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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

NASA CONFERENCE ON THERMAL RADIATION PROBLEMS IN SPACE TECHNOLOGY

A COMPILATION OF SUMMARIES OF THE PAPERS PRESENTED

Langley Research Center
Langley Field, Va.

September 12 and 13, 1960

NASA

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THERMAL PROBLEMS OF SPACECRAFT

INTRODUCTION

This document contains summaries of the talks presented at a small NASA Conference on Thermal Radiation Problems in Space Technology held at the Langley Research Center on September 12 and 13, 1960. The conferees were NASA members and NASA contractors directly concerned with radiation characteristics of materials and with thermal problems of spacecraft. The purpose of the conference was not only to present significant research information but also to provide an opportunity for the conferees to become aware of each other's fields of interest, activities, techniques, and problems.

As arranged herein, the first set of talks, mostly by the contractors, is concerned with surfaces and surface coatings, and with the determination of their pertinent characteristics. The second set of talks, mostly by NASA personnel, is concerned with spacecraft and their thermal problems.

A list of the conferees is included.

Appreciation is expressed to the speakers, both for participating in the conference and for preparing the summaries.

This document is being distributed to the conferees to serve as a record of the conference. It is not a NASA publication and should not be used as a reference in publications.

LIST OF CONFEREES

ALTEKRUSE, John L.	Goodyear Aircraft Corp.
ASKWYTH, William H.	Pratt & Whitney Aircraft
BAKER, Kenneth M.	Bausch & Lomb, Washington, D.C.
BASTIAN, Robert P.	Republic Aviation Corp.
CARUSO, Anthony J.	NASA - Langley Research Center
CASEY, Francis W., Jr.	NASA - Langley Research Center
COOLEY, William C.	NASA - Headquarters
CURRY, Richard	Pratt & Whitney Aircraft
DEUTSCH, George C.	NASA - Headquarters
FUSSELL, William B.	NASA - Goddard Space Flight Center
GAUMER, Roger E.	Lockheed Aircraft Corp.
GEYE, Richard P.	NASA - Lewis Research Center
GILLESPIE, Warren, Jr.	NASA - Langley Research Center
HALL, Joseph F., Jr.	Bausch & Lomb, Rochester, N.Y.
HARRISON, William N.	National Bureau of Standards
HASLETT, Robert A.	Grumman Aircraft Engineering Corp.
HASS, George	U.S. Army Engineering Research & Development Lab., Fort Belvoir, Va.
HASTINGS, Earl, Jr.	NASA - Langley Research Center
HEATH, Atwood R., Jr.	NASA - Langley Research Center
HELLER, Gerhard B.	NASA - Marshall Space Flight Center
HIESTER, Nevin K.	Stanford Research Institute, Menlo Park, Calif.
KATZ, Arthur J.	Grumman Aircraft Engineering Corp.
KATZOFF, Samuel	NASA - Langley Research Center
KOVACIK, Victa P.	Thompson Ramo Wooldridge
LEIGH, Charles H.	AVCO Corp.
LEWIS, Beverly W.	NASA - Langley Research Center
LYONS, George J.	Pratt & Whitney Aircraft
MACLAY, James E.	Jet Propulsion Laboratory
MADDEN, Robert T.	Goodyear Aircraft Corp.
MAKI, Arthur	National Bureau of Standards
MARCO, Donald M.	Goodyear Aircraft Corp.
MASON, Robert M.	NASA - Langley Research Center
McDONOUGH, Ralph	Baird-Atomic, Inc.
McEVILY, Arthur J., Jr.	NASA - Langley Research Center
MISKULIN, Paul	Jet Propulsion Laboratory
NEEL, Carr B.	NASA - Ames Research Center
NICHOLS, Lester D.	NASA - Lewis Research Center

POLLACK, John L.	NASA - Lewis Research Center
RARING, Richard H.	NASA - Headquarters
REIMER, Richard R.	Thompson Ramo Wooldrige
RICHMOND, Joseph C.	National Bureau of Standards
RUBERT, Kennedy F.	NASA - Langley Research Center
SCHACH, Milton	NASA - Goddard Space Flight Center
SCHNITZER, Emanuel	NASA - Langley Research Center
SCHOCKEN, Klaus	NASA - Marshall Space Flight Center
SCHULMAN, Fred	NASA - Headquarters
SHORTEN, Frederick J.	National Bureau of Standards
SPIERS, Robert B., Jr.	NASA - Langley Research Center
STIMLER, Frederick J.	Goodyear Aircraft Corp.
STAUSS, Henry E.	NASA - Headquarters
SUMMERS, Robert A.	Allied Res. Associates, Inc.
THOSTESEN, Thomas O.	Jet Propulsion Laboratory
TROUSDALE, William L.	Pratt & Whitney Aircraft
TURNER, Richard E.	NASA - Langley Research Center
WADE, William R.	NASA - Langley Research Center
WESTON, Kenneth C.	NASA - Space Task Group

MEASUREMENTS OF SPECTRAL AND TOTAL EMITTANCE IN VACUO

By George J. Lyons
Pratt & Whitney Aircraft

In order to optimize the design of space radiators, practical engineering data are required on the emittance characteristics of the materials involved. In order to acquire such data, a program was initiated at Pratt & Whitney Aircraft under contract to NASA to evaluate both the spectral normal and total hemispherical emittance of metals and coated metal surfaces under conditions which would simulate space environment.

Spectral normal emittance measurements were made by comparing the radiation intensity from the surface of a tubular sample with the radiation intensity from a small black-body hole cut in the tube wall. Since both intensities originate from essentially the same temperature, and since the radiation from the hole is essentially the same as that from a black body at the tube temperature, the ratio of surface intensity to black-body hole intensity is a close measure of the spectral normal emittance. This is the method used by Larrabee (ref. 1) and De Vos (ref. 2). A modified Perkin-Elmer dual-beam spectrophotometer with the addition of external optics was used to make the spectral measurements over the wavelength range of 0.45 to 15 microns and at temperatures above 1,400° F. A schematic diagram of the optical system is shown in figure 1.

Total hemispherical emittance was measured by the method used by Jones and Langmuir (ref. 3). This method involves the determination of the temperature and the power dissipation per unit surface area in the central portion of an electrically heated tube or strip specimen. If heat conduction and convection from this central portion, as well as radiation returning to the test section from its surroundings, are negligible, the total hemispherical emittance may then be computed as the power dissipation per unit surface area divided by the emissive power of a black body at the same temperature. Specimen temperatures were measured by means of 0.001-inch wire thermocouples, made of platinum and platinum—10-percent-rhodium, resistance welded to the substrate metal of the specimen. Emittance measurements were made over the temperature range 200° to 2,000° F. The measurements were made in vacuum chambers designed to meet the following basic requirements:

(1) Radiation leaving the surface of the specimen should not be reflected back to the specimen to a significant degree by the walls of the test chamber.

(2) The pressure in the chamber should be as low as possible so that adverse effects of the low outer-space pressures on the radiation properties of material surfaces could be simulated to a reasonable degree.

(3) Conduction losses through the specimen power connections and instrumentation lead wires should be negligible relative to heat rejection by radiation.

Pressures as low as 10^{-9} millimeters Hg were attained by means of ion-gettering vacuum pumps. Axial heat-conduction losses to the electrodes supporting the sample were minimized by supplying supplementary heat to the ends of the specimen in controlled amounts, thus achieving a uniform temperature distribution along the test section. A cutaway sketch of the test chamber used for total emittance measurements is shown in figure 2.

Most of the materials tested were nonmetallic or metallic black coatings which held promise of having high emittance combined with the quality of withstanding both high vacuum and high temperature without deterioration. A large number of coatings were tested on a short-time basis, and then the more promising coatings were subjected to endurance tests up to 550 hours. The materials tested may be classified as uncoated metals, oxidized metals, electroplated coatings, plasma-arc-sprayed coatings, and painted coatings. The emittance values measured in this program varied from less than 0.1 for pure metals to as high as 0.9 for the better coatings.

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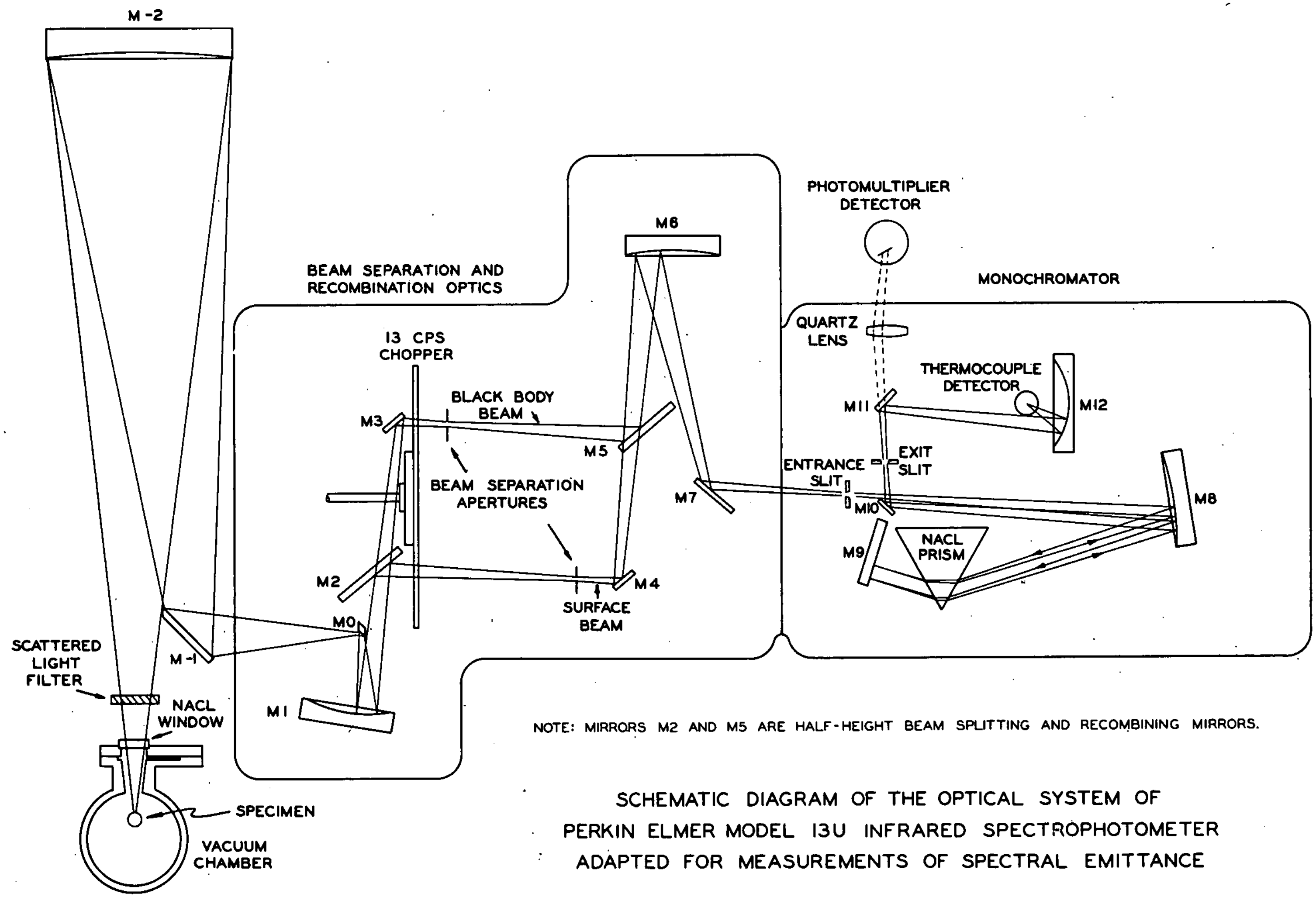
2. De Vos, J. C.: A New Determination of the Emissivity of Tungsten Ribbon. Physica, vol. 20, no. 10, Oct. 1954, pp. 690-714.

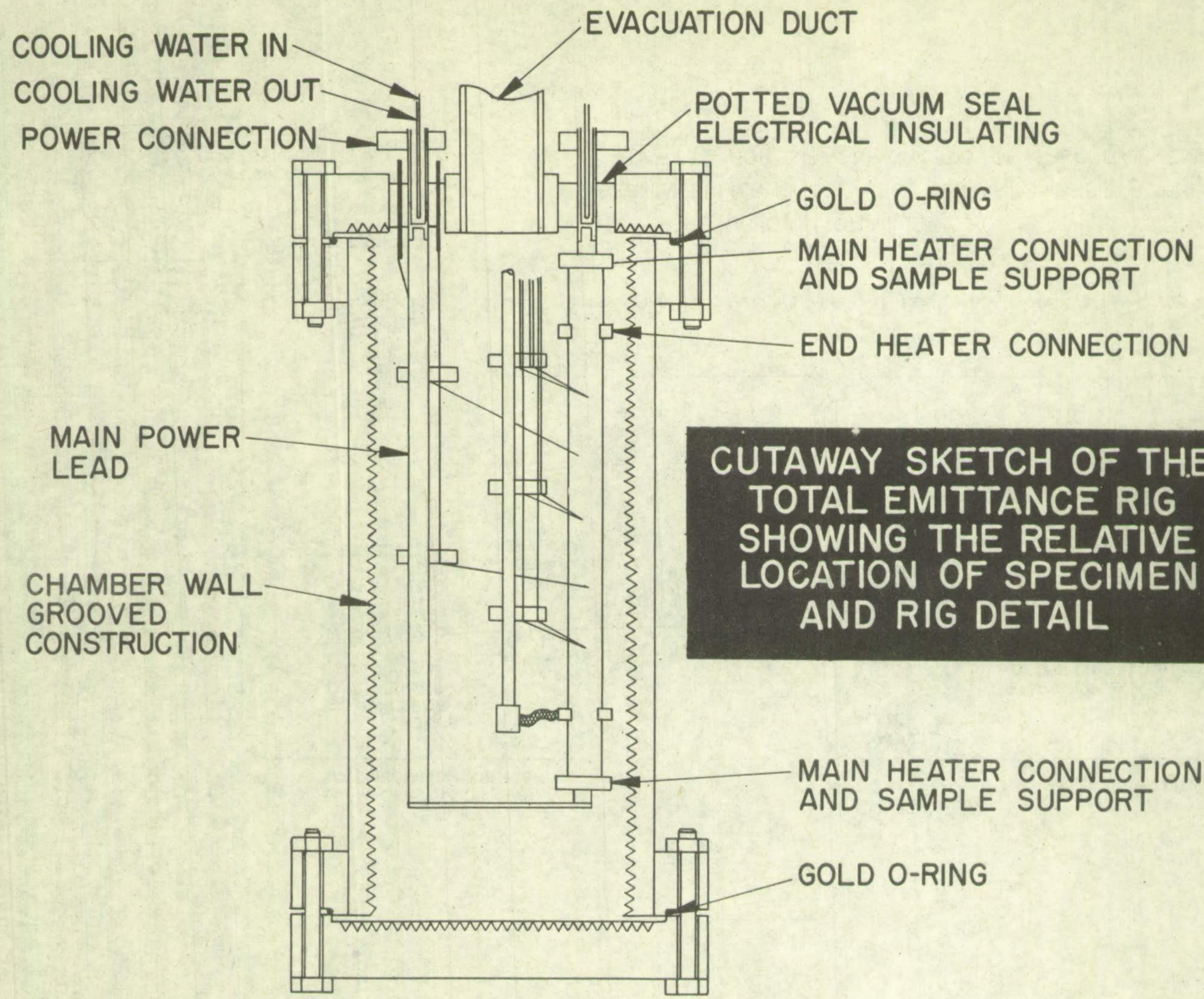
3. Jones, H. A., and Langmuir, I.: The Characteristics of Tungsten Filaments as Functions of Temperature. General Electric Review, vol. 30, June 1927, pp. 310-319; July 1927, pp. 354-361.



NOTE: MIRRORS M2 AND M5 ARE HALF-HEIGHT BEAM SPLITTING AND RECOMBINING MIRRORS.

SCHEMATIC DIAGRAM OF THE OPTICAL SYSTEM OF
PERKIN ELMER MODEL 13U INFRARED SPECTROPHOTOMETER
ADAPTED FOR MEASUREMENTS OF SPECTRAL EMITTANCE

Figure 1



CUTAWAY SKETCH OF THE TOTAL EMITTANCE RIG SHOWING THE RELATIVE LOCATION OF SPECIMEN AND RIG DETAIL

Figure 2

SPECTRAL AND TOTAL EMITTANCE AND REFLECTANCE STUDIES IN THE ENAMELED METALS SECTION OF THE NATIONAL BUREAU OF STANDARDS

By Joseph C. Richmond
National Bureau of Standards

The Enameled Metals Section of the National Bureau of Standards has been engaged in research on emittance and reflectance for the Army Ballistic Missile Agency, now the George C. Marshall Space Flight Center of the National Aeronautics and Space Administration, since 1957. The work has included:

1. Development of instrumentation and procedures for the evaluation of total hemispherical emittance by a modification of the hot-filament method. This work is described in detail in reference 1, and briefly in reference 2.

2. Development of equipment and procedures for the evaluation of spectral reflectance, under conditions approximating normal illumination and hemispherical viewing, of specimens at room temperature or at controlled temperatures somewhat above or below room temperature. The equipment has been designed and constructed, and is now being calibrated. It is described in reference 2.

3. Evaluation of emittance and reflectance properties of a large number of materials of interest to the space program. These included solid specimens of two types of graphite, composite specimens of a number of flame-sprayed coatings applied to various metals and alloys, composite specimens of several alloys to which several types of NBS ceramic coatings had been applied, bare specimens of several alloys in the polished and sandblasted condition, and specimens of several alloys that had been oxidized for various times at 1,000° C subsequent to polishing and sandblasting, respectively. Evaluations included (a) total hemispherical emittance at temperatures of 300° to 1,000° C and (b) spectral reflectance at room temperature under conditions approximating normal illumination and hemispherical viewing, over the wavelength ranges of 0.40 to 1.08 microns and 1 to 15 microns, respectively. Data for these evaluations appear in references 3 and 4.

