Thermal Design of the Orbiting Astronomical Observatory

Louis H. Hemmerdinger*
Grumman Aircraft Engineering Corporation, Bethpage, N. Y.

The design of the passive thermal control system for the 3600-lb NASA (GSFC) Grumman Orbiting Astronomical Observatory is presented. The spacecraft structure can be designed to operate at an average of either −22° or +32°F and maintain the required thermal stability of ±18°F irrespective of pointing angle during its one-year lifetime. A telescope thermally linked to the OAO central tube and insulated from the space environment will be maintained within 3°F of the spacecraft structure. The design utilizes Alzak skins as the thermal control surfaces. Electronic equipment is maintained between 0° and 130°F; it is insulated from the structure and radiates most of its heat to the skins and out to space. Spacecraft structural temperatures are achieved by controlling the heat transfer from the equipment to the structure and rejecting this heat through thermally stable skins. The analyses of the spacecraft thermal balance and electronic equipment design is presented along with the insulation configuration of the spacecraft. The analytic model is then correlated with data from the full-scale thermal vacuum test. A discussion of the test model and setup is presented.

Introduction

THE Orbiting Astronomical Observatory is designed to provide an accurately stabilized, unmanned platform for astronomical observations from above the earth's atmosphere.[1,2] When launched at the Atlantic Missile Range in 1965, it will be the free world's largest and most complex scientific satellite. The immediate aim is to observe the ultraviolet radiation of the heavens which is absorbed by the earth's atmosphere. Emphasis has been placed on designing the OAO for a long life and on maintaining a general purpose type of spacecraft that could be used for a variety of experiments with minimum redesign and retesting. Based upon the results of detailed analytical studies and a development test program, it appears that all temperature requirements of the observatory can be successfully met by passive thermal-control techniques. The passive system containing no moving parts with a minimum of heaters was chosen for maximum reliability. This paper presents the status of the OAO thermal design at the approximate midpoint of the development program.

Thermal Design

Description of Spacecraft

As shown in Fig. 1, the satellite is an octahedron, 80 in. across the flats and 118 in. long. Solar paddles extend perpendicularly from sides C and G at an angle of 33° to the central axis. A cylindrical hole the full length of the spacecraft and 48 in. in diameter accommodates the primary telescope system. Sunshades are provided at the optical openings to prevent sun impingement inside the telescope cavity when oriented to a star.

Internally, aluminum trusses and shelves are integrally connected to the 48-in.-diam structural tube and form 48 truncated bays, as seen in Fig. 2. These bays house the electronic equipment, which is mounted to hinged honeycomb panels. The panels swing outboard to permit access to the equipment and wiring while installed in the spacecraft.

The total weight of the OAO is 3600 lb. Of this, the experimenter is allowed 1000 lb; the structure weighs 800 lb, the thermal control system, 50 lb, and the remaining 1750 lb consists of electronic equipment. Ten-mil polished aluminum skins (Alcoa's Alzak) protect the equipment and are used as both a thermal damper to incident radiation and a heat radiator. The skins are thermally isolated from the structure by means of nylon insulators.

During slewing maneuvers the satellite can revolve about any of its three axes. When positioned to a star, it is roll-oriented such that the earth-sun line is always parallel to the C and G skins (see Fig. 1). This position allows maximum solar paddle output and provides the OAO with a set of stable temperature skins that are used as symmetrical radiators to cool the spacecraft structure. Only bays 1, 2, 5, and 6, on sides C and G, are used for this purpose, while the entire central torus (bays 3 and 4) contains electronic equipment.

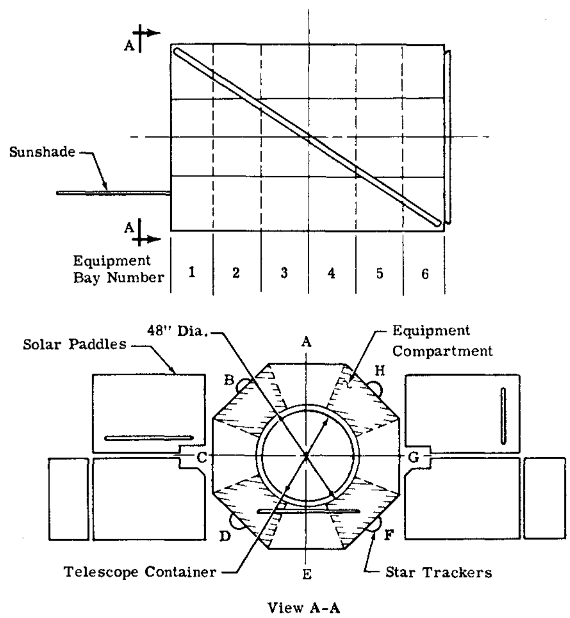

Fig. 1 Orbiting astronomical observatory.

Presented at the ASME 6th Annual Heat Transfer Conference, Boston, Mass., August 13, 1963; received by AIAA October 31, 1963; revision received May 15, 1964. The NASA Goddard Space Flight Center team effort, through the cooperation and assistance of C. Katz, L. Rabb, R. Stroup, and R. Ziemer has contributed greatly to the success of this design. Gratitude is also expressed to O. Sangesland, former OAO Thermal Group Leader, and A. Bartilucci, R. Boyland, R. Hembach, N. Kosowski, R. Sadler, A. Serper, D. Sullivan, and C. Walthers, who, by helping to perform the detailed design work and testing, have made this paper possible.

