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AIMP-D THERMAL DESIGN REPORT

S. OLLENDORF

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AIMP-D THERMAL DESIGN

REPORT

S. Ollendorf

July 1968

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CONTENTS

CONTENTS (Continued)

AIMP-D THERMAL DESIGN REPORT

INTRODUCTION

The primary objective of the AIMP-D spacecraft was to investigate, in the vicinity of the moon, the characteristics of the interplanetary magnetic field, solar plasma flux, dust distribution, solar and galactic cosmic rays, as well as to study the magnetohydrodynamic wake of the earth in the interplanetary medium at lunar distances every 29.5 days. The launch vehicle for the mission was the Thrust Augmented Improved Delta with an FW-4 third stage. The retro-motor, a Thiokol TE-M-458, was used to slow the spacecraft to permit its capture by the lunar gravitational field. Figure 1 shows a typical flight plan. Figures 2 and 3 show an over-all picture and a side view of the spacecraft. The AIMP-D structure consisted of a two-piece magnesium axial-thrust tube with a Delta attach flange

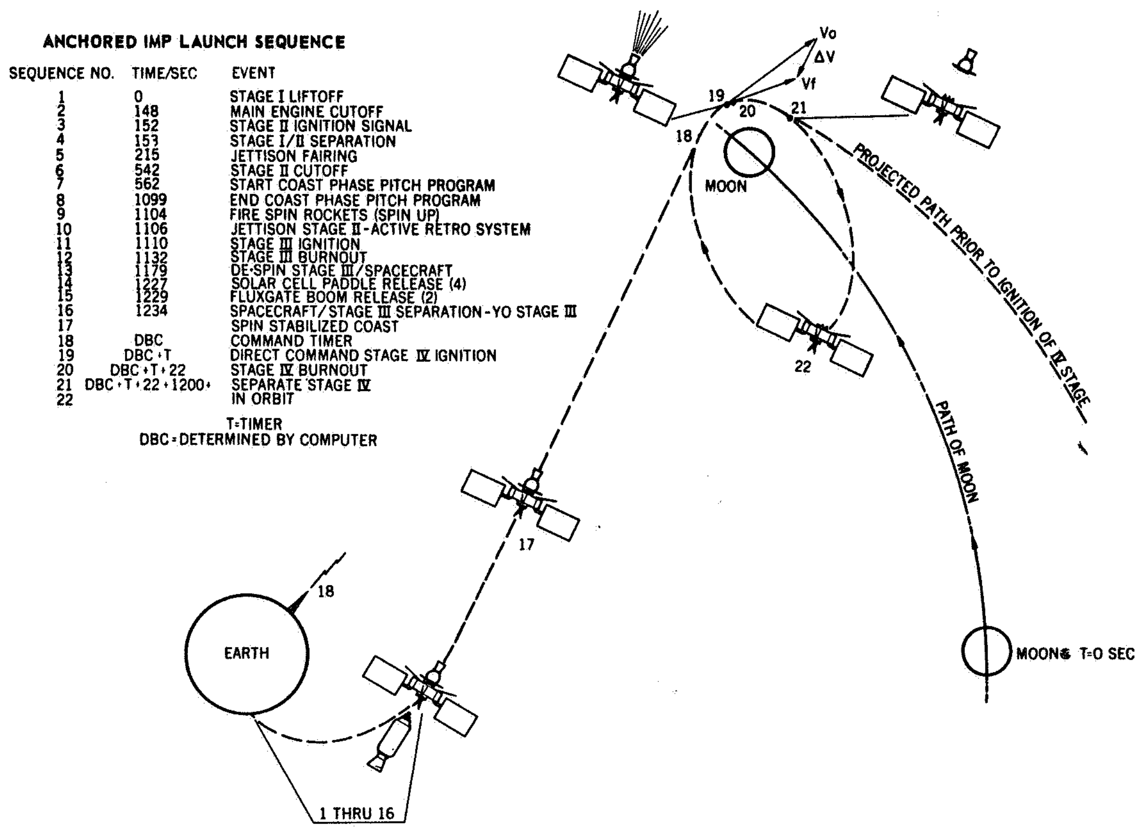


Figure 1. Typical Flight Plan

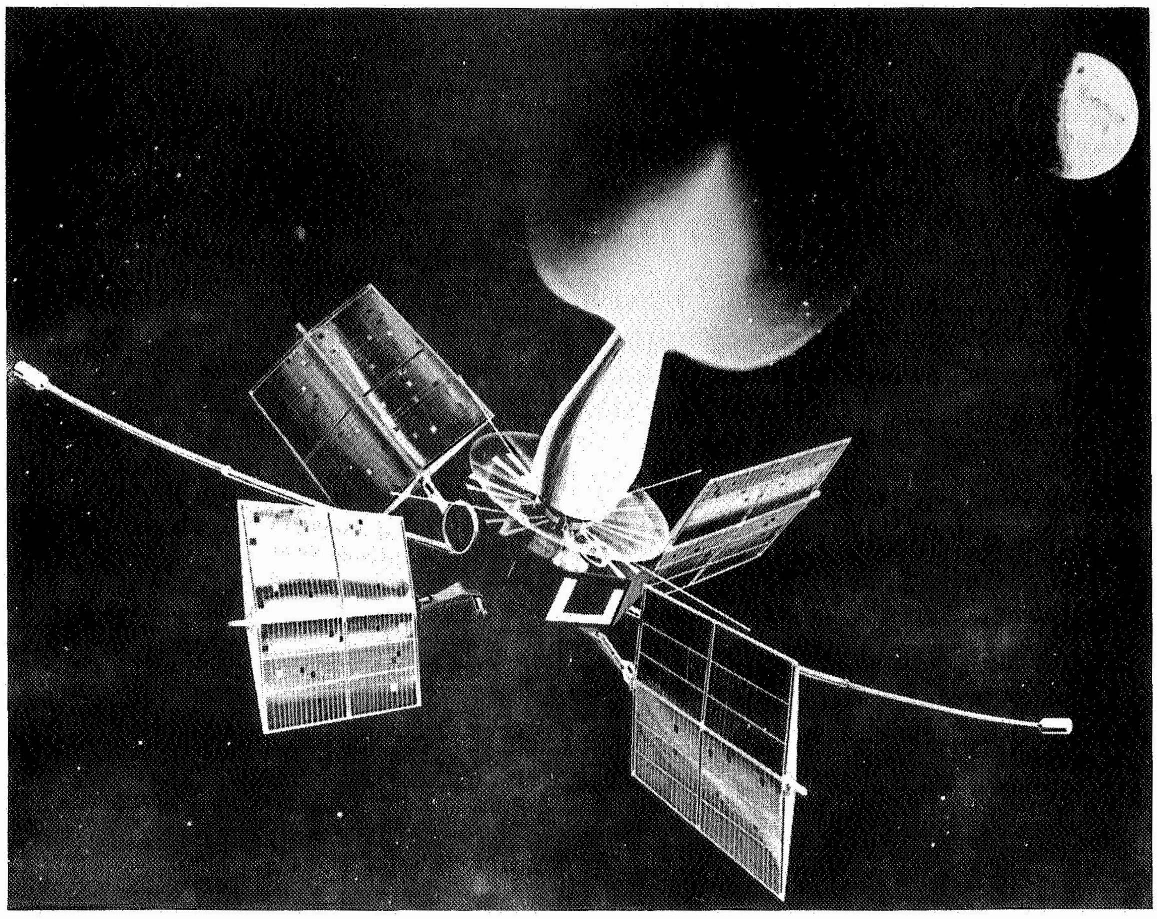


Figure 2. AIMP-D at Retro-Firing

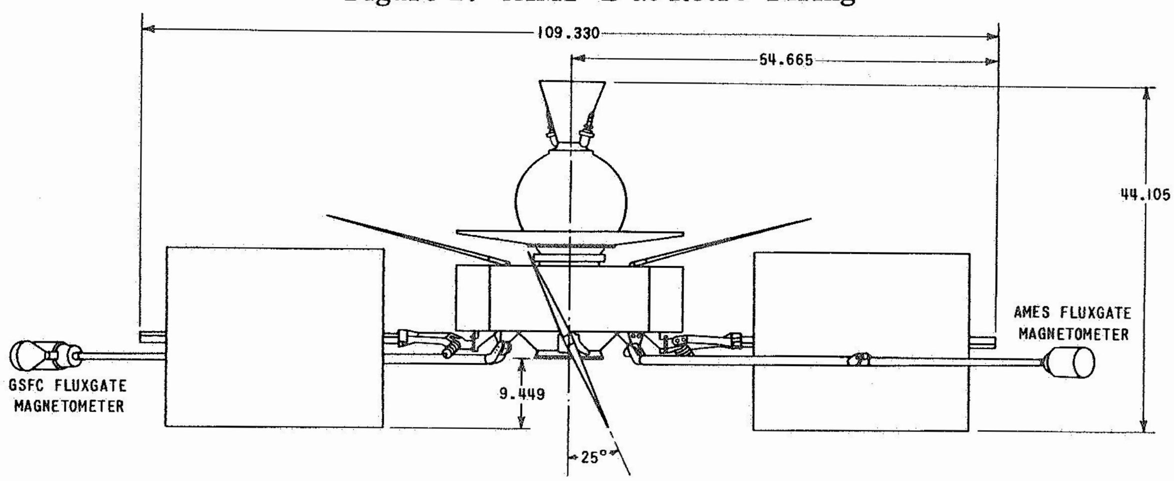


Figure 3. AIMP Spacecraft Side View

on one end and a retro-motor flange on the other; an octagonal aluminum-honeycomb equipment shelf with eight radial support struts; an aluminum-honeycomb top cover; and four support brackets with arms for solar cell paddles and booms for magnetometers. Experiments and electronics were mounted on the periphery of the equipment shelf in modular support frames. The top cover enclosed all equipment mounted on the shelf and provided exterior surfaces for thermal control.

PHASE I - THERMAL CONTROL FEASIBILITY STUDY

During the first part of the AIMP-D program, a design feasibility study was initiated in which various thermal designs were considered. The IMP-A analytical model was used during this design study. The study was primarily initiated to determine whether or not the AIMP-D spacecraft was to have active or passive temperature control. Two basic configurations were used in the study. The first configuration was the transfer configuration which had a rocket engine on top of the spacecraft. The second was the orbit configuration, which did not have the rocket engine attached to it. An analysis was performed applying various percentages of active temperature control in the form of shutters to various surfaces of the spacecraft. Two possibilities that existed were: (1) Shutters on the top and bottom surfaces, and (2) shutters on the top, bottom, and side surfaces. A modified heat transfer computer program which included variable emittance as a parameter, was used to determine what the effect of active temperature control could be. The spacecraft was taken through various sun/spin axis angles and through long shadows to determine its thermal response to the environment. It was found that approximately 40% of the spacecraft top and bottom would have to be actively controlled in order to keep the average temperature between 10° and 25° under steady state conditions and hold the battery above -10° C in a long shadow, which was anticipated to be of the order of eight hours. The findings of this study were given to the AIMP-D project office for their evaluation. It was decided by the project office to use passive thermal design on the AIMP-D spacecraft for the following reasons: (1) covering the spacecraft with 40% active temperature control would be prohibitive from a weight standpoint; (2) due to the fact that many experimenters required quite wide look angles from the side of the spacecraft, active temperature controllers would necessarily restrict them; (3) the uncertainty of active temperature controllers, in the form of shutters, which had never been applied externally to a spinning spacecraft. Calculations showed that the effective absorption of energy could become quite high if the shutters were left open to solar environment. Another significant factor, from a thermal design viewpoint, was that at the 90° solar aspect the spacecraft presented a very small projected area to solar input. The temperatures during this time are quite cold within the spacecraft. If shutters were mounted on the outside surface they would tend to close to retain the heat thereby setting up a radiation barrier to the surface of the spacecraft and possibly causing an even deeper depression in temperature. On the other hand, solar heat on the shutters could cause them to open inadvertently. The opposition of these effects could possibly have lead to an oscillation in the shutter system. These factors, coupled with the schedule and program presentation to NASA Headquarters, which outlined AIMP-D as a slightly modified version of the IMP A, lead the program office to a passive thermal control system for the AIMP-D.

PHASE II - PRELIMINARY THERMAL DESIGN

MISSION REQUIREMENTS

The following mission requirements were stipulated by the AIMP-D project as thermal design conditions:

Launch Phase

During this condition the spacecraft is exposed to fairing heating for approximately 210 seconds with the temperatures of the fairing reaching as high as 250° F adjacent to the spacecraft and as high as 480° F above the spacecraft on the tapered section of the fairing.

Coast Period

Upon fairing ejection the spacecraft, in a non-spinning condition, has a variable solar input imposed on it for a period of approximately 18 minutes while the second stage is pitching over.

Third Stage Burn

After third stage burn the spacecraft paddles are exposed to heating from the third stage case.

Transfer

After ejection into a lunar transfer orbit, the spacecraft receives 100% sun for approximately 72 hours, at a solar aspect which varied from 120° to 150° dependent upon time of launch. No transfer shadows were anticipated during this period.