4. Present work is concentrated on a feasibility study. Most ceramic coatings are normally applied in thicknesses at which they are not completely opaque. The emittance of a composite specimen comprising a partially transparent coating applied to an opaque substrate will be influenced by the emittance properties of both the substrate and the coating material and will vary with the thickness of the coating. Theoretical equations have been developed relating the normal spectral emittance of such a specimen to the emittance of the substrate into the

coating and to the thickness, coefficient of scatter, absorption coefficient, and index of refraction of the coating material. All of these properties vary with wavelength. The study is concerned with the feasibility of setting up tables of these properties, at all the wavelengths of interest, so that the normal spectral emittance curve of a composite specimen of any given thickness of coating applied to any substrate can be computed.

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EMISSIVITY OF METALS AFTER DAMAGE BY PARTICLE IMPACT

By C. H. Leigh
AVCO Corporation

The purpose of this investigation is to determine the effect of simulated micrometeorite impact upon the spectral emissivity of metals. The metals chosen for study are gold, silver, stainless steel, chromium plate, tungsten, and aluminum. Micrometeorite impact will be simulated by means of a hypervelocity gun which fires small particles of known size at velocities in the region of 12,000 to 15,000 feet per second.

The first phase of the investigation is to determine the spectral emissivity of the above metals after they have been made into disks 1 inch in diameter by one-sixteenth of an inch in thickness. All samples are carefully polished mechanically to give a smooth flat surface. The flatness of each surface is determined by means of a talysurf machine which measures across the diameter of a sample. The fine surface structure is also defined by means of an interference microscope using thallium light. The appearance of the fringes indicates the surface flatness and the size and depth of any surface pits, such as those caused by microparticle impact.

The method employed in the determination of the spectral emissivity was the reflectivity method used by Gier, Dunkle, and Bevans in reference 1. This method employs a black-body cavity at approximately 800° C within which the metal sample is placed coplanar with the roof of the black body. The metal sample is maintained at room temperature by means of water cooling or is taken down to the temperature of liquid nitrogen by circulation of liquid nitrogen coolant. Since all samples are opaque, the emissivity is simply $\epsilon = 1 -$ Reflectivity.

The wavelength range covered by the instrument is dependent upon the temperature of the black body and the sensitivity of the spectrophotometer employed to detect the radiation. At 800° C the maximum is at about 2.65μ with 99 percent of the total energy falling below 20μ. At the present time the wavelength range of 2 to 15 microns has been studied, and longer wavelengths will be investigated by increasing the long-wavelength energy by raising the temperature of the hohlraum.

The metal samples used are in the form of disks about one-sixteenth of an inch in thickness. The original metal is ground flat and subsequently polished. The polishing agent depends upon the nature of the metal; the following agents have been used:

Tungsten	500μ size Linde B alumina; soft magnesium oxide
Gold and silver	Abrasive rouge
Stainless steel	Linde B alumina
Aluminum	Magnesium oxide

Grinding is accomplished by using a cloth lap and 1μ diamonds.

The optical path is shown in figure 1. In the present system the single-beam operation of the Perkin-Elmer Model 13 infrared spectrophotometer is used. The reference black-body radiation and the reflected radiation from the sample are selected by rotation of the hohlraum through 180°. Temperature within the hohlraum is continuously monitored with four chromel-alumel thermocouples firmly anchored to the top, sides, and bottom of the cavity walls. The cavity is heated by three independent heaters which are adjusted to give temperature uniformity within the cavity. Temperature uniformity can be maintained at 800° C within ±1° C.

It has been found that a cool sample, that is, a sample at a much lower temperature than the cavity temperature, causes the cavity to lose its temperature uniformity if left in position for any length of time. Point-by-point measurements must therefore be made, the cool sample remaining in position within the cavity only long enough to obtain a measurement, normally about 30 seconds, the sample then being removed. The cavity thus maintains uniform temperature conditions.

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The effect of simulated micrometeorite particle impact on the spectral emittance as determined by the above method will be investigated. The micrometeorites will be simulated by small particles of silica or iron oxide and will be accelerated to high velocity, approximately 1,300 feet per second, by means of a high-velocity gun. It is planned to select particles of known size and density and to load the gun with these particles. The metal sample will be the target, and the whole system of gun and sample will be housed in a vacuum range. Particle impact will damage the surface of the metal sample and the nature of the damage and the effect upon the spectral emittance will be determined.

The reflectance data obtained on a representative material, type 2S aluminum, is shown in figure 2. The temperature of the cavity was 500° C and the sample temperature was maintained at room temperature or liquid nitrogen temperature. The measured curve closely approaches the Hagen-Rubens theoretical curve. Similar data are obtained for all samples.

The state of the surface is illustrated in figures 3, 4, and 5 for pure silver, tungsten, and pure gold, respectively. These photographs show a small portion of the surface under an interference microscope, with thallium light being used for illumination. A particle of 1μ diameter would cause a crater on impact which would cover several of the fringes shown.

REFERENCE

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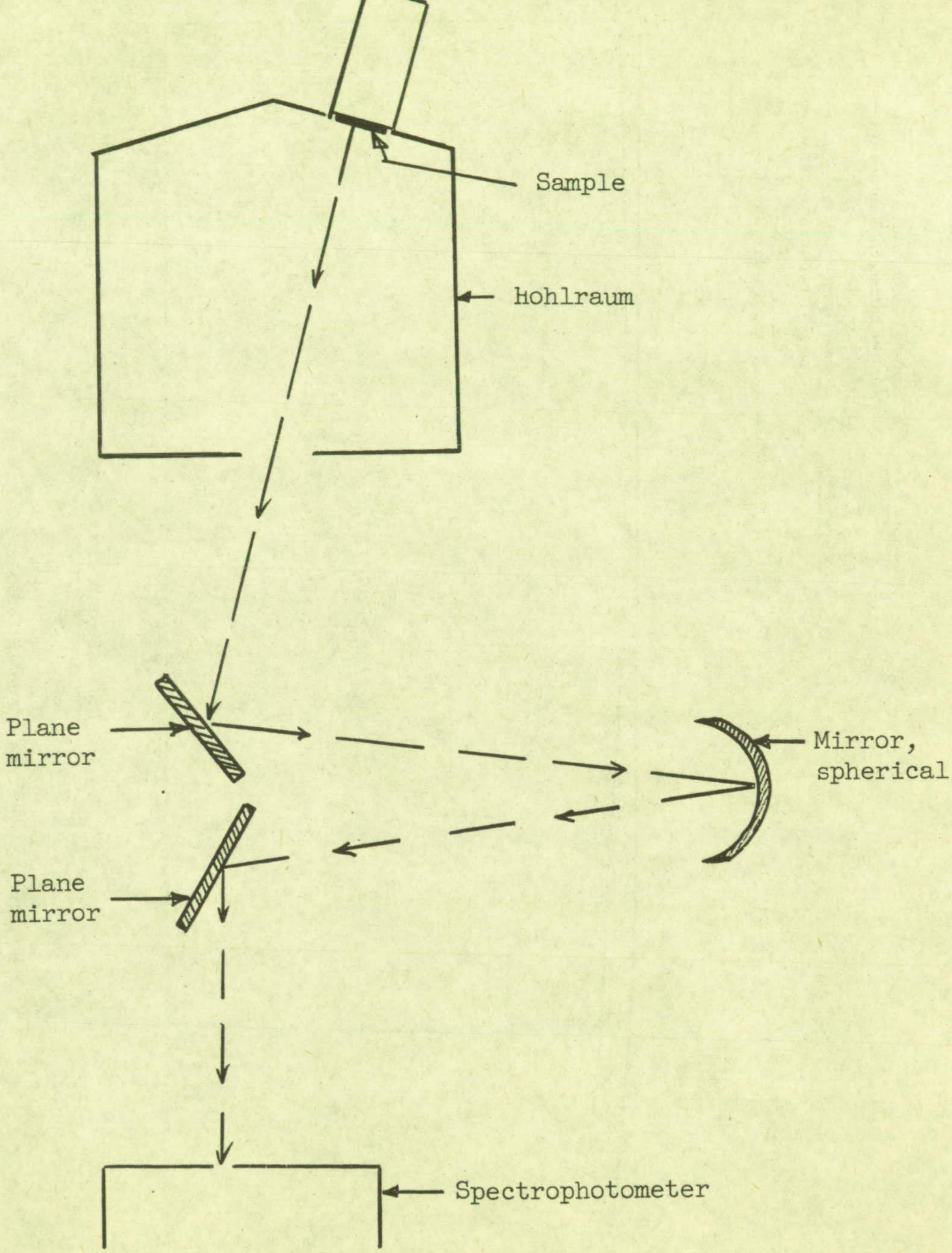

Figure 1.- Modified optical system.

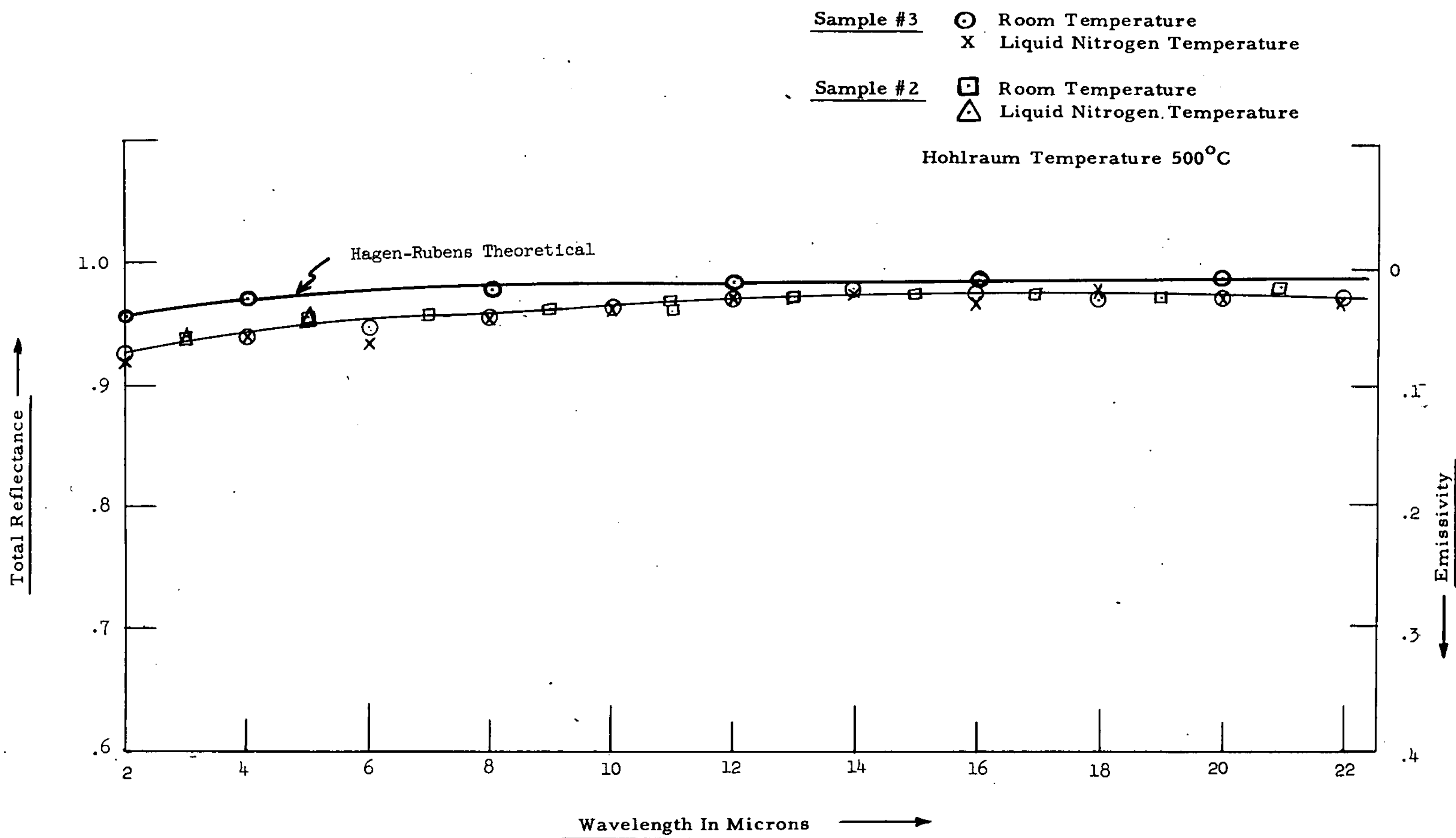

Figure 2.- Total reflectance of 2S aluminum.

