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DOCUMENT NUMBER VOY-C1-TR10
15 DECEMBER 1966

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PRELIMINARY SPACECRAFT
THERMAL INTEGRATION
ANALYSIS
RTG STUDY, VOYAGER TASK C

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PREPARED FOR

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
4800 OAK GROVE DRIVE
PASADENA, CALIFORNIA

UNDER JPL CONTRACT No. 951112
MODIFICATION No. 3

GENERAL ELECTRIC
MISSILE AND SPACE DIVISION
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TABLE OF CONTENTS

<u>Section</u>		<u>Page</u>
1	INTRODUCTION	1-1
2	CONCLUSION	2-1
3	TECHNICAL INVESTIGATIONS	3-1
	3.1 RTG Thermal Interaction During Enshrouded Period in Parking Orbit	3-1
	3.2 Effect of RTG Heat on Spacecraft Elements	3-5
	3.3 RTG/Shroud Interactions	3-8
	3.4 RTG Ground Cooling	3-11

LIST OF ILLUSTRATIONS

<u>Figure</u>		<u>Page</u>
3-1	Model Used to Determine Shroud Temperature Rise	3-1
3-2	Shroud Temperature for RTG-Powered Spacecraft	3-3
3-3	Inside-Shroud Temperature Variation with K/ ΔX Aluminum Honeycombs	3-3
3-4	RTG Root Temperature History	3-4
3-5	Relationship of RTG Heat Rejection Temperatures of Enshrouded and Free-Space Operational Conditions	3-4
3-6	Voyager RTG Thermal Analysis Nodal Breakdown	3-6
3-7	Ascent Heat Flux on Bottom Shroud	3-7
3-8	Voyager RTG Thermal Analysis Nodal Breakdown	3-10
3-9	External Shroud Heat Source	3-11
3-10	Shroud Isotherms due to RTG Thermal Radiation	3-12
3-11	Effect of Cooling Air on Electronics Bay Temperature	3-14
3-12	Effect of Cooling Air on Superinsulation Temperature	3-14
3-13	Effect of Cooling Air on Shroud Temperature	3-15
3-14	Effect of Cooling Air on RTG Temperature	3-15
3-15	Air Conditioning Required for Various Air Cooling Temperatures	3-16
3-16	Electronics Temperature for Varying Inlet Air Flow Rates	3-16

SECTION 1

INTRODUCTION

A series of analyses of various problems relating to the integration of RTG's with the Voyager Spacecraft is presented. These analyses refer to a variety of spacecraft and RTG configurations, inasmuch as the analyses were performed during a period of configuration exploration.

A later Design Definition Report will be issued which will include the thermal integration of the RTG's with the selected spacecraft configuration in detail. The purpose of the studies reported here is to determine the nature and extent of the thermal integration requirements.

SECTION 2

CONCLUSIONS

Preliminary evaluation of the thermal integration of RTG's with the Voyager Spacecraft Bus has resulted in the following conclusions:

- a. Prior to shroud separation, RTG temperatures will not increase more than 35°F over free space equilibrium temperatures.
- b. No directed air precooling of the RTG's is required.
- c. Prior to shroud separation, electronic equipment temperature increases will not exceed 11°F in comparison with a solar-powered spacecraft.
- d. Shroud temperature gradients are not expected to exceed $1^{\circ}\text{F}/\text{inch}$. Maximum local shroud temperatures will not exceed 136°F .
- e. To maintain the electronics equipment during prelaunch at a maximum temperature of 120°F will require air flows of 4 lb/sec at inlet temperatures of 70°F .

SECTION 3

TECHNICAL INVESTIGATIONS

3.1 RTG THERMAL INTERACTIONS DURING ENSHROUDED PERIOD IN PARKING ORBIT

The first analysis performed was that of a one-dimensional model of the RTG/shroud thermal radiation interactions for a typical parking orbit period. This model was selected in order to uncover any gross thermal problems caused by the heat rejection from the enshrouded RTG's. The shroud model consisted of two 60-mil aluminum skins with a honeycomb matrix in between. The outside of the shroud was exposed to direct solar, albedo, and earth infrared radiation in the time sequence of a 90-minute orbit, the first half of which is in sunlight. The shroud temperature was studied using a range of thermal conductance values for the honeycomb portion of the shroud.

