JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

750-39

EXTREME TEMPERATURE REQUIREMENTS FOR SPACECRAFT ELECTRONICS PARTS

FY'71 REPORT

THE ORBITER CASE

September 15, 1970

By

Jerry Swanson

APPROVED: T. C. Gindorf

T. Gindorf, Supervisor
Thermal Vacuum
Environmental Requirements
Group

D. Hess

D. Hess, Manager
Environmental Requirements
Section

J. Small

J. Small, Manager
Project Engineering Division

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

PRECEDING PAGE BLANK NOT FILMED

CONTENTS

I. SUMMARY

The purpose of the Extreme Temperature Requirements (ETR) study is the determination of the extremes of temperature expected to be experienced by electronics components on future spacecraft. This information will be used to 1) evaluate the capability of existing parts and 2) serve as design criteria for new piece parts.

The orbiter investigation is centered on a hypothetical spacecraft which may be instrumented for an orbiter mission around any of the planets. The spacecraft instrumentation in each case is based on the instruments proposed for orbiter spacecraft noted in the Phase I report (JPL document 900-212). Consequently, each spacecraft contains electronics considered representative of those expected to be found on any orbiter spacecraft actually launched in this century.

For the inner planet orbiters, high temperature extremes may occur either while orbiting Venus or Mercury, depending upon the distance of the spacecraft from the planet's surface. Orbits around each planet are examined to determine the worst case effects.

The worst case low temperature extremes incurred by an orbiting spacecraft may be expected to occur during orbiting of Pluto. However, temperatures incurred would be expected to be less extreme than those incurred by flyby spacecraft passing through the void of outer space.

Studies indicate that by proper design temperature extremes can be limited to occur only at locations on orbiter spacecraft remote from the main bus (i.e., on science experiments and attitude control sensors). The types of electronic components contained in units at these remote locations are identified, and thermal studies are performed to determine the severity of temperatures resulting from the thermal variations in the environment. Power and weight penalties resulting from the use of various means of thermal control are evaluated and summarized in more detail in Section VI, Thermal Studies.

II. CONCLUSIONS AND RECOMMENDATIONS

The instruments and component piece parts identified as likely candidates for orbiter spacecraft were found to generally be the same as those identified as likely to be used on flyby spacecraft. The temperature extremes associated with orbiter spacecraft, while not as severe as the extremes associated with the results of the flyby spacecraft investigation, were found to exceed the capabilities of the piece parts if essentially no thermal controls were employed. As a direct consequence, the conclusions and recommendations resulting from the flyby investigation (noted in JPL documents 701-29 and 750-35) are directly applicable to the orbiter investigations, and are briefly summarized below. (More detailed analysis and comments may be found in Sections V and VI, Parts Identification and Thermal Studies.)

Special high or low temperature capability electronic parts need not be developed to guarantee the success of any orbiter mission. Orbiter missions around any planet can be performed with currently existing piece parts. However, the temperature capabilities of those parts generally dictate the acceptable range of temperatures within which satisfactory operations may take place. For orbiter spacecraft, this is expected to result in the establishment of a need for the inclusion of temperature control devices. Thermal control devices, however, consume spacecraft weight and power. Consequently, the penalty incurred by the use of currently existing parts types is revealed in terms of the quantities of weight and power required for thermal control purposes.

A. OUTER PLANETS

Thermal constraints, or penalties, associated with orbiter missions around any of the outer planets could be eased considerably by the uniform use of parts types capable of survival and operation at temperatures as low as -100°C to -125°C (which is about 50°C lower than the currently stated capability of most parts). For an orbiter spacecraft designed similar to the propcsed flyby class Thermoelectric Outer Planet Spacecraft (TOPS) and carrying similar appendage-located instruments, the savings would include a reduction of thermal control power requirements amounting to a value approaching 50 percent of the total originally specified, (corresponding to a savings of approximately 10 watts).

If development of low temperature capability piece parts for outer planet orbiter spacecraft is undertaken, the first approach should consist of an attempt to qualify existing parts types at the desired lower levels. The reason for first approaching the development in this manner is that a considerable amount of uncertainty exists relative to the understanding of actual existing capabilities. Included in low temperature capability development should be an evaluation of the expected useful life and reliability of each piece part when operated at the low temperatures.

The minimum electronics likely to be mounted at non-bus locations on outer planet orbiter spacecraft would consist of a detector element and preamplifier (containing linear bipolar and MOS IC's, metal film and wire-wound resistors, ceramic capacitors, inductors/transformers, and discrete semiconductors such as transistors, diodes, FET's, zeners, and thermistors). If qualification of existing electronics at lower temperature proves fruitless, development of new parts should begin with the above noted types.

B. INNER PLANETS

In the case of the inner planet orbiters, the "penalty" for thermal control may be reduced by the use of uniform high temperature capability electronic piece parts for assemblies appended to the main bus. Most electronic components located in these external appendages currently possess high temperature capabilities on the order of 125° C. However, certain components (primarily sensor and detector elements) have maximum temperature capabilities considerably lower (often on the order of 60° C or less) thereby constraining the maximum operating temperatures to levels lower than 125° C. Providing a uniform high temperature capability of 125° C for all piece parts would provide each instrument with relaxed thermal control requirements by eliminating or reducing shielding requirements, or providing a greater span of allowable operating and survival temperatures. Such an expanded range could be beneficial for use during the transit period between Earth and the orbital destination of the spacecraft.

The design of the physical configuration of most inner planet orbiter spacecraft is likely to locate most appendage instruments in the shade of the main spacecraft bus. (Exceptions might include certain attitude control devices and, in rare cases, a science instrument.) Most instruments are thereby shielded from the variable thermal influence of the Sun at the expense of providing some internal dissipation of power to maintain a minimum acceptable temperature. These same instruments, in planetary orbit, are also unlikely to experience high temperature extremes if a reasonable amount of thermal control is provided.

