NASA SP-8105

NASA SPACE VEHICLE DESIGN CRITERIA (ENVIRONMENT)

SPACECRAFT THERMAL CONTROL

MAY 1973

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

FOREWORD

NASA experience has indicated a need for uniform design criteria for space vehicles. Accordingly, criteria have been developed in the following areas of technology:

Environment
Structures
Guidance and Control
Chemical Propulsion

Individual topics are issued as separate monographs as they are completed. A list of the titles that have been published to date can be found at the end of this monograph.

These monographs are to be regarded as guides to design and not as NASA requirements except as may be specified in formal project specifications. It is expected, however, that the monographs will be used to develop requirements for specific projects and be cited as the applicable documents in mission studies, or in contracts for the design and development of space vehicle systems.

This monograph was prepared under the cognizance of the Goddard Space Flight Center (GSFC) by Robert Lyle, Pericles Stabekis, and Robert Stroud of Exotech Incorporated, Washington, D. C. Scott A. Mills of GSFC served as program coordinator. An Advisory Panel, chaired by R. E. Kidwell, Jr. of GSFC, provided guidance in determining the monograph's scope and assuring its technical validity. The following individuals served as members:

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Comments concerning the technical content of these monographs will be welcomed by the National Aeronautics and Space Administration, Goddard Space Flight Center, Systems Reliability Directorate, Greenbelt, Maryland 20771.

May 1973

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SPACECRAFT THERMAL ENVIRONMENT

1. INTRODUCTION

Contemporary spacecraft are comprised of an array of components which operate effectively and reliably only in a hospitable thermal environment. The thermal control system is designed to provide an environment favorable to the operation of scientific instruments and other equipment vital to the mission by limiting temperature variations in that equipment to within specified design limits. Thermal control should be considered as a system function and requires system trade-offs in configuration, material selection, and operational cycles. The onboard thermal environment is determined by the magnitude and distribution of radiation inputs from the Sun and the planets, heat from internal sources (rockets, isotope heaters, and nuclear power sources), and heat from spacecraft electrical operations. The impact of these inputs is affected by the characteristics of the heat transfer paths within the spacecraft and the heat radiation characteristics of its external surfaces. This monograph provides guidance for assessment and control of spacecraft temperatures. The emphasis is on unmanned spacecraft in space environment; this monograph does not address itself to landers and thermal environment associated with atmospheric entry except where explicitly indicated.

Principal thermal sources external to the spacecraft have been treated in the NASA design criteria monographs SP-8005 (Solar Electromagnetic Radiation) and SP-8067 (Earth Albedo and Emitted Radiation).

Monographs in this series also treat the environments of other planets and space vehicle technology pertaining to structures, guidance and control, and chemical propulsion. All are listed at the end of this monograph.

2. STATE OF THE ART

The early satellites were small, symmetrical in shape, simple in design, and planned for missions of short duration. Temperature control was based primarily on the use of the surface coatings and finishes that would provide at the desired temperature the required balance between energy received and energy radiated to space. Spacecraft have since grown in size, weight, and complexity; missions have become longer and scientific experiments have imposed more stringent requirements on thermal design. As a result, a number of thermal control systems have evolved. Their common purpose is to modify the heat transfer to and from each spacecraft element so that its temperature will remain within the allowable range during the entire life of the mission. Temperature stability and temperature gradients are also primary concerns in the design of the thermal control system.

The state-of-the-art section considers the various sources that influence the spacecraft's thermal environment, relates flight and design experiences, and presents the design techniques that are used for thermal control.

2.1 Heat Balance

Spacecraft temperatures are computed from solutions of simple heat balance equations of the form

$$\text{Heat Stored} = \text{Heat In} - \text{Heat Out} \tag{1}$$

For a spacecraft in orbit above a planetary atmosphere, the heat that is absorbed by the spacecraft includes absorbed sunlight, reflected sunlight (albedo), and emitted radiation. Heat is produced within the spacecraft by power dissipated primarily by electrical and electronic components. Other heat sources may include electric heaters, chemical reactions, and radioisotopes. Heat is rejected from the spacecraft by radiation to space. Heat also is exchanged among spacecraft component parts by radiation and conduction. Convection may be neglected in most unmanned spacecraft because it is generally of concern only when there is a forced convection source as in the case of some pressurized components or during ground hold.

2.1.1 The Thermal Model

A detailed heat transfer analysis of a spacecraft requires a thermal model. The thermal model is a mathematical representation of the thermal parameters of the spacecraft. The model is represented by a set of equations which describes the heat transfer among selected points or nodes of the structure on both internal and external surfaces. These nodes are small isothermal elements into which the spacecraft is divided. The number and location are selected on the basis of accessibility, accuracy requirements, and reasonable use of engineering and computer time.

The heat flow at each node is given in reference 1 by

$$\begin{aligned}(mc)_i \frac{T_i(t+\Delta t) - T_i(t)}{\Delta t} &= a_i P_{s_i}(t) + a_i P_a(t) + P_{I_i}(t) \\ &+ \epsilon_i \, P_{e_i}(t) - A_i F_i \epsilon_i \sigma T_i^{\,4}(t) - \sum_{j=1}^{N} K_{ij} \left[T_i(t) - T_j(t)\right] \\ &- \sum_{j=1}^{N} A_i \mathcal{F}_{ij} \left[\sigma T_i^{\,4}(t) - \sigma T_j^{\,4}\right]\end{aligned} \tag{2}$$

where

i,j	=	node or element designations, including space or radiation sink
T_i	=	temperature, absolute (K)
t	=	time (s)
(t), (t+Δt)	=	indicate time dependence
mc	=	thermal mass (W· s/K)
α	=	solar absorptance
P_S	=	incident sunlight (W)
P_a	=	incident planetary albedo (W)
ϵ	=	thermal emittance
P_e	=	incident planetary emitted radiation (W)
P_I	=	internal power dissipation (W)
A	=	surface area (m^2)
F	=	view factor to space
σ	=	Stefan-Boltzmann constant ($W/m^2/K^4$)
K	=	thermal conductance (W/K)
$\mathcal{F}$	=	radiation interchange factor, for diffuse surfaces including view factors, emittances, and reflections

The result is a set of N finite-difference equations with time dependent terms for heat inputs from the Sun, a planet, and internal power dissipation; a term for heat radiated to space; and terms for conduction and radiation interchange among spacecraft elements or nodes. The N unknown temperatures appear in both the first and fourth powers. Solutions have been obtained by analog methods, but the most commonly used methods involve explicit or implicit solutions by digital computers (refs. 2, 3, and 4). Much of the input data, such as incident energy from the Sun and a planet, and the radiation interchange factors are also computed by digital computer programs, separately or as subroutines of a comprehensive computer program for spacecraft heat transfer analysis.

Equation 2 can be modified to account for aerodynamic heating; re-entry heating; heating from rocket exhaust plumes; heat exchange with specified boundaries such as a test chamber, launch vehicle, or a mounting interface; and temperature dependent parameters such as the emittance of thermal control louvers and the power dissipated by temperature controlled heaters.

2.1.2 The Orbital External Thermal Environment

The heat sources associated with the spacecraft external thermal environment are solar radiation, planetary albedo, and planetary emitted radiation. Although the following discussion of these sources is in terms of an Earth orbit, the same basic approach would apply to orbits of other planets.

At the mean distance of the Earth from the Sun, the magnitude of the solar radiation is about 1353 (±21) W/m^2 (ref. 5). The solar intensity varies as the inverse square of the distance from the Sun. Because of the slight ellipticity of the Earth's orbit, the seasonal variation of the solar intensity is ±3.5 percent of the mean value. Corresponding mean values for the planets Jupiter, Mars, Venus and Mercury are 0.037, 0.43, 1.91, and 6.67 times the solar intensity at the Earth (ref. 5). For Saturn, Uranus, Neptune, and Pluto, the analogous values are 0.011, 0.0027, 0.0011, and 0.00067.

The solar spectrum, shown in figure 1, extends over a wide spectral range. Very nearly 99 percent of the energy is contained within the range of 0.275 to 5.0 μm with a maximum at 0.48 μm.

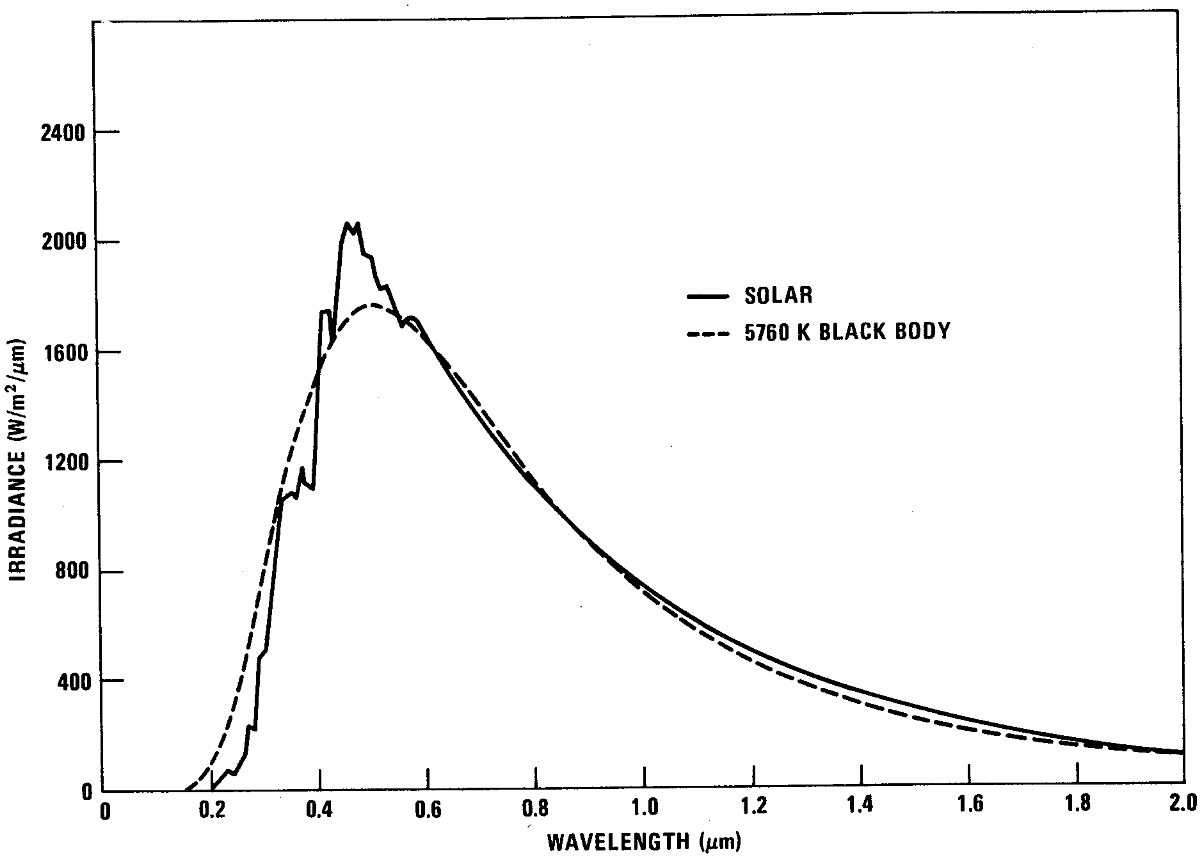


Figure 1.—Solar Spectral Irradiance Curve and Black Body Radiation Curve for 5760K.

When a spacecraft orbits a planet such as the Earth, the spacecraft surfaces receive emitted energy from the planet and reflected sunlight (albedo) from the planet. Reference 6 describes in some detail how Earth albedo and emitted radiation vary over the Earth's surface with terrain, latitude, season, and cloud cover. The recommended average value of Earth albedo of 0.30 leads to an equivalent black body temperature of the earth of 254 K and an average emitted radiation at the Earth's surface of 237 W/m^2.

