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SIMPLIFIED THERMAL ESTIMATION TECHNIQUES FOR LARGE SPACE STRUCTURES

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by

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BOEING
AEROSPACE COMPANY

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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FOREWORD

This document was prepared by The Boeing Aerospace Company for the National Aeronautics and Space Administration - Langley Research Center in compliance with Contract NAS1-13967, "Evaluate the Influence of Operational and System Imposed Requirements on the Structural Design of Large Flexible Spacecraft." This report consists primarily of thermal response data generated during the Task I study, arranged to provide the designer of large area spacecraft structures with a simplified approach to thermal analysis.

Dr. Edwin T. Kruszewski was the NASA contracting officer's technical representative and Mr. E. C. Naumann was the assistant technical representative. Boeing performance under the contract was under the management of Mr. J. W. Straayer. Mr. D. L. Barclay was the Technical Leader. Mr. E. W. Brogren performed the thermal investigations and was the principal author of this document.

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LIST OF SYMBOLS

A	Area
AU	Astronomical Unit (Mean distance from sun to earth, 1.4956×10^8 km)
C_p	Material Specific Heat
d	Diameter
F	Radiation View Factor
h	Orbital Altitude
k	Material Thermal Conductivity
L	Length
$\dot{q}$	Heat Flux
T	Temperature
t	Tube Wall Thickness
α	Absorbtivity or Absorptance
ε	Emissivity or Emittance
θ	Angle, defined where used
λ	Angle, defined where used
μ	Angle, defined where used
ρ	Mass Density
σ	Stefan-Boltzmann Constant, 5.775×10^{-11} kW/m^{2}K^4

Subscripts:

a	Location, defined where used
abs	Absorbed
avg	Average
b	Location, defined where used
c	Circumferential; Cylinder
cond	Conduction
e	Emitted; Applicable to Emitted Radiation Spectrum; Earth
eq	Equilibrium
l	Lateral
n	Normal; Normal Component; Effective Incident Intensity
r	Reflected; Applicable to Reflected Radiation Spectrum
s	Solar; Applicable to Solar Radiation Spectrum; Applicable to Shield
t	Transverse; Applicable to Tube

1.0 SUMMARY

The purpose of this document is to provide the designer of large space structures with a tool for making rapid estimates of the response of these structures to the thermal environments encountered in earth orbits. The document is in two major parts, which, although they are closely related, may be used independently of each other.

Part I of the document consists of charts giving heating rates and temperatures for certain typical large spacecraft structural elements, suitable for developing estimates of whole structure temperatures and for screening structural concepts on the basis of their thermal and thermal-mechanical response.

The thermal response of structures in space is dependent upon a number of variables such as orbital parameters, spacecraft orientation, structural geometry, and material properties. Because of the great number of particular combinations of these variables, a comprehensive treatment of thermal response in a short document was not practical. Instead, this document presents data upon which useful estimates may be based by treating one typical structural member geometric shape (right circular cylinders, i.e., tubes and rods), two orbits (low earth and geosynchronous) and two structural materials (anodized aluminum and graphite-epoxy composite). The choices of cases treated were made to bracket most, but not necessarily all, of the orbit-material characteristic combinations that might be encountered in real designs.

Structural member orientations were treated parametrically in generating most of the data of this document and simplifying assumptions were made in order to eliminate certain higher-order effects which would make graphical presentations unwieldy. Earth-emitted and earth-reflected radiation have possibly significant effects on structural thermal response but introduce a number of additional variables to thermal response analysis, making comprehensive handbook-type accounting impractical. For this reason earth radiation effects are presented in only a very general, limited fashion.

Spacecraft on-board heat sources can vary so widely in output levels, distribution, and other characteristics that it was not considered appropriate to attempt to include their effects in a general document of this type. One

chart, which gives simplified structural temperatures versus arbitrary incident radiant heat flux levels, will enable the user to make estimates of the effects of on-board heating.

The user familiar with the principles underlying the generation of the temperature data presented should be able to perform interpolations to form estimates for particular cases not specifically covered by the charts. A section of Part I describes such a procedure. Because of the simplifications employed in generation of the thermal data, however, caution should be exercised in employing them other than as approximations of the structures' true thermal response.

The deflections and stresses that may result from structural thermal response are usually more important than temperatures alone to the performance and integrity of the structure. The great dependence of a structure's deflections and stresses upon configuration and mechanical properties as well as upon thermal and applied load states made impractical the carrying of a general parametric approach all the way to an evaluation of deflections and stresses. The assessment of thermally-induced distortions and stresses requires a mechanical elastic or plastic analysis of the structure, normally performed with the aid of a digital program.

Part II of the document consists of background information for spacecraft thermal design considerations. Environments, requirements, thermal control techniques, design guidelines, and approaches available for more detailed thermal response analyses are discussed. The intent of this material is to aid the designer in interpreting the data of Part I and in qualitatively evaluating design choices as they may influence the thermal reponse of the spacecraft.

2.0 INTRODUCTION

Studies have shown that during the period 1985-2000 there will be need for large structures in orbit. Antennas as large as 100 meters in diameter with wave front error of only 1 mm are desired for use in multi-beam communication systems and microwave radiometry. Needs for power systems approximately 100 meters in size, providing 1 MW of power at 400 watts per kilogram are predicted for solar electric propulsion and space processing. Platforms of the order of 100 meters in size for use as stable foundations, utility stations, and supports for multi-antenna systems are also being considered.

The success of nearly every mission envisioned for large spacecraft is critically dependent upon the maintenance of a stable, close-tolerance geometry. Thus, the prediction and control of structural deformations under the influence of the orbital environment become major design requirements and the candidate structures' characteristic reponses become important considerations in concept selection. Temperature differences and temperature gradients arising from solar and planetary radiant heating and possible on-board heat sources are significant potential causes of structural deformations.

Assessment of thermal environment influence upon large flexible space structures for this document centers around a "thermal influence coefficient" approach, described in Figure 2.0-1. The analogy with the structural analysis technique implied by the phrase is only a vague one and here "thermal" influence coefficients" are simply structural element temperatures, calculated and applied under certain simplifying assumptions.

The intent of the approach was to yield first-order approximations of structural thermal response, suitable for screening structural concepts and developing design guidelines, although not necessarily adequate for preliminary design support.

The broadest possible applicability was desired for the thermal response data summarized here. This breadth was sought through two techniques. First, those properties that affected response more or less independently, e.g., surface emissivity and element orientation, were treated parametrically. Second, those variables whose effect was more complex, e.g., element cross

"THERMAL INFLUENCE COEFFICIENT" APPROACH

- Transient or steady-state thermal response
- Key assumptions:
  - (1) Structural members treated as isolated, independent bodies absorbing solar thermal radiation and emitting infrared radiation.
  - (2) No longitudinal heat flow or temperature gradients due to longitudinal variations in surface or internal properties or dimensions.
  - (3) No reflected or emitted earth radiation. (These may be considered in special cases.)
  - (4) No shading by up-sun members.
  - (5) No conduction through joints to or from other members, components, or on-board heat sources or sinks.
  - (6) No radiant interchange with other members or heat sources or sinks. (Radiant interchange with extensive shielding surfaces may be considered in special cases.)
- Applicable to any orbit but more accurate for geosynchronous or other high orbits.
- Approach is most accurate for open truss structures consisting of slender constant cross section members, remote from extensive surfaces, more massive components, or heat sources.

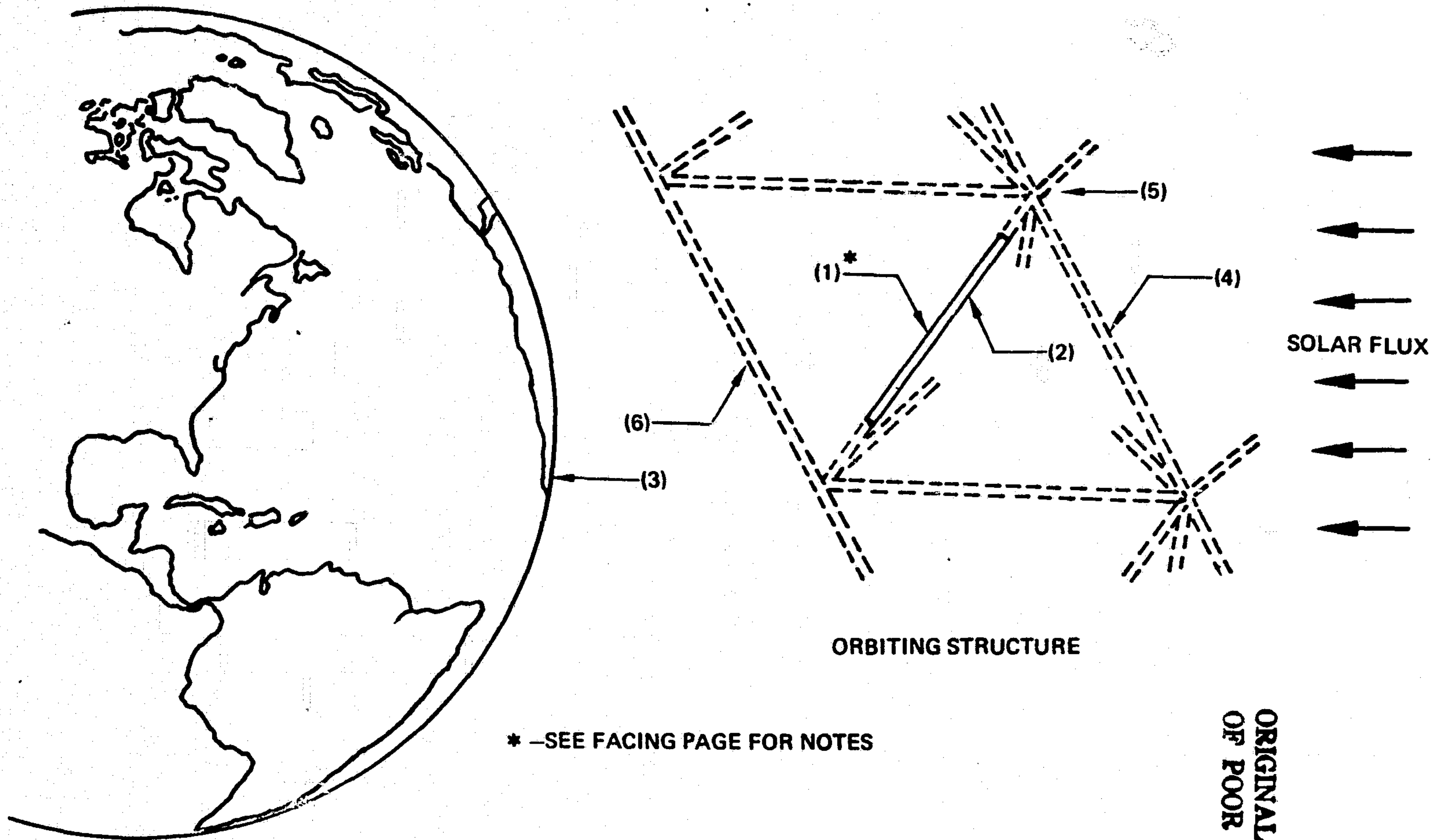

Figure 2.0-1: "Thermal Influence Coefficient" Approach

section geometry, material choice, were treated by considering possible extremes that might be encountered in realistic designs.

3.0 PART I - THERMAL DESIGN DATA

3.1 Introduction

The charts and tables that follow were developed to aid the designer of large space structures in estimating temperatures that those structures are expected to experience due to natural thermal environments in orbit. It must be emphasized that the data are intended for preliminary design and concept evaluation only; detailed design analyses will require more sophisticated temperature predictions.

The thermal analysis approaches used in most of this document are outlined through descriptions of the analytical models in Figure 3.1-1. The two-dimensional model (Fig. 3.1-1(A)) yields distributions of temperature around the circumference of tubes but does not recognize variations along the length of the tube due to longitudinally varying properties or end effects. The simplified two-dimensional case (Fig. 3.1-1(B)) yields a single temperature for the tube cross section. Such an estimate can be quite accurate for rods and tubes with thick or highly conductive walls. For tubes which, in reality, experience significant circumferential temperature gradients, the isothermal cross section estimate closely approximates the actual mean of the cross section distribution. Thus the simplified (isothermal) two-dimension temperatures are useful for estimating longitudinal thermal expansion or contraction. The case (A) temperatures, however, are required for computing thermally induced moments or bending in the tube.

The one-dimensional model (Fig. 3.1-1(C)) yields accurate approximations of temperatures of planar members such as flat reflectors and shields. The zero-dimensional or adiabatic surface model (Fig. 3.1-1(D)) will usually yield an upper bound estimate of the temperatures of surfaces exposed to solar or other major radiation sources. The estimates can be quite accurate for well insulated surfaces that have reached a radiation equilibrium condition. (A case in point is a temperature measured on the insulated aft body skin of a Mercury Spacecraft, shown later in this document in Fig. 3.3-1.)

Except in certain special cases, shading, conduction and radiation heat interchange between members, effects of earth radiation, and effects of on-board heat sources were ignored in generating data for charts of this

SIMPLIFIED THERMAL ANALYSIS MODELS

(A) TWO-DIMENSIONAL

- Most general model of those shown
- Conduction ($\dot{q}_c$) and radiation (Int. $\dot{q}_e$) across section considered
- Longitudinal heat flow assumed zero
- Analyses may be transient or steady-state

(B) SIMPLIFIED TWO-DIMENSIONAL

- No resistance to heat flow across section
- Realistic for high-conductivity members, e.g., Al tubes
- Results: Isothermal tube or rod temperatures, approximations of mean temperatures of non-isothermal tubes

(C) ONE-DIMENSIONAL

- Infinite slab of finite or zero thickness
- Thermal radiation ($\dot{q}_e$) emitted from both surfaces
- Lateral heat flow ($\dot{q}_l$) is zero
- Transverse heat flow ($\dot{q}_t$) finite (thick slab), or infinite (thin film or highly conductive slab)

(D) ZERO-DIMENSIONAL

- Simple balance: Heat absorbed and emitted at point
- No conduction
- No effect of storage or radiation at other locations
- Pertinent material properties: α_s and ε only
- Results: Adiabatic surface or radiation equilibrium temperatures (T_{eq})

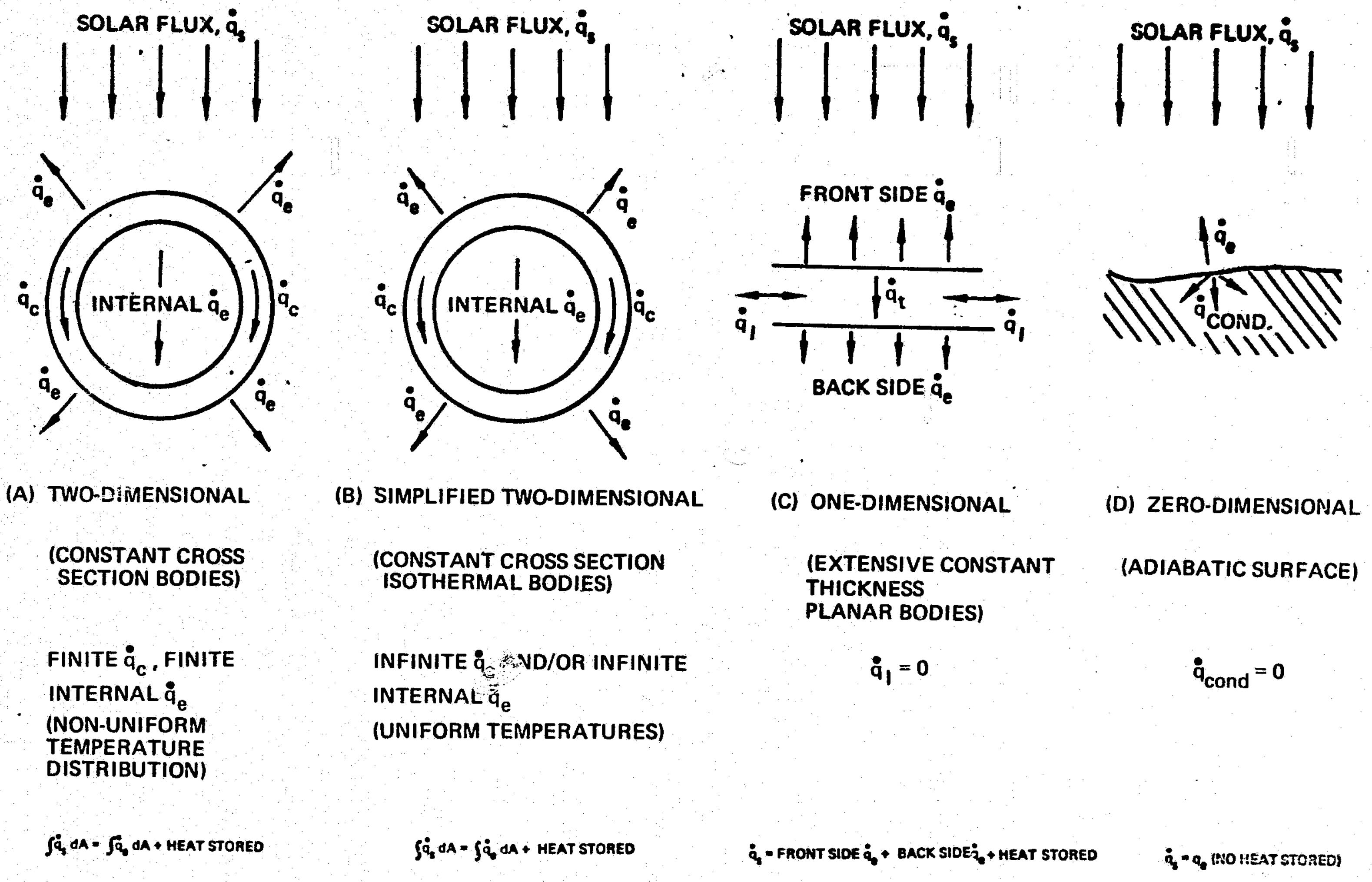

Figure 3.1-1: Simplified Thermal Analytical Models

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document. While most of the neglected effects may not be important to preliminary temperature estimates for large extensive space structures, heating from on-board sources can be important and for certain spacecraft or spacecraft subassemblies, could be the major thermal influence. The stated and implied limitations of the simple models of Figure 2.0-1 and 3.1-1 should be recognized in assessing the applicability of the data to real designs.

More rigorous thermal analysis approaches, which are not suited to handbook-type presentation, are discussed in Part II of this document. Criteria are presented there which will aid in determining whether the approximations of Part I are adequate or a more sophisticated analysis is required.

Most of the graphic presentations of data in Part I of this document are accompanied by facing pages explaining the application of the respective charts. The intent of the material on the facing pages is to minimize the need for the user to refer to main text of the document to make proper use of the charts.

Thermal response analyses requiring, for practical purposes, machine computation, were performed with the aid of the Boeing Engineering Thermal Analyzer (BETA) program (Reference 1). The BETA program uses finite difference techniques to solve transient or steady state thermal diffusion problems in 2 or 3-dimensions. All modes of heat transfer can be modeled and the program has broad capabilities regarding boundary conditions, heat path networks, and material properties.

3.2 Thermal Environment

Approximate levels of natural thermal radiation are described in a number of published sources, e.g., Reference 2, and are summarized in Figures 3.2-1 through 3.2-3. The values shown are total infrared and visible radiation, appropriate to use for thermal response analysis. The spectral distribution of earth reflection radiation is similar to that of the incident solar radiation. Therefore, for purposes of heating estimates at the level of accuracy of this document, materials' solar absorptance values may be used to evaluate thermal response to earth reflection radiation as well as direct solar radiation. Earth emitted radiation is a different matter, however, and a materials' absorptance appropriate for radiation from a black body source of approximately 289K should be used to evaluate response to earth emitted radiation.

3.3 General Radiation Equilibrium Temperature Data

The term radiation equilibrium temperature or adiabatic surface temperature will be used in this document to describe surface response temperatures computed by considering only the balance between incident radiation absorbed and heat radiated away by virtue of the surfaces temperature. Since such computation accounts for the heat balance at only a surface point, it has been referred to as a "zero-dimensional analysis" (Fig. 3.1-1(D)). Resulting temperatures are realistic only for bodies whose thermal conductivity is such that heat conducted from or to the irradiated surface is negligible relative to the heat radiated away from the surface. The only material properties upon which radiation equilibrium temperatures depend are the surface solar absorptance (α_s) and the emittance (ε), or more generally, the ratio of these two properties.

For thin-wall structural members and members constructed of low conductivity materials in a steady radiation environment, radiation equilibrium temperatures (T_{eq}) can reasonably approximate the true temperatures. In the presence of significant levels of incident radiation, T_{eq} values are usually higher than the true temperatues of irradiated surfaces. Values of T_{eq} are particularly useful in comparing effects of different α_s and ε values and effects of different levels of incident radiation.

The following example will illustrate both the use of T_{eq}'s for assessing α_s/ε and incident radiation effects and the procedure for hand calculation of simple zero-dimensional radiation equilibrium (Fig. 3.1-1(D)) temperatures. Let an uncoated aluminum surface with α_s = 0.30 and ε = 0.10 and an effectively adiabatic (perfectly insulated) back surface be exposed to solar radiation (1.39 kW/m^2) at an incidence angle, λ_s, of $\pi/6$ radians. Assume the combination of aluminum thickness and exposure time is such that thermal equilibrium has been reached. The equilibrium radiation temperature, T_{eq}, is found by:

$$\dot{q}_{in} = \dot{q}_{out}$$

$$\dot{q}_s \, \alpha_s \cos \lambda_s = \sigma\varepsilon \, T_{eq}^4$$

SOLAR HEATING

- The figure shows the simple cosine relationship between ambient solar flux intensity at 1.0 AU ($\dot{q}_s$) and the intensity incident upon a plane surface ($\dot{q}_n$), i.e.,

$$\dot{q}_n = \dot{q}_s \cos \lambda$$

- The heat absorbed by the surface is

$$\dot{q}_{abs} = \alpha_s \dot{q}_n$$

- Seasonal variations in $\dot{q}_s$, effects of the divergence of the flux, and basic uncertainty yield a ±4.2% tolerance on the curve. (Reference 2)

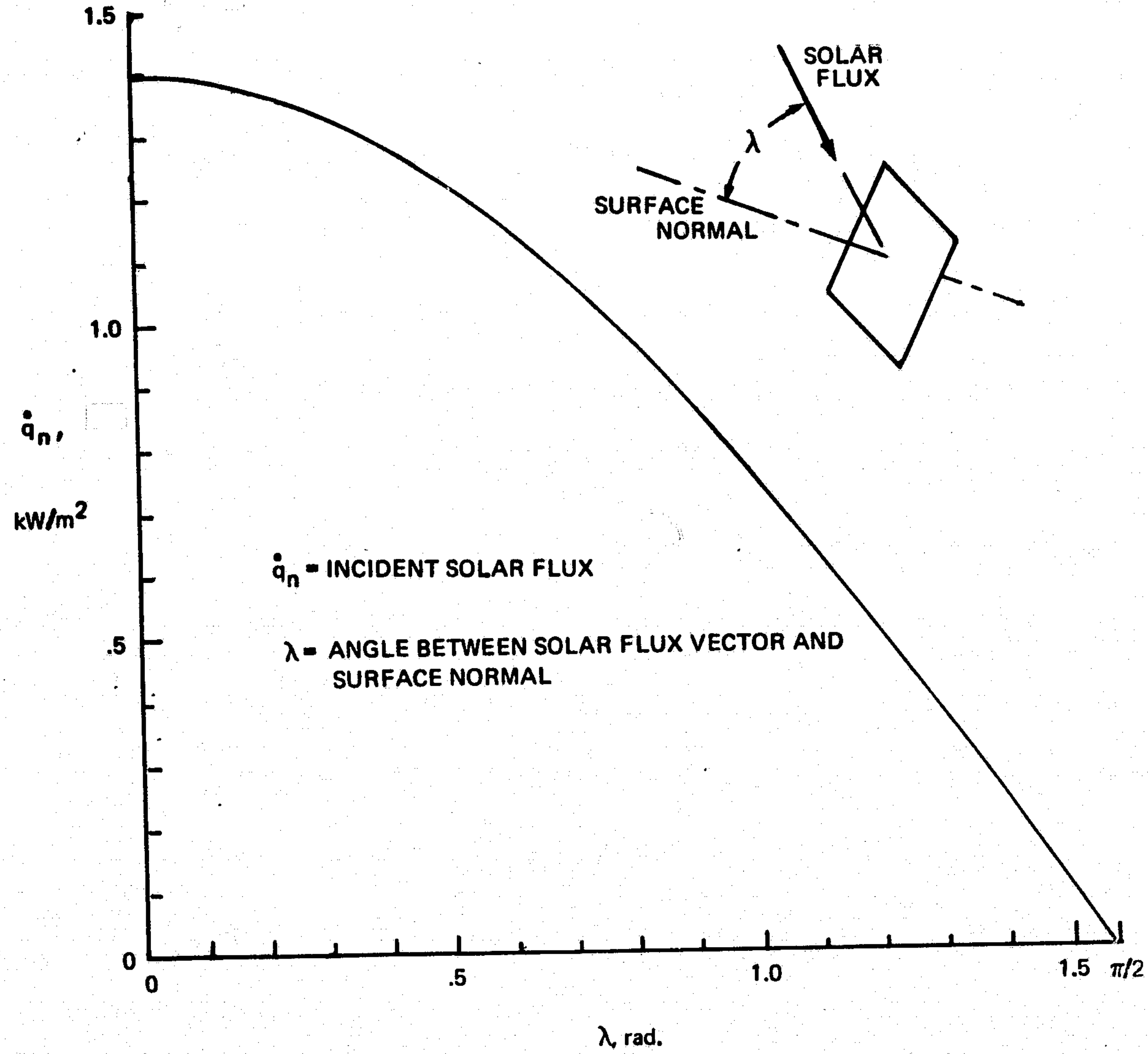

Figure 3.2-1: Solar Heating

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EARTH EMISSION HEATING

- Heat received at the satellite surface is

$$\dot{q}_{n,e} = F\dot{q}_e, \text{ where}$$

$\dot{q}_e$ = flux emitted at the effective earth surface

F = radiation view factor - isothermal sphere to planar element

F = function of h and λ_e

- Heat absorbed by the satellite surface is

$$\dot{q}_{abs} = \alpha_e \, \dot{q}_{n,e}$$

- $\dot{q}_e$ varies diurnally, seasonally, and locally, but not by large amounts.