Orbit

After ejection into orbit, the spacecraft would undergo the following environmental conditions: (a) 100% sunlight for all solar aspect angles; (b) 1 to 2 hours shadows yielding approximately 80% sunlight; (c) shadows up to the order to 6.8 hours; (d) long penumbra type shadows, and (e) the possibility of going into a highly eccentric earth orbit as an alternate mission.

Design Effort

Improvements over IMP A Design - During the early stages of the mechanical design of the AIMP-D, quite a few improvements were instituted over the IMP A design. Specifically, the two hottest components in the spacecraft, namely the transmitter and the prime converter, were located opposite one another in the spacecraft and on the bottom shelf. Conduction paths between facets were improved through the use of aluminum straps and a "C" frame design. See Figure 4. Conductive foil was placed under the transmitter and prime converter together with copper screws attached through the bottom shelf, which acted as heat sinks for these high power dissipators. Aluminum honeycomb was used in the top cover and bottom shelf instead of fiberglass as in the IMP A. The battery was set in the center tube as in previous IMP spacecraft. However, it was more strongly radiatively coupled to the center tube, bottom and top spring seats through the use of all black surfaces.

Rocket Engine Requirements

An important design feature, which was applied to the rocket engine mounted on top of the spacecraft, was the provision of an isolated motor mount made of fiberglass which conductively decoupled the rocket from the spacecraft and protected it from the rocket soakback heat after burnout. A thermal blanket was provided for the fourth stage rocket engine which kept it between the operating limits of 0 to 120° F, as stipulated by the manufacturer, during the long coast period in transfer to the moon. The blanket consisted of ten layers of aluminized Kapton, a high temperature film, embossed to reduce the number of contact points between layers. Alternate layers of glass paper were used as separators between the Kapton layers. The number of layers chosen was found to be optimum in a report on OGO thermal insulation issued by TRW, Inc. (Reference 1.)

In anticipation of heating effects from the rocket exhaust plume, a plume shield was provided. The plume shield consisted of a transparent film on which was vapor deposited an infra-red reflective coating. The object of the plume shield was to transmit solar energy to the top cover for high aspect launch conditions but reflect energy from the rocket plume during rocket firing. The coating chosen was an I-R 71-E Libby-Owens–Ford coating which transmitted on the order of 50% of the solar energy and reflected 80% of the energy beyond one micron.

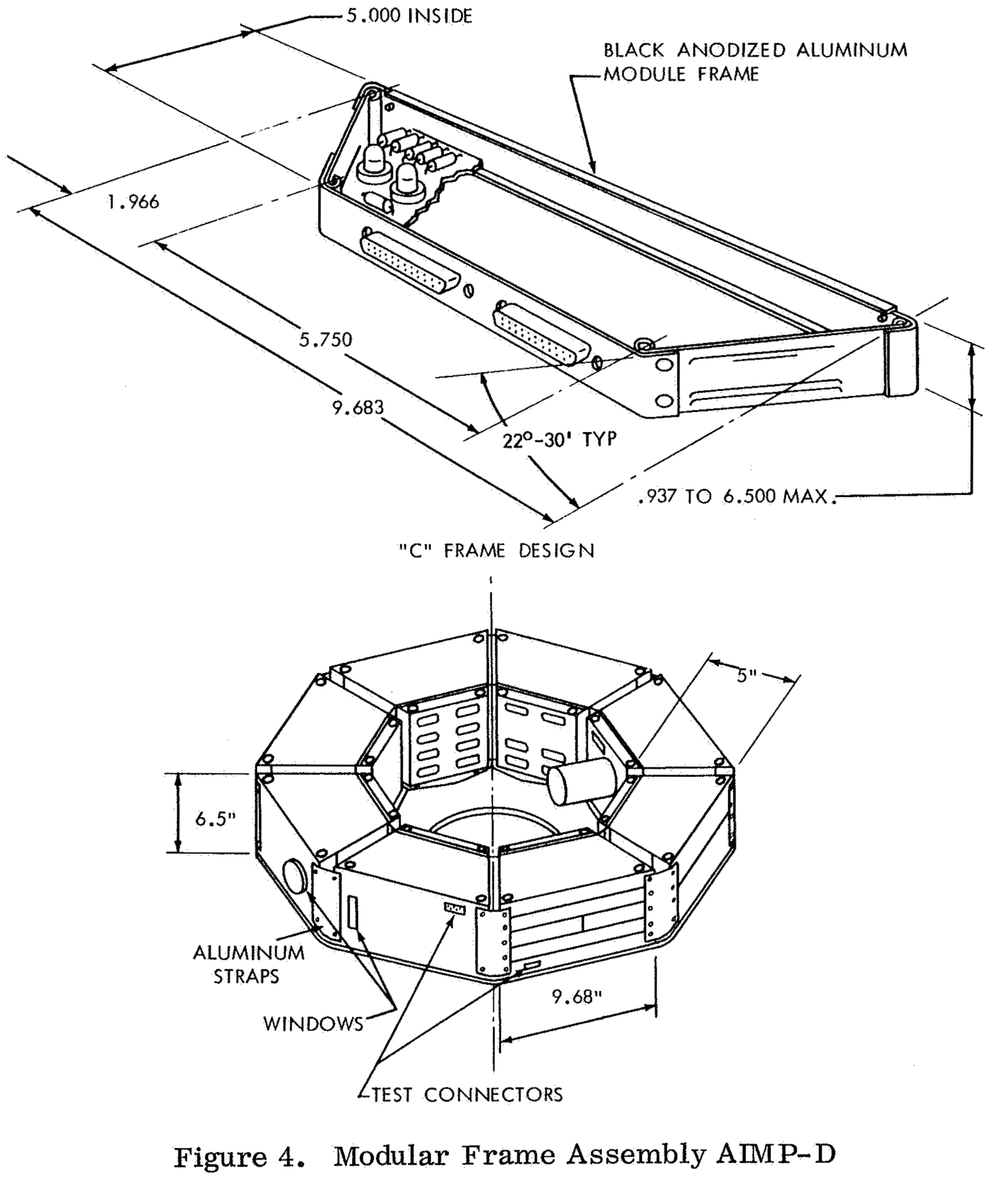


Figure 4. Modular Frame Assembly AIMP-D

Data taken on previous firings (Reference 2) indicated that the rocket plume consisted largely of energies of the order of 1 to 3 microns or greater. The thin film used as a substrate for this coating was Aclar, which has a good resistance to UV shrinkage while maintaining its transmissivity. The Aclar was bonded to a fiberglass support structure with a minimum of blockage area so as not to shadow the spacecraft solar input.

Thermal Analysis

(a) Methods and Computer Programs - Based on this preliminary design, a thermal analysis was performed. A 45 node analytical model (figure 5) was developed which took into account radiative and conductive paths around the spacecraft. Contact conductance values from reference 3 for both non-filled and foil-filled joints were used. The Confac II radiation program was used to determine the radiation coupling factors around the internal portions of the spacecraft. A modified version of the Temperature Control Section Spinning Flat Plate program was used to determine the moon input both from the IR direct heating and the albedo. An albedo factor of .17, maximum, was used (reference 4); surface temperatures of moon are plotted in figure 6. Photos were taken of the space-

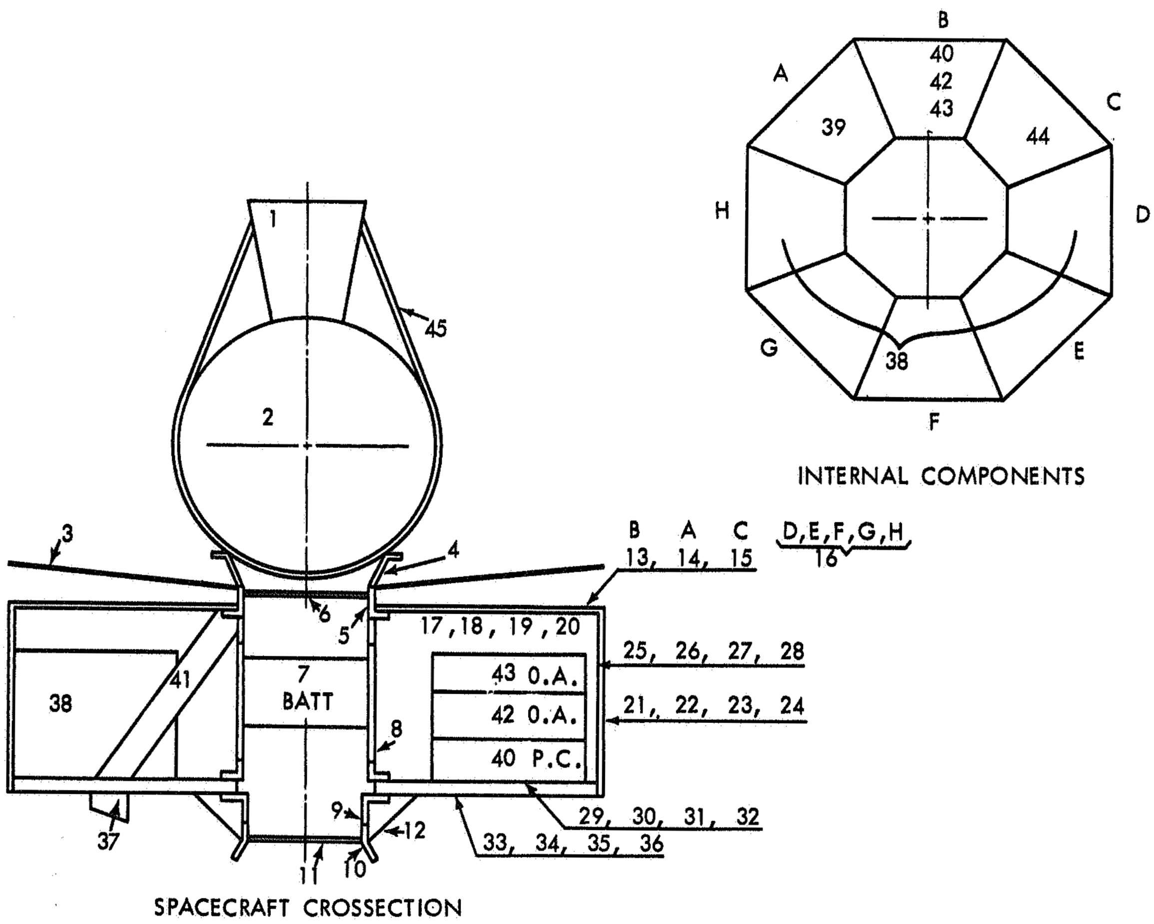


Figure 5. 45 Node Network AIMP-D

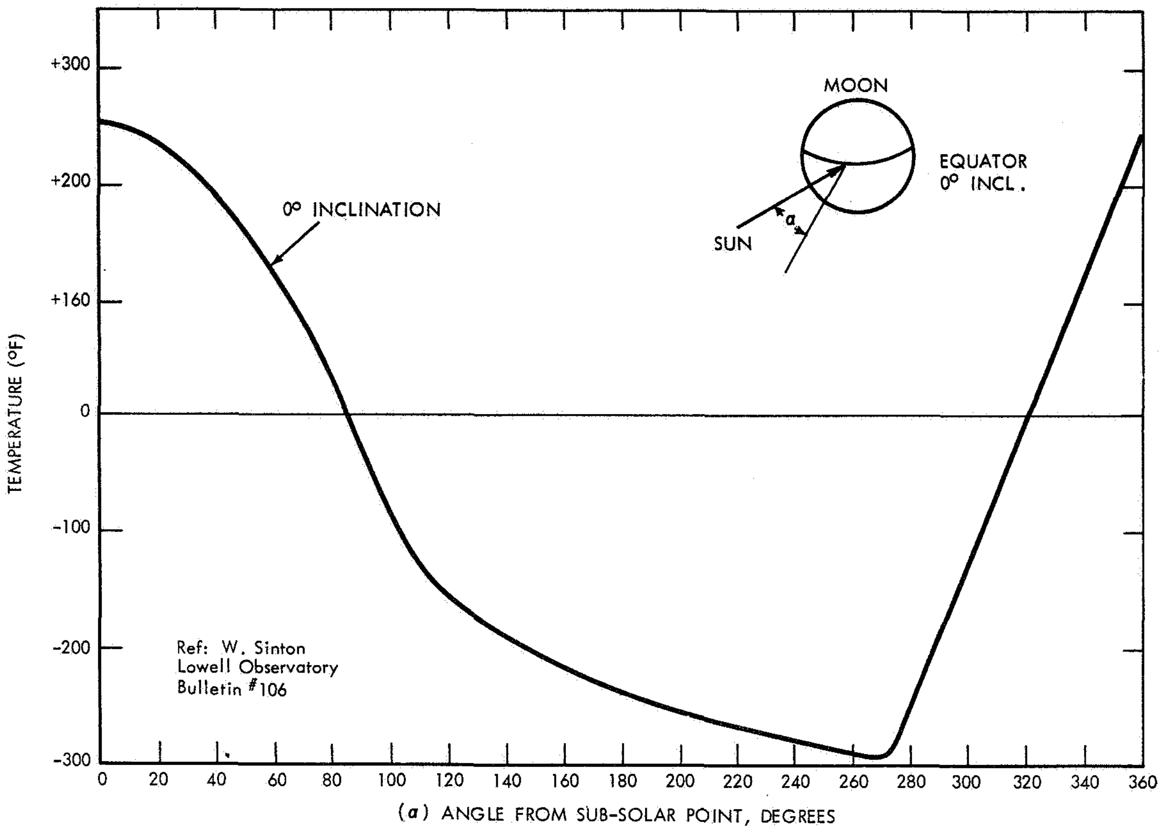


Figure 6. Temperature Profile of Moon

craft throughout all aspects and projected areas of all surfaces measured using a planimeter. A program developed at TRW, Inc., was used to develop plume heating from the rocket engine. Inputs to the program were provided by the Thiokol Corporation on the rocket engine configuration, fuel and nozzle design. From this TRW supplied to Goddard temperature profiles and particle concentration for the solid and aluminum oxide constituents of the rocket engine plume. See figures 7 and 8.