To determine solar radiation, Earth albedo, and Earth emitted radiation is a problem in geometry and integration. With figure 2, radiation to a spacecraft surface element, dA, can be computed by

$$dP_s = S \cos \beta_1 \, dA \tag{3}$$

$$dP_e = \cos \beta_2 \cos \beta_3 \frac{\sigma T_E 4}{\pi} dA dA_E \tag{4}$$

$$dP_a = \frac{Sa}{\pi} \cos\beta_2 \cos\beta_3 \cos\beta_4 \, dAdA_E \tag{5}$$

where

S	=	solar constant
a	=	Earth albedo
T_E	=	equivalent black body temperature of the Earth (254 K)
β_1, β_2, β_3, and β_4	=	angles per figure 2
dA and dA_E	=	surface elements on spacecraft and Earth per figure 2

Equation 4 is integrated over the Earth cap visible from the spacecraft surface to obtain emitted energy incident on a spacecraft surface element for one position of the spacecraft relative to the Earth and the Sun. Equation 5 is integrated over the illuminated area of the Earth cap visible from the spacecraft surface to obtain the incident albedo energy. Incident solar radiation, albedo, and emitted radiation for each spacecraft surface element or node as a function of orbital position and rotation about a spacecraft axis is a tedious calculation which is usually performed by digital computers. A typical method is given in reference 7. The problem of computing incident solar, albedo, and emitted inputs can be complicated by

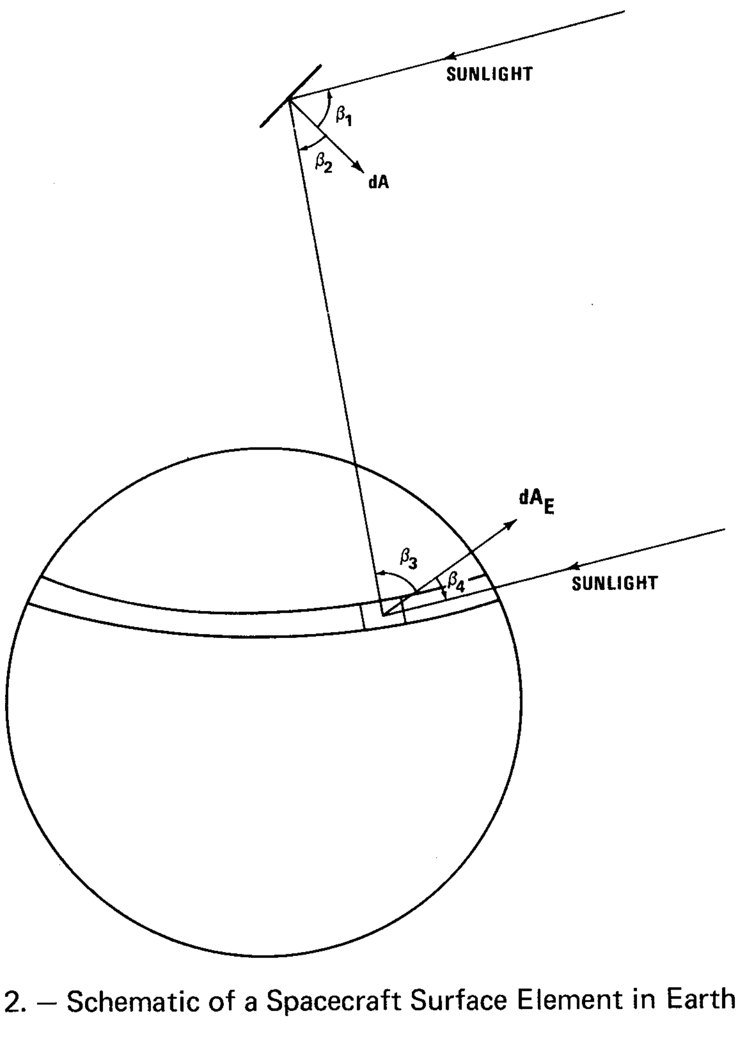


Figure 2. — Schematic of a Spacecraft Surface Element in Earth Orbit.

shadows cast by parts of the spacecraft over some surfaces (ref. 8) and by energy reflected among spacecraft surfaces (ref. 9).

2.1.3 Radiative Properties

On the basis of mission requirements, particular coatings are selected to help achieve thermal control. Figure 3 shows the spectral absorptance of five coatings typically used in spacecraft temperature control: black paint, white paint, vapor-deposited aluminum, silver, and gold.

In spacecraft thermal design it is common practice to use separate symbols to distinguish the absorbing or emissive properties of materials in the solar spectral region from the emissive properties in the infra-red region. Specifically, a is used for absorption of solar and reflected solar energy, and ϵ is used for absorption or emission of infra-red energy.

The solar absorptance, a, is defined by

$$a = \frac{\int_0^{\infty} a(\lambda)\, I_S(\lambda)\, d\lambda}{\int_0^{\infty} I_S(\lambda)\, d\lambda} \tag{6}$$

where

λ = wavelength (μm)
$a(\lambda)$ = spectral absorptance
$I_S(\lambda)$ = solar spectral irradiance (W/cm^2/μm)

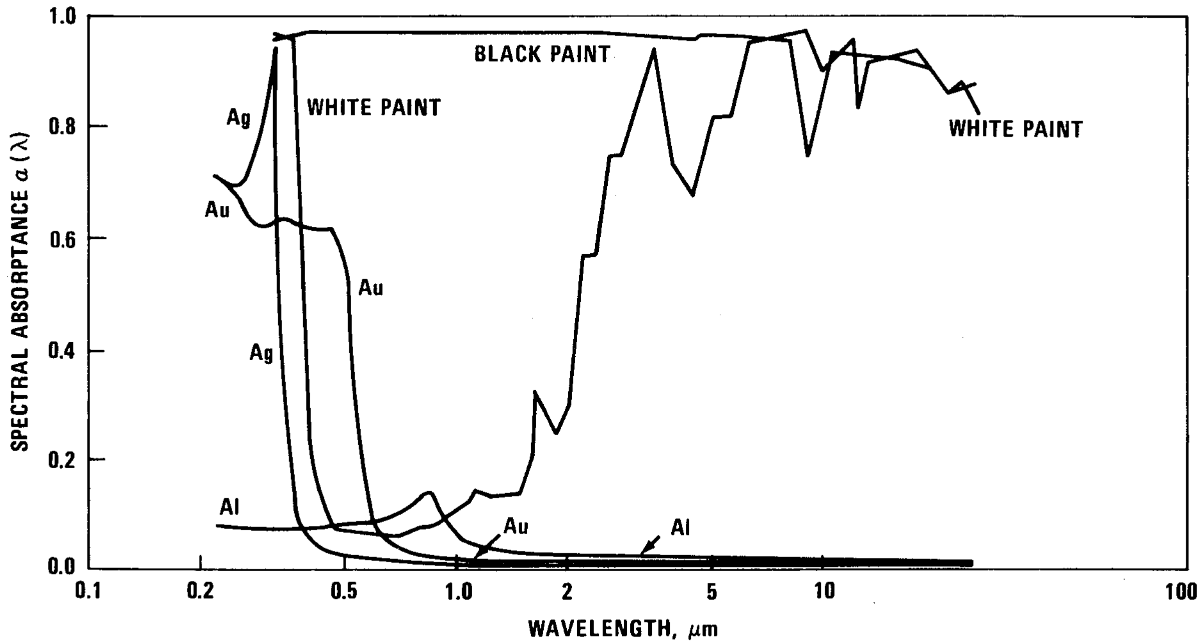


Figure 3. – Spectral Absorptance of Typical Spacecraft Coatings.

In equation 2, solar absorptance, a, is used for both solar and Earth albedo radiation. Reference 6 shows typical albedo spectra from clouds, water, and wheat fields. Although there can be substantial differences between albedo and solar spectra, the albedo energy is concentrated in the same spectral region as the solar energy and the error introduced by using the solar absorptance for albedo is less than the uncertainty in the magnitude of the incident albedo energy.

Because $a(\lambda) = \epsilon(\lambda)$, the thermal emittance of a surface receiving and emitting infrared radiation can be defined by

$$\epsilon = \frac{\int_0^\infty a(\lambda)\, I_b(\lambda)\, d\lambda}{\int_0^\infty I_b(\lambda)\, d\lambda} \tag{7}$$

where

I_b = spectral irradiance for black body (W/cm^2/μm)

Spectral irradiance from a perfectly black body can be computed from Planck's formula (ref.10) by

$$I_b = \frac{2\pi C_1}{\lambda^5 \left(e^{C_2/\lambda T} - 1\right)} \tag{8}$$

where

$C_1 = 0.595 \times 10^{-12}$ W/cm^2
C_2 = 1.44 cm·K

Spectral irradiance from a black body at 5760 K is shown in figure 1 for comparison with the solar spectral irradiance. Similar curves are shown in figure 4 (ref. 11) for black bodies at 218 and 286 K. Note the reduced magnitude and shift of the spectral distribution toward longer wavelengths with the lower black body temperature.

Solar absorptances and emittances for the coatings in figure 3 are shown in table 1. In general, emittance is a function of temperature, but for many materials, an emittance at 300 K can be used over the expected temperature range of a spacecraft with acceptable accuracy. Both normal and hemispherical emittances are included in the table. The normal emittance is determined by reflectance measurements at nearly normal incidence. The hemispherical emittance, which is ordinarily used for heat transfer calculations, can be obtained by applying the approximate conversion factors as shown in figure 5 (ref. 12).

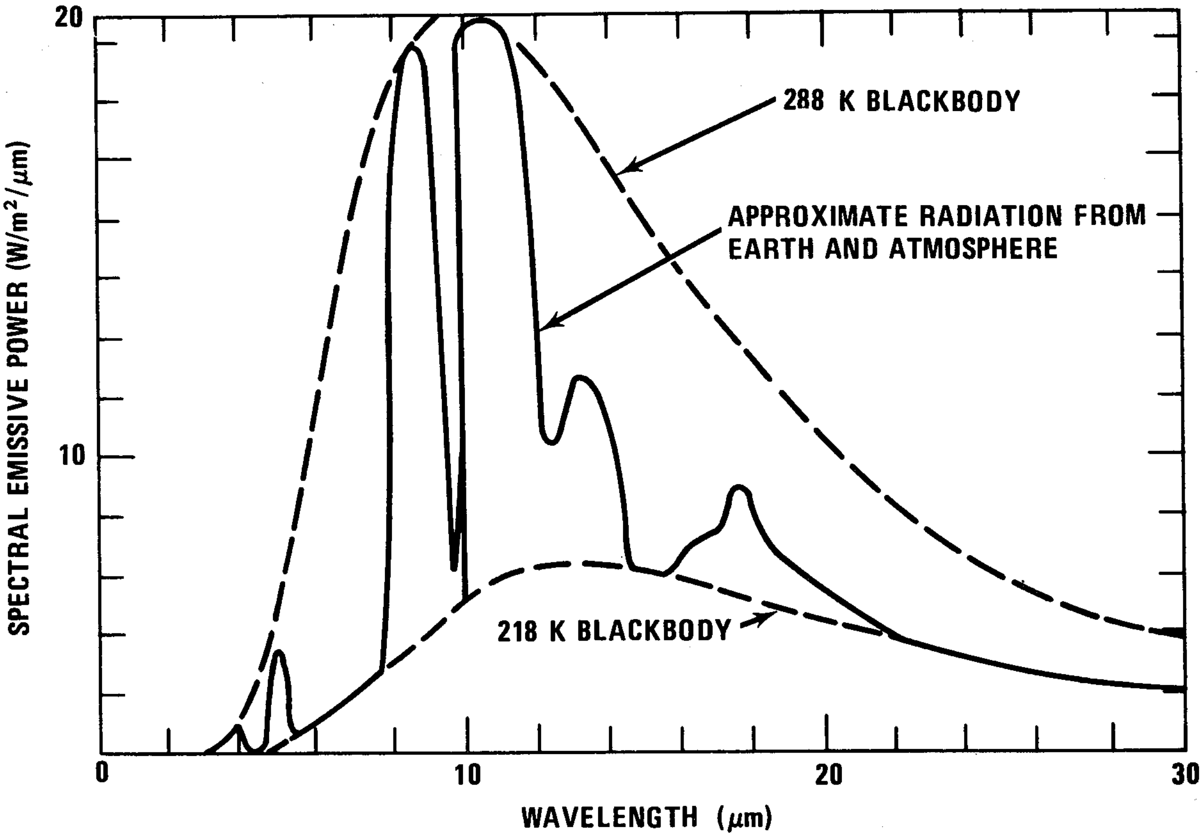


Figure 4. — Blackbody Spectral Intensity Distribution and Estimate of Earth Emitted Radiation.