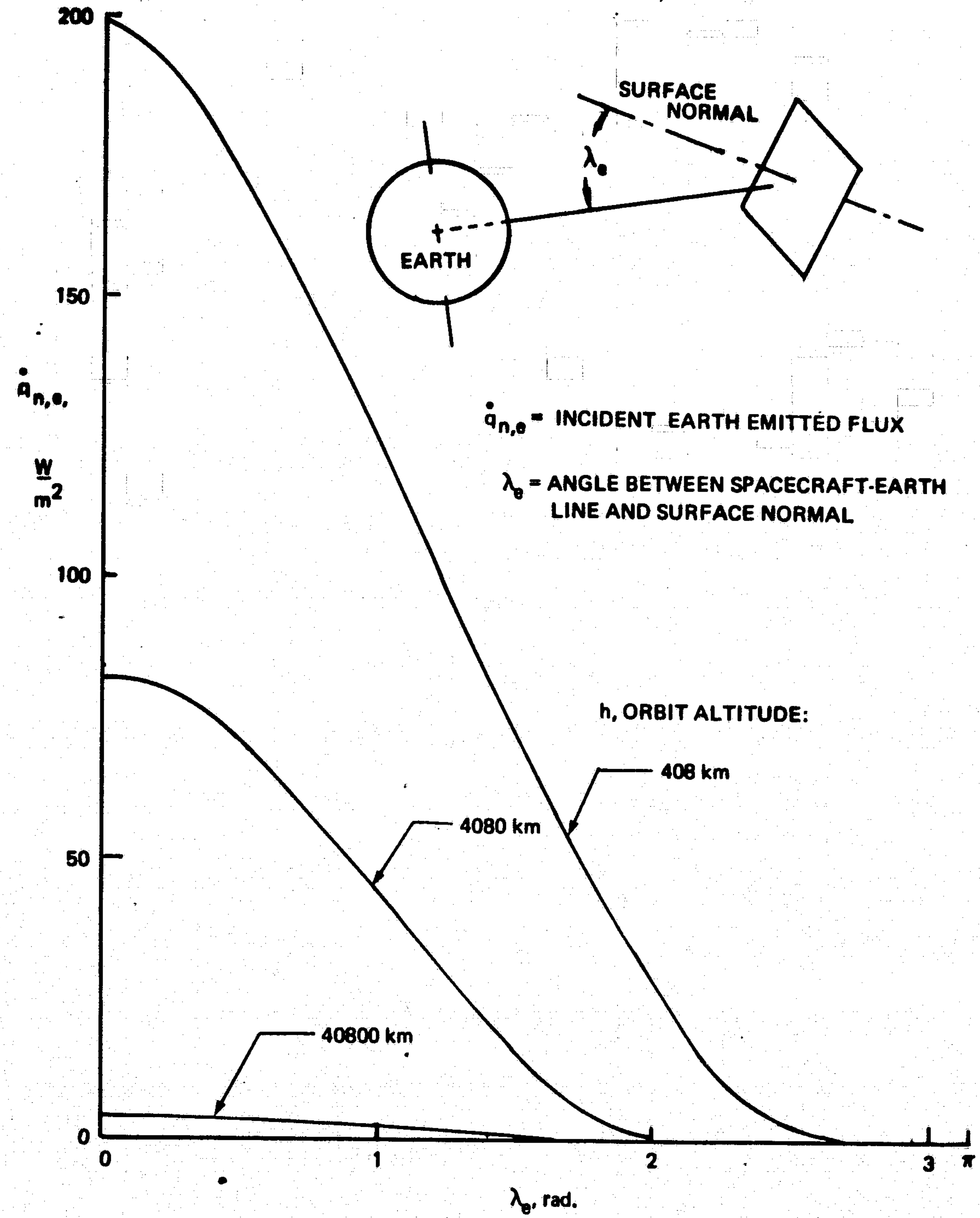


Figure 3.2-2: Earth Emission Heating

EARTH REFLECTION (ALBEDO) HEATING

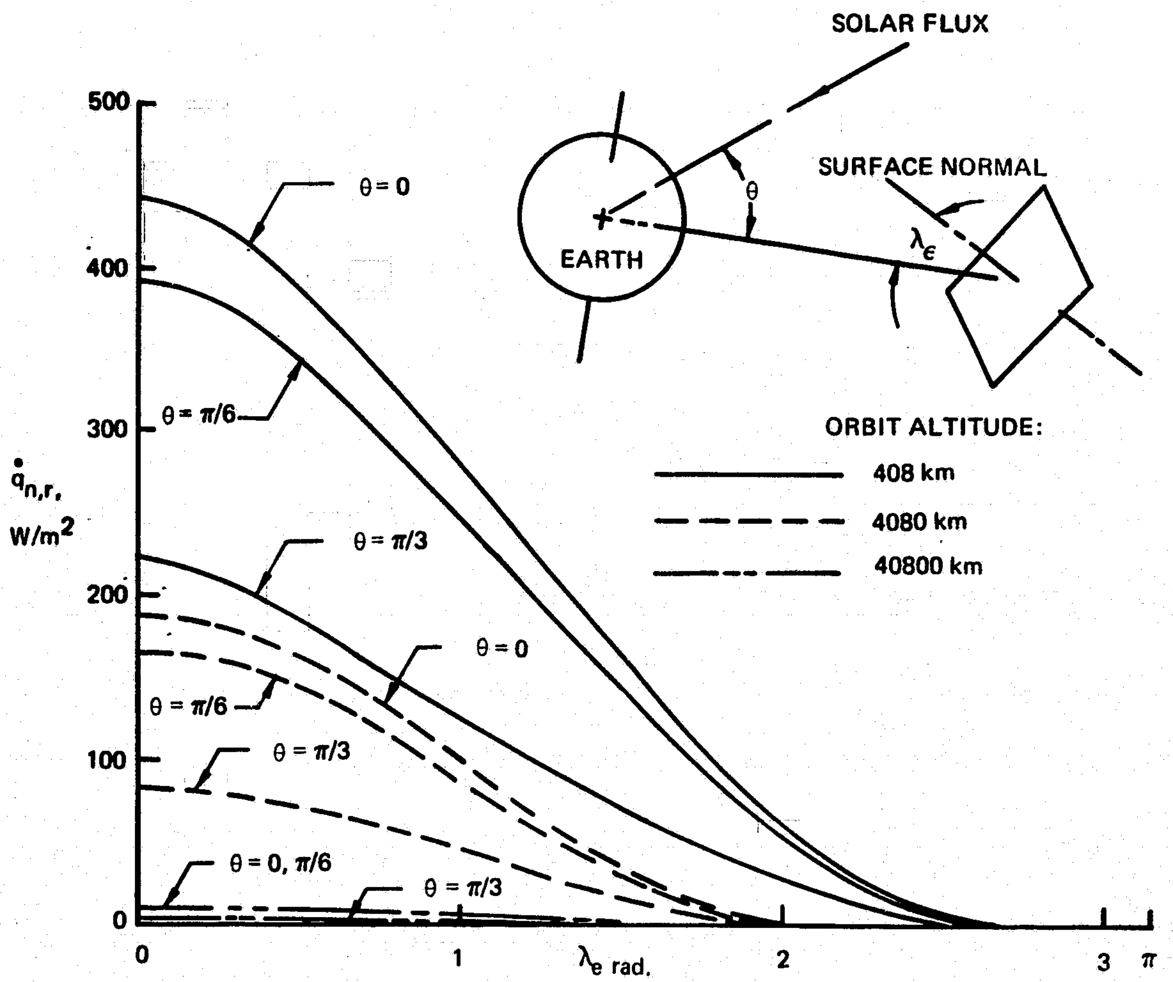
- Reflected thermal radiation based on earth albedo (reflectivity) of 0.36 (Ref. 2) and assumed diffuse reflection from earth surface.

- Heat absorbed by the satellite surface is

$$\dot{q}_{abs} = \alpha_r \, \dot{q}_{n,r}$$

$$\alpha_r \approx \alpha_s \text{ for most applications}$$

- Local value of albedo can vary significantly from the average assumed, due to earth surface character and cloud cover. Reflection may also deviate significantly from the diffuse condition.



$\dot{q}_{n,r}$ = INCIDENT EARTH REFLECTED FLUX

λ_e = ANGLE BETWEEN SPACECRAFT-EARTH LINE AND SURFACE NORMAL

θ = ANGLE BETWEEN SOLAR FLUX VECTOR AND SPACECRAFT-EARTH LINE

Figure 3.2-3: Earth Reflection (Albedo) Heating

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$$T_{eq} = \left(\frac{\dot{q}_s \cos \pi/6}{5.775 \times 10^{-11}}\right)^{.25} \left(\frac{\alpha_s}{\varepsilon}\right)^{.25}$$

$$= 500 \text{ K}$$

If the surface is subjected to an additional arbitrary heat flux of 0.20 kW/m^2, with a normal incidence and a spectral character such that α_s still applies, the T_{eq} becomes:

$$T_{eq} = \left(\frac{\dot{q}_s \cos \pi/6 + .20}{5.775 \times 10^{-11}}\right)^{.25} \left(\frac{\alpha_s}{\varepsilon}\right)^{.25}$$

$$= 520 \text{ K}$$

If the surface is now coated with a white thermal control paint, for which $\alpha_s = 0.28$ and $\varepsilon = 0.90$, and subjected to the original solar-flux-only environment, the T_{eq} becomes:

$$T_{eq} = \left(\frac{q_s \cos \pi/6}{5.775 \times 10^{-11}}\right)^{.25} \left(\frac{.28}{.90}\right)^{.25}$$

$$= 284 \text{ K}$$

The curves of Figures 3.3-1 through 3.3-3 were generated by such calculations.

Radiation equilibrium temperatures are shown in Figure 3.3-1 as a function of total normal component of incident radiation and surface absorptance/emittance ratio. Material categories are indicated for various portions of the α_e/ε range but caution should be exercised in categorizing any particular material, and its actual α_s/ε range should be ascertained and used if possible.

Two data points illustrate the use of the chart of Figure 3.3-1. A temperature of 372K was measured on the aft body of one of the Mercury spacecraft at a point receiving normal solar radiation at 1.39 kW/m^2. The inferred α_s/ε value of approximately 0.8 appears consistent with the expected value for the oxidized metal skin at that location.

The earth experiences complex radiation phenomena with its partially transparent atmosphere and variable surface and weather conditions. As a body, however, it is in approximate thermal equilibrium with its environment and is said to have an average solar absorptance of approximately 0.64 and an average total emittance of approximately 0.56 (Ref. 2). The combination of these values with an average (taken over the entire sphere, night side included) of the normal component of solar flux of 0.348 kW/m^2 yields a T_{eq} of 289K, almost exactly the accepted average sea level standard atmosphere temperature.

In order to provide an indication of the range of temperatures that spacecraft surfaces might experience in low earth orbit, T_{eq} values were evaluated for a simple representative body through a complete orbit. The body was a planar slab with the surfaces on the two sides assumed perfectly insulated from each other. Each surface is thus equivalent to the model of Figure 3.1-1. Two orientations of the slab were considered: in one the slab continuously faces the sun, in a manner representative of a solar collector or cell array; in the other the slab continuously faces the earth, possibly representing an earth sensor array surface or antenna.

For a number of positions in orbit, for both slab orientations and for both surfaces of the slab, the incidence angles, λ, of the solar, earth-emitted, and earth-reflected flux upon each surface were evaluated. Then, using the 408 km orbit altitude curves of Figures 3.2-1, -2, and -3, the incident flux levels were tabulated as a function of position in orbit. Finally, T_{eq} values were read from Figure 3.3-1 at the total incident flux value and plotted as functions of position in orbit. The results are shown in Figure 3.3-2.

The process described in the preceding paragraph was repeated for geosynchronous orbit altitudes by using the 40800 km orbit altitude curves of Figures 3.2-1, -2, and -3. These results are shown as Figure 3.3-3.

For the curves of Figures 3.3-2 and 3.3-3 $\alpha_S/\varepsilon = 1.0$ was assumed but it is clear from Figure 3.3-1 that other α_S/ε values could be treated in the same way. Geometric relations for computation of λ values were based on simple circular ecliptic-plane orbits. In Figures 3.3-2 and 3.3-3 surfaces receiving no incoming radiation appear to experience a radiation equilibrium temperature of absolute zero. Because of almost certain significant heat conduction

RADIATION EQUILIBRIUM TEMPERATURES

- Temperatures were computed for the zero-dimensional case (adiabatic surface). See Fig. 3.1-1(D).

- Material classifications of α_s/ε ranges are general approximations. Obtain value for particular material before using the chart.

- Application of the chart to the earth, treated as an isothermal body with $\alpha_s/\varepsilon = 1.14$, yields a temperature of 289 K, the mean temperature of the sea level atmosphere.

- The chart accurately predicts a maximum temperature measured on the Mercury Spacecraft for $\alpha_s/\varepsilon = 0.80$.

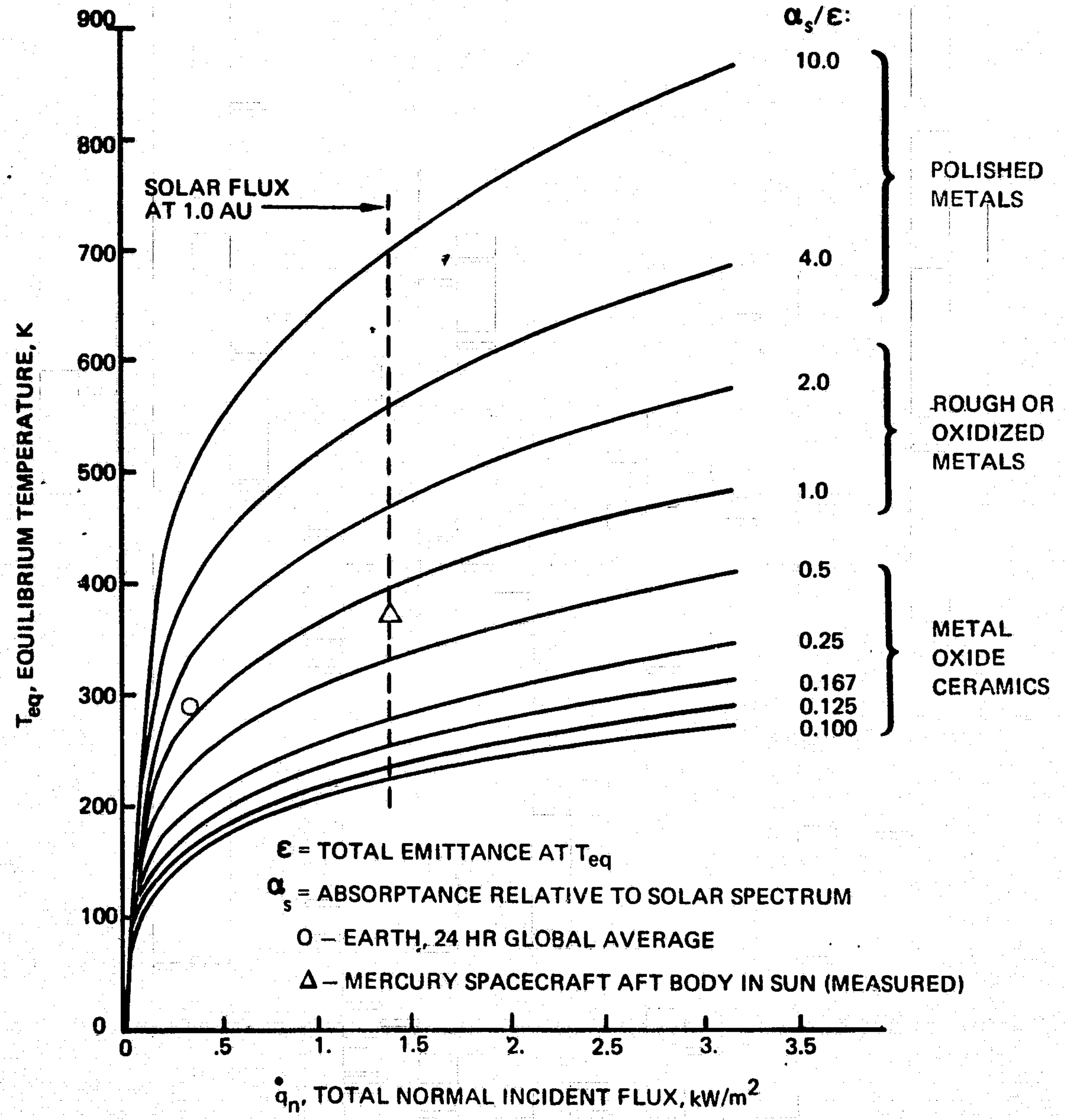

Figure 3.3-1: Radiation Equilibrium Temperatures

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RADIATION EQUILIBRIUM TEMPERATURES FOR ADIABATIC SURFACES IN LOW EARTH ORBIT

- Graph illustrates character of orbital temperature variations. Charts of Sec. 3.4 may be better suited for predicting temperatures of particular structural elements.

- Temperatures were computed for the zero-dimensional case (adiabatic surface). See Fig. 3.1-1(D).

- Temperatures are for $\alpha_s/\varepsilon = 1.0$ only.

- Analysis approach yields unrealistic results (temp. = absolute zero) for surfaces simultaneously facing away from sun and earth.

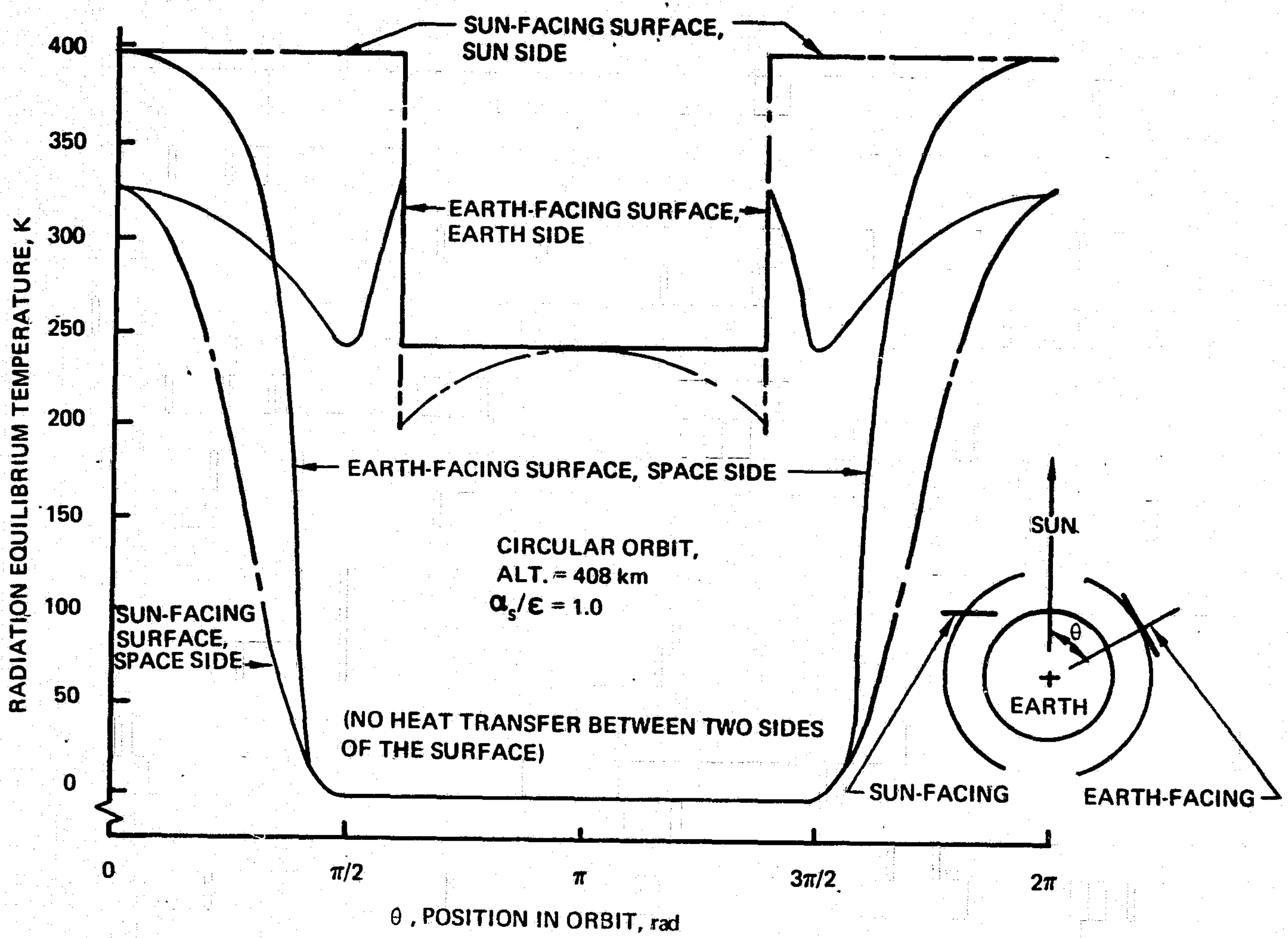

Figure 3.3-2: Radiation Equilibrium Temperatures For Adiabatic Surfaces In Low Earth Orbit

RADIATION EQUILIBRIUM TEMPERATURES FOR ADIABATIC SURFACES IN GEOSYNCHRONOUS ORBIT

- Graph illustrates character of orbital temperature variations. Charts of Sec. 3.5 may be better suited for predicting temperatures of particular structural elements.

- Temperatures were computed for the zero-dimensional case (adiabatic) surface). See Fig. 3.1-1(D).

- Temperatures are for $\alpha_s/\varepsilon = 1.0$ only.

- Analysis approach yields unrealistic results (temp. = absolute zero) for surfaces simultaneously facing away from sun and earth.

- Penumbra effects were ignored.

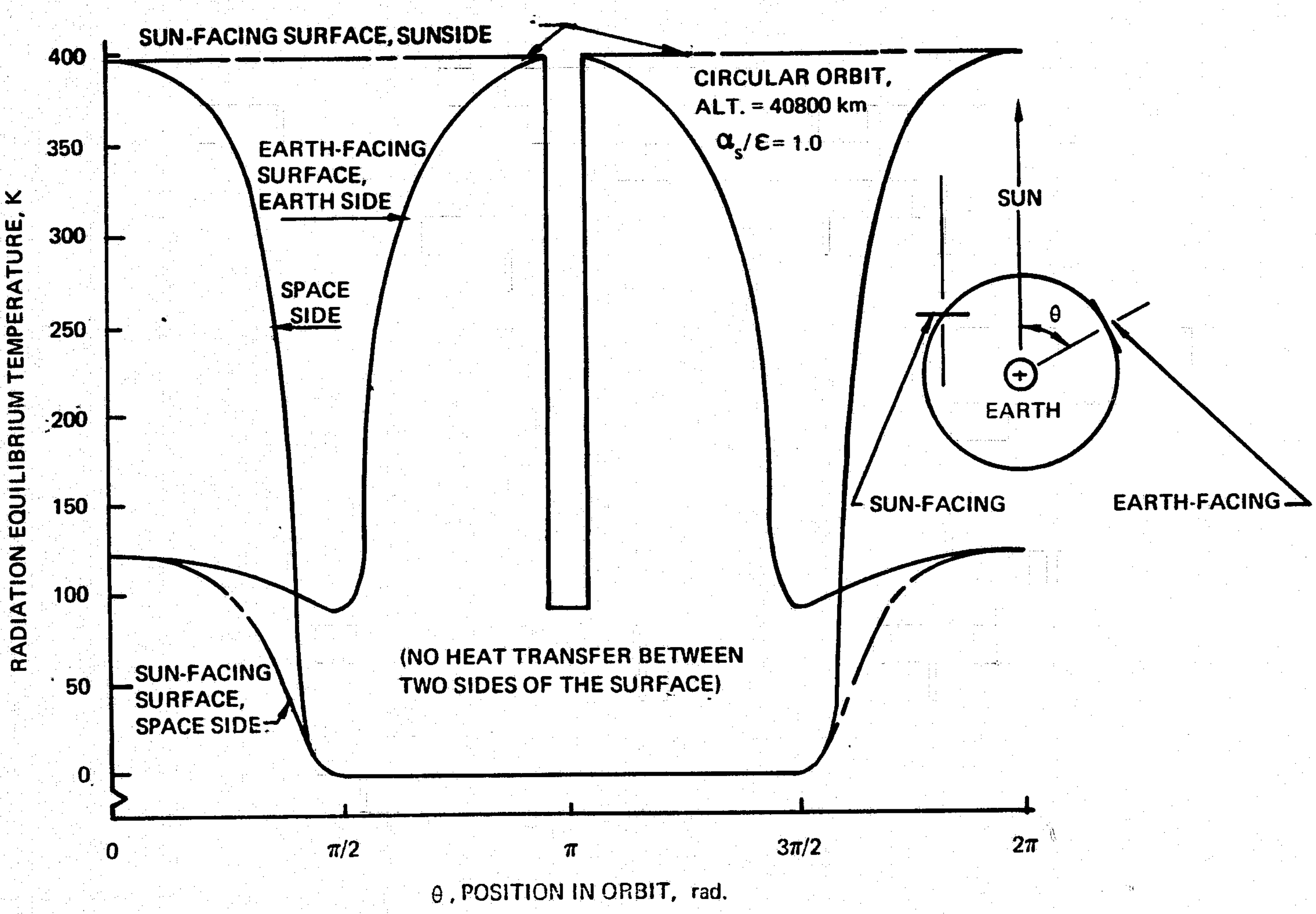

Figure 3.3-3: Radiation Equilibrium Temperatures For Adiabatic Surfaces In Geosynchronous Orbit

and radiation from other parts of the satellite and thermal capacitance of the element, such a temperature is not realistic. Unfortunately, there exists no simple means of estimating a reasonable minimum temperature in orbit, corresponding to the radiation equilibrium approach for conditions when significant heat input exists.

3.4 Low Earth Orbit Transient Temperatures

Many structural elements cannot be expected to reach steady-state temperatures in orbit due to periodic eclipsing. This is particularly probable for elements of relatively high thermal capacitance in low earth orbit. Also, as pointed out in the preceding section, the steady-state or radiation equilibrium approximations are not realistic techniques for estimating minimum element temperatures occurring in earth shadow.

Transient temperatures for this document were computed with the aid of the Boeing Engineering Thermal Analyzer (BETA) program (Ref. 1). The computation took into account the thermal capacitance of the structural members and thermal conduction and radiation in the plane of the members' cross section. Heat transfer along the members' length or heat interchange with other structural members was not considered. In addition to temperatures for isolated members (Fig. 2.0-1), data are presented for certain special cases of structural members in close proximity to extensive opaque, reflecting surfaces, where consideration of radiant interchange between the structural member and the opaque surface was essential to accurate temperature prediction.

The data of the charts of this section are applicable with greatest accuracy to constant cross section members whose length is large with respect to their cross section dimensions and to members without far reaching end effects or significant thermal interaction with other members or components. Although the transient temperature data were generated for particular members having the geometric and material properties noted on the charts or listed in Table 3.4-1, the chart transient temperatures are also applicable to other members whose properties satisfy the following three conditions:

Table 3.4-1: Material Properties For Thermal Response Charts

ALUMINUM			
6061 T-6 WITH HEAVILY ANODIZED SURFACE			
TEMP. (K)	THERM. COND. (W/m-K)	SPECIFIC HEAT [kJ/(kg K)]	
0	74.72	.004197	
20	224.2	.01256	
30	194.3	.04187	ρ = 2713 kg/m^3
90	190.5	.4396	α_s = 0.42
200	160.0	.7411	ϵ = 0.84
260	152.0	.8214	
300	151.9	.8750	
370	160.3	.9361	
420	162.5	.9797	
480	165.1	1.005	
530	167.4	1.026	
590	168.9	1.044	
640	170.0	1.076	
700	171.9	1.114	

GRAPHITE-EPOXY COMPOSITE			
60% FIBER VOLUME; 50% AXIAL PLIES, 50% ±45° PLIES			
TEMP. (K)	CIRCUMFERENTIAL THERM. COND. (W/m-K)	SPECIFIC HEAT [kJ/(kg K)]	
0	~ 0	.000419	
120	3.884	.338	ρ = 1633 kg/m^3
170	5.993	.479	α_s = .916
220	8.032	.620	ϵ = .80
270	9.714	.783	
330	10.14	.976	
400	11.14	1.08	
810	16.98	1.66	

(1) $$\left(\frac{\alpha_s}{\varepsilon}\right)_{\text{new member}} = \left(\frac{\alpha_s}{\varepsilon}\right)_{\text{chart member}}$$

(2) $$\left(\frac{kt}{d}\right)_{\text{new member}} = \left(\frac{kt}{d}\right)_{\text{chart member}}$$

(3) $$\left(\frac{\rho\ C_p\ t}{\alpha_s}\right)_{\text{new member}} = \left(\frac{\rho\ C_p\ t}{\alpha_s}\right)_{\text{chart member}}$$

Condition (3) may be ignored if steady-state thermal response can be assumed. Condition (2) may be ignored for members assumed to have isothermal cross sections (Fig. 3.1-1(B)). Effects of deviations from Condition (1) may be estimated using Figure 3.3-1 in a procedure described in Section 3.6.

A circular orbit with 463 kilometer altitude and an inclination of 0.497 radians, a nominal space shuttle orbit, was selected as a representative case for computation of low earth orbit temperatures. Moderate deviations from these orbit parameters will not significantly change the thermal response characteristics for purposes of this document. Orientation of structural members with respect to the incident solar radiant flux, however, has an important effect on temperatures. The orientation was thus treated as a variable or parameter in the transient temperature charts.

Four representative classes of structural members are treated in the charts of this section; bare aluminum tubes, bare graphite-epoxy composite tubes, insulated tubes, and shielded tubes. These classes were selected as examples to characterize the possible range of thermal response, particularly with regard to temperature gradients and thermal lag in response to changes in environments. Values of α_s/ε different from those indicated on the charts can lead to significantly different member response temperatures. The value of α_s/ε is not treated as a parameter in the charts of this section but the effects of α_s/ε variations may be estimated by use of a procedure described in Section 3.6.

The temperatures of the aluminum tubes, with that material's high thermal conductivity, are not particularly sensitive to wall thickness-to-diameter ratios. Consequently, only two cases of bare aluminum tubes are treated; a thin-wall tube (Figures 3.4-1 and 3.4-2) and a "tube" whose wall is one half the member's diameter, i.e., a rod (Figures 3.4-3 and 3.4-4).

Three graphite-epoxy configurations are presented; a thin-wall tube (Figures 3.4-5 and 3.4-6), a tube with a moderately thick wall (Figures 3.4-7 and 3.4-8), and a rod (Figures 3.4-9 and 3.4-10). Nominal properties for the aluminum and the graphite-epoxy tubes, used to compute the thermal response data, except where otherwise noted, are given in Table 3.4-1.

The aluminum tubes were chosen as representing one extreme of candidate structural materials, having high thermal conductivity (resulting in low temperature gradients) but having high thermal coefficients of expansion. The bare aluminum tubes were assumed to be heavily anodized, resulting in a low α_s/ε, leading to low temperatures in the solar environment. Thermal properties were those of 6061 T-6 alloy. The graphite-epoxy composite was selected as representative of the opposite extreme of structural candidate characteristics. Thermal conductivity is low but the particular composite assumed was one formulated for minimum thermal expansion. No surface coating was assumed, resulting in a α_s/ε slightly greater than unity.

The two remaining classes of structural members covered by the transient temperature data are insulated tubes (Figures 3.4-11 and 3.4-12) and shielded tubes (Figures 3.4-13 through 3.4-15). The insulated tube data are applicable, within limits, to a number of different structural member materials and geometries and also to a variety of insulation schemes. The shielded tube cases include data for both aluminum and graphite epoxy composite tubes, but since the inclusion of a shield introduces additional material properties and may necessitate consideration of earth reflected and emitted flux, the shielded tube charts should be recognized as only special cases out of a possible wide variety of material and configuration combinations.

Additional radiation equilibrium temperatures for cylindrical bodies in the proximity of planar shielding surfaces may be calculated by the following procedure, provided the system satisfies the listed assumptions:

(1) Shield and cylinder in thermal equilibrium (steady-state radiation) and are opaque gray bodies

(2) Cylinder is isothermal in length and cross section

TRANSIENT THERMAL RESPONSE -
ANODIZED ALUMINUM TUBE ELEMENT, LOW EARTH ORBIT

- Temperatures were computed for the transient two-dimensional case. See Fig. 3.1-1(A).
- Isothermal tube analysis (Fig. 3.1-1(B)) would yield almost identical results for this tube.
- Material properties are from Table 3.4-1.
- Earth emitted and reflected radiation ignored.
- Results valid for any d/t = 25.