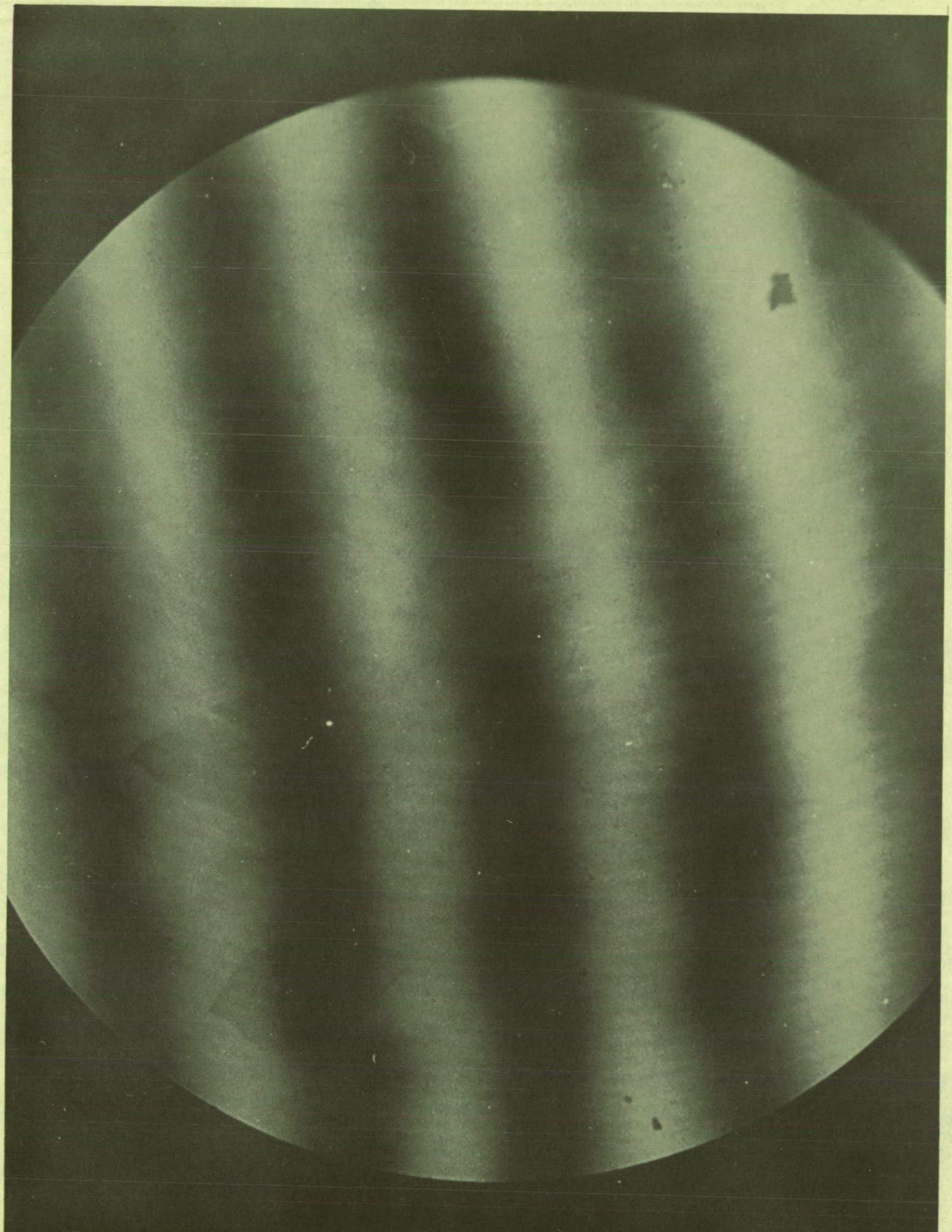

Figure 3.- Pure silver (magnification, X8). L-60-6915

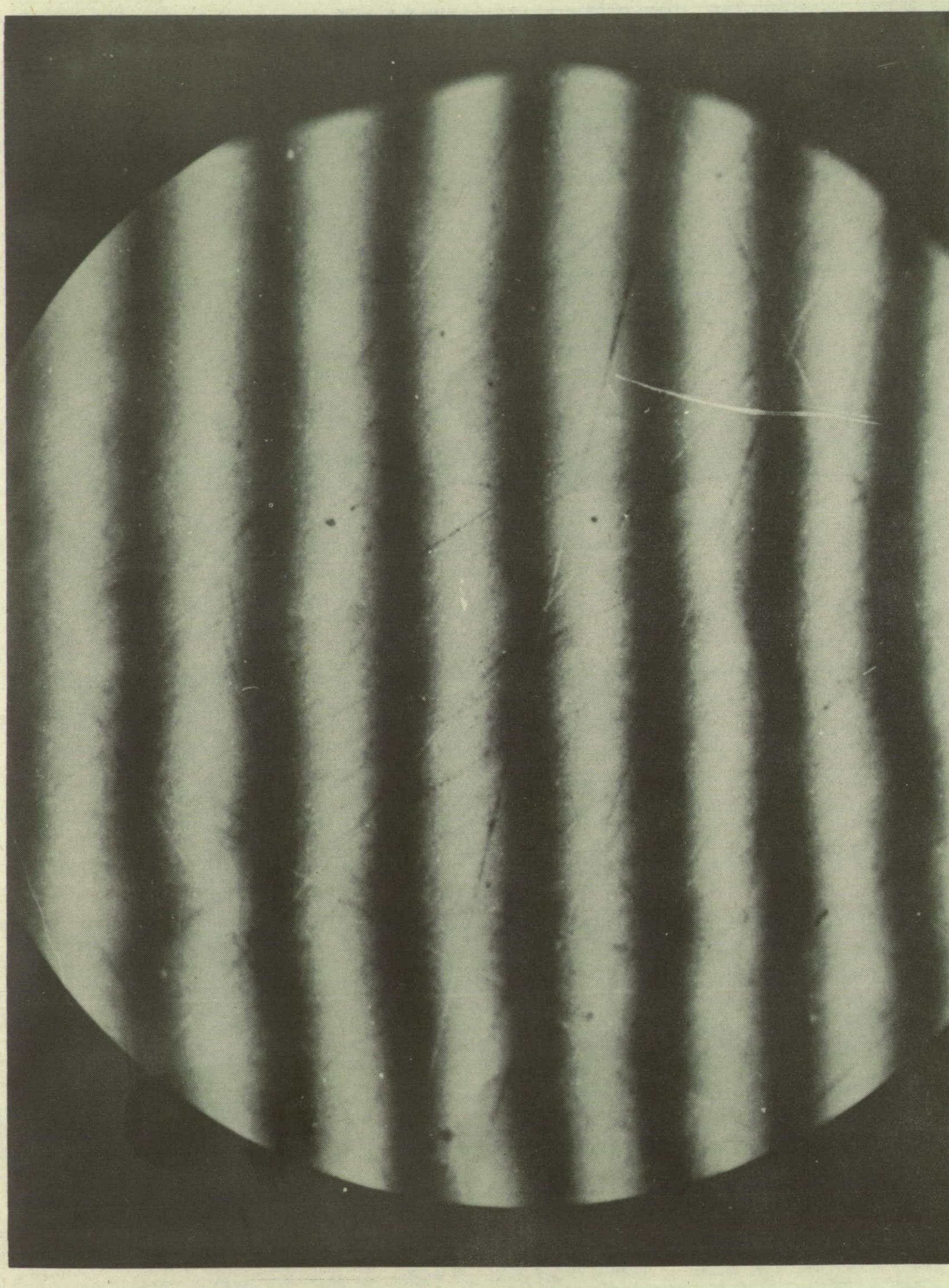

Figure 4.- Tungsten (magnification, X20). L-60-6916

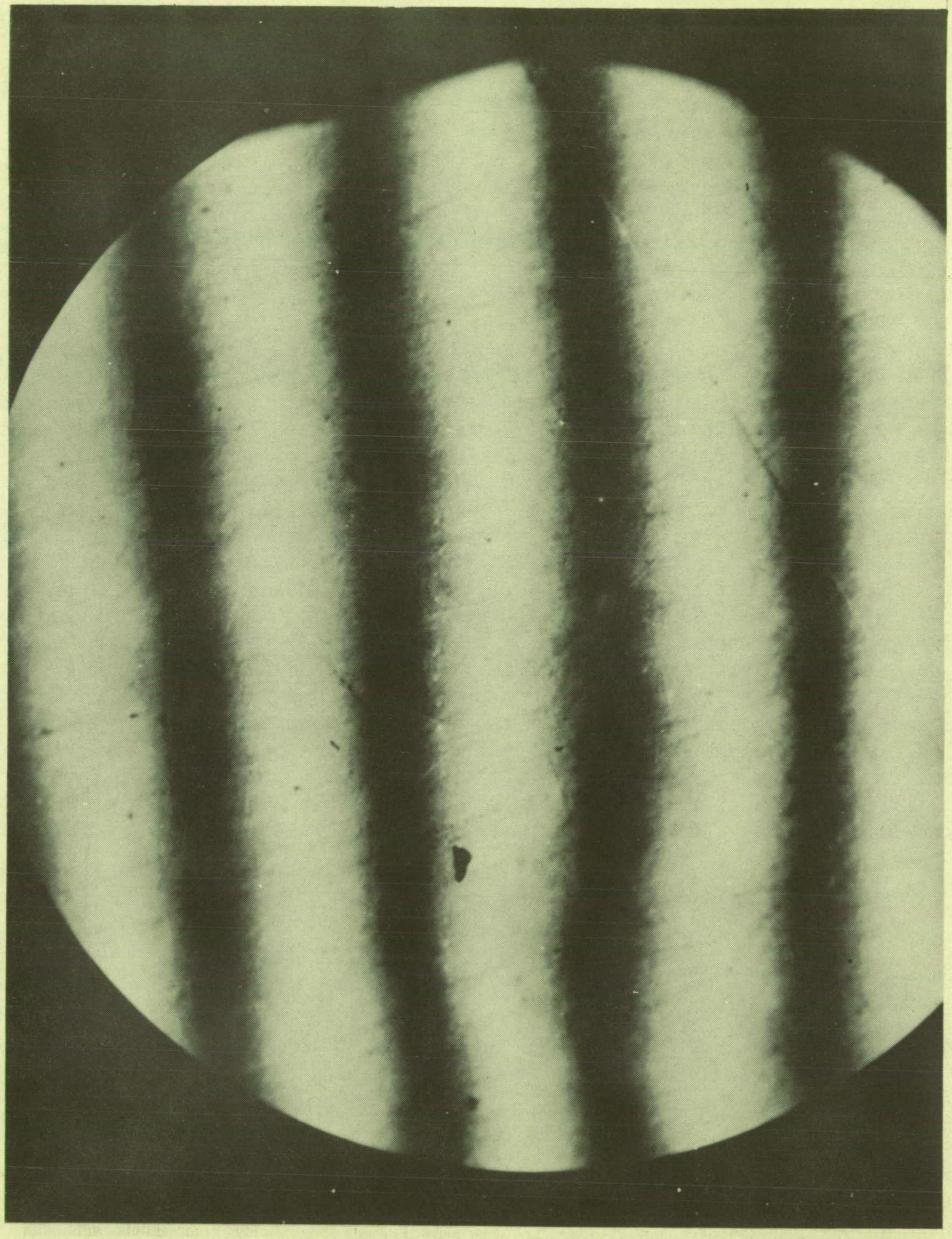

Figure 5.- Pure gold (magnification, X20). L-60-6917

SPECTRAL EMITTANCE AND REFLECTANCE MEASUREMENTS AT THE JET PROPULSION LABORATORY

By James E. Maclay
Jet Propulsion Laboratory

In late 1959 a spectrophotometer laboratory was established at JPL to be used as a support for temperature-control programs. The immediate task was spectroradiometric measurements of light sources to be used in a simulated-deep-space environment chamber. It was also necessary to have permanent provisions for measuring reflectances of surfaces for the calculation of solar absorptance α and thermal emittance ϵ.

The spectral distribution of an unknown source, in this case a Ventarc carbon-arc lamp, was derived from the ratio of detector outputs (with monochromatic light) of the arc lamp and a tungsten lamp, and the spectral intensity of the tungsten lamp. The tungsten lamp is a ribbon-filament lamp calibrated from 0.25 to 2.6 microns. A recording of pen deflection (linearly proportional to detector output) of each source over the visible and near infrared range is made with all parameters (slit width, amplifier gain, and source focusing) identical; then the ratios are computed at discrete wavelengths from these recordings. Because the spectral intensities of the tungsten lamp and a solar simulator are very different in most of the range, an adjustable slit is placed immediately in front of the detector to obtain comparable pen deflections for the two sources. This slit is set by means of a feeler gage, and the attenuation coefficient is evaluated by a transmission measurement. If $I(\lambda)$ refers to spectral intensity, $D(\lambda)$ refers to pen deflection, $R(\lambda)$ refers to the transfer function of the spectroradiometer (including entrance optics, monochromator slit setting, and detector response), G refers to the attenuation from the auxiliary slit, and the subscripts u and s refer to the unknown source and the standardized source, respectively, then

$$I_u(\lambda)\ G_u\ R(\lambda) = D_u(\lambda)$$

$$I_s(\lambda)\ G_s\ R(\lambda) = D_s(\lambda)$$

and after dividing and rearranging,

$$I_u = I_s \frac{G_s}{G_u} \frac{D_u}{D_s}$$

at some wavelength. In use, $G_u = G_s = 1$ at about 1 micron. Below this wavelength, $G_s = 1$ and G_u is less than unity; the reverse is true for wavelengths above 1 micron. The extreme values of G_u have placed a lower bound on the spectral distributions thus far obtained of 0.4 micron. At 0.36 micron, G_u would be 1/240 and at 0.25 micron, G_u would be 1/4800. It is difficult, if not impossible, to obtain such fine adjustments of the slit. There appears to be no easy way to avoid this difficulty, short of extensive filtration.

During the two scans of the known and unknown source, the detectors, with the exception of a 1P28, will drift by as much as 5 percent. In order to eliminate this drift, it is planned to do all future work on a wavelength-by-wavelength basis, with a flip-mirror to obtain the two readings for the ratio at essentially the same time. A thermal detector will probably be used, as a photomultiplier tube requires a long-time-constant filter to remove the arc flicker. In order to reduce the aforementioned difficulty of spectral intensity differences, filtering the tungsten lamp radiation will provide a more nearly solar distribution; thus, G will be fairly constant even though extreme throughout the entire range. This situation will be easier to cope with than an extreme and variable G.

The equipment for reflectance work is in two units. One utilizes an integrating sphere as an accessory on a double-beam ratio-recording spectrophotometer to cover the range of 0.24 to 0.75 micron; the other uses a heated cavity (hohlraum) as an accessory on a double-beam ratio-recording spectrophotometer to cover the range of 0.5 to 25.5 microns.

The integrating-sphere system has been in use only a short time. This device measures percent reflectance by comparison with an MgO standard. Such a comparison is undesirable but difficult to avoid in ratio-recording instruments.

The hohlraum unit has been subject to a great deal of rework to produce reasonable data. The inaccuracy was clearly indicated by reflectance values greater than unity for specular samples. The system employs a double-beam ratio-recording instrument which compares a reference beam coming from the cavity roof with the sample beam originating from the walls and floor and reflected from the sample face. The walls of the cavity are nickel oxide, which has a low reflectance (very black); thus, very little reference energy comes from the walls. Hence, if the cavity walls and floor were hotter than the cavity roof, readings greater than unity would be possible. The original cavity was rolled and welded from $\frac{1}{4}$-inch-thick nickel sheet with the water-cooled sample holder in thermal contact with the roof. One heater element heated the walls. The cavity was replaced by a one-piece cavity with 1-inch-thick walls, and the sample holder was retained by pointed set screws. In addition, another heater was added around the base of the sample-holder entrance. The cavity was instrumented by putting calibrated thermocouples in wells in the cavity walls. Without the auxiliary heater the maximum temperature variation in the cavity was 12° C; with this heater it was reduced to less than 3° C. The same samples that had given reflectance values greater than 100 percent then yielded reasonable data.

Both the hohlraum and integrating sphere are now in use as continuously recording devices, and their data on paint samples agree to within reasonable accuracy with data obtained from point-by-point measurements performed by Professor Gier of UCLA. The values of reflectance in the overlap region between the integrating sphere and hohlraum match quite well. Samples of high, medium, and low reflectance have all been checked. For example, at 0.54 micron a certain paint showed a reflectance of 76 percent in the hohlraum and 77 percent when another sample of the same paint was measured by Dunkle and Gier at Berkeley. At the same wavelength an etched metal sample showed a reflectance of 49+ percent in the hohlraum and a reflectance of 50+ percent in the integrating sphere (relative to MgO).

EMITTANCE MEASUREMENTS AT THE LANGLEY CHEMICAL PHYSICS LABORATORY

By B. W. Lewis
Langley Research Center

Total hemispherical emittance measurements are made routinely for materials which may be heated by electrical resistance methods over the temperature range of 600° to 2,000° F by using a black-body reference method. This employs a conical black body and a thermopile detector with a calcium fluoride lens. (See refs. 1 to 4.) Emittance is obtained by measuring the radiant flux from the specimen strip and comparing it with the flux from an equal area of the black-body cone at the same temperature. The temperature measurements are made by use of thermocouples. It is planned to extend the temperature range of this type of measurement to temperatures above 2,000° F.

Another technique has been investigated for measuring emittance of materials not amenable to electrical heating or thermocouple attachment. This method uses a black-body-cavity furnace similar to that used in reference 5 to measure emittance of transparent materials such as glass. The method employs a heated black-body cavity in which the semicircular specimen is allowed to come to the equilibrium temperature of the cavity and then is rotated in front of a water-cooled viewing port where a sensitive thermistor detector alternately views the specimen surface and the black-body cavity. The ratio of the two readings gives the specimen emittance directly, for the temperature of the black body. The detector output is recorded on a fast Brown self-balancing potentiometer. The furnace is provided with a water-cooled blackened shutter which may be inserted behind the specimen to eliminate any transmitted black-body radiation if the specimen is transparent. This apparatus is capable of measuring total normal emittance over the temperature range of 1,000° to 2,000° F. Preliminary data for boron nitride specimens of two thicknesses are shown in figure 1 where total normal emittance is plotted against temperature for two experimental conditions: (1) black-body radiation incident on the back of the specimen and (2) no black-body radiation incident on the back of the specimen (that is, with a water-cooled shutter behind the specimen). The data appear to indicate a slight transparency for the thin specimen and emittances of from 0.7 to 0.8 over the temperature range.

Spectral emittance measurements are beginning to be made at the Chemical Physics Laboratory. A Perkin-Elmer Model 13-U Universal spectrophotometer with rock-salt prism and thermocouple detector is used for the measurements. The instrument has been slightly modified so that it views a small circular area of an electrically heated specimen strip or a black-body cone in the same position. The emission spectra are recorded by using a single-beam operation with the same slit widths and gain settings for the black body and specimen. Dry nitrogen is passed through the instrument to minimize the water vapor and carbon dioxide absorption bands. Emittances over the wavelength range of 1 to 15 microns have been measured for the temperature range of 900° to 1,800° F for etched oxidized Inconel and from 900° to 1,700° F for oxidized René 41 alloy. The preliminary spectral emittance curves for oxidized Inconel are shown in figure 2 and compared with similar curves at two temperatures for sandblasted oxidized Inconel and electropolished oxidized Inconel. (See ref. 6.) From this comparison and X-ray diffraction analysis of the three different surface-oxidation products it is evident that the surface composition, determined by previous treatment, is the most important factor controlling the spectral emittance of oxidized Inconel.

These preliminary data have been given to indicate the kinds of measurements made.

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VARIATION OF EMITTANCE OF BORON NITRIDE
OF TWO THICKNESSES WITH TEMPERATURE