TABLE I

SOLAR ABSORPTANCES AND THERMAL EMITTANCES FOR THE COATINGS IN FIGURE 3

Coating	Solar Absorptance (a)	Emittance with Normal Incidence at 300 K (ϵ_N)	Hemispherical Emittance at 300 K (ϵ_H)
White Paint*	0.21	0.91	0.86
Black Paint*	0.97	0.92	0.87
Vapor-Deposited Aluminum**	0.08	0.018	0.024
Gold (Au)**	0.19	0.015	0.020
Silver (Ag)**	0.05	0.010	0.013

* Reference 13

** Reference 14

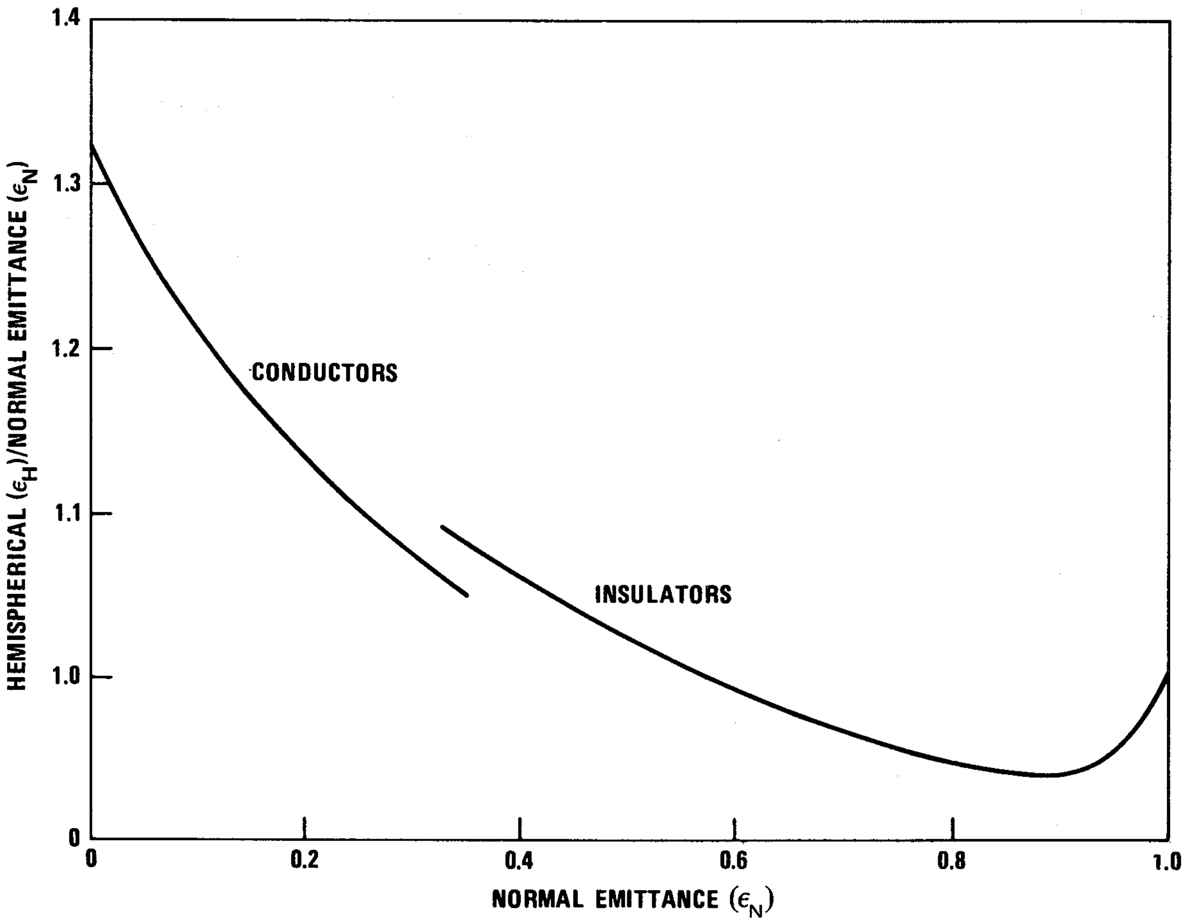


Figure 5. – Ratio of Hemispherical to Normal Emittance for Conductors and Insulators.

Thermal emittance ϵ in equation 2 is used for the absorptance of Earth emitted energy as well as the emission of radiated energy by the spacecraft surfaces. A comparison of the spectral content of Earth emission with black body radiation at a corresponding temperature shows differences in spectral distribution because of absorption and transmission bands in the atmosphere and the differences in temperature at the Earth's surface and through the atmosphere. Again, however, the error for using ϵ in computing absorbed Earth radiation is less than the uncertainty in the magnitude of the Earth-emitted energy.

2.1.4 Insight Into Temperature Control Methods

Some insight into temparature control methods can be gained by considering the two basic approaches, exposure of the spacecraft to the external environment and insulation of the spacecraft from the external environment.

2.1.4.1 Exposure to External Environment

In a simplified situation, the spacecraft is considered in thermal equilibrium as it receives heat only from the Sun and loses heat to space by radiation. Then, the heat balance equation reduces to

$$\text{Absorbed Sunlight} = \text{Heat Radiated to Space} \tag{9}$$

or

$$A_p S a = A\, \sigma\, \epsilon\, T^4 \tag{10}$$

and

$$T = \left[\frac{S}{\sigma} \circ \frac{A_p}{A} \circ \frac{a}{\epsilon} \right]^{1/4} \tag{11}$$

where

A_p = projected area (m^2)
Other terms: as previously defined

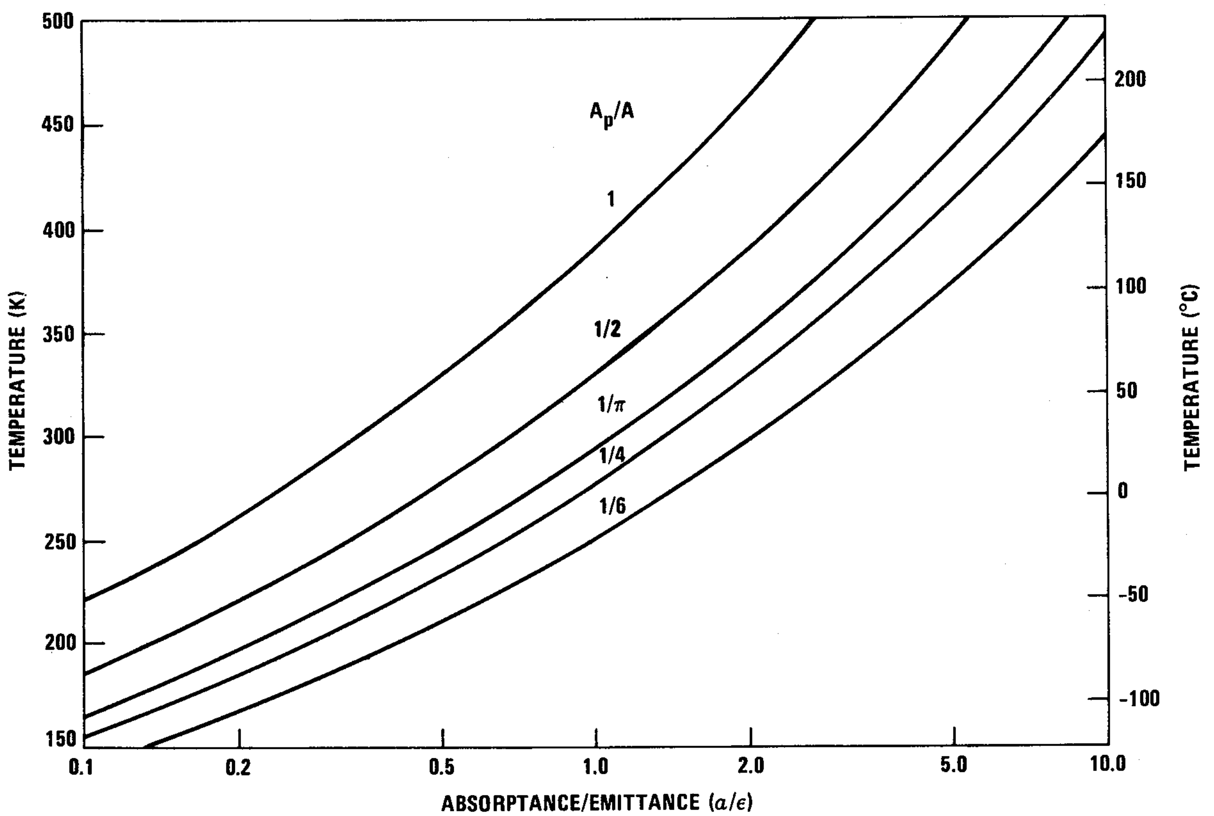


Figure 6. — Temperature as a Function of a/ϵ and of Projected Area to Total Area (A_p/A).

This relationship is shown in figure 6 as a function of the two ratios a/ϵ and A_p/A for the solar constant at the earth.

If the spacecraft temperature in the foregoing case is to be near room temperature, 293 K (68° F), the required product of (A_p/A) (a/ϵ) is 0.31. For example, this condition can be met for a spherical spacecraft (A_p/A = ¼) by selecting surface coatings with an a/ϵ = 1.24.

Another configuration would be a gray (a/ϵ = 1) cylinder with insulated ends and its axis oriented perpendicular to the Sun's rays ($A_p/A = \frac{1}{\pi} = 0.32$). This configuration is a very commonly used thermal design concept for spacecraft in orbits far enough from the Earth that the Earth albedo and emitted radiation incident on the spacecraft surfaces may be neglected.

The termperatures of a spacecraft in sunlight at various distances from the Sun can be computed from equation 11 simply by using the corresponding values of the solar intensities. Figure 7 shows the temperature of a black sphere at different distances from the Sun.

For a spacecraft in orbit near the Earth, the spacecraft receives Earth albedo and emitted radiation. The amount received depends on orbital altitude and also orientation of the orbital plane to the Sun which affects the proportion of time the spacecraft is in the Earth's

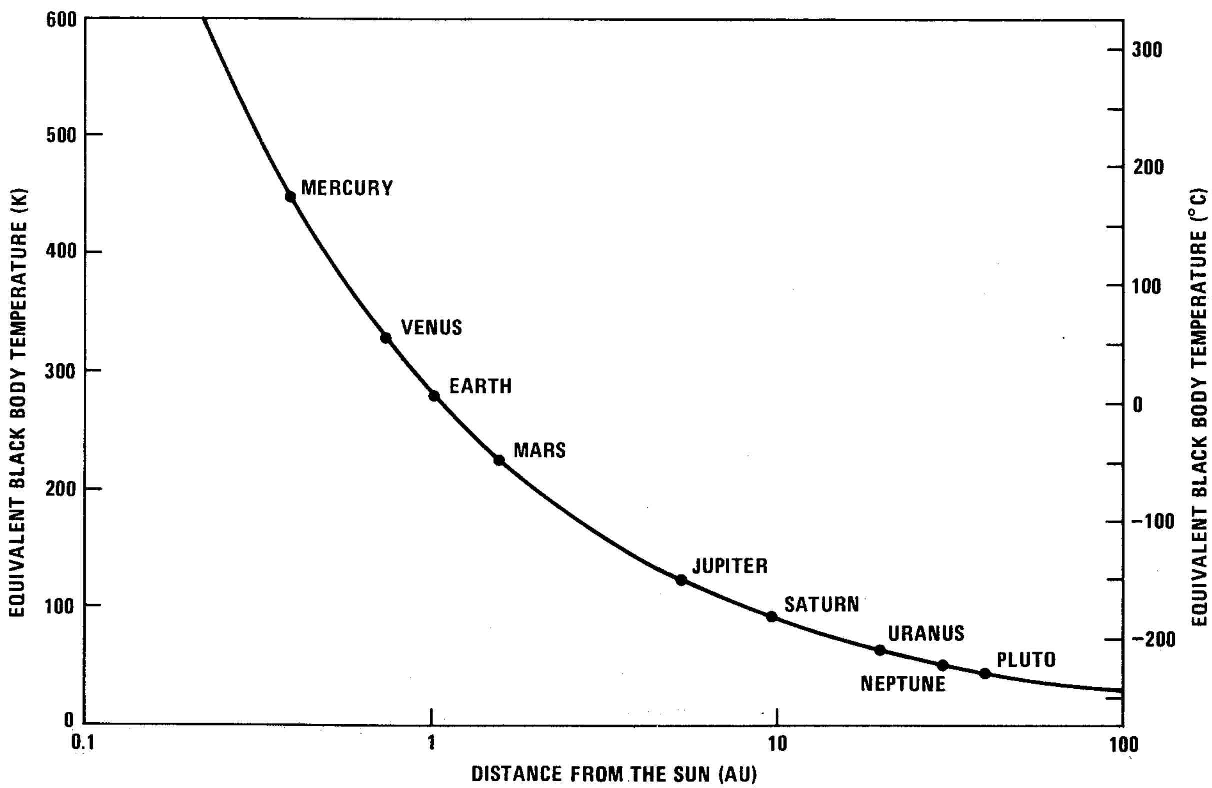


Figure 7.—Temperature of a Black Sphere.

shadow. Radiation from the Earth (considered a black body at 254 K) to a sphere is shown in figure 8. The Earth albedo input to a sphere, shown in figure 9, was based on data from reference 15 but corrected for a solar constant of 1353 W/m^2 and Earth albedo of 0.30. The

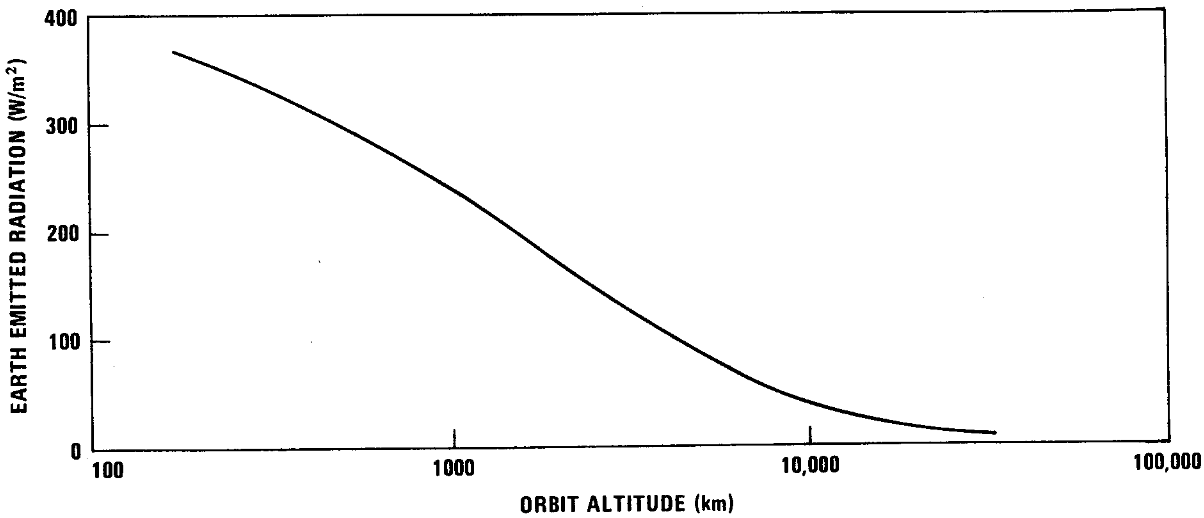


Figure 8.—Earth Emitted Radiation to Sphere of Unit Projected Area (the Earth is considered a black body at 254 K.)