It is important to note that penalties for thermal control can also be reduced for inner planet orbiter spacecraft by increasing the low temperature capabilities of appendage instruments. Shielding or blanketing generally results in appendage located equipments experiencing a more constant temperature environment which is semi-isolated from the external environment, and is often preferred from an operational reliability point of view. Consequently, it is not entirely desirable to provide for the elimination of thermal shields or blankets. When shields or blankets are incorporated in the design of an instrument located on an appendage to the main spacecraft bus, it forces the instrument to rely on its internal dissipation of power (heat) to maintain minimum acceptable temperatures. Since low temperature capabilities of the instrument's piece parts dictate the minimum acceptable temperature of the instrument, it follows that the lower the temperature capability of the piece parts, the lower the minimum acceptable temperature of the instrument. In turn, the lower the minimum acceptable temperature of the instrument, the less heater power required to attain that temperature. The difference in internal power dissipation requirements for instrument heating can be viewed, then, as a reduction in cost for thermal control.

From the standpoint of power and weight penalties incurred, cost savings that may be realized from development of high temperature (100° C to 125° C) electronics for inner planet orbiter spacecraft are probably lower than those that may be realized from the development of low temperature electronics. However, if development of high temperature components is to be undertaken, initial efforts should be directed towards attaining high temperature capabilities or suitable replacements for the following components: Lead Sulfide and Rubidium Absorption Cells; Lithium Drifted, Silicon, and Germanium Solid State Detectors; Cesium Iodide Photodiodes; Photomultiplier, Geiger-Müller, and Vidicon Tubes; Cherenkov and Scintillation Counters; and other solid state devices having high temperature capabilities less than 125° C.

Few inner planet orbiter missions are likely to take place in this century, and of those that will, the number containing instruments requiring the use of special thermal control devices or high temperature capability components is expected to be quite small. If the choice were made to utilize high temperature capability components for such missions, development of high temperature detectors would be required. However, high temperature capability detectors generally have few uses aside from the particular spacecraft instrument for which they are designed. Existing thermal control techniques (as indicated in Section VI) and devices (shields, blankets, louvers, etc.) can provide acceptable temperatures while incurring little or no additional power or weight penalties. Since such devices and techniques are readily available, it is recommended that high temperature capability detector development be waived in favor of utilizing thermal controls to provide satisfactory temperatures on inner planet orbiter spacecraft.

III. APPROACH

The ETR study is being performed in two phases. The first phase (concluded in FY'69 and summarized in JPL Document 900-212) dealt with possible methods of determining temperature extremes likely to be encountered by electronics parts used on future space missions. Most of the studies performed for NASA dealing with future missions were examined and reviewed in an attempt to identify those missions likely to produce extreme piece-part temperatures. In addition, the possibility of extending those missions to more extreme conditions by using the same spacecraft on longer or more distant missions was considered. However, the Phase I mission review showed that none of the missions represented extreme temperature conditions. Furthermore, in all cases studied, spacecraft temperatures were required to be maintained within a nominal (usually room-temperature) range as a design constraint. This constraint was dictated by the capability of current parts and caused the spacecraft configuration and mission profile to be optimized for those ground rules.

What this meant in terms of pursuing the original study approach was that: 1) no extreme missions were discovered, and 2) extension of existing missions to the extreme cases was not possible without re-optimizing for the extended mission environment. This would require a complete new mission study for each case desired.

The revised approach started with the assumption that some form of thermal control (requiring a finite amount of weight and power) would be used on all future space missions. These missions could then be classified (in ascending order of thermal control complexity) into three cases:

(1) Flyby missions, which observe the celestial target for a brief period and at a closest approach distance of a thousand or more kilometers (this category also includes solar and deep space probes).

(2) Orbiter missions, which perform similar measurements much more frequently with many revolutions around a planet and at distances closer to its surface.

(3) Lander missions, which include all survival surface missions (capsules, hard landers and soft landers) that return telemetered data to Earth.

In Phase I, it was shown that the spacecraft bus[1] could be thermally decoupled from the solar environment for flyby missions. Once the bus is successfully decoupled, a reasonable amount (normally less than 4 percent of weight and power) of thermal control is sufficient to maintain acceptable temperatures for the electronics contained inside the bus. Therefore, the extreme temperature problem is reduced to considering only the effects on items which cannot be included in the bus, such as science experiments and attitude control sensors.

For any particular spacecraft using such thermal control methods, the problem of discovering which electronic parts types experience severe temperature environments breaks down into two tasks.

A. PARTS IDENTIFICATION

This task involves listing the electronic piece part types contained (or expected to be contained) in each spacecraft subsystem. As a portion of this task, the present temperature capability of each parts type is identified.

B. THERMAL STUDIES

This task involves estimating the temperature extremes to be encountered during the mission for all of the spacecraft subsystems. Some idea of the penalty imposed by the thermal control requirements can be obtained by a comparison of the estimates of the weight and power required for each subsystem and the temperatures which would be reached if no thermal control were used.

---

[1] "Bus" is defined as the main spacecraft structure (exclusive of appendages such as booms, antennas, solar panels, etc.) which contains the major portion of spacecraft control electronics. It is usually thermally controlled as a unit.

Performing the above tasks for non-bus hardware should yield answers to the following questions:

(1) Is there an extreme temperature problem for electronics parts on the spacecraft?

(2) What parts are involved?

(3) What penalty must be paid, in terms of weight and power, for a thermal control sufficient to bring all affected parts within an acceptable temperature range?

(4) What cost savings could be realized by the development and utilization of parts that would eliminate the current temperature constraints, thereby establishing new limits at less constraining levels?

The first mission selected to provide answers to these questions was the JPL TOPS mission. The TOPS' primary mission is to perform the Grand Tour Flyby of Jupiter, Saturn, Uranus, and Neptune during the unique 1976-79 launch opportunity. Such a mission was considered by the ETR study to be limiting cold-temperature case for any flyby missions planned for this century. The results of that study are contained in JPL Document 701-29.

