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THERMAL ANALYSIS OF THE INTACT
SNAP-19/NIMBUS B/AGENA SYSTEM

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AEROSPACE NUCLEAR SAFETY

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ABSTRACT

This report presents the parametric safety studies conducted to aid in the design of the intact Snap-19 radioisotopic thermoelectric generator and the final aerothermal safety assessment of the generator on the Nimbus B/Snap-19 mission. The safety analysis includes ablation and temperature predictions for a launch pad abort fire and for abort and orbital reentries.

Key words: aerodynamic heating, ablation, reentry

The cutoff date for information in this report is February 1968.

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SUMMARY

As part of the Aerospace Nuclear Safety Program, Sandia Laboratories conducted an aerothermal safety assessment of the intact Snap-19 radioisotopic thermoelectric generator. The thermal response of three configurations of the heat source were computed for a launch pad abort fire resulting from the combustion of the Thorad and Agena fuels. Reentry analyses were conducted for the four most severe environments the generator would experience during any expected reentry of the Nimbus B system in any of the expected configurations. The results of this study showed that the intact Snap-19 would not experience a thermal failure in any of the environment studied.

In accordance with an AEC directive to locate safety problems early in the design of Snap systems and to recommend design changes, Sandia conducted a series of parametric studies and preliminary safety studies to aid the Martin Company in selecting the safest and most efficient design. These studies are summarized in this report.

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THERMAL ANALYSIS OF THE INTACT SNAP-19/NIMBUS B/AGENA SYSTEM

Introduction

The original SNAP-19 radioisotopic thermoelectric generator (RTG) was designed to melt and disperse the fuel during reentry in an altitude-velocity range where the fuel would not melt or break up. However, reentry studies^{1,2} showed that during some abort reentries, the fuel would be released in the melt zone. Feasibility studies conducted by Sandia Laboratories^{3,4} indicated that a modified SNAP-19 would contain the fuel to the time of earth impact for all expected reentries. The heat source was then redesigned by the Martin Company. The intact reentry heat source (IRHS) has the same external dimension as the dispersal heat source and can be used interchangeably with the dispersal capsule in the SNAP-19 generator.

An exploded view of the IRHS is shown in Figure 1. The exterior ablation shield is made of POCO AXF-5Q graphite and has a threaded center closure. The 570 watt load of plutonium dioxide microsphere fuel is contained in a completely filled Haynes-25 capsule which has a filtered vent at one end. Although the capsule melts during most reentries, the solid fuel and molten Haynes-25 are contained until the time of earth impact in a thin-walled tantalum cannister (liner) and the ablation shield. The cannister closure is a slip fit and offers no protection after earth impact. The cannister is supported on both ends by tantalum felt compliance pads. The inner surfaces of the POCO heat shield and the tantalum cannister are coated with zirconium dioxide.

Safety studies of the dispersal and intact heat sources conducted by Martin and Sandia led to the selection of the IRHS for the Nimbus B mission. Figure 2 shows the mounting of the two SNAP-19 generators on the Nimbus B. Figure 3 is a photograph of the SNAP-19 subassembly on the mounting bracket.

The purpose of the safety study presented in this report was to predict the thermal response of the intact SNAP-19 to the most severe environments it would experience during any expected reentry of the Nimbus B system in any of the expected configurations. This approach made it possible to assess the safety of the intact SNAP-19 with the minimum expenditure of time and money. The four trajectories and configurations analyzed were as follows:

- (1) Orbital decay of the generators on the Nimbus with an end-over-end tumbling motion followed by random tumbling spinning IRHS's after generator disassembly. This is the most probable reentry mode and is the basis for direct comparison with the dispersal SNAP-19.⁵
- (2) Orbital decay of a bare IRHS with a random tumbling spinning motion. This mode could occur if an explosion disassembled the generator before reentry and would result in the highest integrated heating and the most ablation of any unstable reentry. By comparison with (1) above, this trajectory shows that the thermal protection of the generator during the most severe reentry conditions is negligible.
- (3) Orbital decay of a bare IRHS with a side-on stable orientation. This orientation produces the highest local ablation and surface temperature and the highest circumferential temperature gradients.
- (4) Guidance failure abort of the generator on the Nimbus and Agena with an end-over-end tumbling motion followed by random tumbling spinning IRHS's after generator disassembly. The -2.026° reentry angle is the largest expected angle for powered reentry. This trajectory produces the highest reference heating rate, the highest rate of heating increase, and the highest average radial temperature gradient.

The end-on and end-over-end spinning orientations, which are not possible, and the less severe Agena thrust termination aborts were not analyzed. The thermal response of oscillating, coning, and side-on stable motions will be between that of the random tumbling spinning and side-on stable motions.

All of the trajectories and configurations were analyzed assuming helium gas in the generator with all interior metal surfaces oxidized. These conditions result in the minimum internal thermal resistance. Trajectories 1 and 4 above were also analyzed assuming internal vacuum and deoxidized internal metal surfaces for maximum internal thermal resistance.

Development Studies

During the development of the IRIS, Sandia conducted periodic preliminary safety analyses of the heat source to uncover any potential safety hazards before the design was finalized. Sandia also conducted several parametric studies to aid in selecting the safest and most efficient design. The parametric studies indicated the range of acceptable values for certain parameters. These development studies were reported in References 6 through 13 and were transmitted to the participating organizations. This section contains a brief summary of some of the earlier development-oriented studies that may be useful in future programs.

Effects of Internal Emissivities and Pressures on IRIS Temperatures (Reference 6)

A parametric study was conducted to determine the effects of internal pressure, internal emissivity factors, and radiation shields on the thermal response of an early intact SNAP-19 heat source. The purposes of the study were to aid in optimizing the capsule, point out configurations and coatings that should be avoided, and provide a preliminary safety assessment of the intact SNAP-19.

The configuration analyzed was defined by an unnumbered drawing supplied by the Martin Company, May 5, 1967. A cross section view of the heat source and generator is shown schematically in Figure 4. The outer dimensions of the heat source were 3.014 inches in diameter by 6.468 inches long. The outer POCO graphite ablation shield was 0.416 inch thick on the sides, with slightly thicker ends. The ablation shield was made in two pieces and was joined with a thread joint in the middle. The thread roots were 0.16 inch from the surface. The inner surface of the POCO was flame sprayed with zirconium dioxide. There was a 0.031 inch gap between the POCO and a Haynes-25 fuel container. The container was completely filled with fuel. The exterior surface of the container was nickel plated. Fibrous pads on each end of the container compensated for differential thermal expansion.

For this configuration, the radiation area factor (F_A) between the Haynes-25 and graphite is approximately one and the emissivity factor (F_ϵ) is given by Christiansen's equation. The range of emissivity factors investigated was 0.05 to 0.80, which encompasses the values obtainable with most existing materials or coatings. An equation was derived to compute an effective emissivity factor with radiation shields present. This equation permits the results of this study to be used if one or more radiation shields are placed between the Haynes-25 and graphite. Pressure in the generator has a large effect on thermal resistance and, as a result, it is an important factor in determining the temperature of the capsule. The following three pressure combinations were analyzed

- (1) One atmosphere of argon in the capsule throughout reentry,
- (2) Loss of capsule pressure when the generator disassembles, and
- (3) Loss of pressure in the generator and capsule before reentry.

The reentry phase of the study was based on a polar orbital decay trajectory. A random tumbling and spinning motion was assumed for both the generator and the exposed heat source. Reference 14 was used to compute ballistic coefficients of 62.3 lb/ft for the generator and 50.5 lb/ft² for the heat source. The reentry velocities and altitudes are given in Figure 5 as a function of time from 400,000 feet altitude.

The HERMAN computer program was used for all internal heat transfer computations. The thermal model shown in Figure 4 was one-dimensional in cylindrical coordinates. When the magnesium case (node M) reached the melt temperature, all converter nodes were removed and the aeroheating was applied to node G1. Conduction through the argon gas was included for the pressurized heat source, but heat transfer was limited to radiation across the gap when pressure was lost.

Aeroheating ratios for free-molecule flow ($\bar{F}_{FM}$) and laminar continuum flow ($\bar{F}_C$) are defined as the ratios of average local heating rates to the reference heating rate. The free molecule reference is a flat plate perpendicular to flow (Figure 6), and the continuum reference is the stagnation point of a one-foot radius sphere (Figure 7). The heating ratios were computed using methods given in Reference 14,

with modifications to account for the flanges on the generator and rounded edges on the capsule. The heating ratios used in this study were

- (1) $\overline{F}_{FM}$ heat source sides = 0.257,
- (2) $\overline{F}_{FM}$ converter sides = 0.255,
- (3) $\overline{F}_C$ heat source sides = 0.599, and
- (4) $\overline{F}_C$ converter sides = 0.377

Ablation studies were not conducted, so the heating rates could not be corrected for surface chemical reactions and heat blockage. The hot wall corrections to the heating rates were made by the HERMAN program.

Initial temperatures of the generator and capsule were computed using the 800°R magnesium case operating temperature as a boundary condition. Figure 8 shows the initial temperatures of the generator and capsule for various emissivity factors, with and without internal pressure. The initial temperatures of all components outboard of the gap between the Haynes and graphite are independent of the conduction and radiation across the gap. If pressure is lost and the emissivity factor is below 0.084, the Haynes will melt before reentry. Therefore, low emissivity factors were to be avoided to prevent long duration contact between the fuel and graphite.