* Orbiting Astronomical Observatory, Thermal Group Leader. Member AIAA.

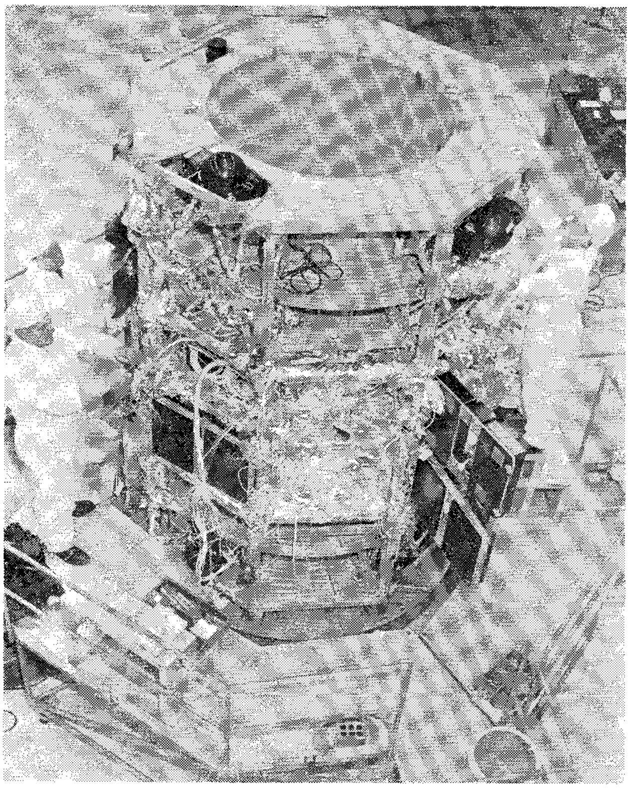

Fig. 2 OAO prototype (dummy equipment).

The telescope can be of any optical design which will fit within a 40-in.-diam container. The assembled container and telescope are then mounted within the OAO 48-in. tube on four aluminum lugs that lie in the same plane. Three of the lugs are shimmed as required for coarse adjustment of the optical axis with the spacecraft axis, and the fourth is torqued for fine adjustment. Alignment of the six star trackers, boresight star tracker, television camera, and three-axes inertia wheels are also referenced to the spacecraft axis. Three star trackers are required for orientation selection. Once acquired to a star, the boresight tracker and primary telescope are used to adjust the spacecraft finely to within $\frac{1}{10}$ sec of arc of the star under observation. Star trackers and boresight tracker have alignment error offset capabilities which are used during initial stabilization to correct for structural distortions because of vibration and change in structure temperature from launch to orbital environment.

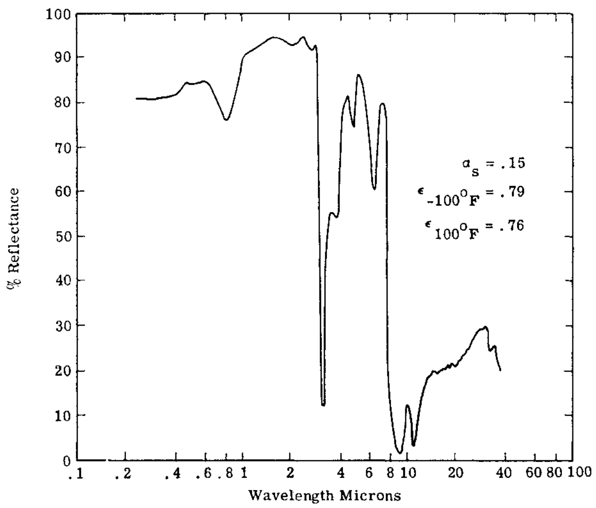

Fig. 3 Spectral reflectance of Alzak, grade S.I.

Environment

The OAO will be launched into a circular 500-mile orbit and have a period of 101 min. It is capable of being oriented to any star for a matter of minutes or days during an estimated lifetime of one year. During this time the spacecraft will be in sunlight from 65 to 80% of the time, corresponding to orbital planes of 0° and 55° to the ecliptic.

The solar constant is assumed to be 440 Btu/hr ft^2 with a yearly average earth albedo rate of 35% of the sun's energy. Earth radiation is taken as that from a −10°F blackbody source.[3] Transient flat-plate calculations have been made according to plate orientations orbiting the earth.[4] From these are derived average orbital heat fluxes owing to solar, albedo, and earth radiation. Since the OAO structure and equipment masses are large, transient heat fluxes are required only in determining temperatures of the skin, solar paddles, sunshade, and external fittings.

The sun spectrum is assumed to be a 10,200°F blackbody when integrating spectral reflectance measurements of materials to determine their absorptance to solar radiation. These same values of absorptance are used when calculating heat fluxes due to albedo radiation. Since the telescopes operate in the deep ultraviolet region of the spectrum, 1100 and 3000 Å, low outgassing materials must be used throughout the satellite to prevent contamination of the mirror surfaces.

