Technical Memorandum No. 33-189

Temperature Control. A Case History of the Mariner Spacecraft

(Revision No. 1)

R. K. Pefley

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Prepared Under Contract No. NAS 7-100
National Aeronautics & Space Administration

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PREFACE

It seems appropriate that any flight vehicle so historically novel as the *Mariner* family of interplanetary probes should have its design evolution recorded for posterity. There are many facets to its design evolution but because of the author's biased inclination and training he has recorded here the evolution as it relates to the thermal behavior or temperature control of the spacecraft.

A second bias should also be recorded. The author has had the good fortune of spending three months of sabbatical leave from the University of Santa Clara with the *Mariner* Engineering Mechanics Group. While three months represent adequate time to set down these words, it does not necessarily provide time to carefully cross-reference all statements that are reported herein as fact. The author accepts sole responsibility for inaccuracies herein reported.

A third bias is the author's interest in the case method of teaching. For this reason, the thermal design evolution is being written in the form of a case history. The inclusion of student activity work, while unconventional from the standpoint of the regular Technical Memorandum, is consistent with the aim and purpose of this particular study.

The material herein presented could not possibly have been gleaned were it not for the willing helpfulness of the Temperature Control Group. The author is indebted to Don W. Lewis, group leader, Marsh Gram, Larry Dumas, Tom Thostesen, and Dave Miller for the discussions, answers, and ideas that resulted from the many questions asked. He is also indebted to J. N. Wilson, Section Chief, and D. W. Lewis for the time spent in reviewing this material.

ABSTRACT

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The logic underlying the thermal design of the *Mariner* spacecraft is reviewed. The temperature control philosophy is developed showing the influence of mission requirements, spacecraft configuration, and operational constraints on the spacecraft thermal design. Detailed engineering consideration is given to the trade-offs, alternatives, and ramifications of using louver systems, super insulation, and surface coatings.

It is concluded that a reasonable thermal design has been conceived and implemented for the *Mariner* spacecraft.

Author

I. DESIGN OBJECTIVE

It is desirable to launch a space vehicle within a 2-year interval that can perform deep space measurements, scan a near planet (Mars or Venus), and transmit all of this data to Earth. The launch vehicle has a modest payload capability.

The preceding statement establishes the principal purposes and constraints guiding the design evolution of the *Mariner* spacecraft to date. The quality of the final design must be weighed recognizing these as principal constraints; manpower, material, and test facility limitations are dictated principally by them.

A. Known Benchmarks at Design Inception[1]

The following parameters are known within reasonable bounds at project initiation.

1. Trajectory Data

The launch period, travel time, and trajectory are known as a result of parametric studies involving planet motion, payload, and launch vehicle capability. Figures 1a and 1b represent the results of this study. Table 1 gives the nominal times associated with the flight events.

2. Flight Events

Table 1. Nominal times for flight events

Event	Time t K. K.
Prior to launch	$t < 0$
Shroud ejection	6 min
Injection	$0 \leq t \leq 40$ min
Solar panel deployment	40 min
Sun acquisition	40 min $\leq t \leq$ 60 min
Space flight	60 min $\leq t \leq$ 5 days
Midcourse maneuver	5 days $\leq t \leq$ 5 days + 130 min
Space flight	5 days + 139 min $\leq t \leq$ 250 days
Planet encounter	250 days $\leq t \leq$ 250 days + 20 hr

3. Flight Orientation

Except for the initial Sun acquisition and midcourse maneuver, the spacecraft will maintain a fixed orientation with respect to the Sun.

[1]The values presented here are nominal values involving the planet Mars. The *Mariner* missions involve different planets, and hence designs for the different missions vary.

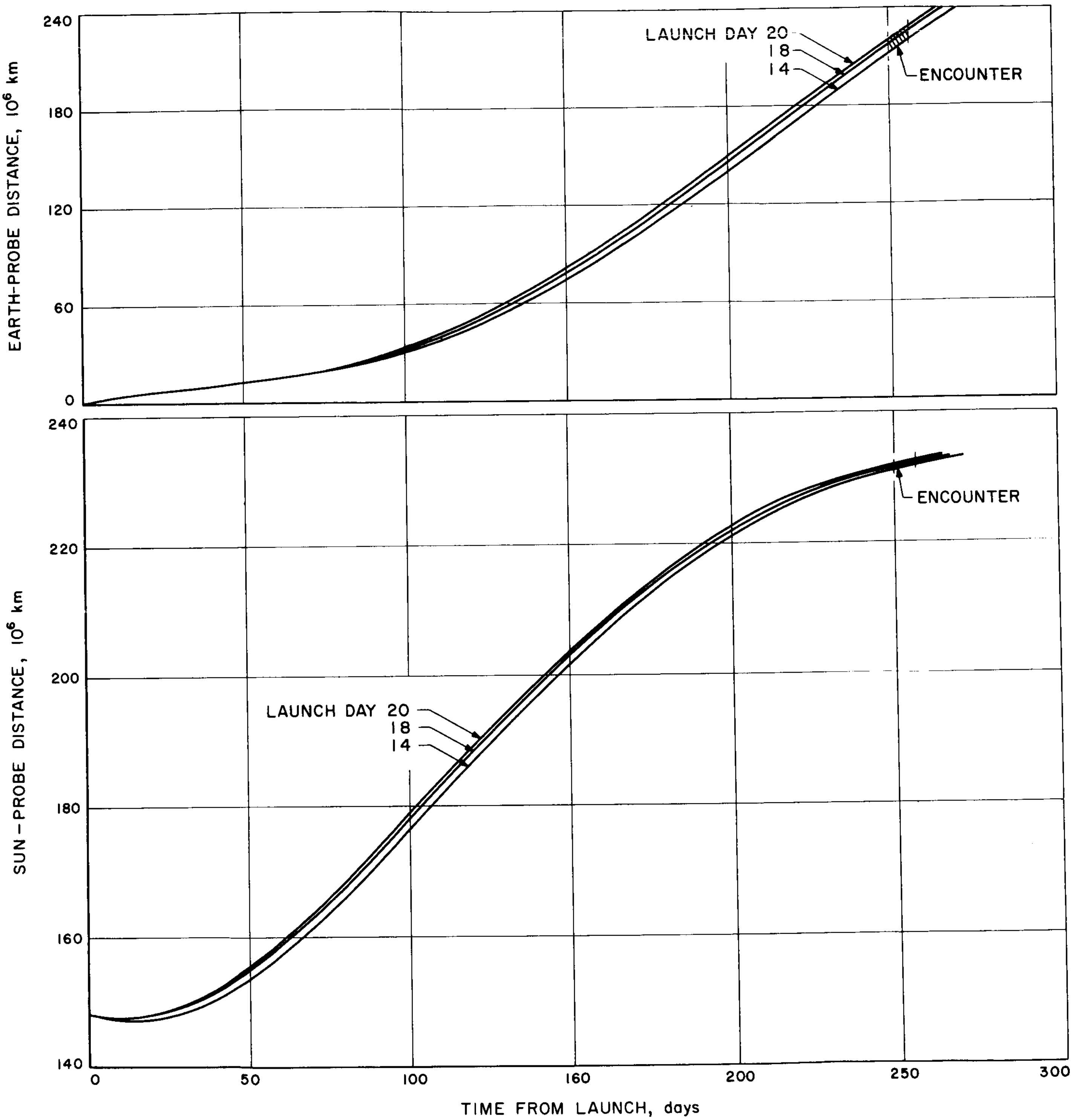

Fig. 1. Earth-probe distance vs time from launch

4. Vehicle Geometry

The spacecraft will have a volume of approximately 30 ft^3 and a maximum projected area of approximately 15 ft^2. These quantities can be scaled downward.

5. Component Location

A main component-cavity will house the bulk of the flight components, but some sensors, antennas, etc. need to be remotely mounted relative to the main component cavity. They will be of differing geometries.

6. Temperature Requirement

A nominal temperature of 70°F with a nominal excursion of ±30°F is desired for the bulk of the components. These values are arbitrarily chosen by the designers because design time is short and relatively little general knowledge is available about composite component behavior and component life expectancy if extreme deviation from this range is encountered.

7. External Thermal Environment

The external thermal environment can conveniently be broken into the following classifications: prior to launch, launch, shroud ejection and trajectory insertion, solar panel deployment and Sun acquisition, space flight, midcourse correction, and planet observation. In the first two environments, the vehicle is encapsulated in a shroud which protects it from aerothermodynamic loadings. Since a cooling sheath can be placed over the shroud and the vehicle air conditioned within the shroud prior to launch, no serious thermal loadings are to be expected during these modes. Sheath and shroud design will be modified as necessary to achieve this condition. As a result, the spacecraft is expected to be at essentially ambient temperature upon shroud ejection, and thermal design concern begins at this point.

8. Internal Thermal Environment

There are two types of internal environments. One is the *main multicomponent cavity* called a "bus" where an isothermal state is to be approached. The other is a component customized cavity which is required either because a component cannot thermally communicate—it is mounted external to the main cavity—or because it needs a particular temperature environment. Electrical heat sources will exist in some of these cavities. They will tend to be concentrated in particular areas and outputs will vary depending upon flight phase.

9. Experimental Facilities

A space simulator environment is available (Fig. 2) for major system checkout. Its principal deviations from true space environment are: (1) the solar simulation while matching the Sun in total intensity does not match it in spectral distribution, being high in the ultraviolet and infrared ranges (Fig. 3), and is not as well-collimated as the Sun, (2) the producible vacuum of 10^{-6}mm of Hg is not as good as the hard vacuum of space, and (3) the walls of the space chamber are not as absorbent of radiant energy as space.

One of the very first questions, and yet one that continues to plague spacecraft designers, is the actual solar constant θ_s and its spectral distribution. Although it appears known in Fig. 3, there is an uncertainty in its value of several percent. A 1% uncertainty in solar constant represents approximately a 1°F uncertainty in spacecraft temperature control and can represent a greater error if there is spectral mismatching also.

B. Student Activitiy for Thermal Design Phase 1

Study the problem statement and the initial mission specifications. Based on this study and other investigations you deem essential, do the following:

1. *Compose an agenda and be prepared to direct a meeting involving all cognizant engineers associated with the spacecraft design. The gist of the agenda should deal with the significance of the general thermal problem as well as specific problem areas as you see them; it should indicate the type of inputs you need to do your job; and it should indicate your design preference in terms of general shape, component location, and structural materials.*

2. *Separately from 1. and based on your preliminary study, specify the analyses you plan to make and prepare a list of projects you need to initiate based on anticipated analytic uncertainties. The project should be of two types: long range which will have inputs in subsequent spacecraft designs, and short range which will provide meaningful inputs in second- and third-generation interplanetary spacecraft.*

JPL 25-ft SPACE SIMULATOR

The JPL 25-ft space simulator tests spacecraft under the interplanetary conditions of extreme cold, high vacuum and intense solar radiation. The principal uses of the facility are to determine spacecraft equilibrium temperatures and the capability of spacecraft systems to perform satisfactorily in expected space environments.