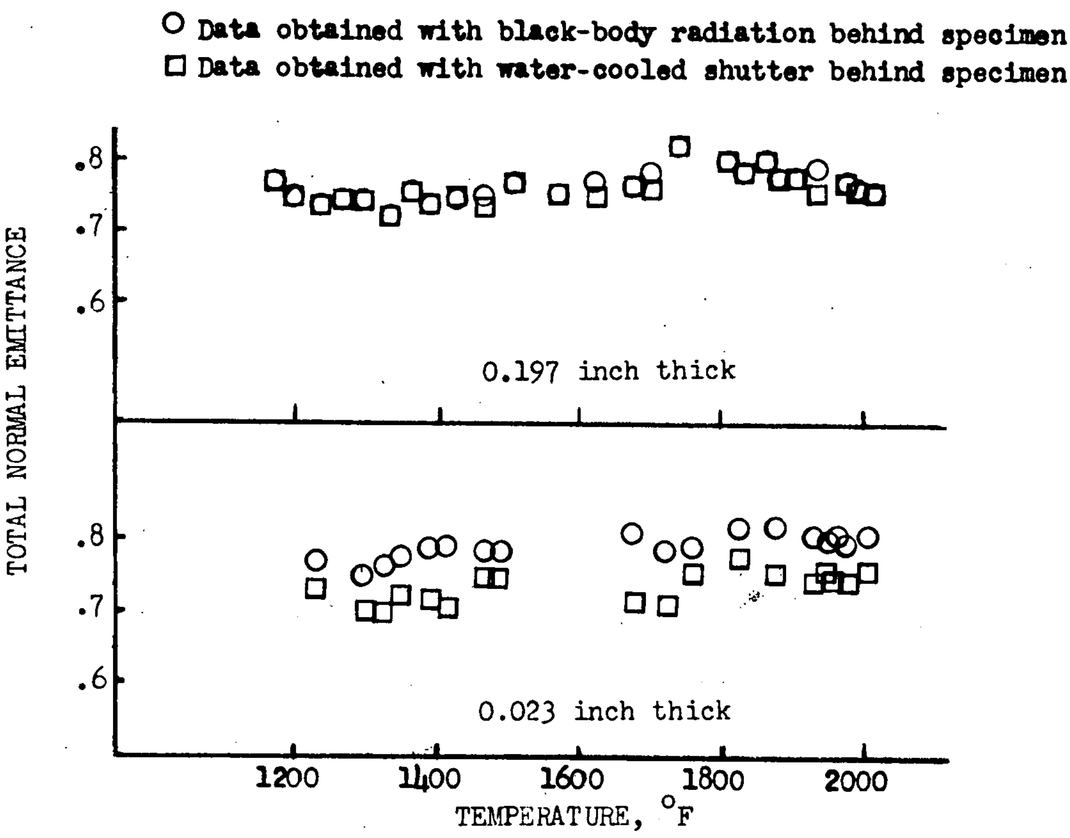

Figure 1

SPECTRAL EMITTANCE VS WAVELENGTH FOR OXIDIZED INCONEL

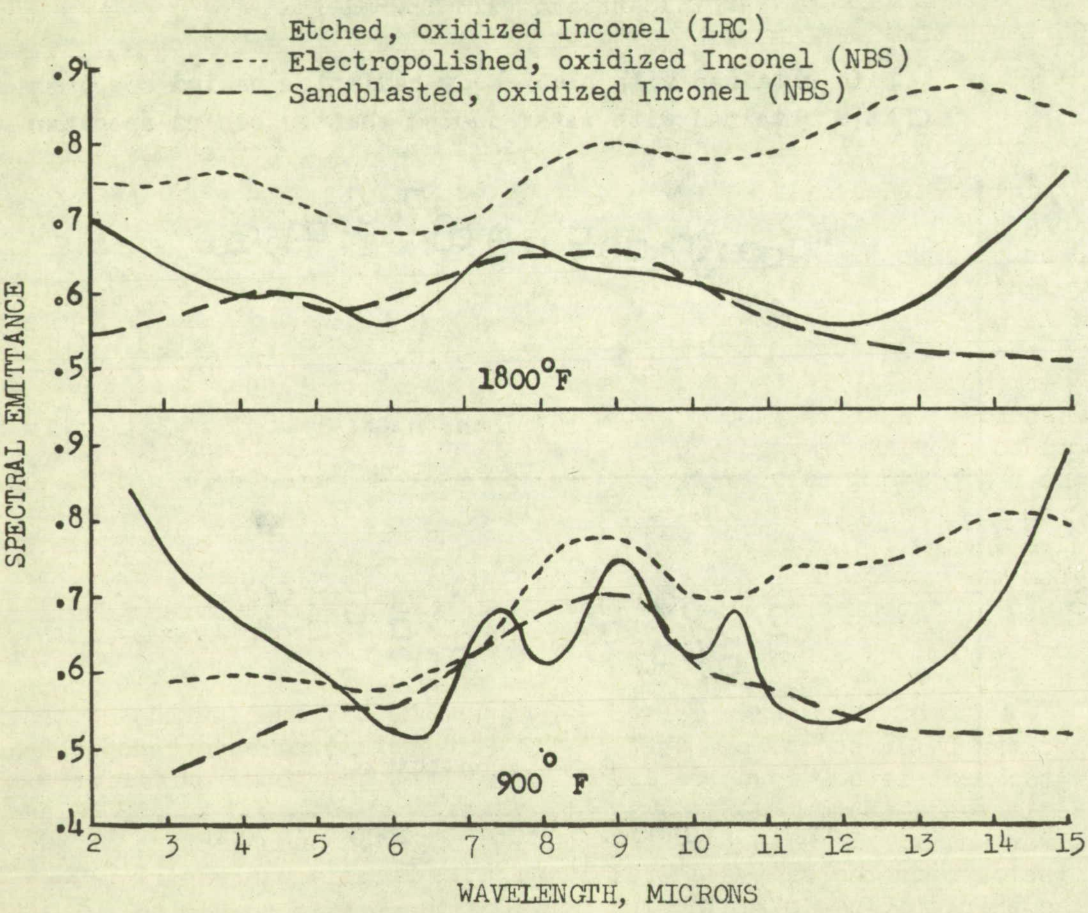

Figure 2

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EQUIPMENT FOR THERMAL-BALANCE STUDIES IN THE LANGLEY INSTRUMENT RESEARCH DIVISION

By Robert B. Spiers, Jr.
Langley Research Center

The Langley Research Center is in the process of obtaining emissivity measuring apparatus. This apparatus will be used to measure the solar absorptance and low-temperature emittance of space-vehicle materials. Two types of instrumentation will be used. One type will be the total hemispherical emissivity apparatus being constructed at this Center. The other will be double-beam photoelectric ratio-recording spectrophotometer apparatus which is being purchased.

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Construction of the total hemispherical emissivity apparatus is due to be completed in 2 months. It will consist of an evacuated stainless-steel cylindrical chamber 1 foot in diameter and 50 inches long. The chamber walls will be cooled with liquid nitrogen. The specimens to be tested will be wrapped around an electrically heated cylindrical Inconel tube 5/8 inch in diameter which is placed on the axis of the cooled chamber. The temperature of the middle 20 inches of the Inconel rod and the cold chamber walls will be measured with thermocouples. The electrical power input to the Inconel rod will be measured. Auxiliary heater coils will be on both ends of the Inconel rod to prevent temperature gradients over the 20-inch test length of the rod. Two metal reflectors will be placed approximately 4 inches from the ends of the chamber in order to give it an effectively infinite length. The emissivity of the specimen will then be calculated from the power input and the temperatures. The specimen temperature range is expected to be from 0° F to 200° F. The vacuum will be 10^{-6} mm Hg.

Most of the spectrophotometer apparatus has been purchased. A Beckman DK-1R spectrophotometer with an integrating-sphere reflectance attachment is now in operation. It measures the total or diffuse reflectance (compared with magnesium oxide) from 2,200 to 26,000 angstrom units. It also measures the scattered and nonscattered transmission over the same spectral range. Difficulty was experienced with the thin coatings of magnesium oxide on the reference plates. Thicker coatings eliminated the difficulty. A Perkin-Elmer Model 13-U spectrophotometer has been purchased and is in operation. It measures the nonscattered transmission from 2,000 to 150,000 angstrom units (0.2 micron to 15 microns). A hohlraum attachment is on order, to be used with this spectrophotometer for measuring the total or diffuse reflectance of samples from 2 microns to 15 microns. The sample temperature is specified to be approximately 120° F when the hohlraum operates from 752° to $2{,}012^\circ$ F. This hohlraum attachment is due to be delivered in November.

AN EXPERIMENT FOR DETERMINING THE STABILITY OF SURFACE COATINGS IN SPACE FLIGHT

By Carr B. Neel
Ames Research Center

Because of the difficulties involved in simulating the adverse effects of the space environment with ground-based facilities for the purpose of investigating surface coatings, the possibility of testing the stability of surfaces during flight in the actual space environment must be considered. A simple, straightforward method for measuring any changes in the thermal characteristics of a surface resulting from conditions in space flight is to measure shifts in temperature of the surface during exposure to sunlight, since any change in the ratio of solar absorptance to surface emittance α/ϵ will be reflected in a corresponding change in surface temperature. This paper describes an experiment of this type which is planned for inclusion on a number of satellites.

An obvious requirement for the experiment is that the test surface be thermally isolated, since extraneous heat losses must be minimized to provide an accurate indication of change in α/ϵ.

A second requirement is that the test surface have a rapid response to changes in heat flux. It is desirable that the surface reach thermal equilibrium during its daytime and nighttime passages around the earth. This facilitates reduction of the data to obtain α/ϵ. If the thermal lag of the surface is so large that equilibrium is not reached, reduction of the data becomes complicated, since a transient thermal analysis would be involved.

The design which was established to minimize the heat losses and thermal lag of the test surfaces is illustrated in figure 1. The test surfaces are mounted on three small Kel-F supports to minimize the conduction path. Radiant heat losses to the mounting cup are minimized by the use of four radiation shields. All interior surfaces are polished and have an evaporated gold finish for further reducing the radiant heat exchange. Thermal lag is minimized by using a thin base disk and thin radiation shields.

Surface temperature is measured by means of a thermistor soldered to the underside of the test surface.

Figure 2 shows the radiation sensors mounted in a cluster of six to permit testing several different surface finishes simultaneously. A seventh surface will serve as a reference for the other surfaces. The reference surface is designed to maintain constant radiation properties in space flight. Comparisons of the temperatures of the test surfaces with that of the reference surface will provide a basis for evaluating changes in the thermal characteristics of the test surfaces.

The design of the reference surface is indicated in figure 2. The surface is composed of razor blades stacked together to form a large number of notches, which cause multiple reflections and eventual absorption of most of the incident radiation. Because of the large number of reflections, any change in the emittance of the individual surfaces of the notches will have only a very small effect on the overall emittance or absorptance of the reference surface. Since the notched surface is somewhat directional at large angles of incidence, the rows of razor blades are arranged in a hexagonal pattern to minimize the directional effect. Due to its high absorptance, the reference surface acts as a black body, and hence is useful for evaluating the radiant energy incident upon the test surfaces.

In order to correct for heat exchanges between the test surfaces and the sensor mounts, the temperature of the base plate is measured by means of a thermistor.

Before installation of the test equipment on a satellite, it is necessary to determine the heat-loss and thermal-lag characteristics of the sensors. In addition, the absorptance and emittance of each of the test surfaces should be known. These properties will be measured by various techniques in a small vacuum chamber.

Temperatures of the sensors will be telemetered to ground periodically during the flight. To conserve telemetry channels, all temperatures will be transmitted on one channel by means of a solid-state commutator switch. The switch contains ten points. Seven of these points are from the thermistors measuring the temperatures of the test surfaces, and one is from the thermistor measuring the base temperature of the sensor mounts for heat-balance corrections. The remaining two points are from standard calibrating resistances, representing two levels of thermistor temperature.

CONSTRUCTION OF RADIATION SENSORS

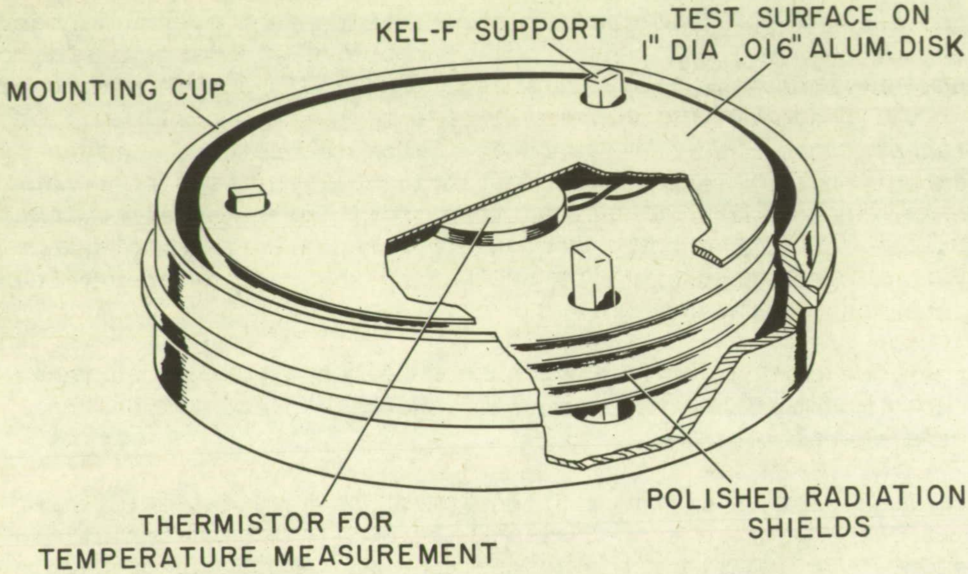

Figure 1

MOUNTING OF RADIATION SENSORS

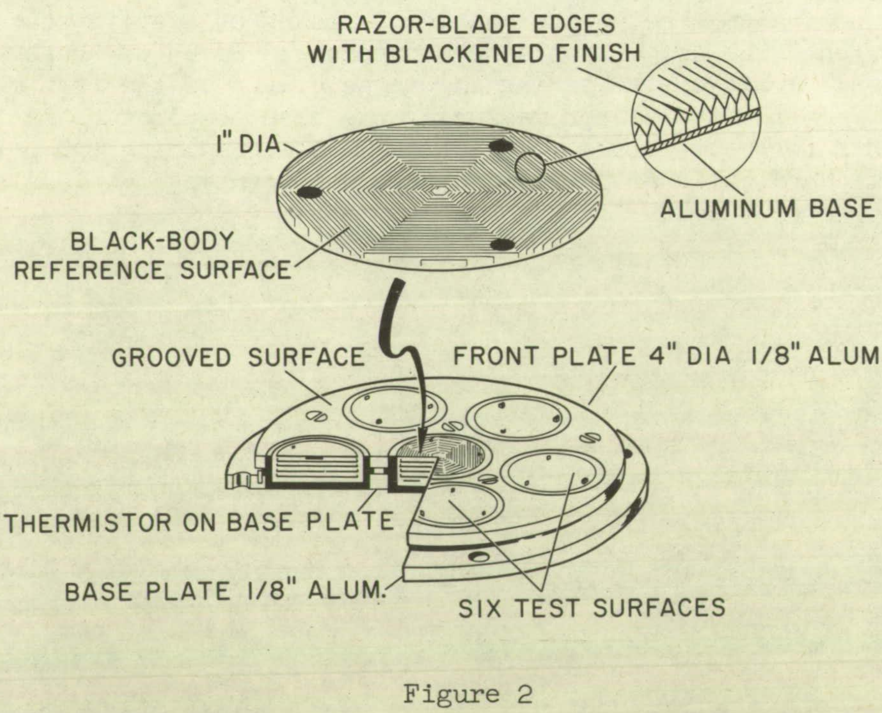

Figure 2

THERMAL RADIATION MEASUREMENTS

By Ralph McDonough
Baird-Atomic, Inc.

Since 1957 the Marshall Space Flight Center has supported a program of thermal radiation measurements for application to satellite temperature control. The work at Baird-Atomic has consisted of providing instruments and making measurements on samples as supplied. Four instruments are currently in operation. These are:

(1) Hemispherical spectral reflectance in the range from 0.4 to 0.7 micron at room temperature is measured by the Color Measurements Laboratory of M.I.T. on a General Electric recording spectrophotometer with integrating-sphere attachment.

(2) Hemispherical spectral reflectance in the range from 1.0 to 3.5 microns at room temperature is measured with an integrating sphere of the type described by Preston (ref. 1). This instrument is a special case of a partial sphere in which the illumination in the sphere is related to the reflectance of the material placed at a large test aperture, which is not directly illuminated. In this case both incident and reflected radiation are hemispherical, in contrast to the more common spheres in which the incident radiation is approximately collimated and normal to the sample surface. A set of interference filters is used for point-by-point wavelength selection with this instrument.

(3) Normal spectral emissivity in the range from 4.0 to 13.5 microns at 360° K is measured by a single-beam, point-by-point comparison of the signals from a sample and a black body, both heated to 360° K. The salt-prism monochromator and bolometer detector of a Baird-Atomic infrared spectrophotometer are used for these measurements. Corrections for room radiation, both that reflected from the sample and that received by the instrument when the chopper blade is in front of the slit, must, of course, be made when working with such low energy levels.

(4) Normal total emissivity at 275° K is determined by comparing the energy received from a sample and a black body, both cooled with ice water. This apparatus is operated in a bell jar at a pressure of approximately 10^{-2} mm to prevent collection of moisture on the samples and also to permit the bolometer detector to be used without any window, thereby increasing its spectral bandwidth. Again, emissivity is related to the measured signals by simple radiation calculations that include the effects of room radiation.

Detailed descriptions of these instruments may be found in ABMA Final Technical Report dated July 29, 1960 (Contract DA-19-020-ORD-4474).

Over 200 samples of various surfaces including metals with different surface finishes, paints, flame-sprayed coatings, plastics, and solar cells have been measured with each of these instruments.

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EMISSIVITY MEASUREMENTS AT THE RADIOMETRY SECTION OF THE NATIONAL BUREAU OF STANDARDS

By Arthur Maki
National Bureau of Standards

The Radiometry Section of the National Bureau of Standards is currently engaged in a project which involves the measurement of the normal spectral emissivity of gold, aluminum, and platinum in the infrared region (3 microns to 14 microns). These measurements are being made over a range of temperatures from 500° K to 750° K for aluminum and to $1,000^{\circ}$ K for gold. The maximum temperature for the platinum measurements is $1,400^{\circ}$ K.

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It is hoped that this work will indicate the manner in which the emissivity changes with temperature so that it will be possible to infer emissivity values beyond the range in which the measurements are being made. For example, the emissivity values obtained for aluminum at various temperatures when plotted against temperature form a smooth curve which may be connected to the room-temperature values of Bennett and Koehler (ref. 1). This curve may be used for interpolation of emissivity values at intermediate temperatures or may be used to extrapolate to slightly higher or lower temperatures. These measurements may also be used to check the validity of various theoretical relationships that predict the emissivity from other facts regarding the metal (refs. 2 and 3).

The apparatus is essentially the same as that described in reference 4 with the following modifications. A double-pass monochromator is now being used in order to eliminate stray radiation which can introduce sizable errors at the low energy levels involved in the lower temperature measurements. The small energy available at the longer wavelengths likewise requires caution lest stray radiation cause incorrect readings. As a further precaution some measurements are made while using a polyethylene black filter which quite effectively removes any stray shortwave radiation.

In order to eliminate errors due to any nonlinearity of the detector and amplifier system, the practice has been adopted of comparing the energy of a black body at a relatively low temperature with the samples at much higher temperatures. The temperatures of the two are varied so that both give equal or nearly equal pen deflections on the recorder which is used for the data output. This procedure requires a point-by-point plot of the data, which is a very tedious procedure. Happily, however, the lack of sharp peaks or discontinuities in the emissivity curve obviates the necessity of having small wavelength intervals. It has been found that points taken 1 micron apart are sufficient for most of the wavelength region covered in this work.

The principal source of error in this work derives from the temperature measurement of the samples. The samples consist of strip lamps which are operated without a glass envelope and therefore are exposed to the air. Thus, convection currents cause fluctuations in temperature and create temperature gradients along the filaments. Errors due to temperature fluctuations are not serious; on the other hand, the temperature gradients can be an important cause of error. The temperature of the specimen is determined by means of a thermocouple attached to one face of the thin metal strip. Great care must be exercised to be certain that the spectrometer is focused on the specimen surface immediately opposite the point of attachment of the thermocouple. For specimen temperatures of 600° K an error of 10° K in the temperature measurement will cause an error of about 10 percent in the measured emissivity value at 4 microns so that it is obvious that the temperature measurement must be as accurate as possible.

Since infrared radiation is not very sensitive to small surface irregularities, the surfaces of the specimens were not specially prepared. All the specimens have a relatively smooth appearance to the naked eye, which is quite sufficient to insure that the surfaces will be optically smooth for the longer wave infrared radiation.

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4. Maki, Arthur G., Stair, Ralph, and Johnston, Russell G.: Apparatus for the Measurement of the Normal Spectral Emissivity in the Infrared. Paper 64C2-29, Jour. Res. of Nat. Bur. Standards, vol. 64C, no. 2, Apr.-June 1960, pp. 99-102.