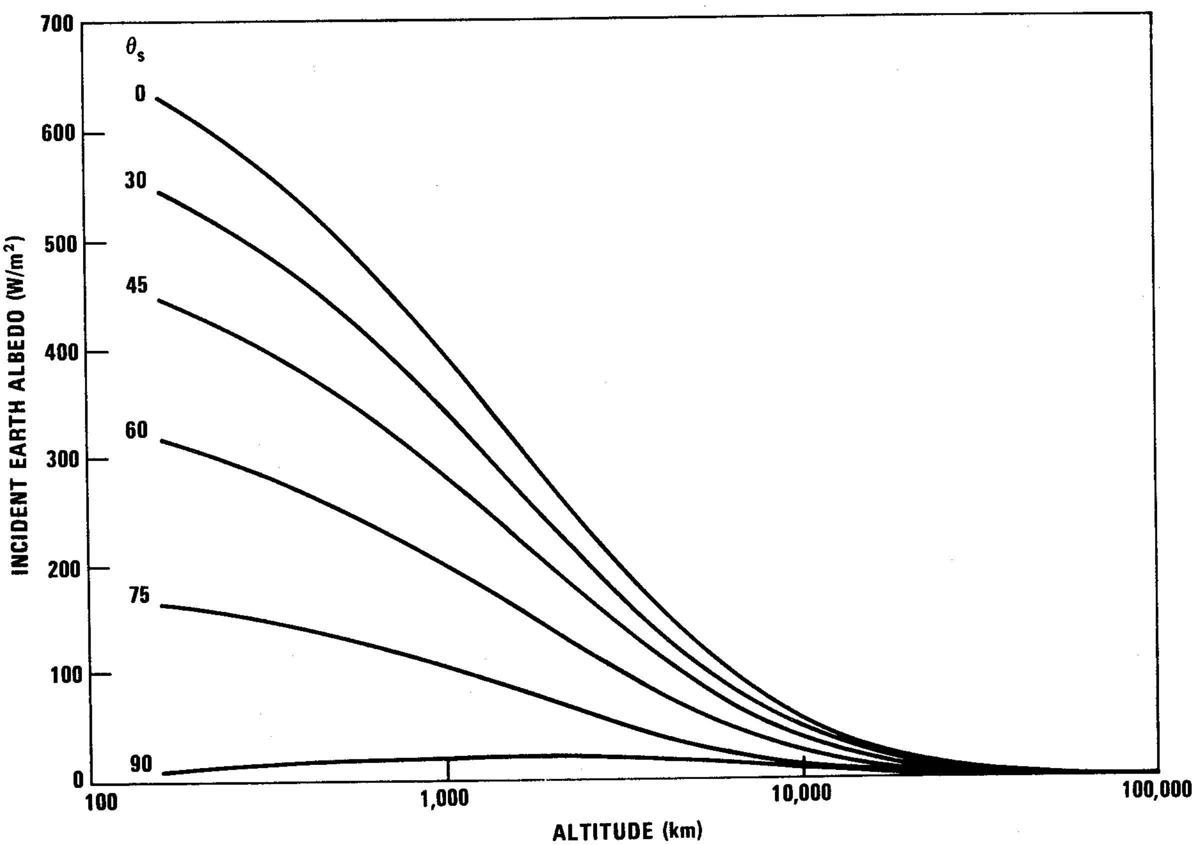


Figure 9.—Incident Earth Albedo per Unit Cross-Sectional Area of a Spherical Satellite as a Function of Altitude.

angle θ_S is the angle between the Earth-Sun line and the Earth-satellite line. Figure 10 shows the variation of Earth radiation inputs with orbital altitude and type of orbit. In developing figure 10, orbits with maximum total orbital average input from solar, Earth albedo, and Earth emitted radiation (full sunlight or twilight orbits) and orbits with minimum inputs (maximum percent of shadow time or noon orbits), were selected to compute the temperatures of a black sphere as a function of altitude for circular orbits. For a spacecraft in near Earth orbits, the temperatures can be expected to change almost 20 K as the heat inputs from the Sun and the Earth vary with relative position of the spacecraft orbital plane. Other factors, such as internal power levels, changes in surface properties of thermal coatings and changes in projected area of non-spherical spacecraft can increase this expected temperature change.

The projected area of a sphere is constant in all directions so that the mean temperature of a sphere in continuous sunlight is independent of its orientation toward the Sun. The temperatures of other shapes are dependent on attitude because of the change in projected area to sunlight.

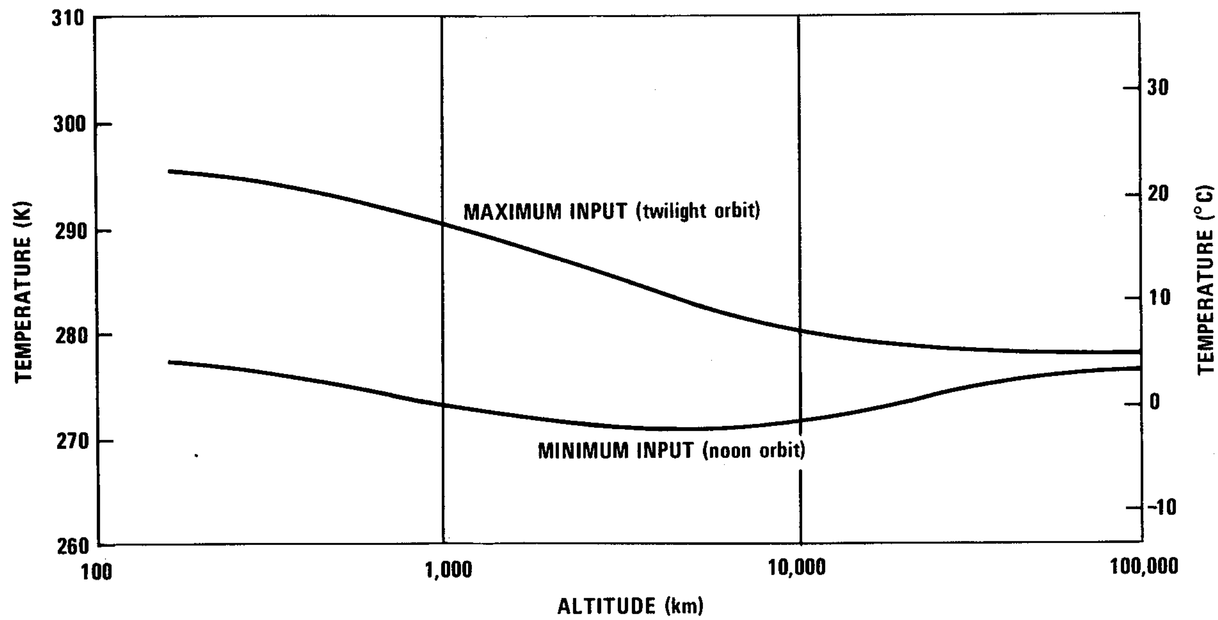


Figure 10.—Altitude vs Temperature for a Sphere in Earth Orbit ($\alpha/\epsilon = 1$).

2.1.4.2 Insulation from External Environment

Another approach to spacecraft temperature control is to insulate the spacecraft from the external environment. This approach requires that at least one surface of the spacecraft be allowed to radiate to space. The temperature is determined simply from the relation

$$P = A\,\sigma\,\epsilon T^4$$

where P is the internal power in the spacecraft plus any net heat absorbed from the external environment by the radiating surface or through the insulation. The power per unit area of

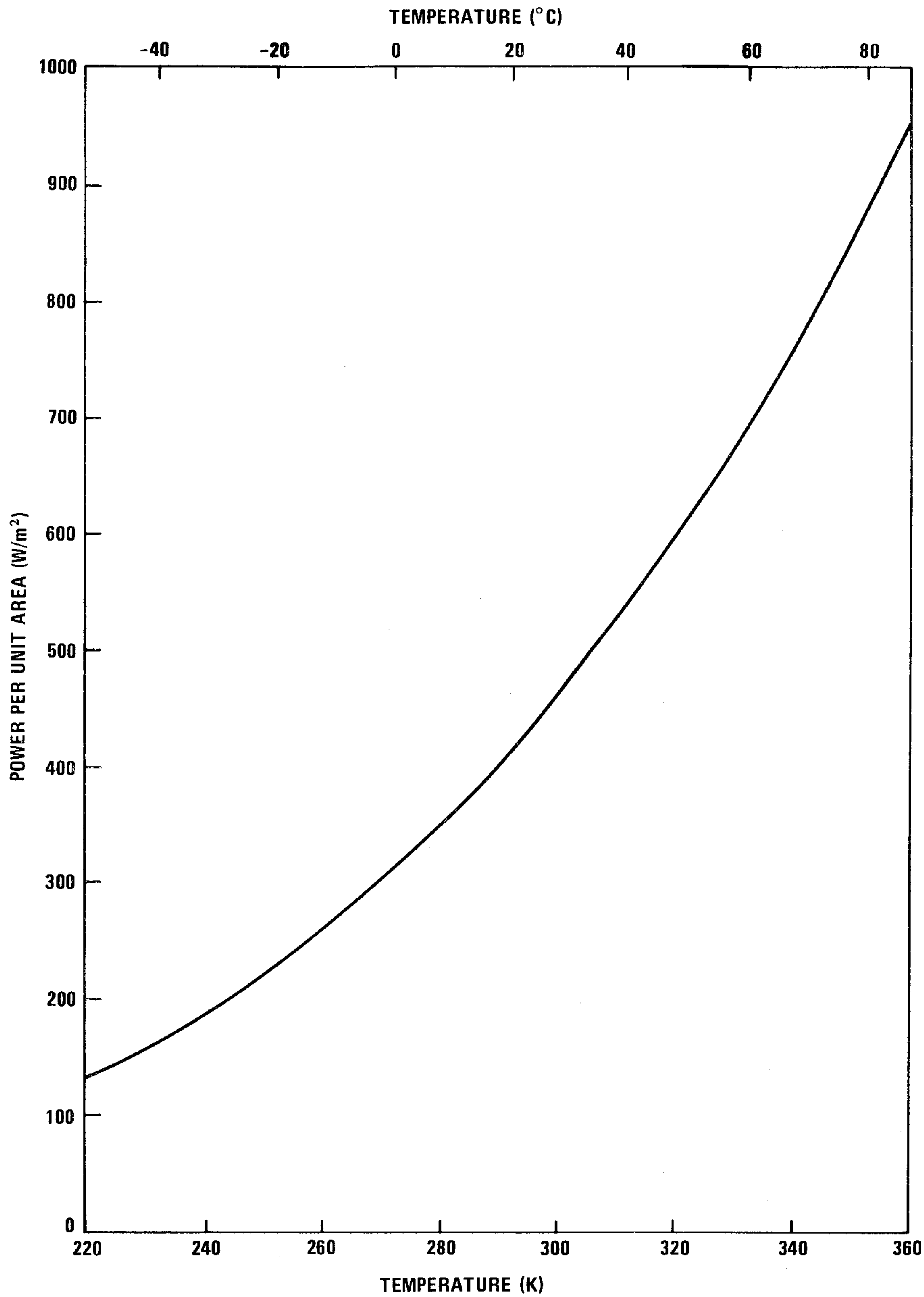


Figure 11.—Power per Unit Area of Black Radiator
(A ϵ = 1) vs Temperature.

a black radiator ($A \epsilon = 1$) is shown in figure 11. Spacecraft temperature can be regulated by controlling the internal power dissipation or the radiating area.

2.2 Elements of Thermal Design

The basic design approach for the temperature control system requires both analysis and testing. The analytical part of the design includes heat-transfer network analysis and the analysis of the overall power profile which lead to the development of a complete mathematical model for the spacecraft. Testing is used to complete and verify the analytical model or, as an integral part of the thermal design, to provide the means of determining design parameters. In certain cases, it may do both. The relative reliance on testing and the relationship of testing to analytical methods are matters of considerable concern and a source of debate among thermal designers of varying philosophies.

The typical evolution of a thermal design can be considered in three stages. These are the conceptual design, the preliminary design, and the detail design. The major task of the thermal designer during the conceptual design stage is to influence the general spacecraft design, particularly its configuration, in such a way that it will readily accommodate effective temperature control. Feasibility studies mission planning, and gross design tradeoffs also occur at this stage.

After the basic design goals and spacecraft characteristics have been established, the preliminary design stage begins. System level tradeoffs are identified and recommended on the basis of project or mission requirements and available time and money. More detailed planning of the spacecraft configuration, such as the overall electronics packaging layout, begin. Preliminary definitions of subsystem thermal requirements and characteristics are obtained and power and weight budgets established. Analyses are provided to support the preliminary design.