Figures 9 through 11 show the time-temperature histories of the Haynes-25 container during reentry for the maximum and minimum emissivity factors. In all cases, the Haynes-25 melts by a large margin. If the conductivity of the internal gas is increased by the use of helium, the initial and impact temperatures of the capsule would be decreased but the peak reentry temperatures would be increased. The increase in atmospheric pressure as the heat source approaches the earth will increase the internal conductivity of the unpressurized capsules and make impact temperature slightly lower than shown in Figures 10 and 11. Figures 12 and 13 show the maximum reentry temperatures and impact temperatures versus emissivity factors for the three pressure combinations. Impact temperatures decrease with

increasing emissivity factors for all internal pressures. Maximum temperatures increase with increasing emissivity factors when internal pressure is maintained throughout reentry. However, if pressure is lost before or during reentry, the lowest maximum reentry temperatures occur when the emissivity factors are between 0.10 and 0.15. Emissivity factors should be kept above 0.2 to prevent capsule melting before reentry and excessively high interior temperatures during reentry when pressure is lost prior to reentry (Figure 12). Figure 14 shows the emissivity factors versus temperature for the coatings that were proposed for the intact SNAP-19 heat source. If the nickel coating is deoxidized in a vacuum, the emissivity factor is almost low enough to melt the capsule before reentry. It was recommended that higher emissivity coatings be used.

Since the proposed intact SNAP-19 capsule was very similar to the intact design in Reference 15, the ablation depths computed in that study should apply to this design. For a random tumbling motion, it was predicted that 0.068 inch of graphite would ablate on the sides. This would leave only 0.09 inch of graphite over the thread roots at the center joint. There would be even less graphite left if the capsule stabilized perpendicular to flow. It was recommended that either the thread diameter be reduced or an end plug be used, as in Reference 15. Melting of the Haynes-25 could cause material compatibility problems. To aid in specifying compatibility tests, temperature profiles of the materials surrounding the Haynes-25 were predicted. These temperatures are given in Figures 15 through 17. The results of this study are valid even if a high melt temperature container is placed between the Haynes-25 and the graphite for compatibility or containment reasons. The only adjustment that must be made is the use of an equivalent emissivity factor. The equivalent emissivity factor is given by

$$F_{\epsilon} = \frac{1}{\frac{1}{F_{\epsilon 1}} + \frac{1}{F_{\epsilon 2}} + \dots + \frac{1}{F_{\epsilon N}}}$$

where $F_{\epsilon 1}$ to $F_{\epsilon N}$ are emissivity factors between the containers. The developments of this and other radiation shield equations are given in Reference 16. If radiation shields (containers) are added, the emissivities of the surfaces must be increased.

The conclusions drawn from this study were as follows.

1. Provided a tumbling spinning motion can be assured, the heat source should not fail thermally during orbital decay. Fuel temperatures remain well below the melt temperature, thermal stresses are low in the graphite, and ablation is not excessive. The proposed design was similar to the design suggested in Reference 15, which was analyzed in more detail and found to be satisfactory.
2. The Haynes-25 will melt during reentry for all conditions investigated. If the melted Haynes causes a material compatibility problem it could be replaced with a coated refractory material, as suggested in Reference 15.
3. During orbital reentry, most of the incoming heat is rejected by radiation. Because radiation is proportional to the fourth power of temperature, large changes in heating rates result in small temperature changes. Therefore, minor errors in heating ratios would not change the results of this study.
4. Internal emissivity factors should be greater than 0.20 to prevent excessively high temperatures before and during reentry.
5. Relocation of the joint in the graphite container would increase the strength after ablation.
6. If pressure is lost in the capsule, an emissivity factor can be found which minimizes the maximum internal capsule temperatures during reentry.
7. Reentry analyses of a stable heat source are recommended using two- or three-dimensional thermal models and aeroheating rates from wind tunnel tests.

Comparison of Thrust Termination Alofts with Orbital Decay (Reference 8)

A study was conducted to determine the relative severity of orbital decay reentry and a series of ballistic reentries resulting from thrust termination of the Agena during first burn. The configuration selected for this study was the intact SNAP-19 heat source reentering with a crossflow, spinning motion. The thermal model was the same as the model described in the previous section of this report except that the nickel plating was removed from the outer surface of the Haynes-25 and the surface was assumed to be oxidized with an emissivity of 0.87. The continuum flow heating ratio (F_C) was computed to be 0.69, using Reference 14. Table I lists the conditions at 400,000 feet and the Agena thrust termination times after launch for the trajectories used in this study.

TABLE I
Initial Reentry Conditions (References 1 and 2)

Thrust termination (sec)	Reentry velocity (ft/sec)	Reentry angle (deg)
285	14,100	-9.4
364	17,300	-7.2
413	20,100	-5.1
453	23,200	-3.1
472	25,200	-1.5
Orbital decay	26,092	-0.118

Figure 18 shows the maximum reentry and earth impact temperatures of the Haynes-25 as a function of reentry velocity. The time that the Haynes-25 is above the melt temperature is given in Figure 19. Orbital decay is the most severe from the standpoint of time above melt temperature (312 seconds). Figures 18 and 19 also show that the Haynes-25 does not melt if reentry velocities are less than 18,300 ft/sec. Figure 20 shows the maximum temperature gradients across the POCO graphite during the heating and cooling phases of reentry. The maximum gradient of 270°R occurs during a 21,500 ft/sec reentry. This value should not cause excessive thermal stresses.

Since the thrust termination aborts are less severe than orbital decay, the remainder of the SNAP-19/Nimbus B reentry safety studies were confined to orbital decay and guidance failure aborts.

Effects of Aeroheating Multipliers (Reference 9)

This study was conducted to determine the effects of variations in aeroheating rates on the fuel containment of the IRHS. The generator was assumed to have an end-over-end motion, and the heat source was assumed to have a random tumbling spinning motion after generator disassembly. The trajectories, thermal models, and heating rates were the same as those used in the final safety analysis and are defined in the Thermal and Ablation Models and Reentry Analysis sections of this report.

Four orbital decay cases were analyzed. These were:

- (1) Nominal heating and internal pressure throughout reentry.
- (2) Nominal heating with no internal pressure,
- (3) 125 percent heating with no internal pressure, and
- (4) 150 percent heating with no internal pressure.

Figures 21 through 24 give the temperature profiles of the hottest locations in the POCO graphite, Haynes-25, and fuel for the four cases. Figure 25 and Table II summarize the most important temperatures for each case.

TABLE II
Critical Reentry Temperatures

	<u>Trajectory time (sec)</u>	<u>Case 1</u>	<u>Case 2</u>	<u>Case 3</u>	<u>Case 4</u>
Maximum graphite temperature, (°R)	6730	3969	3975	4210	4413
Maximum fuel temperature (°R)	6820	3918	3925	4134	4323
Maximum heating graphite temperature gradient	6670	52	48	55	61
Maximum cooling graphite temperature gradient	6810	99	94	112	128

The Haynes-25 reaches a temperature 980°R above the melt temperature during the least severe reentry (Case 1). The fuel remains over 200°F below the melt temperature even with 150 percent heating. Figure 26 shows that the graphite temperatures with 150 percent heating remain at least 750°R below the temperature where graphite strength begins to decrease with temperature.

Information from the remainder of the development studies appears in the Reentry Analysis section of this report.

Thermal and Ablation Models

Ablation computations were made using the Charring Material Ablation (CMA) digital computer program (Appendix A). The CMA program was also used to compute corrections to the aerodynamic heating caused by ablation. CMA is one-dimensional, but it was indirectly coupled to the three-dimensional HERMAN heat transfer program (Appendix B). Nodes can be removed from the HERMAN thermal model when a given time or a computed node temperature is reached. Generator fin nodes were removed when the node reached the solidus temperature of HM 31-A magnesium. All generator nodes exterior to the IRHS were removed when both generator case nodes reached their solidus temperature. Heat of fusion was included for all confined materials, but materials that were unconfined were assumed to be removed by aerodynamic forces at the solidus temperature. Surface node removal of the graphite ablation material coincided with the removal times predicted by the CMA program.

Material properties were used as functions of temperature in the CMA and HERMAN models. With the exception of the thermochemistry properties used in the ablation energy balances (Reference 18), the material properties used in this analysis are given in Table III.