The second mission selected to provide answers to these questions utilized a hypothetical, fully attitude stabilized spacecraft. The spacecraft mission was assumed to consist of interplanetary environment investigations between Earth and 0.1AU of the Sun, and flyby investigations of the inner planets Venus and Mercury. Since no spacecraft in the next 30 years is expected to operationally approach nearer the Sun than 0.1AU, the environmentally produced thermal response of the hypothetical spacecraft, evaluated at 0.1AU, was considered to encompass the limiting high temperature extremes that will be experienced by actual flyby spacecraft in this century (JPL Document 750-35).

Orbiter class spacecraft missions are examined to identify additional answers to the questions stated above. Again a hypothetical, fully attitude stabilized spacecraft was assumed as a model. Two distinct sets of instruments were considered associated with the model: one set representative of spacecraft proposed for Venus or Mercury orbiter missions, and one set representative of spacecraft proposed for Mars or outer planet orbiter missions.

Circular orbits of 500 and 1500 km above the surfaces of both Venus and Mercury were selected for analysis of the high temperature extremes, based on the lowest orbits proposed in missions identified in the Phase I report. Outer planet orbits were not investigated, since the set of instruments found representative of outer planet orbiters was found to be the same as those identified for outer planet flyby spacecraft, and the more conservative estimate of cold temperature extremes had already been established during cold temperature flyby case examination.

IV. HYPOTHETICAL MISSION RATIONALE

The orbiter case study was undertaken with the philosophy that the equipments and environments studied must be representative of typical orbiter missions. No single mission reviewed in the mission studies satisfied this criteria. Instruments recommended for use on spacecraft proposed for orbiting Venus, however, were found to also be recommended for use on Mercury orbiters. Similarly, the instruments recommended for use on Mars orbiters were also noted as generally recommended for use on outer planet orbiters. Consequently, a hypothetical spacecraft mission was synthesized, and assumed to contain a spacecraft consisting of a bus electronics package and a set of appendage located instruments (representative of typical planet orbiter instruments). The mission objectives were defined to be consistent with the generally stated objectives of all proposed orbiter missions. Orbits of 1500 km (identified as typical for proposed inner planet orbits) and 500 km (worst case extreme) were examined.

Orbits lower than 500 km were excluded from examination since 1) no missions were identified that proposed orbits lower than 500 km, and 2) spacecraft approaching nearer than 500 km to an inner planet are more likely to be planetary atmosphere or surface probes, which will be examined in the lander study. Orbits of 1500 km were noted as being fairly representative of orbital heights most likely to be attempted.

The main bus of the spacecraft (containing the bus electronics packages) was assumed to be thermally decoupled from impinging solar energy, consistent with the capabilities deemed reasonable as indicated in the Phase I report. Planetary radiation and albedo were assumed to be capable of influencing bus temperatures by a relatively insignificant amount (see Appendix I). The temperatures of externally located experiment and attitude control packages, however, were considered influenced by the amount of solar radiation, planetary radiation and albedo radiation intercepted. The highest temperatures would be expected to occur where the maximum radiation intercepted was converted into heat (which might be at either Venus or Mercury), and the lowest temperatures would be expected where the total radiation intercepted was minimal.

The worst case high temperatures would be experienced when an assembly is simultaneously irradiated with solar radiation, and planetary infrared and albedo radiation. To insure the most conservative high temperature extreme producing situations were examined, the orbiter study of the inner planets initially assumed all three forms of radiation impinged on the externally located instrument packages of the hypothetical spacecraft.

An orbiting spacecraft would logically intercept the least amount of solar radiation, planetary radiation, and albedo, while in orbit around Pluto, the coldest, most distant planet from the Sun. However, the travel time for an orbiter spacecraft to reach the outer planets is sufficiently long, so that the actual cold temperature extremes expected to be incurred by such a spacecraft will occur in transit, where planetary radiation and albedo are insignificant and solar radiation is minimal. Consequently, that point on a spacecraft's trajectory wherein the minimal amount of solar radiation alone would impinge on a spacecraft was judged most appropriate for determining the worst case cold temperature extremes.

V. PARTS IDENTIFICATION

Knowledge of a precise spacecraft configuration is not required for identification of electronic equipments located outside the main bus. For nearly all spacecraft, most electronic packages located externally consist of science experiments, which require a field of view of the phenomena being observed. The remaining externally located electronics packages generally consist of attitude control system equipment. Both science and attitude control packages may be located on a scan platform or booms.

Lists of electronics parts types for each of the non-bus equipments have been prepared by the cognizant JPL technical divisions. Included in the parts lists are estimates of the current temperature capabilities of each piece part.

Tables 1 through 3 summarize the identification of piece parts and associated temperature capabilities. Basically, each instrument can be considered to consist of a sensing element or detector, a signal amplifier, and circuitry to convert the signal into a data format suitable for transmission to Earth. In all cases, the detector is the one component which must be exposed to the ambient environment where the sensing of raw information takes place. The balance of the electronics associated with each instrument may be located at the same physical location as the detector, or it may be partially included in the bus hardware. For any particular instrument, the minimal amount of electronics likely to be exposed to the environment would consist of a detector element and a pre-amplifier to carry the signal to the bus, where the remaining data handling could take place.

Most of the electronics parts likely to be used on Venus or Mercury science experiments and attitude control devices (aside from detector elements) have a high temperature capability of 125°C. However, nearly all of these packages contain detectors having high temperature capabilities significantly lower than 125°C. As a consequence, high temperature operating limits are generally imposed on instrument designs in order to accommodate detector-dictated high temperature limitations.

If orbiter missions to the inner planets were likely to produce high temperatures (>50°C) in the external electronic equipment, detector devices would be likely to be the first components adversely affected. However, it is

Table 1. Parts Type Breakdown-Science Experiments. All parts have a temperature capability of -55°C to 125°C, except as noted.