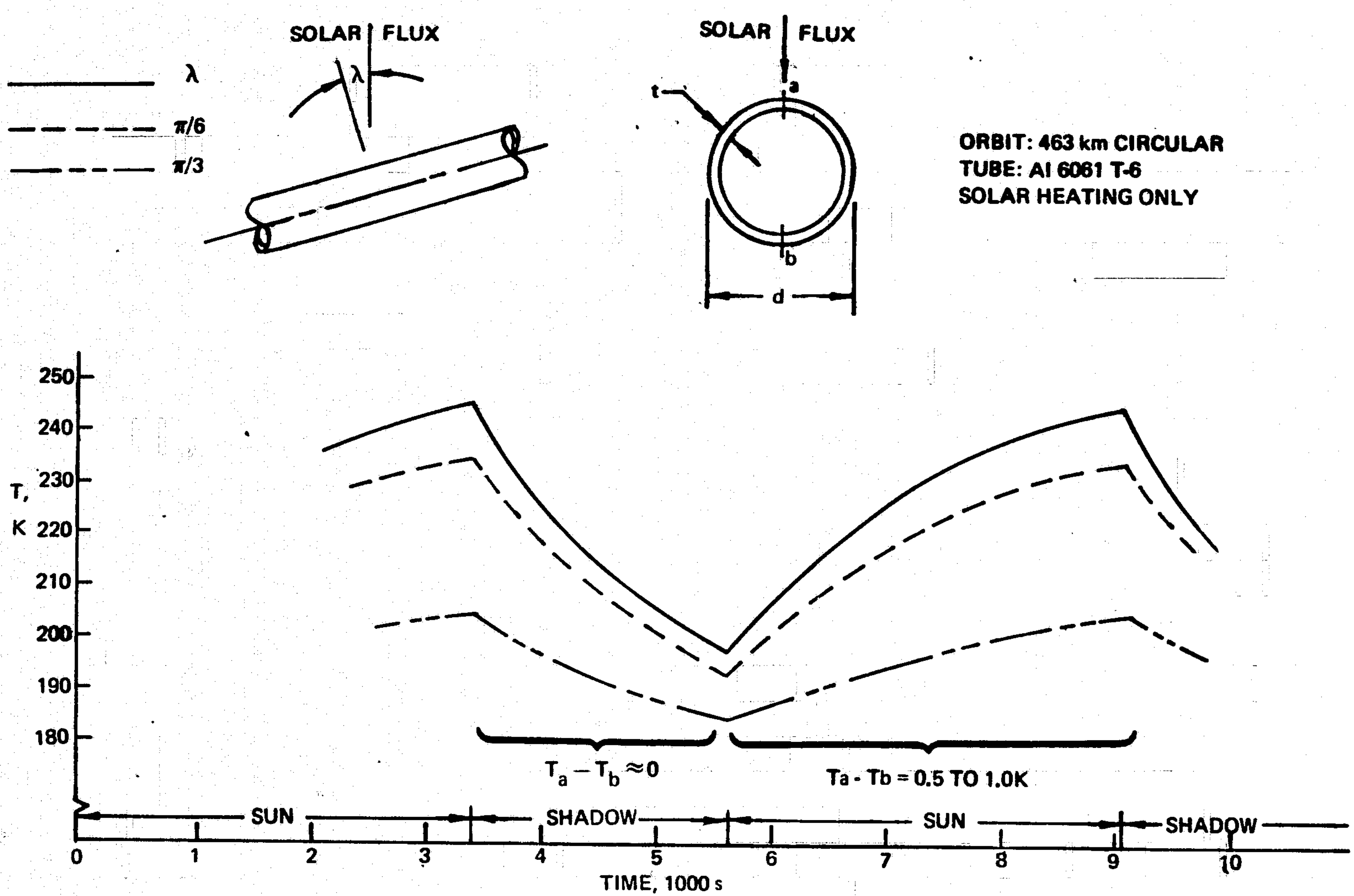

Figure 3.4-1: Transient Thermal Response—Anodized Aluminum Tube Element, Low Earth Orbit, d/t = 25

SUMMARY OF TEMPERATURE EXTREMES - ALUMINUM TUBE ELEMENT, LOW EARTH ORBIT

- Envelope of maximum and minimum values from Figure 3.4-1.
- Isothermal tube values for steady-state in sunlight shown for comparison.
- Temperatures at values of λ approaching $\pi/2$ not realistically predictable by isolated element approach because of probable significant influence from adjoining or nearby members, heat sources, etc.

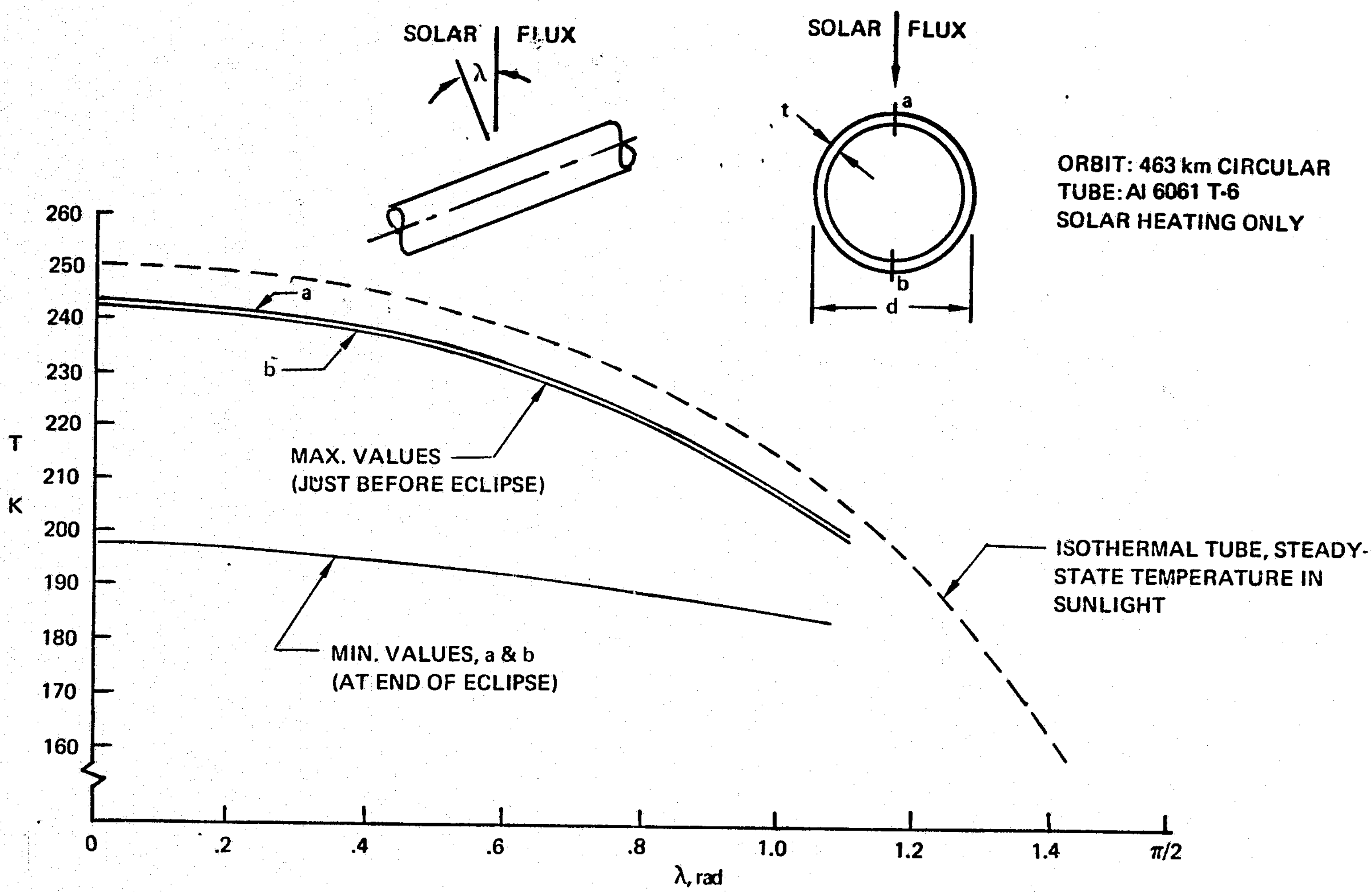

Figure 3.4-2: Summary of Temperature Extremes—Anodized Aluminum Tube Element, Low Earth Orbit, d/t = 25

TRANSIENT THERMAL RESPONSE - ANODIZED
ALUMINUM ROD ELEMENT, LOW EARTH ORBIT

- Temperatures computed for the transient isothermal cross section case. See Fig. 3.1-1(B).
- Results useful as approximations for rods from 1/2 to twice the indicated diameter.
- Material properties are from Table 3.4-1.
- Earth emitted and reflected radiation ignored.

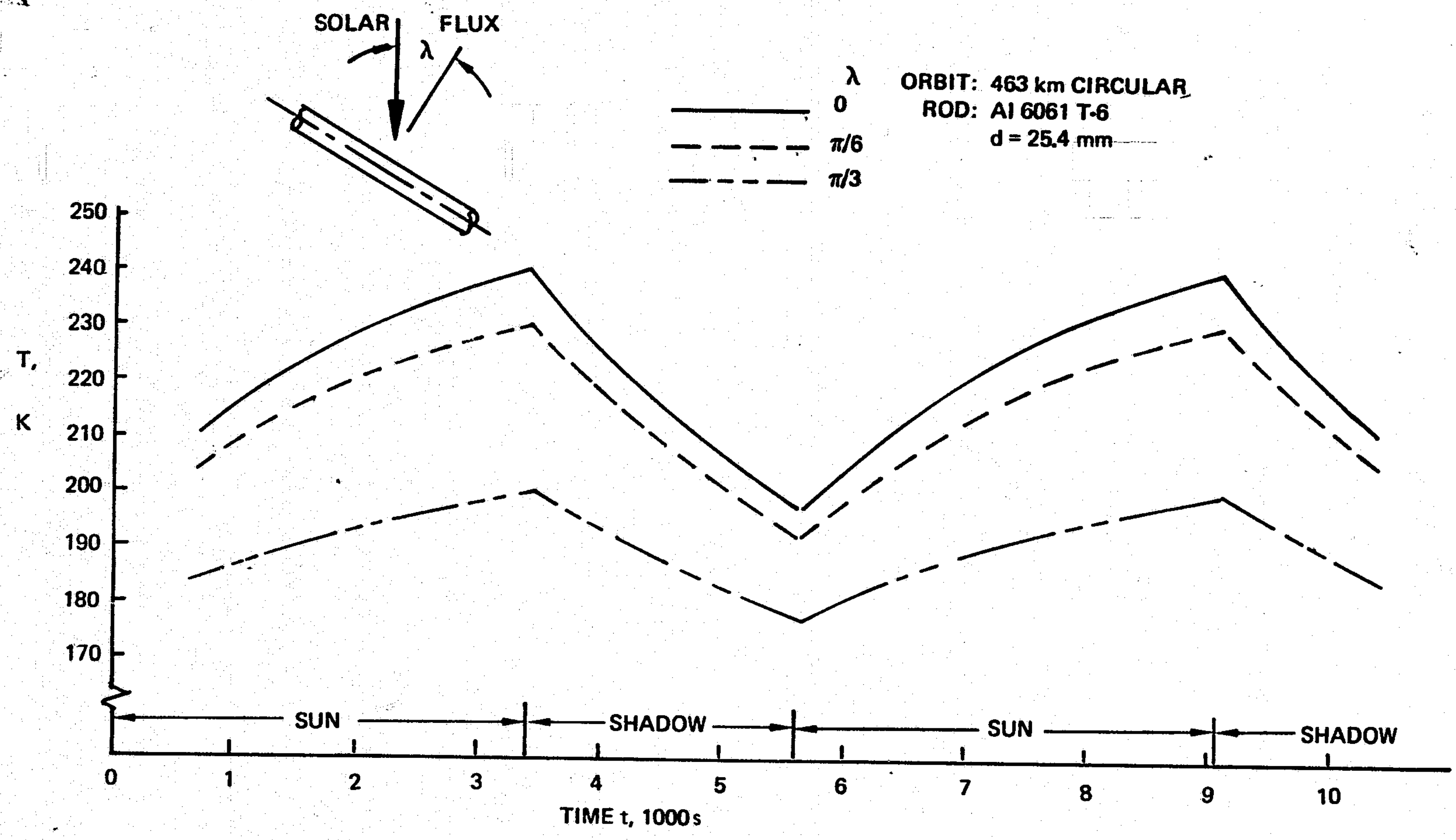

Figure 3.4-3: Transient Thermal Response—Anodized Aluminum Rod Element, Low Earth Orbit

SUMMARY OF TEMPERATURE EXTREMES - ANODIZED ALUMINUM ROD ELEMENT, LOW EARTH ORBIT

- Envelope of maximum and minimum values from Fig. 3.4-3.
- Isothermal rod values for steady-state in sunlight shown for comparison.
- Temperatures at values of λ approaching $\pi/2$ not realistically predictable by isolated element approach because of probable significant influence from adjoining or nearby members, heat sources, etc.

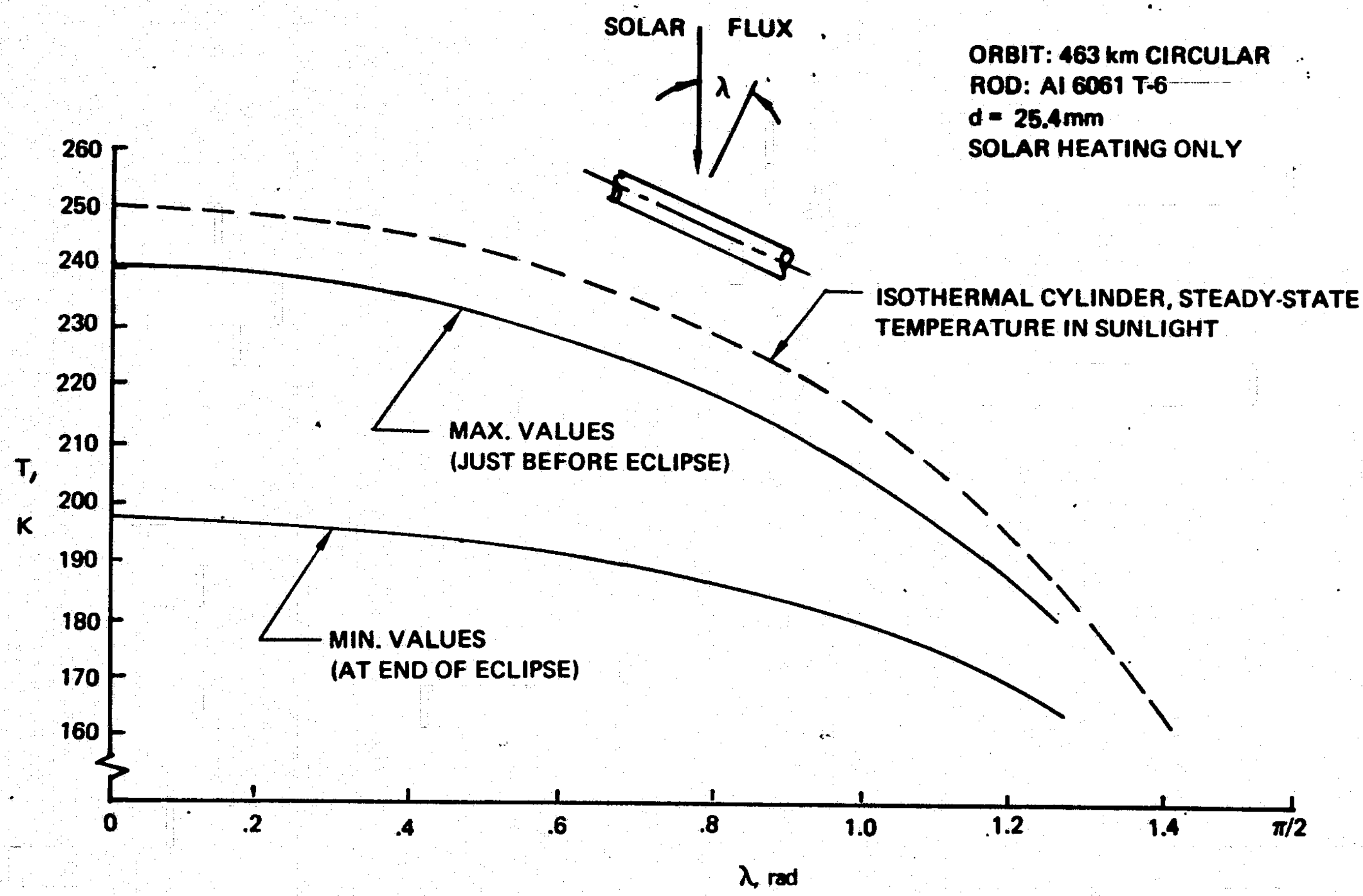

Figure 3.4-4: Summary of Temperature Extremes—Anodized Aluminum Rod Element, Low Earth Orbit

TRANSIENT THERMAL RESPONSE - 0.508 mm WALL GRAPHITE-EPOXY TUBE ELEMENT, LOW EARTH ORBIT

- Temperatures were computed for the transient two-dimensional case. See Fig. 3.1-1(A).
- Steady-state condition reached for major portion of sunlit part of orbit.
- Material properties are from Table 3.4-1.
- Earth emitted and reflected radiation ignored.
- Results valid for any d/t = 100.

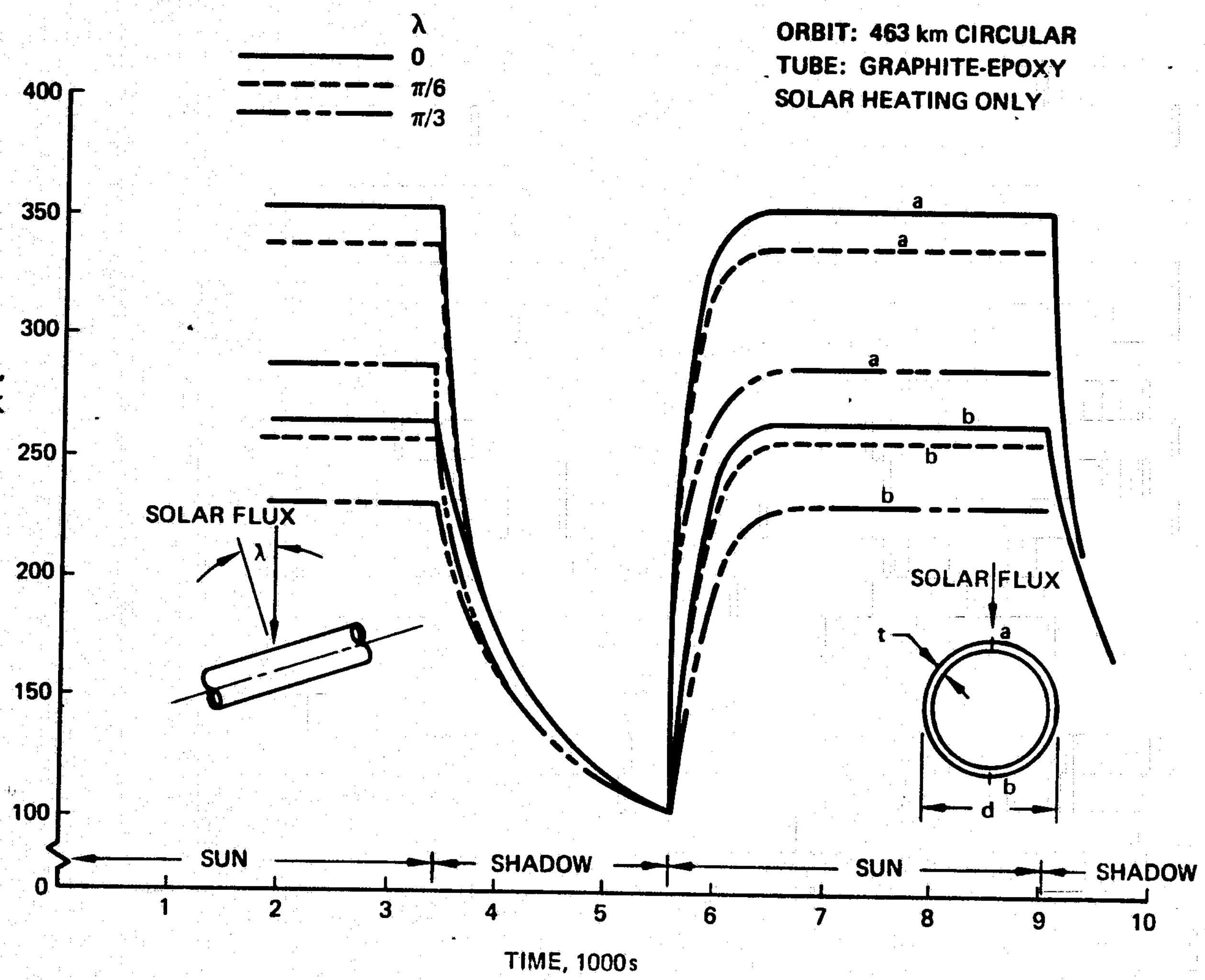

Figure 3.4-5: Transient Thermal Response—Graphite-Epoxy Composite Tube Element, Low Earth Orbit, d/t = 100

SUMMARY OF TEMPERATURE EXTREMES -
0.508 mm WALL GRAPHITE-EPOXY TUBE ELEMENT LOW EARTH ORBIT

- Envelope of maximum and minimum values from Fig. 3.4-5.
- Temperatures for adiabatic surface and steady-state isothermal tube in sunlight shown for comparison.
- Temperatures at values of λ approaching $\pi/2$ not realistically predictable by isolated element approach because of probable significant influence from adjoining or nearby members, heat sources, etc.

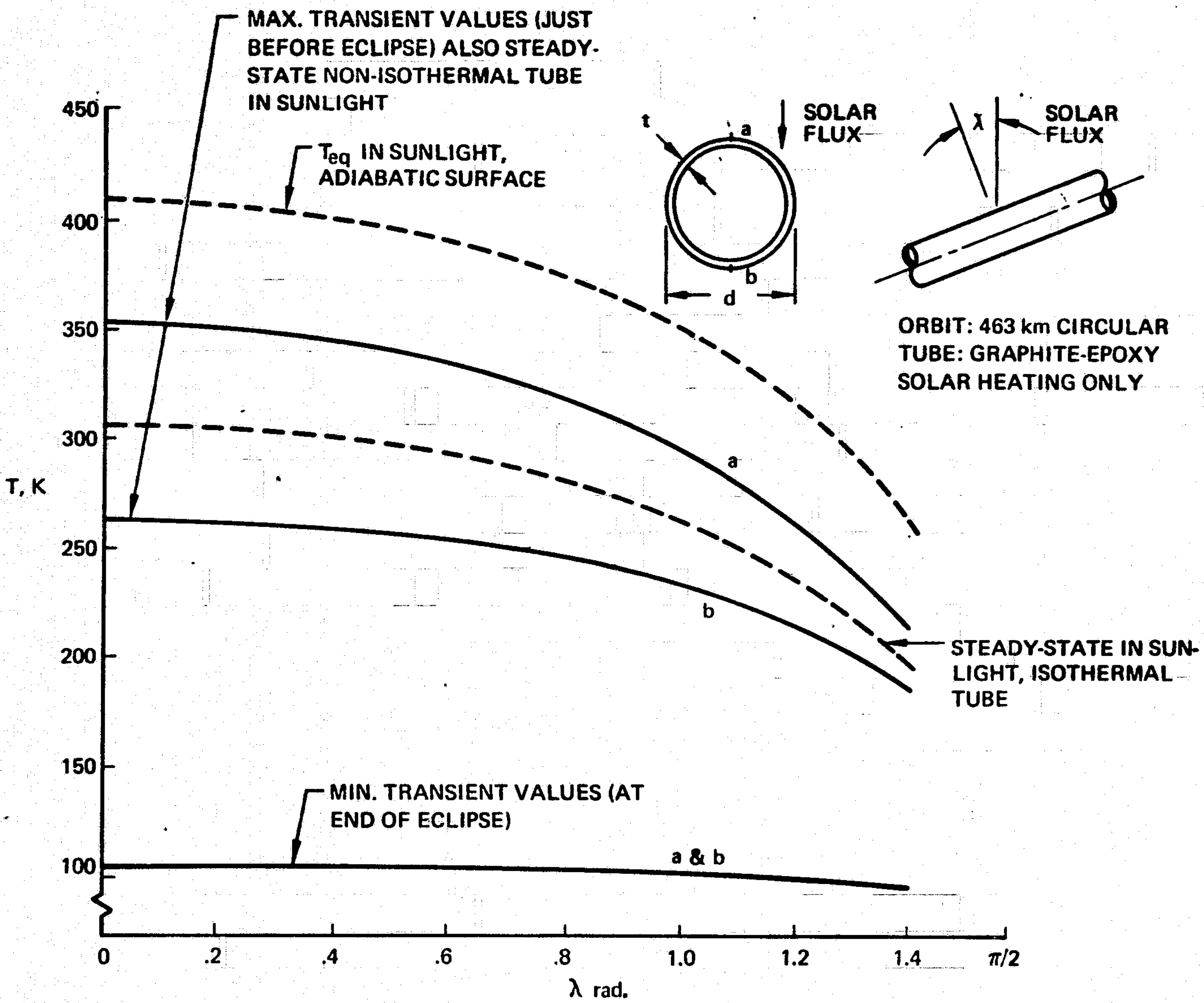

Figure 3.4-6: Summary Of Temperature Extremes—Graphite-Epoxy Tube Element. Low Earth Orbit, d/t = 100

TRANSIENT THERMAL RESPONSE - 2.03 mm WALL GRAPHITE-EPOXY TUBE ELEMENT, LOW EARTH ORBIT

- Temperatures were computed for the transient two-dimensional case. See Fig. 3.1-1(A).

- Steady-state condition closely approached at end of sunlit part of orbit.

- Material properties are from Table 3.4-1.

- Earth emitted and reflected radiation ignored.

- Results valid for any d/t = 25

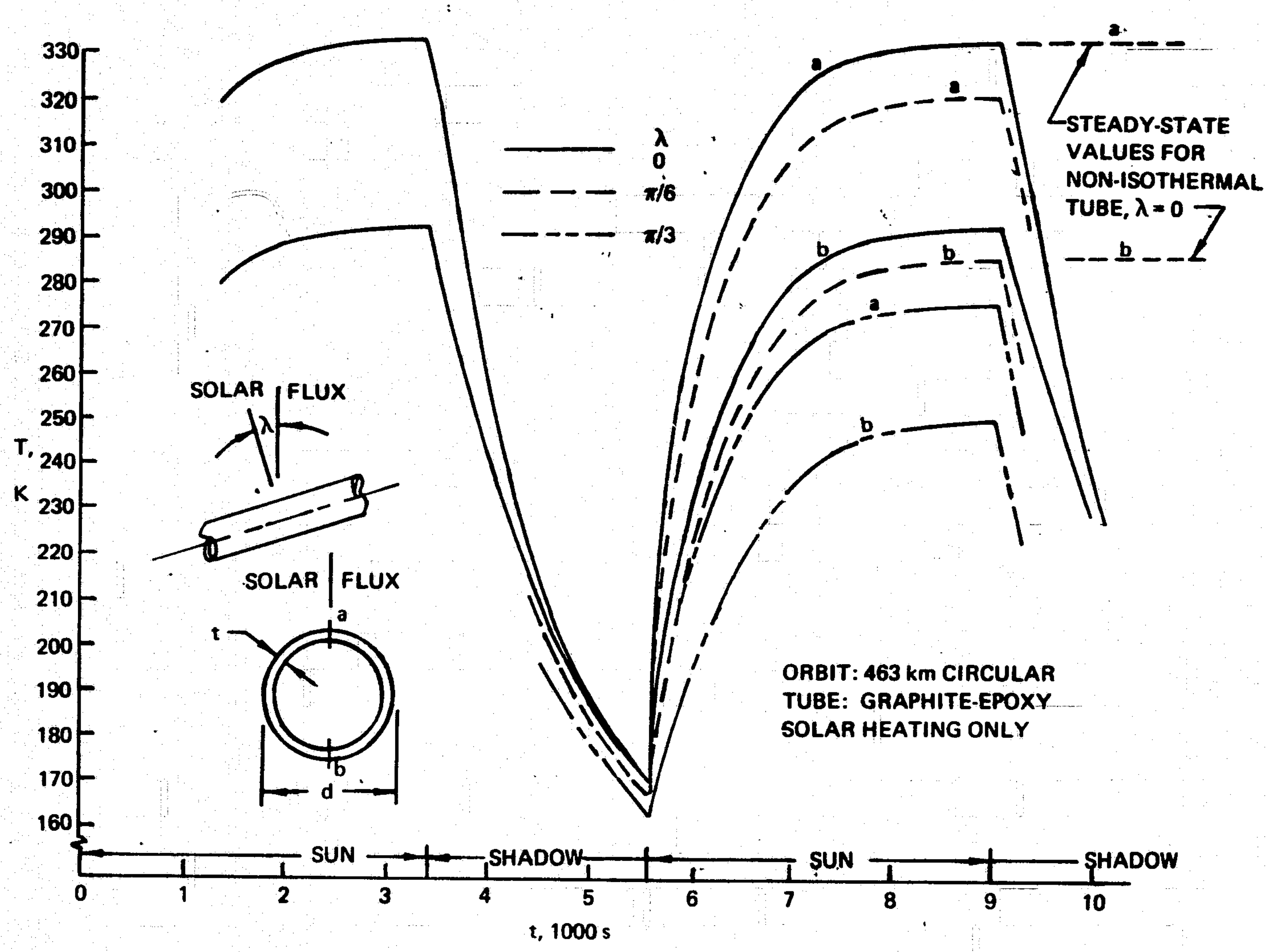

Figure 3.4-7: Transient Thermal Response—Graphite-Epoxy Tube Element, Low Earth Orbit, d/t = 25

SUMMARY OF TEMPERATURE EXTREMES - 2.03 mm WALL
GRAPHITE-EPOXY TUBE ELEMENT, LOW EARTH ORBIT

- Envelope of maximum and minimum values from Fig. 3.4-7.
- Temperatures for adiabatic surface and steady-state isothermal tube in sunlight shown for comparison.
- Temperatures at values of λ approaching $\pi/2$ not realistically predictable by isolated element approach because of probable significant influence from adjoining or nearby members, heat sources, etc.