TABLE III
Material Properties

Tantalum

$\rho = 1038 \text{ lb/ft}^3$	
$C_p = 0.034 \text{ Btu/lb } ^\circ\text{R}$	@ 500°R
0.041	3500
0.053	5885
$K = 0.010 \text{ Btu/ft } ^\circ\text{Rsec}$	500°R
0.014	5885
Melting point = 5885°R	
Emissivity	
Oxidized 0.42	@ 500°R
0.45	3700
As received 0.049	@ 500°R
0.125	3000

Haynes-25 (L-605)

$\rho = 578 \text{ lb/ft}^3$	
$C_p = 0.092 \text{ Btu/lb } ^\circ\text{R}$	@ 500°R
0.120	2000
0.128	3000
$K = 0.0014 \text{ Btu/ft } ^\circ\text{Rsec}$	@ 500°R
0.0043	2100
0.0057	2884
Melting range = 2884 to 3030°R	
Emissivity	
Oxidized = 0.86	@ 500°R
0.87	1700
0.90	2500
Sand blasted - no oxide 0.39	@ 1500°R
0.32	2500
Heat of fusion = 116 Btu/lb	

TABLE III (Cont)

POCO Graphite AXF-5Q

$$\rho = 115 \text{ lb/ft}^3$$

$$C_p = 0.10 \text{ Btu/lb } ^\circ\text{R}$$

0.30

0.42

0.55

0.63

@ 500°R

1000

2000

4500

6000

$$K = 0.0204 \text{ Btu/ft sec } ^\circ\text{R}$$

@ 500°R

0.0126

1200

0.0064

2300

0.0047

3300

0.0040

6000

$$\text{Emissivity} = 0.70$$

@ 500°R

0.75

1000

0.83

2000

0.87

3000

0.92

4000

0.96

6000

ATJ Graphite

$$\rho = 108 \text{ lb/ft}^3$$

$$C_p = 0.100 \text{ Btu/lb } ^\circ\text{R}$$

0.350

0.447

0.497

0.525

@ 500°R

1500

2500

4000

6000

$$K(\text{cross grain}) = 0.0150 \text{ Btu/ft sec } ^\circ\text{R}$$

@ 500°R

0.0084

1500

0.0054

2500

0.0053

4000

0.0058

6000

TABLE III (Cont)

Emissivity = 0.70	@ 500°R
0.80	1500
0.86	2500
0.92	4000
0.96	6000

HM 31-A Magnesium

$$\rho = 113 \text{ lb/ft}^3$$

$C_p = 0.248 \text{ Btu/lb } ^\circ\text{R}$	@ 500°R
0.272	1000
0.360	1580

$$K = 0.0168 \text{ Btu/ft } ^\circ\text{Rsec}$$

$$\text{Melting range} = 1580 \text{ to } 1660^\circ\text{R}$$

$$\text{Heat of fusion} = 142 \text{ Btu/lb}$$

Emissivity

Polished 0.12	@ 500°R
0.16	1100
Oxidized 0.22	900

6061 Aluminum

$$\rho = 169 \text{ lb/ft}^3$$

$$C_p = 0.23 \text{ Btu/lb } ^\circ\text{R} \quad @ 670^\circ\text{R}$$

$$K = 0.0276 \text{ Btu/ft sec } ^\circ\text{R} \quad @ 540^\circ\text{R}$$

$$\text{Melting range} = 1540 \text{ to } 1660^\circ\text{R}$$

$$\text{Heat of fusion} = 170 \text{ Btu/lb}$$

Emissivity

As received 0.09	@ 600°R
Oxidized 0.20	600
0.33	1500

TABLE III (Cont)

Lead Telluride (Mean P&N)

$$\rho = 480 \text{ lb/ft}^3$$

$$C_p = 0.0305 \text{ Btu/lb } ^\circ\text{R}$$

$$0.0344$$

$$0.0560$$

$$K = 0.000686 \text{ Btu/ft sec } ^\circ\text{R}$$

$$0.000183$$

$$0.000264$$

$$@ 620^\circ\text{R}$$

$$960$$

$$2000$$

$$@ 400^\circ\text{R}$$

$$1200$$

$$3000$$

$$\text{Melting temperature} = 2143^\circ\text{R}$$

$$\text{Heat of fusion} = 35 \text{ Btu/lb}$$

Fuel

$$\rho = 396 \text{ lb/ft}^3$$

$$C_p = 0.0605 \text{ Btu/lb } ^\circ\text{R}$$

$$K \text{ (1 atmos helium)} = 0.000172 \text{ Btu/ft sec } ^\circ\text{R}$$

$$\text{(1 atmos argon)} = 0.000083$$

$$\text{(vacuum)} = 0.000069$$

$$\text{Melting temperature} = 4520^\circ\text{R}$$

$$\text{Heat of fusion} = 101 \text{ Btu/lb}$$

$$\text{Heat generation} = 68.1 \text{ Btu/ft}^3 \text{ sec}$$

Tantalum Felt

$$\rho = \text{compressed} = 270 \text{ lb/ft}^3$$

$$C_p = 0.034 \text{ Btu/lb } ^\circ\text{R}$$

$$0.041$$

$$0.053$$

$$@ 500^\circ\text{R}$$

$$3500$$

$$5885$$

$$K \text{ compressed} = 0.0010 \text{ Btu/ft } ^\circ\text{Rsec}$$

$$0.0014$$

$$@ 500^\circ\text{R}$$

$$5885$$

$$\text{Melting temperature} = 5885^\circ\text{R}$$

ZrO₂ (CaO stabilized) coating

Melting temperature = 5172°R

Emissivity = 0.84	@ 500°R
0.39	2255
0.51	4000
0.63	5000

Sodium Silicate - Zirconia coating

Emissivity = 0.85	@ 500°R
0.40	2000
0.40	3000
0.92	3500
0.92	5000

Helium

<u>K(Btu/ft sec °R)</u>	<u>T(°R)</u>
2.31×10^{-5}	500
3.39	900
4.93	1440
6.25	1980
7.93	2700
9.94	3600
1.18×10^{-4}	4500
1.37	5400
1.54	6300

Argon

2.684×10^{-6}	500
4.237	900
5.926	1440
7.291	1980
9.010	2700
1.111×10^{-5}	3600
1.309	4500
1.506	5400
1.664	6300

Emissivity factors used for radiation heat transfer computations across the gaps with oxidized and unoxidized surfaces are given in Figure 27.

Since the generator configuration was always assumed to be tumbling and spinning and the heat transfer through the insulated ends was negligible, both internally generated and aerodynamic heat flows in the generator were assumed to be radial. The generator thermal model is shown in Figure 28. Circumferential heat transfer in the case and the varying cross section for heat transfer are accounted for as shown in Figure 28. The exterior surfaces of the generator are covered with a sodium silicate-zirconia emissivity coating. The surfaces of the finned generator radiate heat between the surfaces as well as radiating to space. The radiation area factors were computed using the CONFACT II program. The area factor model is shown in Figure 29. The model length was 19.38 inches which represented the two attached generators. When a node was removed by melting, the area factor to that node was added to the factor to space. Table IV lists the area factors for the generator.

TABLE IV
SNAP-19 Generator Radiation Area Factors

<u>Areas</u>	<u>Factor</u>
Case to space (1 to S)	0.6596
Case to pin tips (1 to 3)	0.0404
Case to fin roots (1 to 2)	0.3000
Fin roots to space (2 to S)	0.6085
Fin roots to fin tips (2 to 3)	0.0654
Fin tips to space (3 to S)	0.7790

Two heat source models were required because of the different motions assumed during reentry. The following dimensions were used for both models.

Haynes-25 capsule length	5.390 inches
OD	2.120
thickness	0.060
Tantalum liner length	5.440
ID	2.135
thickness	0.010
POCO Graphite Ablator length	6.470
OD	3.000
center side thickness	0.400
center end thickness	0.383
end internal radius	1.170
corner radius	0.250
Tantalum Felt Pads arc length	1.750
thickness	0.100

Uniform gaps were assumed with no changes for thermal expansion. Gap dimensions were as follows.

Generator ATJ to heat source POCO graphite	0.002 inch
POCO to tantalum liner (center 1.70 in.)	0.022
POCO to liner taper for 0.6 in. on each end	0.022 to 0.060
Liner to Haynes-25 capsule	0.007

The thermal resistance across the gap was bracketed by assuming either helium or vacuum in the gaps.

For random tumbling spinning reentry, there are no circumferential temperature gradients but there are axial gradients because of the aeroheating distribution and internal heat source construction. The two dimensional thermal model in the x and r directions used for tumbling reentries is shown in Figure 30. This is the model that was coupled to the generator model to compute steady state temperatures in space and to compute reentry temperatures in the generator.

The two-dimensional thermal model in the R and θ directions used for the cross flow stable reentry orientation is shown in Figure 31. The circumferential divisions were selected based on heating and pressure distributions. The radial divisions in the POCO ablation material were based on the results of the ablation study. The 0.05 inch thick nodes (G1 - G4, G6 - G8, and G10) are the nodes that are predicted to ablate by the CMA program and were removed from the thermal model at the appropriate times.

The CMA ablation models were one-dimensional with surface areas corresponding to all the surface nodes of the two heat source thermal models. These 10 ablation models made it possible to predict ablation distributions over the body, to input ablation corrected aeroheating rates to all surface nodes of the thermal models, and to input the proper times for node removals in the thermal models.

Reentry Analysis

Four trajectory-configuration-orientation combinations were selected for the reentry safety analysis to cover the most severe cases. If the heat source survived these reentries, it would survive the less severe reentries and no further reentry analyses would be necessary. Reference 8 showed that thrust termination aborts are less severe than orbital decay, and further analysis of this group of reentries was not needed. The highest heating rates and largest temperature gradients were expected during a steep guidance failure abort. The steepest credible guidance failure abort for the Nimbus B mission would result in a reentry with an inertial velocity of 26,000 ft/sec and an inertial reentry angle of -2.06 degrees at 400,000 feet. The initial flight path angle measured counterclockwise from East (Tau) is -108 degrees. This trajectory was used with the IRHS initially in the generator on the Agena and Nimbus B to produce a thermal shock at the time of generator disassembly. The other three trajectories were decays from polar orbits. One was initially an end over end tumbling generator on the Nimbus B followed by a random tumbling spinning heat source after generator disassembly. Another was a random tumbling spinning heat source for the entire reentry. Comparison of the last two configurations determines the effect of the generator during reentry. The third orbital decay configuration was a cross flow stable IRHS throughout reentry. This orientation produces the most local ablation and highest local temperatures.