INSTRUMENT \ PART TYPE	Digital ICs	Linear Bipolar ICs	Linear MOS ICs	Carbon Resistors	Metal Film Resistors	Discrete Semiconductors[1]	Tantalum Capacitors	Glass Capacitors	Ceramic Capacitors	Paper/Mylar Capacitors	Inductors/Transformers	Crystals (Quartz)	Bipolar/MOS ICs	Bipolar/MOS MSI ICs	Bipolar/MOS LSI ICs	Lead Sulfide/Rubidium Cells[2]	Lithium Drifted Detectors[3]	Silicon/Germanium Detectors[4, 15]	Cesium Iodide Photodiodes[5]	Photomultiplier Tubes[6]	Geiger-Müller/Vidicon Tubes[7]	Cherenkov Counters[8, 9]	Scintillation Counters[8,9]	Quartz Sensors	Ni Fe Cores, Copper Wire Wound[14]	Lithium Cores[16]
Rubidium Vapor Magnetometer[10]	X	X	X	X	X	X	X	X	X	X	X		X	X	X	X										
Fluxgate Magnetometer	X	X	X	X	X	X	X	X	X	X	X		X	X	X										X	
Helium Magnetometer[11]	X	X	X	X	X	X	X	X	X	X	X		X	X	X	X										
Foil Sensor Micrometeroid Detector	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X											X
Acoustic Sensor Micrometeroid Detector	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X											X
Energy Spectrum Plasma Probe	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X											
Directional Plasma Probe	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X											
Solar Proton Particle Detector	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X		X		X							
Neutron Particle Detector	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X			X		X						
Electron Particle Detector	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X			X			X					
Ionization Chamber Particle Detector	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X						X			X		
Gamma-Ray Detector	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X					X		X	X			
X-Ray Detector	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X						X					
Trapped Radiation Detector	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X			X			X					
Infrared Spectrometer	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X			X								
Ultraviolet Spectrometer	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X					X						
Infrared Radiometer[12]	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X											
Microwave Radiometer[13]	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X											
Multicolor or White Light Photometer	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X					X						
Ultraviolet Photometer	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X					X						
Wide Angle Television	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X						X					
Narrow Angle Television	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X						X					
Radar Altimeter	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X											

[1] Includes Transistors, Diodes, FET's, Zeners, Etc.

[2] Maximum Temperature Capabilities: Lead Sulfide Cell, 55°C; Rubidium Absorbtion Cell, 40°C.

[3] Maximum Temperature Capability: Lithium Drifted Solid State Detectors, 50°C.

[4] Maximum Temperature Capabilities: Silicon Solid State Detectors, 40°C; Germanium Solid State Detectors, 50°C.

[5] Maximum Temperature Capability: Cesium Iodide Photodiode, 50°C.

[6] Maximum Temperature Capability: Photomultiplier Tubes, 0°C to 70°C.

[7] Maximum Temperature Capabilities: Geiger-Müller Tubes, 20°C to 55°C; Vidicon Tubes 25°C.

[8] Maximum Temperature Capability: Cherenkov and Scintillation Counters, 40°C.

[9] Minimum Temperature Capability: Cherenkov and Scintillation Counters, -20°C.

[10] Also contains plastic lenses with a maximum temperature capability of 65°C.

[11] Also contains a polarizer with a maximum temperature capability of 65°C.

[12] Also contains a Bismuth-Antimony Thermopile with a maximum temperature capability of 30°C.

[13] Also contains a ferrite switching device and special hot carrier diodes with temperature capabilities of -20°C to 70°C.

[14] Minimum Temperature Capability: Copper Wire Wound Ni Fe Cores, -20°C.

[15] Minimum Temperature Capability: Silicon Solid State Detectors, -10°to -40°C.

[16] Minimum Temperature Capability: Lithium Cores, -20°C.

Table 2. Potential Science Detectors for Orbiter Missions

Detector	Current Estimation of Temperature Capability, °C		Possible Instruments
	High	Low	
Image/Sound Orthocon	20	-40	Television Cameras
Germanium Bolometer	50	-150	IR Radiometers
Lead Sulfide Cells	55	-250	Vector Helium Magnetometers; Planet Sensors, Approach Guidance
Copper Wire Inductors Wound on NiFe Cores	60	-20	Fluxgate Magnetometers
Doped Detectors (for IRR use)	50	-50	IR Radiometers
Crystal Microphones	125	-55	Acoustic Sensors, Micrometeroid Detectors
Photomultipliers with Si, SiO_2 etc. surfaces	0 to 70	-100	Optical Sensors, Radiation Instruments
Curved Gold Plated Magnesium Disks; Faraday Cups	>125	-100	Energy Spectrum Plasma Probes, Directional Plasma Probes
Lithium Drifted Solid State Detectors	40	-55	Solar Proton Particle Detectors
Cesium Iodide Photodiodes	40	-55	Solar Proton Particle Detectors
Silicon Solid State Detectors	15 to 40	-10 to -40	Neutron Particle Detectors; Cosmic Ray, Trapped Radiation, Low Energy Radiation Detectors
Germanium Solid State Detectors	50	-40 to -55	I. R. Spectrometers, Radiation Detectors
Helium Cells	100	-50	Magnetometers
Photomultiplier Tubes	0 to 70	-55 to -70	UV Photometers, Neutron Particle Detectors, Gamma-Ray Detectors, UV Spectrometers, Multicolor or White Light Photometers
Geiger-Müller Tubes	20 to 55	-30 to -100	Electron Particle Detectors, Ionization Chamber Particle Detectors, X-Ray Detectors, Trapped Radiation Detectors
Cherenkov Counters	40	-20	Gamma-Ray Detectors
Scintillation Counters	40	-20	Gamma-Ray Detectors
Bismuth-Antimony Thermopiles	30	-200	IR Radiometers
Ferrite Switching Devices, Hot Carrier Diodes, Gallium-Arsenide Semiconductors	70	-20	Microwave Radiometers
Vidicon Tubes	25	-40 to -65	Television Cameras
Electrometers (Vibrating Reed)	75	-50	UV Spectrometer, IR Radiometer/Spectrometer, Magnetometer
Electromagnetic Antennas	100	-150	Microwave Radiometers, Ionization Analysis of Planetary Atmospheres, Occultation Experiments, Radar Altimeter
Electron Multipliers			
(a) Multistage Secondary Electron Simulator	20	-100	UV, Mass Spectrometer
(b) Secondary electron conductor	20	-100	UV, Mass Spectrometer
(c) Electrostatic and Electromagnetic Chance Multiplier	20	-100	UV, Mass Spectrometer
Silicon Sensors	-10	-65	TV
Photo Cathode Devices	20	-65	TV
Rubidium Absorption Cells	40	0	Rubidium Vapor Magnetometer

Table 3. Parts Type Breakdown - Attitude Control System. All parts have a Temperature Capability of -55°C to +125°C, except as noted.