The shroud thermal model is illustrated in Figure 3-1. The inner shell receives thermal radiation from the RTG's, radiates to the spacecraft, and has a conductive coupling to the outer shell of the shroud. In this model both the inner and outer shells of the shroud are assumed to be isothermal. Later investigations examined the effects of the circumferential distribution of solar, earth infrared, and RTG heat fluxes.

Figure 3-1. Model Used to Determine Shroud Temperature Rise

The average shroud temperatures are shown in Figure 3-2. The outer skin temperature drops rapidly from its initial assumed value of 70°F , while the inner surface experiences small changes. When the spacecraft passes into the earth's shadow both temperatures drop considerably.

The average shroud temperature for the first orbit is shown in Figure 3-3 as a function of the honeycomb thermal conductance and shroud surface area. Investigations of the thermal conductance of 4-inch-core aluminum honeycomb showed expected values as high as $3.0 \text{ BTU/hr-ft}^2\text{-}^{\circ}\text{F}$, and may be considerably lower. The effects of having less than the full shroud area contributing to RTG waste heat rejection were investigated by obtaining temperature histories of the shroud in which only two-thirds of the area and heat capacity were employed. Analyses with less than the full shroud area and heat capacity were performed, since it appears that the shroud will receive thermal radiation from the RTG's which are partially focused on a portion of the shroud.

A second set of calculations was made in which the shroud was divided into two sections, one facing the sun and one facing the earth. Each section was isothermal. The RTG's radiated to both sections, which in turn could exchange thermal radiation with each other. Comparison of results of these cases with the previous simplified one-dimensional approach showed that the inside of the sunlit half of the shroud was 10°F hotter in the split-shroud analysis. Further details of the shroud temperature were obtained through a detailed, multinodal thermal analysis, and are reported in later sections of this report.

These values of the heat sink temperature permitted determination of the RTG temperature variation during the initial parking orbit. Figure 3-4 shows the temperature of the RTG (fin root) during this period. Since an RTG design had not been established, generators with operational heat rejection temperatures of 300, 400, and 500°F were studied. The temperature of the RTG's rises rapidly, and within the 90-minute period almost reaches the enshrouded equilibrium condition. The relationship between the free space operational and enshrouded equilibrium temperatures is shown in Figure 3-5. The effect of the initial temperature on the temperature after 90 minutes in parking orbit with the shroud on was

Figure 3-2. Shroud Temperature for RTG-Powered Spacecraft

Figure 3-3. Inside-Shroud Temperature Variation with $K/\Delta X$ Aluminum Honeycombs

Figure 3-4. RTG Root Temperature History

Figure 3-5. Relationship of RTG Heat Rejection Temperatures of Enshrouded and Free-Space Operational Conditions

examined for initial RTG outer case temperatures of 150, 200, and 250^oF, and little difference was found. It was concluded that since the enshrouded RTG equilibrium temperature is only about 35^oF above the free space operational condition, and since the RTG is capable of being overheated 50^oF early during its life without detrimental effects on life or performance, there was no value to the RTG operation to be gained by precooling the RTG's.

3.2 EFFECT OF RTG HEAT ON SPACECRAFT ELEMENTS

A thermal analysis was performed on the Voyager spacecraft to determine the effects of the RTG waste heat on the spacecraft temperatures during flight in a parking orbit with the vehicle enshrouded. The reference spacecraft configuration used was that of the General Electric Task B design, with four 150 watt (elec) RTG's spaced evenly around the circumference at the plane of the solar array (see Figure 3-6). The spacecraft was modeled using 37 nodes to represent the various elements. Included in the analysis are the effects of the thermal radiation from the RTG's on the shroud and spacecraft, including those of multiple reflections. The major assumptions include:

- a. Isothermal inner and outer shroud surfaces
- b. Thermal conductance of shroud 1.0 Btu/hr-ft²-0^oF
- c. Mass and area of shroud 2/3 actual value
- d. Shroud outer coating $\alpha = 0.3$, $\epsilon = 0.8$
- e. Shroud inner coating $\alpha = \epsilon = 0.9$

The last assumption -- that of a high emissivity coating on the inside of the shroud -- represents a departure from the Task B configuration. The solar-cell-powered spacecraft employed a low emissivity coating on the inside of the shroud, since there was no heat rejection requirement prior to shroud separation. An investigation was made of the effects of emissivity of the inner shroud surface on spacecraft temperatures for a solar-powered vehicle during ascent (when aerodynamic heating occurs). The aerodynamic heating is shown in Figure 3-7 for a location 30 feet from the stagnation point of the nose cap. Since aerodynamic heating of side boundary layers is substantially less than that in stagnation

Figure 3-6. Voyager RTG Thermal Analysis Nodal Breakdown

Figure 3-7. Ascent Heat Flux on Bottom Shroud

regions, these heat fluxes are appreciably lower than those which would be encountered with a smaller (diameter and length) shroud which permitted the spacecraft to have a view of the inward side of the stagnation region of the nose cap. In the Saturn V/Voyager case the upper planetary vehicle is shielded from receiving radiant heat from the inner surface of the nose cap portion of the shroud by a dirt and contamination barrier. The higher emissivity on the inside of the shroud helps to keep the shroud isothermal inasmuch as heat from the sunlit side is radiated away by the shaded portion of the shroud.

The spacecraft temperatures were compared for two cases: with and without the RTG heat. This comparison is presented to show the effects of the RTG's in comparison with a solar-cell-powered vehicle. The temperature differences are felt to be more significant than the absolute values. Table 3-1 shows the thermal effects of the RTG's on the spacecraft during enshrouded flight. The temperature changes are within the limits which can be met by the subsystem capabilities.

Table 3-1. Thermal Effect of RTG's on Voyager Spacecraft During First 90 Minutes

Location	Node No.	Maximum Temperature with RTG's Minus Maximum Temperature without RTG's	Minimum Temperature with RTG's Minus Minimum Temperature without RTG's
Middle of Main Propulsion Tank	4	1	0
Middle of Main Propulsion Tank	5	1	0
Spacecraft Structure	15	4	0
Equipment Bay	26	11	0
Superinsul. Top Layer on Outside of Spacecraft	28	46	11
Inside of Shroud	36	19	70

NOTE: (a) Temperatures in $^{\circ}\text{F}$; (b) 4 RTG's, 150 watts each.

3.3 RTG/SHROUD INTERACTIONS

The preliminary analysis, based on a one-dimensional isothermal shroud model, indicated that the RTG's and associated spacecraft components would not overheat during enshrouded operation in parking orbit. A more detailed analysis of the enshrouded spacecraft was performed in order to determine the shroud temperature variation. This analysis was to be used to obtain shroud hot spots and to determine their effects on spacecraft elements. Shroud hot-spot information was to be used to determine the extent of thermal distortion of the shroud. However, since a new spacecraft configuration has been developed, this work on thermal distortion has been postponed until the two-dimensional temperature profile is available for the new configuration. At this time the present configuration shroud distortion problem can be examined.

The RTG configuration considered was that of four 150-watt finned cylindrical units. It was expected that these would produce maximum shroud hot spots. The RTG model was that of

a right circular cylinder with the same temperature as the fin root and a circumferential area equal to the effective heat rejection area of the finned cylinder. The length used in the analysis was equal to that of a 150-watt RTG.

The nodal model is illustrated in Figure 3-8, which shows the spacecraft model and the manner in which the shroud was divided for the analysis. The shroud was first cut into four segments for the purpose of radiation exchange with the spacecraft. One quadrant of the shroud was then subdivided into twenty-five segments so that the temperature distribution of the shroud could be determined. An interior and exterior node was located at each shroud division. Each of the twenty-five segments was 3.15 feet square. The RTG centerline was opposite the center of node 48, which is 30% of the height of the heat rejecting portion of the shroud.