Trajectories

Trajectories were run with the TTA computer program using an oblate rotating earth model. The continuum flow stagnation point reference heating rate equations used in TTA are given in Appendix C. Ballistic coefficients were computed using the force data given in References 19 through 21 and the techniques given in Reference 14. The resulting ballistic coefficients for the various configurations, motions, and flow regimes are tabulated in Table V. The limits of transition flow were taken from Appendix B of Reference 2.

TABLE V
Ballistic Coefficients ($W/C_D A$), lb/ft^2

<u>Configuration</u>	<u>Free molecule flow</u>	<u>Continuum flow</u>
Side-on IRHS	21.1	32.4
Random tumbling IRHS	21.9	45.4
End-over-end Nimbus and Agena	--	40.0
End-over-end Nimbus	--	45.0

IRHS Transition Altitudes 380,000 to 260,000 feet

Figures 32 through 35 give the altitude, velocity, dynamic pressure, and reference heating rates versus time from 400,000 feet altitude for the four trajectories. The free molecule reference is the heating rate to a flat plate perpendicular to flow, and the continuum reference is the stagnation point heating of a one-foot radius sphere.

Aerodynamic Heating

The heating ratios used in the HERMAN internal heat transfer digital computer program and the CMA ablation program are defined as the ratio of local heating rates to the reference heating rate. The continuum flow ratio (F_c) is ratioed to the stagnation point heating of a one-foot radius sphere. The free molecule ratio (F_{FM}) is ratioed to the heating of a flat plate perpendicular to flow. Average local heating ratios ($\bar{F}$) were computed for the various motions using methods described in Reference 14.

Continuum flow heating ratios for the generator were based on data from wind tunnel tests conducted by Martin-Marietta (References 22 and 23). Average heating ratio distributions over an unattached generator were computed for 1/3, 2/3, and full length fins using data from Reference 23. A limited amount of data from Reference 22 indicated that the Nimbus had no effect on the heating rates to the generator case (Nodes H1 and H2) but reduced the heating to the fins by 29 percent. The heating distributions given in Figure 36 are corrected for heat blockage by the Nimbus. Details of the data reduction leading to the generator heating rates are given in Reference 5. Heating rates for the generator with no fins were computed using data from References 13, 24, and 25. Due to the complex shape of the generator and the limited number of heat meters on the generator models, the average local heating ratios of a tumbling spinning generator can only be considered as approximations. The accuracy also suffers because there were heat meters at only two generator locations during the tests which defined the blockage effects of the Nimbus. Fortunately, the generator disassembly altitude has little effect on the survivability of the IRHS for any of the critical reentry trajectories, as is shown in the Reentry Temperatures section of this report.

Free molecule heating rates to the IRHS were computed using the equations in Reference 14. Continuum flow heating ratios for the IRHS were based on data from hypersonic wind tunnel tests conducted by Sandia in the Rhodes and Bloxsom 60 inch hot-shot tunnel and the Cornell Aeronautical Laboratory 96 inch shock tunnel (References 13, 24, and 25). The average heating ratios for each surface node of the generator and IRHS, for each configuration and motion, are given in Table VI. The node names refer to locations on the thermal model.

Initial Temperatures

Initial temperatures for all trajectories were computed using two-dimensional HERMAN models (Figures 28 and 30) with the generator radiating to space. An open electrical circuit and the internal generation rate at the time of fuel loading (570 watts) were assumed for conservatism. Computations were made for three combinations of pressure and internal surface conditions which range from the maximum to the minimum possible internal thermal resistance. The results given in Table VII show that the capsule and fuel will not melt in space regardless of internal resistance.

TABLE VI

Aerodynamic Heating Ratios

Location, Node, and Configuration	Motion	$\overline{F}_{FM}$	$\overline{F}_c$
Generator fin ends (H4), full fins	End-over-end tumbling	--	0.46
Generator fin bases (H3), full fins	End-over-end tumbling	--	0.37
Generator fin bases (H3), 2/3 fins	End-over-end tumbling	--	0.41
Generator case (H1 & H2), full fins	End-over-end tumbling	--	0.18
Generator case (H1 & H2), 2/3 fins	End-over-end tumbling	--	0.16
Generator case (H1 & H2), no fins	End-over-end tumbling	--	0.25
IRHS, end center (GE1)	Random tumbling spinning	0.255	0.524
IRHS, end edge (GC2)	Random tumbling spinning	0.255	0.629
IRHS, side end, (GC1)	Random tumbling spinning	0.259	0.592
IRHS, side center (GS1 & GS3)	Random tumbling spinning	0.259	0.503
IRHS, side, 0° to 20° (G1, G2, G3, G4, G5)	Side-on stable	0.985	2.06
IRHS, side, 20° to 40° (G6, G7, G8)	Side-on stable	0.873	1.75
IRHS, side, 40° to 60° (G10, G11)	Side-on stable	0.655	1.09
IRHS, side, 60° to 90° (G13)	Side-on stable	0.274	0.445
IRHS, side, 90° to 180° (G15)	Side-on stable	0.00	0.085

TABLE VII

Open Circuit Intact SNAP-19 Temperatures in Space Temperature

Node	Helium internal gas and Oxidized metal surfaces (°R)	Internal vacuum oxidized metal surfaces (°R)	Internal vacuum deoxidized metal surfaces (°R)
H4	822	822	822
H3	828	828	828
H2	835	835	835
H1	835	835	835
TE	1371	1419	1406
HA	1374	1422	1409
GS1	1378	1671	1660
GS2	1380	1673	1662
GS3	1383	1712	1705
GS4	1392	1723	1717
GC1	1379	1705	1699
GC2	1382	1708	1703
GC3	1382	1709	1703
GE1	1386	1715	1709
GE2	1388	1717	1711
T1	1410	1744	1739
T2	1394	1724	1718
L1	1428	1766	1762
L2	1399	1730	1725
C1	1455	2135	2343
C2	1417	2043	2244
F1	1460	2138	2346
F2	1796	2396	2562
F3	2003	2545	2693
F4	2106	2626	2765
F5	2128	2644	2782
F6	1419	2044	2245
F7	1714	2296	2458

The temperature of node C1 is higher than the temperature of C2, which indicates that there is less resistance to heat flow through the tantalum felt pads than across the gap, even with helium in the gap. Radial temperature distributions in fuel can be computed in closed form using equations developed in Appendix D.

Ablation

Ablation depths for all surface nodes for the four trajectories were computed with the CMA ablation program (References 26 and 27). The maximum and minimum ablation for each trajectory are plotted versus trajectory time in Figures 37 through 40. The maximum ablation location is node GC2 for a tumbling IRHS and nodes G1 to G5 for a stable IRHS. The location of the minimum ablation for a tumbling IRHS is node GS1, and for a stable orientation the minimum location is node G15. The ablation distribution for a tumbling IRHS is shown in Figure 41. The ablation is nearly uniform with a slight increase in the corner region. The circumferential ablation distribution for the cross flow stable reentry is shown in Figure 42. The ablation losses are large only in the stagnation region with a large reduction in ablation between 30 and 60 degrees from the stagnation line.

The center joint location of the heat shield is the most vulnerable to an ablation failure for all reentry motions or orientations. Figure 43 shows the increased erosion around the center joint during a cross flow stable plasma jet test. This is the only marginal area for ablation on the heat shield. If the IRHS reenters in a cross flow stable orientation, a small area of the heat shield will have ablated down to the thread roots at the time of earth impact even without increased erosion along the joint. However, this large amount of material removal is restricted to a 60 degree arc in the stagnation region. The maximum ablation on the aft side is only 0.007 inch. For all tumbling and/or spinning motions, the maximum ablation depth would be less than half way to the thread roots and would not weaken the heat shield enough to cause disassembly before earth impact unless rotational or tumbling rates became excessive. In all cases, over half of the ablation takes place after peak heating when surface temperatures and air density are both high. Approximately 25 percent of the surface recession takes place after the aerodynamic heat pulse is over.

Figure 44 shows the parameter $\frac{\dot{m}}{C_m}$ (which governs the ablation rate) as a function of temperature and pressure for the three ablation regimes. It can be seen that a catastrophic ablation failure would not occur unless the temperature is high enough to cause sublimation controlled ablation. The heat shield remains at least 950°R below the sublimation controlled regime for all of these most severe reentry conditions.