REVIEW OF RADIATION RESEARCH AT THE LOCKHEED MISSILES AND SPACE DIVISION

By R. E. Gaumer
Lockheed Aircraft Corporation

The discussion was introduced by a general description of the work being performed at Lockheed Missiles and Space Division (LMSD), with particular emphasis upon the necessity for development and evaluation of temperature-control surfaces to meet the requirements of spacecraft missions. A summary of the existing theory applicable to thermal radiation was given, and it was concluded that the theory is of very limited value in the solution of the practical problems of temperature control of spacecraft. A discussion of a comprehensive literature survey which is operative at LMSD was given and particular attention was called to a review article by G. A. W. Rutgers. (See ref. 1.)

The effects of the thermal radiation parameters of an exterior satellite surface on the orbital time-average equilibrium temperature of the surface were discussed. Sources of radiant energy were reviewed, including the accuracy with which albedo, earthshine, and insolation are known. It was pointed out that the most significant parameter affecting thermal control is the α/ϵ ratio, which is the ratio between the solar energy absorptance and the infrared energy emittance. The parameters which significantly affect the α/ϵ ratio of a surface were listed and discussed. These parameters are: surface temperature, the spectral distribution of incident energy, surface roughness and cleanliness, the angle of incidence of radiation, and the chemical and mechanical surface condition. A few possible thermostatic surfaces were discussed, such as ferromagnetic substances operating near their Curie temperature and phototropic substances.

Passive thermal control was discussed in terms of four basic surface types, and a comparison was drawn between idealized and presently existing surfaces. The four basic surface types are solar reflectors, solar absorbers, flat reflectors, and flat absorbers. White paint is a common example of a solar reflector with $\alpha/\epsilon \approx 0.28$. A sapphire mirror is a potential solar reflector with a lower α/ϵ ratio. Common solar absorbers are gold or aluminum in a highly polished condition with $\alpha/\epsilon \approx 0.3/0.03 = 10.0$. Flat reflectors such as an aluminum-pigmented silicone paint are characterized by $\alpha/\epsilon \approx 0.3/0.3 = 1.0$. Black paints and black anodized surfaces are representative flat absorbers characterized by $\alpha/\epsilon \approx 0.9/0.9 = 1.0$.

The method by which the four basic thermal-control surfaces are combined into a mosaic for temperature control was outlined, including recommended analytical approaches. The possible usage of such specialized devices as interference filters, vacuum-deposited films, and Tabor coatings was briefly discussed.

A major portion of the talk was devoted to a discussion of the factors affecting choice of materials for thermal-control purposes, the primary criteria for choice of materials being work stability and reproducibility. The effects of the environment on thermal-control surfaces were divided into two phases, a prelaunch and a postlaunch phase. The significant parameters in the prelaunch environment are oxidation, corrosion, contamination, and fingerprints. The high temperatures and aerodynamic shear associated with ascent heating were discussed in terms of their effect upon such thermal-control materials as paints and other organic coatings.

It was pointed out that the most significant aspects of the orbital phase, insofar as temperature-control surfaces are concerned, are ultraviolet radiation and temperature extremes. Other space environment characteristics which might be expected to have significant effects upon materials in space are micrometeoroid erosion, sputtering by atmospheric particles and by protons, high-energy radiation of various types, and, conceivably, the effects of a charge accumulation upon a vehicle.

The reentry phase is similar to the ascent phase in its effect upon materials, the most notable difference being the greater magnitude of heat flux, which leads to the necessity for ablative materials.

The nature of the thermal-radiation experimental work in progress at LMSD was discussed, including the basic instruments for the determination of emissivity such as the Cary integrating sphere, the hohlraum and associated double-beam spectrophotometers, and various calorimetric devices. A device for the direct determination of the α/ϵ ratio was outlined in detail.

Contemplated work in the near future includes the determination of the angular dependency of emissivity and absorptivity, the emissivity of materials at high temperatures, the effects of surface roughness on thermal-radiation interchange, and the radiation characteristics of pure metals.

Some 500 samples have been measured to date and their emissivities have been determined in the wavelength range of 0.2 to 20 microns. Environmental simulation apparatus which is operative at the present time includes ascent heating simulation, temperature extreme and temperature cycling simulation, simultaneous ultraviolet and high-vacuum exposures, and adhesion tests under conditions of aerodynamic shear. The development program underway at LMSD was outlined and the goals of the program stated. Materials development efforts are in progress in the following areas:

Organic paints
Inorganic paints
Pigment concentration and variation

Anodizing processes
Protective coatings
Metallic vacuum deposition
Electroplating
Metal foils
High-temperature adhesives
Chemical cleaning processes
Polishing processes
Stable oxides
Refractory metals

A report summarizing this work is presently available for internal use. This report is also being prepared as an external document and will soon be available to interested persons. If copies are desired, please notify the author, Dr. R. E. Gaumer, Research Specialist, Lockheed Aircraft Corporation, Missiles and Space Division, Thermodynamics, Department 53-15, Building 104, Sunnyvale, California.

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DEVELOPMENT OF A SOLAR SIMULATOR

By John L. Pollack
Lewis Research Center

This paper describes the development of a solar simulator in use at the Lewis Research Center. The simulator is intended to duplicate the total radiance and the spectral distribution of radiation of the sun in the vicinity of the earth and before it is modified by passing through the atmosphere.

Figure 1 shows the spectral energy curves related to the sun. Above the atmosphere about 8 percent of the radiation is in the ultraviolet region (below 0.4 micron), 41 percent in the visible region (between 0.4 and 0.7 micron), and 50 percent in the infrared region (over 0.7 micron).

The solar constant is the irradiation per unit area normal to the direction of the sun, outside the atmosphere, at the mean earth-sun distance. The most generally accepted value for the solar constant is 1,400 watts/meter^2. The value is considered accurate to within 3 to 5 percent. By way of comparison, the value of the solar constant at the earth's surface is about 2/3 of that for an air mass of zero, or about 930 watts/meter^2.

One other characteristic of sunlight should be noted, namely, that at 93 million miles its rays are essentially parallel and for the small distances involved in earth-satellite orbits inverse-square-law effects are negligible. Any sun substitute should have radiation that does not vary along the light path.

Theoretically there are many combinations of light sources and filters which could be made to produce the sun's spectrum. From a practical engineering standpoint the source should meet the following conditions:

1. The source should be relatively compact and adaptable to conventional optical systems in order to permit control of total radiation and of uniformity of intensity of radiation and also to permit collimation in order to minimize inverse-square-law effects on the test specimen.

2. It must have a reasonably continuous spectral output.

3. It must be capable of use for 30-minute periods.

4. If possible, the source should be located outside the environmental chamber so that the radiation enters the chamber through a suitable window. Placing the source outside the chamber eliminates all the complex problems of heat dissipation from the source, alinement and adjustment of optics from outside the chamber, and reradiation to the specimen from the optical components.

The high-intensity carbon-arc lamp seems most nearly to meet these specifications. Reflection optics or a quartz transmission system will provide for nearly complete spectral transmission between 0.3 and 3 microns; this region contains 97 percent of the sun's radiation.

The spectral distribution of the carbon arc varies with current density, electrode type, and condenser optics but it can be controlled to approximate closely the solar distribution.

Figure 2 compares the spectral radiation of the sun with that of the carbon arc. The dashed curve is the sun's distribution and the upper solid curve is the distribution from an existing high-intensity carbon-arc lamp in use with an environmental chamber. These curves were obtained by using a Model 12A Perkin-Elmer infrared spectrometer with a lithium fluoride prism, modified for use as a spectral radiometer. The curve represents the radiation after it has passed through the optics, and under conditions such that its total intensity is equal to that of sunlight. The sketch at the right of the curve shows the Beck type arc that was used. It shows the negative electrode and the cored positive electrode, along with tail flame emanating from the cup-shaped positive electrode. The total spectrum is the sum of the spectra from the electrode and the tail flame. Also plotted are the separate spectral distributions of each of these regions. The electrode radiation is produced by the vaporized carbon at 3,900° K and the volatilized core material - generally rare earths of the cerium group. The tail-flame radiation is produced almost entirely by the rare earths and is rich in the ultraviolet and near-visible portions of the spectrum.

The optical system in use now utilizes quartz lenses in a system that is similar to the condenser projection optics in a commercial slide projector. The condensing lenses form an image of the arc in the plane of the projection lens. Specially shaped masks in this plane can control the relative amount of radiation from the electrode (zone 1) and from the tail flame (zone 2) so that their sum will equal the solar distribution.

Figure 3 is a picture of the space and solar simulator used to determine the equilibrium temperature of the skin of a micrometeorite package. Starting from the right are the Beck type carbon arc, the boom that holds the lens system, a right-angle mirror, an inclined quartz window reflecting the light to a radiation detector, and finally the entrance to the space chamber. The total radiation in the target plane is monitored and controlled by a light-sensitive device that receives reflected radiation from the window in the optical path. The sensor is located at the mirror image of the target. Any change in energy density caused by arc instability, improper electrode feed rate, fluctuation of lamp power-supply output, or even dirt on the optics causes this sensor to operate a controller mechanically coupled to the condensing lens. Motion of the lens toward or away from the arc increases or decreases the amount of light intercepted by the lens and maintains constant intensity in the target plane.

The system is initially calibrated for an intensity of 1 solar constant by use of a total-radiation-pyrometer thermopile. An output in millivolts is obtained from the pyrometer by pointing it at the sun on a clear day at noon. Suitable empirical corrections give the output for 1 solar constant above the atmosphere. The thermopile is then placed in the plane of the target and the arc intensity is adjusted to obtain the same output. A second independent calibration was obtained by using a black patch of known $\frac{a_{solar}}{e}$ ratio in the space simulator. Both methods agreed within 2 percent.

The solar simulator is thus a working system that gives a solar constant accurate to 5 percent with 1-percent regulation for periods up to 45 minutes. Uniformity of illumination over the entire area, which in this chamber is 6 inches in diameter, is within 3 percent. Total and spectral radiation outputs from lamps consuming up to 22.5 kilowatts in the arc have been measured. Results indicate that for solar-simulation sources that use this type of system about 4 watts of electrical power into the arc is required per square centimeter for 1 solar constant of radiation (using f/1 on the first condenser). As an example, an environmental chamber with a 3-foot-diameter test section will require an electrical input of about 25 kilowatts.

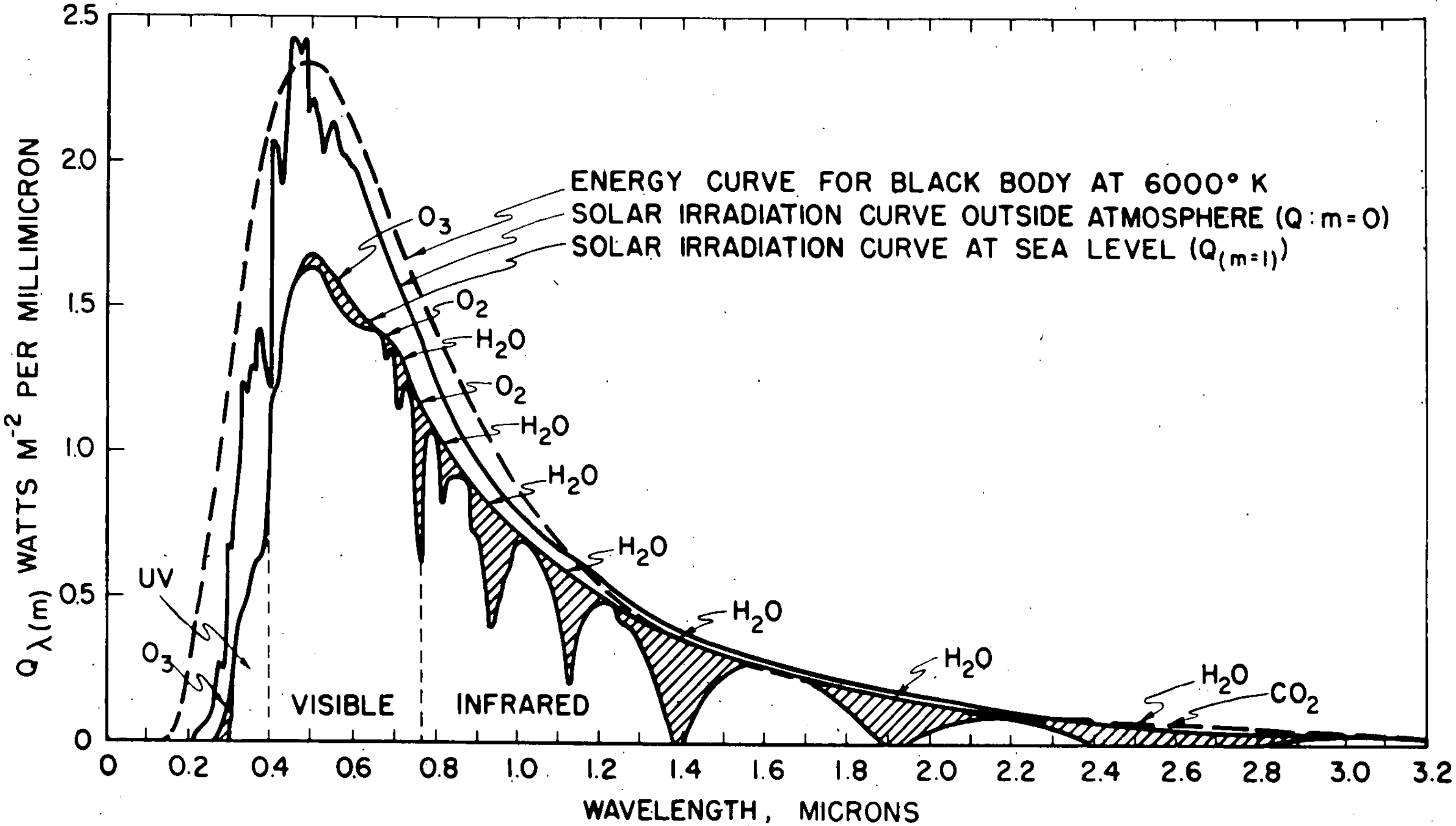

Figure 1.- Spectral energy curves related to the sun.
(From Handbook of Geophysics, revised edition, U.S. Air Force. The Macmillan Co., 1960.)

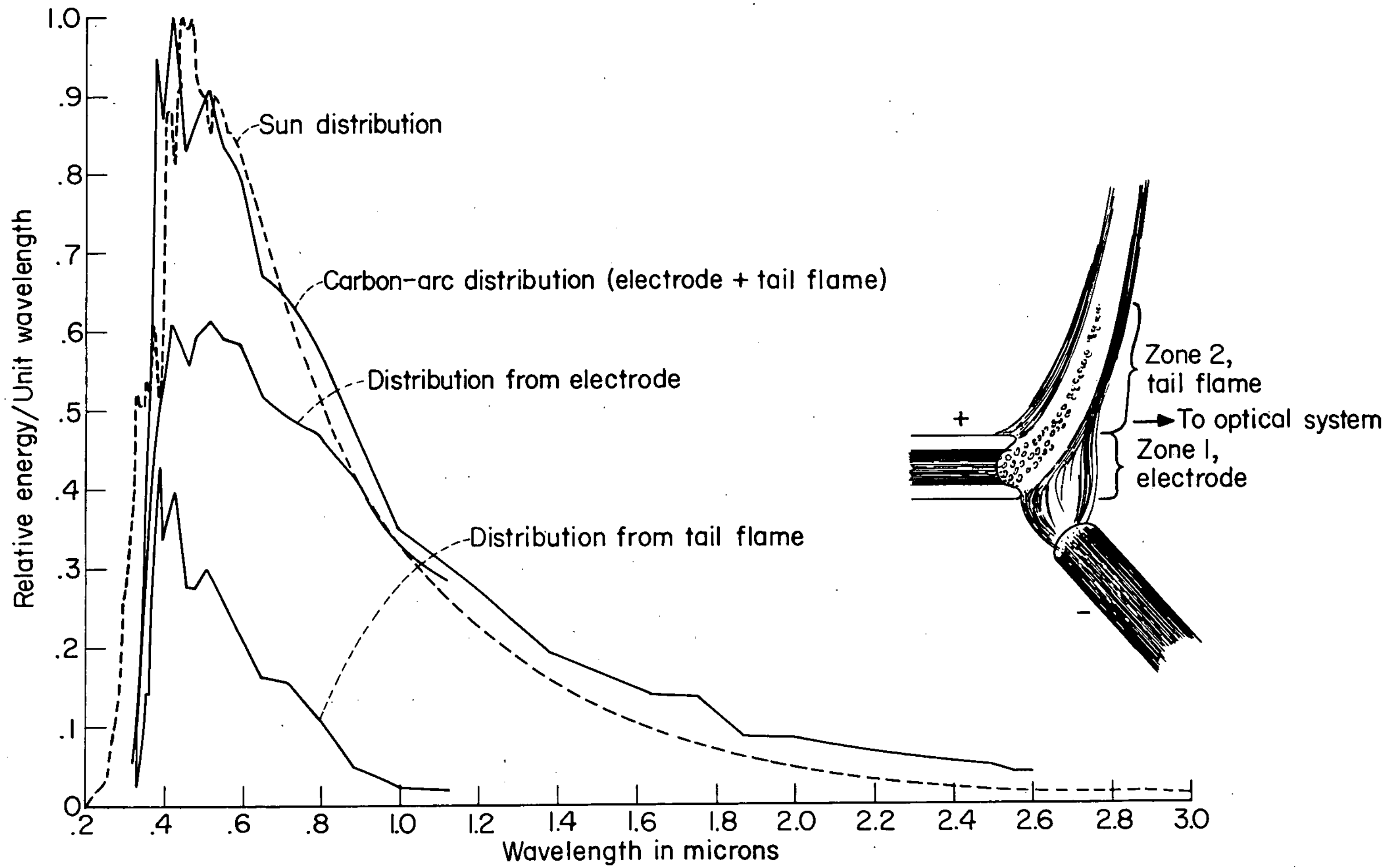

Figure 2.- Radiation of sun and carbon arc.