Results of Spacecraft Analysis

Using the analytical model, coating patterns were determined which attempted to keep spacecraft temperatures between 0°-40° C for both the transfer and orbit configurations. See figures 9 to 12.

It was found that the sun spin axis angle had to be restricted during transfer to the moon to angles greater than 30° (see figure 9). This was due to the fact that at aspects between 0 and 30 degrees shadowing from the fourth stage

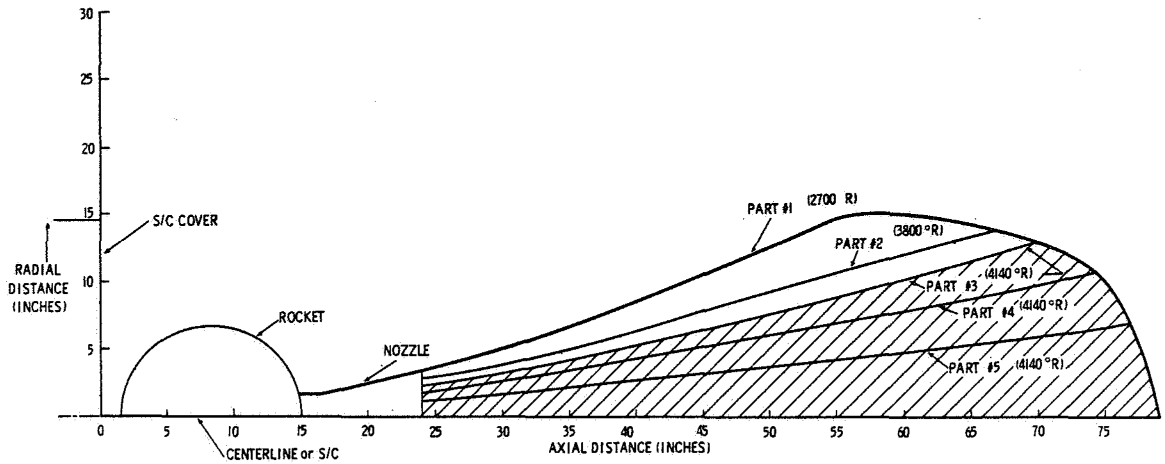


Figure 7. Exhaust Plume Characteristics Thiokol-TE-458 Rocket, Particle Temperature (° R)

rocket engine reduces solar input and depresses the internal temperatures below the desired lower limits of 0° C. It was also found that a requirement to hold orbital elements to pericynthions and apocynthions greater than approximately 2,000 KM and 3,500 KM respectively would limit the effects of moon radiation and albedo on the spacecraft. These were the only restrictions imposed by the temperature control section on the project, with respect to mission constraints.

Applying the methods developed by TRW, reference 2, the effective emittance of the plume was determined and average radiating temperatures at various look angles were determined (see Table I). Using this data, heating rates to the spacecraft cover were determined. The maximum heating rate at the outermost edge of the cover calculated to be 2.45 solar constants.

Experiments and Solar Paddle Analysis

Experiments and solar paddles were handled on an individual basis.

UNIVERSITY OF IOWA

This experiment consisted of a T-shaped section with a set of three Geiger tubes set 90° apart. This section was coupled very well to the experiment card internal to the spacecraft. A P-N junction and brass cup surrounding it, were also external to this experiment. The extreme most Geiger tubes which looked out along the spin axis were set 11 inches apart. Foam potting internal

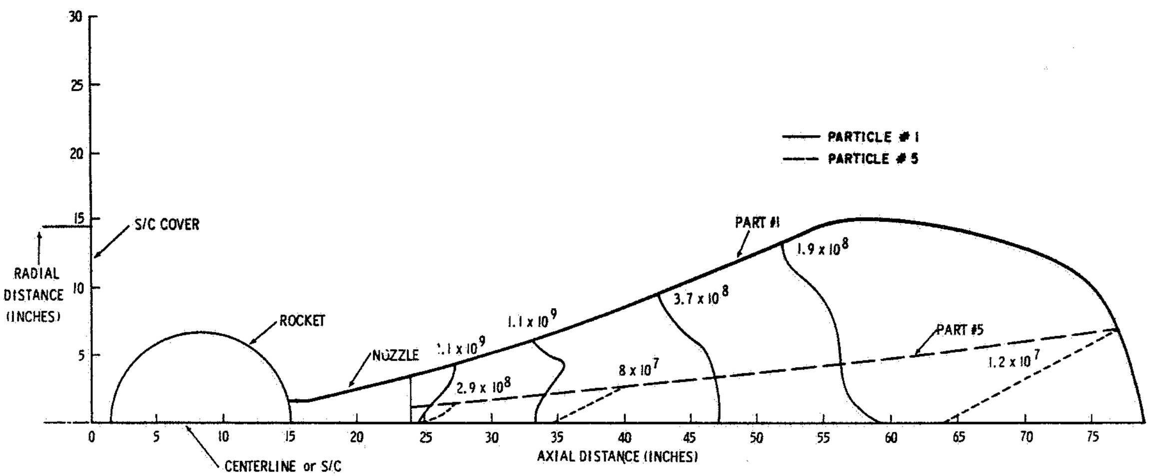


Figure 8. Exhaust Plume Characteristics Thiokol-TE-458, Particle Concentration (/ft.^{2})

to the T-section decoupled each tube from one another. The required temperature limits were 0 to 40° C. In order to keep these tubes at the proper temperature in the near-isolated condition from each other and from the spacecraft, thermal coatings had to be applied to the outside surfaces of the experiment. These consisted of roughly 93% evaporated aluminum and 7% white paint which gave an α / ϵ ratio of 1.1 together with a low emittance $\epsilon = .142$ which helped keep it from getting too cold during long shadow conditions.

MIT

This experiment was mounted on the side of the spacecraft and consisted of a 4-1/2" diameter gold cup. There were no special thermal coatings applied to this experiment. Its appearance to space was a black body and was strongly coupled internally to its card inside the spacecraft.

GODDARD MAGNETOMETER

The thermal analysis for this experiment was performed by R. Hoffman of the Thermal Systems Branch who had previous experience on similar experiments for the Pioneer spacecarft. The experiment was mounted on a boom approximately 5 feet from the center of the spacecraft. The design philosophy here was

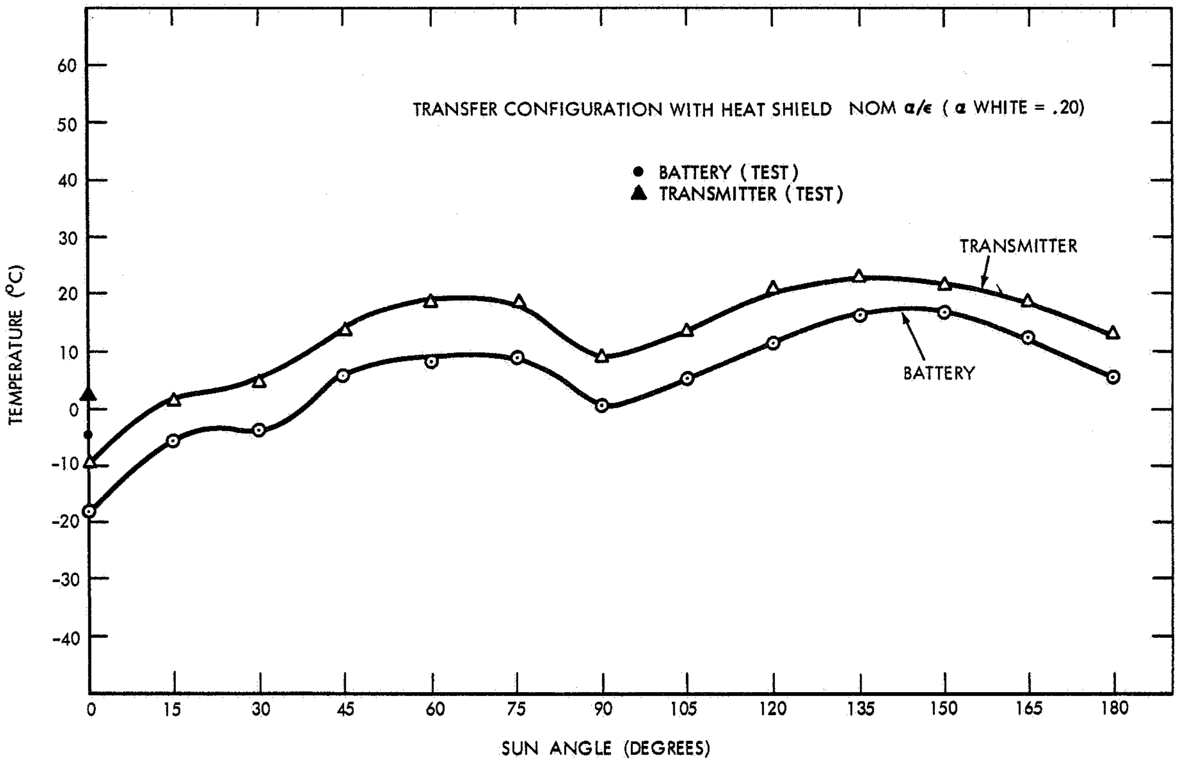


Figure 9. Engineering Test Unit Battery & Transmitter Temperatures

to keep a paraffin filled flipper mechanism internal to the experiment below its actuation temperature 60° C. Applying heat to a flipper actuator, approximately 4 watts for 10 minutes, would cause a rise in temperature above 60° C and actuate the flipper assembly. In order to minimize heat leaks from the flipper assembly, the flipper was conductively and radiatively decoupled from the canister itself. This was done by means of Teflon inserts and bushings at the flipper assembly and vapor deposition of aluminum on the inside surfaces of the canister. The two main design conditions that the flipper had to work under were the 90° aspect, which gave minimum sunlight and minimum projected area and 30° aspect which gave maximum sunlight and maximum projected area. By applying 18% of white paint and 82% evaporated aluminum to the sides of the canister and 7% black 93% evaporated aluminum to its end, an α / ϵ ratio of 1.04 was achieved. This gave roughly a 30° C internal temperature during 100% sunlight maximum area and -5% internal temperature during 90° aspect minimum area.