INSTRUMENT \ PART TYPE	Digital ICs	Linear Bipolar ICs	Linear MOS ICs	Carbon Resistors	Metal Film Resistors	Discrete Semiconductors	Tantalum Capacitors	Glass Capacitors	Ceramic Capacitors	Paper/Mylar Capacitors	Inductors/Transformers	Crystals (Quartz)	Bipolar/MOS ICs	Bipolar/MOS MSI ICs	Bipolar/MOS LSI ICs	Cadmium Sulfide Detectors[3]	Silicon Solid State Detectors[4]	Gallium Arsenide Light Sources[4]	Vidicon Tubes[5]	Image Dissector Tubes[6]
Cruise Sun Sensors	X			X[1]		X[2]	X									X				
Acquisition Sun Sensor and Sun Gate		X		X												X				
Digital Position Encoder						X											X	X		
Approach Guidance Tracker	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X			X	
Canopus Tracker		X		X	X	X	X		X	X	X									X

[1]Wire wound precision resistors, also.

[2]Includes Transistors, Diodes, FET's, Zeners, Etc.

[3]Maximum Temperature Capability is 75°C; Minimum temperature Capability is -45°C.

[4]Maximum Temperature Capability is 40°C.

[5]Maximum Temperature Capability ranges from 25°C to 30°C; Minimum temperature Capability is -20°C.

[6]Maximum Temperature Capability is 75°C; Minimum temperature Capability is -15°C.

not likely that high temperatures will be a problem for most instruments. (See Section II, Conclusions and Recommendations, and Section VI, Thermal Studies.)

The majority of the electronics parts likely to be used on outer planet science instruments (again aside from detectors elements) have a low temperature capability of -55°C. Detector elements, however, are not as easily categorized. Detector elements with possible low temperature problems include silicon solid state radiation detectors (-40° C), selenium compound vidicons (-40°C), certain Photomultiplier tubes (-10°C to -20°C), certain Geiger Müller tubes (-30° C), and certain ferrite switching devices, hot carrier diodes and Gallium Arsenide Semiconductors (-20°C).

Attitude control sensors for outer planet orbiters use parts types similar to those contained in the science units. For example, the approach guidance tracker uses a type of TV camera for a sensor, with electronics located in the bus. An existing configuration of a Canopus Tracker has a quoted lower temperature limit of -15°C, which may be mainly due to its ancestry as a bus instrument on previous Mariner missions. The parts types listed for it include items used in the science units; thus, it would seem that operation at lower temperatures could be accomplished, or as a last resort, the unit could be placed within the bus.

In general, most spacecraft missions that have taken place to date have focused most attention on upper qualification temperature limits. The relatively minor significance attached to the low temperature limits may be more a reflection of current MIL-Specification performance guaranteed by manufacturers than an absolute floor on low-temperature capability. In other words, some of the parts with quoted low end capabilities of -50°C or so may actually be capable of operation at significantly lower temperatures. Current qualification specifications do not require verification of such capability since previous missions permitted operation of the parts at comfortably higher temperatures. In these cases, qualification of parts for lower temperature operation may be attainable while avoiding some of the costs otherwise incurred in the research and development of new piece parts.

VI. THERMAL STUDIES

Identification of extreme temperatures that may be encountered is, in reality, a complex task. This becomes evident when it is realized that various means of thermal control are available to provide nearly any range of temperature desired for operation of any non-bus instrument. However, the greater the amount of thermal control required, the greater the penalty incurred, in terms of power and weight needed to provide that thermal control. Ultimately, increases in the thermal control requirement can be seen to be associated with increases in cost per data bit retrieved from the mission.

From the standpoint of obtaining the maximum scientific information from spacecraft mission for the least cost, the use of instruments capable of operation throughout a wide range of temperatures is highly desirable. The use of such instruments would minimize the power and weight requirements for thermal control devices.

However, the range of temperatures within which any instrument may operate is generally limited by the thermal capabilities of its components. To increase this range, development of components with greater thermal capabilities would be required. Ultimately, a trade-off must be made between paying the price for thermal control and paying the price for development of wide temperature capability components.

If increased piece part capabilities could result in the elimination of one or two pounds of weight used solely for thermal control purposes, perhaps an additional scientific instrument could be placed on board. In this way the scientific worth of the spacecraft might be increased. On the other hand, piece part development would not be economically sound if the development costs exceeded the cumulative savings that result from the reduction of thermal control weight and power requirements.

A. ASSUMPTIONS AND APPROACH

Orbiter spacecraft can be classified into two distinct categories. One category would include those orbiters projected for use around Venus or Mercury, where the proximity to the Sun and certain physical properties of each planet combine to produce an environment that makes available a significant supply of heat capable of increasing the temperature of the spacecraft. The other category of orbiter spacecraft primarily includes those orbiters projected for use around any of the outer planets. This category is characterized by the lack of heat sources in the environment that are capable of significantly influencing spacecraft instrument temperatures (aside from certain IRR devices).

For the purpose of evaluating temperature extremes incurred by orbiter spacecraft, thermal studies should be performed on representative equipments in each category. However, thermal studies were not performed on outer planet orbiter spacecraft. The basis for this action rests on two points. First, the instruments identified as logical choices for use on outer planet and Mars orbiters were noted to be generally identical to those identified as logical choices for use on outer planet flyby spacecraft. Secondly, the temperature extremes expected to be incurred by these instruments on an orbiter spacecraft mission would in no case be more extreme than if the instruments were on an outer planet flyby spacecraft mission. Consequently, the results drawn from the thermal studies performed relative to flyby spacecraft experiencing cold temperature extremes are directly applicable to Mars and outer planet orbiter spacecraft as well, and have been included in Section II, Conclusions and Recommendations.