The foregoing processes are continued in greater depth in the detailed design stage. To assist in detailed thermal analysis of spacecraft designs, extensive use is made of large, high speed digital computers. General computer programs are available which aid the development of a thermal model. Three such programs are described in detail in references 2, 3, and 4.

The thermal design of a particular spacecraft may rely strongly on an established design of a spacecraft of similar concept. Examples are the Mariner, OGO, and Nimbus series.

2.3 Thermal Control Techniques

The two basic methods used to control spacecraft temperatures are termed passive and active.

2.3.1 Passive Thermal Control

The passive method is defined, for the purposes of this monograph, as one that maintains the component temperature within the desired range by control of conductive and radiative heat paths through selection of the geometrical configuration of surfaces and optical properties of materials. Such a system requires no moving parts, moving fluids, or power input other than the power dissipation of spacecraft functional equipment. Passive thermal control techniques include thermal coatings, thermal insulations, heat sinks, and phase change materials (that change in state). Passive radiator coolers are also used but are discussed in section 2.3.2.5 for convenience.

2.3.1.1 Thermal Coating Materials

The external surfaces of a spacecraft radiatively couple the spacecraft to space, the only heat sink available. Because these surfaces are also exposed to external sources of energy, their radiative properties must be selected to achieve the balance at the desired temperature between internally-dissipated and external sources of power and the heat rejected to space. The two properties of primary importance are the emittance of the surface ϵ and the solar absorptance α. Figure 12 indicates the range of these properties for different types of materials. Two or more coatings can be combined in an appropriate pattern to obtain some desired average surface values of α and ϵ, e.g., a checkerboard pattern of white paint and polished metal. The information shown in figure 12 does not consider the applicability of these coatings to a specific surface; practical constraints often seriously limit the coatings that may be used (ref. 17).

The values of the radiative properties are subject to uncertainties arising from four sources: (1) property measurement errors, (2) manufacturing reproducibility, (3) contamination before, during and after launch, and (4) space environment degradation. The effect of these uncertainties upon a thermal design is discussed in reference 18.

The errors made in property measurements result from basic deficiencies in the available instrumentation. In general, the smaller the value of a property, the greater the uncertainty in the measurement. Variations in manufacturing must be controlled by proper quality assurance, e.g., statistical sampling of the coated surfaces. Contamination can result from such sources as improper handling of thermal coatings, outgassing from the shroud during ascent, and condensation of outgassed constituents of other parts of the spacecraft, e.g.,

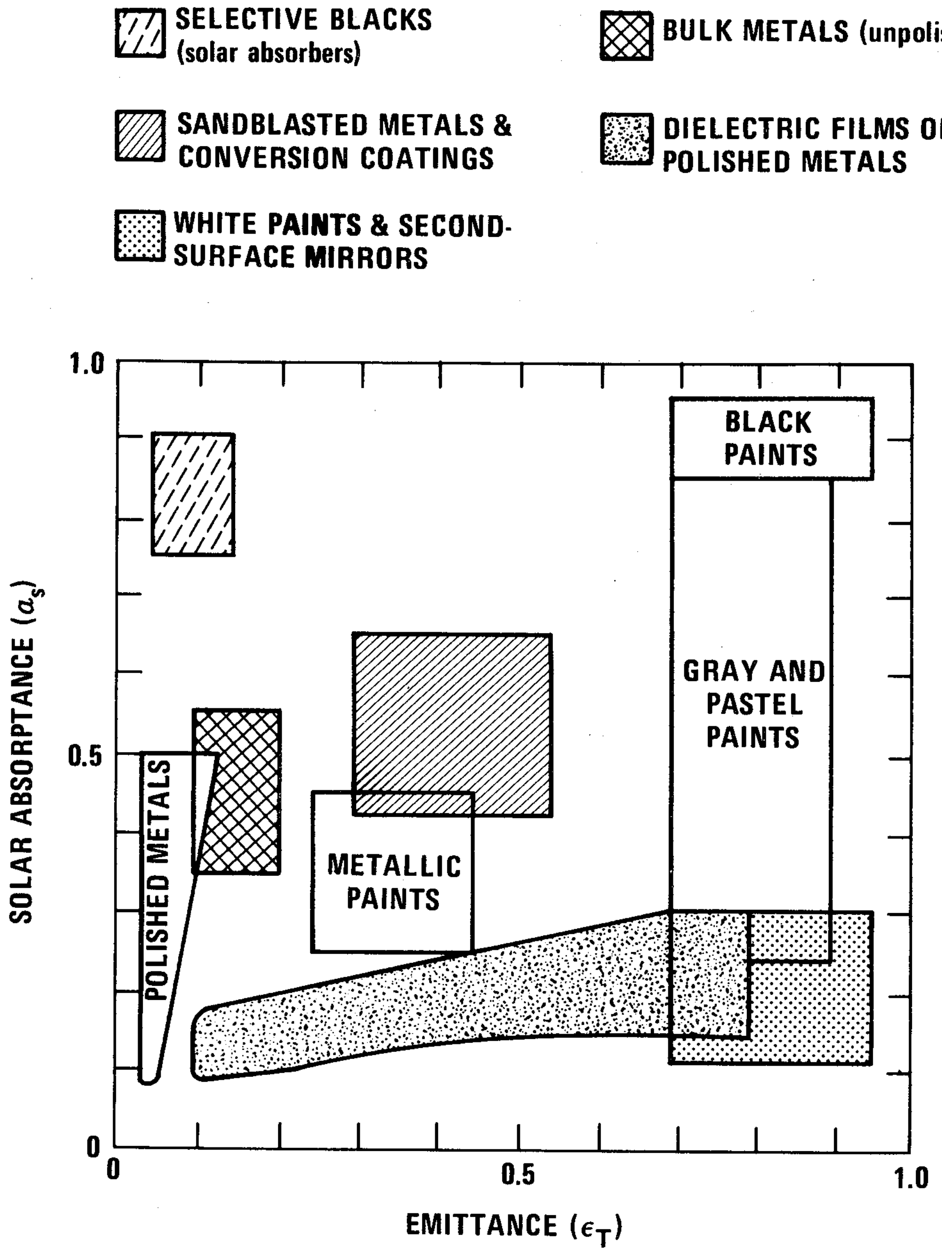


Figure 12. – General Types of Surfaces (from ref. 16.).

volatile materials and other thermal coatings. All of the factors must be evaluated and considered when selecting coatings, and the sensitivity of the thermal design requirements to these uncertainties must be examined carefully.

Degradation of thermal coatings in the space environment results from the combined effects of high vacuum, charged particles, and ultra violet radiation from the Sun. The last two factors vary with the mission trajectory and orbit. Degradation data are being obtained continually from flight tests and laboratory measurements. Normally, the thermal design of a spacecraft uses extrapolated data from flight and the laboratory to establish the worst case conditions for analysis, i.e., values for α to end of projected life of spacecraft.

2.3.1.2 Thermal Insulation

Thermal insulation is designed to reduce the rate of heat flow per unit area between two boundary surfaces at specified temperatures. Insulation may be a single, homogeneous material such as a low-thermal-conductivity foam or an evacuated, multilayer, insulation system in which each layer acts as a low-emittance radiation shield and is separated by low-conductance spacers. Descriptions, applications, and typical performance data for different types of thermal insulation are presented in references 19, 20, and 21.

Multilayer, evacuated insulations are widely used in the thermal control of spacecraft and components to (1) minimize heat flow to or from the component, (2) reduce the amplitude of temperature fluctuations in components because of time-varying external radiative heatfluxes, and (3) minimize the temperature gradients in components caused by varying directions of incoming external radiative heat. Typical examples of the applications of multilayer insulation for thermal control are:

- The heat flow to a cryogenic propellant tank may be reduced to minimize the boil-off of stored cryogen.
- The heat loss from an isolated spacecraft component may be reduced to minimize the heater power required to maintain the component within specified temperature limits.
- A portion of an orbiting planetary spacecraft or components thereof may be covered with insulation to minimize heat loss when shadowed from the Sun and to minimize heat inputs when exposed to direct sunlight, thereby reducing the internal temperature variation with orbital position.
- The exterior of a large spaceborne telescope mounted within a spacecraft may be insulated so that relatively large temperature gradients in the surrounding spacecraft structure, caused by uneven solar heating or nonuniform internal power dissipation, will not distort the supporting structure and thus degrade optical performance.

Multilayer insulations are thin radiation shields, usually consisting of 25 μm thick polyester or polyimide layers or films that are metallized with aluminum or gold on one or both sides by vacuum-deposition to achieve a low emittance. The thickness of the vacuum-deposited metal films is usually made 250 Å or larger to provide a surface which has a room-temperature emittance of 0.025 or less. Many different spacer materials are used to separate the radiation shields and minimize the shield-to-shield conductance. Common materials include glass-fiber paper, plastic and silk netting, thin sheets of foam, and embossed plastic film. Another common multilayer insulation system does not utilize any discrete spacer but uses 25 μm polyester films, aluminized on one side. These radiation shields are then crinkled so that the conductance from shield to shield is minimized by only having point contacts over a small fraction of the area.

For effective performance, the residual gas pressure within multilayer insulations must be less than 10^{-4} torr; for this reason and to protect the insulation from damage, particular attention must be paid to adequate venting during ascent. Multilayer insulations are usually vented through the edges of panels or by perforations in the shields. Loose flakes that remain after manufacturing must be prevented from escape during venting.

The thermal performance of a multilayer insulation system is usually characterized in terms of effective emittance or effective thermal conductivity perpendicular to the radiation shields as given by

$$\bar{\epsilon} = \frac{q/A}{\sigma T_h^4 - \sigma T_c^4}$$

$$\bar{k} = \frac{\ell(q/A)}{T_h - T_c}$$

where

$\bar{\epsilon}$ = effective emittance
q/A = rate of heat flow per unit area
T_h, T_c = hot and cold boundary temperatures (T_c may be the space temperature in some spacecraft applications.)
σ = Stefan-Boltzmann constant
$\bar{k}$ = effective thermal conductivity
ℓ = insulation thickness

For the theoretical case of conductively isolated radiation shields, the rate of heat flow per unit area is inversely proportional to the number of low-emittance radiation shield surfaces and proportional to the emittance of the shield surfaces.

However, there is no simple method for predicting performance of multilayer insulation prior to installation. Theory and measurements of emittance and conductivity on large, flat, insulation samples are insufficient for prediction because performance after installation is affected by the following variables:

- Interstitial gas pressure
- Degree of compression of insulation caused by installation or by thermal contraction
- Details of the interaction of the heat flow within the insulation to structural or other penetrations and to seams and edges
- Methods of attachment
- Hot and cold boundary temperatures
- Launch mechanical and acoustic noise environment and pre-launch storage environment (temperature, humidity, etc.)
- Size and percent open area of perforations if used for venting

In general, the deviation of thermal performance of an installed system from predicted or sample measurements increases as the surface area decreases. For a small component with a surface area of less than $0.1 m^2$, the heat flow per unit area may be an order of magnitude higher than the heat flow through an identical system applied to a $5m^2$ area because of the greater influence of seams and penetrations on the smaller area, as illustrated in figure 13 (ref. 22).