The solar, albedo, earth, and aerodynamic heat fluxes to the exterior of the shroud are given in Figure 3-9.

Figure 3-10 shows the temperature distributions on the interior of the shroud at 15, 30, 45, and 90 minutes after launch. The maximum temperature of the shroud is 136^oF at a location remote from the electronics equipment. The largest temperature gradients do not exceed 1^oF/inch.

The average temperatures of the inside of the other quadrants of the shroud are presented in Table 3-2.

Table 3-2. Average Temperature of Earth- and Horizon-Facing Quarants

Time After Launch (Minutes)	Earth-Facing Quadrant	Horizon-Facing Quadrant
15	104	91
30	112	93
45	112	92
60	106	86
75	103	82
90	102	80

Figure 3-8. Voyager RTG Thermal Analysis Nodal Breakdown

Figure 3-9. External Shroud Heat Source

3.4 RTG GROUND COOLING

During pre-launch operations the RTG heat must be removed from the spacecraft. If left alone the RTG would radiate thermally to the shroud, and the air inside the shroud would circulate due to free convective bouyancy forces. The purpose of this section is to investigate the need for auxiliary air circulation for RTG heat removal.

A 10°F temperature rise of externally supplied air would result in an air flow of 7.6 lb/sec per spacecraft. With this air flow the temperature of the electronics bays rises less than 20°F (worst case with injected air at 90°F) above the value to be expected with a solar-cell-powered spacecraft. It appears that gross pre-cooling of the air is not required and that the temperatures to be expected in the electronics bays are not excessive.

The analyses were performed using the multinodal thermal model of the spacecraft previously described. Air, introduced at temperatures of 50, 70 or 90°F at the bottom of the shroud, rises by free convection, thereby cooling the RTG's, superinsulation, and

15 Minutes
After Launch

30 Minutes
After Launch

45 Minutes
After Launch

60 Minutes
After Launch

75 Minutes
After Launch

90 Minutes
After Launch

Figure 3-10. Shroud Isotherms Due to RTG Thermal Radiation

equipment bays. The air is assumed to be vented or removed from the top of the shroud. The outside of the shroud is subjected to solar, albedo, and earth infrared fluxes, and the heat is dissipated by means of radiation and natural convection to a 90°F environment.

It was assumed that internal mixing of the air circulating within the shroud with that injected for cooling purposes would result in a temperature rise (of the air flowing over the electronics) of 5°F above the inlet air temperature. This assumption is based on the in-pad cooling experienced gained during the Nimbus B ground tests.

Figure 3-11 shows the dependence of the temperature of the electronics bays on that of the cooling air. The influence of effective convection area of the RTGs is shown parametrically. Figures 3-12 and 3-13 similarly show the temperatures of the superinsulation and the average of the inner surface of the shroud. The RTG fin root temperature is shown in Figure 3-14. The free-space operational fin root temperature is 400°F . The air conditioning tonnage is presented in Figure 3-15.

The dependence of the temperature of the electronics on the air flow rate is shown in Figure 3-16. With no air cooling the temperature rises to 163°F , in comparison with 119°F for the solar-cell-powered spacecraft. Air flows of 3.5 to 4 lb/sec significantly reduce this difference. Further work on the reference design will examine transient operation to determine the extent to which cooling air is required during various transportation phases.

Figure 3-11. Effect of Cooling Air on Electronics Bay Temperature

Figure 3-12. Effect of Cooling Air on Superinsulation Temperature

Figure 3-13. Effect of Cooling Air on Shroud Temperature

Figure 3-14. Effect of Cooling Air on RTG Temperature

Figure 3-15. Air Conditioning Required for Various Air Cooling Temperatures

Figure 3-16. Electronics Temperature for Varying Inlet Air Flow Rates