Inner planet (Venus and Mercury) orbiter thermal studies were undertaken. Even though the instruments proposed for inner planet orbiter spacecraft are nearly identical to those identified as likely candidates for inner planet flyby spacecraft, the additional impinging energy attributed to planetary radiation was considered of sufficient magnitude to warrant analysis of its effects. As in the high temperature flyby examination, a parametric approach was employed to determine the impact of variations in solar intensity, planetary infrared radiation, albedo, and certain physical properties of instruments. Solar radiation intensities were limited by orbital considerations to values no greater than that normally intercepted at 0.31 AU (Mercury perihelion).

The nature of an inner planet orbital mission increases the complexity of a spacecraft's thermal system relative to that of a flyby mission through the same point in space. As in the flyby case, the initial phase of an orbiter mission encounters solar radiation which varies in intensity between Earth and the destination planet. In orbit, not only is the direct solar irradiation of concern, but also the constant proximity of the spacecraft to the planet, which allows appendage located equipment to also be thermally influenced by planetary radiation. The planetary radiation is derived from two sources: the albedo, or reflected solar energy, and the planet's infrared radiation (which is characteristic of a planet's equilibrium temperature). Another thermally significant difference occurs to spacecraft on orbits traversing through the shaded or night side of a planet, which results in the cyclic application and removal of the solar and albedo irradiation of externally located equipment.

Based on the previous study of flyby missions, an optimally configured cubical model of an instrument housing was assumed. Primary solar-irradiation control was incorporated in the form of a second-surface mirror for the surface of the sunlocked face. As in the flyby investigation, the cube was defined to contain an area of 0.5 ft^2 per face. The initial thermal-control parameter examined was the type of housing material selected for the non-sunlocked faces, since certain physical properties of the surface material control the quantity of radiant energy absorbed from the planet as well as the quantity reradiated to space. Other instrument characteristics considered included the length of the housing along the sun line, which related the lateral reradiating area and the supplemental internal heating.

Certain other parameters related to the orbit selected for the spacecraft, planetary radiation properties, and instrument geometrical properties were considered. The main parameter governing the radiation intensity from a planet is defined by the altitude of the orbit. Two values of orbit altitude were evaluated, 500 and 1500 km. The values were considered representative of nominal and worst case orbits likely to be selected for actual missions.

The second parameter closely examined relates to the radiation-intensity distribution of a planet. Certain positions in specific orbits provide more conservative extremes in terms of total received radiation than others. In particular, the first portion is located where a spacecraft crosses the planet-sunline at noon, and the second point is located when the planet-sunline is crossed at midnight, (which can also be referred to as light or subsolar and dark or shadowed points). At the former, all three energy sources impinge on a spacecraft, in maximum concentration. At the latter, only the planetary infrared radiation is present.

Values of various planetary characteristics pertinent to the investigation, and used in the analysis, are given below in Table 4.

The analysis was performed for circular orbits around Venus, Mercury perihelion, and Mercury aphelion. Rather than constructing a thermal system for the computer program consisting of the Sun, a planet and the instrument, the planet was replaced by a factor representing the amount of energy radiated from the planet which is absorbed by the entire instrument. The most critical segment of this portion of the analysis occurred during the determination of the viewing factor for the planet. This criticality was due to the fact that each face of the instrument housing had to be separately analyzed to determine the resultant heat input, which was not the same for all surfaces due to the differences in the orientation of each surface relative to the Sun, planet, and space view factors. The factors were ultimately determined utilizing tables based on an analysis of Earth orbits, which were recommended for use in the determination of view factors for orbiters of all the other planets.[2]

Table 4. Planetary Properties and Characteristics

Characteristics / Position	Displacement, AU	Solar Flux,* BTU/hr ft^2	Planet Diameter, 10^6m	Planet Albedo	Planet Emissivity	Surface Temperature °F	
						Noon	Midnite
Mercury Perihelion	0.31	4533	4.84	0.06	0.235	750	70
Mercury Aphelion	0.47	1792	4.84	0.06	0.235	490	70
Venus	0.72	823	12.1	0.76	0.06	890	665

*Based on 430 Btu/hr ft^2 for Earth

[2]Stevenson, J.A. and J.C. Grafton; "Radiation Heat Transfer Analysis for Space Vehicles," ASD Report 61-119, Part I, North American Aviation Inc., SID 61-91, JPL 95036, December 1961.

As mentioned earlier, except for polar orbits along the terminator, spacecraft orbits about a planet will be in the shadow of the planet for a certain percentage of the total orbit time (up to approximately 50 percent). Such spacecraft will be subjected to varying amounts of radiation. In particular, when crossing the terminator between the light and dark sides, an instrument will be impressed with an almost discontinuous change in energy level. As a consequence, an instrument's temperature response can be expected to experience a time lag which is a function of its thermal properties, particularly its heat capacity. In turn, the temperature of the instrument may not reach its terminal, steady state value. The value it does attain depends on the change in the stimulating energy and its duration, which is determined by the orbit period.

The calculated circular orbital periods for orbiters at various altitudes are listed in Table 5. The time during which the energy distribution is constant can be considered to vary from the entire orbit period (for polar terminator orbits) to a very small duration on the order of one-half hour or less (for elliptical orbits centered on the planet-sun line).

The total effect of this variability, as experienced by the instrument, will be an energy impressment that oscillates in magnitude. When the duration of impressment of a relatively constant magnitude energy is on the order of one-half hour or less, the thermal response of the instrument may prevent the instrument from attaining the terminal value indicated. On the other hand, when the duration is greater than one-half hour, the size of the instrument and its heat capacity may be sufficiently small such that the terminal steady state values (based on qualitative comparisons with spacecraft analysis) may be attained. Thus, for the purpose of this ETR study, the terminal, steady state value of temperature serves as a conservative approximation of the actual extremes that may be incurred.