The vacuum test chamber is a right circular cylinder 27 ft in diameter and 52 ft high. The top head of the chamber contains a 25-ft-diameter parabolic mirror. The bottom head, extending 5 ft below the floor level, contains numerous "feed through" or instrumentation ports for making electrical and mechanical connections to spacecraft and test equipment inside the vacuum chamber. The bottom of the chamber is also provided with a separately supported platform for mounting a vibration driver.

A cylindrical solar dome caps the vacuum chamber, increasing the over-all height of the simulator to 80 ft. Simulated solar radiation, originating in the solar dome, passes into the vacuum chamber through a 36-in. quartz lens mounted in a cylindrical well in the top of the chamber. The light reflects from a multifaceted reflector onto the cooled parabolic mirror which directs the radiation as an "off-axis" collimated beam into the test area.

The heat sink of deep space is simulated by liquid nitrogen-filled black panels (shrouds) which line the walls and bottom of the chamber.

OPERATIONAL SEQUENCE

A. Pump-Down

Phase 1	Phase 2	Phase 3
Wind tunnel compressors Chamber reaches 4mmHg in 5 min	Mechanical pumps Chamber reaches 5×10^{-2} mmHg in 25 min	Diffusion pumps Chamber reaches 5×10^{-6} mmHg in 2 hr

B. Shroud Cooling

Shroud temperature reduction with LN_2 starts when chamber pressure reaches 10^{-4}mmHg. Cooling to $-270°F$ takes 45 min

C. Solar Simulation

Lamps turned on as required during temperature reduction

D. Recompression and Warm-Up (3½ hr)

LN_2 drained from shrouds. Solar lamps turned off as required	Heated GN_2 pumped through shrouds	Pressure increased to 10^{-3}mmHg with dry GN_2	Diffusion and mechanical pumps disconnected from chamber	Chamber recompressed with filtered dry air

CROSS SECTION OF SPACE SIMULATOR

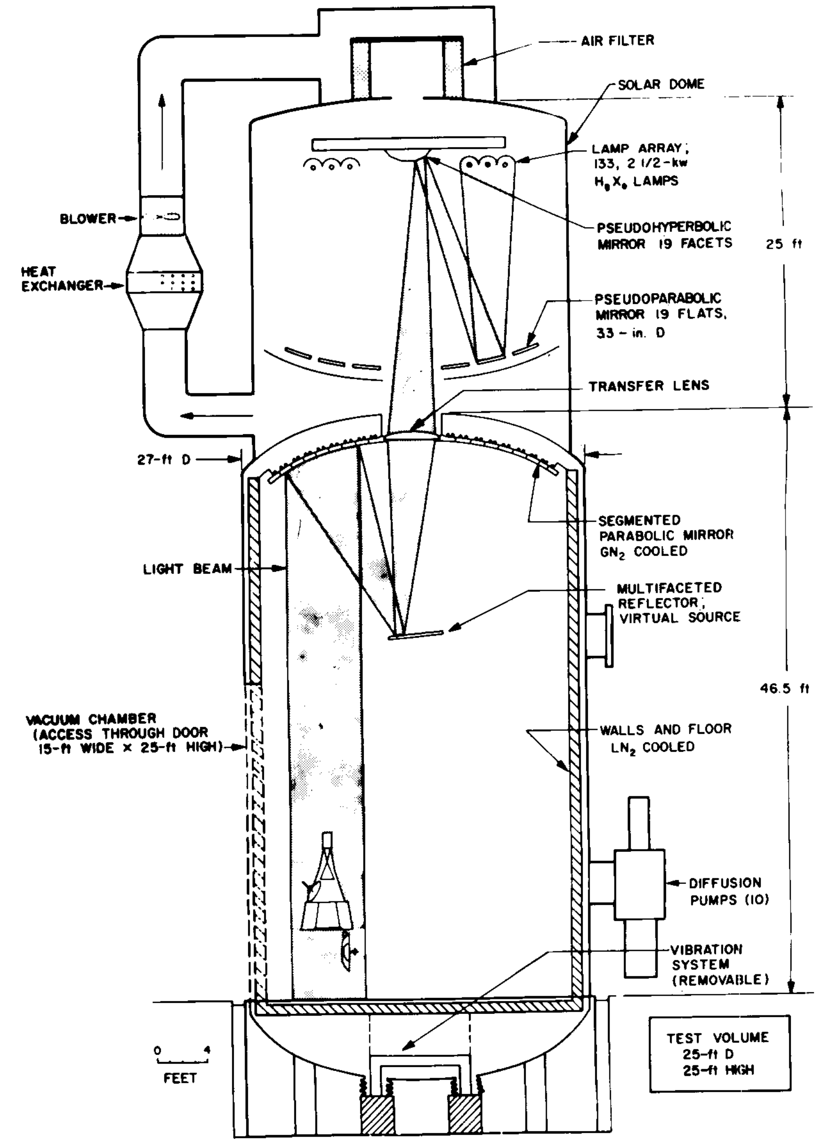

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Fig. 2. Space simulator

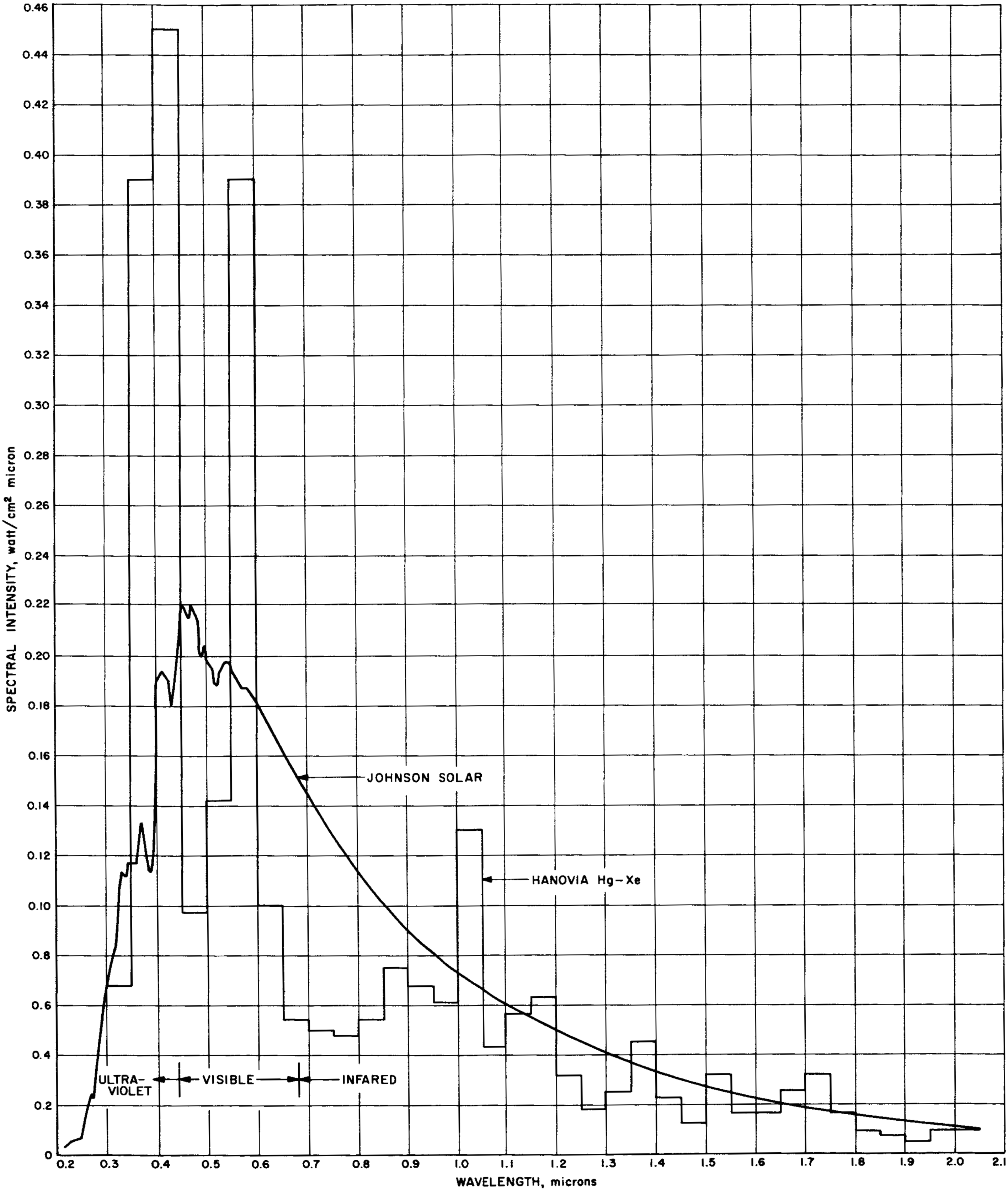

Fig. 3. Solar constant

II. THERMAL DESIGN PHASE 1

A. Temperature Control Problems

The thermal designers of the *Mariner* spacecraft started searching for general answers to temperature control by asking themselves the following questions:

1. What is the range of external environment to be encountered?
2. How do the internal heat loads compare with external radiation loads?
3. If thermal radiation modulation is required, what kinds of radiation valves are available for controlling radiant heat flow, which types are preferred and how much modulation is required of these valves based on (1) and (2) above?
4. What preferred spacecaft shapes are desirable?
5. What investigations need to be made?

Their thinking appears to have evolved the following logic.

Initially, by gross simplification, the spacecraft can be treated as an isothermal, steady state mass that is radiantly coupled between the Sun and space and has thermal energy dissipation within. We then have[2]

$$(\theta \alpha A) + P = \delta \epsilon A T^4 \qquad (1)$$

or upon rearrangement

$$T = \left[\left(\frac{(\alpha A)_s}{\epsilon A} \right) \left(\frac{\theta_s + \dfrac{P}{(\alpha A)_s}}{\delta} \right) \right]^{1/4}$$

This equation in conjunction with Fig. 4 gives us a clear introductory picture of the spacecraft thermal design problem.[3]

In Fig. 4, the upper branch of the abscissa in conjunction with the ordinate and topmost line shows the solar heat flux and black surface equilibrium temperature as functions of solar-planet distance. The lower branch of the abscissa in conjunction with the lower three parametric lines and the ordinate establish the required $\alpha A_s/\epsilon A$ for a given temperature and solar distance if the internal power generation is zero.

[2]The nomenclature is listed at the end of the report.

[3]Planet data on which Fig. 4 is based is from Ref. 2.

If we assume that P is zero or a constant relative to θ_s (a reasonable assumption since solar panel input is proportional to θ_s) and that $0 \leqq t \leqq 150°F$ is a permissible operating temperature range, the spacecraft can fly from Earth to Venus or Mars without changing its value of $\alpha A_s/\epsilon A$. Assume P is zero. For Venus $\alpha A_s/\epsilon A$ may have a value anywhere between 0.30 and 0.18 and its temperature change will be approximately $+70°F$. For Mars, it may have a value between 0.55 and 0.37 and its temperature swing will be approximately $-90°F$. If P/θ_s is assumed to be a constant the parametric temperature curves retain their slope and relative position and shift to the right. As a result a similar statement to that for $P = 0$ can be made but $\alpha A_s/\epsilon A$ will be smaller. For planets more remote than Venus or Mars, a fixed value of $\alpha A_s/\epsilon A$ is impossible if the temperature range of 0 to 150°F is to be maintained.