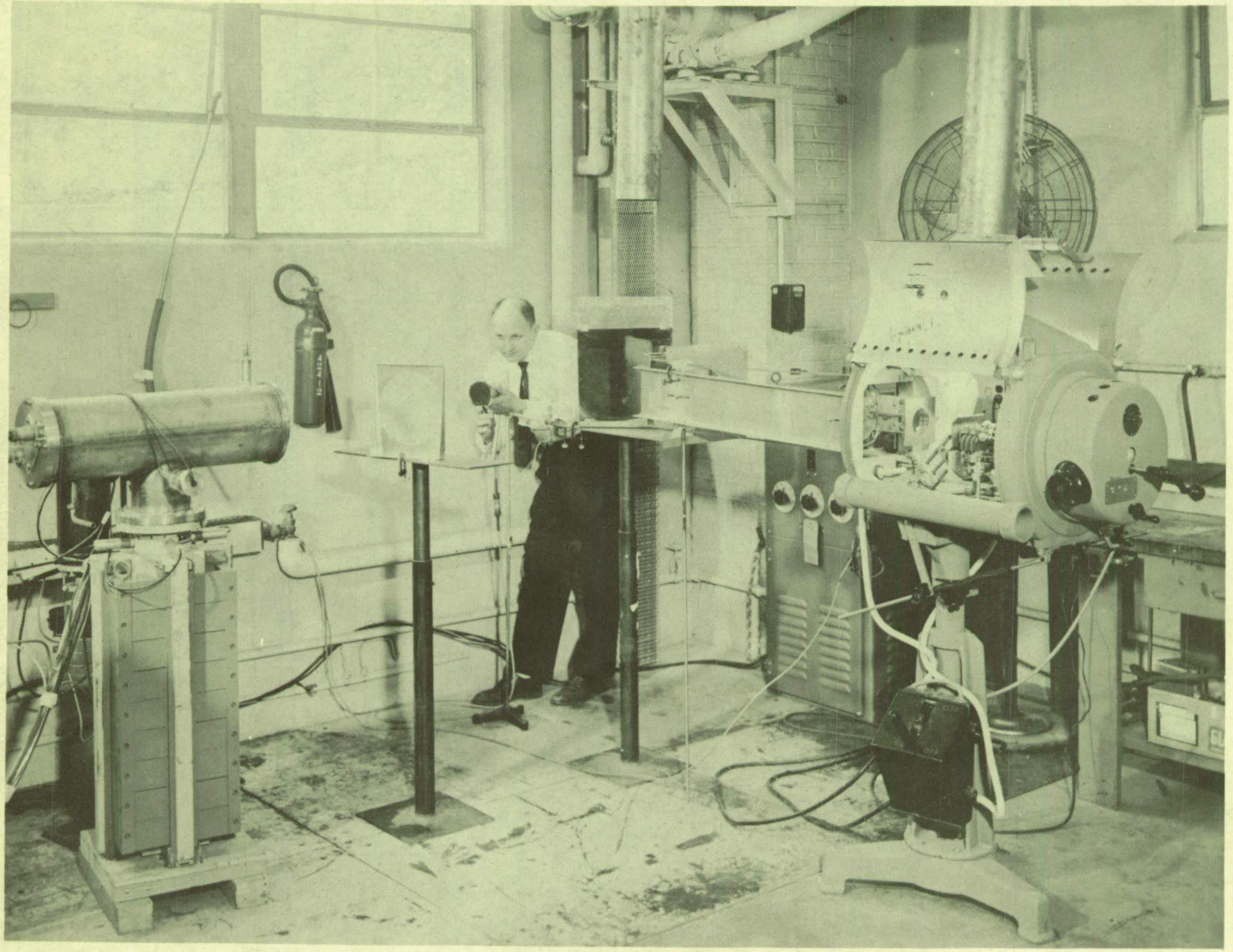

Figure 3.- Arrangement of space and solar simulator. C-53591

SPACE-ENVIRONMENT SIMULATION AT THE LEWIS RESEARCH CENTER

By Lester D. Nichols
Lewis Research Center

A knowledge of the environmental conditions in space and an understanding of the possible effect on objects in that environment is necessary in order to make a full-scale, systematic exploration of outer space. Once these conditions have been determined, the understanding of the behavior of objects in space may be gained by simulating the environment in the laboratory and conducting experiments in the simulated environment. Among the conditions in space which have been determined and may be simulated are the pressure, the temperature, and the radiant energy. Quantitatively, simulating these conditions would require a pressure as low as 10^{-16} mm Hg, a sink temperature of approximately 3° to 4° K, and radiation similar to that from a $6,000^{\circ}$ K black body.

The features of the space environment directly affect the thermal balance of the spacecraft. The temperature of the object is that temperature which provides equilibrium between the energy absorbed by the object and the energy emitted by the object. Hence, a knowledge of the absorption and emission characteristics for materials under space environmental conditions must be obtained. High-speed particles may impact and erode the material surfaces and in that way alter the radiation properties of the materials and finally change the equilibrium temperature. Also, since most of the radiant energy (that from the sun) is from one direction, the heat transfer within the body will provide surface-temperature variations which depend upon the shape and orientation of the object and the properties of the material. These variations may be effectively studied in the simulated environment. Finally, because of the vacuum the heat transferred to and from the object must be accomplished solely by radiation. Thus, if heat-cycle power-generation equipment is to be used, the necessary rejection of heat must be accomplished by radiation from waste heat radiators whose design must be investigated in order to determine optimum configurations.

At the Lewis Research Center, some of the features of the space environment will be approximated in a space-environment-simulation chamber 3 feet in height by $1\frac{1}{2}$ feet in diameter. Within this tank a pressure of 10^{-10} mm Hg will be maintained, the tank walls are to be cooled to 4.2° K (the normal boiling point of liquid helium), and a source of solar radiation (having an intensity equal to that at the mean earth-sun distance) capable of irradiating a 12-inch-diameter disk will be provided. The walls of the tank have an absorptivity of approximately 0.9 to reduce reflection of radiation back to the object from the wall.

This space simulator is shown in cutaway view on figure 1. The inner tank (shown as the darkest area on the figure) is surrounded by a vacuum, a liquid nitrogen shield, and an outer vacuum. Aluminum-foil radiation shields are shown in the evacuated regions. The tank was provided with a liquid helium jacket both to maintain the space sink temperature of 4.2° K, and to serve as a cryogenic pump to reduce the pressure within the tank to 10^{-10} mm Hg after roughing and diffusion pumps have brought the pressure down to 10^{-6} mm Hg.

A program of studies to be made in the space simulator has been determined. A large number of tests have been considered which will determine the effect of the solar radiation on the average temperature and on the temperature variations of space vehicles. The effect of the emissivity and absorptivity of the material on the average temperature and the effects of geometry and configuration on the temperature distribution and heat transfer will be determined experimentally. Particle impaction effects on the radiation properties of material will also be studied. Because the walls are at liquid helium temperatures, it is possible to study the problem of storing cryogenic fuels (liquid hydrogen, for example) in space.

Another group of tests is designed to determine the effect of the environment in transferring heat to and from the vehicles. Included in this category is a study of temperature-control devices (such as shields), collectors of solar energy for direct energy conversion devices, and waste heat radiators necessary in maintaining continuous power generation from heat-cycle devices. The space-environment simulator provides an opportunity to investigate complex configurations which cannot be handled analytically.

Another area in which the space-environment chamber may be utilized is the development of equipment which will operate in space. It will be necessary to test this equipment under "actual" conditions in order to determine the proper design and materials that will operate satisfactorily under these conditions. Such items as vacuum seals, bearings, and motors must be developed and as part of this development program they may be tested in the space-environment simulator. Other areas in which this chamber may be used include radiation pressure measurements (associated with inflatable satellites) and heat transfer to and from superconductivity devices operating in a vacuum (motors and gyroscopes).

This tank is a pilot model for a larger facility which is being built. The larger space-environment simulator will be approximately 9 feet in height by 6 feet in diameter having a solar source capable of irradiating a disk nearly 3 feet in diameter. The remaining features of the tank are very similar to those of the pilot model. The larger tank provides facilities for the increased research program outlined above.

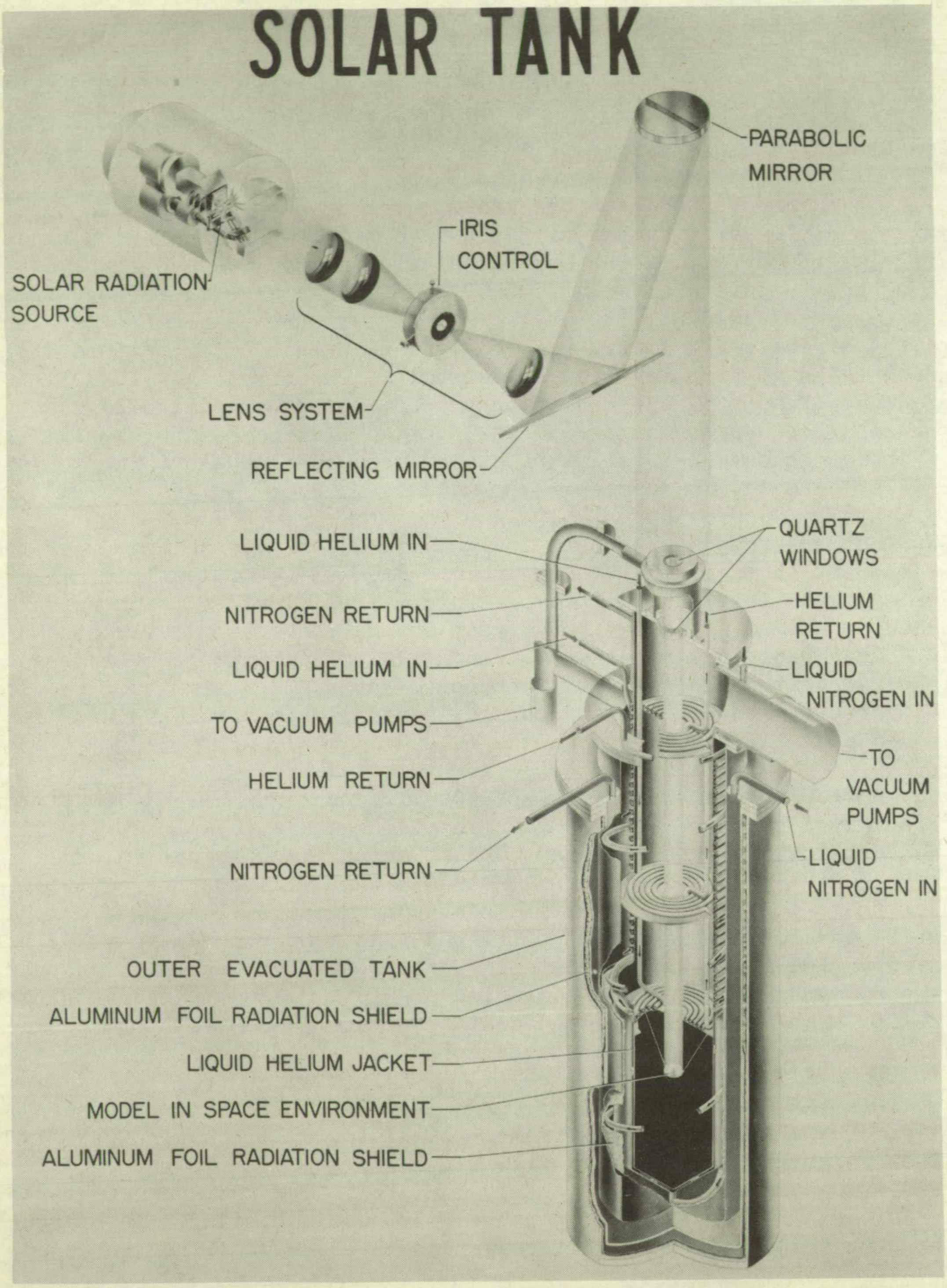

Figure 1 C-51438

RANGER SPACECRAFT

By T. O. Thostesen
Jet Propulsion Laboratory

The thermal-control philosophy of the spacecraft currently under development by the Jet Propulsion Laboratory is design by passive means to maintain all components within the tolerances specified by cognizant engineers. Due to the complexity of the configurations, calculations are, of necessity, fairly generalized and final design is based upon tests in an environmental chamber.

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The Ranger series spacecraft is designed with a basic structure which is common to all models, with additional hardware to suit the individual mission. This basic structure of Rangers A-1 and A-2 is seen in figure 1 as the hexagonal instrument section, the erectable solar panels, the movable antenna, and the omniantenna. The Ranger A-1 and A-2 configuration is for engineering tests and space exploration, with the scientific instrumentation isolation requirement dictating the spread-out design. The spacecraft stands 12 feet high, weighs 700 to 800 pounds, and has an internal power of 150 watts. Rangers A-3, A-4, and A-5 are designed to rough land a capsule on the moon. For these, a capsule and retrorocket replace the scientific instruments, occupying the space inside the tower structure.

The spacecraft must survive many environments. Chronologically, they are:

1. Folded configuration inside an aerodynamic shroud on the pad

2. Thermal flux from shroud aerodynamically heated during boost phase

3. Coasting up to 30 minutes attached to Agena stage after booster and shroud are separated

4. Agena stage burning

5. Coasting and tumbling after separation from Agena until it passes from earth's shadow

6. Upon reaching sunlight, panels open and begin sun acquisition

7. Antenna seeks earth after spacecraft locks onto sun

8. Space phase - "steady state" with vehicle's vertical axis locked on sun, communicating with earth

The philosophy is to design for the sun-acquired mode, making allowances for the transient conditions.

In order to test the design, an environmental facility is being completed at JPL. An existing 6-foot-diameter by 8-foot-tall cylindrical vertical vacuum tank has been modified by the addition of a liquid nitrogen liner and solar simulation. Depending upon the amount of outgassing, the vacuum is in the range of 10^{-5} to 10^{-6} mm Hg. The hot-wall empty pressure is 6×10^{-6} mm. A 60-inch-diameter illuminated area 38 inches above the cold floor is achieved with a solar intensity variable up to a maximum equal to that at Venus by means of two carbon arc lamps. Four lamps are mounted on top of the chamber and each focuses a diverging beam of light through an individual quartz port. The lamps are used in opposing pairs, the pairs being interchanged every 20 minutes as the electrodes expire.

The chamber presents many problems. One of the primary problems is that the spacecraft is taller than the chamber. Therefore, the spacecraft will be tested in three sections: the hexagonal instrument section, the scientific instrument platform seen just above the hexagonal instrument section, and the group comprising the omniantenna and the magnetometer.

The second primary problem is the solar simulation. The axes of the opposing lamps are 28° apart, and the light diverges 15° from the axis of each lamp. The illumination pattern from the lamps is nonuniform, and the variation of intensity of the solar simulation with vertical location is 3 percent per inch at the design plane of illumination.

The complex shadow pattern of the forward structure on the lower areas is almost impossible to simulate in the nonparallel light from two sources. Therefore, allowances will have to be made in the interpretation and interpolation of the results of the tests in the environmental facility.

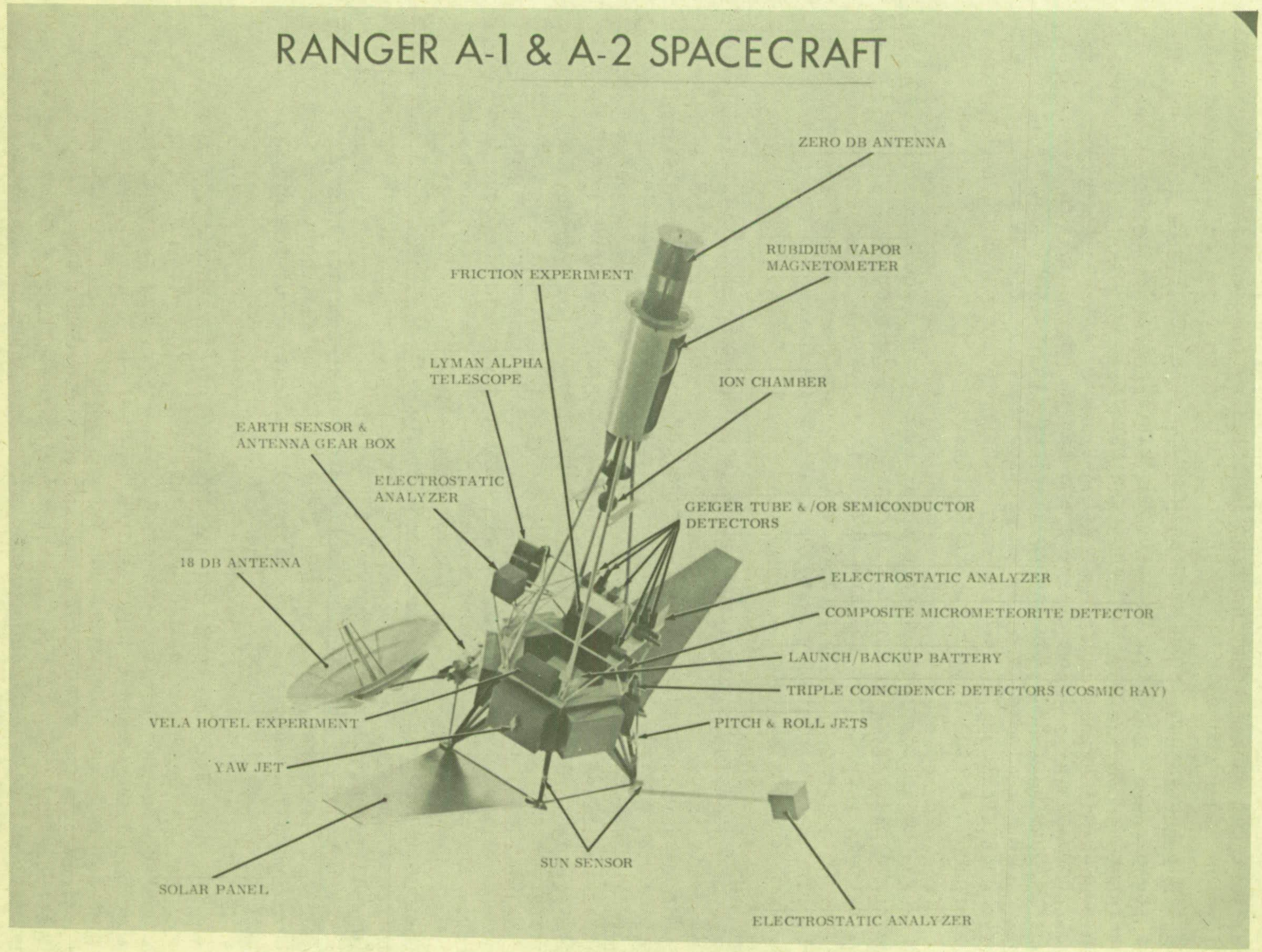

Figure 1 L-60-6914

A MANNED ORBITAL SPACE LABORATORY

By Emanuel Schnitzer
Langley Research Center

A study is under way of a manned orbital space laboratory, some of the purposes of which would be to determine man's adaptability to space and to study structures and systems in space before committing manned spacecraft to long-range missions. It uses an inflatable torus as laboratory and living quarters and has an erectable solar collector as the source of heat for the power plant (fig. 1). The station rotates six times per minute in order to provide some artificial gravity together with stabilization. An escape taxi, which is not shown in the figure, is attached to the bottom of the station.