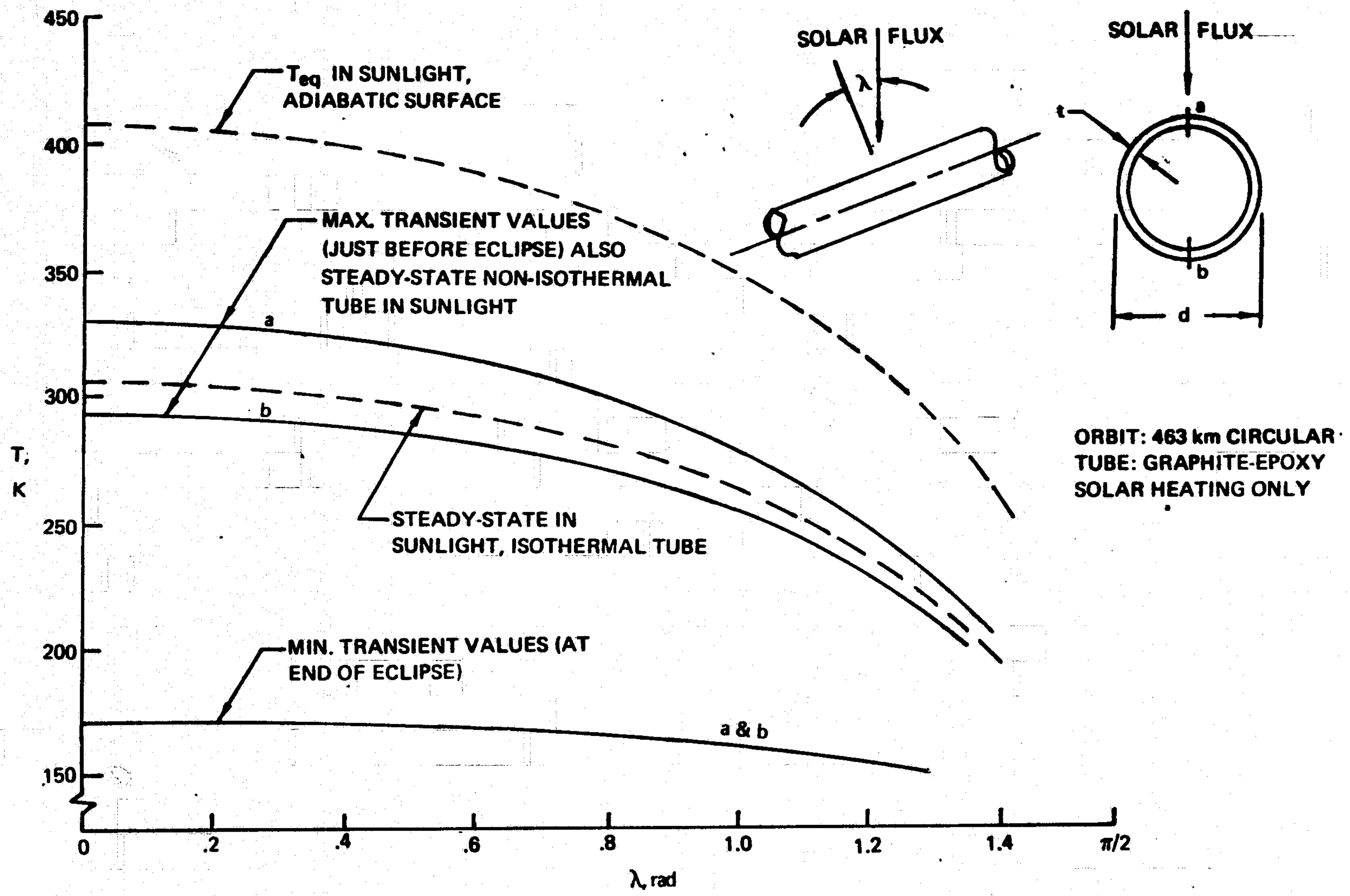

Figure 3.4-8: Summary of Temperature Extremes—Graphite-Epoxy Tube Element, Low Earth Orbit, d/t = 25

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TRANSIENT THERMAL RESPONSE - GRAPHITE-EPOXY ROD ELEMENT - LOW EARTH ORBIT

- Temperatures were computed for the transient isothermal cross section case. See Fig. 3.1-1(B).

- Results useful as approximations for rod from 1/2 to twice the indicated diameter.

- Material properties are from Table 3.4-1.

- Earth emitted and reflected radiation ignored.

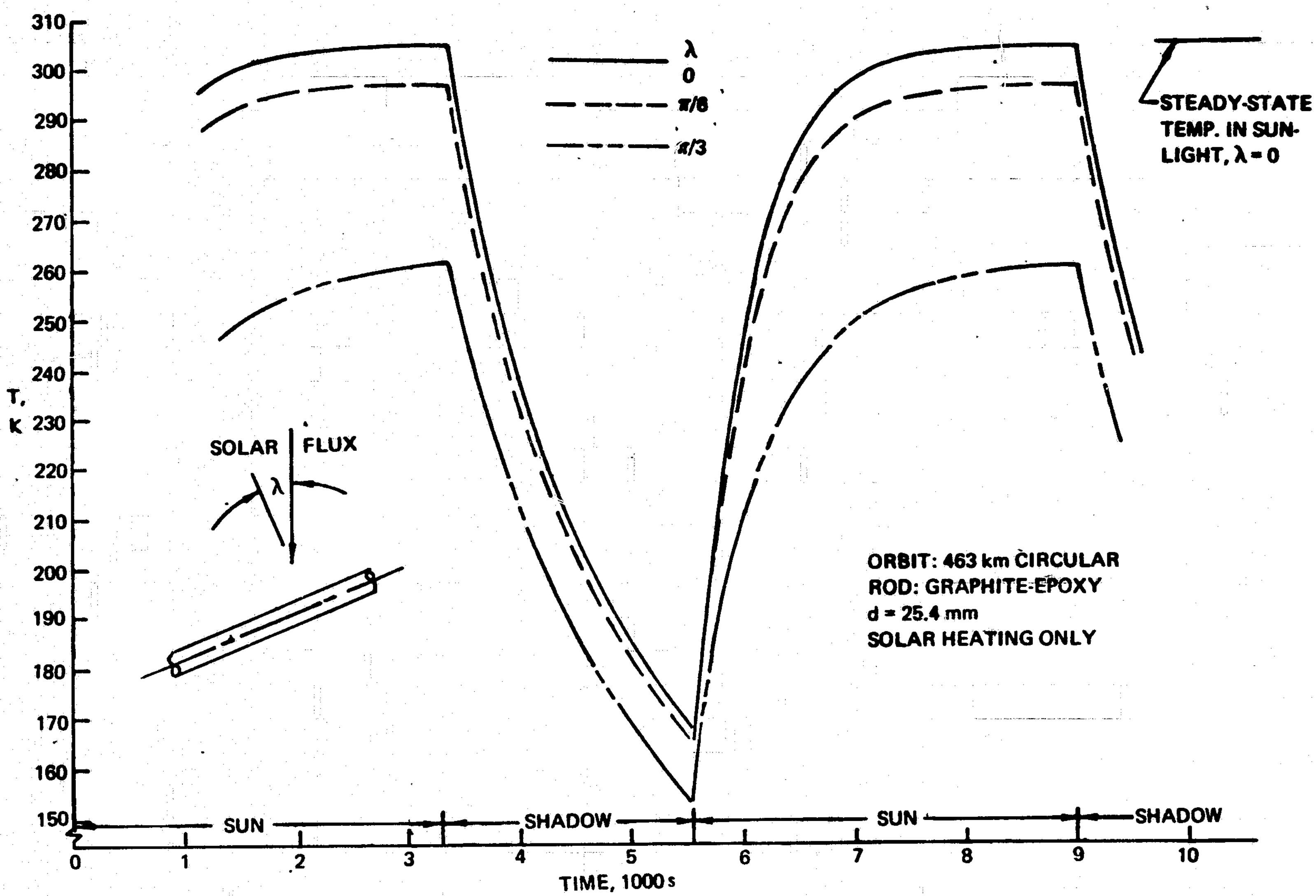

Figure 3.4-9: Transient Temperature Response—Graphite-Epoxy Rod Element, Low Earth Orbit

SUMMARY OF TEMPERATURE EXTREMES - GRAPHITE-EPOXY ROD ELEMENT, LOW EARTH ORBIT

- Envelope of maximum and minimum values from Fig. 3.4-9.

- Maximum values are equivalent to isothermal rod steady-state in sunlight temperatures.

- Temperatures at values of λ approaching $\pi/2$ not realistically predictable by isolated element approach because of probable significant influence from adjoining or nearby members, heat sources, etc.

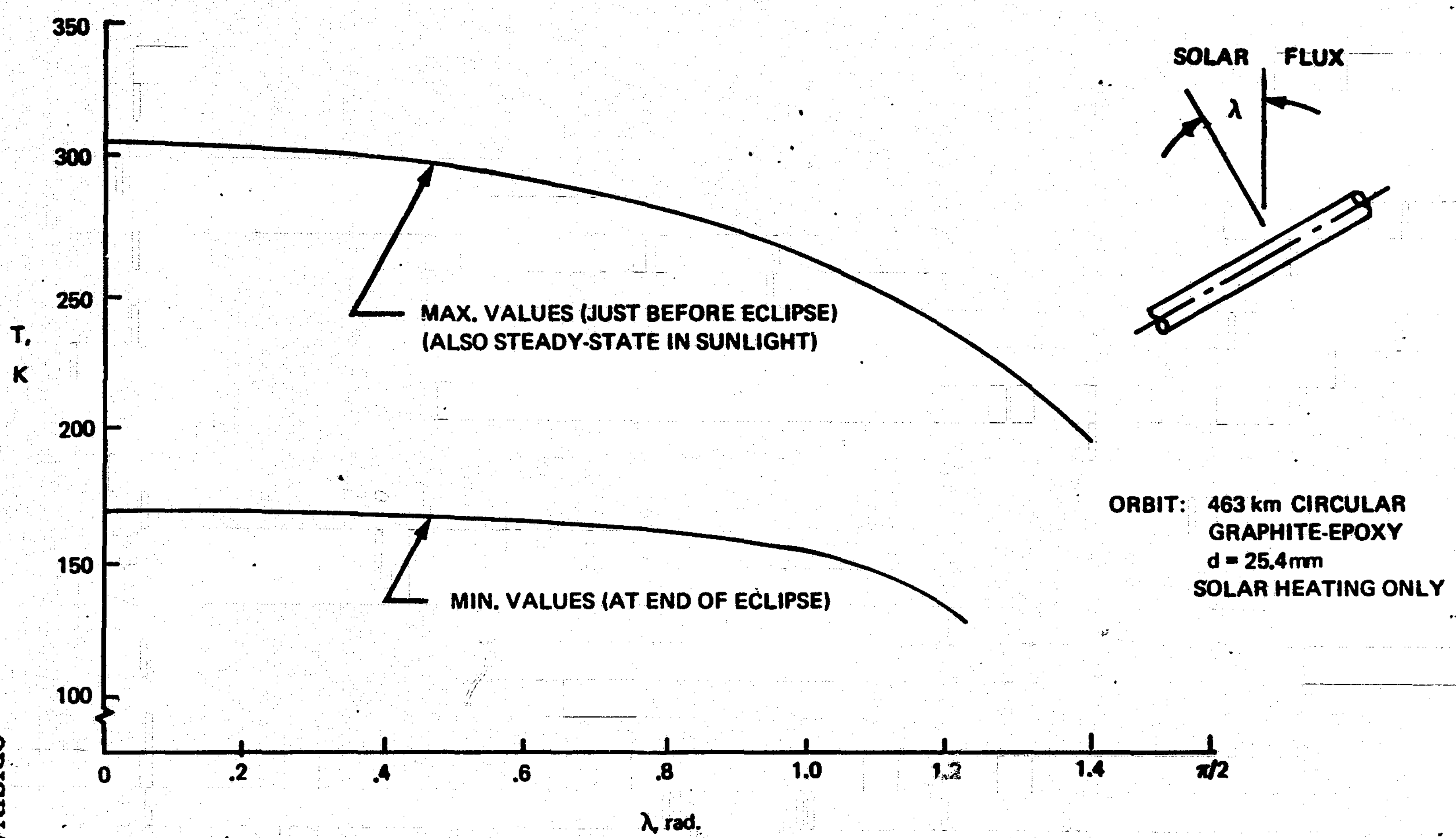

Figure 3.4-10: Summary of Temperature Extremes—Graphite-Epoxy Rod Element, Low Earth Orbit

TRANSIENT THERMAL RESPONSE -
INSULATED TUBE ELEMENT, LOW EARTH ORBIT

- Temperatures were computed for the transient two-dimensional case. See Fig. 3.1-1(A)
- Tube temperatures may require many orbits to reach values shown if initial temperature differs greatly.
- Tube and insulation temperatures appear constant during major portions of orbit phases but actually experience very slow continuous changes, due to very slow heat gain (in sun) and loss (in shadow) of the system.
- Insulation surface temperatures in sunlight may be closely approximated by the adiabatic surface (Fig. 3.1-1(D), Zero-Dimensional) approach.
- Insulation properties from Reference 3. Tube properties were those for Al 6061 T-6, d = 50.8 mm, t = 2.03 mm. Results are valid for any tube with similar dimensions and mass.
- Earth emitted and reflected radiation ignored.

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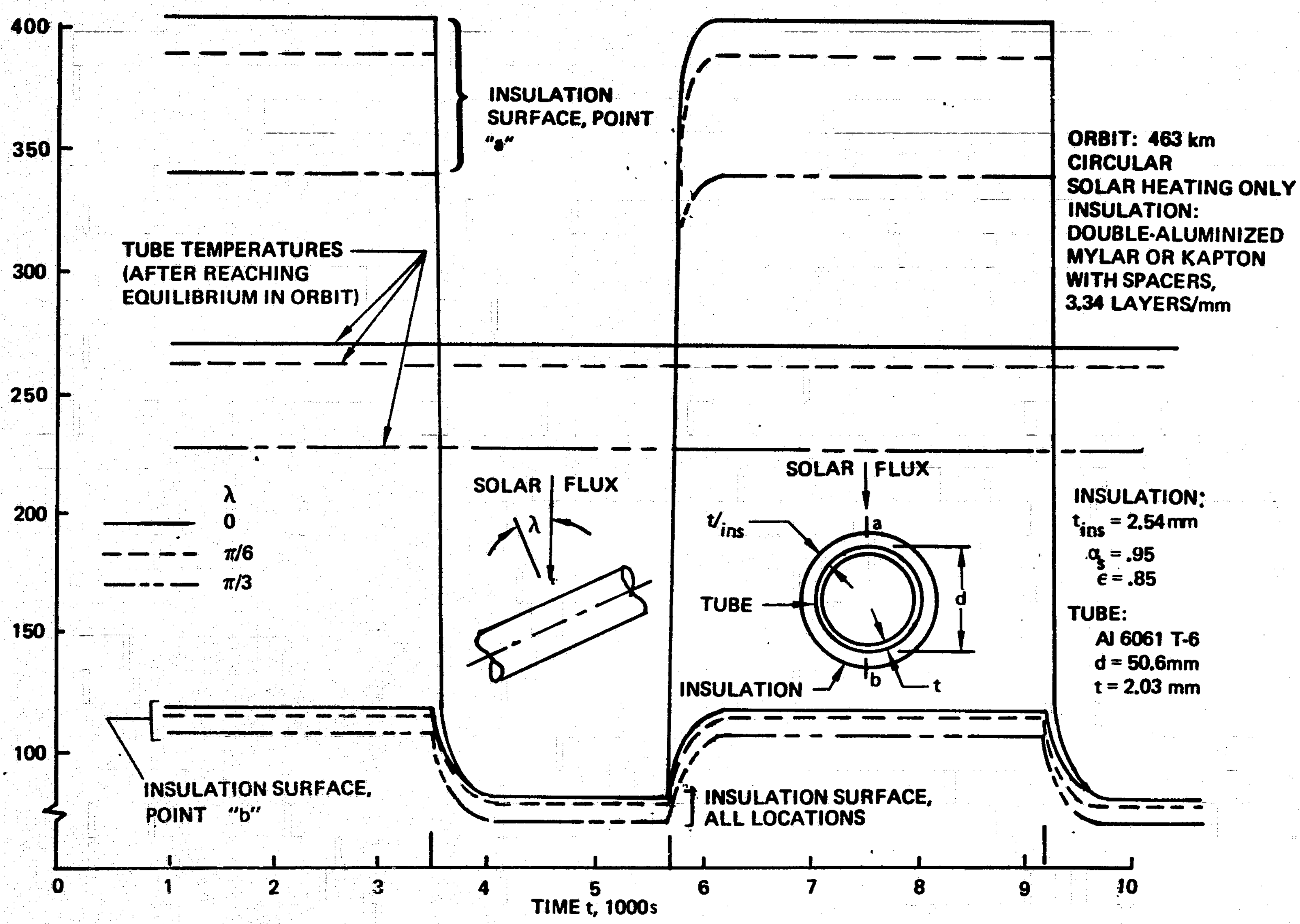


Figure 3.4-11: Transient Thermal Response— Insulated Tube Element, Low Earth Orbit

STABILIZED TEMPERATURES -
INSULATED TUBE ELEMENT, LOW EARTH ORBIT

- Curve is cross plot of tube temperatures from Fig. 3.4-11.
- Results may be generalized to apply to:
  a) Any insulation having the indicated surface properties and insulating capabilities such as to effectively isolate tube from orbital sun-shadow heating variations,
  b) Any structural tube whose mass per unit length is large relative to that of the insulation.
- Results valid only after initial transient (due to initial tube temperature differing from stabilized value) has become insignificant.
- Temperatures at values of λ approaching $\pi/2$ not realistically predictable by isolated element approach because of probable significant influence from adjoining or nearby members, heat sources, etc.

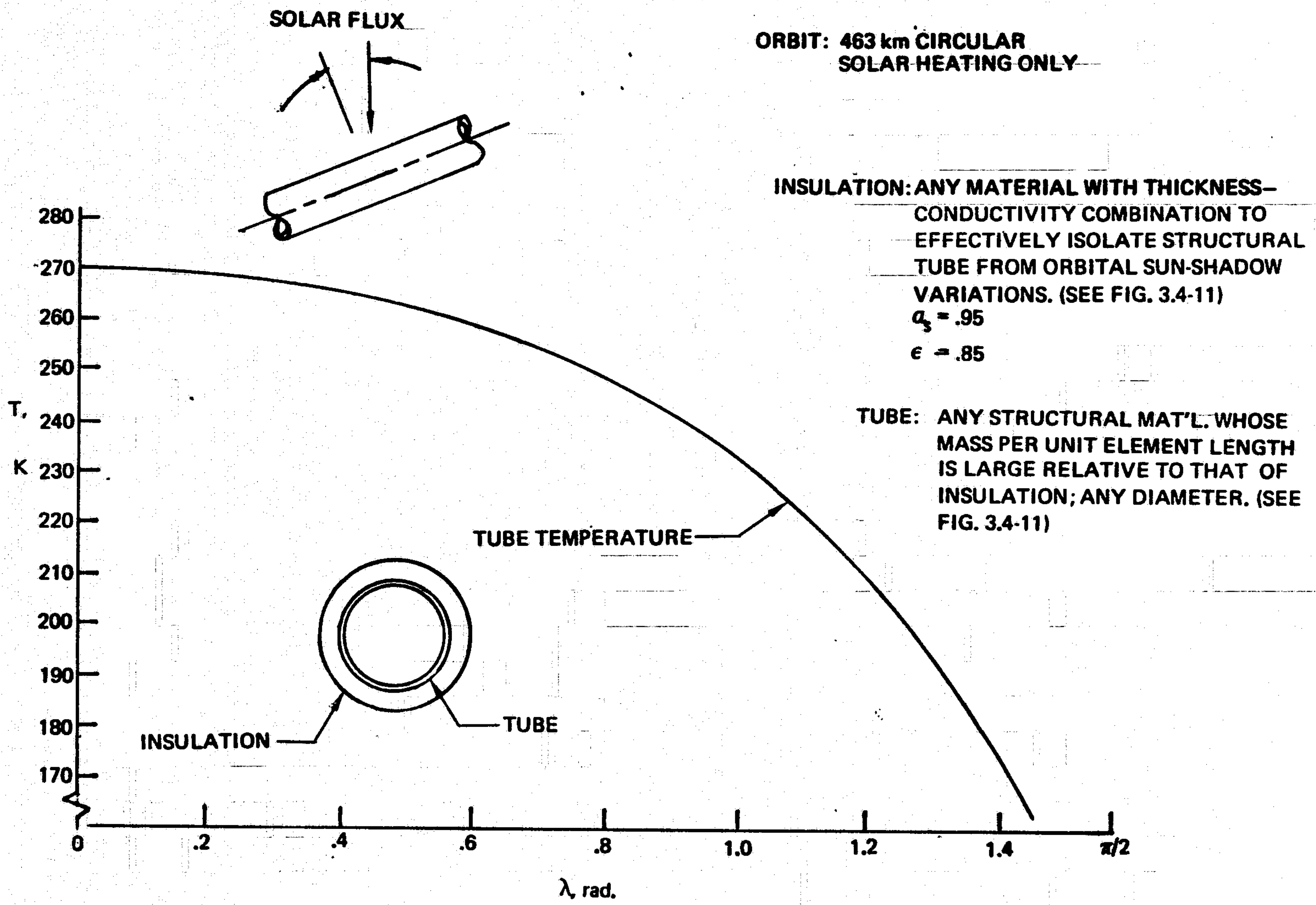
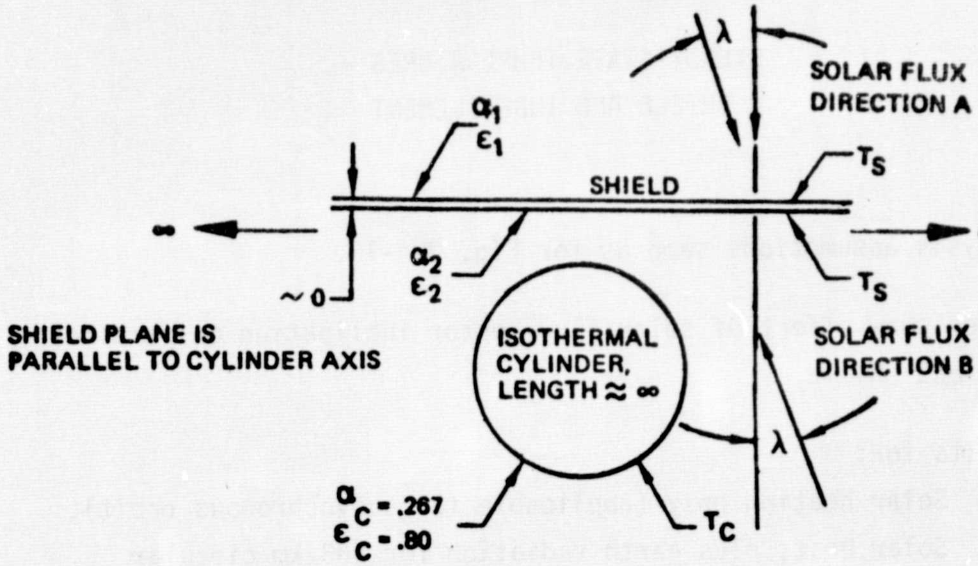


Figure 3.4-12: Stabilized Temperatures–Insulated Tube Element, Low Earth Orbit

SAMPLE EQUILIBRIUM TEMPERATURES ON SHIELDED ISOTHERMAL CYLINDER

- The data show trends of shield effects for particular sample cases.
- Analysis assumptions:
  - (1) Cylinder - two-dimensional, isothermal, steady state (Fig. 3.1-1(B)).
  - (2) Shield - one-dimensional, zero effective thickness, steady-state (Fig. 3.1-1(C).
  - (3) Shield-cylinder interchange - infinite plane, infinite cylinder radiation view factor.
- Earth emitted and reflected radiation ignored.
- Case No. 2 properties were chosen to represent a typical solar cell array and support structure (Ref. 2).
- Conclusions:
  - (1) Depending upon shield properties, shielding will not always decrease shielded member's temperature (Case 1. vs. Case 2.)
  - (2) Shield with particular properties can significantly reduce shielded member's temperature (Case 3.)
  - (3) Shield on side of member away from sun (Case 4) can significantly increase tube temperature over the no-shield (Case 1) value.



ACTUAL DESIGN APPROXIMATED	α_1	ε_1	α_2	ε_2	T_S (K)	T_C (K)
1. HEAVILY ANODIZED Al CYLINDER WITH NO SHIELD, OR WITH SOLAR FLUX PARALLEL TO SHIELD	–	–	–	–	3	226
2. CYLINDER OF 1. WITH A TYPICAL-PROPERTY SHIELD. SOLAR FLUX DIRECTION A.	.8	.9	–	.9	323	268
3. CYLINDER OF 1. WITH MAX. EFFECTIVE SHIELD. SOLAR FLUX DIRECTION A.	.2	.8	–	.02	419	157
4. PROPERTIES OF 2. WITH SOLAR FLUX DIRECTION B.	.8	.9	.3	.9	253	321

Figure 3.4-13: Sample Equilibrium Temperatures on Shielded Isothermal Cylinder (Solar Flux Only)

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STEADY-STATE TEMPERATURES - SHIELD AND TUBE ELEMENT

- Analysis assumptions same as for Fig. 3.4-13.
- Curves show effect of solar flux vector inclination on parallel tube and shield.
- Results for:
  (1) Solar heating only (applicable to geosynchronous orbit),
  (2) Solar heat, plus earth radiation for 463 km circular orbit with shield-tube unit always normal to earth radius (local vertical).

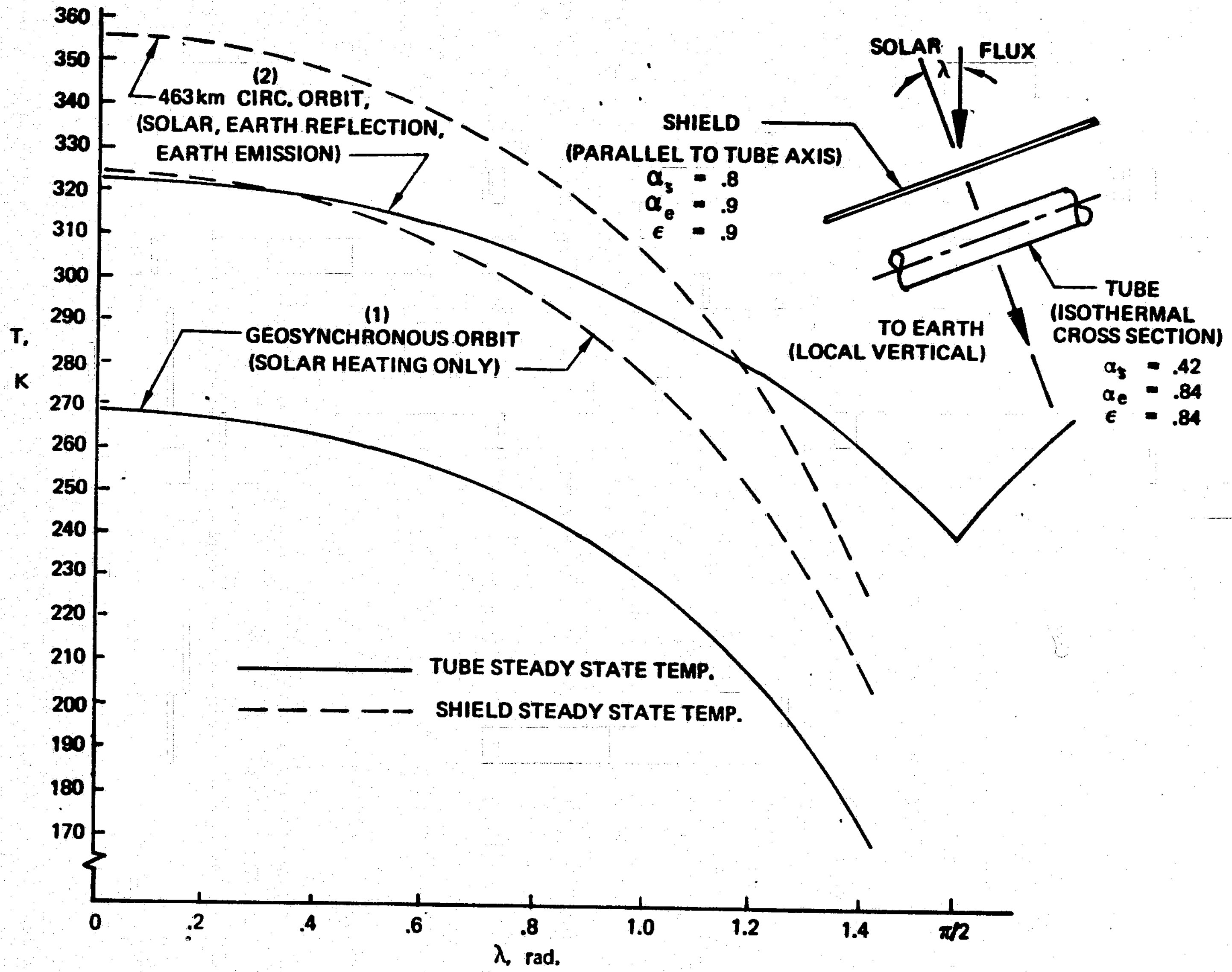


Figure 3.4-14: Steady-State Temperatures—Shield and Tube Element

STEADY-STATE TEMPERATURES - SHIELDED ISOTHERMAL CYLINDER IN SUNLIGHT

- Temperatures are steady-state with earth radiation ignored.
- Curves show dependence of cylinder temperature upon shield surface properties. Formula gives dependence upon solar flux direction.
- Cylinder emittance and absorptance are assumed equal, for which condition temperatures are independent of cylinder surface properties. This assumption is generally valid for a wide variety of materials and wide range of temperatures.
- Isothermal cylinder temperatures closely approximate mean temperatures of non-isothermal cylinders. If circumferential temperature distributions are needed, e.g., to evaluate thermally induced bending, more rigorous analysis is required.