Skins

Thermal control is dependent on the external skins. All skins are attached to the structure through pin-connected nylon fasteners, so that the wide fluctuation in skin temperatures (−200° to 40°F) does not create expansion problems or thermal leakage paths from the structure. In order to maintain the required equipment and structure temperature ranges and to reduce the wide fluctuations in thermal flux absorbed by the satellite because of orbiting and orientation changes, the skins must have a low absorptance α_s in the solar spectrum and high emittance ϵ between −200° and +100°F.[5–7]

Consultation with Alcoa resulted in the procurement of a commercial product called Alzak Lighting sheet, class S.I. The Alzak is composed of 3003 base metal and 99.9% aluminum cladding which is electrolytically brightened and coated with a light clear anodize. Using this material as a solar reflector, development tests were run for α_s/ϵ and for uv effects on the material. Measurements indicate a solar absorptance of approximately 0.15 and total normal emittance of 0.77 (see Fig. 3).

Testing of the uv degradation phenomenon commenced about a year ago. Early measurements used a carbon arc lamp for solar simulation and a liquid N_2 baffled diffusion

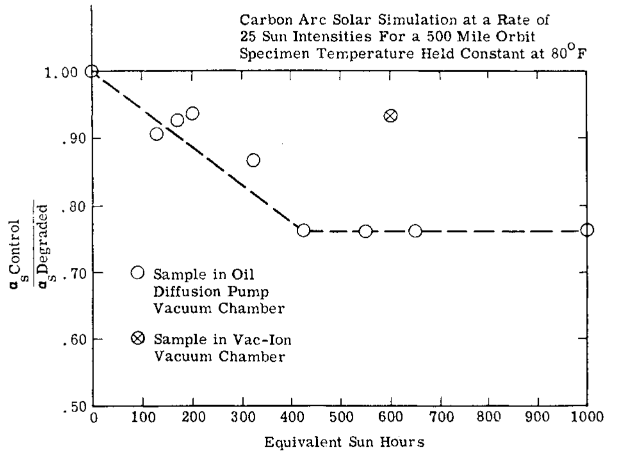

Fig. 4 Ultraviolet effects on Alzak, grade S.I., in a vacuum.

type of vacuum chamber with quartz lens. Under a vacuum of 10^{-6} mm Hg and at a constant temperature of 80°F, a 25% degradation of Alzak occurred (i.e., from a base of α_s = 0.15 to 0.20) over a period of 1000 equivalent sun hr taken at a rate of 25 sun intensities. Suspicion that the chamber was contaminating the sample by slight backstreaming of the diffusion pump oil led to a revised test, making use of a vac-ion pump with a chamber having metal seals and avoidance of all organic materials near the specimen. Measurements made with this test setup showed less than 10% degradation after 600 hr of effective sunlight (see Fig. 4). It appears that the degrading, which is a significant yellowing of the near mirror-like surface, is increased by organic contamination of the Alzak surface. In the manufacture and storage of the skins, the mirror surface is coated with vinyl to prevent scoring and fingerprints. Samples that have completed the manufacturing process are currently under further investigation in a turbomolecular chamber.

Electronic Equipment

Each bay in the spacecraft, except the C and G bays at levels 1, 2, 5, and 6, are available to hold electronic equipment. At present 23 of the 40 bays are occupied by 35 black boxes. An additional six bays contain the star trackers. The packages are sized for the bay dimensions and, if small, are tied with other boxes to insure a high thermal inertia. They are then bolted to honeycomb panels facing the skins, as shown in Fig. 2. A silicone rubber compound (RTV 40 with 25% alumina) is used to provide good thermal contact between box and honeycomb. The honeycomb is hinged to fiberglass mounts from the structure, so that the equipment can be swung outward for easy access after it is installed. The box is internally designed to transfer all its heat to the heat-sink surface that is mounted on the honeycomb panel. Figure 5 illustrates a typical bay installation with its associated heat flows. A simplified method of analyzing the equipment thermal design follows: When the extremes of power dissipation and environment heat fluxes averaged over an orbit are known, the equipment heat-sink temperatures can be calculated. Initially the box is assumed to be perfectly insulated from the structure with all its heat rejected to the skins. The temperature of the equipment heat sink is then obtained after algebraic elimination of the skin temperature:

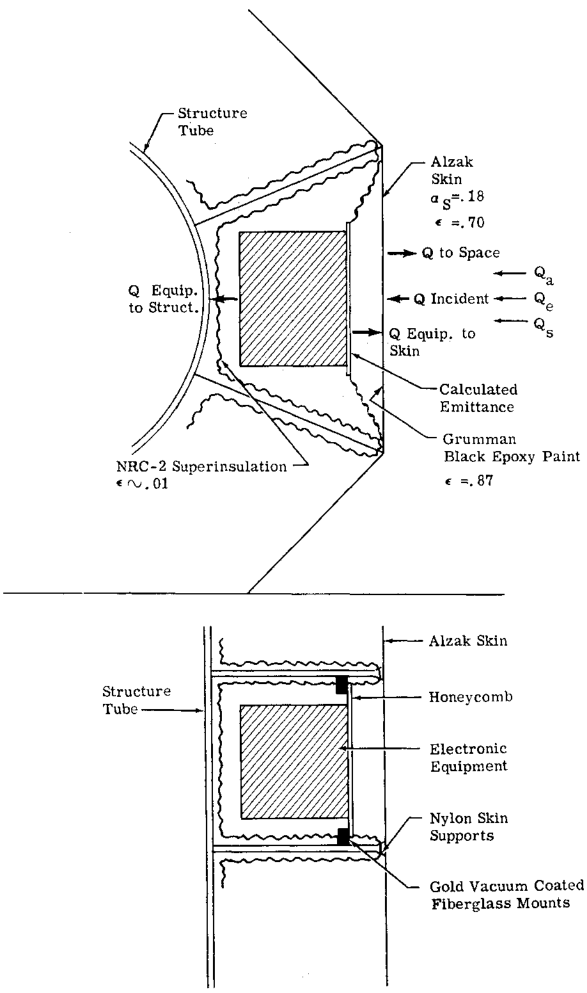

Fig. 5 Equipment bay design.

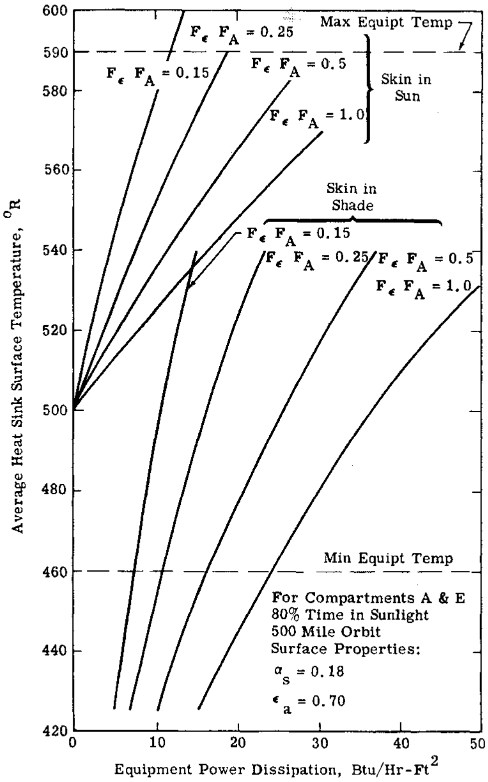

Fig. 6 Average equipment temperature vs internal power dissipation.

$$\sigma T^4_{\text{eq}} = (Q_s + Q_a)\frac{\alpha_s}{\epsilon_A} + Q_E\frac{\alpha_E}{\epsilon_A} + Q_{\text{eq}}\left(\frac{1}{A_A\epsilon_A} + \frac{1}{A_H\epsilon F_A}\right) \quad (1)$$

where for bay E in the sun (Fig. 6), with skin in sun—averaged over an orbit:

T_{eq} = equipment temperature, 540°R for this example
σ = Stefan-Boltzmann constant 0.173×10^{-8} Btu/hr ft^2 °R^4
Q_s = orbital average solar heat flux to skin, 352.0 Btu/hr ft^2
Q_a = orbital average albedo heat flux to skin, 3.33 Btu/hr ft^2
Q_E = orbital average earth radiation to skin, 16.97 Btu/hr ft^2
α_s = absorptance of Alzak skin to solar spectrum, 0.18
ϵ_A = emittance of Alzak skin at 0°F, 0.70
ϵ_B = emittance of Alzak inner surface, black epoxy paint, ϵ = 0.87
α_E = absorptance of Alzak skin to earth radiation, $\alpha_E = \epsilon_A$
Q_{eq} = equipment heat dissipation, 34 Btu/hr
A_A = Alzak skin area, 4.5 ft^2

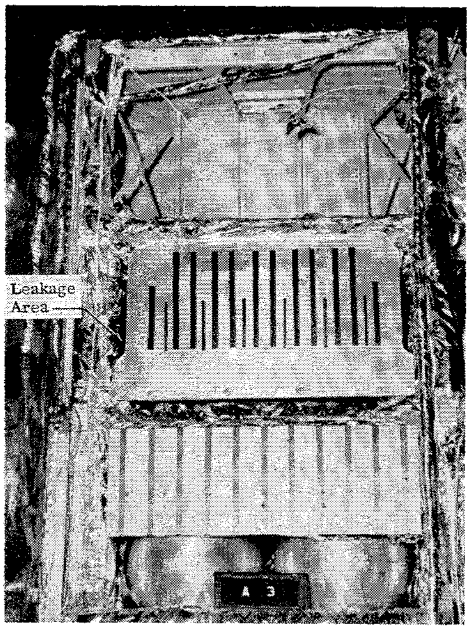

Fig. 7 OAO full-scale test model.