Reentry Temperatures

The internal heat generation, the long reentry times, the low heat capacity of the surface structures, and the relatively high internal thermal resistances caused the generator and IRHS surface temperatures to remain close to the radiation equilibrium temperatures for all four trajectories. This can be demonstrated by comparing the computed temperatures with the equilibrium temperatures given in Reference 28. The computed generator temperature was only 14°R below the equilibrium temperature at the time of disassembly during a decay from orbit. The peak graphite temperatures were 45°R below equilibrium during orbital decay and 65°R below equilibrium during a guidance failure abort. Figures 45 and 46 further illustrate that most of the aeroheating is radiated to space and little is conducted into the vehicle. During the abort trajectory, 72 percent of the aeroheating is radiated to space before the time of peak surface temperature. After peak temperature is reached, more heat is radiated to space than is added by aeroheating. During an orbital decay, 80 percent of the aeroheating is reradiated prior to peak temperature. Figure 47 shows the effects of all the corrections on the cold wall heating rate leading to the net heat conducted inward from the surface. The summation of the hot wall correction and ablation effects are small for most of an orbital decay trajectory. These corrections do not change predicted temperatures significantly, but they do increase the predicted ablation because of the heat added at low altitudes where there is a larger supply of oxygen. The largest correction comes from the radiation to space. This is the correction that affects IRHS temperatures and is an indication of how close the surface is to the instantaneous radiation equilibrium temperature. When the temperature of a vehicle is near the radiation equilibrium temperature throughout reentry, any thermal protection at high altitude has little effect on maximum reentry temperatures. Therefore, the generator was not expected to protect the IRHS. This was substantiated by the comparison of the

orbital decay tumbling trajectories of the bare IRHS and the IRHS in the generator. The maximum peak temperature difference was 8°R , and the maximum ablation difference was 0.0022 inch. Because of the similarity of these two trajectories, the temperature plots of the bare capsule are not presented.

Figures 48 and 49 give the temperatures and disassembly sequence of the generator case for the abort and orbital decay trajectories. The temperature gradients indicate that heat was transferred from the fins to the case during reentry. The reduction in the slope of the case temperature curves after the fins are removed show that the fins increase the generator disassembly altitude.

The temperatures of the hottest heat shield and capsule nodes are given in Figures 50 through 54. Temperature profiles are given for heat sources with internal pressure and oxidized surface and with internal vacuum and deoxidized surface for the abort trajectory (Figures 50 and 51) and the tumbling orbital decay trajectory (Figures 52 and 53). These conditions bracket the internal thermal resistance and give the range of possible reentry temperatures. The Haynes-25 was assumed to remain in place in this analysis even after it melted. Actually, the molten Haynes will mix with the fuel and have the same temperature distribution as the fuel. In all cases, parts of the Haynes capsule reach a temperature at least 640°R above the melt temperature. Although the temperature plots indicate that the Haynes will resolidify before earth impact, any Haynes at the center of an evacuated, deoxidized IRHS will be molten at impact.

Figure 54 shows the circumferential temperature gradients in the graphite and Haynes-25 for a cross flow stable orbital reentry. The maximum heating and cooling radial temperature gradients across the graphite at the center joint are given in Table VIII.

TABLE VIII

Maximum Radial Temperature Gradients
Across the Heat Shield During Reentry

<u>Trajectory</u>	<u>Heating (°R)</u>	<u>Cooling (°R)</u>
Guidance failure abort, pressurized	117	100
Guidance failure abort, vacuum and deoxidized	79	78
Orbital decay, tumbling, pressurized	49	90
Orbital decay, tumbling, vacuum and deoxidized	34	63
Orbital decay, side-on stable, pressurized	241	103

Temperature gradients are higher in the pressurized IRHS because lower internal resistance permits more heat to be transferred to or from the inner surface of the graphite. Internal pressure and surface conditions have almost no effect on the surface temperature of the graphite, but the peak outer surface temperatures of the Haynes-25 would range from 105 to 82°R hotter in a pressurized IRHS.

The reentry temperatures of the fuel are needed to predict melting and sintering of the fuel and the helium release rate of the fuel. Figures 55 through 58 give the fuel temperature ranges for tumbling heat sources with and without internal pressure and oxidized surfaces. The heat source ends have the least thermal resistance between the fuel and the IRHS surface and respond to exterior environmental changes most rapidly. Peak fuel temperatures are less sensitive than peak capsule temperatures to internal thermal resistance. Figure 59 gives the fuel temperature range for a stable cross flow orbital reentry of the IRHS. This figure illustrates how the heat flow changes from radially outward prior to reentry, to flow from the stagnation line and out the aft side during reentry. The fuel does not melt during reentry regardless of trajectory or configuration. The temperature gradients in the fuel at any given time should help to spread the helium release over a longer period of time and hence reduce peak internal pressure.

This study predicted no thermal failures of the IRHS for any of the trajectories, provided there are no structural changes of the IRHS brought about by mechanical or chemical degradation. However, if the heat shield could fail by

other means it would be necessary to know the fuel and capsule temperatures and the altitude and velocity of failure to predict fuel release and melt. Figure 60 shows the location of three significant events for each trajectory and orientation with respect to the fuel melt curve taken from Reference 29. All trajectories pass under the fuel melt curve and all capsules are liquid at the time of peak heating. All trajectories pass through the melt zone before reaching peak deceleration. Table IX lists the parameters which could influence a mechanical failure at the times of capsule melting, peak heating and peak deceleration for each trajectory. Temperatures of the tantalum liner are not included because the liner has no positive closure. All graphite temperatures are below the temperature where graphite strength decreases with temperature, as shown in Figure 26.

Launch Pad Abort Fires

The temperature distributions of the IRHS in a launch pad abort fire were computed for 3 initial configurations of the intact Snap-19 generator (Reference 30). These were:

1. The entire generator attached to the Nimbus. All external parts of the generator were assumed to fall free of the IRHS when the generator melted.
2. A bare IRHS blown free of the generator and sitting on end and exposed to the fire.
3. A bare Haynes-25 capsule separated from the generator and heat shield and exposed to the fire.

Oxidation and sublimation of the graphite were not included in the launch pad abort fire computations. The reactions in this environment are not well defined, but the surface recession of graphite is not expected to be excessive.

TABLE IX

Parameters That Influence Fuel Release in the Melt Zone

Guidance Failure Abort on the Agena, Random Tumbling Spinning Pressurized IRHS			
Parameter	Event		
	Haynes melt	Peak heating	Peak deceleration
Trajectory time (sec)	245	305	385
Altitude (1000 ft)	236	205	132
Velocity (ft/sec)	25,800	22,700	9,700
Deceleration (g)	0.975	2.53	7.90
Fuel temperature range (°R)	2285 - 2877	2775 - 3361	3429 - 3545
Ablation at joint (in.)	0.0056	0.012	0.026
Graphite temperature, GS1 (°R)	3202	3515	3203
Orbital Decay on the Nimbus B, Random Tumbling Spinning Pressurized IRHS			
Parameter	Event		
	Haynes melt	Peak heating	Peak deceleration
Trajectory time (sec)	10,410	10,500	10,555
Altitude (1000 ft)	247	193	135
Velocity (ft/sec)	24,800	20,700	10,600
Deceleration (g)	0.545	3.22	8.21
Fuel temperature range (°R)	2823 - 2883	3242 - 3447	3326 - 3492
Ablation at joint (in.)	0.0095	0.018	0.032
Graphite temperature, GS1 (°R)	3018	3496	3285
Orbital Decay of a Side-on-Stable Pressurized IRHS			
Parameter	Event		
	Haynes melt	Peak heating	Peak deceleration
Trajectory time (sec)	10,240	10,482	10,544
Altitude (1000 ft)	295	204	139
Velocity (ft/sec)	25,900	20,900	9,600
Deceleration (g)	0.078	3.11	7.94
Fuel temperature range (°R)	2110 - 2882	2731 - 4336	2785 - 4029
Ablation at joint (in.)	0.0005-0.016	0.00026-0.068	0.0042 - 0.110
Graphite temperature, joint (°R)	2029 - 3098	2584 - 4601	2594 - 4186

The fire temperatures and durations were computed using methods given in Reference 31. The thermal environment was based on combustion of 160,000 pounds of liquid propellant from the Thorad and Agena vehicles. Temperatures given in Reference 31 were computed from experimental data assuming radiation from a black body as the only method of heat transfer from the fire to the sensors. The effective black body radiating temperatures given in Table X were used in this study.

TABLE X
Effective Black Body Radiating Temperatures
for Launch Pad Fire

<u>Time (sec)</u>	<u>Temperature (°R)</u>	<u>Environment</u>
0	5410	Fireball
2.3	4761	Fireball
3.7	4302	Stem
4.0	3042	Stem
4.7	2297	Residual fire
1800	2297	Residual fire
1900	560	Ambient
2500	560	Ambient

It was assumed that the generator and heat shield was filled with argon and that helium was in the capsule. An emissivity of 0.95 was assumed for the bare capsule to simulate a coating of soot from the fire. These assumptions produce the maximum heat transfer to and through the heat sources.