It is immediately seen that for bus temperatures to be held to 70°F with a $\pm 30°F$ temperature swing even without internal power modulation a flight change in $\alpha A_s/\epsilon A$ is mandatory even for near planets.

Since $\alpha A_s/\epsilon A$ must be modulated, should ϵA or αA_s be controlled or both? Since ϵA can be factored from the right hand side of Eq. (2), varying it will have the greatest effect on T and hence if equal percentage changes are possible, it should be modulated.

The design conforms to these conclusions.

A further examination of Eq. (2) raises the question, should αA_s be as large or as small as possible? Of course ϵA must be adjusted accordingly. Since P and θ_s both vary during flight we have by differentiating Eq. (2)

$$dT = \frac{1}{4T^3 \delta \epsilon A} \left[(\alpha A)_s (d\theta)_s + dP \right] \qquad (3)$$

It is obvious from inspection of this equation that if $d\theta_s$ and dP have the same sign, αA_s should be as small as possible. If they have opposite signs, $(\alpha A)_s (d\theta)_s$ should approximately equal dP to minimize dT.

Since the *Mariner* craft fly primarily dependent on solar power except for short-time variations, the thermal designers sought a low αA_s, for θ_s and P tend to vary in the same way under these circumstances.

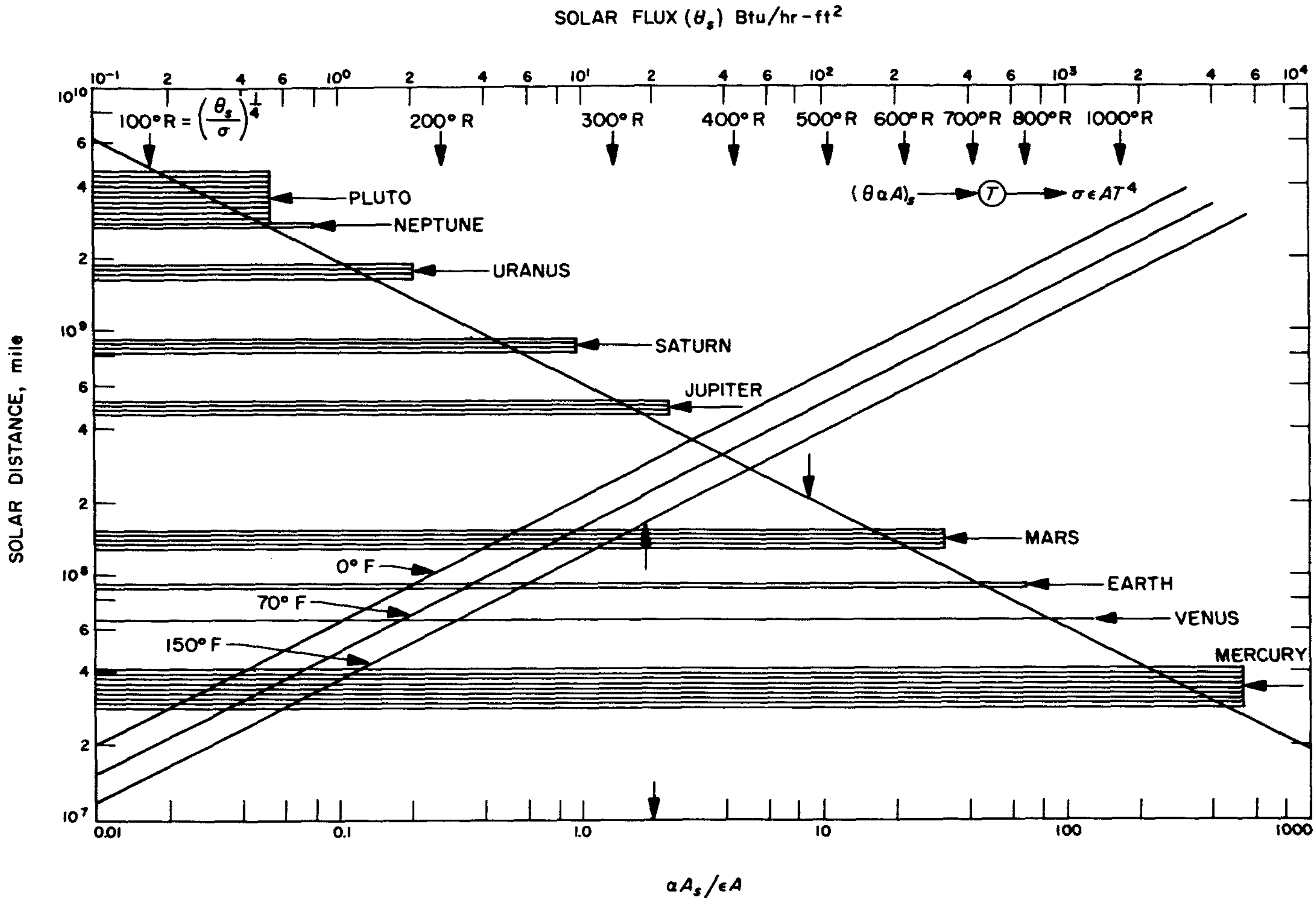

Fig. 4. Solar heat flux and black surface equilibrium temperature vs solar-planet distance

There are added advantages to small αA_s. The heat that must be conducted to the radiating surfaces from solar input is minimized, thereby reducing temperature differences required for the passage of heat. Also, low αA_s means low ϵA hence a high resistive coupling between the bus and its thermal surroundings. This results in slow response to external thermal environment changes.

There are disadvantages to this design. It is difficult to handle large momentary internal heat boosts such as that associated with the midcourse motor firing because of the high radiative resistance coupling (low ϵA) between the spacecraft and the surroundings. To cope with this type of surge requires a large thermal energy storage capability—not likely for a light weight spacecraft—a good capability for modulating ϵA, or the thermal pulse source must be isolated from the remainder of the craft.

Also, as the values of ϵA and αA_s are reduced, the uncertainty in T grows. Typically a measuring system for determining ϵ and α_s has a constant increment uncertainty in its measurement and as ϵ and α_s approach zero the percentage uncertainty in their values grows. Referring to Eq. (2), assume for demonstration purposes that as a first approximation $P = 0$, $\alpha_s = \epsilon$, and $dQ = -d\epsilon$; then the uncertainty in T is given by

$$\frac{\Delta T}{T} \simeq \frac{1}{2}\frac{\Delta\epsilon}{\epsilon} \tag{4}$$

Figure 5 is based on this equation and the assumption that $T = 530°R$ and ΔT is $\pm 30°F$. It graphically demonstrates the exactness with which ϵ must be known as it approaches zero in value.

Spacecraft external shape is of thermal importance. Again referring to Eq. (2), if A_s and/or A vary due to flight maneuver requirements the temperature balance will be disturbed. Further, the spacecraft is more sensitive to a percentage change in A than to a like percentage

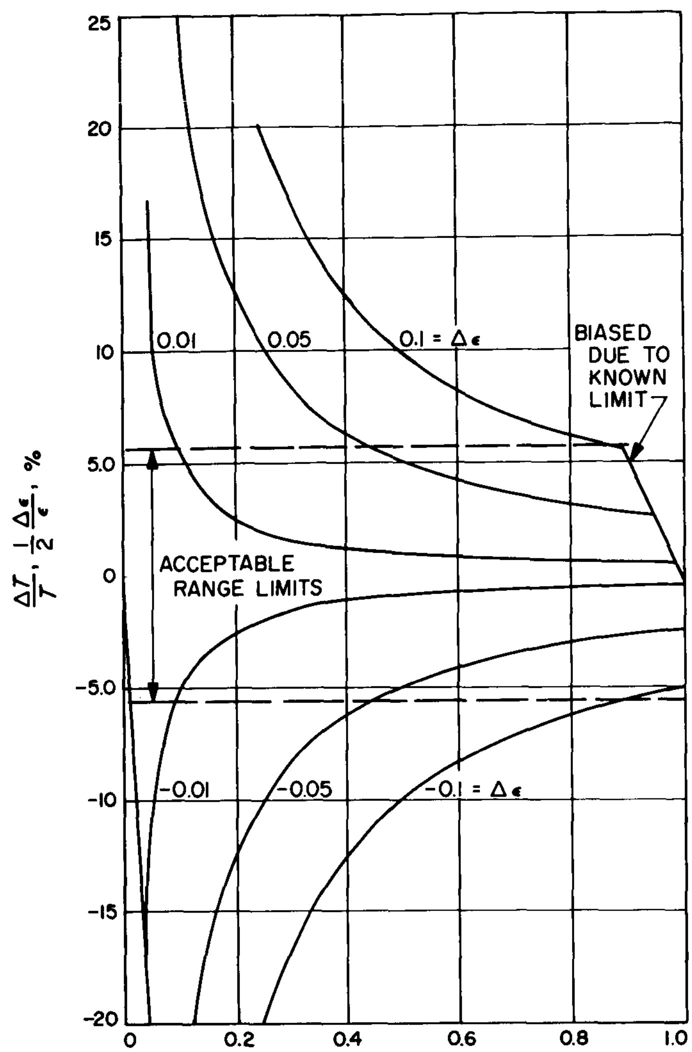

Fig. 5. Maximum spacecraft temperature uncertainty based solely on absorptivity, emissivity uncertainty

change in A_s. For zero internal power, A_s/A cannot shift more than 5¾% without requiring partial compensation by an opposite change in α_s/ε to stay within temperature limits. If $P/\alpha A_s >> \theta_s$, then εA must become independent of spacecraft orientation.

The Sun occupies such a small solid view angle as seen from the spacecraft that for a fixed geometry spacecraft the emission area may be assumed constant, and hence for fixed external geometry spacecraft, only ε can vary, and one of the following conditions must be true to minimize the required excursion of ε to control T as the spacecraft changes its helio-alignment.

1. $P/\alpha A_s >> \theta_s$ for all flight alignments.

2. αA_s is isotropic.

3. The spacecraft has a preferred helio-orientation and sufficient thermal capacitance which, in conjunction with its capability for changing ε, permits it to survive short deviations from its preferred helio-alignment.

The designers adopted possibility 3 for it was also harmonious from a solar panel efficiency and communications point of view.

The early activities of the thermal designers were oriented by the considerations just postulated. They proceeded to nail down the following in conjunction with cognizant engineers of other disciplines:

1. The internal power dissipation and its modulation as a function of flight path position and its preferred location within the vehicle.

2. The geometry of the vehicle.

3. The physical location and temperature *sensitivity* of sensors mounted external to the bus.

The *Mariner* spacecraft by this time has evolved into a nominal geometry, a specific example of which is portrayed by the spacecraft shown in Fig. 6a, 6b, and 6c (reproduced from Ref. 3).