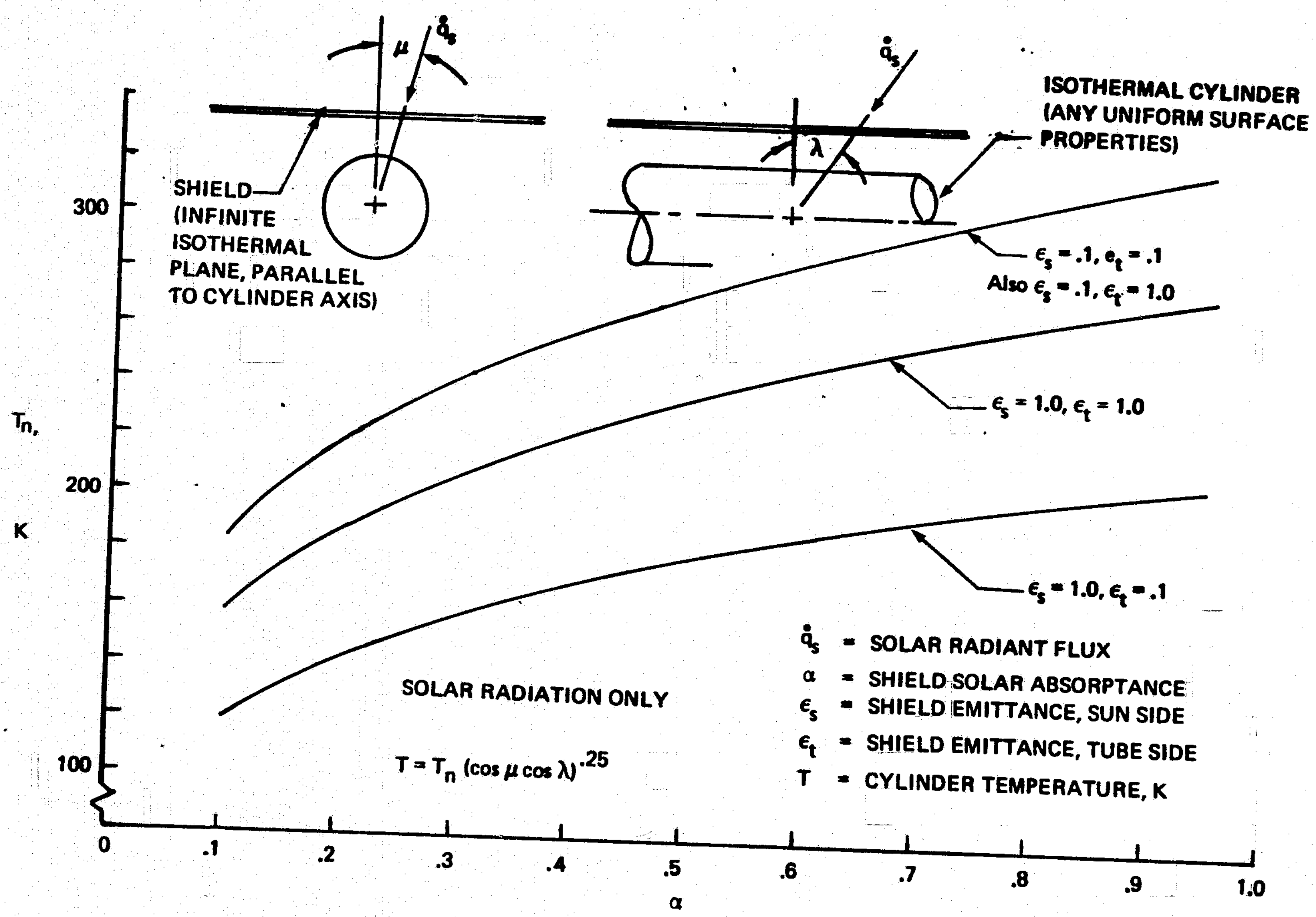


Figure 3.4-15: Steady-State Temperatures—Shielded Isothermal Cylinder In Sunlight

(3) Shield is isothermal in planform

(4) Shield has zero thermal thickness (same temperature both surfaces)

(5) Shield width is very large relative to shield-cylinder spacing

(6) Shield plane is parallel to cylinder axis

(7) Ambient environment consists of solar flux only

(8) Shield temperature is unaffected by shading or radiation from cylinder

(9) Cylinder absorptivity is invariant with wavelength

(10) Shield surface reflects and emits diffusely.

(Nomenclature for the following relations is from Fig. 3.4-13.)

Shield:

$$\dot{q}_{in} = \dot{q}_{out}$$

$$\dot{q}_s \cos \lambda \, \alpha_\chi = \sigma \varepsilon_1 T_s^4 + \sigma \varepsilon_2 T_s^4$$

($\chi = 1$ for Solar Flux Direction A

$\chi = 2$ for Solar Flux Direction B)

$$T_s = \left[\frac{\dot{q}_s \cos \lambda \, \alpha_\chi}{\sigma (\varepsilon_1 + \varepsilon_2)} \right]^{.25}$$

Cylinder:

$$\dot{q}_{in} = \dot{q}_{out}$$

$\dot{q}$ (absorbed from shield emission) + $\dot{q}$ (absorbed from shield reflection) + $\dot{q}$ (absorbed from solar flux) = $\dot{q}$ (emitted by cylinder)

$$\sigma F_s \varepsilon_2 \alpha_c T_s^4 + \dot{q}_s \cos \lambda \, F_s (1-\alpha_2)\alpha_c + \dot{q}_s F_c \alpha_c$$

$$= \sigma \varepsilon_c T_c^4$$

($\dot{q}_s$ at cylinder is zero for solar flux direction A)

F_s = geometric view factor, infinite plane to infinite cylinder = 0.50

F_c = geometric view factor, collimated source to infinite cylinder
$= 1/\pi$

$$T_c = \left[\frac{\sigma\, F_s\, \varepsilon_2\, \alpha_c\, T_s^{\,4} + \dot{q}_s \cos\lambda\, F_s(1-\alpha_2) + \dot{q}_s\, F_c\, \alpha_c}{\sigma\, \varepsilon_c}\right]^{.25}$$

Some of the charts of this section show only steady state temperatures, e.g., the shielded member charts, and some show steady-state temperatures in addition to transient temperatures. The steady-state values are temperatures that would exist after indefinite exposure to the environment. Some temperatures were computed for assumed isothermal tubes, i.e., infinite conductance between the irradiated and shaded sides of the tube, some were computed for the true conductance, and some were for thermally isolated surface points, i.e., zero material conductance. The latter assumption yields the adiabatic surface or radiation equilibrium values described earlier. In each case the assumptions employed in generating the data are indicated on the charts.

With a few exceptions all preceding data in Section 3.4 (Figure 3.4-1 through 3.4-15) ignore effects of earth emitted and reflected radiation. From examination of the potential magnitudes of earth radiation relative to solar flux levels (Ref. 2), it is clear that earth reflection and emission can have a significant effect on space structure response temperatures, particularly when absorbed solar flux levels are low. The inclusion of earth radiation effects in a thermal response analysis introduces additional independent variables to the heat balance solution. The new variables are needed to describe the structural member's orientation relative to the earth flux, the member's absorptance values at the effective earth emission and reflection spectra, and the position in orbit (affecting the reflected flux incident intensity).

The many ways that the variables needed to evaluate the solar and earth radiation effects can combine, make impractical the development of general yet comprehensive structral temperature charts for low earth orbits with earth radiation included. It is possible, however, to evaluate the possible range of additional heat input to structural members from earth emission and reflection at any particular level of solar heat input, as determined by the solar flux incidence angle, λ. Thus it is possible to compute radiation equilibrium temperature ranges as a function of λ and the members' surface properties. Because the various incident heat flux components may have different vector directions such temperature calculations may be performed only under the isothermal cross section assumption, i.e., the computed temperatures represent the mean values around a cross section. The usual assumptions of slender, constant-cross-section members with negligible end or joint effects apply also.

Figure 3.4-16 is a chart of upper and lower bounds of isothermal tube equilibrium temperatures, generated by the approach described in the preceding paragraph. The two structural members treated are the same graphite-epoxy and anodized aluminum tubes described in Table 3.4-1. A temperature at the upper limit of a band on the figure will exist when the member is oriented to receive the maximum earth emission and reflection possible at the particular value of λ. The band lower limits are the solar-heating-only curves and apply for members oriented so that received earth radiation is zero.

The data of Figure 3.4-16 do not apply when the orbiting structure is in the earth's shadow. For that part of the orbit both solar flux and earth reflected flux are zero and only the earth emitted flux remains. Since earth emitted flux is approximately constant through any constant altitude orbit, the band of possible equilibrium temperatures in the shadow is defined by a constant upper and a constant lower limit. Furthermore, since for nearly all surfaces absorptance relative to earth emission closely approximates emittance, it may be universally assumed that $\alpha/\varepsilon = 1.0$ in earth shadow, and the temperature band in the shadow is the same for all materials.

The simplistic temperature-estimating approach described yields a temperature of absolute zero as the band lower limit in earth shadow. As pointed out previously, such an estimate is, in all likelyhood, very unrealistic, since under conditions of zero ambient heat input, transient and heat interchange effects will become quite significant. Therefore, only the upper limit of the temperature band in earth shadow has much usefullness as a source of structural estimates. Even this boundary is not an absolute limit, nor are any of the other boundaries of Figure 3.4-16 absolute limits, since transient effects can yield temperatures outside the bands.

The band of possible (under the assumptions employed) temperatures for isothermal tubes in the earth shadow portion of a 463 km circular orbit is bounded by a constant zero K and a constant 182K.

3.5 Geosynchronous Orbit Transient Temperatures

Transient temperatures for structural members in geosynchronous orbit were computed in the same way as those in low earth orbit. Heat flow along the length of the constant-cross-section members and interchange between members was ignored. The orbit chosen for environment definition was an eclipsing circular geosynchronous orbit, taken at a time during its history when maximum duration eclipsing of the spacecraft occurs. As for most of the low earth orbit analyses, earth emission and reflection were neglected, an omission of insignificant effect on geosynchronous temperatures.

Two structural members are covered by the geosynchronous orbit transient temperature curves, the aluminum tube, Figures 3.5-1 and 3.5-2, and the thin-wall graphite-epoxy tube, Figure 3.5-3 and 3.5-4. In addition to these figures, all steady-state temperature data from preceding Section 3.4 figures, except where earth radiation is considered, are applicable to the sunlit portions of geosynchronous and intermediate orbits as well as to low earth orbits.

3.6 Extrapolation to Arbitrary Environment Levels and Surface Properties

The wide range of possible combinations of solar heating, earth reflection heating, earth emission heating, and spacecraft component heating lead to the probability of many situations not covered directly by the charts of this document. If the structural temperatures can be satisfactorily approximated by radiation equilibrium temperatures, then the curves of Figure 3.3-1 may be used directly at the appropriate values of $\dot{q}_n$ and α_s/ε.

The possibilities of extrapolating temperatures from the transient temperature response curves, e.g., Figure 3.4-1 are quite limited, but some scaling of temperatures, using trends indicated on Figure 3.3-1, is possible. As an example, suppose the peak temperatures at the end of the sunlit period in the nominal low earth orbit for an aluminum tube with a 50.8 mm diameter and a 2.03 mm wall but with $\alpha_s/\varepsilon = 1.0$ are desired. From Figure 3.4-1, it is seen that for $\alpha_s/\varepsilon = 0.50$, the temperatures for $\lambda = 0$, .524 rad., and 1.047 rad. are,

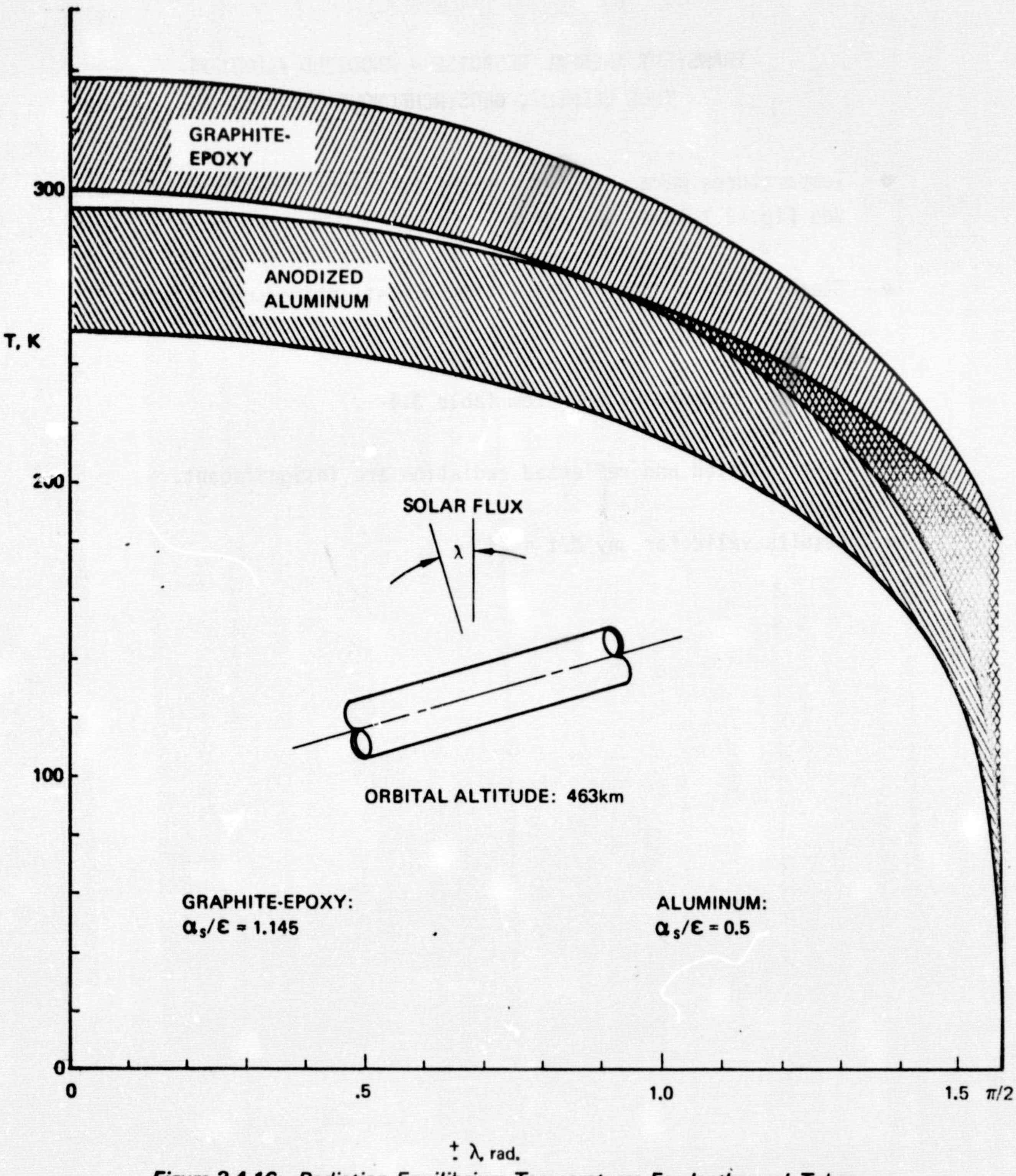


Figure 3.4-16: Radiation Equilibrium Temperatures For Isothermal Tubes In Low Earth Orbit—In Sunlight With Earth Emitted And Reflected Radiation Included

TRANSIENT THERMAL RESPONSE - ANODIZED ALUMINUM TUBE ELEMENT, GEOSYNCHRONOUS ORBIT

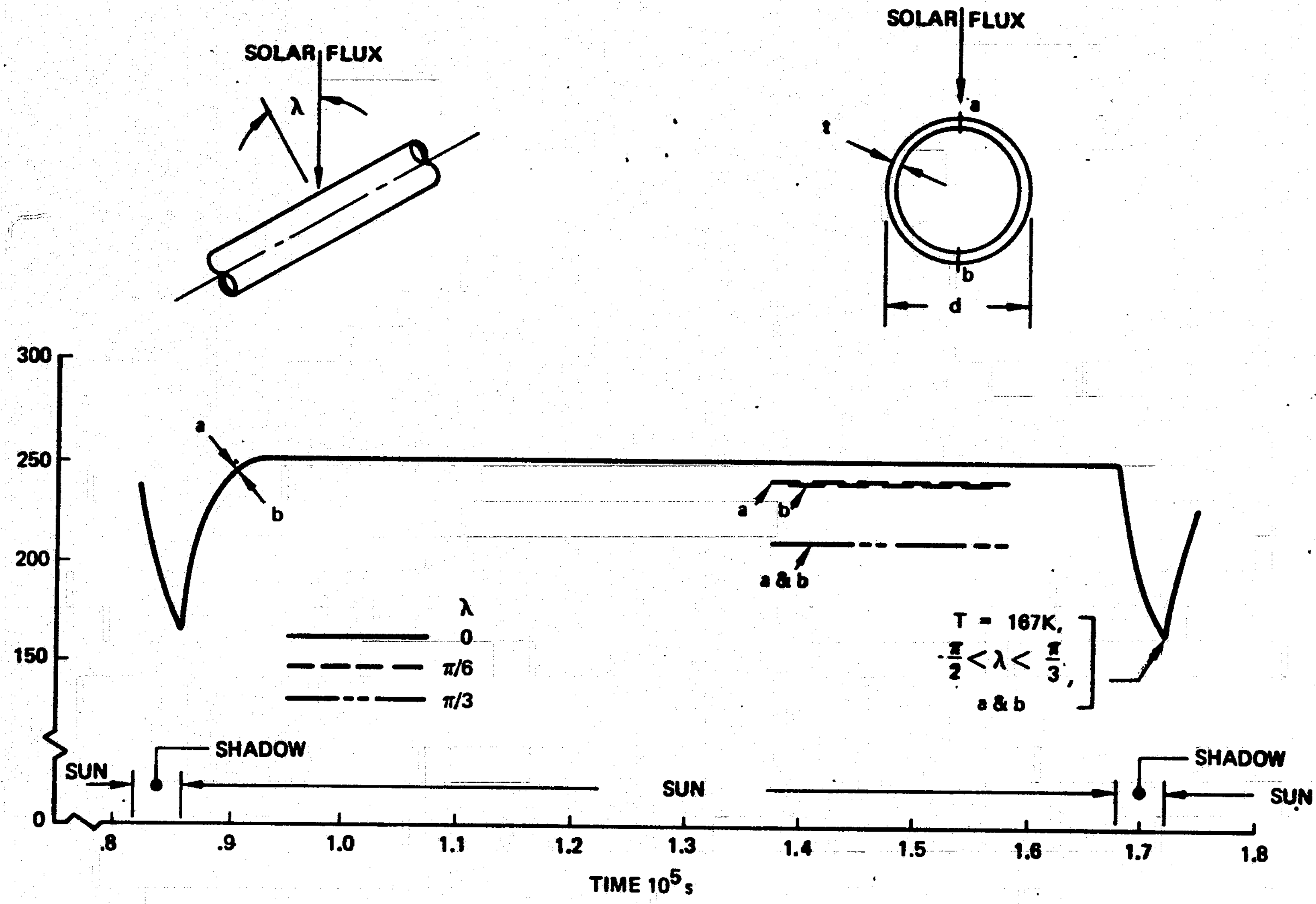
- Temperatures were computed for the transient two-dimensional case. See Fig. 3.1-1(A).

- Steady-state condition exists for almost entire sunlit part of orbit.

- Material properties are from Table 3.4-1.

- Earth emitted and reflected radiation are insignificant.

- Results valid for any d/t = 25



ORBIT: ECLIPSING GEOSYNCHRONOUS
TUBE: Al 6061 T-6

Figure 3.5-1: Transient Thermal Response—Anodized Aluminum Tube Element, Geosynchronous Orbit, d/t = 25

SUMMARY OF TEMPERATURE EXTREMES - ANODIZED ALUMINUM TUBE, GEOSYNCHRONOUS ORBIT

- Envelope of maximum and minimum values from Fig. 3.5-1.

- Maximum transient values would be closely approximated by isothermal tube steady-state in sunlight temperatures (not shown).

- Temperatures at values of λ approaching $\pi/2$ not realistically predictable by isolated element approach because of probable significant influence from adjoining or nearby members, heat sources, etc.

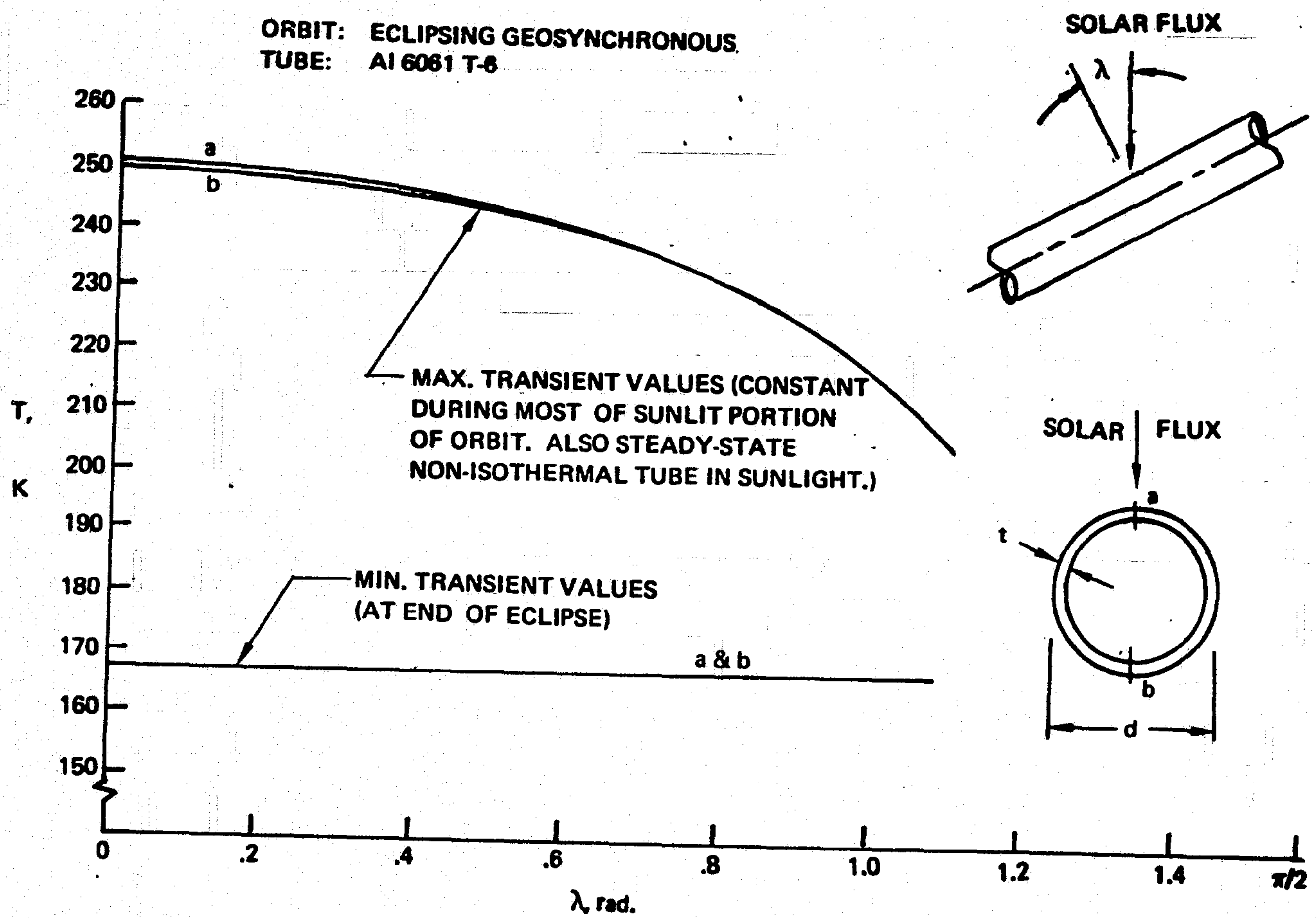


Figure 3. 5-2: Summary of Temperature Extremes—Anodized Aluminum Tube Element, Geosynchronous Orbit, d/t = 25

TRANSIENT THERMAL RESPONSE -
GRAPHITE-EPOXY TUBE ELEMENT, GEOSYNCHRONOUS ORBIT

- Temperatures were computed for the transient two-dimensional case. See Fig. 3.1-1(A).
- Steady-state condition exists for almost entire sunlit part of orbit.
- Material properties are from Table 3.4-1.
- Earth emitted and reflected radiation are insignificant.
- Results valid for any d/t = 100.

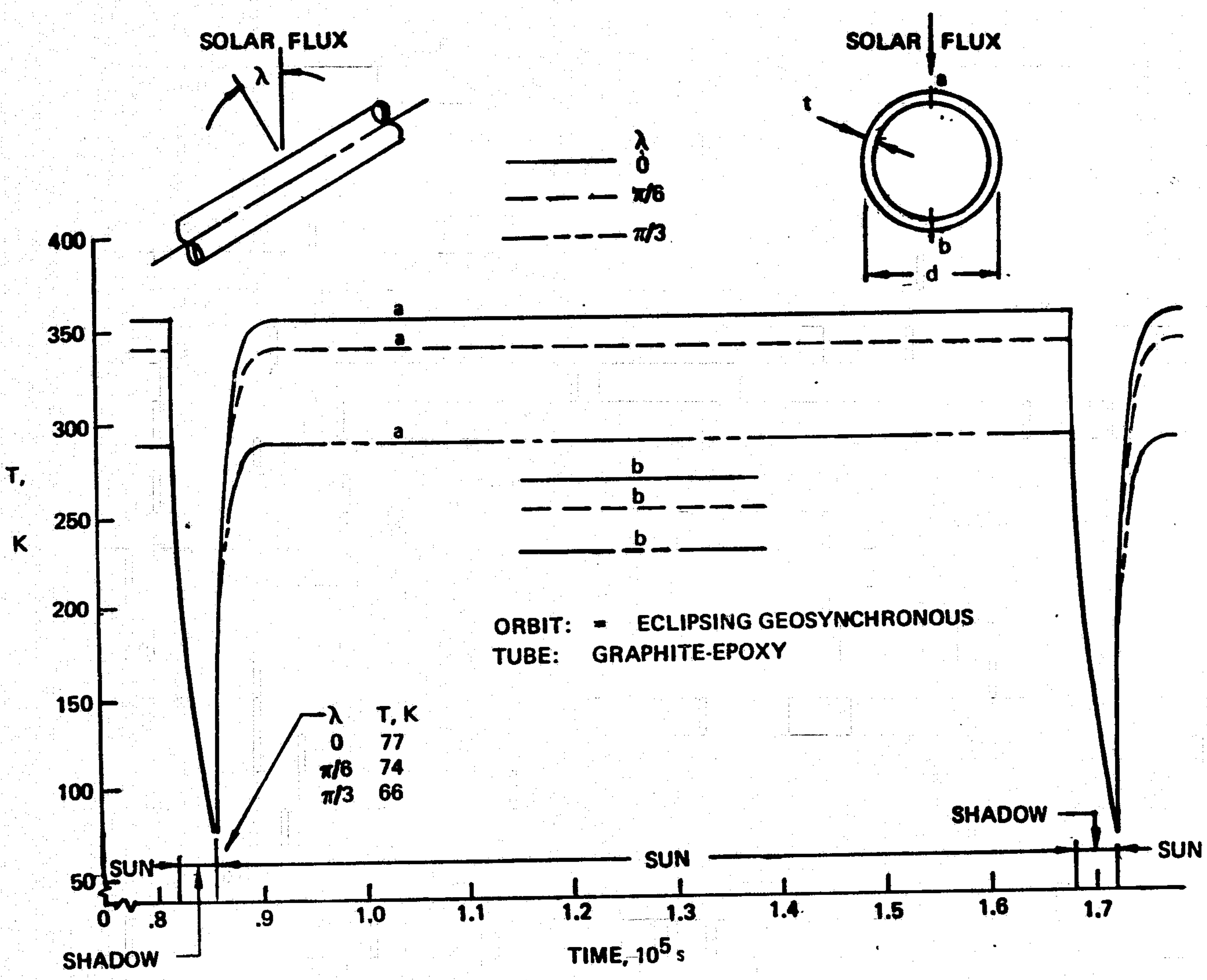


Figure 3. 5-3: Transient Thermal Response—Graphite-Epoxy Tube Element, Geosynchronous Orbit, d/t = 100

SUMMARY OF TEMPERATURE EXTREMES - GRAPHITE-EPOXY TUBE ELEMENT, GEOSYNCHRONOUS ORBIT

- Envelope of maximum and minimum values from Fig. 3.5-3.
- Temperatures at values of λ approaching $\pi/2$ not realistically predictable by isolated element approach because of probable significant influence from adjoining or nearby members, heat sources, etc.