Temperature distributions in the generator and IRHS were computed using the HERMAN computer code and the thermal models shown in Figures 28 and 30. The temperature profile of the hottest location on the outside of the graphite heat shield is given in Figure 61. The temperature rises rapidly in the fireball and then drops before reaching the steady state temperature in the residual fire. The heat capacity of the inner part of the IRHS accounts for the dips in the curves. Heat is being transferred inward to the left of the minimum points and outward to the right of the minimum points. Comparison of the two curves shows that, although the generator remains intact for only 2 seconds, it protects the IRHS long enough to eliminate the fireball temperature peak. Figure 62 shows the temperatures of the hottest point on the inside of the heat shield for both initial configurations. The thermal lag of the heat shield is sufficient to eliminate the temperature peak of the short duration fireball and stem. Figure 63 gives the maximum Haynes-25 capsule temperatures for the three initial configurations. One node of the bare capsule configuration exceeded the solidus temperature of Haynes-25 (2884°R) for 1.5 seconds. However, the external environments are not well enough defined to conclude that a bare capsule will definitely melt in a launch pad fire. This analysis did not include a configuration change with capsule melting. The capsule will not exceed 2511°R if it is protected by the heat shield. Maximum fuel temperatures versus time are given in Figure 64. Peak fuel temperatures occur during the residual fire for all initial configurations and are well below the melt temperature.

The maximum temperature gradients in the heat shield occur between nodes GS3 and GS4 which are 0.35 inch apart. The gradients for both initial configurations are given in Figure 65.

The generator disassembles two seconds after fuel ignition and may not offer any impact protection when the generator falls to the earth or from falling debris. If the capsule is blown free of the heat shield prior to fuel ignition, there is a possibility of the capsule rupturing in the fire. The intact Snap-19 will not fail thermally during a launch pad abort fire in any of the other initial configurations investigated.

Conclusions

This study analyzed the four severest reentry trajectories, orientations, and configurations from the standpoint of maximum temperatures, temperature gradients, and ablation depths. The IRHS has a large thermal margin of safety if reentry occurs with a tumbling and/or spinning motion. It is highly improbable that the IRHS will reenter stable side-on with no rotation throughout the trajectory. However, for this most severe condition, this study shows that the IRHS will not fail thermally although the margin of safety will be small. The only types of disassembly considered in this study were ablation of the IRHS and melting of the generator. The effects of chemistry, internal pressure, and thermal and aerodynamic stresses were not included. If other studies indicate configuration changes caused by mechanical or chemical failures, the results of this study may be invalid. The Haynes-25 capsule will melt during all orbital decay reentries and during guidance failure aborts, but the heat shield is designed to contain the fuel. The Haynes-25 will resolidify before earth impact for all trajectories if the IRHS is initially pressurized. The Haynes-25 at the center of an evacuated, deoxidized IRHS will be molten at impact.

Table XI summarizes the thermal parameters pertinent to the survival of the SNAP-19 IRHS. Temperatures in the table were based on the assumption that after the capsule melts the fuel is in contact with the liner and the molten Haynes-25 is mixed evenly throughout the fuel.

The following conclusions about the IRHS during reentries and launch pad abort fires can be drawn from this study. Since the heat sources remain near the radiation equilibrium temperature for the most severe reentries, small errors in the trajectories or aeroheating rates will not have a significant effect on temperature or ablation predictions. In addition, there are large temperature or ablation margins on which to base the conclusions.

TABLE XI

Maximum Temperatures, Temperature Gradients, and Ablation Depths During Reentry

Parameter	Abort, randomly tumbling IRHS, initially in Nimbus B or Agena	Orbital decay, randomly tumbling IRHS initially in Nimbus B	Orbital decay, randomly tumbling bare IRHS	Orbital decay, side-on stable bare IRHS
<u>Graphite Heat Shield</u>				
Peak temperature ($^{\circ}\text{R}$)	3745	3673	3668	4601
Impact temperature ($^{\circ}\text{R}$)	2058	2068	2062	1988
Heating radial temperature gradient ($^{\circ}\text{R}$)	117	49	43	241
Cooling radial temperature gradient ($^{\circ}\text{R}$)	100	90	92	103
Ablation at end of heat pulse (in.)	0.0448	0.0450	0.0490	0.1280
Ablation at earth impact (in.)	0.0562	0.0603	0.0625	0.1710
<u>Tantalum Liner</u>				
Peak temperature ($^{\circ}\text{R}$)	3562	3516	3510	4394
Impact temperature ($^{\circ}\text{R}$)	2117	2129	2122	2007
ΔT below melt at T_{Max} ($^{\circ}\text{R}$)	2323	2369	2375	1491
<u>Haynes-25 Capsule</u>				
Peak temperature ($^{\circ}\text{R}$)	3562	3516	3524	4394
Impact temperature ($^{\circ}\text{R}$)	2854	2875	2859	2748
ΔT above melt at T_{Max} ($^{\circ}\text{R}$)	682	632	640	1510
<u>Fuel</u>				
Peak temperature ($^{\circ}\text{R}$)	3562	3516	3524	4394
Impact temperature ($^{\circ}\text{R}$)	2854	2875	2859	2748
ΔT below melt at T_{Max} ($^{\circ}\text{R}$)	957	1007	999	129

- (1) The graphite heat shield will not ablate down to the center joint before the end of the heat pulse for any trajectory or orientation. If the heat source reenters in a cross flow stable orientation, a 60 degree arc will ablate to the thread roots before earth impact. The graphite surface temperatures remain below the sublimation temperature for all reentries. Thermal gradients in the graphite will be small.
- (2) The tantalum liner will not melt.
- (3) The Haynes-25 capsule temperatures will exceed the solidus temperature by over 600°R for all reentries. Most or all of the Haynes will resolidify before earth impact, but it will not contain the fuel. The capsule will not melt in a launch pad abort fire unless the capsule is blown free of the generator, heat shield, and liner before fuel ignition.
- (4) The fuel will remain below its melt temperature.
- (5) The generator offers practically no thermal protection for the IRHS during orbital decay or guidance failure aborts. For some early Agena thrust termination aborts the generator would keep the capsule from melting. The generator has no significant thermal effects in a launch pad abort fire.

Figure 1. SNAP-19 intact reentry heat source

Figure 2. Nimbus B spacecraft with the SNAP-19 generators in the exit shroud

Figure 3. SNAP-19 generator subassembly

NOTE:

INNER SURFACE OF POCO GRAPHITE IS COATED WITH ZIRCONIUM DIOXIDE AND OUTER SURFACE OF THE HAYNES-25 IS COATED WITH NICKEL.

Figure 4. Intact SNAP-19 (May 5, 1967)

Figure 5. Altitude and velocity for intact SNAP-19 orbital decay

Figure 6. Reference free molecule heating rates ($\text{Btu/ft}^2 \text{sec}$)

Figure 7. Reference continuum heating rates for intact SNAP-19 orbital decay

Figure 8. Initial temperatures of the May 5, 1967, intact SNAP-19 generator as a function of the emissivity factor (F_ϵ) between the graphite and Haynes-25

Figure 9. Haynes-25 temperature
1 atmosphere of argon in capsule
throughout reentry

Figure 10. Haynes-25 temperature
1 atmosphere of argon in capsule until generator disassembly
with no pressure in capsule thereafter

Figure 11. Haynes-25 temperature
Capsule pressure loss before reentry

Figure 12. Maximum Haynes-25 temperatures during reentry

Figure 13. Haynes-25 temperatures at earth impact

Figure 14. Emissivity factors (F_e) between nickel and zirconium dioxide

Figure 15. Temperatures of materials bounding Haynes-25, 1 atmosphere pressure in capsule throughout reentry

(a)

(b)

Figure 16. Temperatures of materials bounding Haynes-25, pressure loss at generator disassembly

(a)

(b)

Figure 17. Temperatures of materials bounding Haynes-25, capsule pressure loss before reentry

Figure 18. Maximum and impact temperatures of Haynes-25 in the intact SNAP-19 heat source for orbital decay and preorbital aborts

Figure 19. Time Haynes-25 is above the melt temperature in the intact SNAP-19 heat source for orbital decay and preorbital aborts

Figure 20. Maximum temperature gradients across the POCO graphite in the intact SNAP-19 heat source for orbital decay and preorbital aborts

Figure 21. Orbital decay temperatures of a tumbling SNAP-19 IRHS with internal pressure throughout reentry

Figure 22. Orbital decay temperatures of a tumbling SNAP-19 IRHS with pressure loss before reentry

Figure 23. Orbital decay temperatures of a tumbling SNAP-19 IRHS with pressure loss before reentry and 125 percent theoretical aeroheating

Figure 24. Orbital decay temperatures of a tumbling SNAP-19 IRHS with pressure loss before reentry and 150 percent theoretical aeroheating

Figure 25. Maximum orbital decay temperatures of a tumbling SNAP-19 IRHS demonstrating the margin of safety

Figure 26. High temperature properties of graphite

Figure 27. Emissivity factors across gaps

(a)

(b)

Figure 28. SNAP-19 generator thermal model

S_1 = RADIAL OPENING

S_2 & S_3 = OPENINGS ON EACH END
OF THE GENERATOR

THE BOUNDARY BETWEEN 2 AND 3
CORRESPONDS TO THE BOUNDARY
BETWEEN NODES H3 AND H4 IN
THE HERMAM MODEL

Figure 29. Radiation area factor model

Figure 30. X-R Heat source thermal model

Figure 31. R-θ Heat source thermal model

(a)

(b)

Figure 32. Guidance failure abort trajectory of an end over end tumbling Agena and Nimbus and a random tumbling IRHS $\gamma_0 = 2.026$ degrees

(a)

(b)

Figure 33. Orbital decay trajectory of an end over end tumbling Nimbus and a random tumbling IRHS