A_H = honeycomb, equipment radiating area, 2.6 ft^2
F_ϵ = effective emittance between honeycomb and black inner skin surface, assume 0.5
F_A = geometric form factor from honeycomb to skin, approximately 1.0

Using tolerance values of Alzak (α_s = 0.18, ϵ = 0.70), Eq. (1) is represented graphically in Fig. 6. The equipment honeycomb temperatures for sides A and E of the OAO are plotted as a function of the power dissipation per square foot of honeycomb radiating area and the emittance form factor product between honeycomb and skin. Figure 6 is used to determine the honeycomb emittance that will maintain the equipment temperature between 0° and 130°F. In using Fig. 6, the average heat loss from the equipment to the structure through the NRC superinsulation and each fiberglass fitting is first calculated. The sum of these heat losses (approximately 5 Btu/hr per bay) is subtracted from the equipment power dissipation; an $F_\epsilon F_A$ line is then determined for the maximum and minimum conditions that will maintain the equipment temperature between 0° and 130°F.

For this type of installation $F_A \sim 1.0$. The honeycomb emittance ϵ_H is calculated assuming parallel flat-plate radiation:

$$\epsilon_H = \frac{1}{\dfrac{1}{F_\epsilon} - \dfrac{A_H}{A_A}\left(\dfrac{1}{\epsilon_B} - 1\right)} \quad (2)$$

Once determined, the emittance of the honeycomb panels is increased from its initial alodine surface ($\epsilon \simeq 0.1$) to the desired value by painting area-weighted 1-in. black epoxy stripes on the honeycomb (see Fig. 7).

Equipment such as star trackers, boresight tracker, television camera, firing squibs, and pneumatic controls that include jet nozzles, radiate directly to space rather than to the skin panels. These devices have undergone separate analyses which take into account duty cycle, heat flux to and from space, conduction, and radiation to the spacecraft, and are treated separately from the completely passive control system used for the electronic equipment.

Insulation

A multiple-radiation foil insulation controls the heat flow between the equipment and structure. During initial studies, the National Research Corporation developed an aluminized Mylar superinsulation (NRC-2) which has exceptional insulating qualities.[8] Using a National Bureau of Standards type of calorimetric device, NRC data for the insulation in vacuum indicated an effective emittance equivalent to the theoretical curves for infinite parallel radiating plates, using an emittance value of 0.05 for the aluminum side (25 × 10^{-6} in. in thickness) and 0.4 for the $\frac{1}{4}$-mil Mylar side. Numerous tests of single-bay and three-bay installations showed an increase over this emittance by a factor of 2 for 25 layer blankets. Analysis of full-scale test data indicated that this factor is closer to 3.5 times the theoretical value for internal insulation and approximately 5 times for the 10-layer blankets on skin panels covering nonequipment bays (see Fig. 8).

Interior blankets are installed along all of the structure exposed to equipment surfaces. Blankets overlap each other by a minimum of 6 in. to reduce end effects but are attached loosely and held to the structure with aluminized Mylar tape. To insure a minimum of ballooning or tearing of the insulation during launch, each overlap is left open to enable air to escape freely. Altitude-pressure tests representative of launch conditions show that ballooning does occur, but the insulation remains in place with no apparent tearing. The ballooning slowly subsides within minutes after reaching the high-altitude condition. All equipment, wiring, and piping are supported from the structure using fiberglass mounts and brackets which have been vacuum coated with gold to complete the separation of heat flow from equipment to structure.

Experiment Telescope

The experiment package is comprised of the primary telescope system plus associated electronic equipment. All electronic equipment that is not required to be within the telescope envelope is stored in bays E4 and E5 and is handled in the same manner as other spacecraft electronic equipment. Any type of optical telescope may be housed within the 40-in.-diam container. Interior surfaces of the telescope and the sunshade are painted black, in order to prevent incident earth albedo reflections from washing out the star image on the detector.

In order to achieve thermal stability, the telescope is insulated from open-end radiation to space and is thermally linked to the spacecraft via conduction through four aluminum lugs and via radiation between the 40-in. container and the OAO's 48-in. structure tube. The spacecraft structure is then designed to provide the desired mean telescope operating temperature of either −22° or +32°F. This eliminates

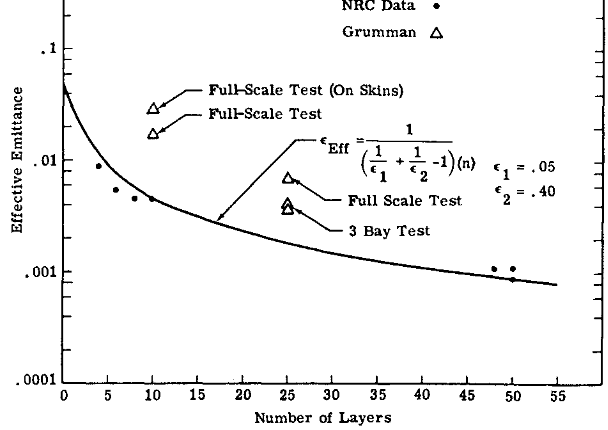

Fig. 8 Effective emittance of NRC-2 superinsulation.

temperature differences between spacecraft and telescope which would cause misalignment of the star trackers and telescope axis. By insulating the telescope from space and having the telescope and spacecraft thermally linked, the telescope gains the stability inherent in the OAO and becomes insensitive to the radical environment changes at the open end. For this design, the telescope operates within 3°F of the structure; however, it also assumes thermal gradients similar to those of the OAO's structure tube, which reach a maximum of 20°F axially and 15°F circumferentially.[9] If the telescope were to be completely insulated from the OAO and uninsulated from space, the tube temperature would extend over a range, such as shown in Fig. 9, as a function of pointing angle.