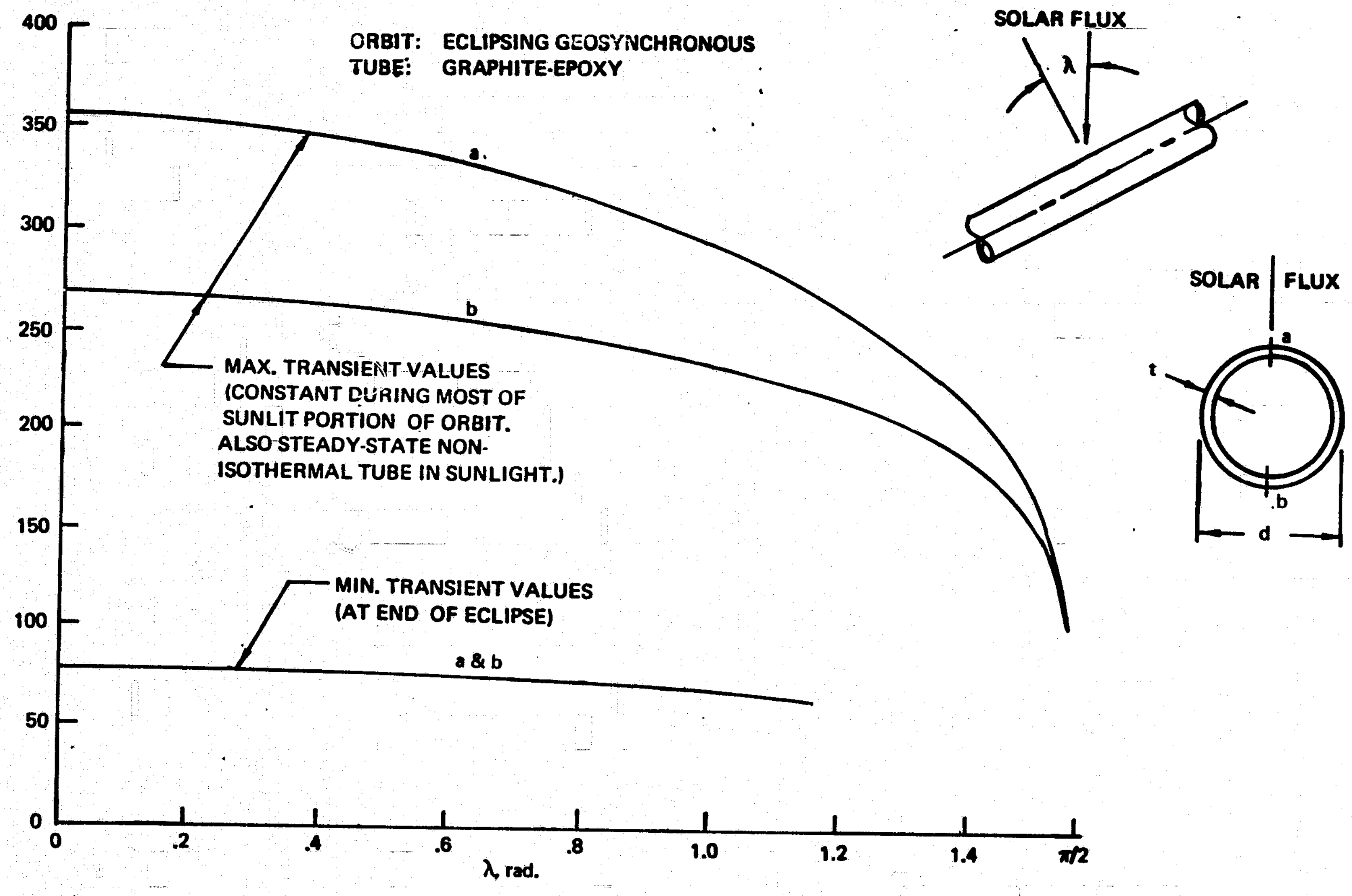


Figure 3.5-4: Summary of Temperature Extremes—Graphite-Epoxy Tube Element, Geosynchronous Orbit, d/t = 100

respectively, 245K, 234K, and 204K. If these temperatures are treated as radiation equilibrium values, they are found to correspond to approximate heating rates 0.409, 0.396, and 0.204 kW/m^2, respectively. At these heating rates, then from Figure 3.3-1, an increase in α_s/ε from 0.5 to 1.0, yields new temperatures 294K, 281K, and 239K. These temperatures may be regarded as rough approximations to the desired peak transient temperatures. Since the temperature differential around the tube cross section (T_a - T_b in Figure 3.4-1) is primarily dependent upon the tube wall conductance, the (T_a - T_b) values are not expected to change for the new α_s/ε. There exists no conventient way to estimate the new minimum temperatures of the tube nor to extrapolate to new tube diameter-to-wall thickness ratios or new tube material properties other than surface properties. As conduction and/or thermal capacitance effects become more significant to thermal response, the extrapolation technique using Figure 3.3-1 becomes less accurate.

If structural temperatures can be approximated by steady state isothermal tube values, Figure 3.3-1 may again be used for extrapolations. For extrapolation to α_s/ε values other than those specifically covered by the charts, the procedure is the same as described in the preceding paragraph, beginning, for example, with values read from the broken curve of Figure 3.4-2.

For extrapolation to heating rates other than those specifically covered, the procedure is somewhat more involved but reasonable estimates of steady-state isothermal tube temperatures can be made. Suppose, for example, the standard aluminum tube of Figure 3.4-2 is illuminated by the sun at a value of λ_s = .524 rad., and by earth reflection and emission normal to the tube's axis (λ_e = 0). Further consideration of tube flux vector directional relationships is not necessary because of the isothermal cross section assumption. From Figures 3.2-1, -2, and -3, it is seen the projected area of the tube is subjected to $\dot{q}_e$ = 1.204 kW/m^2, $\dot{q}_e$ = 0.196 kW/m^2, and $\dot{q}_r$ = 0.443 kW/m^2, respectively. The data for a 408 km orbit were taken as a satisfactory approximation to the data for the 463 km orbit of the example. The isothermal cross section assumption, plus a further assumption of equal absorptance values for all three radiant sources, permits the three intensities to summed as scalars.

$$\dot{q}_{tot} = \dot{q}_s + \dot{q}_e + \dot{q}_r = 1.843 \text{ kW/m}^2$$

Since the flux is intercepted over an area equivalent to the tube's projected area but distributed over the tube's entire surface area, the total flux is reduced by the ratio of projected area to surface area, yielding the average radiant intensity per unit of surface area.

$$\dot{q}_{avg} = \dot{q}_{tot} \frac{dL}{\pi dL} = 0.587 \text{ kW/m}^2$$

The $\dot{q}_{avg}$ value may now be used to enter the curves of Figure 3.3-1, and at the desired α_s/ε, read the T_{eq} value as a steady-state isothermal tube temperature. In the sample case, with $\alpha_s/\varepsilon = 0.50$, the resulting temperature is 267K.

A potentially weak assumption in the preceding scheme is the use of a common absorptance value for all three sources of radiation. For many materials the absorptance at the effective earth emission spectrum will differ significantly from α_s. Since the earth emission flux is often the smallest of the three sources, however, the error arising from an incorrect absorptance value may not be serious.

3.7 Examples of Data Application

Three sample problems will be described to illustrate the use of the thermal response charts. As the first example, consider the tetrahedral truss configuration shown in Figure 3.7-1. All members of the truss are represented in the sketch of a typical repeating prime module shown in that figure. It is assumed that all structural members (elements numbered 1 through 9 on the prime module sketch) are 6061 T-6 aluminum tubes, 0.1016 meter in diameter with 4.06 mm thick walls. The members are all of equal length, 6.096 meters. The aluminum surfaces are anodized to yield surface properties, $\alpha_s = 0.42$ and $\varepsilon = 0.84$. Since the ratio of wall thickness to diameter is the same as for aluminum tubes treated in Section 3, and the α_s/ε value is that of the Section 3.4 and 3.5 aluminum tube charts, the handbook data apply directly to this example.

It is further assumed that heat flow through joints between members is

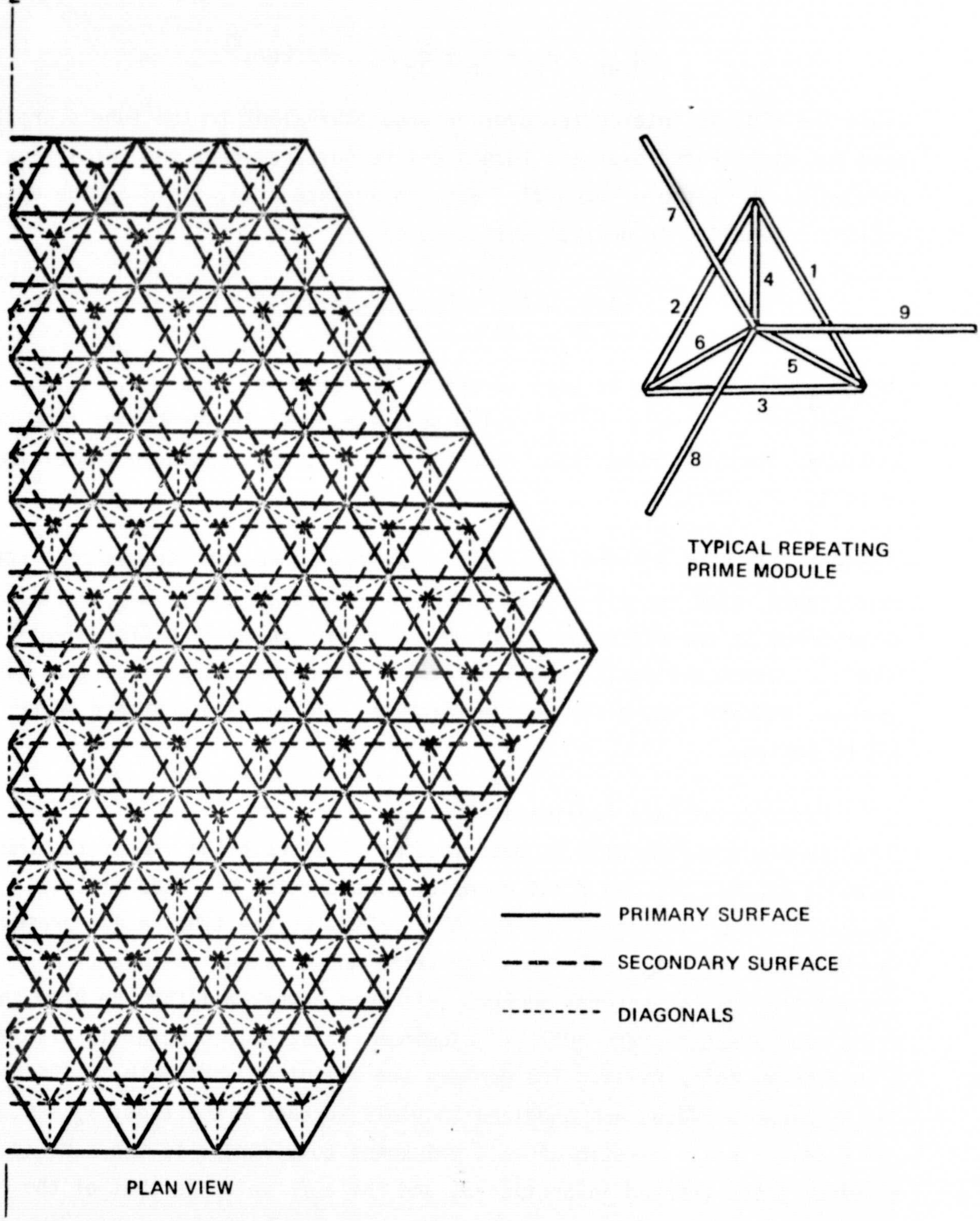


Figure 3.7-1: Tetrahedral Truss Configuration

negligible, that there is no shielding or heating by other spacecraft components, and the mutual shadowing of truss members is negligible. The validity of the last assumption can be assessed for solar heating normal to the plane of the truss by examining the ratio of member diameter to truss depth. For the geometry described the diameter of an upper surface member subtends an angle of 0.0204 rad. when viewed from the plane of the lower surface and vice versa. This angle is somewhat greater than the .0092 rad. subtended by the sun at 1.0 AU, resulting in the existence of umbra type shadows. The ratio of member length to diameter, however, is such that only a very small part of any member's total surface will be in shadow at any time, and the assumption of negligible shadowing is considered acceptable. Finally, for purposes of this example the structure is assumed to be in a circular geosynchronous orbit, with the plane of the truss normal to the sun. It is assumed that the members' nominal length is their actual length at a temperature of 289K.

Let the objective of this example be to find the member temperatures and the thermal distortion potential of the truss while in the stabilized sunlit portion of its orbit. Figure 3.4-16 shows that the tubular members are at the non-isothermal tube steady-state temperatures for a large portion of the sunlit part of the orbit. The figure also shows that the temperature differences between the illuminated and shaded sides of the tubes are not significant. Therefore, the isothermal tube temperatures are satisfactory approximations for this case.

Although the member temperatures could be read and interpolated from Figure 3.4-16, a more accurate presentation of isothermal aluminum tube steady-state temperatures in a solar-only heating environment is found in Figure 3.4-2.

Evaluation of the individual members' orientation relative to the solar flux vector yields the values of λ required to read Figure 3.4-2. The steady-state temperatures lead directly to values of temperature change from the initial condition. Then, using an approximate thermal coefficient of expansion of 20.7 μm/m-K, unrestrained potential length changes may be calculated. These results, for the members identified on the prime module of Figure 3.7-1, are tabulated in Table 3.7-1.

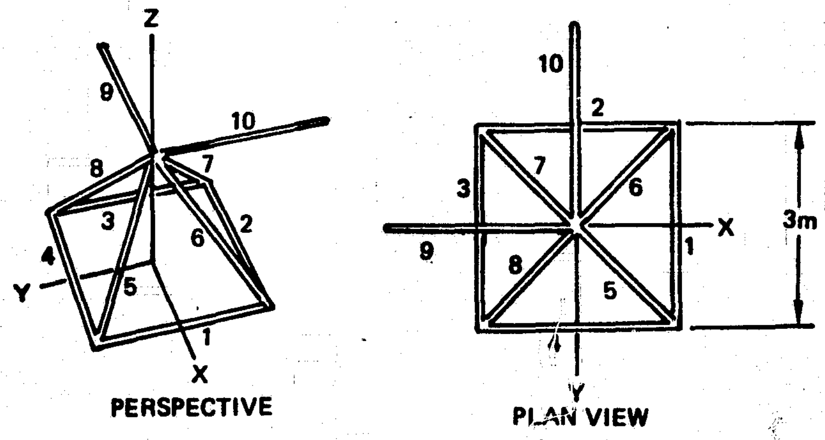
Table 3.7-1: Sample Temperatures and Length Changes For Tetrahedral Truss

MEMBER NO. SEE FIG. 3.7-1	λ	T	UNRESTRAINED ΔL
	rad	K	mm
1, 7	0	251	-4.80
2, 8	0	251	-4.80
3, 9	0	251	-4.80
4	.955	217	-9.09
5	.955	217	-9.09
6	.955	217	-9.09

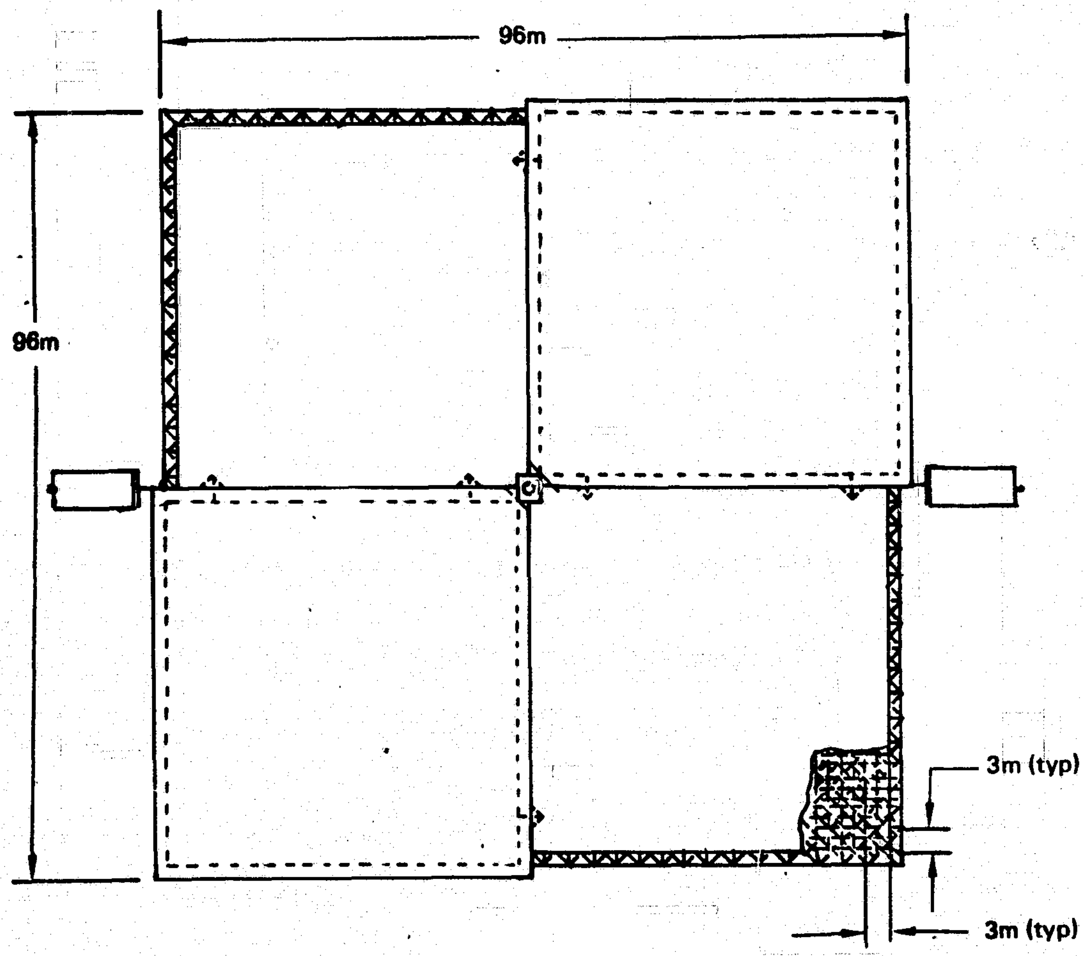
If the truss were constructed with flexible or pinned joints, the only consequence of the thermal response would be a slight charge in the geometric proportions (plan dimensions vs. depth) of the structure. Rigid joints, on the other hand, would provide resistance to the changes to member length, introducing both axial and bending stresses and distortion of the members. Accurate assessment of structural distortions in this case requires a formal stress analysis.

As a second sample problem illustrating the use of the document, cyclic temperature variations will be estimated for a large space antenna structural frame, consisting of repeating equilateral pentahedral figures, shown in Figure 3.7-2. A single prime module of such a structure is shown in the figure. Every member of the repeating part of the frame is represented, as far as orientation in space and geometric relation to other members are concerned, by one of the numbered members in the prime module views of Figure 3.7-2.

The structural members forming the pent[illegible] trusses were all assumed as 50.8 mm in diameter with a 0.508 mm wall [illegible]ckness. The material was graphite-epoxy composite whose properties were consistent with those on which the graphite-epoxy charts of Section 3.4 and 3.5 are based.



PENTAHEDRAL TRUSS
PRIME MODULE



ANTENNA PLAN VIEW

Figure 3.7-2: Large Space Antenna Structural Frame

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The large space antenna was assumed to be in a circular equatorial geosynchronous orbit with its axis (the normal to the plane of the structural frame) continuously passing through the earth's center, as shown in Figure 3.7-3. It was assumed that eclipsing of the satellite by the earth would not occur. Actually, in most equatorial or near-equatorial geosynchronous orbits eclipsing will occur but, because of the tilt of the earth's axis, will be limited to two brief periods each year. The assumption of no eclipsing avoids the problem of predicting the transient effects of eclipse passage while resulting in incorrect temperatures for only a small part of any prolonged time in orbit.

The effect of the earth axis tilt was ignored for the purpose of evaluating geometric relations between the antenna and the solar flux. As a result, the temperature predictions will fail to show a very small annual variation which in reality would be superimposed on the much more significant daily cycles. As a result of ignoring earth axis tilt, the X-axis of the antenna (Figure 3.7-3) remains normal to the solar flux and, through simple geometric relations, the solar flux incidence angle, λ, for each member (Figure 3.7-2) may be computed as a function of θ only.

The relating of λ to θ for each of the module members makes possible an evaluation of temperature versus θ from the charts of Section 3.5. The neglect of eclipse transients and the fact that changes in orientation relative to the solar flux occur rather slowly in geosynchronous orbit permit accurate approximations of member temperature histories as a series of steady-state values, i.e., a quasi-steady state approach. Figure 3.5-2 shows transient temperatures of graphite-epoxy tubes in geosynchronous orbit but the data of Figure 3.5-3 are more convenient to read at intermediate θ values and are adequate because of the no eclipsing assumption.

Temperatures read from Figure 3.5-3 as λ varies with θ are plotted in Figures 3.7-4 through 3.7-7. Member numbers 1, 3 and 10 became parallel to the solar flux at $\theta = \pi/2$ radians and at $\theta = 3\pi/2$ radians. On the basis of the simplified analytical approach, temperatures at these times would drop to absolute zero. In reality, thermal capacitance and heat flow by conduction and radiation from adjacent members will become significant and prevent temperatures from dropping to the isolated-member steady state value. Also,