The collector reflects the solar energy into the hemispherical mercury boiler from which the turbogenerator is operated. The boiler, of course, should have maximum absorptance for solar radiation, but it also should have minimum thermal emittance. The condenser, shown in the figure as a plate between the collector and the boiler, must have maximum thermal emittance.

The inflatable torus is constructed like an automobile tire, with plies of fabric cord, possibly of dacron, bonded together with an elastomer like butyl. The torus is connected by four passageways, like spokes, to a rigid central cylinder containing life-support equipment, electronics, and so forth with inflation tanks stacked around the periphery. The collector and torus are launched in the folded condition and erected when in orbit; the crew then enters from the taxi.

The station is heated by solar radiation, possibly by providing some windows in the collector, with black areas on the torus immediately below the windows. In order to minimize loss of heat to space, the remainder of the torus should have minimum emittance. It is estimated that if the black areas are 5 percent of the total torus area, a mean internal temperature of 70° F could be achieved, with 10° F variation in wall temperature when the station passes between sunshine and shadow. It is hoped also that the surface radiation characteristics can be selected such that, if the stabilization fails and the torus tumbles or turns partly toward the sun, habitable temperatures will still be maintained within the torus.

High reflectance and low emittance (that is, a metallic-type coating) may be a basic answer to this last problem. Obtaining such a coating on the thick elastomer skin, however, may be a formidable problem; it may be difficult enough to find any type of thermal coating that not only has the usual desirable properties but is also sufficiently flexible for use on erectable elastomer structures. It is especially desirable not only to keep the internal temperature of the torus within habitable limits but also to keep the temperature of the skin itself everywhere below that at which it loses strength and above that at which it becomes brittle.

Other problems of a similar nature are the development of passive heaters for heating food or for the various chemical processes of the life-support system, and of passive refrigerators for chilling food, for dehumidification, and for removal of toxic gases.

In order to solve these problems, theoretical calculations are being undertaken of temperature-time history distributions over the structure; also, experimental studies are being made in a space-environmental facility as shown schematically in figure 2. This figure shows a thermal model in the evacuated chamber. Solar heating is simulated by ring heaters covering the parabola. Liquid nitrogen cooled walls simulate cold space. Reflected sunlight from the earth is simulated by banks of albedo lamps illuminating a frosted glass plate. The glass plate is cooled in order to simulate thermal radiation from the earth. Inside the model are provided heaters to simulate heat inputs from crew and equipment, along with a fan to circulate air to duplicate the ventilation system of the space station. Thermocouples would be provided to measure the temperature distribution on the model. The thermal-control coatings could be varied until proper distributions are obtained.

In order to supplement the previously described tests, measurements will be made of the thermal conductivity and thermal storage of composite materials developed for the walls of the inflatable space station. Measurements will also be made, in a vacuum, of the thermal properties of the honeycomb plate of which the solar collector is made. Studies will be made of thermal-control coatings which might be applied to flexible materials. Such coatings must be capable of withstanding packaging, abrasion during launch, high vacuum, ultraviolet radiation, and micrometeroid damage.

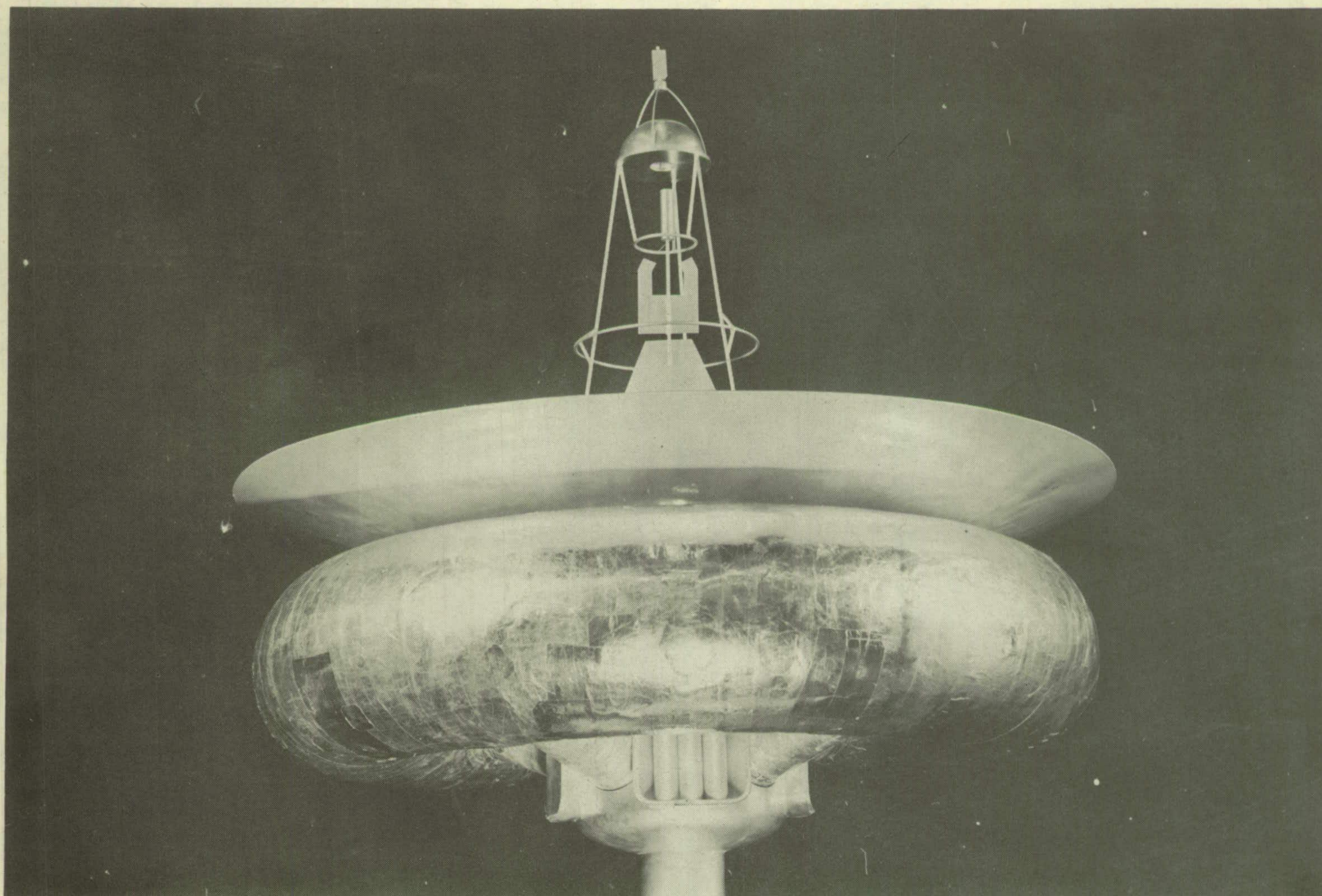

Figure 1.- Model of erectable-torus manned space laboratory. L-60-6205

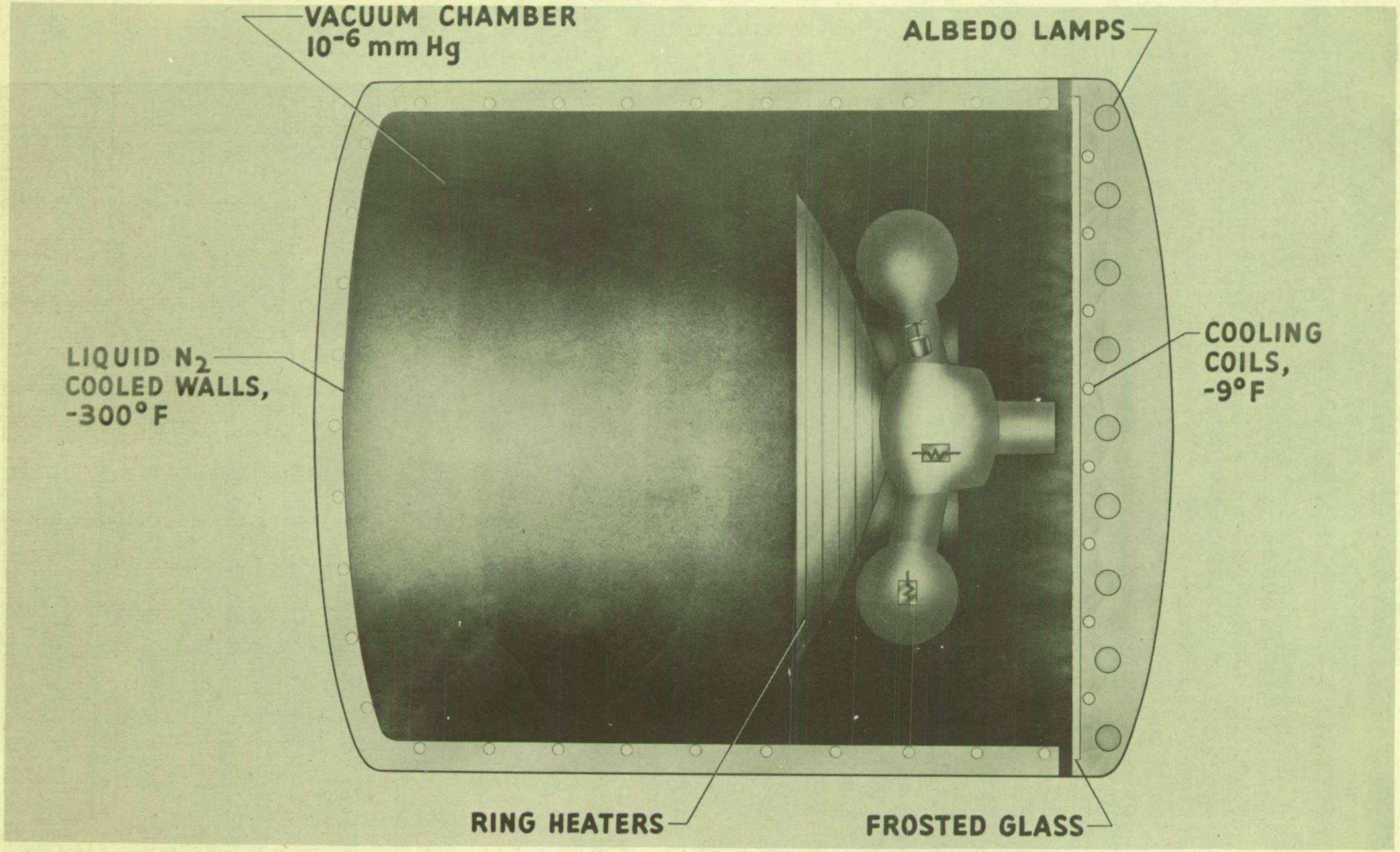

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Figure 2.- Temperature distribution tests of manned orbital laboratory in space-thermal simulator.

THERMAL DESIGN OF THE MICROMETEOROID SATELLITE S-55

By Earl C. Hastings, Jr., and Richard E. Turner
Langley Research Center

The thermal design of the micrometeoroid satellite S-55 involves both experimental and analytical approaches in selecting materials and coatings. Figure 1 is a cutaway drawing of the S-55 satellite, the purpose of which is to obtain scientific and engineering design data on the frequency and penetration hazard of micrometeoroids at altitudes between about 250 nautical miles and 700 nautical miles.

The passive method of thermal control used involves the selection of materials and coatings that give the desired ratio of absorptivity to emissivity a/e for keeping the telemetry temperature within narrow limits and also to prevent overheating of the separate experiments. The selection of a material or coating for this purpose, however, is dictated not only by its absorptivity and emissivity values, but also by its reliability and the constancy of these values under long exposure to the space environment. Several test programs have been conducted in order to evaluate the materials and coatings being considered. Some of these are as follows:

(1) Ultraviolet radiation in a vacuum to study discoloration and weight change

(2) Solar radiation in a vacuum to determine maximum equilibrium temperature, discoloration, and weight loss

(3) Thermal cycling and thermal shock to study material integrity (leaking, spalling, melting, etc.)

(4) Proton radiation to observe effects on color, crystal structure, and strength

(5) Determination of effects of heat associated with coating application on the leak rate of pressurized parts

(6) Absorptivity and emissivity measurements

The experimental tests outlined and the maximum use of coating methods successfully employed on previous satellites should provide high reliability of the material used for the thermal design of this vehicle.

A theoretical analysis was made to determine the values of a/e required for different areas in order that the telemetry remain within the desired temperature limits. External energy sources contributing to the heat balance are the earth and the sun. The analysis must take into account not only these radiations but also the conductive and radiative heat transfer between different parts. The work of computing the heat balance may be appreciably reduced by limiting the computations to only those conditions that produce the highest and lowest telemetry temperatures. At launch, the satellite will be spinning about its longitudinal axis; however, with gradual loss of rotational kinetic energy, but with constant angular momentum, it should soon be spinning about the axis of maximum moment of inertia, which is at right angles to the longitudinal axis. This type of spin (tumbling) is expected to occur soon after injection; therefore, calculations were made only for this condition. The maximum-temperature condition occurs when the satellite is in continuous sunlight with its spin axis parallel to the sun's radiation and its maximum area projected to the sun; the minimum-temperature condition corresponds to a minimum-sunlight orbit, with the spin axis normal to the sun's radiation. Figure 2 shows, for these two cases, the time variation of surface temperature of the nose cone, where the telemetry is housed. For the initial period, when the satellite is spinning about its longitudinal axis, permissible temperatures are obtained by selecting a suitable launch time.

The heat balance for the S-55 satellite is based on the same idealizations that have been made in previous heat balance studies except that the spherical-body altitude correction factor is not used for the spin stabilized case. Appreciable error can result from its use if the satellite is not approximately spherical and the altitude variation is large.

In general, the heat balance for the S-55 satellite contained one equation for each of thirteen parts. These equations accounted for conduction and radiation between components and were solved simultaneously on the IBM 7090 electronic data processing machine.

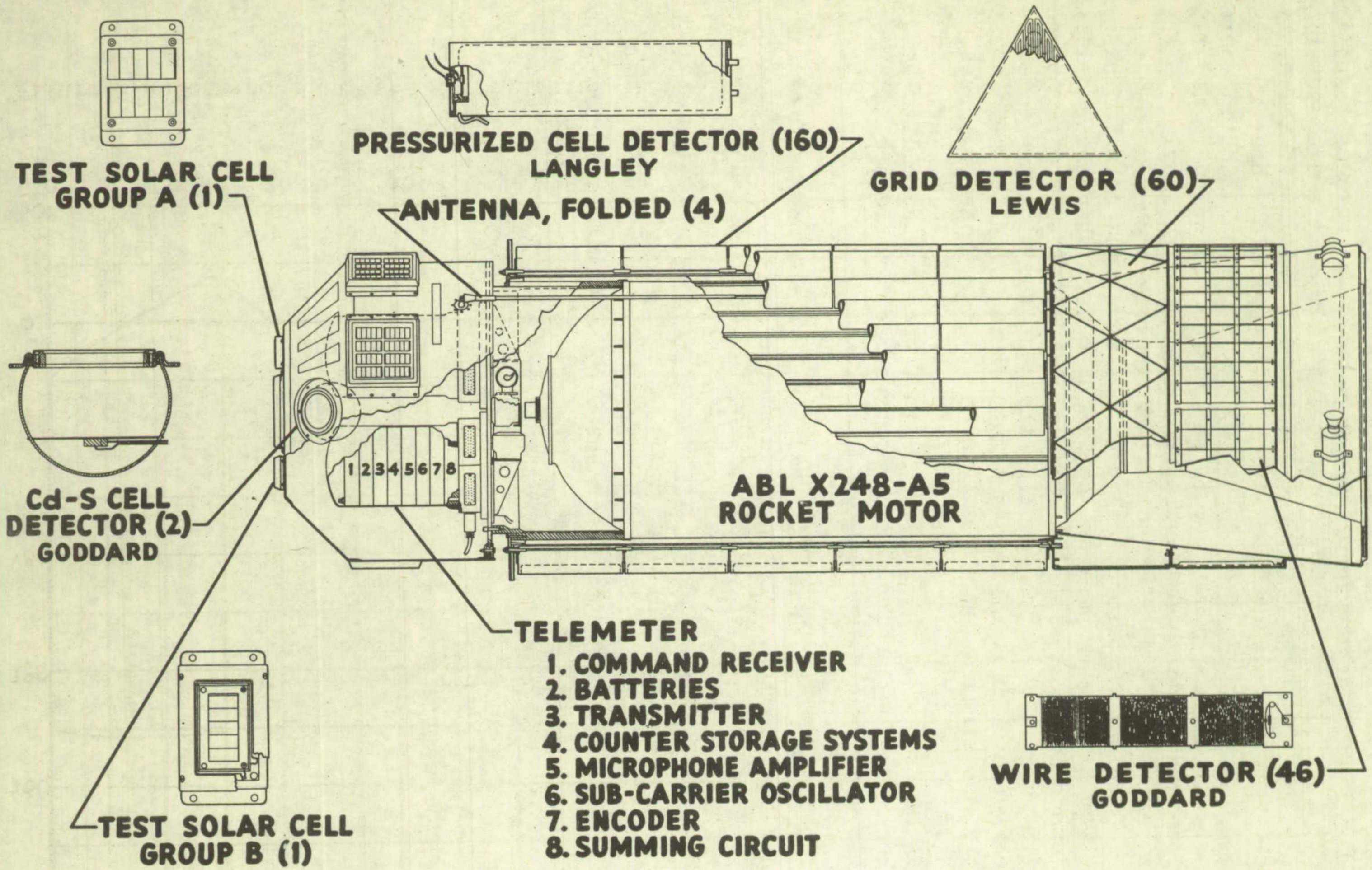

Figure 1.- Cutaway drawing of the micrometeoroid satellite S-55.