A nominal internal electric power dissipation has also been estimated as a function of flight phase and geometric position in the main bus (Table 2). The temperature

Table 2. Power dissipation (watts)

Bus Bay No.	Cruise	Midcourse Maneuver	Encounter
1 Electronics	18	29	16
2 Propulsion	0	Motor Firing	0
3 Electronics	12	12	29
4 Electronics	11	11	11
5 Electronics	7.5	7.5	10
6 Electronics	40	40	40
7 Electronics	10	54	10
8 Electronics	30	50	30
Note: best estimate for design 150 (+30, −0) watts			

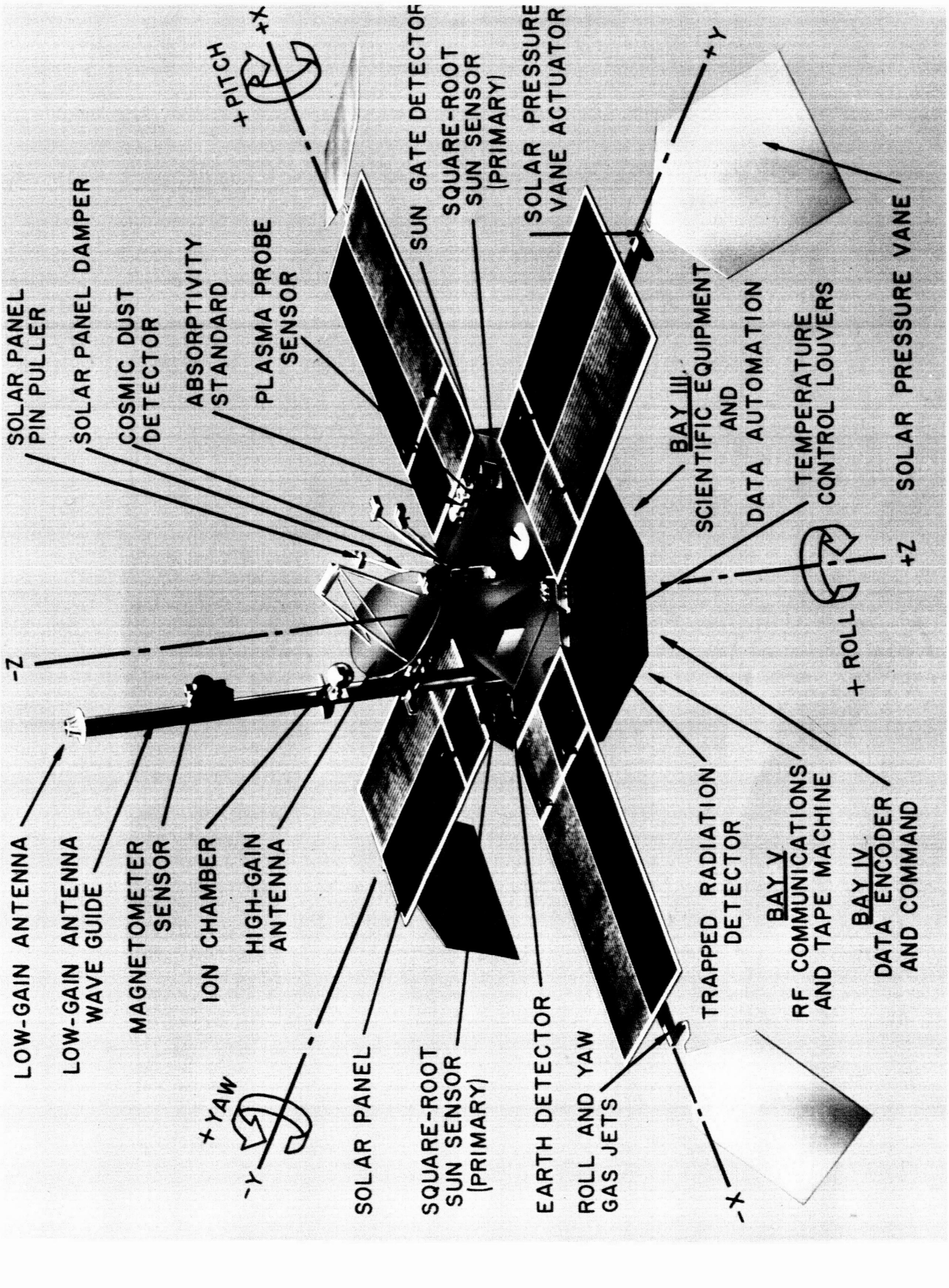

Fig. 6a. *Mariner* spacecraft

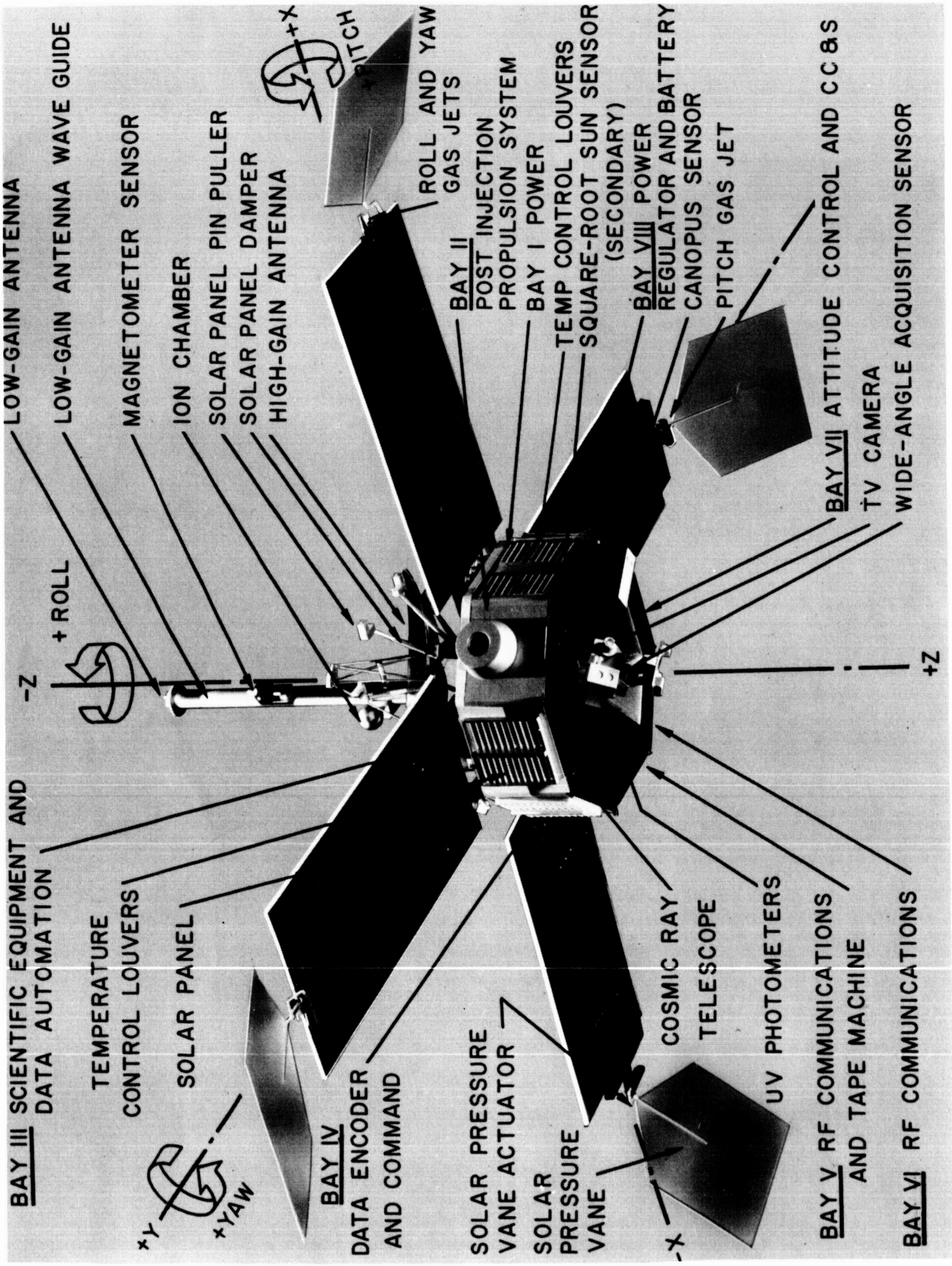

Fig. 6b. *Mariner spacecraft*

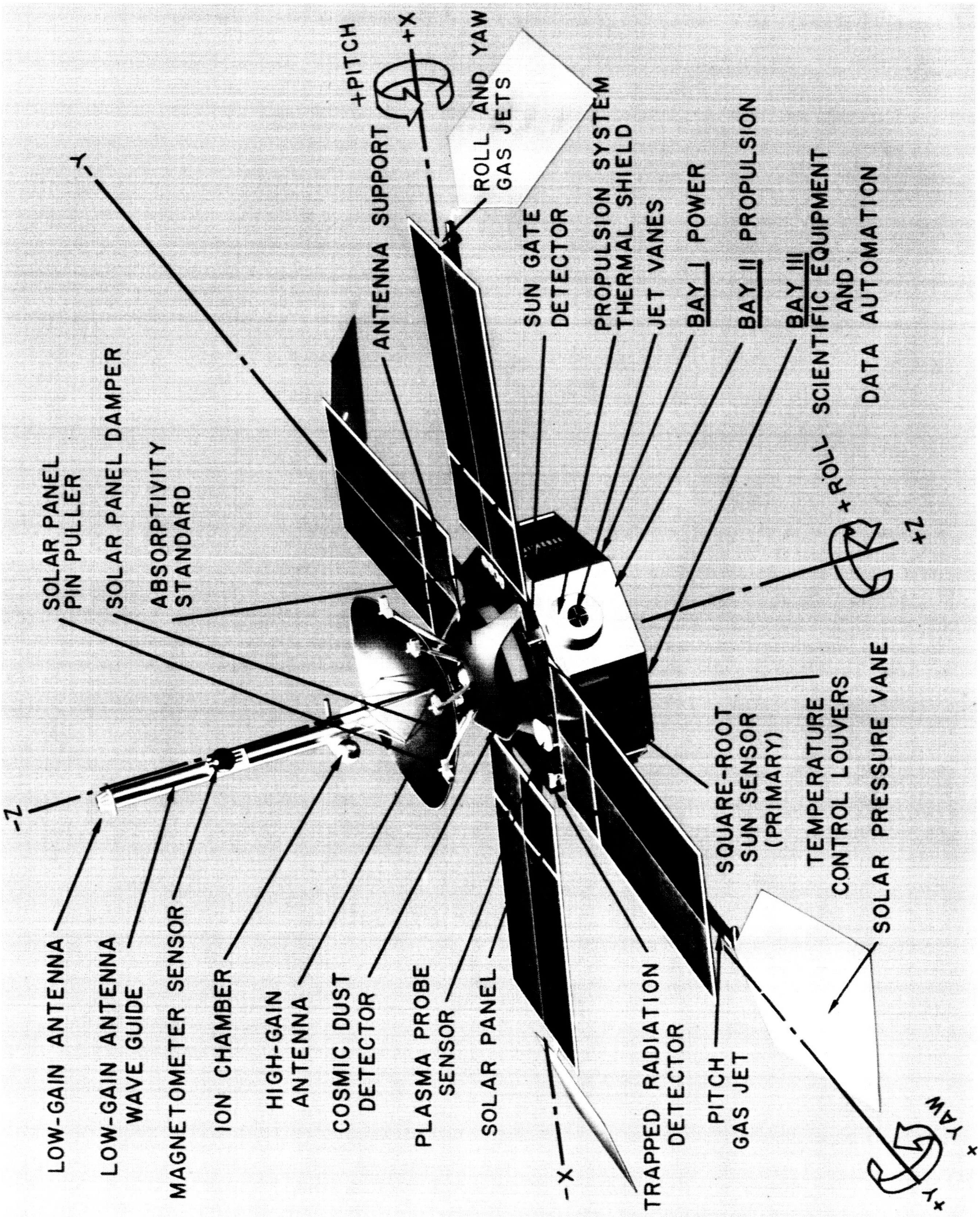

Fig. 6c. *Mariner spacecraft*

control goal for the components in the main bus cavity has previously been mentioned as 70 ±30°F.