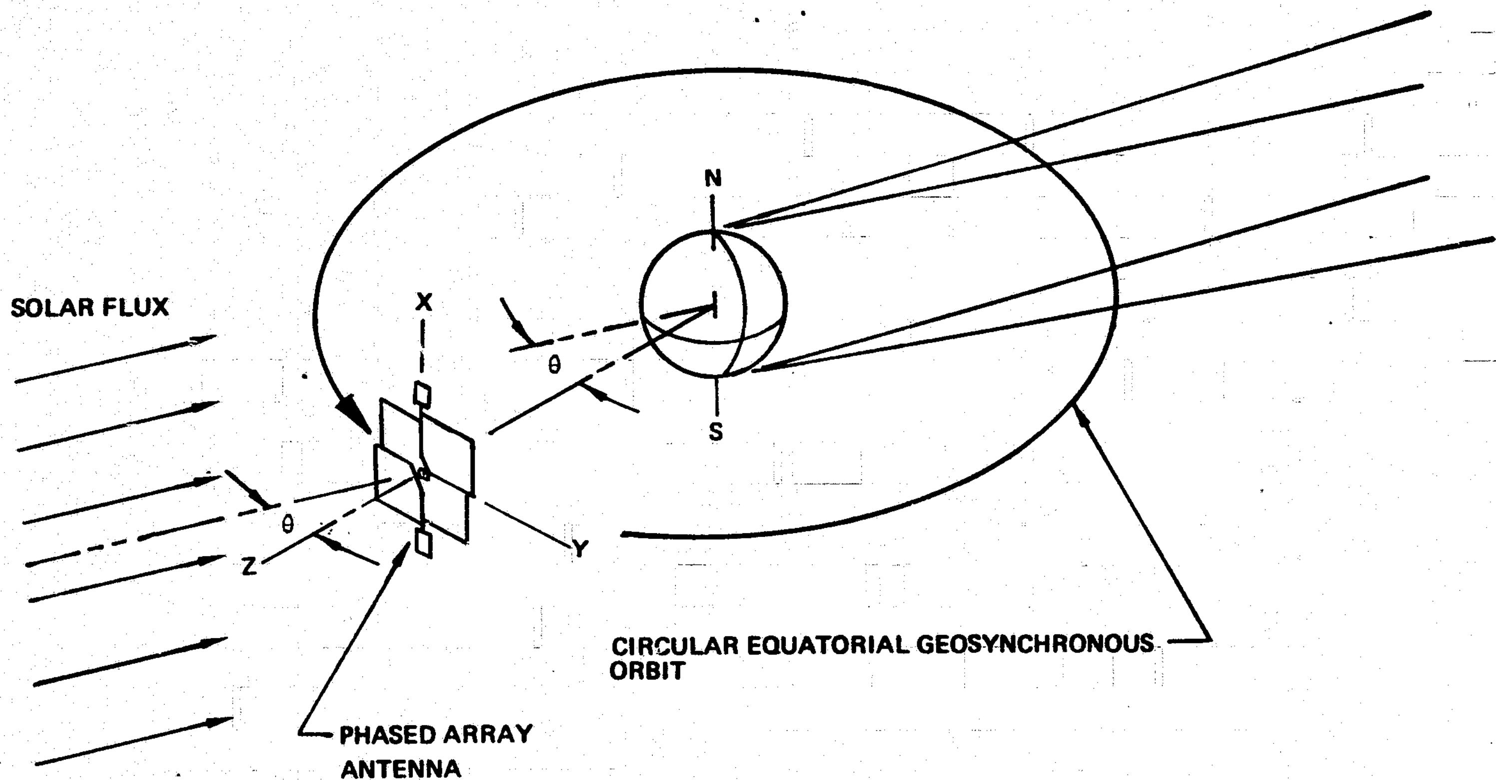

Figure 3.7-3: Simplified Orbit Geometry

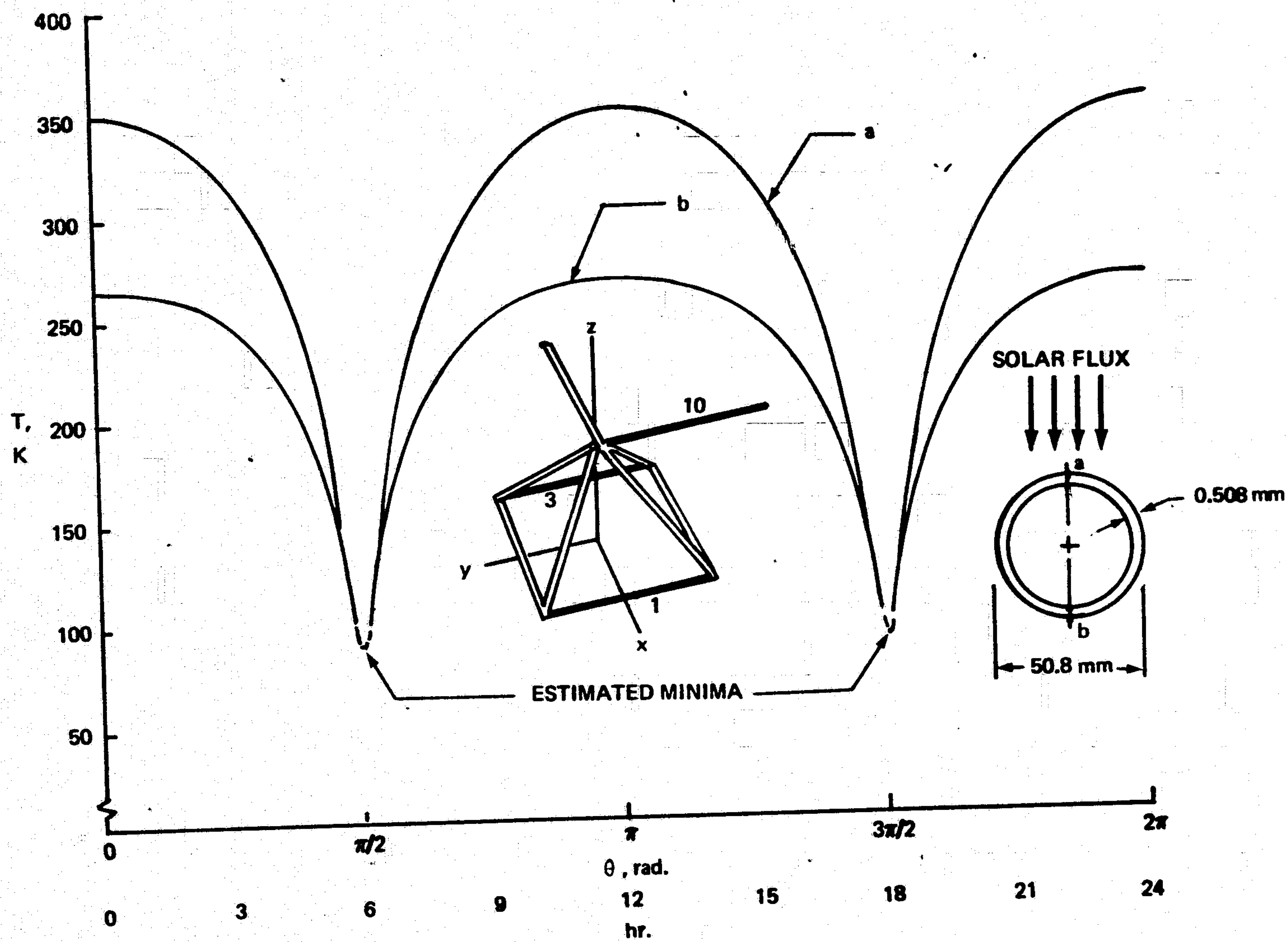

Figure 3.7-4: Temperature Histories—Members 1, 3, and 10

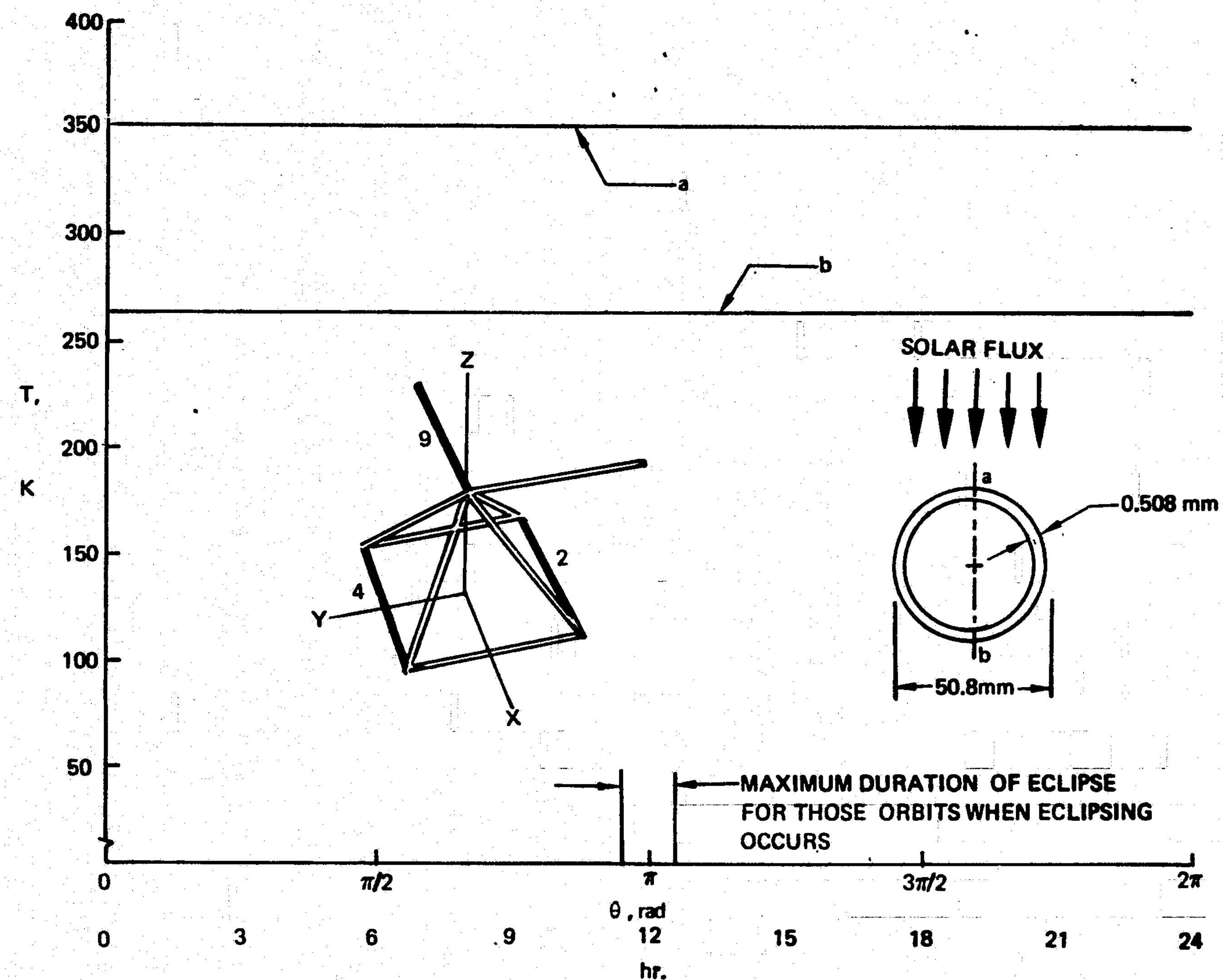

Figure 3.7-5: Temperature Histories—Members 2, 4, and 9

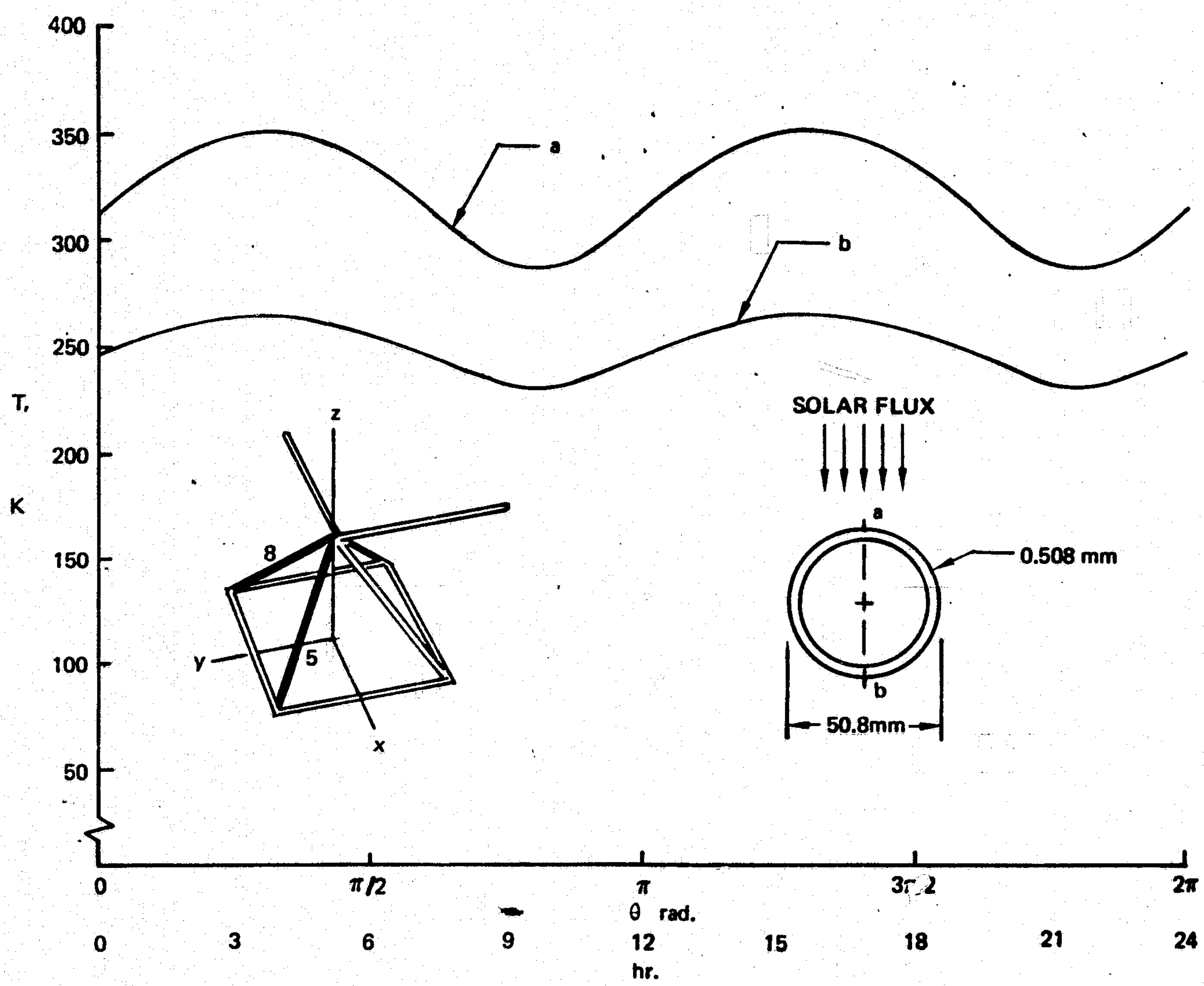

Figure 3.7-6: Temperature Histories—Members 5 and 8

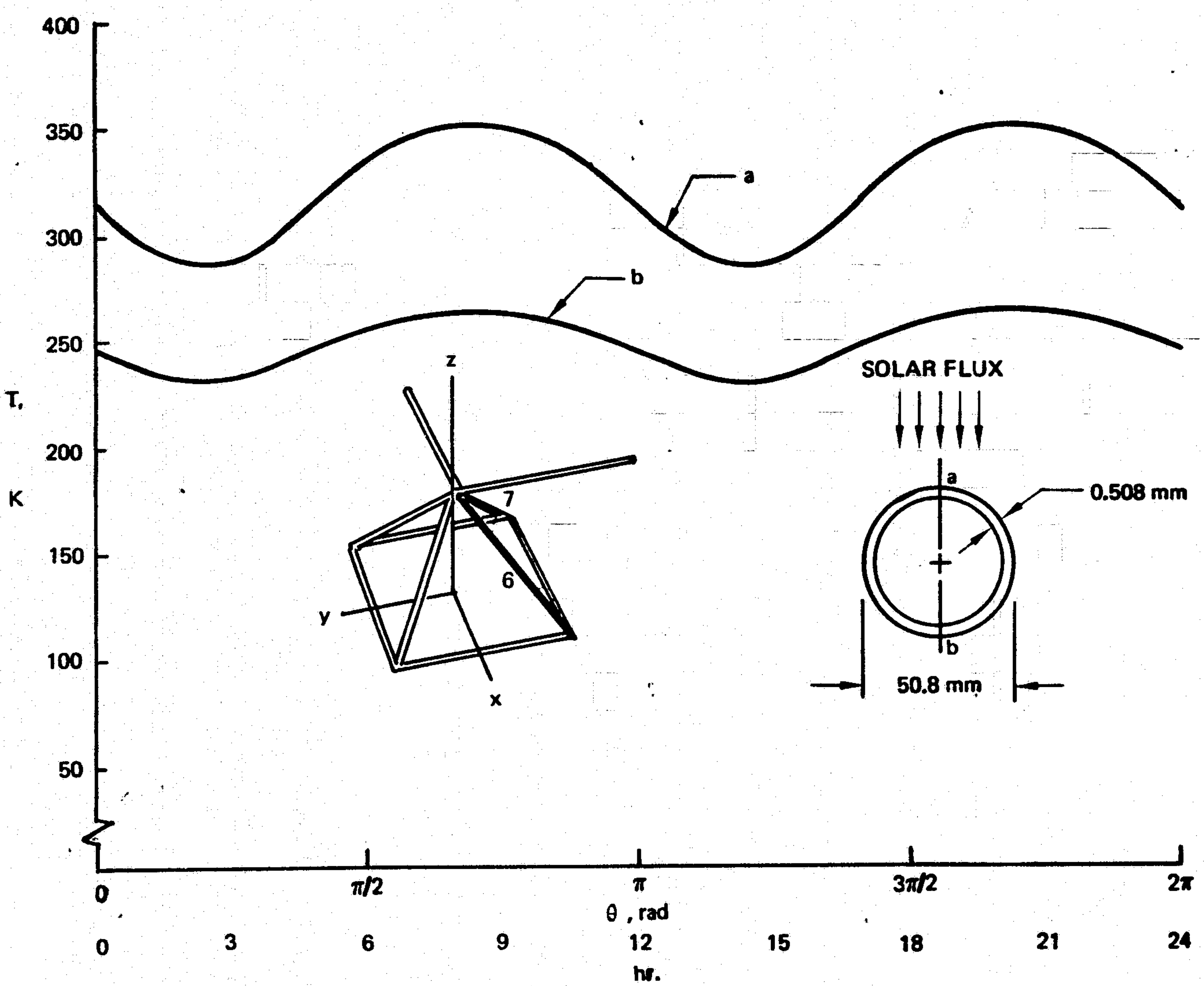

Figure 3.7-7: Temperature Histories—Members 6 and 7

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divergence of the solar flux results in some illumination to surfaces that are parallel to the mean solar flux vector. Temperatures at these times may be ignored or simply estimated, as was done for Figure 3.7-4.

For times during the life of the satellite when eclipsing occurs, the maximum duration of shadow passage (umbra plus penumbra) is approximately 72 minutes or 2.98 rad. $< \theta <$ 3.30 rad. This period is indicated in Figure 3.7-5 and although eclipsing would affect temperatures of all members during that period, no estimates of altered temperatures were made.

The third sample problem illustrates the use of Figure 3.4-16 in estimating temperatures of members of a structure in low earth orbit, including the effects of earth emission and reflection. The structure assumed is that of Figure 3.7-1, with members composed of tubular graphite-epoxy composite. The orbit is circular,near the ecliptic plane, and at an altitude of 463 km. The structure is oriented in orbit so that the plane of the truss continuously faces the earth. The temperatures of member number 6 (Fig. 3.7-1) will be estimated in the illustration.

By standard trigonometric relationships the member's solar flux incidence angle, λ, may be related to the position in orbit, θ, as defined in Figure 3.7-3. Alternately, λ may be related to the time point in the orbit, which has a period of 5630 seconds. The result is shown as Table 3.7-2.

Using the values of λ read from Table 3.7-1, the range of temperatures for member number 6 may be read from Figure 3.4-16. These temperatures are plotted as functions of θ or time in Figure 3.7-8. Between the 1720 second and 3910 second time lines the structure is in the earth's shadow and the temperature range is given by the notes accompanying Figure 3.4-16.

Table 3.7-2: Geometric Relations For Member No. 6 Of Tetrahedral Module

(REFERENCE FIG. 3.7-1)

TIME FROM NOON POSITION IN ORBIT (s)	GEOMETRIC POSITION IN ORBIT, θ (rad)	SOLAR FLUX INCIDENCE ANGLE, λ (rad)
0	0	-.9547
469	$\pi/6$	-.4747
938	$\pi/3$	.0244
1408	$\pi/2$	.5236
1720*	1.92	.8467
1877	$2\pi/3$	1.000
2346	$5\pi/6$	1.278
2815	π	.9547
3284	$7\pi/6$	.4747
3753	$4\pi/3$	-.0244
3910*	4.36	-.1920
4222	$3\pi/2$	-.5236
4691	$5\pi/3$	-1.000
5161	$11\pi/6$	-1.278
5630	2π	-.9547

*-TIME OF PENUMBRA PASSAGE

Figure 3.7-8 shows a transient isothermal temperature curve for member number 6 for comparison with the temperature range band estimated by the method of Section 3.4. The transient curve was obtained by a rigorous digital analysis and, for purposes of the comparison, may be considered the exact solution. The computed curve and the estimated band differ in two ways. First, the computed transient solution recognizes the member's heat capacitance, which causes a lag in its thermal response to rapid changes in radiation environment. Second, the computed solution incorporates discrete levels of absorbed earth-emitted and earth-reflected radiation, as determined by variations in flux incidence angles and by variations in ambient levels of reflected flux through the orbit.

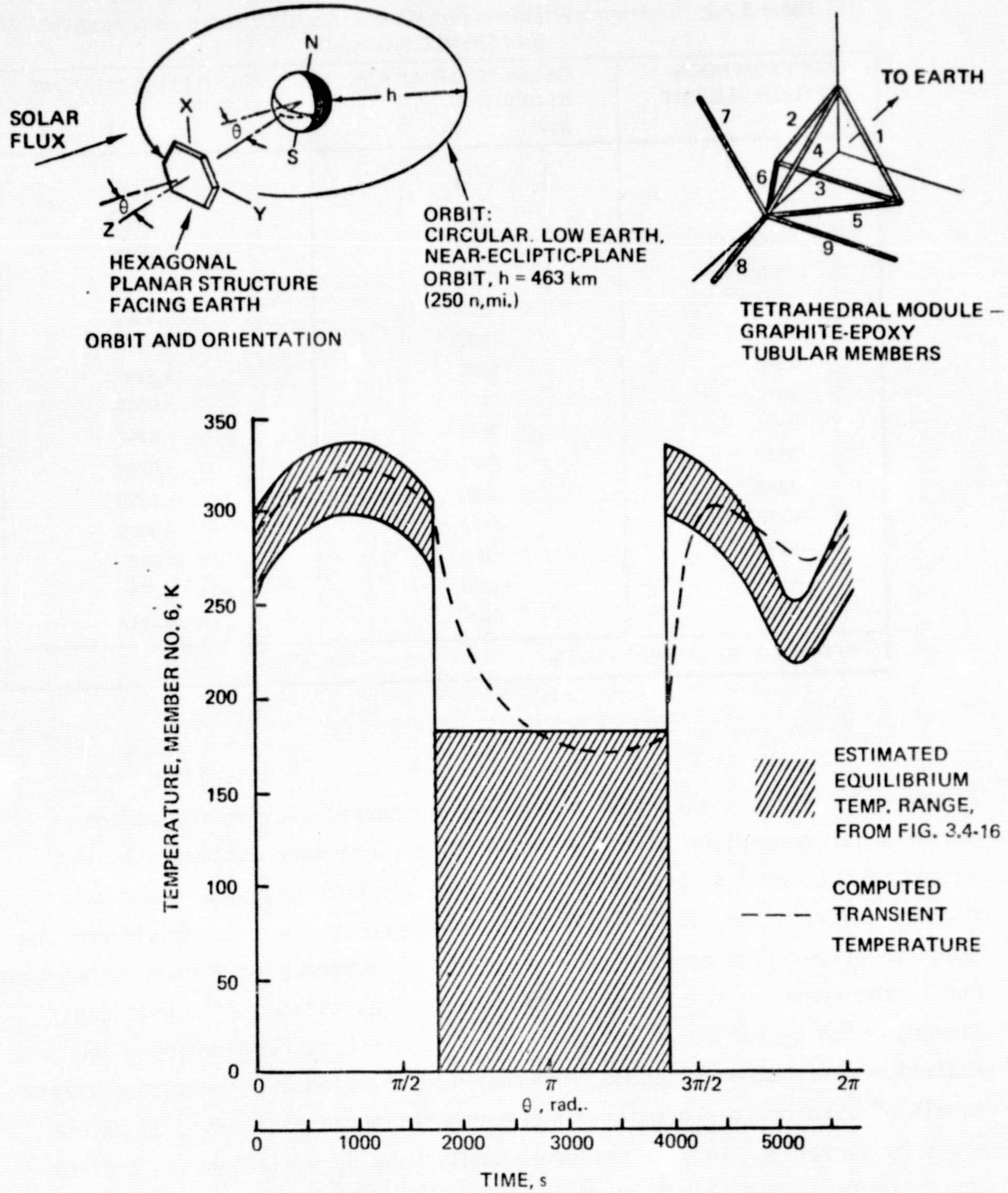

Figure 3.7-8: Sample Temperatures In Low Earth Orbit With Earth Radiation Included-Estimated Range and Actual Transient Values

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In the example of Figure 3.7-8 or any similar problem, the analyst with only the temperature range bands before him obviously cannot make an accurate estimate of member temperature during shadow passage. Some assistance in narrowing the band during shadow passage can be obtained by reference to the appropriate transient thermal response curves from Section 3.4, in this case, Figure 3.4-5. Although the curves of Figure 3.4-5 and those of Figure 3.4-4 for aluminum tubes do not include earth radiation, they do indicate the maximum rate at which the tubes will cool following sudden removal of all ambient heating upon shadow entry. The curves for cooling in the shadow may therefore be used to form a new, more realistic lower (or upper) boundary for the range of temperatures in the earth shadow portion of the orbit. The application of this modification to the sample estimate is shown in Figure 3.7-9.

SEE FIG. 3.7-8 FOR ORBIT AND STRUCTURE DEFINITION
AND FOR BASIC TEMPERATURE RANGE ESTIMATES

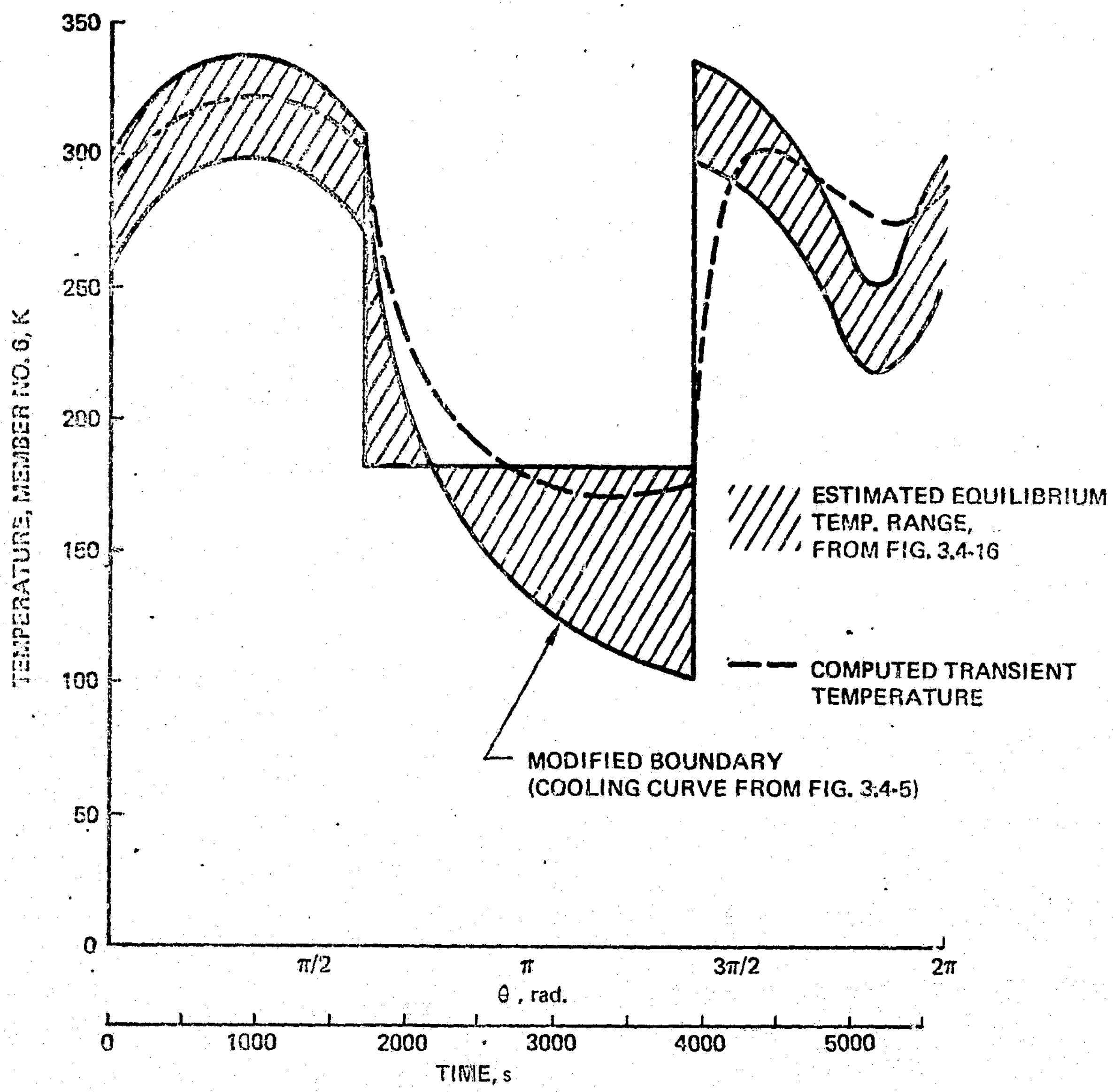

Figure 3.7-9: Sample Temperatures In Low Earth Orbit With Earth Radiation Included-Modified Boundary In Earth Shadow Passage

4.0 PART II - THERMAL DESIGN BACKGROUND

4.1 Design Requirements

Thermal design requirements for large space structures can be defined in two categories. First, the design must ensure against the loss of structural integrity due to thermal effects throughout all life phases. Material degradation or permanent property changes due to excessively high or low temperatures must be avoided. Damage from overstressing or buckling due to thermal deformations cannot be allowed. Data in this document are applicable to environments encountered during deployment and erection, operation or check-out in low earth orbit, transfer to operational orbit, and operation in final orbit. Structural thermal design integrity is also required in the prelaunch ground environment and during launch but these phases are not treated here.

The second general thermal design requirement is that the structure maintain some specified degree of geometric stability during the operational mission. The stability requirements must be met both in terms of the static geometric shape as affected by temperature distributions and material thermal expansion characteristics, and dynamic stability of the structure as affected by temperature-dependent elastic properties and motions induced by periodic temperature changes.

4.2 Mission Thermal Environments

An infinite variety of earth orbits is possible, each orbit having its own particular thermal environment history. Also, thermal environment patterns can change significantly as a particular orbit experiences diurnal and seasonal changes in earth-sun-spacecraft geometric relationships. Certain common thermal environment characteristics, however, can be identified with certain broad classifications of orbits or orbit-spacecraft orientation combinations.

Low Earth Orbit

Thermal environments in low earth orbit are characterized by eclipsing on every orbit, nearly equal time in sun and shadow, and rapid passage through the penumbra, as illustrated in Figure 4.2-1. As a result, rapid excursions of spacecraft surface temperatures are unavoidable. Earth-emitted flux may be

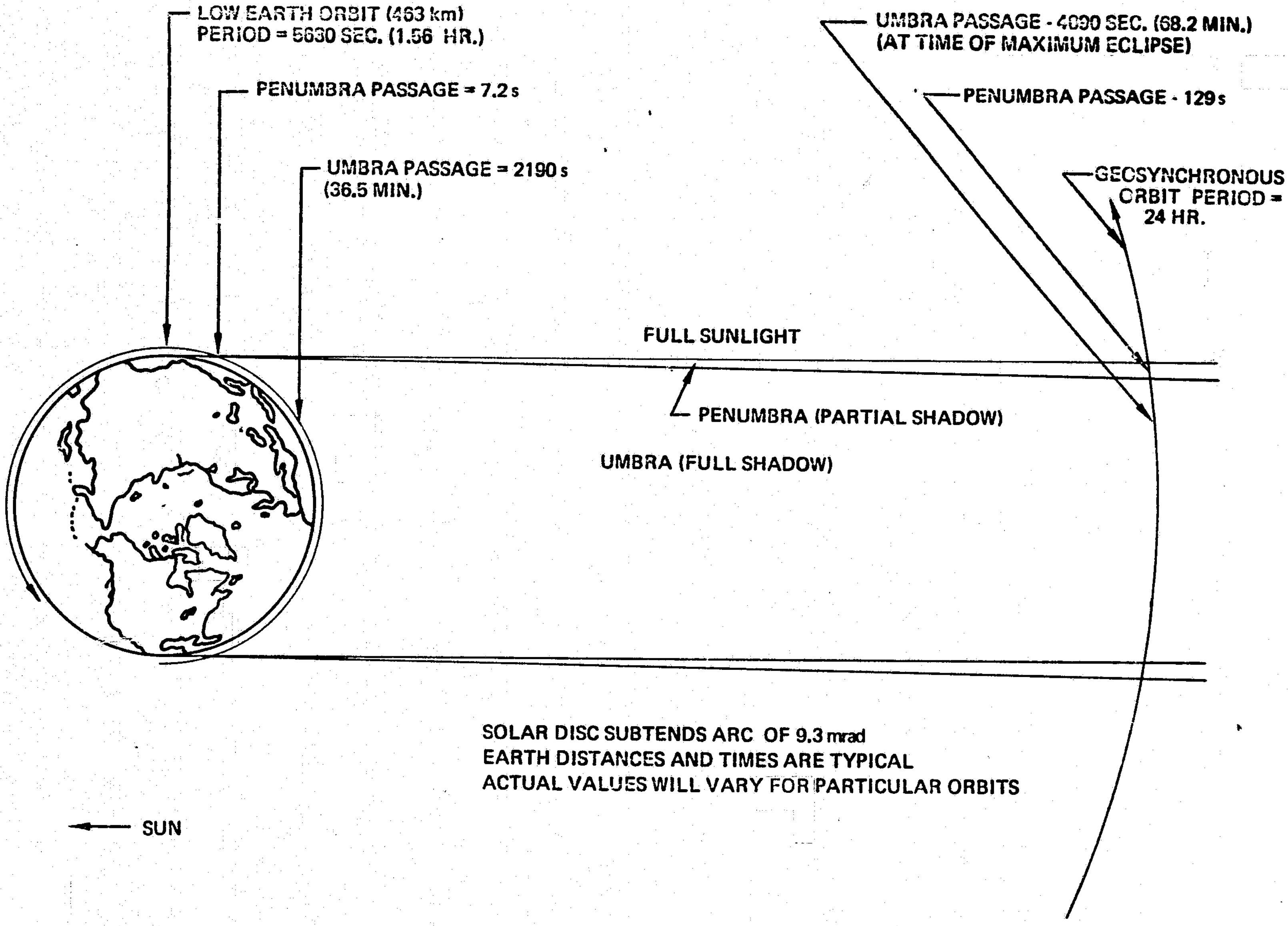

Figure 4.2-1: Orbit and Earth Shadow Geometry

signficant in moderating the low temperatures experienced on the night side. Earth-reflection and earth emission on the other hand, can add significantly to satellite heat load on the day side.

Geosynchronous Orbit-Space Fixed

A spacecraft or portion of a spacecraft whose orientation is fixed relative to a target in interstellar space, including the sun (Figure 4.2-2(A)), receives solar radiation whose direction and intensity are nearly constant with time. Eclipsing by the earth may be limited to a few orbits at two periods each earth year. (See Figure 4.2-3.) Maximum eclipse duration is roughly twice as long as low earth orbit night passage and penumbra duration is also of longer duration, as shown in Figure 4.2-1. Earth emission and reflection are insignificant to all but the most detailed thermal analyses.

Structural thermal response in space-fixed geosynchronous orbit is characterized by temperatures at or approaching the steady-state values for the particular configuration and orientation. Excursions will, of course, occur as a result of eclipses but for many applications, e.g., solar power satellite or solar observatory, structural deflection tolerances may be relaxed during these periods when the primary mission is interrupted.

Geosynchronous Orbit-Non Space Fixed

A spacecraft or portion thereof which constantly faces the earth (Figure 4.2-2 (B)) or performs a space-scanning function will receive solar radiation from constantly varying directions. Thermal response will be characterized by large slow excursions of surface temperatures of individual structural members. Changing orientation with respect to incident solar flux as well as changing shadow patterns from satellite components or other structural members must be considered. In addition, eclipsing patterns and effects similar to those mentioned for the geosynchronous orbit-space fixed case may be encountered.