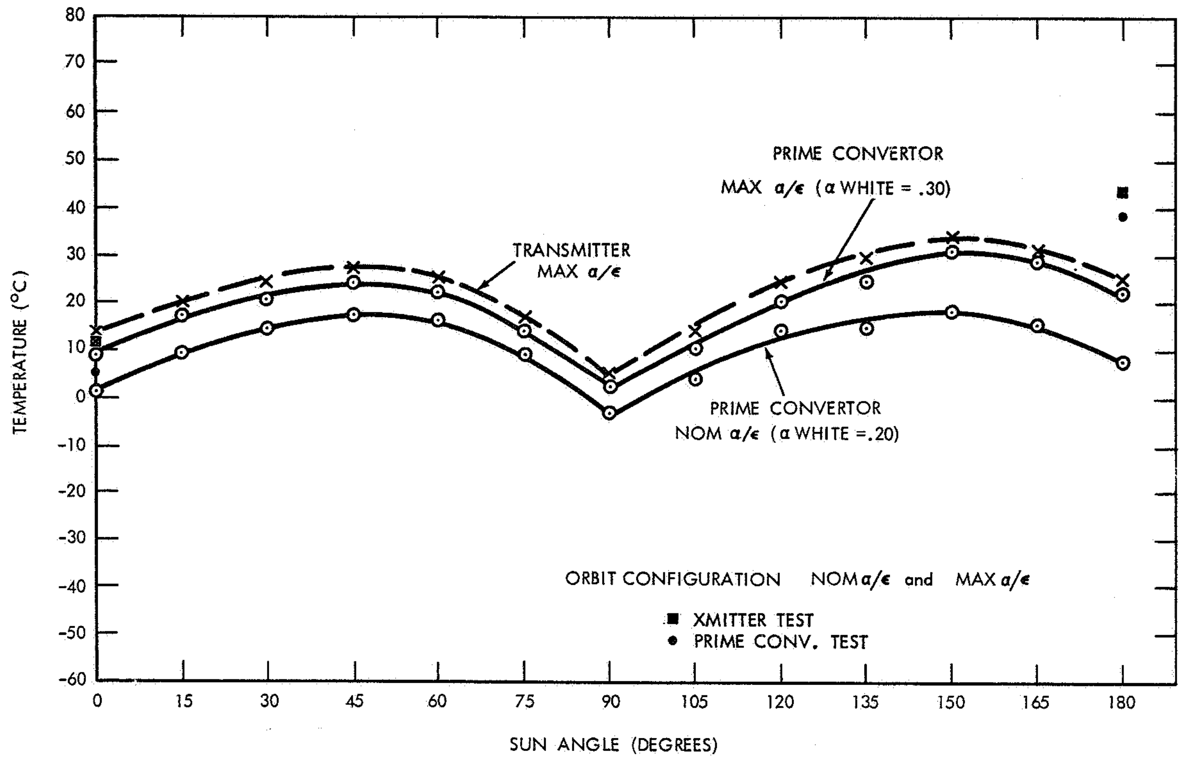


Figure 10. Engineering Test Unit Prime Converter and Transmitter Temperatures

AMES RESEARCH CENTER MAGNETOMETER

This experiment was thermally designed by the Honeywell Radiation Center of Boston, Massachusetts. Goddard's only responsibility was to coat the booms according to the experiment specifications of Honeywell. They stipulated during the program that the booms should be coated with evaporated aluminum. The canister was coated by them with 13.5% white paint and 86.5% evaporated aluminum.

UNIVERSITY OF CALIFORNIA

This experiment consisted roughly of a three-inch diameter gas-filled ball detector which was conductively coupled internally to its spacecraft card. As the experiment protruded above the top surface of the spacecraft cover a hemispheric bubble was provided in the top to enclose the experiment. In order to decouple the experiment from the top cover, which in the high solar aspects could get relatively warm, a polished surface was provided on the inside of the bubble. The experiment in turn had a low emissivity surface on the detector.

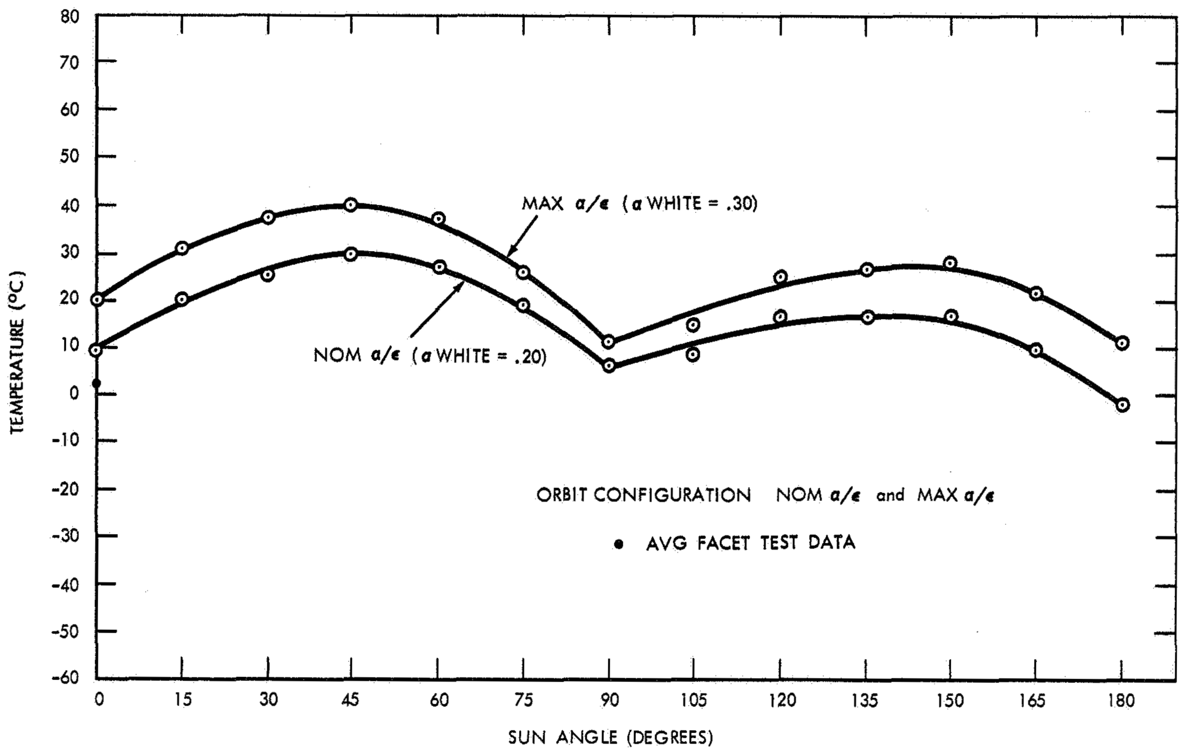


Figure 11. Engineering Test Unit Average Facet Temperature

TEMPLE UNIVERSITY

The experiment consisted of an eight-inch diameter glass plate with styrofoam backing. With the styrofoam acting as a good thermal isolator it was found necessary to coat the outside surface of the micrometeorite detector of this experiment with an α/ϵ equal to 1.0, to keep it below the specified 100° C level. The coating chosen was 6-1/4 wave lengths of aluminum oxide (α_s = .10 - .12, ϵ = .12 - .13).

SOLAR PADDLES

Using α/ϵ measurements from a typical solar paddle module and projected areas from photos, orbital temperatures were determined (see figure 13).

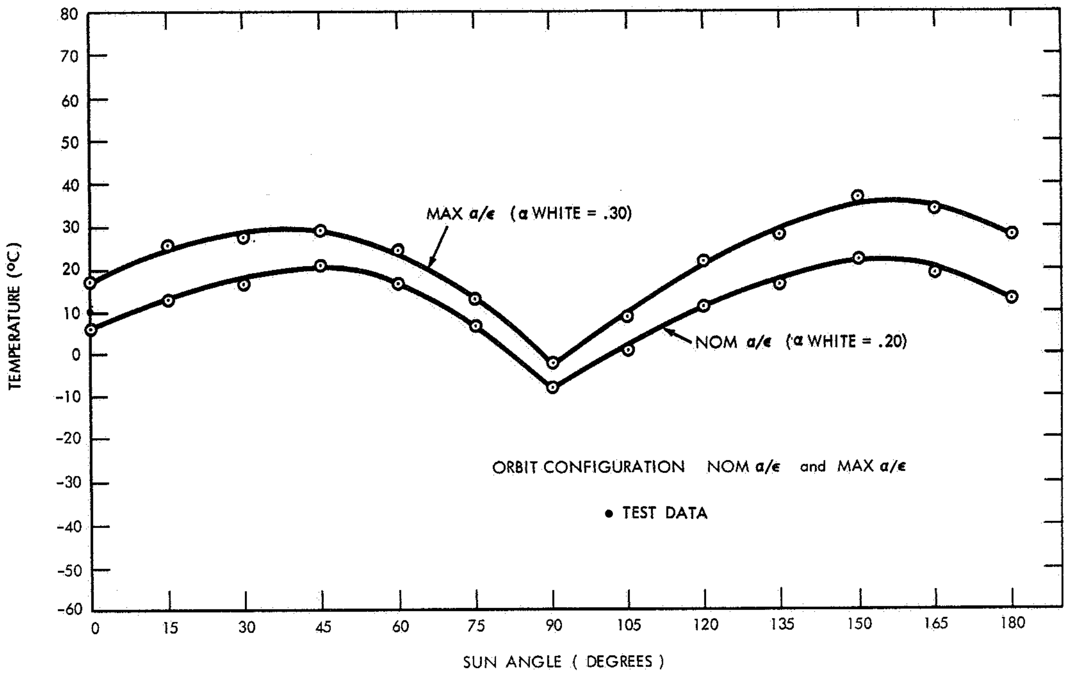
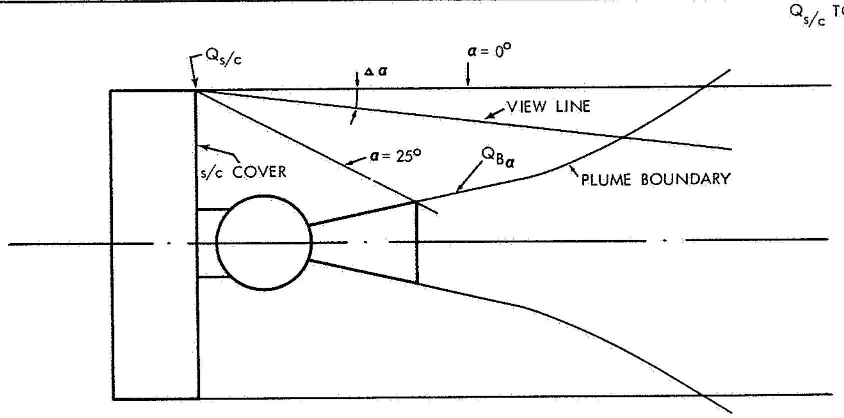


Figure 12. Engineering Test Unit Battery Temperature

Table I

SIZE / VIEW ANGLE $\Delta\alpha$	NUMBER OF PARTICLES x 10^8 /ft^2 1	2	3	4	5	τ OPTICAL THICKNESS	ϵ_a APPARENT EMISSIVITY	$\bar{T}$ AVERAGE TEMP.°R	$Q_{B\alpha}$ FLUX at PLUME BOUNDARY BTU/ft.sec.	$Q_{s/c}$ FLUX at s/c COVER BTU/ft.sec.
0 - 5	5.9	1.7	.5	0	0	.03	.01	3309	.565	.017
5 - 10	10.5	3.0	1.4	.7	.2	.11	.01	3610	.80	.018
10 - 15	17.1	4.9	2.2	.8	.5	.14	.03	3662	2.54	.048
15 - 20	28.7	7.4	3.5	1.4	.9	.23	.05	3763	4.8	.079
20 - 25	46.1	12.1	6.0	3.1	1.7	.41	.10	3720	9.1	.137

$Q_{s/c}$ TOTAL = .301
= 2.45 SOLAR CONSTANT



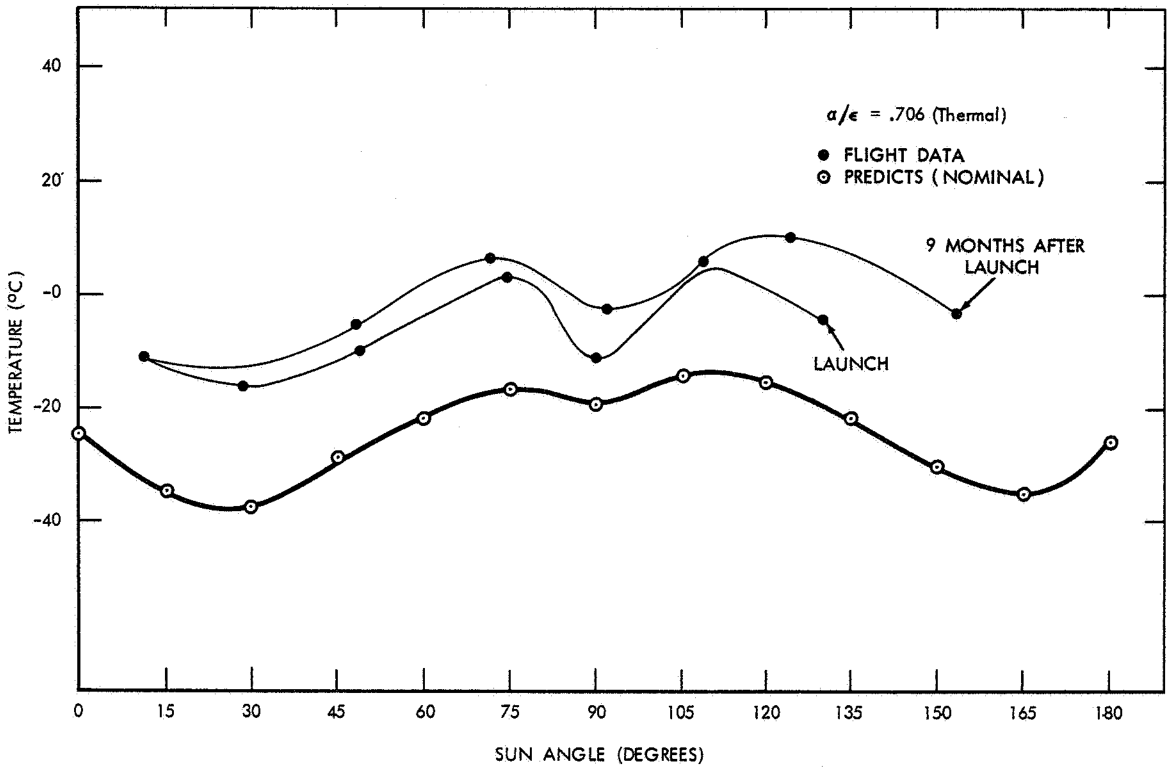


Figure 13. Paddle Temperature AIMP-D

SOLAR CELL DAMAGE EXPERIMENT

This experiment is located outboard on the side of the spacecraft cover. The maximum temperature level desired was 40° C. Because the solar absorptance of each band of cells could vary from .6 to 1.0, it was felt that coupling it strongly to the cover, which runs considerably cooler than 40° C, would counteract the large amount of absorbed solar energy. To this end, aluminum mounting screws were used and the experiment painted black on the side facing the spacecraft skin.