Two spacecrafts are currently being designed. The first OAO Spacecraft *A* incorporates two experiment packages: the Smithsonian Astrophysical Observatory's Project Celescope[10–12] and the Wisconsin University Package.[11, 12] Each of these points outward through opposite ends of the central tube and are mated at the midsection of the spacecraft. The *A* observatory is to operate at an average structural temperature of −22°F with a permissable deviation of ±27°F for its one-year lifetime. Spacecraft *B* contains the single-ended NASA Goddard Space Flight Center Package[11] and is to operate at +32°F with an allowable deviation of ±18°F. Both observatories can be aimed at any point in the sky, except the 45° cone about the earth-sun line when facing the sun.

Structure Thermal Balance

In discussing the OAO thermal balance, it will be assumed that the telescopes of Spacecraft *A* are thermally coupled to the structure and are insulated from space. This is the representation used in the full-scale thermal test. The problem is then to design the OAO structure to operate at −22°F. The telescope, if properly insulated, will run within 3°F of this mean temperature.

The spacecraft structure gains heat via the three following distinct modes:

1) Radiation of heat from equipment to structure through NRC-2 superinsulation.

2) Conduction of heat from the equipment bays to the structure through the fiberglass fittings. Included in these are equipment mounts, wiring brackets, and piping brackets.

3) External fittings directly attached to the structure radiating to and receiving heat from the space environment. Included in these are gas jet nozzles, solar paddle arms, solar paddle latches, sunshade latches and hinge points, separation feet, coarse and fine sun sensors.

The amount of heat conducted by sources mentioned in items 2 and 3 depend upon the orientation of the spacecraft. Careful thermal design of each source minimizes heat transfer to the structure. The sum of the heat flows from items 2 and 3 to the spacecraft amounts to a maximum of 80 Btu/hr and a minimum of 60 Btu/hr.

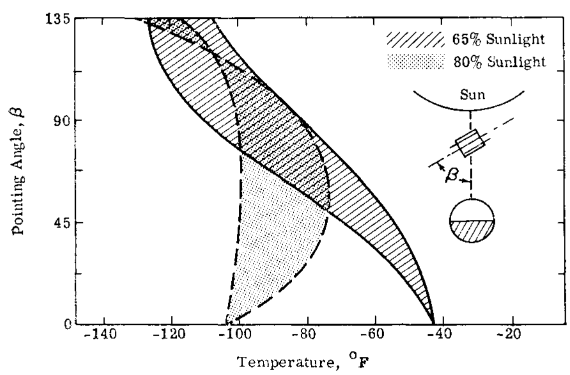

Fig. 9 Equilibrium temperature of open end.

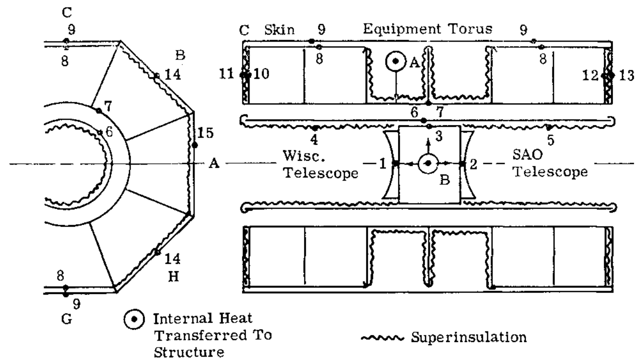

Fig. 10 Thermal schematic of spacecraft *A*.

The primary mode of heat transfer is through the bay insulation. This is controlled by varying the insulation layers to achieve the proper heat flow to the structure, so that external environment effects on the spacecraft temperature level are minimized.

The observatory looses heat to space through three areas: a) heat transferred via radiation from the structure in bays C-1, 2, 5, and 6, and G-1, 2, 5, and 6 to their corresponding thermally stable skins and out to space; b) heat transferred from the OAO structure to the telescope container and out of the insulation through the telescope aperture to space; and c) heat transferred in nonequipment bays and forward and aft skins from the structure through insulated skins and out to space.

Area a) is completely controllable and is the valve in the thermal design of the OAO. Areas b) and c) are not controllable heat leaks but have been insulated to an optimum point.

A simple IBM program was developed to make parametric calculations of the mean structural tube temperature of the OAO. A thermal nodel of the 15-node radiation program is shown in Fig. 10. The model nomenclature is defined as follows (all values are approximate):

Q_A = total heat transferred to the structure through superinsulation, fiberglass fittings, and external fittings, 160 Btu/hr
Q_B = telescope internal heat dissipation, 20 Btu/hr
1 = Wisconsin telescope mirror; $\epsilon = 0.03$, $\alpha = 0.03$
2 = SAO telescope mirror; $\epsilon = 0.03$, $\alpha = 0.03$
3 = telescope bulkhead
4 = Wisconsin telescope insulation, 10 layers NRC-2
5 = SAO telescope insulation, 10 layers NRC-2
6 = telescope containers 40-in.-diam × 118 in.
7 = OAO structure tube, 48-in.-diam × 118 in.
8 = inner Mylar layer on C, G cooling skins
9 = C, G cooling skins levels 1, 2, 5, 6
10 = forward bulkhead
11 = forward skin
12 = aft bulkhead
13 = aft skin
14 = nonequipment skins, bays B, D, F, H
15 = nonequipment skins, bays A, E

External heat inputs of solar, albedo, and earth radiation are averaged over an orbit for the particular plate orientation. Each calculation of the mean spacecraft structure temperature is obtained as a function of spacecraft orientation and heat flow from equipment to structure. The solution of these equations for the full-scale test is plotted as band A of Fig. 11. Spacecraft temperature T_7 is plotted against equipment and external heat load to the structure Q_A as a function of the observatory orientation.