4.3 Thermal Control Approaches

Design approaches for ensuring that a structure meets the thermal design requirements may be classified under three categories for discussion purposes. Combinations of schemes are, of course, possible and probably necessary for design optimization. The three general approaches are to: (1) minimize

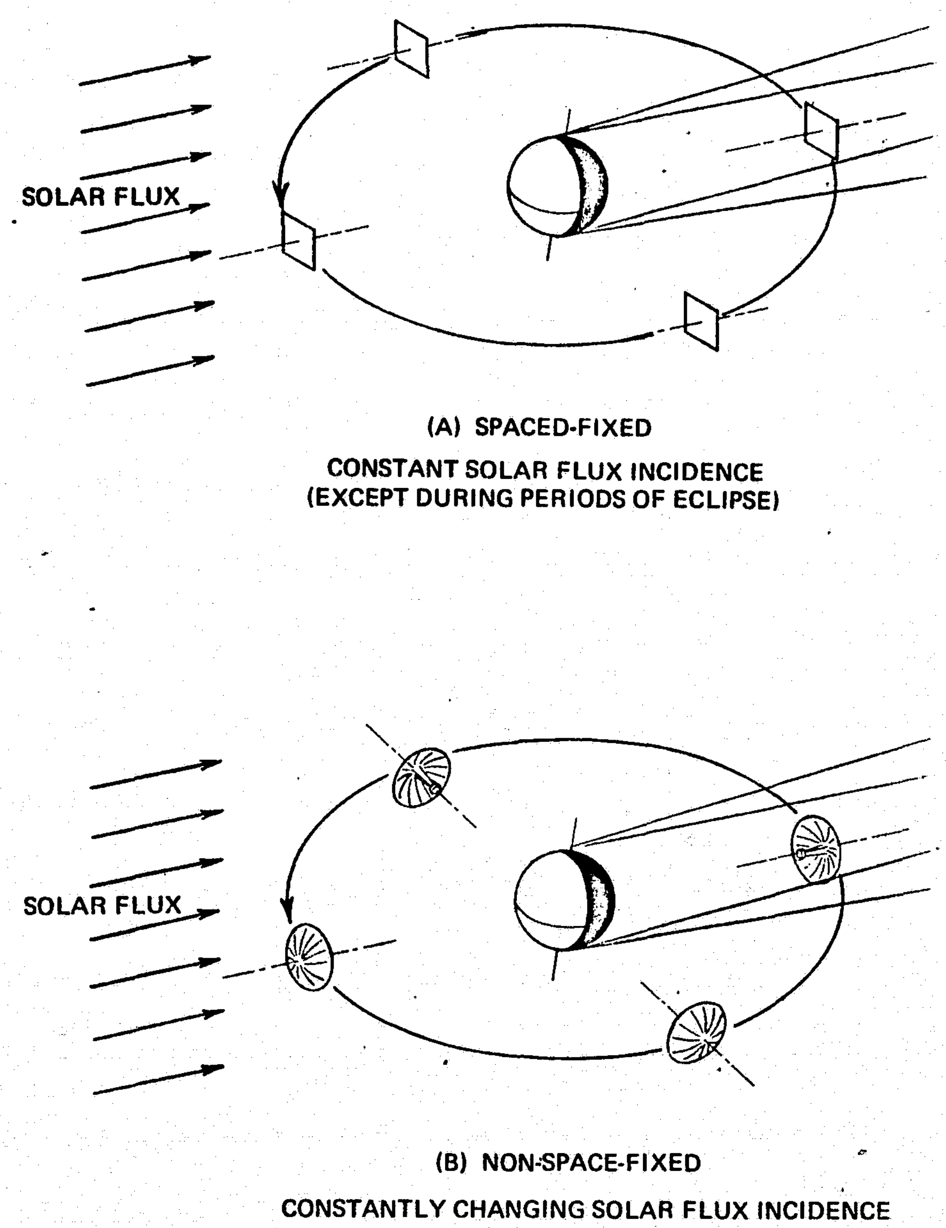

Figure 4.2-2: Satellite Orientations In Geosynchronous Orbits

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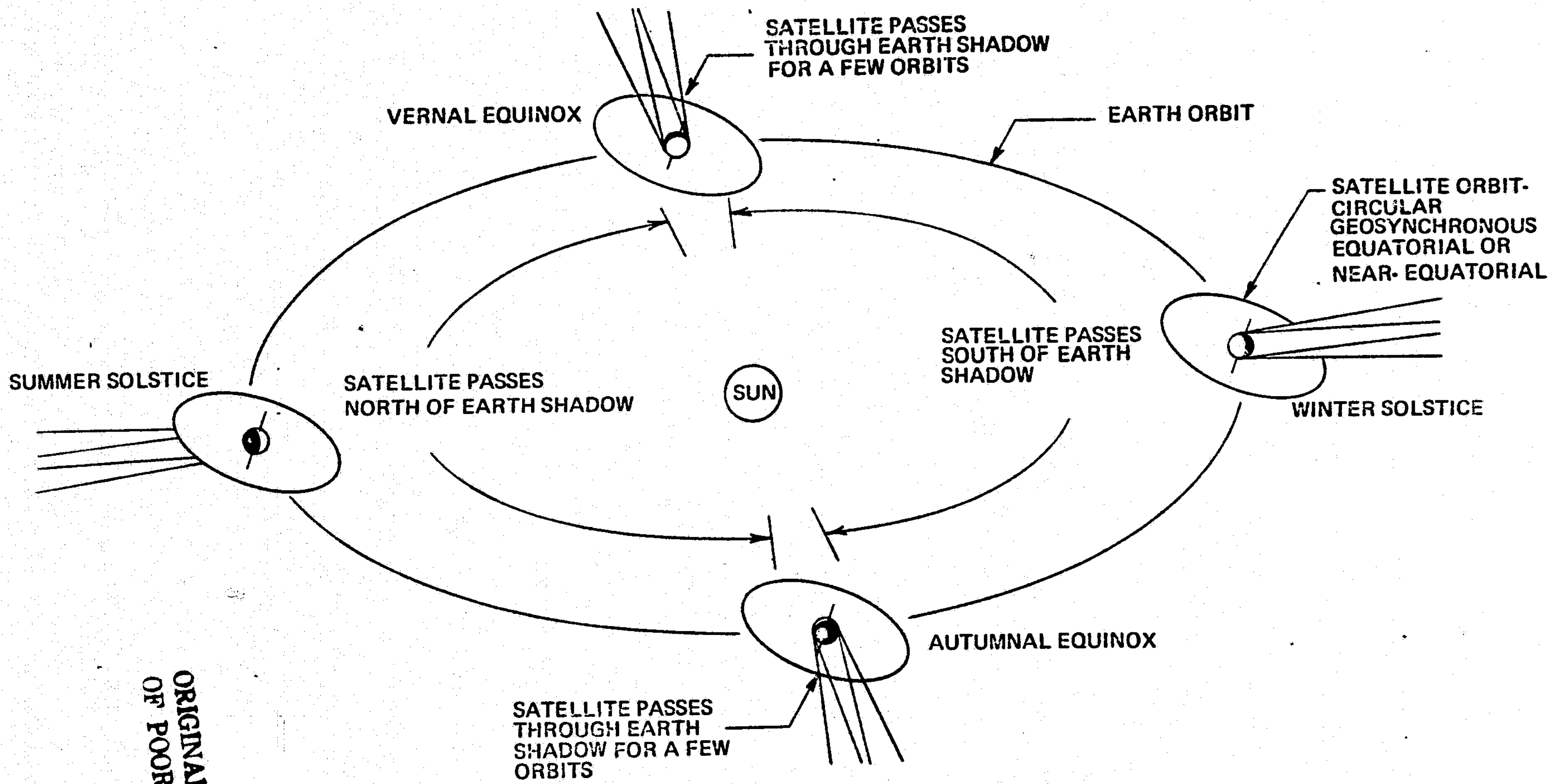

Figure 4.2-3: Geosynchronous Satellite Eclipse History

thermal environment levels or variations thereof, (2) minimize thermal response to environment, and (3) minimize mechanical response to the thermal state.

Minimizing environment levels is included as a candidate approach mainly for the purpose of listing all possibilities and will be discussed only very briefly. Isolating the whole of a large structure from the environment, e.g., solar radiation, by shielding is often not practical. Some degree of shielding might accrue to a structure which supports a non-structural solar cell array or other such continuous extensive opaque component. Shielding of individual structural members would involve mechanical attachments and geometric proximity, resulting in significant member-shield thermal interaction, more properly placing that scheme in the category of minimizing thermal response.

Minimizing thermal response to the environment, i.e., minimizing thermal gradients and thermal excursions, can be accomplished through either passive or active schemes. Passive schemes including coatings, paint, or other surface treatment to produce desired values of solar spectrum absorptance and infrared emittance, and passive shielding such as insulation or shadow shields. Thermal response can also be influenced in a passive sense through selection of high thermal conductance designs to minimize gradients and high thermal capacitance to minimize rates of temperature change with changes in environment. Thermal response control can be achieved through semi-active schemes such as thermally actuated louvers, discs, etc., which automatically vary the average surface absorptance and emittance in such a way to reduce temperature variations. Finally, active control systems which transfer heat by thermodynamic, mass transfer, or electric means can be used for thermal response control. Heat pipes are included in this latter category even though they may require no external power or control system.

Minimizing mechanical response to the structure's thermal state is not a thermal control in the direct sense but does offer a means of meeting the stress, distortion, and geometric stability aspects of the thermal design requirements. Means of minimizing mechanical response can, like the thermal response control techniques, be classified as passive, semi-active, and active. The principal available passive scheme is the use of materials having low thermal expansion coefficients. The use of low elastic modulus secondary members or components, e.g., reflector mesh or shield standoffs, to minimize loading of primary structure by deforming non-structural components also falls under this classification. Semi-active mechanical response control can be achieved through compensating devices such as temperature-actuated jack screws or other such mechanisms employing differential thermal expansions, mechanically amplified, as their principle of operation. Fully active control systems, probably employed at the mounting points of the geometrically-critical operational systems (reflector panels, antenna arrays), could be used to compensate for thermally induced structural deflections.

4.4 Thermal Control Alernatives - Applicability and Assessment

The description of thermal control approaches in Sec. 4.3 included suggestions for certain practical alternative schemes for achieving varying degrees of thermal control. The consideration of orbital mission thermal characteristics yields insight as to applicability of these thermal design alternatives. Other aspects of the alternatives, such as weight, cost, fabricability, and power requirements, cannot be assessed in any degree of detail for this discussion but will be considered in the summary assessment that follows.

Surface Property Tailoring

The tailoring of surface properties (absorptance and emittance) by use of coatings, paint, or other surface treatments offers a simple, light-weight, and reliable means of maintaining a particular desired temperature in a given radiation environment. A wide range of property values are generally available, affording considerable freedom in temperature selection. Base material-coating compatibility considerations may introduce some limitations on obtainable characteristics and most surfaces will experience some change in properties with time in the space environment. Surface property tailoring cannot counteract the effects of changing orientation of the surface relative to the direction of incoming radiant flux, nor for major variations in the level of incident flux, such as by eclipsing or shading.

Thermal Conductance and Capacitance

Designing for high conductance or capacitance through choice of structural materials or member dimensions will aid in minimizing temperature gradients or rates of temperature change. A change in materials or dimensions purely to effect an increase in conductance or capacitance will often incur weight or other structural capability penalties that are unacceptable in relation to a modest thermal control advantage gained. This fact, plus other constraints on material choices, limits the usefulness of this approach as a primary thermal control scheme. The tailoring of conductance and capacitance should be regarded only as a secondary device, augmenting the effectiveness of other features of thermal design. Since capacitance affects temperatures only during transients, conductance is normally the more important property for design considerations.

Passive Shielding

Passive or fixed shielding schemes (insulation or shadow shields) offer a reliable means of influencing structural temperatures, with some capability for reducing effects of changing orientations relative to incident flux and of periodic eclipsing or shading. Tailoring of surface properties, of both shield or insulation and protected members, is an integral part of this scheme. Capability is limited, however, as insulation cannot isolate a member from a changing thermal environment indefinitely, and shadow shields cannot maintain a constant structural temperature for all orientations (see Fig. 3.4-13). Potential installation problems exist with shields or insulation and some cost and weight penalties are almost certain.

Semi-active Shielding

Semi-active shielding includes all design devices by which shield geometry or average surface properties are changed by some mechanical device which responds and is driven by a temperature change. Such schemes, which have been extensively used for thermal control of small spacecraft components, are really extensions of the passive shielding approach. Capability is extended but is still subject to the general limitations of shielding. Furthermore, significant additional penalties in weight, cost and reduced reliability can occur.

Active Thermal Control

Active thermal control schemes, ranging from closed, self-activated heat pipes to heat pump systems, are probably not practical for large expanses of structure and are listed here only for completeness. Weight, cost, power requirements, and possible reliability problems must receive close attention in considering this approach as a means of achieving structural thermal control.

Minimum Thermal Expansion Material

The use of structural materials having zero or very low coefficients of thermal expansion is a reliable means of meeting thermal design requirements, attractive because it bypasses the problems of controlling temperatures. Material choices are severely limited and the scheme would not be a strong candidate except for the availability of certain structural composites. Use of other thermal design schemes may be required in conjunction with the minimum expansion approach, because of probable inability to provide truly zero expansion and because of possible material temperature limitations other than those related to expansion characteristics. Some weight penalty may be incurred through minimum expansion design, as compared with designs optimized purely for mechanical strength or stiffness.

Deflection Compensation

Grouped under deflection compensation are schemes ranging from semi-active, temperature-activated devices which compensate on a member-by-member basis, to active control systems employing sensors and servomechanisms to maintain a stable platform. Such systems have a potential for full satisfaction of geometric stability requirements and, in fact, may be required to compensate for effects of unavoidable non-thermal loads regardless of the degree of stability provided by schemes that minimize thermally-induced distortions. Deflection compensation devices are certain to add weight and cost, introduce reliability problems, and possibly require expenditure of power. The demands placed on such systems, if required, should, therefore, be minimized through the concurrent use of more passive thermal control schemes.

4.5 Thermal Design Guidelines

Several useful observations may be made from the data of Section 3.0 and the analyses from which they were derived. Earth emission and reflected radiation may be ignored for most preliminary thermal evaluations. Radiation reflected or emitted from other spacecraft components near the structural elements, however, may be very significant.

Many uninsulated candidate structural elements' temperatures approach or equal their steady state values near the end of sunlit periods in low earth orbit. In geosynchronous orbit steady state temperatures for these members exist for most of each sunlit period. A moderate amount of insulation on a typical structural element appears to effectively eliminate sun-shadow temperature fluctuations, at least for low earth orbits.

The fourth-power relationship between temperatures and radiated heat flux results in wide differences in rate of heat loss between high-temperature surfaces and low-temperature surfaces as a spacecraft passes into the earth's shadow. High-temperature areas cool rapidly and low-temperature areas cool more slowly, causing the spacecraft as a whole to approach a uniform temperature condition during eclipse periods. Conduction and radiant interchange between members would act to bring temperatures during eclipse even more nearly uniform than would be predicted on the basis of isolated structural members.

Design guidelines for specific structural material and configuration properties fall into two categories: (1) unconditional guidelines, i.e., those for which no exceptions or qualifiers are required, and (2) conditional guidelines, i.e., design characteristics which are usually but not always desirable. Even without consideration of non-thermal implications (weight, cost, and other structural requirements), very few unconditional guidelines can be stated. Therefore, the list of unequivocal desirable design features is brief, and careful attention must be given to all possible implications of any thermal design feature. The preceding statement should, perhaps, be regarded as the cardinal thermal design guideline.

Unconditional Guidelines

The following characteristics are desired features for all large space structures. Even these features must be subject to compromise, however, when non-thermal structural or system requirements are considered.

1. Minimum thermal expansion coefficients of structural materials.
2. No permanent structural material property degradation between approximately 140K and 420K.
3. Maximum structural depth-to-member diameter ratio to minimize structural temperature gradients due to member shading.

4. Material thermal properties stable in the space environment.

Conditional Guidelines

The following characteristics may be generally desirable or desirable under certain circumstances. Some of the characteristics are seen to be mutually conflicting. Conditions or exceptions are described to the extent that such a general listing will permit.

1. A low value of solar absorptance (α_s) - to - infrared emittance (ε) ratio will minimize temperatures of sunlit surfaces, probably minimizing sun-shade temperature variations. Low temperature limits, arising either from requirements of compatibility with other temperatures or from material considerations may introduce exceptions.
2. A low value of ε is desirable where heat retention is desired, e.g., a surface which never faces the sun or is periodically shaded.
3. A high value of ε may be desired where radiant heat exchange between members will act to reduce undesirable temperature gradients.
4. High thermal conductivity or conductance is desired to minimize temperature gradients except where isolation of non-structural on-board heat sources is desired.
5. High thermal capacitance (density-specific heat-volume product) for structural members is desirable to minimize rates of change of temperature, but compatibility of transient temperatures between interacting members may override a desire to simply minimize rate of change in a particular member.
6. High elastic modulus or stiffness of critical-dimension members, in order to resist deflections, is desired except in circumstances where overstressing would result.
7. Low elastic modulus or stiffness of secondary components, e.g., reflector mesh or shield stand-offs, to minimize temperature-induced loads in primary structure, is desired.
8. Statically determinate structure is desired to minimize thermal stresses but this feature must be weighed against the value of rigid joints in enhancing overall structural rigidity, and the difficulty of providing freely rotating joints.

9. Configurations employing repeating identical subassemblies (prime modules), resulting in uniform member density, appear to be preferable to radial type configurations but more analyses are needed to confirm this guideline.

Additional general thermal design guidelines arise from the consideration of the varying degrees of suitability of the thermal design alternatives of Section 4.3 in the differing mission environments of Section 4.2.

In low earth orbits attractive means of meeting thermal design requirements appear to be use of low-expansion materials, active temperature or deflection control, and the use of insulation on high thermal capacitance members. Because of the inevitable day-night solar flux variations, surface property tailoring or simple shielding appear inadequate as primary means of thermal response control.

In a space-fixed geosynchronous orbit structural thermal response is characterized by temperatures at or approaching the steady-state values for the particular configuration and orientation. Excursions will, of course, occur as a result of eclipses but for many applications, e.g., solar power satellite or solar observatory, structural deflection tolerances may be relaxed during these periods when the primary mission is interrupted.

It is possible for the space-fixed geosynchronous orbit that thermal response control could be achieved primarily through selection of appropriate surface properties (absorptance and emittance). Use of high conductance members and joints and attention to configuration characteristics (member orientation and shadowing) can assist significantly in minimizing temperature gradients. All other alternate schemes; low expansion materials, insulation, active thermal and deflection control, are available and potentially beneficial, particularly if close control of geometry through eclipses is required.

Attractive thermal design alternatives are limited for the case of the non-space-fixed geosynchronous orbit. Obviously, environmental variations are too wide for primary control of temperature variations through use of surface property tailoring or simple shielding. Insulation and high thermal capacitance may not be effective because of the long times available for heat to penetrate to or be lost from structural members at the extremes of environmental levels. As always, low-expansion materials offer an effective means of meeting deflection tolerances, as do means of active or semi-active mechanical deflection compensation.

4.6 Thermal Analysis Approach Choices

At the outset of any effort to compute thermal response of space structures decisions must be made as to the use of approximations or simplifying assumptions, and the inclusion or omission of effects of marginal consequence. For purposes of assessment of analysis approach possibilities the choices may be considered under three aspects of thermal analysis: (1) environment prediction, (2) individual member response, and (3) member interaction.

Environment Prediction

Thermal environments to which space structures are subjected can consist of thermal radiation from both natural and man-made sources. The man-made sources (on-board heat sources or those on other spacecraft in close proximity) have been ignored throughout this document. Their possible characteristics and output levels are so varied that treatment in a general discussion is impossible and any analysis which includes significant man-made heat sources must be considered a special case.

The principal analysis approach decision to be made relative to natural environment prediction pertains to the inclusion or omission of earth-reflected and earth-emitted radiation. The levels of earth radiation are less than that of solar radiation (except, of course, during eclipse) but may be significant to bodies in low earth orbit or surfaces shielded from the sun. Earth radiation can almost always be ignored for geosynchronous orbits or missions beyond that distance from the earth. For this reason, during the remainder of this section, the choice between inclusion or omission of earth radiation in the thermal environment definition will be assumed to be dictated by the type of orbit, characterized as low earth or geosynchronous.

The geosynchronous orbit environment definition is simple since only one thermal radiation flux vector, with a fixed direction and a constant, well-established, magnitude is involved. The low earth orbit environment is considerably more difficult to define because of the various ways in which the significant flux vectors can add and the uncertainties associated with seasonal, diurnal, and local variations in earth albedo.

Member Response

Response of individual structural members to thermal influences in orbit may be treated as transient or steady-state. The transient case is always the more accurate but the simpler steady-state analysis, which assumes the heat flow to the member exactly equals the heat flow from the member, can result in quite accurate temperature predictions. The choice will be influenced by the members' thermal capacitance relative to their areas available for heat transfer, by the rate of change of imposed thermal environment levels, and by the need for response assessment during transitory phases of flight.

Steady-state member response requires only a simple analysis, consisting of solution of the radiation heat balance and heat flow within the member. The transient approach is more complex, requiring definition of member thermal capacitance and in effect, the solution of a more difficult form of differential equation. Except for the very simplest cases, machine computation is the only practical means of performing transient analyses.

Member Interaction

Individual structural members and other space craft components can be considered isolated as far as their thermal response is concerned or can be considered as interactive. Of course, the interactive case is the only strictly correct approach but the isolated response approach can yield accurate approximations, depending upon configuration characteristics. Interactions can occur as conduction or radiation heat interchange, or simple shading of one member by another. Significance of interactions will depend upon member conductance and upon general configuration geometry as it influences radiation view factors and blockage.

The isolated member approach is always relatively simple, usually requiring only a two-dimensional and sometimes only a one-dimensional analytical model. The interactive approach can be simple to very complex, depending upon configuration characteristics and the level of detail modeled. A three-dimensional analytical model will almost always be required and machine computation is practically essential. If combined with the transient choice with respect to member response, as will often be appropriate, digital computer time requirements may be quite large.

Summary

The thermal analysis approach choices as categorized in preceding paragraphs may be viewed as three variables, each of which can take one of two possible values. Thus there exist eight permutations representing the eight different combinations of choices regarding analysis approach. A graphical figure illustrating the combinations of choices was devised as a means of summarizing the characteristics of and relations between the various choices. The figure, an octahedron, is illustrated in perspective in Figure 4.6-1 and a template for construction of the three-dimensional figure is shown in Figure 4.6-2. The three pairs of opposing apexes of the octahedron are identified with the three pairs of individual choices and the eight sides correspond to the eight three-component combinations of choices. On the version of Figure 4.6-2, some of the important characteristics of each combination have been listed on the faces of the octahedron.

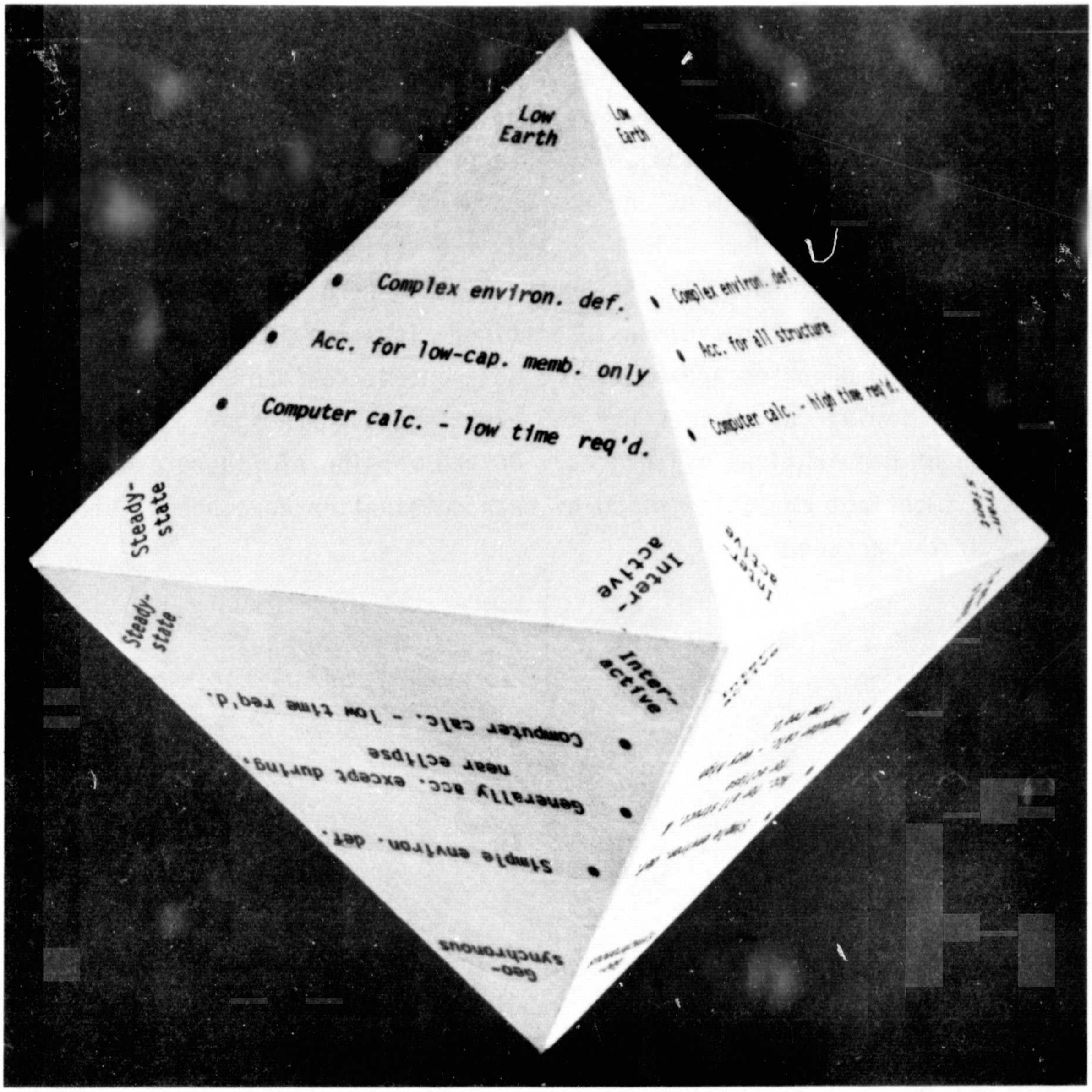

Figure 4.6-1: Octahedral Diagram—Combinations Of Thermal Analysis Approaches

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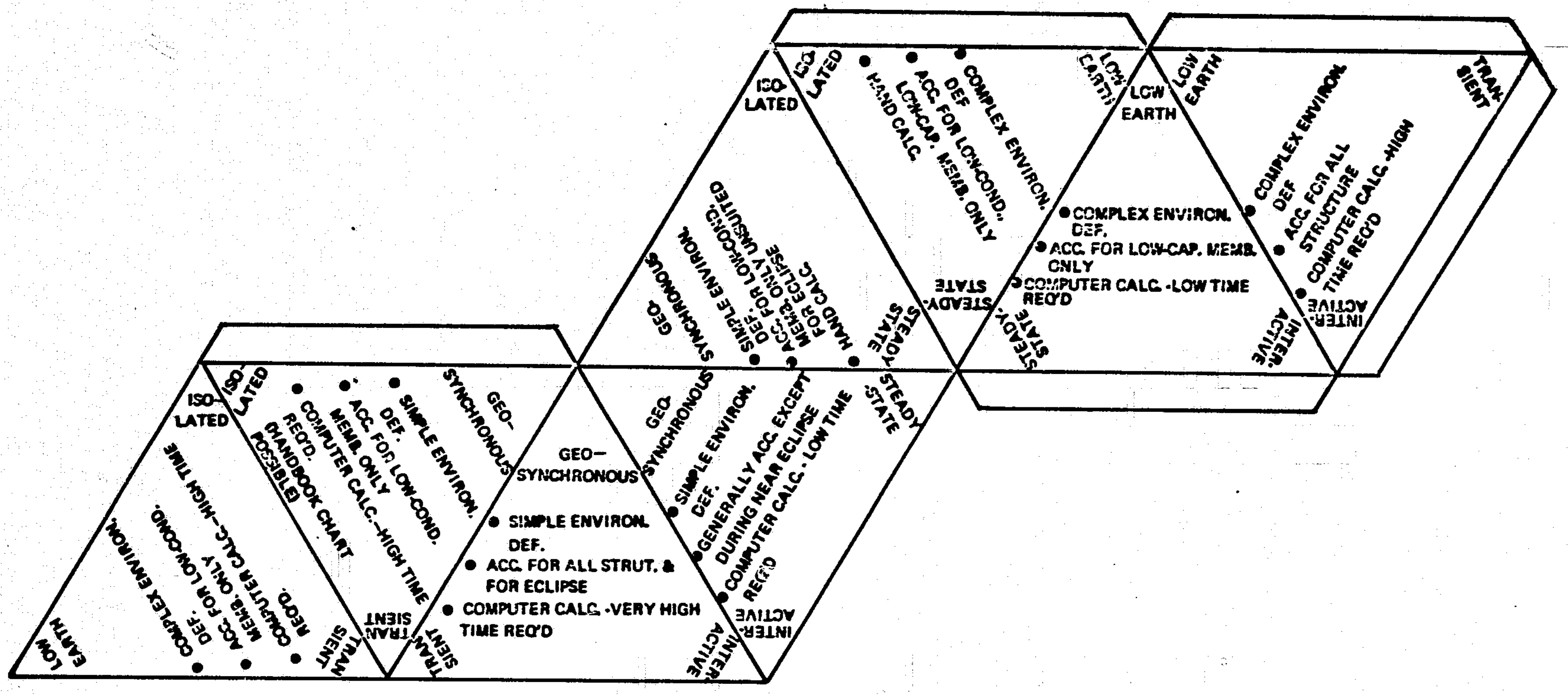

Figure 4.6.2: Pattern For Octahedral Diagram-Combinations Of Thermal Analysis Approaches

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Part I of the document consists of charts giving heating rates and temperatures for certain typical large spacecraft structural elements, suitable for developing estimates of whole structure temperatures and for screening structural concepts on the basis of their thermal and thermal-mechanical response.

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