ASSOCIATED THERMAL TESTING

Prior to the final design of the AIMP-D spacecraft, various tests were performed to confirm the assumptions made in the thermal analysis. These tests were performed by members of the Thermal Systems Branch Lab and recorded in the references noted.

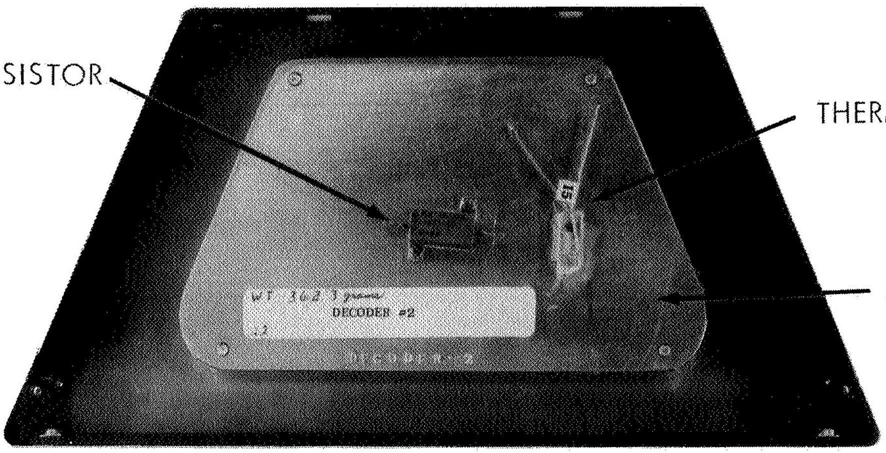


Figure 14. Typical Module ETU

SPACECRAFT THERMAL MODEL - ENGINEERING TEST UNIT (ETU)

A significant amount of data was obtained from a full-scale mock-up of the AIMP-D spacecraft. This model was built by the Mechanical Systems Branch for vibration, shock and acceleration tests. It was turned over to the Thermal Systems Branch for coating with thermal patterns, after heaters and thermistors had been inserted internally to simulate spacecraft components. Each experimentor provided an engineering model of his experiment which fitted its thermal design as closely as possible. The only "live" component in the spacecraft was the battery from which data was obtained on charging rates and general behavior during long shadow periods. Each card in the spacecraft had a 10 or 15 watt resistor mounted to a fiberboard plate. A thermistor was located just adjacent to the resistor. An aluminum plate was inserted in each card to mock-up its weight (see figure 14). The thermal coatings chosen for the top cover consisted of 12% Cat-A-Lac black paint, 4% Methyl Silicone + TiO_2 white paint and 84% buffed aluminum. The side of the spacecraft was coated with Cat-A-Lac black paint except for facets D and F which had 36% white and 64% black. The bottom shelf had 34.3% white paint, 4.6% black and 61.1% buffed aluminum with the bottom support cone 12% white and 88% evaporated aluminum. An Aclar-heat shield with the IR71-E coating was applied to the top surface of the spacecraft and the inert loaded rocket engine with ten layers of aluminized crinkled mylar was placed atop of the spacecraft. The spacecraft was placed in the Thermal Systems Branch 10' x 15' chamber and exposed to solar inputs at the 0 and 180° aspect. Because there was no spin capability in this chamber, the spacecraft was suspended by wires from a rail attached to the chamber door. Following the 180° aspect steady state condition, the spacecraft was allowed to go through a 6.8 hour long shadow.

Temperatures within the spacecraft and on the skins were monitored during all the testing. To simulate the orbit configuration, the rocket engine and plume shield were removed from the top of the spacecraft and again re-illuminated at 0° aspect. In order to establish a reference for the thermal blanket design, a bare rocket engine, inertly loaded and with only a polished outer case, was suspended by wires in the cold section of the chamber and allowed to cool down for approximately 36 hours with its temperature monitored. During this period the temperature fell to - 38° F. It was concluded that the blanket was indeed required. The most significant thing learned during this test was that when the spacecraft was illuminated with solar inputs at the 180° aspect, temperatures both on the skin and internally were 10 to 20° C higher than predicted. However, when the spacecraft was turned over with the top looking at the sun, and presented a primarily black and buffed surface to the sun, temperatures were within a few degrees of predictions. At first it was suspected that reflections from the bottom cone were adding additional inputs to the bottom shelf. To evaluate this the bottom cone was removed and the spacecraft was re-illuminated. Temperatures did not significantly fall over the steady-state condition previously tested with the cone. It was concluded that the cone was not the problem.

In order to study further the behavior of the spacecraft when illuminated on the bottom, various values of absorptivity for white paint were put into the analytical model. It was found that a value of absorptivity for white paint between .40 and .45 gave very good results. These values determined in the tests were more than a factor of 2 higher than measured in the lab using reflectance techniques. White samples exposed with the spacecraft and checked after the test only indicated a degradation of a few percent. (α_s = .18 before test to .22 after test.) The fact that we had closely matched predictions for the zero degree aspect gave credibility to the analytical model and turned suspicion towards the white paint. Measurements made at the Lockheed Aircraft Corporation (reference 10) on their "Thermotrol" (methyl silicon and titanium oxide) white paint indicated that these paints when measured in a vacuum under solar environment degraded quite drastically, i.e., α_s = .38, after only 50 sun hours of exposure, however, recovered when exposed to atmosphere for a short period of time (α_s = .2). This supports the data which was gathered during the test when the spacecraft showed abnormally high absorptances for white paint.

During the long shadow condition the spacecraft temperatures depressed to the predicted values between -45 to -50° C. This showed that the over-all emittance and thermal mass used in the calculations was correct. Large gradients measured between the cover and the bottom shelf indicated that in this model conduction was practically zero between these two points. Conversely, the gradient was very small between hot components such as the prime converter and transmitter, and the bottom shelf indicating that conduction was very good between these points. The effective absorption and emittance calculations from solar energies passing through the heat shield were confirmed when the spacecraft was illuminated at the zero degree aspect with the heat shield in place on the spacecraft. It was also learned that applying a single resistor to a non-conductive plate was not the best way to simulate card power dissipation. This was borne out by the gradient that was found between the area near the highest power dissipator, the prime converter, and its adjacent frame. A gradient of the order of 10° to 15° C was found. This was not the case in the actual spacecraft prototype test. A more adequate method would have been to wrap wire totally around the board and have the card power dissipated over a much larger area.

SOLAR CELL MODULE EXPERIMENT

A 3" x 3" sample of a solar cell composite typical of the solar paddle was developed by the Space Power Section. It consisted of actual cells mounted on both sides of a honeycomb substrate taken from an average section of the paddle. This module was suspended in a Thermal Systems Branch carbon arc facility and an α/ϵ measurement was made. The α/ϵ measured was .8. This proved to be 20% less than that which was measured on other IMP paddles. Thes measurements were confirmed by IMP C flight data which used a paddle similar to the AIMP-D. A conductance value of 5.85 BTU/HR FT^{2}° F was also determined from this test.

EVALUATION OF THE EFFECT OF DECONTAMINATION ON THERMAL COATINGS

As AIMP-D is a lunar mission, NASA Headquarters stipulated that the surfaces of the spacecraft must be decontaminated. In order to study the effects of these decontaminants on thermal coatings, various samples were prepared of spacecraft thermal coatings and decontaminated with the following materials: (1) 88% mixture of Ethylene oxide plus 12% freon and (2) 99% pure isoproproponal alcohol. After decontaminating with these materials, the samples were exposed to 300 hours of solar radiation. White paint samples that had been exposed to ethylene oxide and freon degraded quite drastically (α_2 = .484; however, those exposed to alcohol showed very little degradation. It was therefore concluded that the spacecraft could not be exposed to high concentrations of ethylene oxide gas and freon, but cleaned with alcohol.

BOLT CUTTERS

Bolt cutters used to separate the fourth stage were exposed to a cold environment for 72 hours. Final temperatures were recorded and used as a qualification temperature for the bolt cutters used on the spacecraft (see reference 5).

THERMAL BLANKET EVALUATION

An inert rocket engine was covered with 10 layers of aluminized crinkled mylar placed on a plate which simulated spacecraft interface temperatures and exposed to a cold wall and vacuum for 72 hours. The mylar was used instead of Kapton due to a procurement delay in getting this material (its behavior as a superinsulation is closely equivalent). Temperatures monitored at various points in the rocket engine propellant indicated that the temperature was kept above 0° F. Gradients within the propellant were 2 to 3° F (see reference 6). The over-all effective emittance of the rocket including the nozzle was calculated as .026 from this cool down data.

HONEYCOMB EFFECTIVE CONDUCTANCE

A 1-1/2" x 1-1/2" x 1/4" sample of honeycomb was prepared painted with a white high emittance coating on the front side and super insulation and a heater on the backside. This sample was placed in a chamber, given a fixed amount of heat input to the backside and allowed to radiate to a cold wall from its white surface. The temperature gradient across the honeycomb was measured at steady state and conductance values calculated. Conductance calculated was 24.2 BTU/HR $FT^{2\circ}$ F. In a second test another sample, which had conductive epoxy within the cells of the honeycomb, was evaluated. Conductance of this material proved to be lower than the unfilled honeycomb material, namely, 17.6 BTU/HR $FT^{2\circ}$ F. (see reference 7). This result showed that filling the cells with a relatively poor conductive material did not compensate for the loss of radiative path.

FAIRING CONTAMINATION

Decontamination and cleaning procedures of the Delta fairing were checked to determine if any residue left by the cleaning material would outgas during f fairing heating onto the spacecraft. A typical sample of the fairing was obtained from Douglas Corporation and cleaned with PCA 113 Freon cleaner. It was then brought up to 480° F in a vacuum chamber in approximately 5 minutes. Aluminized slides below the fairing sample were checked for any contamination. The results of reflectance measurements showed the slides did not have any appreciable residue upon them (see reference 8).

EVALUATION OF HIGH TEMPERATURE RESISTANCE THERMOMETER AND FILM H EXPOSED TO HIGH TEMPERATURE

After fourth stage burnout, the rocket case temperatures can reach as high as 660° F. In order to determine the effect upon a resistance thermometer located on the case and the super insulated blanket at such high temperatures, a thermometer was prepared on a hot plate backed by 10 layers of Kapton and raised sufficiently fast to these high temperatures. The thermometer showed good response and maintained its integrity all the way up to 600° F with the bonding only failing when the temperatures reached of the order of 1,000° F. The Kapton only charred on the bottom-most layer but stayed intact throughout the test. The Kapton tape, used to bond sections of the blanket, gave good adhesion all the way up to 700° F; however, lost its adhesive properties beyond that temperature. As the temperatures expected will not exceed 600° F to 700° F the materials were found to be quite adequate (see reference 9).

PHASE III - FINAL DESIGN

REVISIONS TO ANALYTICAL MODEL

After the ETU test showed that the gradients in the shelf and cover were less than predicted by the analytical model, it was decided to see if the difference was produced by the coarseness of the nodal breakdown. Accordingly, an 89 nodal layout was prepared which broke the shelf and cover into finer nodes. Re-running the orbital conditions on this program showed very little effect on the internal components of the spacecraft. However, it did effectively reduce the shelf and cover gradients.