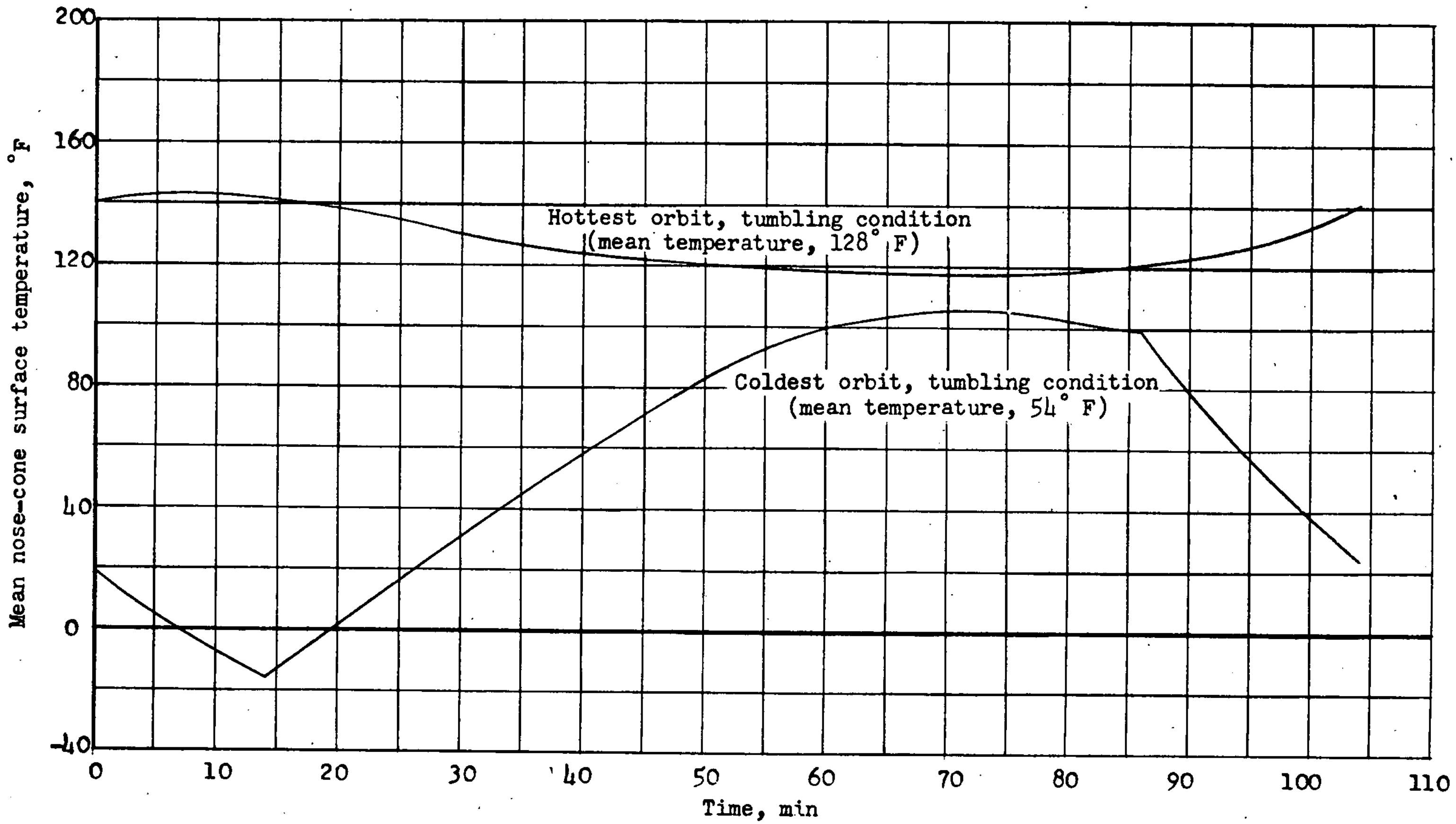

Figure 2.- Nose-cone surface temperatures in hottest and coldest orbits for the tumbling condition.

ORBITAL THERMAL CONTROL OF THE MERCURY CAPSULE

By Kenneth C. Weston
NASA Space Task Group

The approach to orbital thermal control of the Project Mercury capsule environment is relatively unsophisticated compared with that for many unmanned satellites. This is made possible by the relatively short orbital flight of about $4\frac{1}{2}$ hours and by the presence of the astronaut who is able to monitor the capsule systems and compensate for undesirable thermal conditions.

Figure 1 shows the general external features of the Mercury configuration as it appears in the orbital phase of flight. The conical afterbody is a double-wall structure. The inner wall serves as a pressure vessel for the manned compartment, and the outer wall, of shingle-type construction, acts as a radiating shield during reentry. Surface treatment of the shingles calls for a stably oxidized surface to minimize reentry temperatures. The shingles are supported by insulated stringers attached to the inner skin. Areas between stringers are insulated by blankets of Thermoflex insulation. This insulation is especially effective at high altitude due to the reduction of its thermal conductivity with decreasing pressure. As a result of the design of the afterbody for the severe reentry conditions, the heat balance on the manned compartment indicates the necessity for moderate internal cooling to compensate for the heat generation due to human and electrical sources. This cooling is achieved by the controlled vaporization of water in the cabin and astronaut-suit heat exchangers.

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Another problem of interest is that of maintenance of the retropack temperature within the limits determined for the safe and timely operation of the retrograde rockets for initiation of descent from orbit. Since the operational characteristics of solid-propellant rockets vary with grain temperature, it is of utmost importance that the retropack temperature be carefully controlled. In order to maintain tolerable internal temperatures, 40 percent of the retropack aluminum surface is stripe painted with a black silicone paint, and a 1/2-inch layer of insulation is mounted inside the outer case. Uncertainties in property values, behaviour of materials under orbital conditions, the command capability of the astronaut to hold various spatial orientations, and the extreme importance of control of the retro-rocket temperatures necessitate further precautions. Proper control is further assured by electric blankets surrounding the rocket casings, which can control the casing temperature between 49° F and 60° F. These blankets are normally unused but can be switched on by the astronaut when a warning light on the instrument panel indicates that the temperature of the casing has dropped below 40° F.

The two examples discussed typify the approach to thermal control of the Mercury capsule. The brevity of the mission and the ability of the astronaut to monitor and control the capsule systems have made possible fairly straightforward orbital thermal control which insures a high reliability necessary for mission success.

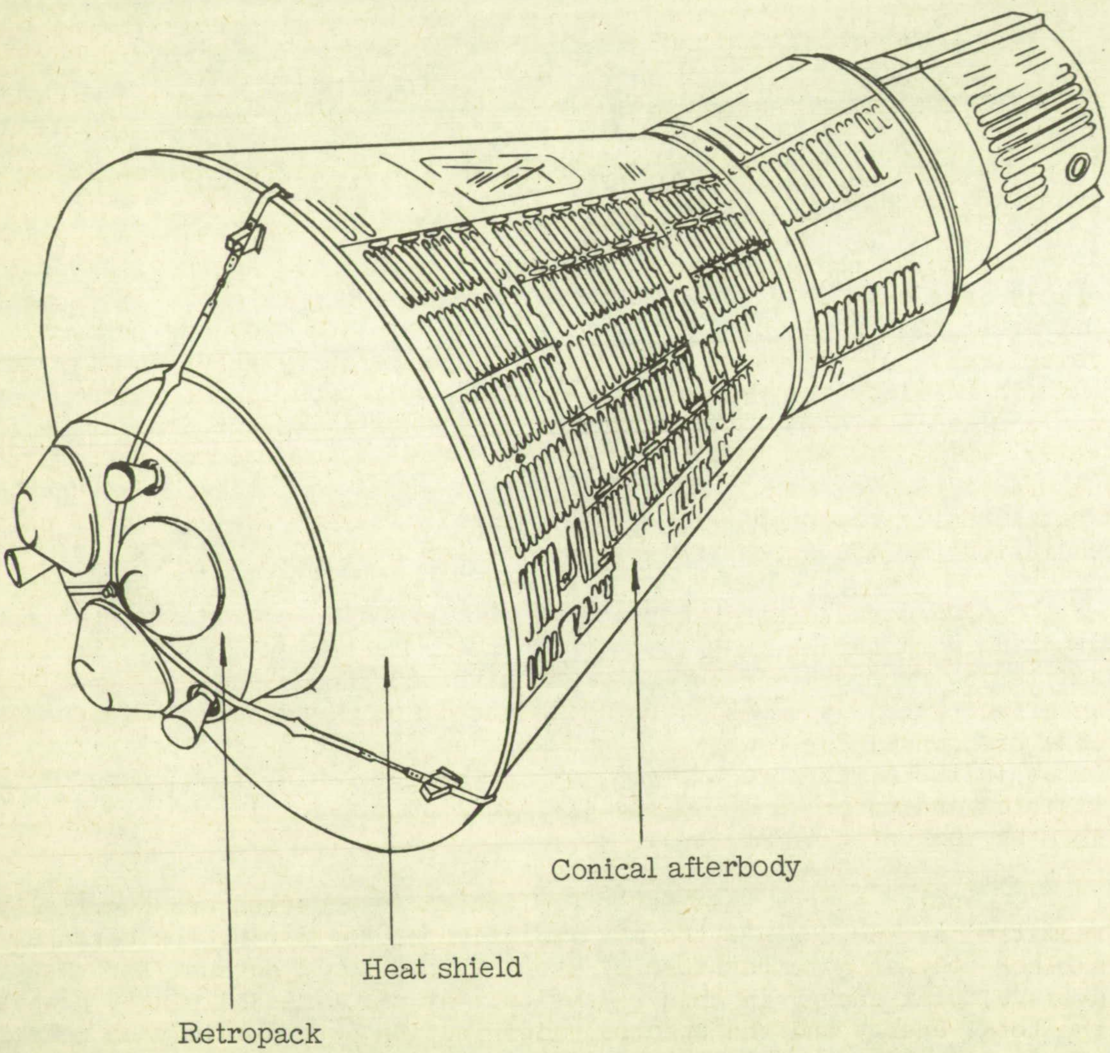

Figure 1.- Mercury orbital configuration.

THERMAL IRRADIATION OF AN EARTH SATELLITE

By S. Katzoff
Langley Research Center

The radiations that significantly affect the thermal balance of an earth satellite are:

(1) Direct solar radiation

(2) Solar radiation reflected from the earth

(3) Thermal radiation from the earth

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The total energy and the spectrum of the direct solar radiation are known to adequate accuracy.

The solar radiation reflected from the earth is known with considerably less certainty. The earth's average albedo is about 35 percent. Different latitudes, however, have average albedos above or below this value (ref. 1). Furthermore, there is considerable variation with time and place, since the reflectance of solar radiation is determined by the sun's elevation angle, the nature of the terrain (desert, forest, snow, water, etc.) and the weather (absolute humidity, cloudiness, height and nature of clouds, etc.). Accordingly, it would be desirable to have statistically reasonable upper and lower limits for the reflected solar radiation for use in thermal-balance design studies.

The spectral distribution of the reflected solar radiation is substantially different from that of the solar radiation itself. Rayleigh scattering from the air enhances the blue end of the spectrum (ref. 2), an effect which is usually very pronounced for cloudless conditions and less pronounced for overcast conditions or for a snow-covered ground. Ozone in the atmosphere absorbs the farther ultraviolet and some of the visible; and water vapor in the air, or liquid water in the clouds, absorbs much of the infrared.

The solar energy that is not reflected is absorbed and eventually reemitted as the earth's thermal radiation. Considering the earth as a black body at a temperature of about 250° K would account for the average total energy in this radiation. At any time and place, however, the total energy and the spectral distribution depend mainly on meteorological conditions, since the ground or cloud temperature and the atmospheric conditions are of primary importance in determining the emission in the different spectral regions of the infrared. Total energy emission may vary by a factor of 2 or more from one place to another. Figures 1 and 2, presented as examples, are block-type spectra constructed from the results given in references 3 and 4 for two different localities in the United States on a given day. There is a striking difference in shape, and the ratio of the areas is about 2:1. Again, a statistical study to establish design limits for thermal design would be desirable.

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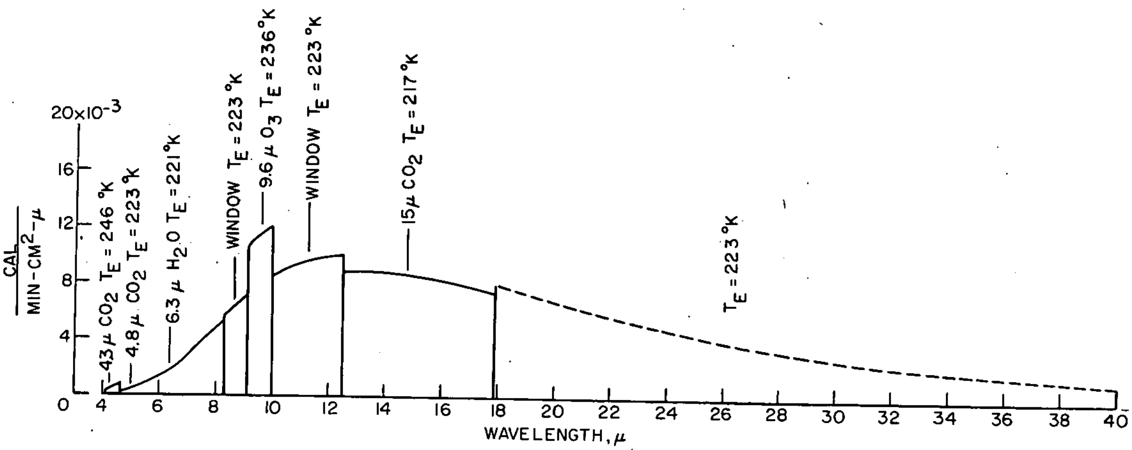

Figure 1.- Estimated spectrum of the vertically outgoing thermal radiation over Norfolk on March 6, 1960.

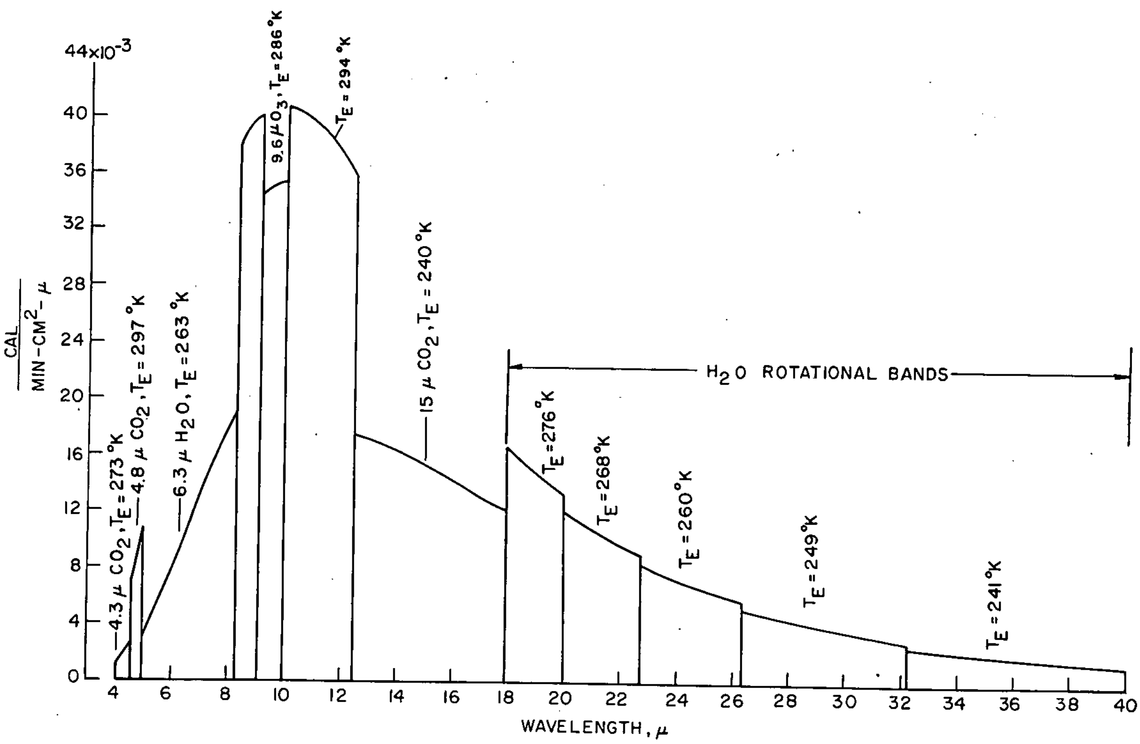

Figure 2.- Estimated spectrum of the vertically outgoing thermal radiation over Miami on March 6, 1960.

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