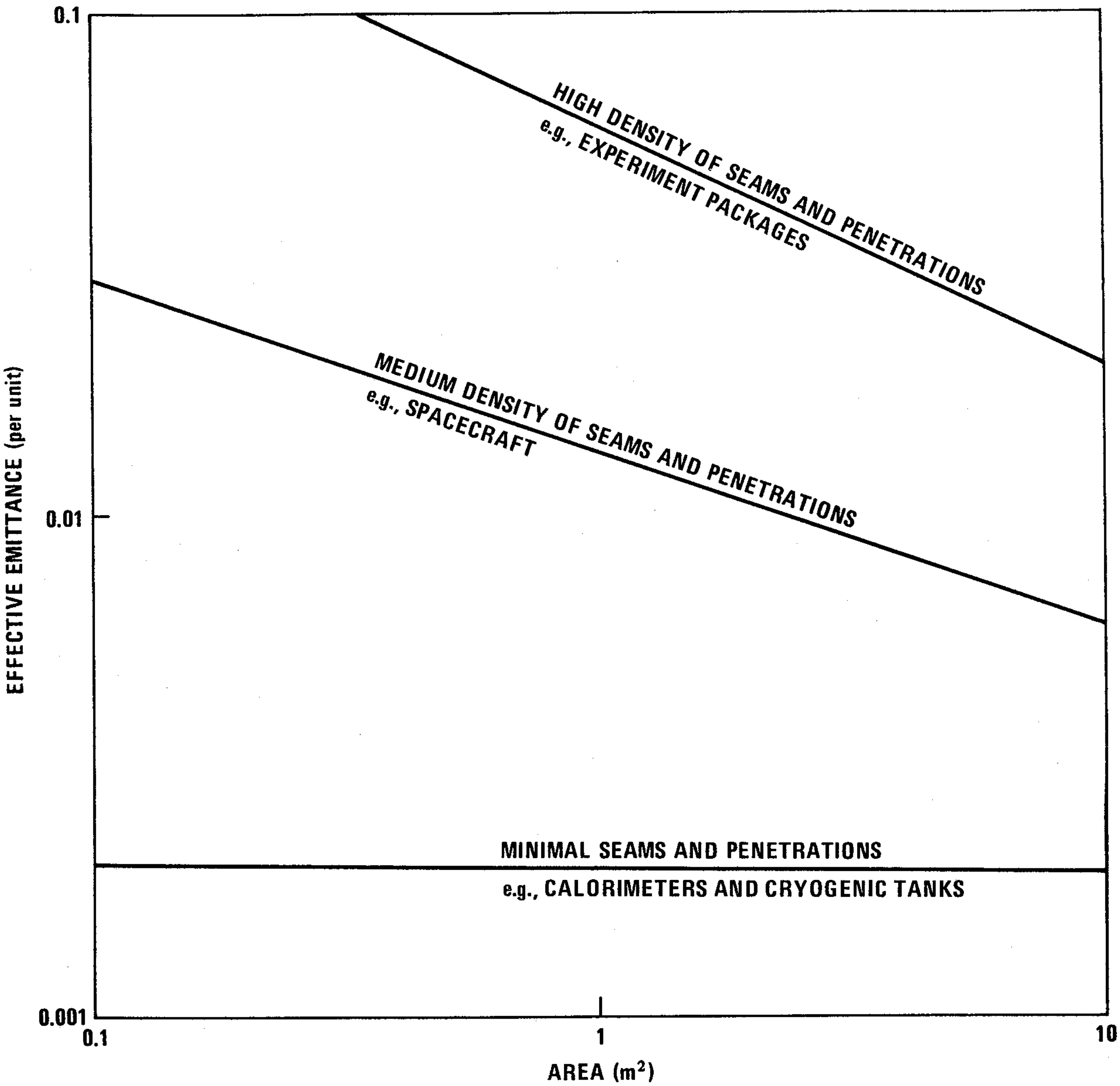


Figure 13. – Effective Emittance vs Area

2.3.1.3 Heat Sinks

Heat sinks are materials of large thermal capacity which are placed in thermal contact with the component whose temperature is to be controlled. When heat is generated by the component, the temperature rise is restricted because the heat is conducted into the sink. The sink will then dispose of this heat to adjacent locations through conduction or radiation. Heat sinks can serve the same function in reverse; i.e., they can prevent severe cooling during periods of low heat absorption or generation. They are extensively used to control the temperature of electronic packages which have cyclical variations in power dissipation. Heat sinks are quite reliable. The equipment and structure of the spacecraft itself usually provides a heat sink. The addition of weight for purely heat sink purposes is not common.

2.3.1.4 Phase Change Materials

Phase change materials are those that can change from one physically distinct and mechanically separable state to another distinct form such as from a definite crystalline to a liquid state. Phase change materials used for temperature control are those whose melting point is close to the desired temperature of a component. Then the latent heat associated with the phase change provides a large thermal inertia when the temperature of the attached component is passing through the melting point. However, the phase change material cannot prevent a further temperature rise when all the material is melted.

Phase change materials are used in electronic component thermal control systems to enable cyclically operating components to remain very nearly isothermal at all times, in thermal energy storage devices to store energy isothermally for later release, and in space flight experiments to maintain thermal stability. Reference 23 presents detailed information about aspects of phase change materials technology and discusses their numerous space and terrestrial applications.

2.3.2 Active Thermal Control Techniques

Active thermal control techniques include electrical heaters, heater-pump-radiator (HPR) fluid systems, thermal louvers, heat pipes, and spaceborne cooling systems. The latter category includes passive radiator coolers which are discussed in this section for convenience rather than under passive techniques (sec. 2.3.1).

2.3.2.1 Electrical Heaters

Electrical heaters (resistance elements) are commonly used to maintain component temperatures close to desired values. The heater is typically part of a closed-loop system that includes a temperature sensing element and an electronic temperature controller. Electrical heaters are used in on-off control modes, ground-controllable modes (including command modes), or simply in continuously-on modes. Small radioactive isotope heaters which continuously supply heat to a component at a constant rate may also be used instead of a variable heat dissipation.

2.3.2.2 Heater-Pump-Radiator (HPR) Fluid Systems

The HPR fluid thermal control system is a dynamic system used for the addition or removal of heat from a component to maintain its temperature within the operational range. In general, a temperature sensor or thermostat detects excessive changes in the temperature of the component and signals the heater and pump to adjust input heat and fluid flow accordingly. For cooling, the system relies on a radiator whose temperature is lower than that of the component in order to receive the heat from the fluid and radiate it to deep space. When only heating is needed for the entire mission, the radiator can be deleted. If cooling only is needed, the heater can be deleted. The reliability of the HPR fluid system depends on the successful operation of the pump and heater. The system heat transfer capability could be jeopardized by a fluid leak in the loop.

2.3.2.3 Thermal Louvers

Louvers provide a simple, reliable method of active temperature control by varying the effective emittance of a spacecraft radiator with temperature. The most commonly used configuration (ref. 24) consists of a series of polished aluminum blades arranged in venetian blind fashion over a high emittance radiator. Each blade is attached to a shaft supported at the ends by bearings. A bimetallic spring attached to the shaft of each blade varies the blade angle with temperature and changes the exposure to space of the radiator surface. Other mechanical configurations and temperature actuators have been used. For example, the Nimbus louvers (ref. 25) employ a fluid-driven bellows; blade position in flight is given by telemetry.

The major problems in the application of thermal louvers are:

- Mechanical bearing lubrication, alignment, and launch vehicle- or spin-induced loads
- Calibration of blade angle with temperature
- Time delay and temperature gradients between radiator and louver actuator temperatures
- External heat inputs from direct and reflected sunlight and from emitted and reflected radiation from spacecraft appendages, such as solar paddles (ref. 26). (External heat inputs reduce the heat rejection capabilities of the louver system and can result in excessive blade temperatures as discussed in reference 27.)

2.3.2.4 Heat Pipes

In its basic form, a heat pipe is a very simple, self-contained device (fig. 14). The walls of an enclosure are lined with a "wicking" material saturated with a "working fluid." Heat is then conducted from a source such as electronics through the heat pipe walls and into the working fluid. The additional heat causes the evaporation of working fluid which then travels by the induced pressure gradient to a colder portion of the pipe. The vapor carries with it the latent heat of vaporization which is released as the vapor condenses in a colder portion of the pipe. The heat is then conducted through the wall to a heat rejection system such as a radiator. Meanwhile, the condensed fluid is pumped back to the hot end by the capillary action of the wicking material to complete the cycle. In some applications, heat pipes exhibit an effective thermal conductivity that exceeds solid copper by orders of magnitude. The heat pipe in this basic form is useful in "isothermalizing" spacecraft structures such as equipment shelves and telescope optical tubes by conducting thermal energy efficiently from hotter to colder regions. Reference 28 discusses a typical application.

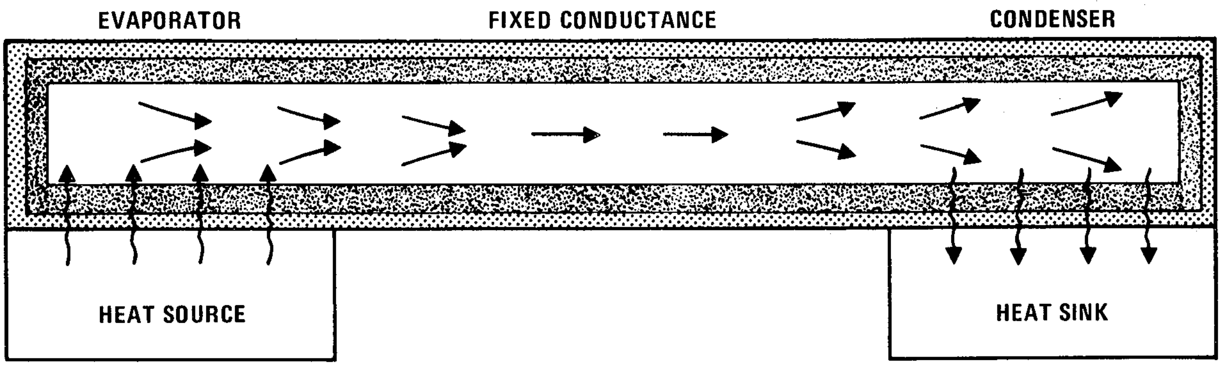


Figure 14. – Heat Pipe Concepts

This basic heat pipe has a fixed, high conductance and must, therfore, be designed for given heat source and sink conditions. Deviation from these conditions results in the overcooling or overheating of the heat source. A heat pipe, however, that is designed to vary its effective conductance in response to changing conditions can be used to control the source at a near constant temperature. A typical application of a "variable conductance heat pipe" is shown in figure 15 and discussed in reference 29. Feedback control provides greater sensitivity than

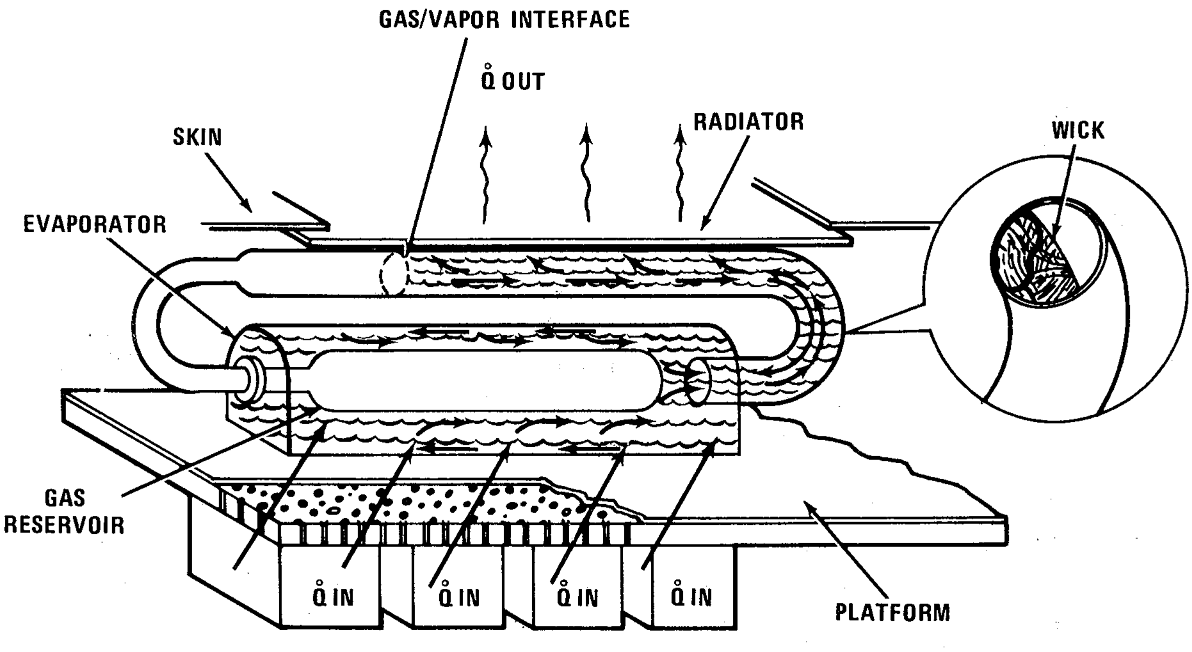


Figure 15. – Heat Pipe Experiment on OAO-C ($\mathring{Q}$ Equals Thermal Energy per Unit Time).

that available with standard variable conductance techniques. Heat pipes also are designed to provide heat transfer in one direction only (thermal diode). Reference 30 thoroughly discusses the theory and design of variable conductance heat pipes. Thermal diodes are discussed in reference 31 and feedback control heat pipes in reference 32.

When integrating heat pipes into a spacecraft design, care should be taken to keep the pipes essentially level so that during testing the height to which the condensate must be returned does not exceed the capillary pumping capability.

2.3.2.5 Spaceborne Cooling Systems

Spaceborne cooling systems are used to provide refrigeration for specific components such as infrared detectors (used for remote sensing applications).

The basic methods for providing refrigeration are:

- Passive radiative coolers which provide refrigeration at cryogenic temperatures by radiation to the space environment
- Open-cycle systems which use stored high-pressure gas or a stored cryogenic in liquid or solid form
- Closed-cycle systems which utilize a mechanical refrigerator to provide cooling at low temperatures
- Thermoelectric cooling systems

A review of spaceborne cooling systems is given in reference 33.

Passive radiators for cooling detectors have been flown, and others are being developed for a number of spacecraft experiments in the mid-70s. Passive radiators require no power input, but in practical sizes can handle refrigeration loads of only about 10 milliwatts at temperatures above 80 K.

Open-cycle systems that use the Joule-Thomson (J-T) expansion process provide refrigeration by discharging a high-pressure, ambient-temperature, stored gas through a counter-flow heat exchanger and an expansion valve. When a relatively small amount of total refrigeration capacity is required at low temperatures after an extended inactive period, J-T systems are particularly attractive.