(a)

(b)

Figure 34. Orbital decay trajectory of a random tumbling IRHS TTA-17647

(a)

(b)

Figure 35. Orbital decay trajectory on side on stable IRHS
TTA-17646

Figure 36. SNAP-19 Generator heating ratios on Nimbus

Figure 37. Guidance failure abort ablation end over end tumbling generator and random tumbling spinning IRHS TTA-17644 and 18200

Figure 38. Orbital decay ablation end over end tumbling generator and random tumbling spinning IRHS TTA-17645 and 18201

Figure 39. Orbital decay ablation for a random tumbling spinning IRHS - TTA-17647

Figure 40. Orbital decay ablation of a side-on stable IRHS TTA-17646

Figure 41. Random tumble-spin capsule alone
ablation depth at earth impact
TTA-17647

Figure 42. Orbital decay side-on, stable capsule
alone ablation depth at earth impact
TTA-17646

Figure 43. Ablation of a cross flow stable IRHS after a plasma jet test

Figure 44. Ablation rate parameter of graphite as a function of temperature and pressure

Figure 45. Surface energy balance (from the time of IRHS release to time of maximum temperature 72 percent of heating is reradiated)

Figure 46. Surface energy balance (from the time of the IRHS release until the time of maximum surface temperature 80 percent of the net aeroheating is radiated from the surface to space)

Figure 47. Surface heat transfer at Node GC2 during orbital decay TTA-18201

Figure 48. Generator case temperatures for a guidance failure abort end over end tumbling on the Agena

Figure 49. Generator case temperatures for orbital decay, end over end tumbling on the Nimbus

Figure 50. IRHS heat shield and capsule temperatures, guidance failure abort on the Agena, random tumbling spinning, pressurized

Figure 51. IRHS heat shield and capsule temperatures, guidance failure abort on the Agena, random tumbling spinning, internal vacuum and deoxidized metal surfaces

Figure 52. IRHS heat shield and capsule temperatures, orbital decay on Nimbus, random tumbling spinning, pressurized

Figure 53. IRHS heat shield and capsule temperatures, orbital decay on Nimbus, random tumbling spinning, internal vacuum and deoxidized metal surfaces

Figure 54. IRHS heat shield and capsule temperatures, orbital decay, side on stable, pressurized

Figure 55. SNAP-19 IRHS fuel temperature range for guidance failure abort, 1 atmosphere of helium internal pressure, oxidized metal surfaces, random tumbling spinning motion. Initially on the Agena and Nimbus B

Figure 56. SNAP-19 IRHS fuel temperature range for guidance failure abort, internal vacuum, deoxidized metal surfaces, random tumbling spinning motion. Initially on the Agena and Nimbus B

Figure 57. SNAP-19 IRHS fuel temperature range for orbital reentry, 1 atmosphere of helium internal pressure, oxidized metal surfaces, random tumbling spinning motion. Initially on the Nimbus B

Figure 58. SNAP-19 IRHS fuel temperature range for orbital reentry, internal vacuum, deoxidized metal surfaces, random tumbling spinning motion. Initially on the Nimbus B

Figure 59. SNAP-19 IRHS fuel temperature range for orbital reentry, 1 atmosphere of helium internal pressure, oxidized metal surfaces, side-on stable motion

Figure 60. Events on intact SNAP-19 reentry trajectories

Figure 61. Hottest outer heat shield temperature, node GC2
in a launch pad abort fire

Figure 62. Hottest inner heat shield temperature, node GS4
in a launch pad abort fire

Figure 63. Hottest capsule temperature, node C1 in a launch pad abort fire

Figure 64. Hottest fuel temperature, node F1 in a launch pad abort fire

Figure 65. Maximum graphite temperature gradients, between nodes GS3 and GS4 (0.35 inch) in a launch pad abort fire

APPENDIX A

Ablation Program

Graphite ablation was computed using the Sandia Charring Material Ablation code described in Reference 32. Each iterative step of the program consists of three main events based upon the laws of conservation of mass and energy: internal decomposition, if applicable; internal energy balance; and surface energy balance. Completion of each iteration gives new values of nodal density, pyrolysis gas flow, nodal and surface temperatures, and ablation rate.

Equations governing internal decomposition include density changes, heat of decomposition, local generation of pyrolysis gas, and the mass flow rate of pyrolysis gas past a given point. The internal energy balance includes sensible energy stored in the node, net conduction into the node, chemical energy accumulation, net energy convected into the node by pyrolysis gas entering the node, and net energy convected from the node by pyrolysis gas generated in the node. The surface energy balance includes convection, diffusion, gas phase removal, radiation in and out, condensed phase removal, conduction, energy in pyrolysis gas, and energy in surface char. The convection and diffusion terms include aeroheating and heat of combustion. The convection term is modified by gas injection into the boundary layer and by the hot wall correction factor.

The program outputs the temperatures of the ablation material and equivalent heat sink nodes, heat conducted from the surface, radiation in and out, convection, diffusion, surface recession, char formation, and depth of pyrolysis as a function of time.

The ablation program is one-dimensional and cannot handle internal heat generation or internal radiation. These limitations make it necessary to use the surface recession rate and net heating outputs of the ablation program as inputs to the more versatile HERMAN heat transfer program (Appendix B) to get the temperature distribution throughout a reentering vehicle.

APPENDIX B

HERMAN-64 Heat Transfer Program

This program for the CDC-3600 digital computer is used for transient thermal analysis of composite structures with radiation from the surface and heat input from either convection or internal generation. Ablation is handled by removing nodes from the network either when they reach a given temperature or at a given time. The thermal-diffusion equation is solved by dividing the conducting material into a finite number of segments or nodes in such a manner that the behavior of the thermal parameters can be described by average values. The thermal behavior of the node can then be described by the following system of ordinary differential equations:

$$C_{po} \rho_o V_o \frac{dT_o}{dt} = \sum \frac{K_i S_i}{\Delta X_i} (T_i - T_o) + \sum f_i \sigma \epsilon_i A_i (T_i^4 - T_o^4) + Q_o \quad (B-1)$$

where

- C_{po} = specific heat of node o
- ρ_o = density of node o
- V_o = volume of node o
- T_o = temperature of node o
- T_i = temperature of node i
- K_i = thermal conductivity between nodes o and i
- S_i = area between nodes o and i
- X_i = a distance such that $(T_i - T_o)/X_i$ is an approximation to T/X in the direction normal to the surface between nodes o and i.
- A_i = area of a surface which is receiving radiation of effective temperature T_i
- ϵ_i = emissivity of A_i

- f_i = form factor of A_i
 σ = Stefan-Boltzmann constant
 Q_o = heat input to node o.

The term on the left represents the heat stored on node o. On the right, the first term represents the heat received by conduction, the second term represents the heat received by radiation, and the third term represents the heat received by internal generation or convection.

APPENDIX C

Trajectory Program

Trajectories were computed using the Sandia TTA digital computer program described in Reference 33. The subprogram used to calculate the convective stagnation point aerodynamic heating employs the equations presented below.

$$\dot{q}_c = \frac{17,600}{\sqrt{R}} \left(\frac{\rho_\infty}{\rho_o} \right)^{0.5} \left(\frac{V_\infty}{V_c} \right)^{3.15} \quad (C-1)$$

where

$\dot{q}_c$ = aerodynamic convective heating rate to the stagnation point of a sphere of radius R ft (Btu/ft² sec)

R = radius of the sphere, ft

$\frac{\rho_\infty}{\rho_o}$ = the ratio of local free-stream air density to the free-stream air density at sea level

V_∞ = vehicle velocity (ft/sec)

V_c = $\sqrt{gr}$ = the circular orbital velocity (ft/sec)

The equation used to calculate the integrated convective stagnation point aerodynamic heating ($\dot{Q}_c$) is:

$$\dot{Q}_c = \int \dot{q}_c dt + \dot{Q}_{co} \quad (C-2)$$

where

$\dot{Q}_{co}$ = the integrated convective stagnation point aerodynamic heating at time zero

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APPENDIX D

Temperature Distributions in Annular Spheres and Cylinders With Uniform Heat Generation

Steady state temperatures of cylindrical and spherical fuel configurations can be computed analytically provided heat is transferred only in the radial direction. The equation for a cylindrical capsule does not hold if there are large heat losses from the end of the cylinder.