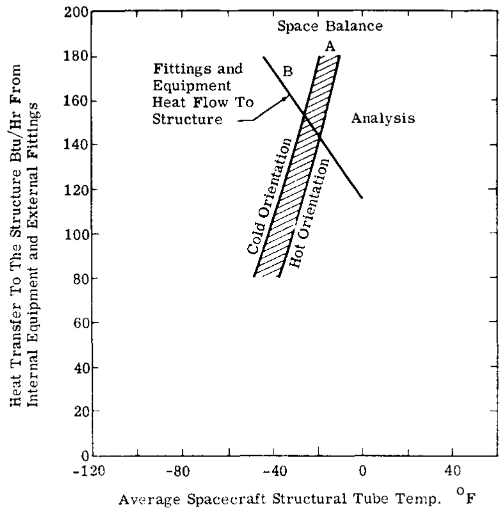

Fig. 11 Spacecraft thermal balance.

The second phase of the analysis determines the heat inputs to the structure from each equipment bay and all external fittings attached to the structure. These are calculated as a function of spacecraft operating temperature T_7, orientation, and equipment duty cycle. In Fig. 11, line B is the estimated heat input for the full-scale thermal test. Only one line is presented as an average heat input for the hot and cold orientations. The juncture of band A and line B is the operational envelope of the spacecraft. As more insulation is applied to the C and G skins, band A tends to move further to the right, resulting in higher observatory temperatures. For this design the over-all emittance between structure and C and G skins is 0.2.

When the insulation design is set and orientation extremes are determined, a second analysis is performed to verify the design and calculate temperature gradients about the OAO structural tube. In this analysis the OAO structure and telescope are broken into 220 nodes, and temperatures are computed by using a transient conduction-radiation[13] program on the IBM 7094. Both analyses have predicted the mean spacecraft structure temperature within 3°F of each other. The average axial gradient analyzed for this design was 10°F; its low point is at the ends of the spacecraft and its high in the equipment torus areas at bay levels 3 and 4. Average circumferential gradients were 4°F, having a maximum value on the sun side of the observatory. It was assumed during this analysis that all NRC-2 superinsulation had an effective emittance twice that of the theoretical value.

Spacecraft Testing

A full-scale thermal vacuum test of the OAO, duplicating the spacecraft *A* thermal design, was performed on the structure model. The spacecraft was insulated with 25-layer blankets of NRC-2 superinsulation in each equipment bay, except the C, G bays at levels 3 and 4. These bays, holding heavy constant heat dissipating equipment that acts as a stable heat source, were insulated from the structure by 10 layers of superinsulation. Each piece of equipment was constructed of heavy aluminum plate to form the proper box dimensions and weight. A heater was installed inside each box to simulate equipment heat dissipations. Equipment boxes were mounted to honeycomb panels using an RTV wetting agent. Thermocouples were attached to each box and heat-sink surface. All equipment mounts and wiring and piping brackets were made of fiberglass vacuum deposited with gold. Wiring and piping was simulated using 3- × 3-in. black sheet metal attached to the brackets. The alodined honeycomb equipment panels were then striped with black epoxy paint to the required emittance as determined from the equipment analyses (see Fig. 7). Instrumentation consisted of a total of 360 thermocouples made from 30-gage copper constantan wire. Tests were carried out in the Grumman high-vacuum chamber (15 × 20 ft) with liquid N_2 cold walls.

To simulate the solar, albedo, and earth radiation, Nichrome strip heaters were installed on the inner surface of the Alzak skins, and wattage inputs were programed for each skin surface. Each skin was chem-milled from its normal 10-mil thickness to 7 mils. A layer of Mylar tape was applied and, over this, 0.002-in. Nichrome heater strips were fastened end to end, covering the entire skin surface. Black epoxy paint was sprayed over the Nichrome heater strips. Variacs controlled each skin panel from a central control station. In equipment bays, skins were insulated from the spacecraft structure with 10 layers of superinsulation, except for the C and G skins at levels 1, 2, 5, and 6. These skins were specially prepared using only one sheet of aluminized Mylar. The Mylar side (ϵ = 0.28 as measured spectrally) faced the anodized OAO structure of the C and G bays, and the aluminized side was painted black and faced the black epoxied Nichrome heaters for an over-all emittance of skin to structure of approximately 0.2.