The power requirement and permissible temperature range estimate for externally mounted sensors is shown in Table 3.

With this information available, the thermal designers proceeded to design in detail the thermal protection system for the main bus and external sensors. What follows is an attempt to trace the advances, the festers and eruptions, and the fixes that resulted in a flyable system.

Table 3. Power requirements and temperature range for externally mounted sensors

Sensor	Power, watts	Allowable temperature swing, °F			
		Operating		Nonoperating	
		Minimum	Maximum	Minimum	Maximum
Planet science					
Scan sensor and preamplifier	0.58	−40	+167	−50	+212
Scan actuator	2.90	−30	+200	−100	+200
TV camera head	1.1	−4	+104	−58	+212
UV photometers	2.50	−20	+212	−20	+212
Experimental assemblies					
Cosmic dust collector	0.23	−40	+212	−100	+298
Helium magnetometer	1.00	−40	+131	−40	+139
Trapped radiation detector	0.44	+14	+122	−22	+140
Ion chamber	0.43	−22	+158	−58	+212
Guidance					
Canopus tracker	3.0	−30	+100	−30	+100
Sun sensor		+30	+130	+30	+130

B. Student Activity for Thermal Design Phase 2

Based on the reported early activities of the Mariner *thermal designers, comment on the following: The logicalness of their approach to the problem, and the suitability of their proposed activities. Further, of the logic and activities with which you disagree, specify alternates and how you would proceed to deal with your proposed approach.*

III. THERMAL DESIGN PHASE 2

A. Main Bus Cavity Temperature Control

There are five general design problems to be dealt with in controlling the temperature of the main bus cavity. They are to achieve:

1. A low αA_s,
2. A matching value of εA,
3. A method of modulating εA,
4. A satisfactory coupling of components within the bus so that internal temperature variation is minimized, and
5. A thermal resistance-capacitance coupling of the bus such that the bus can handle short-time flight deviations from helio-orientation without excessive temperature change.

Achieving a good solar shield (low αA_s) with a minimum of weight is very important in current design circumstances, for the maximum possible heat flow through the bus via solar input is an order of magnitude greater than the internal heat flow and we desire it to be much less. For such solar shields, one looks to multilayer materials of very low thickness that are physically separated. The thermal circuit for such a multilayer shield is shown in Fig. 7.

Two points of major design interest for this type of shield are the heat flow per unit area q'' and the surface temperature T_1. The importance of small q'' has already been recognized, and low T_1 is important from a material-survival point of view.

For the circuit shown, these quantities are related to other known parameters by

$$T_1^4 = \frac{\theta_s \alpha_s n(2-\varepsilon) + \delta\varepsilon T_b^4}{\delta\varepsilon[1+n(2-\varepsilon)]} \tag{5}$$

$$q'' = \frac{\theta_s \alpha_s - \delta\varepsilon T_b^4}{1+n(2-\varepsilon)} \tag{6}$$

For $n >> 1$ (10 or more) these expressions become

$$T_1^4 \simeq \frac{\theta_s \alpha_s}{\delta\varepsilon} + \frac{T_b^4}{n(2-\varepsilon)} \tag{7}$$

$$q'' \simeq \frac{\theta_s \alpha_s - \delta\varepsilon T_b^4}{n(2-\varepsilon)} \tag{8}$$

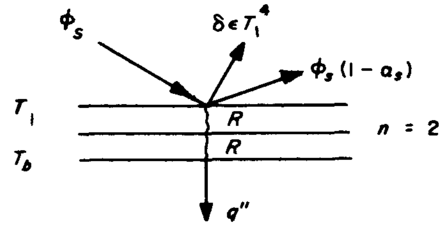

Fig. 7. Thermal circuit for multilayer shield

It is evident that reducing α_s has a favorable effect on reducing both T_1 and q''; its reduction can actually reverse the sign of q''. Increasing ε has a lesser effect on decreasing T_1 and q''. If q'' is near zero raising ε can reverse the sign of q''. Increasing n has an effect similar to decreasing α_s or increasing ε except that it cannot reverse the sign of q''. It will have a greater effect on q'' than upon T_1.

There are materials that are exceedingly attractive for minimizing α_s/ε and n, yet are reasonably tough structurally, have a fair temperature-survival range, and are very light in weight. They are metallized transparent plastic sheets. An example is aluminized mylar which has been used extensively in varying thicknesses in this application since the design inception of the *Mariner* program. It has an $\alpha_s \simeq 0.17$ and an $\varepsilon \simeq 0.35$ when the mylar side is out and for a thickness of ¼ mil.

Figure 8 shows the contrast in thermal resistance when the layers of aluminized mylar are either conductively or radiantly coupled. The thermal resistance contrast is enormous. Since the layers are separated by wrinkling, a very great design uncertainty exists, for, depending upon the degree of wrinkling and the method of holding the sheets together, the thermal resistance can lie anywhere between the two extremes shown. This design difficulty has been encountered but has not been satisfactorily resolved. One design avenue has been to use a large n (30 sheets) but this obviously is not a satisfactory substitute for assuring that the layers are indeed separated.

For the surface area of the bus that does not see the Sun except under maneuver circumstances, a low ε is required to match the low effective α_s achieved by the solar shield of multilayer mylar. This, too, can be achieved by the multilayer mylar. Placing the multilayer aluminum side out will minimize the effective ε. However, if several

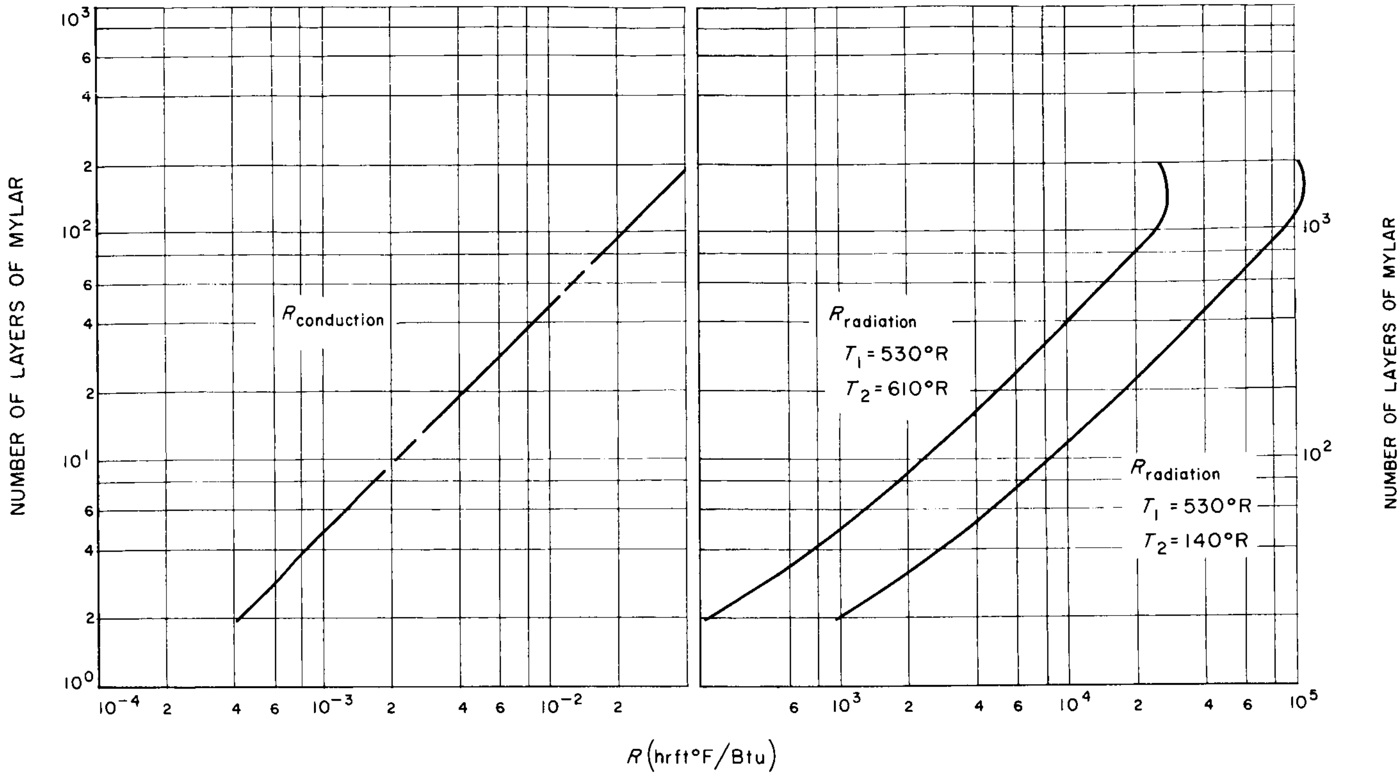

Fig. 8. Thermal contact resistance (conductive vs radiative ¼-mil aluminized mylar)

layers are used, and effectively separated, the preferred orientation is not significantly different, and will be swamped out by the contact resistance uncertainty. The aluminum side is out primarily to keep the mylar from deteriorating during the short time exposures to the Sun which are associated with space flight maneuvering.

Experience to date is that the sheet separation uncertainty and edge effects are so great that design confidence in this material appears to be diminishing rapidly.

Effective use of multilayer mylar to achieve low αA_s and low ϵA results in essentially complete thermal isolation of the bus except for the intentionally exposed areas designated as heat dump areas where the thermal valves are located and the required surface protrusions for sensor mounts, solar panels, antennas, etc. Obviously it is a design desire to keep the flow via these appendages at a known and minimum amount. It is also desirable to locate them near the thermal valves so that bus temperature variation associated with the uncertainty in their predicted heat input is minimized.

The design approaches this goal to a fair degree but the solar panel hinge points and superstructure anchorage points are not as close to the thermal valves as one might expect.

The need for thermal valves which modulate A or ϵ or both has previously been established as a necessity. Spacecraft thermal valving has been studied extensively and many types have been proposed. The requirements are that the valves be nearly weightless, have a high degree of dependability, and have a significant range of heat flow modulation approaching black body performance as an upper limit of heat dissipation. The heat flow range must be modulated over a relatively narrow temperature range.