USE OF GE OPTIMIZATION PROGRAM TO HELP ESTABLISH THERMAL COATINGS

During the final design effort, a computer program, which determines the optimum external thermal control coatings for the desired spacecraft internal temperatures, became available. This program was developed by GE for NASA. It consisted of an analytical model similar to the one used in the thermal analysis. A reduction from the original 45 nodal layout to 34 nodes was required. Solar fluxes with a 100% sun condition in the orbital configuration were applied to the program. The program was asked to optimize thermal coatings using an upper limit of 35° C for the battery as the basic design criteria. Maximum absorptivity values for white paint were used in the program ($\alpha_s = .4$), as brought out by the ETU thermal testing and the Lockheed data, and a maximum value for the buffed aluminum surfaces (a_s = .22). The program satisfactorily gave values for effective absorption and emittance to be applied to each external node of the spacecraft. With this exceptionally high value of absorptivity for white paint, the program added roughly 25% more white area to the bottom shelf of the spacecraft than had previously been used on an ETU model. With such large percentages of white paint on the bottom shelf the battery temperature at the 90° aspect was depressed roughly 5° C below previous predictions. This was not felt to be prohibitive with respect to the battery. Applying minimum values of absorptivity and emissivity to the various spacecraft coatings gave a spread of the order to 12 to 20° C to spacecraft temperatures dependent upon solar aspect angle, (see figures 15, 16 and 17. Using the minimum coating values, an analysis was made

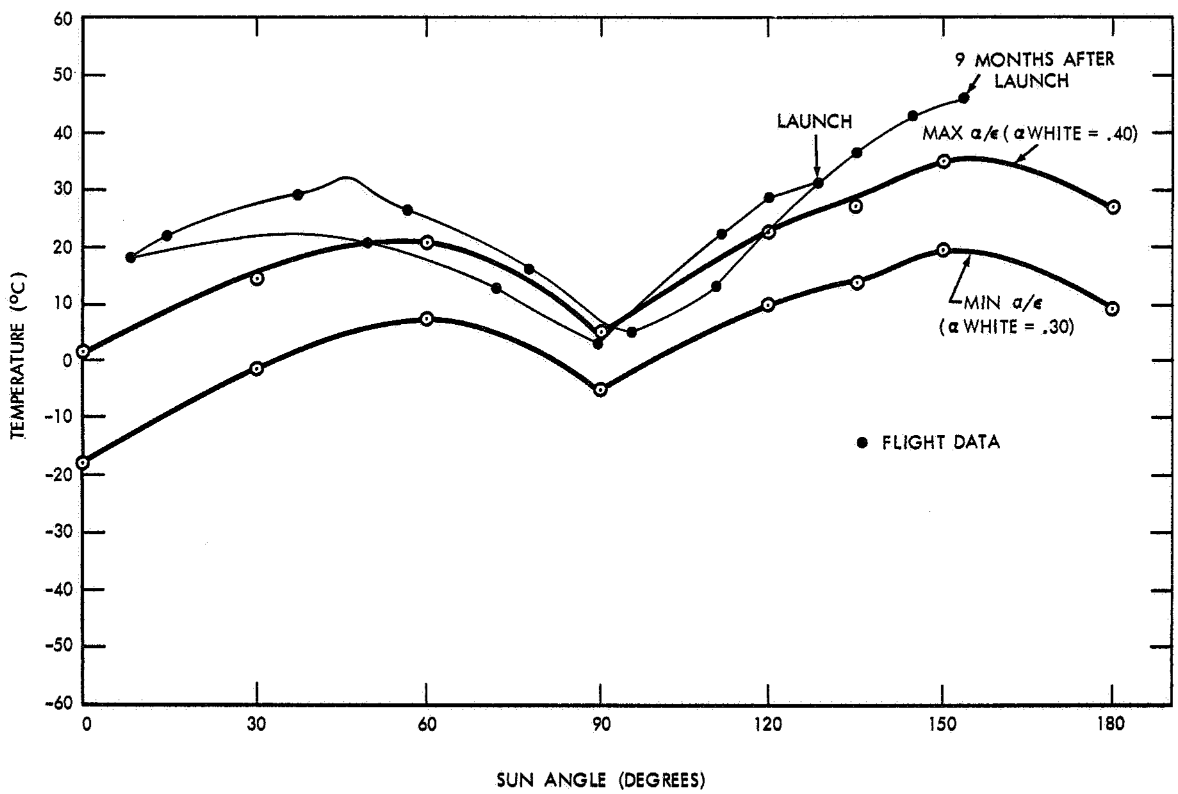


Figure 15. Flight Spacecraft Transmitter Temperature

of the spacecraft temperatures during the launch and coast phases. The most responsive spacecraft components are shown in Figure 18. Temperatures shown in sketch are those maximums on the fairing asked to be held by the Douglas Company.

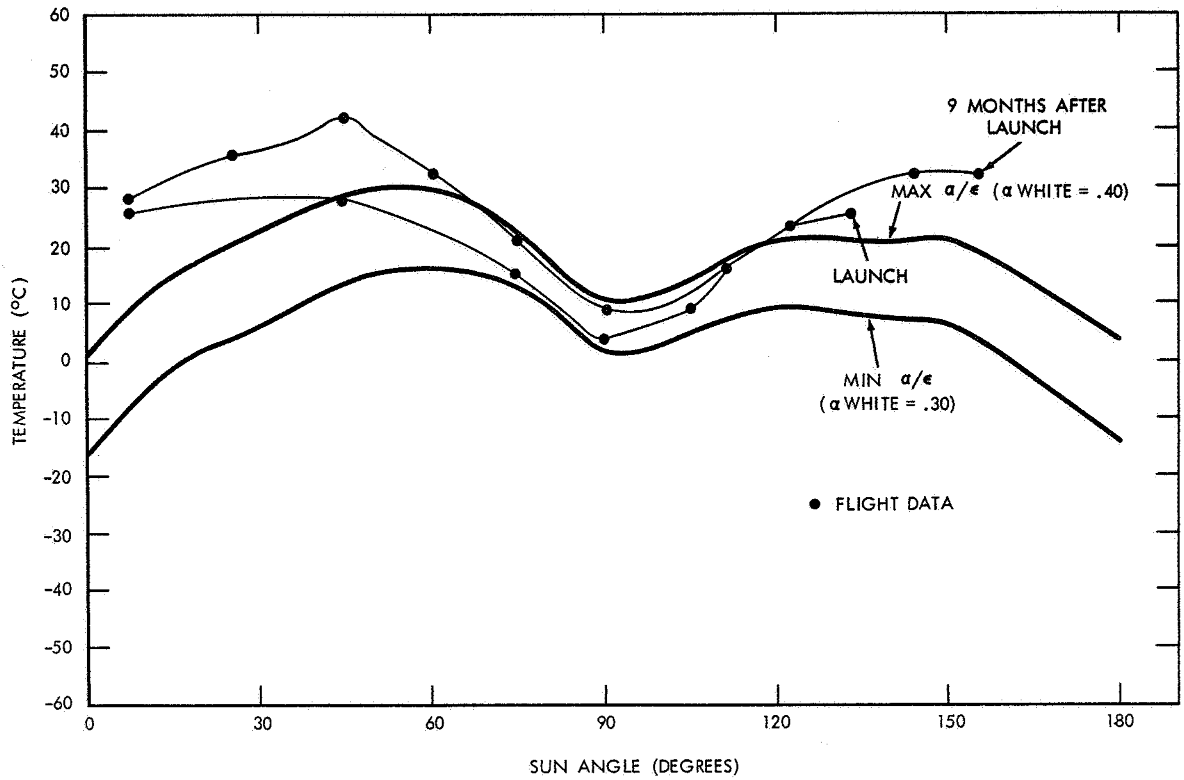


Figure 16. Flight Spacecraft Average Facet Temperature

The final coating pattern developed for the AIMP-D using this computer program and rounded off for practical purposes, was of the following design (see figure 19):

TOP COVER

All facets with the exception of facets B and F had 10% white paint and 90% buffed surfaces. Facet B had 17% white paint and 83% buffed surfaces. While facet F had 23% white paint and 77% buffed surfaces. All facets on the side of the spacecraft were painted black with the exception of facet F which had 12% white paint and 88% black.

BOTTOM SHELF

All facets with the exception of facets B and F had 27% white paint and 73% buffed aluminum. The areas under facets B and F had 95% white paint and 5% buffed aluminum.

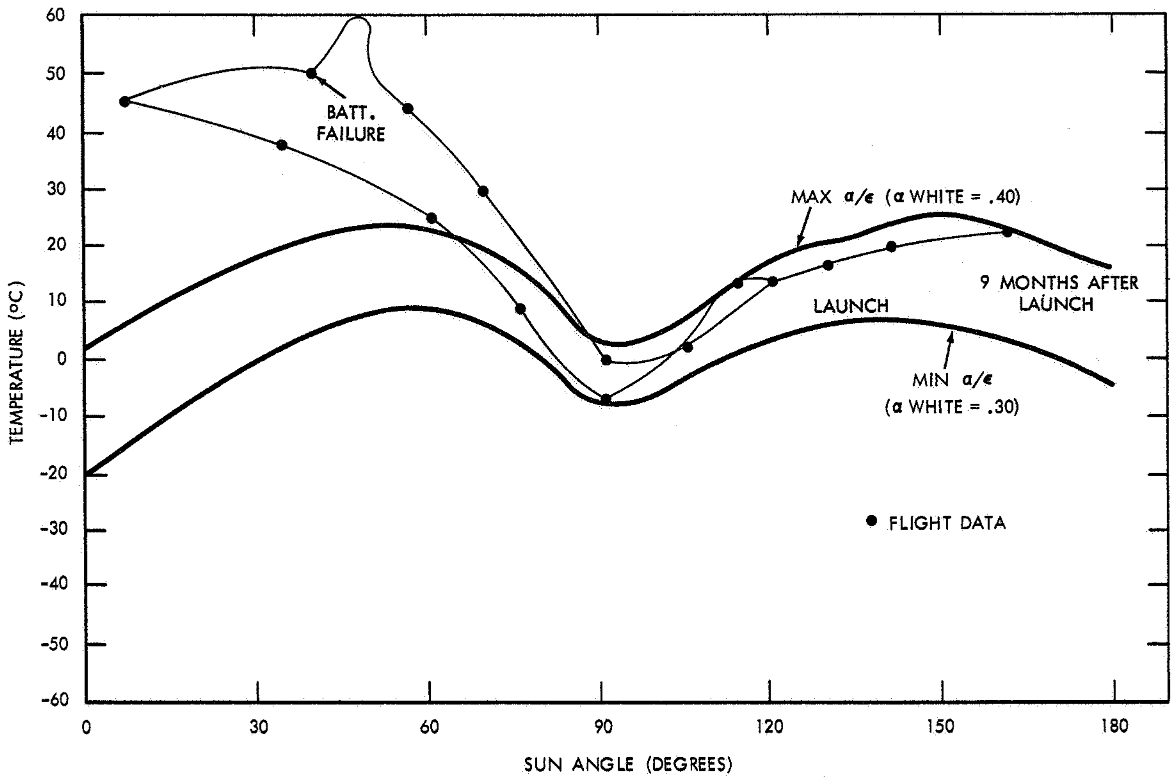


Figure 17. Flight Spacecraft
Battery Temperature

CONE

The support cone under the spacecraft had 45% white paint and 55% evaporated aluminum.

TOP SPRING SEAT

The top spring seat had 55% white paint and 45% evaporated aluminum.

BOTTOM SPRING SEAT

The bottom spring seat had 15% white paint and 85% evaporated aluminum.

TOP CENTER TUBE

The top center tube had 33% evaporated aluminum and 67% black paint.

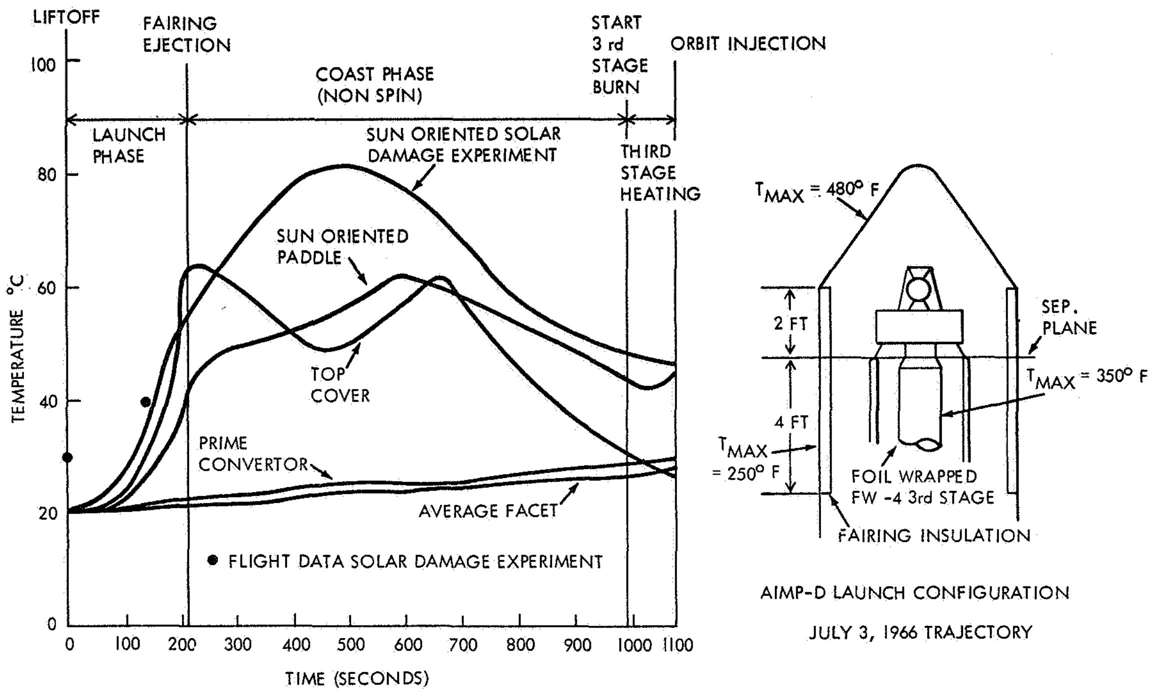


Figure 18. Launch and Coast Phase Temperature

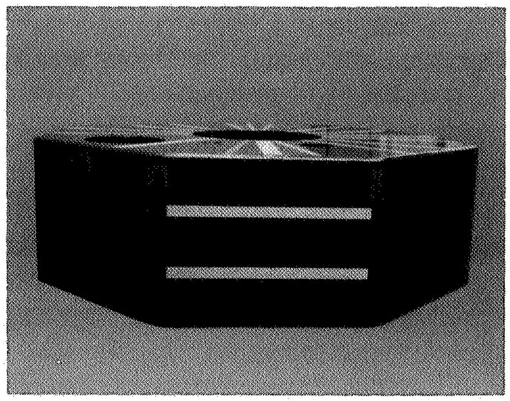
BOTTOM CENTER TUBE

The bottom center tube had 10% white paint and 90% buffed aluminum. The total percentages of coatings weighted by area for the entire spacecraft were 21% white paint, 35% black paint, 38% buffed aluminum and 6% evaporated aluminum.