Open-cycle systems that use solid cryogens are being developed for future spacecraft experiments. In these systems a cryogen is solidified within a very well-insulated, cryogenic container prior to launch. The ullage space above the stored solid is evacuated to maintain the cryogen in a solid state. Heat from the object to be cooled causes the cryogen to sublime, and the resulting vapor is vented to space. Typical applications involve continuous cooling loads of less than 100 milliwatts for periods of 1 year or less at temperatures of 15 to 100 K.

Closed-cycle refrigeration systems are currently under development for future spacecraft missions. In these systems a mechanical refrigerator is supplied with electrical power from the spacecraft system, and the heat load at the cryogenic-load temperature is pumped up to the temperatures of a space radiator which usually is operated at room temperature. The space radiator is used to dissipate the refrigeration load plus all of the energy put into the refrigerator itself. A description of various types of refrigerators and performance data is given in reference 34. Typical applications for closed-cycle refrigeration systems involve cooling loads of 1 watt or more at temperatures of 10 to 80 K for periods up to six months or more in space.

Thermoelectric cooling systems utilizing the "Peltier"effect find limited application for refrigeration loads of a fraction of a watt or less at temperatures above approximately 150 K (ref. 33). No moving parts are used, but the ratio of useful refrigeration divided by the input power is usually quite low.

2.4 Flight and Design Experience

Flight and design experiences demonstrate the evolution of various thermal control techniques which have been required by the increasing complexity of spacecraft and their missions. Table 2 gives the types of thermal techniques that have been used for a number of spacecraft, grouped according to their orbits.

2.5 Testing

Thermal testing is performed at various stages of spacecraft development according to the needs of the particular program. A typical phasing of testing with other development events is given in table 3.

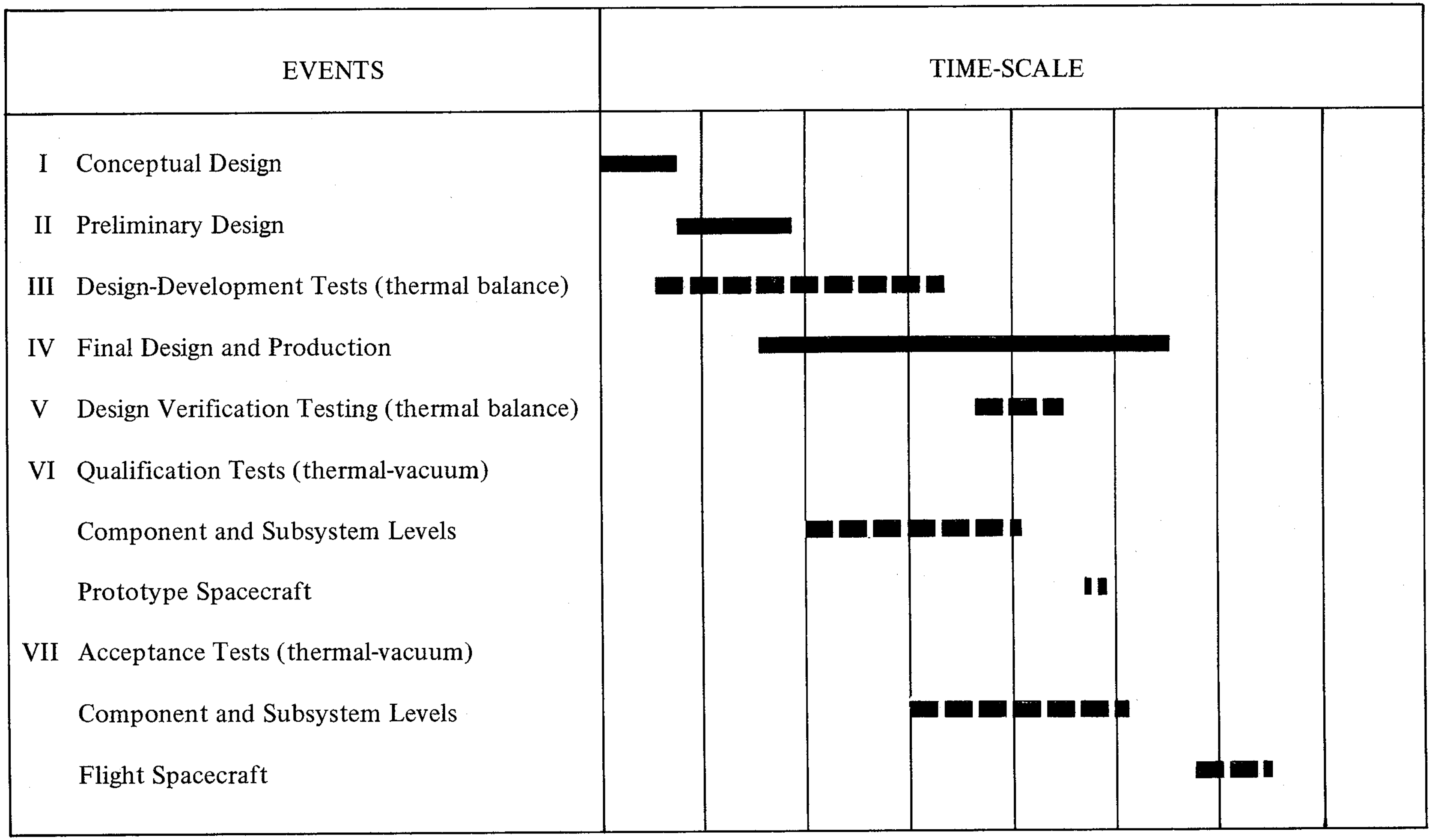
As shown in table 3, the two principal types of testing are thermal balance and thermal-vacuum (soak) tests. The objectives of the thermal balance testing are (1) to evaluate the ability of the thermal control system to maintain the spacecraft thermal environment within established structural, experiment, and subsystem temperature limits and (2) to verify the validity of the analytical model. The objective of the thermal-vacuum tests is to confirm that the system will operate satisfactorily at expected (or more extreme) operating temperatures.

Qualification thermal-vacuum testing has the main purpose of exposing adverse effects on spacecraft performance that result from weakness in thermal design. Acceptance thermal-vacuum testing is to expose adverse effects that result from defects in materials or workmanship related to thermal design. Both types of testing involve collection and analysis of spacecraft performance data; the role of the thermal control engineer is to ensure that the spacecraft is exposed to the specified environments.

TABLE 2
SPACECRAFT THERMAL CONTROL

Type of Orbit	Spacecraft	Attitude (Stabilization)	Design Features (Thermal Control)
Near Earth	Vanguard (ref. 35)	(Spin)	Spherical shape (Reflective surface, vapor-deposited coatings)
	Explorer 1 through 4 (ref. 36)	(Spin)	Cylindrical shape, limited life (Insulation, surface treatment)
	Pegasus (ref. 37)	(None)	(Multi-layer insulation, surface coatings, internal louvers)
	Nimbus 1 (ref. 38)	(Solar and Earth)	Noon orbit (Surface coatings, multi-layer insulation assemblies, louvers)
	RAE (ref. 39)	(Gravity gradient Earth)	(Surface coatings, double wall to minimize internal temperature gradients)
	OSO (ref. 40)	(Spinning Sun-stabilized solar array)	(Surface coatings)
	OAO (ref. 41)	(Solar and stellar)	(Surface coatings, louvers, heaters, multi-layer insulation)
Highly Elliptical	OGO 1 (ref. 42)	(Solar and Earth)	(Multi-layer insulation, surface coatings, adjustable louvers, heaters)
	IMP (ref. 43)	(Spin)	(Black paint, solar cells on cylindrical sides, insulated ends)
Synchronous	ATS-E (ref. 44)	(Spin)	Spinning cylindrical solar array (Insulated ends)
	ATS-F (ref. 45)	(Spin)	(Insulation, louvers, heat pipes)
Lunar	Surveyor (ref. 46)		(Multi-layer insulation, surface coatings, temperature-actuated switches, heaters)
	Lunar Orbiter (ref. 47)	(3-axis) (3-axis)	(Equipment mounted to white painted radiator facing Sun, other faces insulated, thermostatically controlled heaters on certain components)
Planetary	Mariner (ref. 48 and 49)	(3-axis)	(Multi-layer insulation, sun shades, surface coatings, louvers, heaters)
	Pioneer (ref. 50)	Spin axis Earth-oriented (spin)	(Multi-layer insulation, louvers, surface coatings, electric heaters, radio-isotope heaters at remote locations)

TABLE 3
TYPICAL SEQUENCE OF EVENTS IN SPACECRAFT DEVELOPMENT CYCLE



2.5.1 Thermal Balance Testing

Thermal balance testing for design-development is performed to provide design information on those components for which the thermal design is difficult to analyze, stringent temperature constraints are imposed, or it is necessary to establish the feasibility of the design approach. Thermal balance testing also is conducted for design verification.

The thermal balance testing is usually performed on a subsystem as well as the integrated spacecraft. Subsystem testing provides a better understanding of a particular component than the testing of the integrated spacecraft and it allows for higher quality testing at lower costs. It cannot, however, reveal interactions among subsystems.

The thermal balance testing at the spacecraft level can be performed on the fully integrated flight spacecraft, a fully integrated prototype, or a thermal-structural model.

The use of a fully integrated prototype or a thermal-structural model allows for more options in test operations and better instrumentation than if testing is performed on the flight model. Design modifications, if needed, can be more easily accomplished at this relatively early date. The prototype and thermal structural model can also be subjected to a wider temperature range which results in more detailed baseline thermal information. A disadvantage of using a prototype spacecraft is the possibility that the actual flight spacecraft will be changed in design from the prototype. In the case of a thermal-structural model the major disadvantages are differences between the thermal model and the actual flight spacecraft and the normal practice of not using live equipment in the thermal model.

The advantages of testing the flight models are (1) that major changes are not made before launch except when necessitated by test results and (2) flight model testing may be the only opportunity to test the complete spacecraft.

The use of the fully integrated flight spacecraft for thermal balance testing has the following disadvantages: (1) it is too late in the development program for any changes to be made without affecting schedule and cost; (2) the component cannot be tested over a wide enough temperature range to obtain baseline design information because of the temperature range established by mission requirements; (3) facility operation may pose the risk that the allowable temperature limits of the spacecraft would be inadvertently exceeded, and (4) the constraints on test operations and instrumentation to minimize risk of spacecraft damage may hinder testing.

2.5.2 Thermal Vacuum Testing

The thermal-vacuum testing is performed at the component, subsystem, and integrated spacecraft levels. On the component level, the testing is generally performed at the vendor's facility to ensure that the unit meets reliability and quality assurance requirements. On the subsystem level, there are design qualification and flight acceptance tests. The purpose of the design qualification test is to prove the component design by checking its performance capability in vacuum under temperature stress more severe than predicted for the mission. A prototype component is generally used for design qualification testing. The flight acceptance test is performed on a flight model component and its purpose is to locate latent material and workmanship defects in a component of proven design by checking its performance capability under vacuum at the temperature extremes expected in flight.

The purpose of acceptance testing on the flight model of the spacecraft is to check the interaction between subsystems as well as to ascertain the proper operation of all systems.

2.5.3 Test Conditions

The test levels in thermal balance testing are set to simulate the external environment (solar radiation and deep space) or to approximate the anticipated energy flux levels at the boundaries of the spacecraft. These levels are then used in the mathematical model in order to permit valid comparison with test. For thermal-vacuum testing, temperatures are set equal to or higher by some margin than expected flight temperatures. For both types of testing, the component electrical dissipation rates, and duty cycles are set to values appropriate to the mission mode being tested. In some cases, it may be technically and economically advantageous to perform a combined thermal balance and thermal-vacuum test.

There are several different methods for simulating flux in thermal balance testing (ref. 51). Advantages and disadvantages are presented in table 4. Temperature conditioning in thermal-vacuum testing is usually accomplished by varying the test chamber wall temperature or by monitoring the test article on a temperature controlled baseplate.

2.5.4 Test Duration

The duration for thermal balance testing can be determined in two ways: (1) test conditions are established and held until the test article reaches temperature stabilization and (2) test conditions are varied to simulate transient conditions in the same time frame as expected in flight.

Test durations for acceptance thermal-vacuum testing must be long enough to demonstrate that the unit can survive launch and flight. Test times for qualification testing are not as easily defined because testing is not performed on a flight unit and the test levels are more severe than encountered in flight. For information purposes, a typical qualification test cycle curve is presented in figure 16 (from ref. 52).