The ideal valves from a weight and bulk point of view would be molecular. As a result of temperature change, they would modify their values of α and ϵ in a desired way. No known structural or surface coating material has the degree of molecular thermal valving desired, and this type of valve is not feasible at this stage of design.

Thermal valves utilizing natural circulation (produced by capillarity and surface tension) of fluids are attractive if large amounts of energy are to be transported and dumped. Their disadvantages are weight, the possible loss of fluid over a prolonged space flight and the uncertainty of their heat transfer behavior in a zero gravity field. Due to problems associated with thermal valves involving fluids and the fact that the heat transfer requirement is modest in the *Mariner* craft, this class of valve was not considered appropriate.

The remaining valves are those that mechanically change the thermal resistance between the spacecraft and its surroundings. There are two general ways of doing this. One is to use louvers or vanes of highly reflective material which block openings in the walls of the bus cavity. As temperatures rise, bimetal springs rotate these elements causing them to uncover the openings. A direct radiative coupling between the bus cavity contents and outerspace is thereby achieved. The emissivity of polished aluminum vanes is about 0.04. The cavity itself can be made to approach a black surface; and as a result an ideal valve of this type can achieve a thermal resistance ratio in the vicinity of 40:1 based on the area of the opening it blocks. The external surface shape of the main bus dictates the best configuration for the valve and its opening, and is based on the design objective: to maximize the resistance range for an available valve area with a minimum of weight addition.

Another way of changing the resistance is to use a series of parallel plates over the opening. This valve then is similar to the multilayer shield which was discussed previously. Consider the valve as consisting of two plates. By bringing the plates in good contact, direct conductance is achieved between the inner surface of the valve blocking the cavity opening and its outer surface. By separating the two plates, heat must flow by radiation between the two plates. If the interfaces are polished aluminum, the interior and exterior surfaces ideally black, and the conductive resistance negligible, it is found that the radiative resistance ratio for this type of valve is in the vicinity of 25 for black internal and external faces and polished aluminum interfaces. If it is assumed that the inner plate is the heat source, the range is increased to 40, but the approach to black body radiation will not be as good as that possible with the rotating vane type of valves. The parallel plate valve has some additional questionable features. It is not easy to modulate. It is more attractive as an "on-off" valve. Further, when it is dumping heat, it is difficult to say what the plate contact resistance will be in a hard vacuum. Originally there was also concern for cold welding of the polished surfaces in a hard vacuum when they are pressed together. This concern does not now appear justified.

All factors considered, it appears that the *Mariner* spacecraft designers made a valid decision when they adopted the louver design.

The rectangular louver boxes are self-contained sets of aluminum slats that are individually controlled by bimetal springs. They are mounted on the sides of the spacecraft and fit the sections of the octagonal bus. They open such that the louvers tend to serve as sun shields in normal flight attitude. An important aspect of louver design is the bimetal actuation spring. This must be effectively coupled thermally to the innerzone of the bus and isolated otherwise. Considerable effort has been expended to achieve this goal.

It is fair to ask why some of the louvers are not in the base pointing away from the Sun since this is their most effective location. An infrared planet scanner was originally to be mounted on the science platform in this area. It needed to be very cold and would be disturbed by radiation from the louver area. The scanner was subsequently removed but the louvers by this time had been committed to side mounting.

Internally the main bus cavity has localized heat sources the larger of which need to be located in close proximity to the louvers or have good conductive and radiant coupling with a louver area. The polygon design was chosen in part because it allowed mounting of components on bedplates which matched the lateral surface areas of the polygon. These plates can be directly exposed to the outside or immediately adjacent to the louvers thus minimizing heat flow resistance. Each pie segment of the main bus cavity is designed to be thermally self-reliant within the temperature range desired for the main bus cavity. Then the internal thermal coupling among the pie segments, which is maximized by painting the interior black, serves as a safety valve in that it tends to damp out unplanned temperature discrepancies.

There are several complexities associated with "inside cavity heat flow analysis." Heat flows by a mixture of multiple reflections and re-radiations from a myriad of geometries and surface emissivities and reflectances. In series and in parallel with this radiant transfer is conduction through a variety of materials and geometrical shapes and across bolted, riveted, and spot-welded joints.

How does one analyze such a complex heat flow circuit? This is a question that is repeatedly asked by the spacecraft temperature control group. They have generated approximate thermal circuit models that can be treated by computer programs. The comparison of the results so obtained with the temperatures obtained from spacecraft in the solar simulator has produced discrepancies that are great enough that confidence in the analytical modeling has never been fully established. At the same time, the temperature performance of the actual spacecraft in spaceflight has been sufficiently far from the solar simulation performance that full confidence in temperature verification by testing has not been established. It has improved as solar simulation has become more authentic.

Analysis currently is limited to identifying the potential temperature trouble spots. These are invariably hot spots. Examination is made to assure that maximum conduction coupling has been achieved. Then adjustable radiation coupling links are sized such that temperatures are brought into the desired range with a margin for further adjustment remaining in the coupling link. Adjustable coupling links are radiation resistances which can be changed. Area cutouts, louvers, surface finish, and coatings are examples.

The most serious stumbling block to successful conductive heat flow analysis in addition to geometry is joint conductance. Joint conductance is not well understood in our earth environment. This ignorance is compounded by the lack of knowledge of the effects of severe launch vibration and joint outgassing in the hard vacuum of space. Spacecraft temperature control modeling studies have been initiated in part to try and understand some of the complexities of joint conductance. (See Ref. 4.)

Radiantly, the nonisothermal aspect of surfaces and the multisurface environment are the two major impediments to analysis.

Since it is reported that the *Mariner* spacecraft components are predominately coupled within the bus by radiation, it is of interest to question whether this is indeed the best thermal coupling made to accentuate. For heat flow through a series of n radiating black plates of negligible conductive resistance, we have for the radiant resistance

$$R_r = \frac{n}{4A\sigma \overline{T}^3} \tag{9}$$

For the spacecraft, a nominal value of $\overline{T}$ is 530°R and if we consider the least radiation resistance possible $n = 1$, then

$$R_r = \frac{0.96}{A}\left(\frac{\text{hr ft}^2\ ^\circ\text{F}}{\text{Btu}}\right) \tag{10}$$

Conductive resistance is given by

$$R_c = \frac{L}{kA'} \tag{11}$$

Assuming the spacecraft structural material is magnesium ($k = 100$ Btu/hr ft °F), we have

$$R_c = \frac{0.01L}{A'}\left(\frac{\text{hr ft}^2\ ^\circ\text{F}}{\text{Btu}}\right) \tag{12}$$

Upon comparing this resistance with the radiant resistance it is evident that on an equal area basis it would take nearly 100 ft of magnesium to offer the same resistance to heat flow as two black radiating plates.

The spacecraft mass is in the vicinity of 500 lb_m of which 20% may reasonably be assumed to be structural materials in the bus. This is approximately 1 cu ft of magnesium. If we imagine that the bus structure is made up of magnesium plates $1.5 \times 0.02 \times 2$ ft, we would have about 15 pieces. Assuming the nominal power of 150 watts dissipated by the bus is conducted through the long axis of these 15 pieces in parallel, we encounter about a 20°F temperature difference. This ΔT is only one half the maximum ΔT actually measured in the spacecraft.

Consider a cube of material of dimension L. Let there be n cubic voids when counted along any of the axes of the cube, (each of dimension xL.) Assume one-dimensional heat flow in the cube. What is the ratio of conductive to radiative resistance through the cube and the total resistance as functions of density and number of voids?

The mean conductive area is

$$L^2 - (nxL)^2 = L^2(1 - n^2x^2)$$

As a first approximation the conducting length is L.

Then the conductive resistance is

$$R_c = \frac{L}{kL^2(1 - n^2x^2)} = \frac{1}{kL(1 - n^2x^2)}$$

The mean radiative area is $(nxL)^2$

The number of pairs of radiating surfaces in series is n.

For black body radiation the radiant resistance is

$$R_r = \frac{n}{8(nxL)^2\overline{T}^3} = \frac{1}{8nx^2L^2\overline{T}^3}$$

Then

$$\frac{R_c}{R_r} = \frac{\delta n x^2 L^2 \overline{T}^3}{kL(1 - n^2x^2)} = \frac{\delta n x^2 L \overline{T}^3}{k(1 - n^2x^2)} \quad (13)$$

and

$$R_t = \frac{R_c R_r}{R_c + R_r} = \frac{1}{kL(1 - n^2x^2) + \delta n x^2 L^2 \overline{T}^3} \quad (14)$$

The specific gravity in terms of pure metal density is

$$sg = 1 - n^3x^3$$

For the spacecraft, the main cavity and its components weigh about 400 lb_m and occupy 30 ft^3. Choosing a nominal value of $\rho = 100\ lb_m/ft^3$, the sg is about 0.10. It follows that $nx \simeq 0.96$. Substituting $\overline{T} = 530°R$, $L = 2$ ft, and $k = 100$ Btu/hr ft °F into Eq. (13) we have

$$\frac{R_c}{R_r} = \frac{0.173 \times 2 \times 150 \times (0.96)^2}{100 \times 100\,[1 - (0.96)^2]\,n} = \frac{0.06}{n}$$

This indicates that the mean conductivity of all spacecraft components including magnesium could be as low as 1/20 of that of magnesium before R_c/R_r would approach unity even when n is set equal to one. Since n is certainly greater than one, it appears that the conductive path should offer less resistance to heat flow than radiant coupling.

Based on these cursory observations, one is inclined to believe that dense packaging and good conductive coupling would be of prime importance in the spacecraft bus design. This would include utilization of all support structures in an integrated thermal and structural design sense. Such does not appear to be the case in current *Mariner* spacecraft design.

The transient thermal behavior of the spacecraft is complex. Some of the larger masses that are deeply buried within the bus require 15 minutes before showing any significant response to a shift in external thermal environment. Small masses near the uninsulated walls of the spacecraft will show evidence of this shift in less than a minute. By treating the spacecraft as a lumped mass system consisting of the main bus and solar panels, the thermal response to specified environment shifts can be calculated. Transient tests in the solar simulator are used to verify that individual component temperatures do not exceed the estimates based on the coarse models used for predictions.

Diagnostic testing to observe shifts in thermal joint resistance due to vibrations and outgassing and the relative magnitudes of component radiation and conduction resistance could be carried out using vibration and solar simulation facilities and the internal heat modulation capability built into the temperature control model of the spacecraft.

Some such tests have been carried out but the joint conductance shift due to vibration and outgassing and the relative magnitudes of conduction and radiation heat transfer within the main bus cavity are not adequately established. It is believed that the conductive heat transfer plays a minor role in internal heat exchange.

It appears that more intense synthesizing of analytical, test, and flight performance with results that can be obtained from specialized tests aimed at basic understanding is mandatory if "hunt and peck" temperature control is to be reduced.