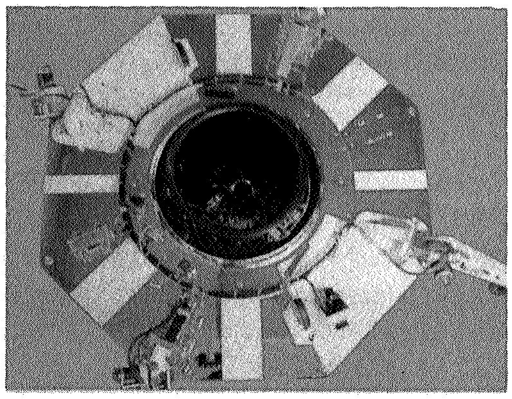
The over-all effective emittance of the total spacecraft was calculated to be .461 using maximum tolerance on all thermal coatings.

OTHER SPACECRAFT DESIGN CHANGES

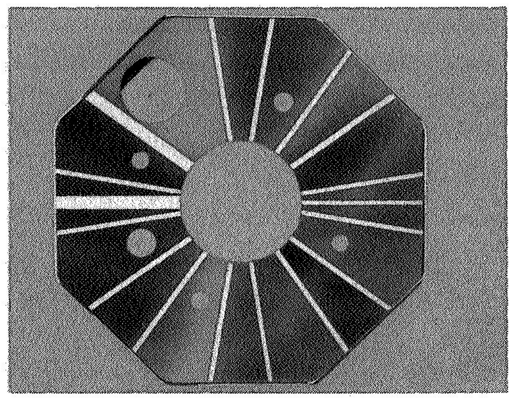
The plume heating study by TRW, Inc., together with measurements made at the AEDC during an actual rocket firing, showed the heat fluxes from the plume to the spacecraft cover to be of the order of 2.5 solar constants. The thermal mass of the spacecraft cover was sufficient to absorb this heat for the 20 to 25 second burn time. Consequently, the plume shield which had been



SIDE COVER



BOTTOM SHELF



TOP COVER

Figure 19. A-IMP-D External Thermal Coating Patterns

developed for the spacecraft was found not to be required and was removed from the design. The ETU testing on the spacecraft showed that the bottom and top spring seats got relatively warm during the 0° and 180° solar aspects. In order to safeguard the battery from these exposures, the inside surfaces of the bottom and top spring seats were changed from black paint to polished surfaces.

PHASE IV - TESTING

PROTOTYPE SPACECRAFT

A full systems test was performed in the T&E 7' x 8' solar simulation chamber on a prototype AIMP-D spacecraft. The spacecraft was supported from the gimbal at the fourth stage interface and illuminated at the 90°, 150°, and the

180° solar aspects. The first test point checked was the 150° solar aspect. The average intensity during this test to the spacecraft was roughly .87 solar constants. This was due to the variation in depth of the carbon arc solar simulator. Data taken on various points of the spacecraft, both internally and externally, very closely matched predictions for this aspect at the reduced solar intensity. The next test performed on the spacecraft was the 90° solar aspect under the 2.4 hours shade and 10.7 hours sun condition. Three orbits were run and temperatured monitored internally and externally to the spacecraft. This test point again closely matched predictions using an α_s = .363 for white paint spectrally matched for the carbon arc. The spacecraft was then elevated to a 180° solar aspect and allowed to stabilize. The temperatures did not approach predictions either internally or externally due to the assumption of zero conductance at the shelf/cover interface. When a conductance value was applied at this point, the temperatures match very well.

The temperature of the battery was then raised until it reached its minimum level which would be expected at the 150° solar aspect. The reason for this was that gimbal failure prevented achievement of a steady state condition at the 150° solar apsect before entering the shadow. The solar simulator was then turned off for a period of 6.8 hours simulating the longest shadow anticipated following a 150° solar aspect angle steady state condition. Spacecraft temperatures fell to near -60° C internally at the end of the 6.8 hour period. This was 15° C cooler than was determined in the Thermal Systems Branch test of the ETU model. These colder temperatures were attributed to the amount of additional white paint that had to be added to the prototype spacecraft. This data point also matched the predicted values very well.

ROCKET HEATING AND SOAKBACK TEST AT AEDC

The ETU model of the AIMP-D spacecraft with a fourth stage rocket engine attached to it was sent to AEDC for actual rocket firing. The spacecraft with the fourth stage rocket engine was attached to a spin rig in a high altitude chamber simulating 150,000 feet altitude. Thermistors were placed in critical internal components such as the battery, prime converter, top cover, center tube and Iowa experiment. Flight type thermistors were placed on the fourth stage rocket engine used to monitor its temperatures during and after firing. Heat rate sensors in the form of pyroheliometers, calorimeters and a radiometer were placed on the spacecraft and in the direct view of the rocket exhaust plume. Two pyroheliometers were placed on the outermost portion of the top cover viewing into the rocket exhaust. These measured the heat flux from the rocket plume

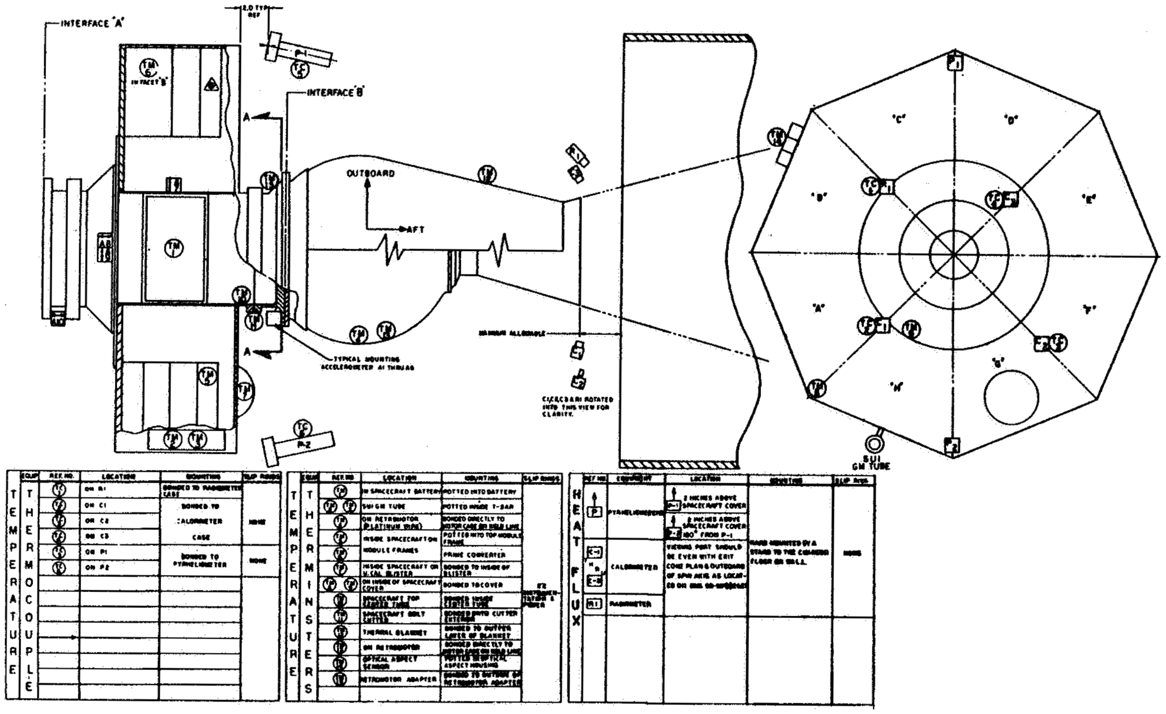
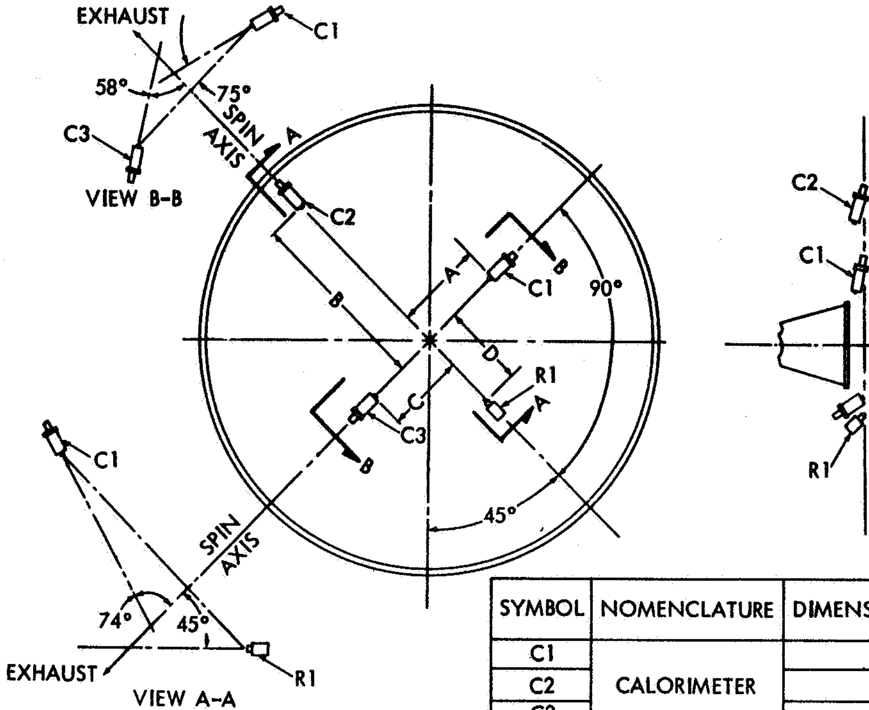


Figure 20. Spacecraft Instrumentation
for AEDC Firing

which directly impinged on the top cover. One pyroheliometer (P-2) had an IR71-E filter in it and the other was bare (P-1). See Figures 20, and 21.

The reason for the filter was to evaluate just how much energy from the plume would be rejected by the IR71-E filter. It would also give an indication of just what the composite nature of the spectrum of the plume was. Two cones were fitted over the pyroheliometers to restrict their view to the plume alone exclusive of the nozzle, thermal blanket and chamber. Calorimeters were set very close to the exhaust plane at the rocket exit plane to determine heat fluxes as close to the plume as possible. This data would be used to correlate to the TRW, Inc., computer program that was run. Fluxes measured during the firing were of the same order of magnitude as was predicted from the computer program. See Table II. Correction factors had to be applied to each flux value depending upon the view the sensor had of the actual plume. The reading taken back at the spacecraft cover was lower than an equivalent element on the spacecraft cover would actually see. This was caused by the restrictive cone that was applied to the pyroheliometer. It not only restricted the view the pyroheliometer had of the extraneous environment, but inadvertently restricted the view it had of the exhaust plume. Through the use of the CONFAC II program, the proper view factors were determined for



SYMBOL	NOMENCLATURE	DIMENSIONS, IN.	FIRING 01	FIRING 02
C1	CALORIMETER	A	6.66	4.160
C2	CALORIMETER	B	11.91	11.91
C3	CALORIMETER	C	5.54	5.54
R1	RADIOMETER	D	5.66	4.29

Figure 21. Heat Flux Sensor Locations
for the TE-M-458 Motor Firing

the heat sensor. Temperature measurements made during and after rocket firing showed that the spacecraft responded very little to soakback heating. All internal components as well as external surfaces responded just a few degrees above test cell temperatures during the two hour period following rocket firing (see figure 22). The only spacecraft member that went up in temperature was the fourth stage fiberlgass rocket mount. It reached a maximum of 190° F. This was not felt to be prohibitive. The resistance thermometer on the rocket case showed a temperature response similar to thermocouples placed at similar points on the rocket engine.

Post firing inspection after the test showed a large deposition of material on the top cover adjacent to the super-insulated thermal blanket. This material turned out to be boiled off evaporated aluminum and lacquer from the nozzle which found its way down the thermal blanket to the spacecraft surfaces through venting ports. The evaporated aluminum was placed on the rocket nozzle to reduce the heat leak to space during the transfer trajectory. Subsequently the blanket design was modified to allow venting to take place at the top of the blanket rather than at the bottom which faced the spacecraft.