Cylinders

The basic steady state conduction equation in the r-direction with internal generation in cylindrical coordinates is:

$$\frac{1}{r} \frac{d}{dr} \left(r \frac{dT}{dr} \right) + \frac{q}{K} = 0 \quad (D-1)$$

which can be written as

$$r \frac{d}{dr} \left(r \frac{dT}{dr} \right) + \frac{q}{K} r^2 = 0 \quad (D-2)$$

Using the boundary condition

$$\frac{dT}{dr} = 0 \text{ at } r = r_i$$

and integrating Eq. (D-2) yields

$$dT = -\frac{q}{2Kr} (r^2 - r_i^2) dr \quad (D-3)$$

Integrating again with the boundary condition

$$T = T_o \text{ at } r = r_o$$

yields

$$T = T_o + \frac{\ddot{q}}{4K} \left[r_o^2 - r^2 - 2r_i^2 \left(\ln \frac{r_o}{r} \right) \right] . \quad (D-4)$$

The last term in the brackets drops out for solid cylinders, and the equation reduces to the familiar:

$$T = T_o + \frac{\ddot{q}r_o^2}{4K} \left[1 - \left(\frac{r^2}{r_o^2} \right) \right] . \quad (D-5)$$

Spheres

The basic steady state conduction equation in the r-direction with internal generation in spherical coordinates is:

$$\ddot{q} = -K \left(\frac{2}{r} \frac{dT}{dr} + \frac{d^2T}{dr^2} \right) \quad (D-6)$$

which can be written as

$$\ddot{q}r^2 = -K \frac{d}{dr} \left(r^2 \frac{dT}{dr} \right) . \quad (D-7)$$

Solving Eq. (D-7) in the same manner as Eq. (D-2), the resulting temperature distribution for a spherical fuel configuration is:

$$T = T_o + \frac{\ddot{q}}{6K} \left[r_o^2 - r^2 - 2r_i^3 \left(\frac{1}{r} - \frac{1}{r_o} \right) \right] . \quad (D-8)$$

The nomenclature for this appendix is:

$\ddot{q}$ = heat generation rate per unit volume

K = thermal conductivity

r_i = inner radius

r_o = outer radius

r = any radius between r_i and r_o

T = temperature at r

T_o = temperature at r_o

APPENDIX E

Capability of the IRHS to Reenter Intact From Lunar Missions (Ref. 34)

This supplementary study has no bearing on the safety of the SNAP-19 IRHS/Nimbus B mission, but it is intended to help determine the suitability of the SNAP-19 IRHS for future missions. Configurations and orientations were selected to produce the most severe thermal environments. The maximum and minimum credible reentry angles for lunar return were investigated.

Previous experience with similar systems indicated that the interior heat source temperatures and ablation depths are greatest during orbital decay and that outer temperatures and temperature gradients are highest during lunar return reentries. Maximum thermal shock occurs if the heat source is initially in the generator and is not released until reentry heating rates have increased to a substantial value. Since peak outer surface temperatures of the IRHS are not reduced significantly by protecting the heat source early in the trajectory, the most severe initial configuration for lunar return would be the IRHS in the generator. A side-on-stable orientation of the IRHS after release from the generator produces the highest local surface temperatures and the largest thermal gradients. Therefore, the parameters selected for the 2 reentries considered in this study were:

1. Reentry angles of -6.25° and -38.01° ,
2. Reentry velocity of 36,084 ft/sec
3. The IRHS initially in the SNAP-19 generator on a tumbling Nimbus B, and
4. A side-on-stable orientation of the IRHS after generator disassembly.

The methods of analysis, ballistic coefficients, heating ratios, and thermal models (Figures 28, 30, and 31) used for the lunar return study were the same as those used for the orbital decay and abort trajectories in the SNAP-19/Nimbus B safety study. Altitudes, velocities, dynamic pressures, and the reference heating

rates versus time from 400,000 feet altitude are given for both trajectories in Figures E-1 through E-4.

Ablation of the IRHS heat shield and heating rate corrections were completed using the Charring Material Ablation (CMA) program. Tables E-I and E-II give the CMA corrected heating rates to the five thermal model surface nodes. Figures E-5 and E-6 give the maximum and minimum recessions of the heat shield for each of the trajectories. The 0.1966 inch of ablation that occurs during the -6.25° lunar return reentry exceeds the ablation of any of the Nimbus B mission reentries. This value would increase approximately four percent if the heat source were released from the generator prior to reentry. During both lunar return trajectories, the graphite surface temperatures exceed the minimum sublimation controlled ablation temperatures. This means small increases in surface temperature would result in large increases in ablation rates. During the -6.25° trajectory, the maximum surface temperature is 550°R into the sublimation controlled regime; during the -38.01° reentry, the maximum temperature is 1420°R into the sublimation regime.

Heat source temperatures were computed with the HERMAN heat transfer program. Time-temperature plots of the stagnation line and aft side of the outer surface of the graphite heat shields and the Haynes-25 capsules are given in Figures E-7 and E-8. The low melting temperature generator fails early in the trajectory, and the IRHS is near the radiation equilibrium temperature throughout reentry. As a result, the generator offers little protection for the IRHS during lunar reentries. During a -6.25° reentry, all of the capsule melts. The maximum liner temperature is near the melting temperature, and the fuel melts during a -6.25° reentry (Figure E-9).

Table E-III summarizes the maximum temperatures, temperature gradients, and ablation depths for both trajectories. The -38.01° reentry produces the highest temperature gradients and highest heat shield temperatures. Maximum radial temperature gradients are given in Table E-III, and circumferential gradients can be obtained from Figures E-7 and E-8. The -6.25° reentry produces the most ablation and highest temperatures of the liner, capsule, and fuel.

Although the IRHS has a substantial thermal margin of safety for orbital missions using the Nimbus B safety criteria, it does not appear to be adequate for lunar missions because of excessive ablation, thermal stresses, and high temperatures.

TABLE E-I
Net Aeroheating Rates for the $\gamma_o = -6.25$ degree
Lunar Return Trajectory

Time (sec)	Surface Nodes				
	<u>G1</u>	<u>G6</u>	<u>G10</u>	<u>G13</u>	<u>G15</u>
0	6	5	3	1	0
30	163	136	85	35	8
40	354	301	181	75	14
50	611	522	327	136	25
60	796	699	466	194	36
66	851	752	518	217	41
72	854	761	524	224	42
78	822	729	504	214	41
84	744	659	452	191	36
90	638	560	377	158	30
100	441	381	247	103	19
110	262	225	144	60	11
120	129	111	73	30	5
140	31	27	18	7	1
170	8	6	3	2	0
250	5	4	3	1	0
350	5	4	3	1	0
440	4	4	3	1	0

TABLE E-II

Net Aeroheating Rates for the $\gamma_o = -38.01$ degree
Lunar Return Trajectory

Time (sec)	Surface Nodes				
	<u>G1</u>	<u>G6</u>	<u>G10</u>	<u>G13</u>	<u>G15</u>
0	6	5	3	1	0
5	484	414	266	110	17
9	1386	1185	751	314	58
10	1983	1696	1075	449	84
11	2514	2221	1422	598	113
12	2531	2400	1728	737	141
13	2321	2180	1642	711	139
14	1771	1569	1044	465	94
15	793	683	457	209	43
17	94	84	62	36	9
20	4	5	7	7	2
30	4	3	3	3	1
50	4	4	3	2	1
100	4	4	3	1	0
250	4	4	3	1	0

TABLE E-III

Maximum Temperatures, Temperature Gradients, and Ablation Depths
During Lunar Return Reentries at $V_O = 36,084$ ft/sec

Parameter	Lunar return $\gamma_O = -6.25^\circ$ side-on-stable IRHS in generator	Lunar return $\gamma_O = -38.01^\circ$ side-on-stable IRHS in generator
Graphite Heat Shield		
Peak temperature ($^\circ\text{R}$)	6345	7619
Impact temperature ($^\circ\text{R}$)	2015	1984
Heating radial temperature gradient ($^\circ\text{R}$)	1455	4836
Cooling radial temperature gradient ($^\circ\text{R}$)	222	300
Ablation at end of heat pulse (in.)	0.1383	0.0469
Ablation at earth impact (in.)	0.1966	0.0954
Tantalum Liner		
Peak temperature ($^\circ\text{R}$)	5680	4155
Impact temperature ($^\circ\text{R}$)	2037	2006
ΔT below melt at $T_{\max}$ ($^\circ\text{R}$)	205	1730
Haynes - 25 Capsule		
Peak temperature ($^\circ\text{R}$)	5680	4155
Impact temperature ($^\circ\text{R}$)	2770	2585
ΔT above melt at $T_{\max}$ ($^\circ\text{R}$)	2796	1271
Fuel		
Peak temperature ($^\circ\text{R}$)	5680	4155
Impact temperature ($^\circ\text{R}$)	2700	2585
ΔT above or below melt at $T_{\max}$ ($^\circ\text{R}$)	1157 Above	368 Below

Figure E-1. Lunar return trajectory of a side on stable IRHS
 $\gamma_0 = -6.24$ degrees, TTA-21256

Figure E-2. Lunar return trajectory of a side on stable IRHS
 $\gamma_0 = -6.25$ degrees, TTA-21256

Figure E-3. Lunar return trajectory of a
side on stable IRHS
 $\gamma_0 = -38.01$ degrees
TTA-21257

Figure E-4. Lunar return trajectory of a
side on stable IRHS
 $\gamma_0 = -38.01$ degrees
TTA-21257

Figure E-5. Lunar return ablation of a side-on-stable IRHS
 $\gamma_0 = -6.25$ degrees

Figure E-6. Lunar return ablation of a side-on-stable IRHS
 $\gamma_0 = 38.01$ degrees

Figure E-7. Lunar return temperatures of the IRHS heat shield and capsule, side-on-stable, $\gamma_0 = -6.25$ degrees

Figure E-8. Lunar return temperatures of the IRHS heat shield and capsule, side-on-stable, $\gamma_0 = -38.01$ degrees

Figure E-9. Maximum lunar return temperatures of the IRHS liner and fuel, side-on-stable, $\gamma_o = -6.25$ degrees

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