External fittings were wound with Nichrome wire and wrapped with superinsulation. The double-ended experiment package was simulated by insulating each container facing the open end with 10 layers of NRC-2 superinsulation. The bulkhead was fabricated using aluminum disks covered with one sheet of aluminized Mylar to simulate both mass and mirror emittance. All insulation within the cavity except the simulated mirror was spray-painted with black epoxy paint. Radiant interchange to space was simulated by covering the open end with 25 layers of superinsulation but leaving a 9-in. hole to simulate a blackbody radiator. A heater and three thermocouples were installed in the cavity. The cavity temperature could therefore be adjusted, and heat leakage to space could be matched against analytical values. Gravity vector gages were installed on all the star trackers, inertia wheels, and telescope axes to measure distortion referenced to the spacecraft axis. No solar paddles or sunshades were used. To provide the proper heat fluxes to the C and G skins during the test, it was necessary to calculate an effective incident heat rate, which included solar paddle heating and blockage, although in the test these skins radiated directly to the liquid N_2 cold walls. A similar analysis was done for the experiment cavity to include sunshade effects.

Spacecraft *A*, short of wiring and piping and assembled as thermally exact as possible, was lowered into the vacuum chamber. Backfilling with liquid N_2 to 1-mm-Hg pressure accelerated cooling of the spacecraft to the −22°F structure temperature region. The test was conducted under a vacuum of 10^{-7} mm Hg and lasted 36 consecutive days.

Test Results

In general, test results verify the adequacy of the thermal design and the validity of the previous analytical work. The sum of the problems and their explanations are presented below.

Equipment Cooling

Initial tests of the electronic equipment thermal design, oriented to and away from the sun and at maximum and minimum power dissipation, indicated that the equipment temperature spread was within the calculated values. However, the mean temperature level of the equipment ran about 30°F cooler than anticipated. In instances where the boxes completely filled a bay, such as the Battery Charger and IBM packages, the level was within 5°F of the estimates. When the boxes were small and there were gaps between the box heat-sink surface and NRC-2 insulation (see arrow in Fig. 7), test results ran well under the calculated values.

Since the time of this test, the electronic design has been revised so that all such voids will be filled with superinsulation. Thus the radiating area-emittance-form factor relationship from honeycomb to skin is completely controlled.

Structure

The mean structure temperature successfully operated within 5° of the required −22°F design. The simulated telescope structure temperature was within 2°F of the spacecraft average for each orientation.

To determine the mean spacecraft temperature, gradients and distortions, equipment temperature levels were raised to operate in the 65°F temperature region estimated for the normal equipment design. All skin inputs were based on average orbital heat inputs for a particular orientation. The spacecraft was operated in this manner for 5 days until an equilibrium condition was reached, whereupon distortion and final temperature measurements were made. Skin heat inputs were then varied for a second orientation.

Thermal distortion between star trackers and telescope axis because of orientation changes did not exceed ±30 sec of arc. This is well within the field of view of all trackers and does not interfere with the primary telescope sensing circuit used to bring the observatory within the required $\frac{1}{10}$ sec of arc.

The spacecraft skins were automatically cycled through typical orbital conditions of sun and shade for one orientation for two days, wherein the average temperature of 48 thermocouples on the spacecraft tube did not deviate by more than $\frac{1}{2}$°F. Distortion measurements made during this period showed no changes from their steady-state values, indicating substantial thermal damping of the structure from orbital transient heat fluxes.

The average temperature of the spacecraft for hot and cold structure orientations were −28° and −21°F, respectively, vs the analytical values of −19° and −26°F, as shown in Fig. 11. The −28°F condition occurred because of two errors in the test: 1) all skins other than the C and G control skins were operated at two-thirds the required power, and 2) the equipment average temperature level was 55°F compared to the estimated 65°F. (However, for the cold case, the equipment temperature level was 76°F instead of the estimated 65°F.).

Using the forementioned test inputs, re-analysis showed that the hot orientation would have operated below the cold orientation, as it did. Considering the correct orbital heat inputs, the spacecraft appears to have been designed approximately 5°F above the required −22°F level.

Average measured gradients of 16°F axially and 9°F circumferentially were greater than anticipated, although the gradient shapes were similar to those previously analyzed. Test data of gradients and temperature level were fitted to the large IBM program, leading to the conclusion that the NRC-2 superinsulation, as installed in the OAO, has an emittance of 3.5 times the theoretical value for 25-layer blankets and approximately 5 times the value for 10-layer blankets used on the skins.

Further Testing

A thermal vacuum development test of the *B* Spacecraft was completed in May 1964. The specimen was the *B* flight spacecraft utilizing real piping and wiring, dummy equipment, and a simulated experiment package. The test confirmed the analytical technique and thermal design in achieving a 32°F spacecraft.

In late 1964, the prototype of spacecraft *A* will be thermal-vacuum tested for 30 days. The model will contain a prototype experiment package and real equipment operated at orbital duty cycles. The thermal instrumentation is similar to that of the flight spacecrafts. Subsequent to this test, any correction required to bring the mean spacecraft temperature level to the −22°F design will be made by adjusting the internal emittance properties of the C and G skins.

The *A* flight spacecraft will be acceptance-tested using heater skins to simulate the space environment. After the test the spacecraft will be delivered to Cape Kennedy for flight preparations and launching.

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