Table II

Heat Sensor Data - AEDC Tests

Test	Sensor	Heat Flux, Q (Actual)*	Heat Flux, Q (Calculated)*	Test Instrument Temperature (° F)
Number 1	P-1	0.0372	0.015	76
	P-2	0.0076	0.0025	76
	R-1	1.69	1.26	96
	C-1	1.56	1.01	112
	C-2	0.624	0.99	104
	C-3	1.69	1.29	132
Number 2	P-1	0.025	0.015	78
	P-2	0.0083	0.0025	72
	R-1	1.44	1.56	74
	C-1	1.46	1.62	96
	C-2	0.619	0.99	120
	C-3	1.27	1.29	132

*Units are Btu/ft^2-sec.

MAGNETOMETER TESTS

The GSFC magnetometer experiment was first tested in the T&E 7' x 8' chamber using the carbon arc solar simulator. A fixture was prepared which simulated the spacecraft shadowing. See Figure 23. A short boom was attached to the fixture and the Goddard package fixed to it. The magnetometer was taken through various sun and shade conditions. The first data point was the 30° solar aspect, 100% sunlight condition. During this test the flipper came to within a few degrees of predicted values and successfully flipped the magnetometer when heat was applied. Following this test, the magnetometer was held at the 90° solar aspect in 100% sunlight until stabilization. The temperatures at this aspect fell to -17° C and the flipper refused to flip the magnetometer. This was also the case during the 82% sun condition (2.4 hour shade/10.7 hours sun) when the temperature dropped to -30° C. The post test inspection of the magnetometer flipper showed a

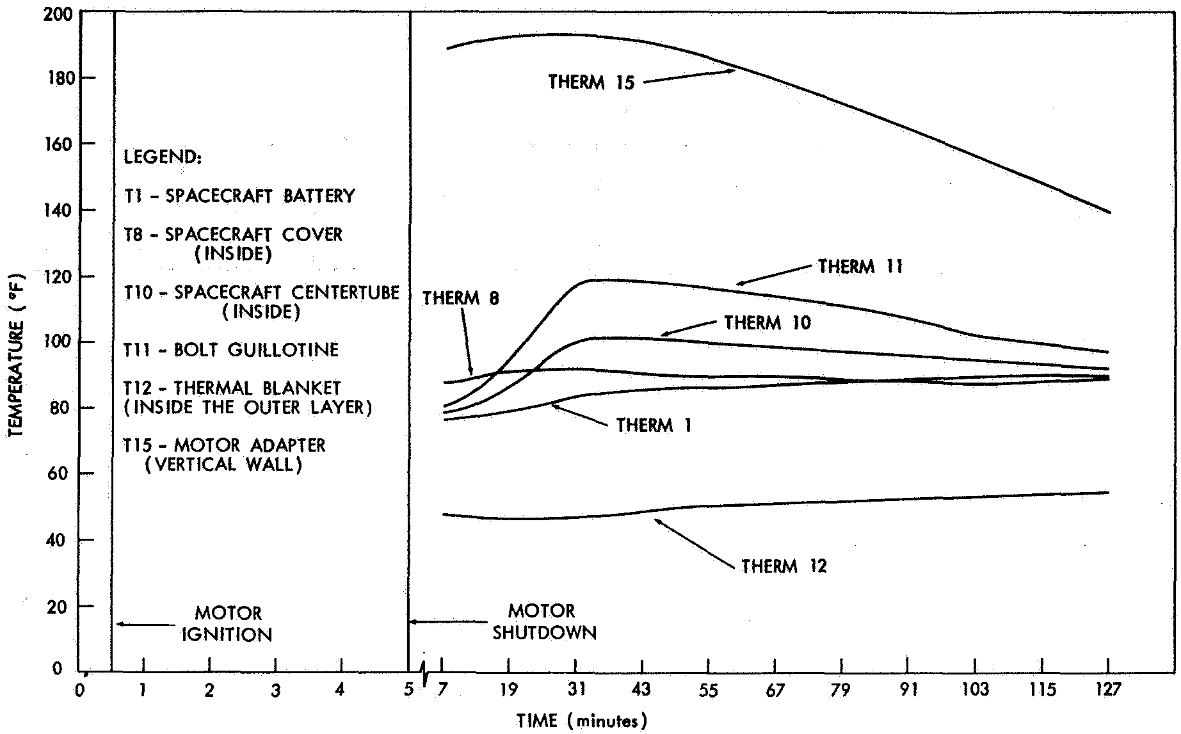


Figure 22. Spacecraft Temperature Response
to TE-M-458 Rocket at AEDC

bushing had come loose at these low temperatures and jammed the flipper mechanism. A modification was made to this bushing and the flipper magnetometer was reassembled. To confirm the design, the magnetometer was retested in the Thermal Systems Branch solar simulator chamber. The first data point achieved was a 90° solar aspect steady state condition. During this condition the flipper did work. This data point was followed by a 2.4 hour shadow which caused the temperature to fall to -18° C. Another successful flip was then achieved. A simulated 6.8 hours shadow depressed the temperature in the magnetometer to -50° C. This proved to be beyond the capability of the flipper and no actuation took place. Subsequent attempts to actuate the flipper as the internal mechanism heated up were however successful. This did not constitute a failure mode as it was not a requirement to actuate the flipper after initial emergence from such a deep shadow.

FLIGHT DATA

On July 1, 1966, the AIMP-D was launched at Cape Kennedy. Within the first few hours after launch it was found that attaining a lunar orbit would not be possible and it was decided to try for an alternate mission around the earth. To

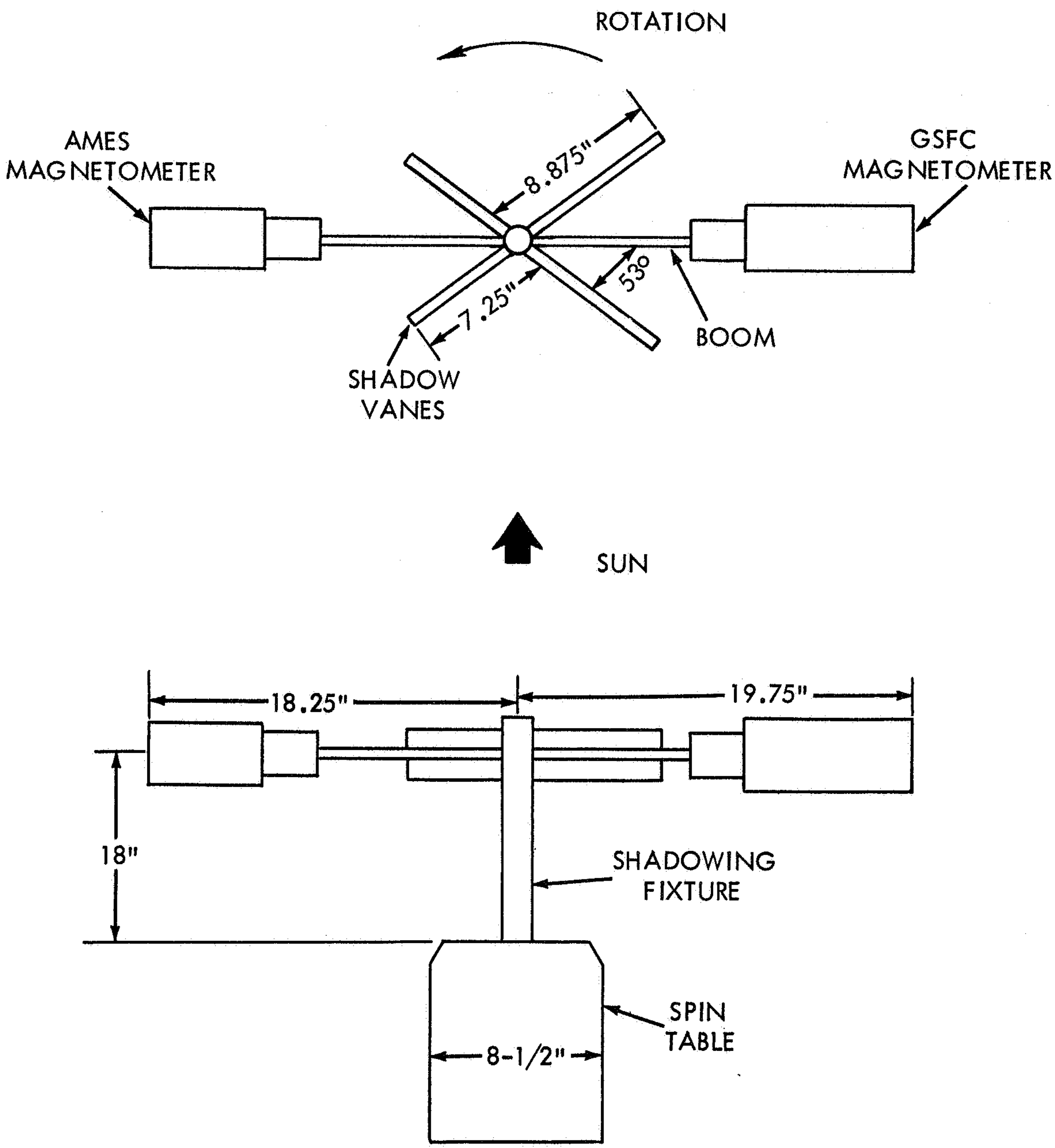


Figure 23. Magnetometer Test Future

achieve this the retro-rocket was fired six and one-half hours after lift-off. The final orbit parameters were:

Apogee	445,854 KM
Perigee	28,532 KM

Inclination	29.04°	Eccentricity	.8846

A six-month life with no significant shadows occurring for the first 3 months was expected.

The data received by telemetry during the launch and coast phase was limited to the period from liftoff to just prior to fairing ejection. The only component that showed any response during this time was the solar damage experiment which rose 10° C (see Figure 18). The entire coast phase was not recorded due to loss of tracking at the Cape, after the spacecraft went over the horizon, and inadequate pickup of the spacecraft by down-range tracking stations.

After retro-fire the rocket was kept with the spacecraft for two hours. During this period the soakback to the internal components was of the order of 5° C. Rocket case temperatures rose to a maximum of 670° F as expected.

The spacecraft achieved steady state conditions about 24 hours after liftoff at a solar aspect of 129°. Shelf-mounted components, such as the transmitter, showed a 5° C higher than predicted value (see figure 15) at maximum α_s. This was also true, to a lesser degree, of the average experiment mounted on the equipment shelf (see figure 16). The battery, which was a good indication of spacecraft mean temperature, ran between the maximum and minimum α / ϵ curves (see figure 17). Experiments mounted on the top of a stack, such as the University of California, also fell within the maximum and minimum α / ϵ band. It seemed from this data that components such as the transmitter and prime converter, which had practically 100% white paint under them, were more sensitive to degradation than components with less white paint on internally mounted components. This effect is reduced as the solar aspect changes and the sun moves up towards the sides of the spacecraft. This can be seen by the curves where temperatures begin to fall into the band as the sun moves to the broadside position. It is white paint exhibited a degradation to at least $\alpha_s = .40$ or better very similar to that seen in test. The largest deviation from predictions came in the solar paddle temperatures which ran 15°-20° C higher than prediction but well within acceptable performance range. There were several factors which could have caused this. During the latter days of spacecraft integration the paddles had to to be reworked and 50% of the cells replaced. Past data indicates that solar absorptivity varies from one cell manufacturer to another by as much as $\Delta \alpha = .05$. There were also anomalies in the data reduction programs which processed paddle flight data from various tracking stations. All of these factors add to the uncertainties.

Outboard experiments such as the solar cell damage experiments and the GSFC magnetometer ran within 5° C of predictions. The thermal actuator in the magnetometer worked very well and successfully flipped several times.

As the mission progressed (see figures 15, 16, 17) and the sun began to rise on the top cover it became apparent that all temperatures began to divert from their predictions. It was obvious that a change to the thermal properties of the top cover had taken place. White paint degradation was first suspected, however, the quantity of paint on the top was not sufficient to cause such behavior. Other factors such as micrometeorite damage or rocket plume contamination were then investigated. Using the analytical model it was found that a doubling of the absorptance and emittance on the cover would best fit the data. Both phenomena could have accounted for this as ground tests on solid particle bombardment and plume contamination of buffed aluminum showed similar changes in properties. The follow on flight, the AIMP-E mission, actually measured a change in reflectance on a buffed aluminum sample mounted on a special contamination experiment. (Ref. 11). Ground calibrations of this experiment indicated that the reflectance changed by at least a factor of two in the solar region. It was of interest to note that this change occurred 3 minutes after motor burnout indicating that the spacecraft passed through a cloud of contaminants rather than having it blown back during pocket firing. Fortunately this spacecraft had a multilayer blanket on its top cover to protect it from contamination and its thermal performance was not affected.

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