B. Student Activity for Thermal Design Phase 3

1. *Prepare a written critique of the efforts of the thermal designers as reported in Thermal Design Phase 2 (Sec. III). Be prepared to summarize your impressions orally and suggest alternate ways of approaching the temperature control problems.*

2. *There are many sensors that must be mounted external to the main bus. Fig. 6 verifies this circumstance and Table 3 summarizes their internal thermal output and their allowed temperature excursion. Would you anticipate any temperature control difficulties with these components? If so, for what reasons? What avenues of temperature control would you tentatively consider for these small modules? Rank in their order of anticipated usefulness and explain why you so ranked them.*

IV. THERMAL DESIGN PHASE 3

A. Externally Mounted Sensors

Because the *Mariner* program is aimed at deep space and planet exploration, it is to be expected that many sensors will be mounted external to the main bus cavity. Figure 6 verifies this circumstance. Table 3 summarizes the internal thermal output and the allowed temperature range for some of these externally mounted components. It is apparent from the allowed temperature range that if these components could all be effectively slaved thermally to the main bus, temperature bounds should not be exceeded. That they cannot all be is evident from their location in Fig. 6.

For first considerations the sensors can be considered to be isolated isothermal bodies with internal heat dissipation. As such, our prior considerations regarding temperature control of the main bus are generally applicable. However, there are some significant differences. The shape and composition of a portion of the sensor's surface as well as its physical location on the spacecraft usually are dictated by its sensing requirements. The sensor may be small and yet have sizeable internal power generation. Small size also means fast response to a changing thermal environment.

If we consider a sensor as a small sphere that is radiantly coupled between the sun, space, and the spacecraft itself, we have for its steady state temperature

$$T^4 = \frac{(F\theta_s) + \dfrac{P}{A} + (\delta F T^4)_{sc}}{\delta\,(F + F_{sc})} \qquad (15)$$

As pointed out previously, if the sensor is dominantly coupled to the spacecraft, T will modulate with T_{sc}. This coupling obviously would not be sought if it were with the main bus thermal shields. When coupling with the main bus is not feasible, the temperature excursion experienced by using nonmodulating temperature control is satisfactory for near-planet flights, fixed helio-orientation, and constant or insignificant values of P.

Internal power may not be insignificant in these sensors. It may easily exceed solar load. For example, the radiation load on a one-inch-diameter sphere with 5 watts dissipated internally is approximately 5 times that of the maximum solar load for near planet flight. The internal power dissipated by a black sphere at 70°F that sees space only is

$$P = 0.89d^2 \qquad (16)$$

where P is in watts and d is in inches. It is obvious from this that a one-inch-diameter sphere can handle about one watt and maintain a temperature of 70°F, provided it is isolated from the sun.

A criticality of sensor temperature control design becomes evident. If it must be situated where the sun sees it and has internal power dissipation, it needs to radiate very effectively from its entire surface area and at the same time have very low absorption of solar radiation to keep its temperature within bounds.

This requires a surface which has a low absorption in the wavelength band of most intense solar radiation and high emission in the infrared region. Such surfaces are commonly referred to as low α/ϵ surfaces.

Figure 9a, 9b, and 9c presents the normal surface reflectivity for some paints and metals. It is evident that of those presented, there is a considerable possible range of α/ϵ. Recognition of this readily available range of α/ϵ has proven to be the temperature control group's dictum—"to control temperature paint it or plate it and shape it." Indeed for the solar and internal loads currently being experienced by the sensors, extensive use of paints and platings is being used. Over a half dozen different paints are applied to the various sensors and appendages. Primarily this is done to achieve temperature control by altering the value of α/ϵ. In one or two instances, special paints are used for reasons relating to sensor behavior other than temperature control.

Paint, while achieving a good flexibility in α/ϵ, has some inherent unsatisfactory features. Because it is mechanically bonded, it may have pockets of fluids beneath it. The pressure exerted by the fluids can cause chipping as the spacecraft moves into the hard vacuum of space. If entrapped fluid causes blisters or flaking there is serious alteration in the surface heat transfer resistance. Scattered chips can cause electrical shorts, confuse star-seeking guidance sensors and alter sensor behavior if they deposit on sensor windows. There are no known nondestructive methods of examining a paint coat for minuscule impurity deposits between it and its base. Meticulous care in painting is the primary assurance of good bonding.

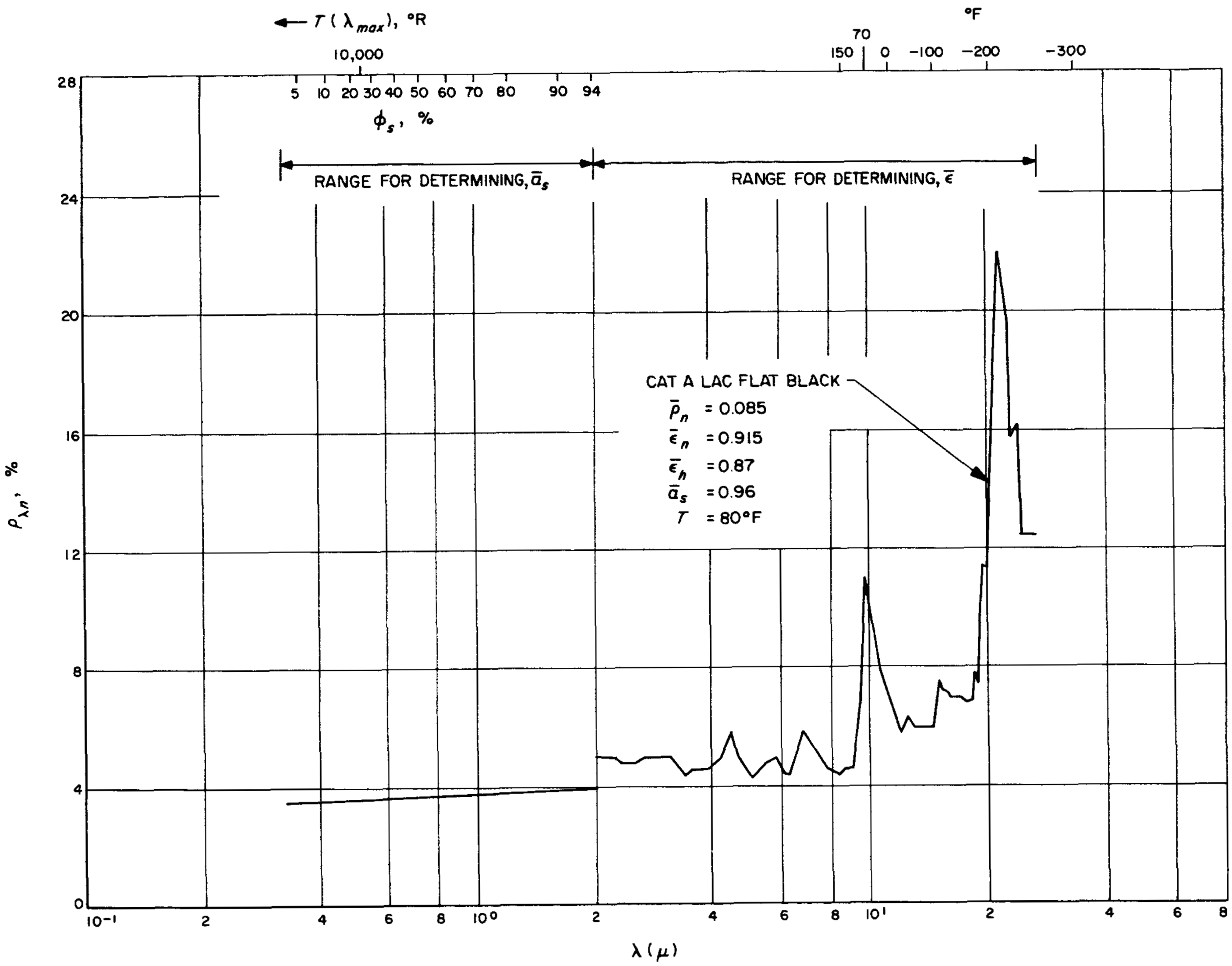

Fig. 9b. Normal surface reflectivity for some paints and metals

A significant uncertainty in evaluating the external thermal environment is solar reflection particularly from specular reflection associated with low α_s surfaces. For the complex geometry of the spacecraft it is difficult to analytically deal with specular reflectance and because of the difference between the collimation angle of solar rays and solar simulator rays, experimental results are not always definitive. Consequently, the thermal designers have adopted the approach of using diffuse surfaces as a precautionary measure. The thermal shield on the upper or solar side of the bus is a good example. The multilayer mylar, aluminum side out, is covered with black dacron. Although the insulation surface runs hotter because of the black dacron, it does assure that there will be no expected high reflective loadings on one of the sensors that could arise if the aluminum were exposed.

The solar cell panels have offered no particular difficulty to the thermal designers. Although their effectiveness diminishes with a rise in temperature, they can survive the temperature swing encountered by flights to the nearest planets. They are coated black on their back or nonsolar side to suppress their temperature as much as possible.

An interesting anomaly is the positioning of the solar panels. In the *Mariner* flight to Venus, which is a progressively warmer flight, the panels were hinged from the lower side of the bus so that the bus was exposed to the hot side including solar reflectance of the solar panels.