TABLE 4
HEAT FLUX SIMULATION TECHNIQUES FOR THERMAL BALANCE TESTING

METHOD	ADVANTAGES	DISADVANTAGES
Test Chamber Wall (Temperature Adjustment)	Cheapest simulation method	Incident heat flux is uniform over test specimen
Solar Simulation	Closest to actual environment	Expense
Quartz Lamps	High heat flux rates attainable	Lack of spatial uniformity of heat flux and spectral simulation
Heater Skins	Spatial and temperature variations of absorbed heat flux can be simulated	Complexity of test support equipment prevents verification of energy absorbed by test specimen

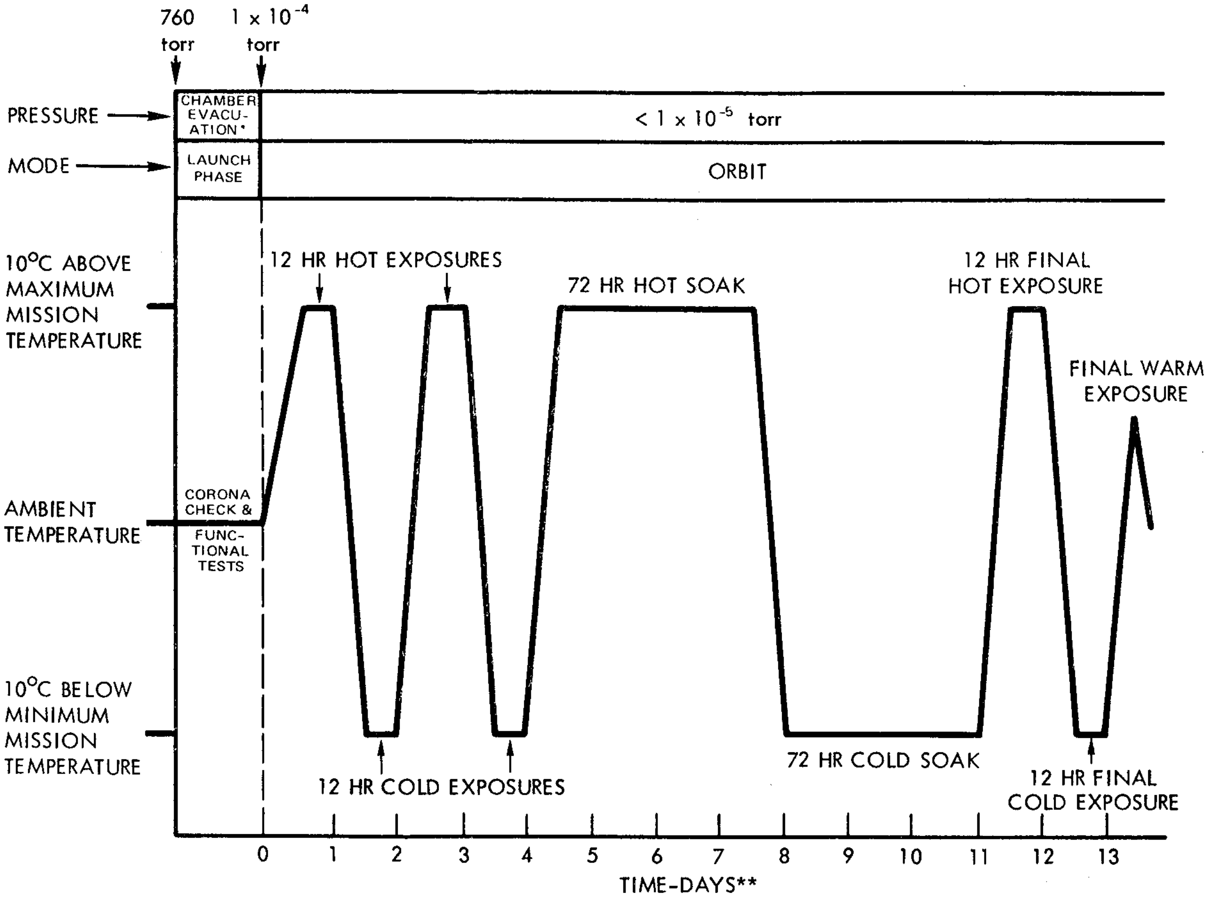
The question of adequate test durations has recently been examined in some detail and the results of one effort which provides some insight are given in reference 53. Other considerations and philosophy are given in reference 54.

2.5.5 Testing Uncertainties

Tests often are performed without regard to possible errors in the test chamber. The consequent test results could dictate an unnecessary redesign or confirm thermal adequacy of a deficient design. Chamber errors result from

- Conduction transfer from fixtures used in mounting and supporting test articles in the chamber
- Infrared energy inputs to test articles from the chamber and reflection from chamber walls and fixtures
- Monitoring errors (calibration and measurement)
- Thermal losses to wiring
- Simulation errors.

Methods of translating the energy errors into temperatures are given in reference 55.



*Chamber evacuation takes about 2 hours so time scale is disproportionate here.
**The plot shows 12 hours for each temperature transition; actually, the transition periods vary widely with spacecraft, mission, and test equipment.

Figure 16.—Typical Thermal-Vacuum Test Cycle for Prototype Spacecraft.

2.5.6 Other Considerations

Several factors that can affect the over-all test program adversely are

- Cost constraints
- Space limitations of the facility
- Constraints on time available for testing
- Degree of simulation attainable –
  - Solar collimation, uniformity and spectrum
  - Earth emission and albedo
  - Aerodynamic heating
  - Gravity effects (especially for heat pipe testing)

To overcome the space limitations of test facilities and cost constraints, consideration has been given to thermal scale modeling. The utility of this technique was demonstrated by use of a 0.43-scale thermal model for the Mariner 4 spacecraft (ref. 56).

3. CRITERIA

In the design of a spacecraft, consideration must be given to the control of the onboard thermal environment to ensure that no subassembly or component is subjected to temperatures which would jeopardize its performance in the mission. To meet this objective, a rigorous temperature control program should be pursued through the planning, design, manufacture, assembly, testing, and launch preparation stages of the flight project.

3.1 Definition of Requirements

A careful review should be made of the spacecraft and its mission, including trajectory and orbit, types of experiments and equipment to be carried, operating cycles and operational lifetime of the spacecraft, spacecraft configuration, attitude control, and primary power system. General requirements for thermal control should be identified from this review.

3.2 Identification of Worst Case Environments

All heat sources contributing to the spacecraft's thermal balance should be identified and their characteristics defined. Sources external to the spacecraft include solar radiation and planetary albedo and emission. The inputs from these sources should be carefully evaluated for all spacecraft locations during the mission and the spacecraft's power dissipation should be similarly analyzed. Worst case environments should then be determined for each heat source and combination of sources.

3.3 Specification of Component Temperature Limits

Component temperature requirements should be accurately determined and an evaluation should be made of the potential effects of all identified heat sources on the local thermal environment of spacecraft subsystems to identify where temperature control is insufficient to satisfy performance and reliability requirements of the mission.

3.4 Temperature Control

After the assessment of potential temperature control problems and the evaluation of the thermal environment, a thermal design effort should be instituted on the basis of priorities and constraints compatible with mission objectives. Candidate thermal design concepts should be developed with emphasis on simplicity. Thermal control should be applied at the spacecraft level and be developed through a combination of analytical and experimental methods.

The thermal design must be coordinated with other spacecraft design activities to facilitate identification of trade-offs in configuration, component material selection, and operational cycles which might reduce the severity of thermal control problems.

The temperature control effort is generally an iterative process. Therefore, to the greatest extent practicable, flexibility should be retained so that the most appropriate thermal control techniques can be applied readily during development of the thermal control system.

3.5 Testing

A test plan should be developed that obtains necessary baseline data, verifies the validity of the analytical design, and demonstrates performance of the thermal control system as specified. The extent of testing should be determined by mission requirements and the uncertainties in the thermal design.

4. RECOMMENDED PRACTICES

It is important that serious thermal control problems be foreseen or predicted as early as possible in the design and development of a spacecraft to facilitate corrective measures. To accomplish this, a total spacecraft approach to thermal control must be initiated at the beginning of the design stage and carried through to the completion of construction, testing, and necessary modifications. This overall program should treat, as quantitatively as possible, the expected thermal environment associated with the particular mission, the acceptability standards of sensitive systems and components, and the full range of practicable methods for treating identified problems.

4.1 Assessment of Spacecraft Requirements

Spacecraft requirements are determined by mission objectives, trajectory and orbit considerations, and spacecraft operational schedules. Consideration of these factors is prerequisite to the design and implementation of a thermal control program. This program should operate throughout design, development and all flight preparations to ensure that temperature problems do not degrade mission performance. Because the thermal control program is an evolutionary process, the various stages in development of the thermal control system, outlined in section 2.2, should allow for maximum flexibility within reason. Identification of needed technology to initiate advanced development should be a primary concern of feasibility studies in the early stages of thermal design because the adequacy of time to develop technology unique to the project depends on early recognition of needs.

Interface and test specifications should be clearly defined and a sensitivity analysis performed to ensure the adequacy of the temperature control measures. Worst case environments should be identified to establish the extreme hot and cold conditions that might be encountered.

4.2 Selection of Thermal Control Techniques

Two general types of methods are used to control spacecraft temperature, passive and active, as described in section 2.3. With passive control, average spacecraft temperature is predetermined primarily by the optical properties of the external surfaces in the presence of the orbital environment of the mission. Active control systems regulate the temperature by adjusting the surface properties or the internal power dissipation.

Passive temperature control is primarily used for spacecraft in fairly circular Earth or lunar orbits. Such spacecraft remain approximately 1 AU from the Sun. In general, passive control is effective when there exists a relative constancy of thermal environment and geometric uniformity of the spacecraft. The basic limitation of passive temperature control is that once the spacecraft is in orbit, the temperature is determined solely by the total energy input rate (internal and external). Typically, energy input rate changes with time and a corresponding range of temperatures is experienced. The design objective in a passive system is to keep this temperature range in allowable bounds. The temperature range is affected by changes in internal heat dissipation and changes in energy input from differing orbital positions. In addition to the changes and uncertainties in average temperatures, there are temperature gradients on external surfaces and among interior parts. There are other uncertainties, too. Measurements of solar absorptance and emittance of thermal coatings may vary, and coatings may be degraded by preflight handling and the launch and space environments.

When it is concluded that the temperature range could be widened by these uncertainties so as to affect the reliability of the spacecraft's functions or when a narrower temperature range is necessary for the operation of a particular component or experiment, active control techniques should be used.

Active control techniques are mandatory for some missions. In interplanetary missions, the combined variations in solar intensity and internal power dissipation are too large for passive methods. Spacecraft in highly elliptical Earth orbits require active techniques to control the diverse environmental inputs they will experience. Active temperature control is also required for a component part of a subsystem with unusual temperature limits necessitating control of temperature or of temperature gradients. For example, to maintain the stable frequency of an oscillator, temperature control to within a fraction of a degree may be necessary. Similar control of temperature gradients in a telescope mount may be required to prevent misalignment of the optics.

4.3 Selection of Materials

Proper selection of materials having the desired conductive, convective, and radiative properties is essential for a successful thermal design. Conductive properties are generally well-defined bulk material properties. There are several materials which can be used to provide conductive paths or needed thermal insulation. Little information, however, is available for conduction across interfaces except for idealized cases. Convective properties for forced convection, which are also fairly well established, are associated with fluids used in active closed temperature control systems. Selection of these fluids should depend on their heat transfer properties in both the gaseous and liquid states and compatibility with their container material.

Radiative properties relate primarily to the thermal coatings and surface finishes used on the spacecraft. Unlike conductive and convective (except for the behavior of fluids in a zero-gravity environment) properties, radiative properties are affected by space environment. Although it is extremely difficult to determine the reaction of a given thermal coating material to the space environment, it is necessary to establish a means of determining the best substances available for a given application. There are several measurable characteristics that should be given careful consideration in the material selection process. Materials to be used for thermal coatings should

- Have reproducible optical properties
- Show high resistance to the space environment
- Adhere well to a variety of substrates, including metals, platings, and plastic with minimum substrate preparation requirements
- Have capability of being cleaned of contaminants, such as particulate dirt, oil, and smudges, preferably through use of solvents or abrasive paste
- Have capability of having optical properties of contaminated surfaces restored to their original values by cleaning without increasing the rate of degradation in space
- Have capability of being cured at room temperature
- Be easily repairable and possess a "touchup" capability with good adherence to either paint or substrate with optical properties similar to the original finish
- Have a long shelf life
- Have minimum tendency to outgas

The discussion of material selection has been restricted so far to the needs of the thermal control system. It should be recognized, however, that there may be requirements associated with other systems of the spacecraft that compete with thermal control. Therefore, a close coordination is necessary between the various spacecraft design groups to select materials that are compatible with efficient thermal control and serve overall spacecraft needs.

4.4 Testing

The thermal design program should include a series of tests to determine the validity of the analytical model and evaluate the performance of the thermal control system in the expected environment. Both thermal balance and thermal-vacuum tests may be necessary. These tests are described in detail in section 2.5. The thermal balance test should be conducted with the spacecraft under vacuum and thermal conditions suitable for evaluating the particular design under consideration. The test may be performed on a thermal model, prototype, or flight spacecraft. When the test has been performed on the thermal model or prototype spacecraft, the flight acceptance test should demonstrate that the proven thermal design has been duplicated in the flight spacecraft.

Temperature conditions in the thermal-vacuum test of the prototype spacecraft should be more severe than the maximum and minimum temperatures predicted for the mission. The purpose of the more severe temperature stress is to demonstrate a design safety margin and accelerate failure in marginal designs. The test should be conducted by forcing extreme temperatures at subsystem locations. This can be accomplished by (1) modifying the operational modes of the spacecraft and (2) adjusting local thermal boundary conditions to provide additional heating or cooling as may be required.

For testing at the subsystem level, special attention should be given to those subsystems which have the greatest sensitivity to adverse temperature conditions or constitute vital parts of the thermal control systems.

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