Table 5. Circular Orbital Periods About Inner Planets, in Hours

Planet \ Orbital Altitude, km	500	1000	1500
Mercury	1.91	2.50	3.02
Venus	1.63	1.82	2.01

The method of thermal analysis employed consisted of constructing an analytical thermal balance in order to identify the contributing mechanisms. Then, the detailed thermal system with each contributing component was defined. Finally, these components were reformed into the arrangement which best facilitated the use of the Thermal Analyses System (TAS) computer program. The computer yielded the instrument housing equilibrium temperature results as each parameter was submitted in sequence.

B. DISCUSSION AND CONCLUSIONS

Comfortable temperatures can be attained by appendage instruments through the application of good thermal control practices. Without the use of thermal control techniques, spacecraft instruments could experience temperatures ranging from less than 200°C below zero to more than 100°C above. However, by effectively eliminating (or controlling) the heat absorbed from the Sun and any orbited planet, instrument temperatures can be made primarily dependent upon the amount of heat dissipated internally. For instruments similar to the cubical model assumed in this study, dissipation of only a few watts internally through thermostatically controlled heaters will result in nominal temperatures being constantly maintained.

To determine what the "worst case" extremes without thermal control could be, a cubical model dissipating no power internally was examined in terms of its interaction with the environment. Other than assuming the instrument had a second surface mirror on its sunlocked face (to limit the maximum temperature attained through solar irradiation), as an extreme case, no forms of thermal control were considered employed. The temperatures experienced by such an instrument were found to vary from 100°C below zero (essentially no heat being absorbed) to more than 100°C above zero (where both the Sun and the orbited planet contribute sizeable quantities of heat). Furthermore, severe temperature oscillations could occur when a spacecraft is in a planetary orbit that causes it to pass in and out of the planet's shadow. Figures 1 through 3 in Appendix II present the graphical illustrations of those effects as a function of housing materials and orbital distance from the planet.

As a more realistic case, a model embodying the application of good thermal design practices was studied and a completely different picture evolved. First, the instrument was considered to be shielded from the Sun by the main spacecraft bus or other shielding techniques. This move was based on the point that most orbiter appendage instruments are designed to view the planet, not the Sun, and consequently do not have to be in the Sun. Thus, the heat variable attributable to the Sun was eliminated.

Secondly, the heat absorbed by the instrument from planetary radiation was minimized. This was accomplished by covering the instrument with a superinsulating blanket. Using a thermal blanket not only reduces the quantity of energy absorbed, but also acts as a thermal oscillation damper. The latter action occurs by introducing a longer time constant for instrument response to abrupt changes in the magnitude of impinging radiation.

Thirdly, internal dissipation of power through thermostatically controlled heaters was used as means of heating the instrument. The heat was used to achieve a relatively constant temperature, at levels generally considered acceptable, for any inner planet orbiter greater than 500 km from the surface of a planet.

Finally, the use of thermostatically controlled louvers and/or the addition of radiating fins, in lieu of the thermal blanket, on that side of the instrument that never views the planet was considered as a means of handling any excessive amounts of heat generated internally. (Production of heat in excess of that required to attain a comfortable temperature could come about through the normal operation of the instrument, wherein it may dissipate more power than basically needed for the production of an acceptable thermal environment.)

For illustrative purposes, a cubical aluminum model (containing approximately 600 in.^{3}) was examined. This configuration and size model was considered generally representative of typical instruments proposed for orbiter spacecraft.

In the first case examined, the model was assumed to 1) be shielded from the Sun by the main spacecraft bus, 2) be covered by a super-insulating thermal blanket 3 mils thick, weighing less than 0.1 lb, 3) contain heaters, thermostatically controlled, and 4) dissipate one watt of power during normal operation. Analysis indicated that during interplanetary flight, 2 watts of internal power dissipation would produce a constant 22° C instrument temperature. In the worst case planetary orbit, 500 km above Venus, the one watt of instrument operating power supplemented by one watt of thermostatically controlled heater power (controlled to open at 25° C) would result in instrument temperatures fluctuating a few degrees centigrade around 25° C.

In the second case examined, the original model was modified in two ways. First, instead of only one watt being dissipated internally, twelve watts were considered dissipated during orbital operations. (This value was based on a conservative estimate of the highest power density instruments likely to be used on orbiter spacecraft.) The second modification consisted of removing the constraint that the entire instrument package be covered by a superinsulating thermal blanket. As in the first case, the instrument was assumed to have no requirement to directly view the planet.

By varying certain surface properties of the instrument, comfortable temperatures were found to be obtainable. For example, analysis indicated that the instrument would attain a temperature of 25° C while orbiting 500 km above Venus, if one sixth of its total surface area was coated with a high emissivity material ($\epsilon = 0.9$) which viewed space alone and the balance of the instrument was covered by 3 layers of superinsulation weighing 0.1 lb. In interplanetary flight a total of 13 watts dissipated through the thermostatically controlled heaters would be required to maintain the same temperature.

The final case considered the problems involved when some portion of an instrument must view the planet. The original cubical model was assumed to contain a TV camera, consisting of optics, vidicon, and signal preamplifier. A 4.6 in. diameter quartz lens was conservatively selected to represent that surface area of the instrument requiring a view of the planet; this estimate was based on the proportion of Mariner Mars 1971 (MM'71) TV lens' areas to the total TV surface areas. Eight watts were assumed to be internally dissipated, similarly based on the most conservative power density relationship existing in the MM'71 TV cameras.

In a 500 km orbit above Venus, the cubical TV model was found to be capable of attaining temperatures between 20 and 30°C when certain thermal control techniques are applied. For example, in one configuration, a quartz reflector could be placed in front of the TV lens, thereby restricting the energy entering the lens system. With a 3 layer superinsulation thermal blanket covering the rest of the cube (except for 1/6 of its total surface which views space and has an emissivity property of $\epsilon = 0.9$), a temperature of 20° C would be expected. During interplanetary flight, the same temperature would result with the total dissipation of 9-1/2 watts of power through thermostatically controlled heaters.