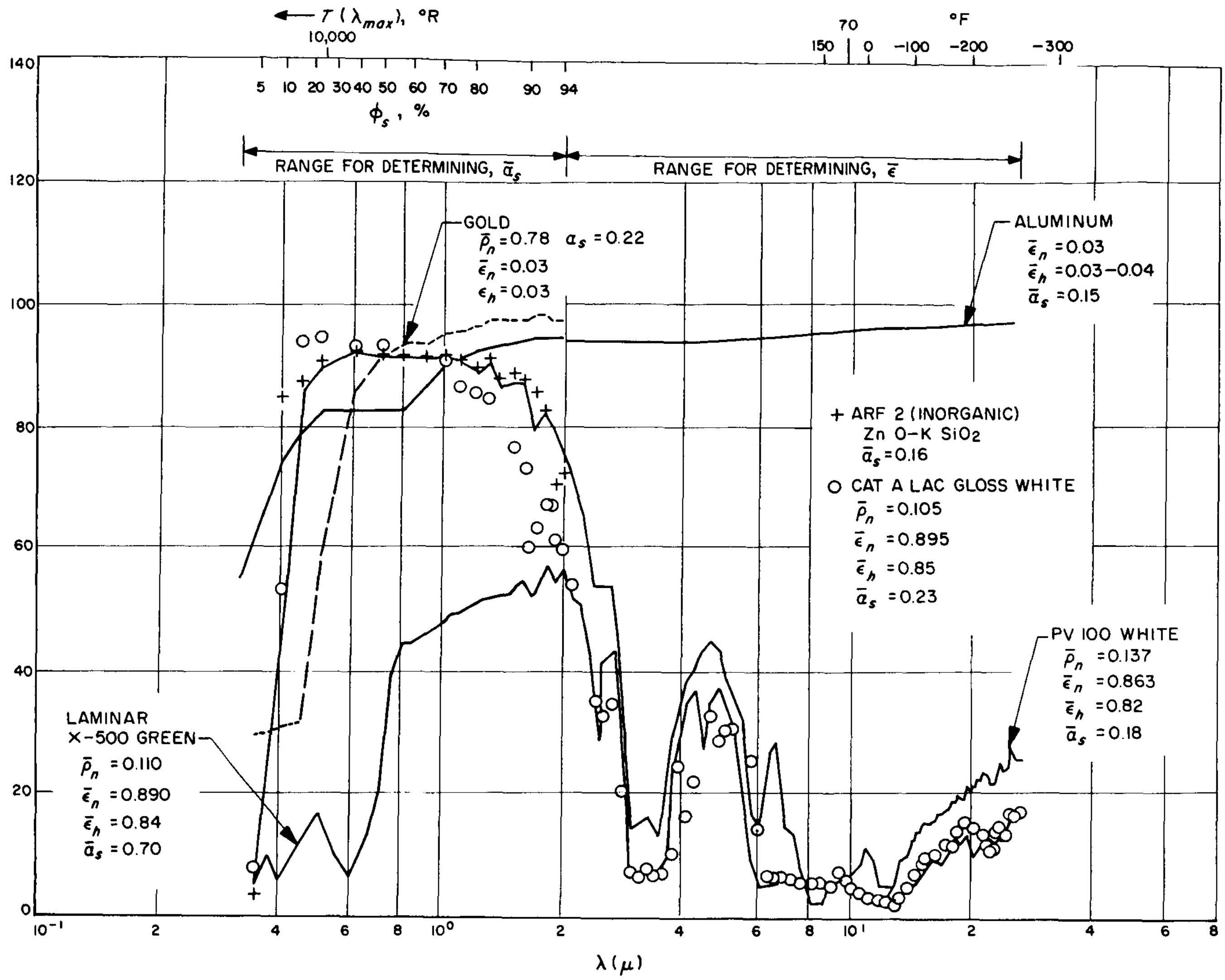

Fig. 9a. Normal surface reflectivity for some paints and metals

The value of α/ϵ is dependent upon the thickness of paint. As a matter of fact, one aspect of paint thickness that will bear further investigation is shown in Fig. 10.[4] Note that as the thickness of the paint diminishes the value of α_s increases. The substrate metal on which the paint is deposited has an α_s that is below all of these values. This requires that as the paint thickness approaches zero the value of α_s must pass through a maximum. The magnitude and location of this maximum and the justification of its existence are not known to the writer.

Ultraviolet degradation of painted surfaces is also a problem. As a result, paints are selected not only for their value of α/ϵ but for their surface adherence capability and resistance to ultraviolet degradation as well.

Difficulties are encountered when the internal power of sensors or the attitude of the spacecraft varies. An efficient way of modulating the value of ϵ for the sensor is then desirable. There is no evidence that an efficient small scale set of louvers or equivalent that will fit a variety of geometric shapes has been developed. Bimetal snap action disks that can be bonded to the surfaces of a sensor might be one method of achieving this goal.

With a good capability for modulating ϵ of the sensors, the reliability upon paints and the proliferation of types of paint may be advantageously curtailed.

[4]The data is taken from Fig. 34 of Ref. 5.

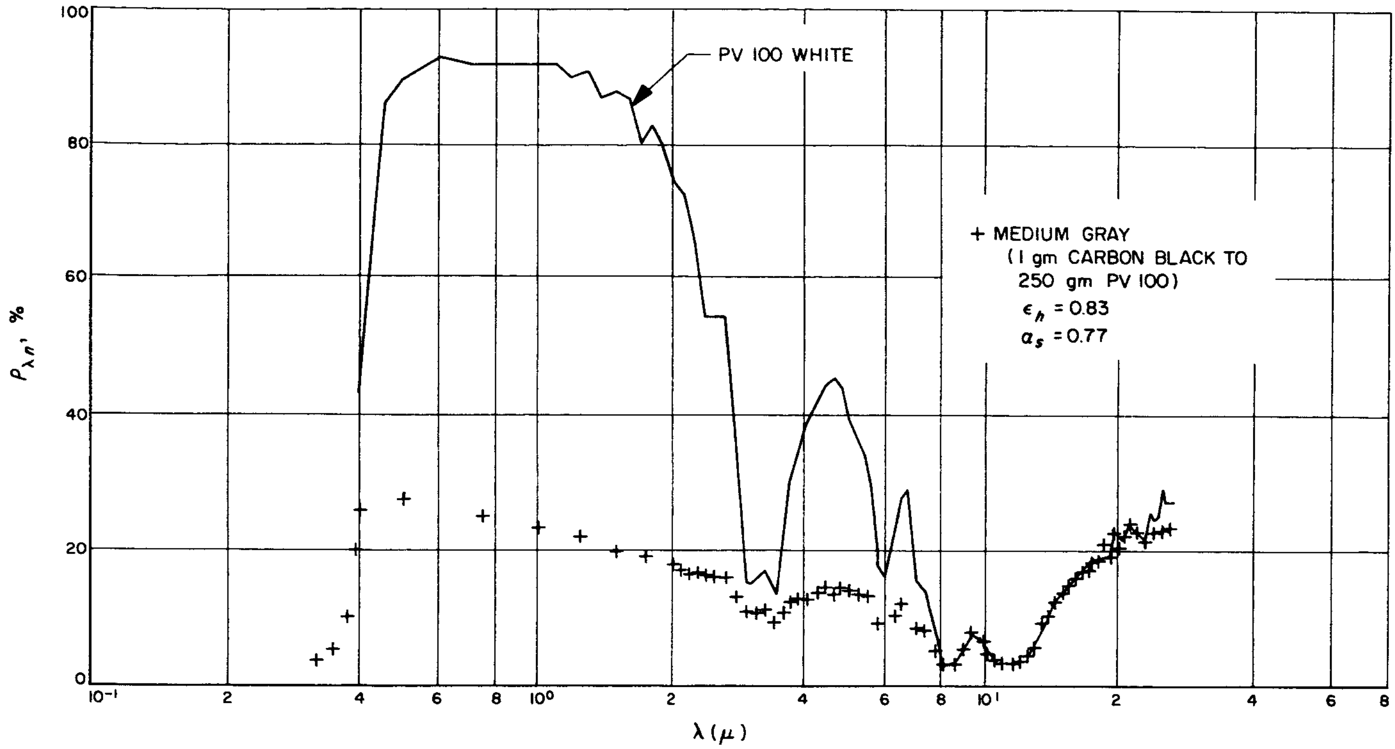

Fig. 9c. Normal surface reflectivity for some paints and metals

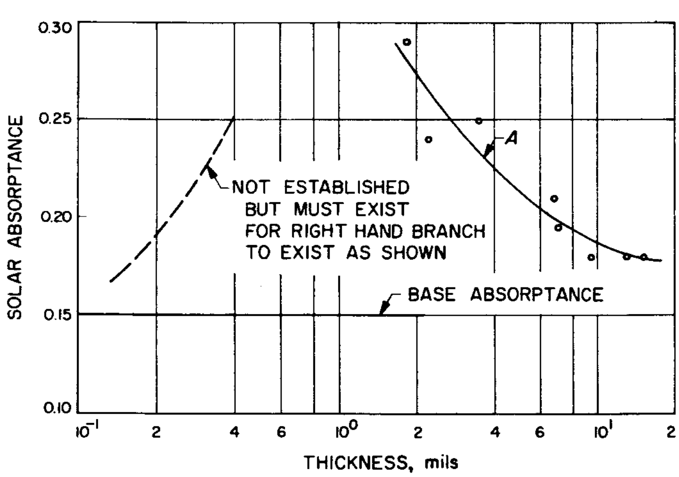

Fig. 10. Solar absorptance vs thickness (white paint)

For the *Mariner* flight to Mars which is a progressively colder flight, the panels are hinged to the top of the bus and the bus sees the lower side of the panels. The panels appear to be incorrectly mounted from this temperature control point of view. Their position has been chosen for reasons other than temperature control on the Mars flight. The temperature control group accepted the position for it avoids the uncertainty associated with specular solar reflectance from the panels to the bus.

Currently, solar cell efficiency is in the vicinity of 5%. If an order of magnitude improvement in solar cell efficiency could be achieved, they would become excellent heat sinks and their surface temperatures could be reduced in the process. There is no evidence of such an achievement in the offing.

B. Student Activity for Thermal Design Phase 4

Based on your understanding of the Mariner *temperature control design, appraise the* Mariner *spacecraft from a temperature control point of view. In addition, anticipate future temperature control problems that may be encountered as the unmanned planetary spacecraft program unfolds. Suggest projects that might be initiated which may have a long range pay-off in spacecraft thermal design evolution.*

V. THERMAL DESIGN PHASE 4

A. Summary

That the *Mariner* spacecraft temperature control design group has achieved a reasonable degree of success is demonstrated by the successful *Mariner R* flight to Venus. That they do not have full cognizance of their problems is attested to by the fact that the *Mariner R* ran at flight temperatures considerably above predicted values and above upper design limits during the latter part of the mission (see Ref. 6). The proof of improved cognizance of their problems will become evident by better correlation between predicted and flight temperatures in future spacecraft.

B. Areas for Further Study

It is the writer's impression that the temperature control group feels that the following problem areas need to be carefully explored if temperature control is to continue successfully as internal thermal loadings become more intense and missions which include planet landings become possible.

1. The solar constant (a misnomer since solar intensity varies with time) must become better known in total intensity and spectral distribution.
2. A dependable analytical method for predicting temperatures in a complex geometric structure experiencing radiant and conductive heat transfer must be found.
3. A part-joining technique must be developed for both static and dynamic joints that will maintain a constant and high heat transfer capability after being subjected to mechanical vibrations, thermal loadings, and the hard vacuum of space.
4. A lightweight, highly dependable thermal value is desirable that will accommodate small parts such as sensors and that has a greater modulation and absolute temperature range than the louvers currently being used.
5. A higher degree of reliability must be established in the insulating quality of the multilayer-type thermal shield.
6. There is a need for coatings that have the same heat transfer dependability as the base materials that they coat. Alternatively, a way of varying the base material's radiation properties is desired.
7. A solar simulator is needed that more closely approximates the sun in collimation and spectral distribution.
8. A lightweight thermal bus bar should be developed for transferring heat from one zone of the spacecraft to another.

As the *Mariner* and companion spacecraft designs continue to evolve, one expects to look back at these early attempts and based on historic parallelism see them as primitive vehicles. Yet, upon examining the intelligent and elaborate iterative processes that have transpired to achieve these spacecraft, one finds this difficult to believe. Obviously, the manifold aspects of the design evolution of spacecraft is one of the more fascinating aspects of space exploration.

NOMENCLATURE

A radiating surface area or conducting area

F combined emissivity and view factor

k thermal conductivity

L length

n number of pairs of radiating surfaces in series

P heat generation rate

q'' heat flow per unit area

R thermal resistance

T, t Rankine, Fahrenheit temperature

x dimensionless ratio

α absorptivity

Δ finite difference

ε emissivity

θ radiation intensity per unit area normal to ray

δ Stefan-Boltzmann constant

Subscripts

$_c$ conduction

$_r$ radiation

$_s$ solar

$_{sc}$ spacecraft

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