original Weight	4.5	3.3	1.25	3.0	6.3
Surface Recession, inches @ the Stagnation Line	0.121-0.130	0.091-0.108	0.018-0.027	0.085-0.093	0.141-0.160
90° from Stag Line	0.0054-0.006	0.0045-0.0055	0.003-0.0073	0.003-0.0083	

trajectory. The short time scales under which this test program was conducted precluded the development of a continuously varying programmed pulse. Therefore, two different heating rate pulses were utilized. The heating rate and stagnation pressure pulses are shown in Figures 56 and 57, respectively. The reentry environment parameters for each of these pulses are given in Table VIII. Pulse A provides a constant reference heating rate of $533 \text{ BTU/ft}^{3/2}\text{-sec}$ which is equivalent to the peak value experienced on this trajectory. The time of exposure to this rate was selected to provide a time integrated heat input equivalent to that experienced on the trajectory. The body stagnation pressure for this run was 0.12 atmosphere which is 50 percent of the trajectory value at peak heating. Pulse B is a three-step pulse intended to approximate the -6.25° supercircular trajectory and to provide an over test of the total heat load. This pulse provides a time integrated heat input 36 percent greater than is incurred on this trajectory. Unit FB-3 was subjected to pulse A, while units FB-2 and FB-4 were subjected to pulse B.

All three test units for this series had the titanium fragmentation shield instrumented with temperature sensitive paints. These paints undergo a phase change at a precise, known temperature level. Each fragmentation shield had five different paint stripes around the cylindrical periphery and one stripe on each end. Thus, indicators of seven different temperature levels were available on each fragmentation shield; they covered a temperature range of 400°F to 800°F . In addition, a radiometer indication of the graphite surface temperature was provided. Dimensional and weight changes were recorded for each run in addition to motion picture coverage of each run.

Table VIII

Test Conditions for Series B

	Pulse A	Pulse B
Reference Heating Rate $\dot{q}\sqrt{R}$, $\text{BTU/ft}^{3/2} \text{ sec}$	533	40 secs @ 240 20 secs @ 533 82 secs @ 240
Body Stagnation Pressure PT2, atmospheres	0.12	40 secs @ 0.067 20 secs @ 0.12 82 secs @ 0.067
Gas Stagnation Enthalpy H_0 , BTU/lb	22,200	40 secs @ 14,000 20 secs @ 22,200 82 secs @ 14,000
Total Run Time, seconds	57	142

Test Series C

Although none of the tests had disclosed evidence of titanium fragmentation shield melting, test series B did indicate that the fragmentation shield reached temperatures higher than anticipated. Therefore, it was decided to rerun one of these test specimens. Specimen FB-3 was selected because it had experienced the minimum mass loss and surface recession. The highest fragmentation shield temperatures are anticipated on the orbital decay reentry trajectory; therefore, the test series A environment was chosen for this repeat run. For this run, specimen FB-3 was rotated 90° about the heater cylindrical axis so that the stagnation line for its second run was 90° removed from its first stagnation line.

The titanium fragmentation shield was sandblasted and repainted with seven new temperature indicating paints. For this test, paint melt-temperatures spanning the range from 900°F to 2500°F were used. In addition, a chromel-alumel thermocouple was spot welded to the fragmentation shield. Motion picture coverage and dimensional change data were again provided.

Discussion

All test specimens appear to have survived the test environments intact with no particular adverse effects. The test of specimen FB-4 in test series B was terminated 11 seconds prior to the end of pulse B when the model support sting failed. This test specimen fell onto the water cooled shroud covering the model insertion mechanism and finally lodged against a water cooled model support arm. When this specimen was recovered, a radial crack through the graphite wall at the stagnation line was observed. This crack extended nearly the entire length of the model. Movies of this run failed to disclose the presence of the crack at the time of sting failure. Specimen FB-2 was tested to the same pulse and survived the test intact. No adverse effects on this specimen were noted. Therefore, it is concluded that the crack in specimen FB-4 resulted either from impact as a result of its fall or from the extreme cooling resulting from contact of the hot model with a water cooled surface.

The ablation results for each test run are tabulated in Table VII. A slight variation in the surface recession occurs over a length of the test specimen; therefore, a range of values from minimum to maximum is given. For the orbital decay simulation in the side-on orientation, approximately one-fourth of the heat shield wall is ablated at the stagnation line.

The titanium fragmentation shield survived each test run with no indication of melting. However, the three specimens run in test series B did reach temperatures on the fragmentation shield higher than were anticipated. Each of the three fragmentation shields were striped with seven different temperature indicating paints. The paints had melt temperatures spanning the range from 400°F to 800°F. All three fragmentation shields reached temperatures in excess of 800°F. In order to further explore the temperatures that are attained by the fragmentation shield, test series C was scheduled. Test specimen FB-3 had the fragmentation shield repainted for the temperature range of 900°F to 2500°F. In addition, this unit was instrumented with a thermocouple to obtain the temperature-time variation. No data were received from the thermocouple because it apparently failed at the time the model was inserted

into the flow. However, further data on fragmentation shield temperatures were provided by the paints on this run. The series of paints used indicated that along the stagnation line the titanium fragmentation shield reached 2000°F at the end opposite the threaded plug, while the end adjacent to the threaded plug did not reach 1700°F. Along the line 180° removed from the stagnation line (base of model) the temperatures of the fragmentation shield were considerably lower, apparently ranging from 900°F to 1000°F. At present, the higher temperatures reached by the titanium fragmentation shield are attributed to the insulating properties of the fibrous carbon insulation being lower than was originally anticipated.

Conclusions

1. The ALRH was subjected analytically to reentry environments considered severe by the SNAP-27 Reentry Working Group.
2. The ALRH graphite heat shield survived the multiple revolution reentries prior to the final revolution for all perigee altitudes. For perigee altitudes of 280,000 and 290,000 feet, the Ta-10W capsules were released below 144,000 feet altitude after peak heating in the final revolution.
3. The ALRH will contain the fuel inventory within the Ta-10W capsules for the two multiple revolution reentries where the graphite heat shield is ablated through. The radiation equilibrium temperature based on Ta melt temperature is not exceeded following release of the Ta-10W capsules.
4. The ALRH has adequate heat shield thickness to survive expected graphite removal from all prompt reentries, including earth orbital decay and lunar returns from -6.25° to 90°.
5. The temperatures of all internal components are predicted to remain well below melt temperature for all reentries provided the heat shield remains intact.
6. Thermal stress failure of the graphite heat shield is predicted analytically for lunar return reentries at angles in excess of -6.25°.
7. Release of the bare titanium capsule following thermal stress failure of the graphite heat shield leads to release of fuel in the melt zone.
8. Although thermal stress failures were predicted analytically for some cases, simulation of supercircular reentry trajectories at reentry angles steeper than the nominal -6.25° is beyond the present capability of ground test facilities. Therefore, it is not possible to demonstrate experimentally where the failure limit of the ALRH is for reentry environments.
9. 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. Apollo lunar radioisotopic heater

Figure 2.
Variation of hypersonic ballistic coefficient with altitude for the Mound 15-watt heater for three orientations

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Figure 36. ALRH thermal response during -90° lunar return reentry in side-on-stable mode, circumferential temperature distribution

Figure 37. ALRH thermal response during -90° lunar return reentry in side-on-stable mode, radial temperature distribution

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