As noted previously, other thermal control techniques are available to limit assembly temperatures, when the particular situation warrants some other approach. For example, thermostatically or mechanically controlled louvers with different emissivity coatings on each side might be used to eliminate a thermal problem in one situation. In another, radiative fins might be used to increase the total radiative area, thereby limiting the maximum temperatures attainable. In any case, it does not appear likely that high temperature extremes will be a serious problem for inner planet orbiter spacecraft.

APPENDICES

APPENDIX I
SPACECRAFT BUS THERMAL ISOLATION

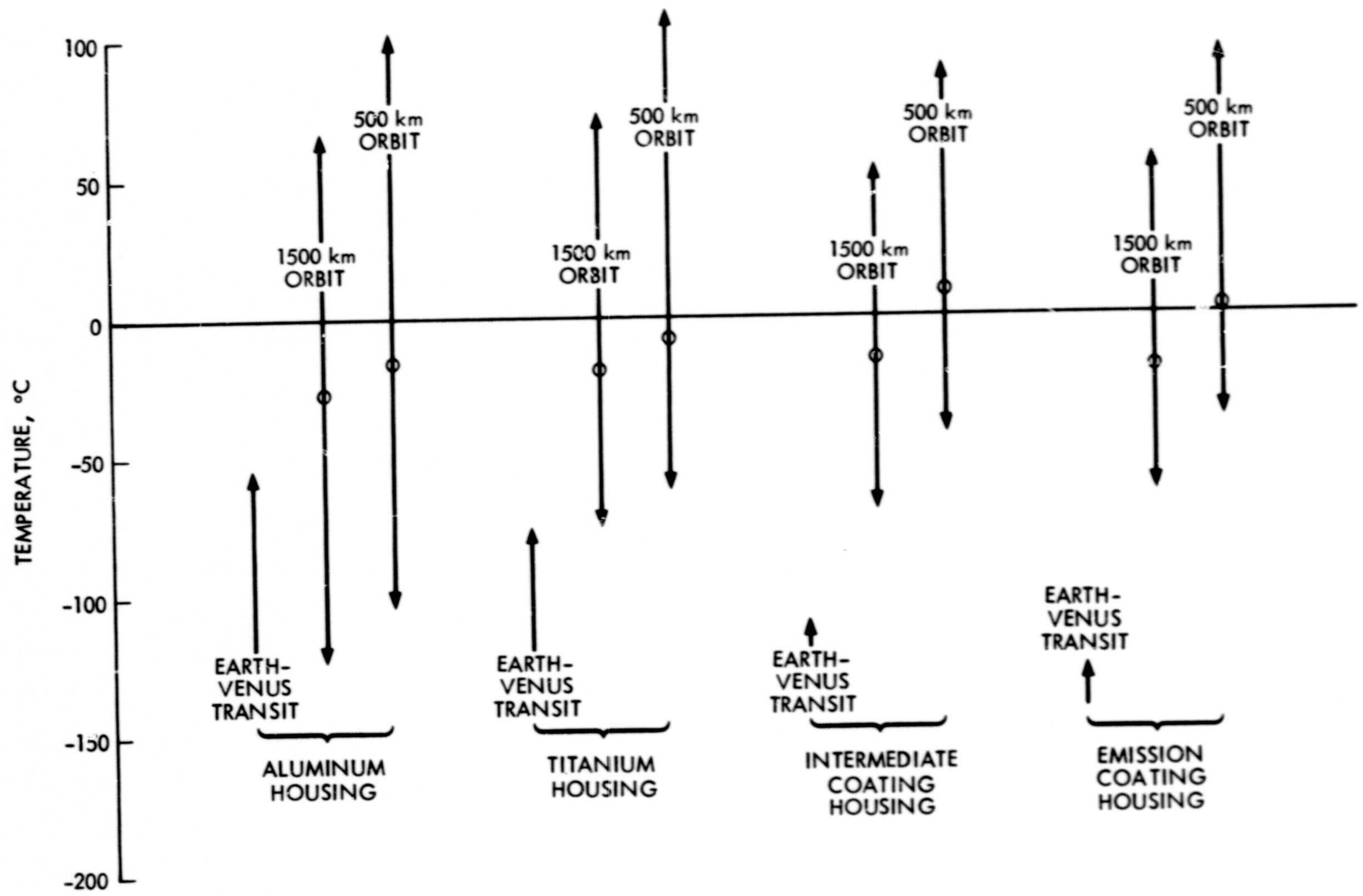
Since the planetary radiation is sufficiently intense to require some consideration, even though small, of the thermal control of instruments orbiting the inner planets, there remains some concern for the thermal isolation of the orbiting spacecraft bus. The experience of thermal control space technologists with spacecraft design and operation--Mariners '69 and '71--indicates that the potential of the techniques of control is more than adequate to maintain design temperatures of the bus during planned orbits about the inner planets.

However, to confirm this judgment, a model spacecraft bus was evaluated relative to the worst case orbit that might be attempted - a 500 km polar orbit around Venus in a plane containing the Sun-planet line. The model bus was geometrically defined to be a cube containing 45 square feet of external surface area, within which 350 watts of power was continuously being dissipated (roughly corresponding to the Mariner class spacecraft size and power dissipation). The two opposing faces of the cube parallel to the plane of orbit were considered coated with a high emissivity coating ($\epsilon = 0.9$). With the remaining four surfaces of the cube covered by several layers of superinsulation (such that the cube was essentially thermally isolated from the effects of planetary and solar irradiation), an average bus temperature near 25°C would be attained.

Temperatures near this value would prevail throughout the entire mission (except for transients induced by midcourse maneuvers, etc.) as long as 350 watts continue to be dissipated. Changes in the power dissipation levels could be handled in a manner similar to the way Mariner spacecraft have handled the problem-incorporating thermostatically controlled louvers on the highly emissive surfaces such that temperature excursions from the nominal level are minimized.

From this analysis, it is concluded that bus temperatures of an inner planet orbiter spacecraft can be made essentially thermally independent of the planet as well as the Sun. Consequently, comfortable bus temperatures can be attained (in the same manner as currently used) by controlling the amount of heat radiating from the bus relative to the amount of heat generated within the bus.

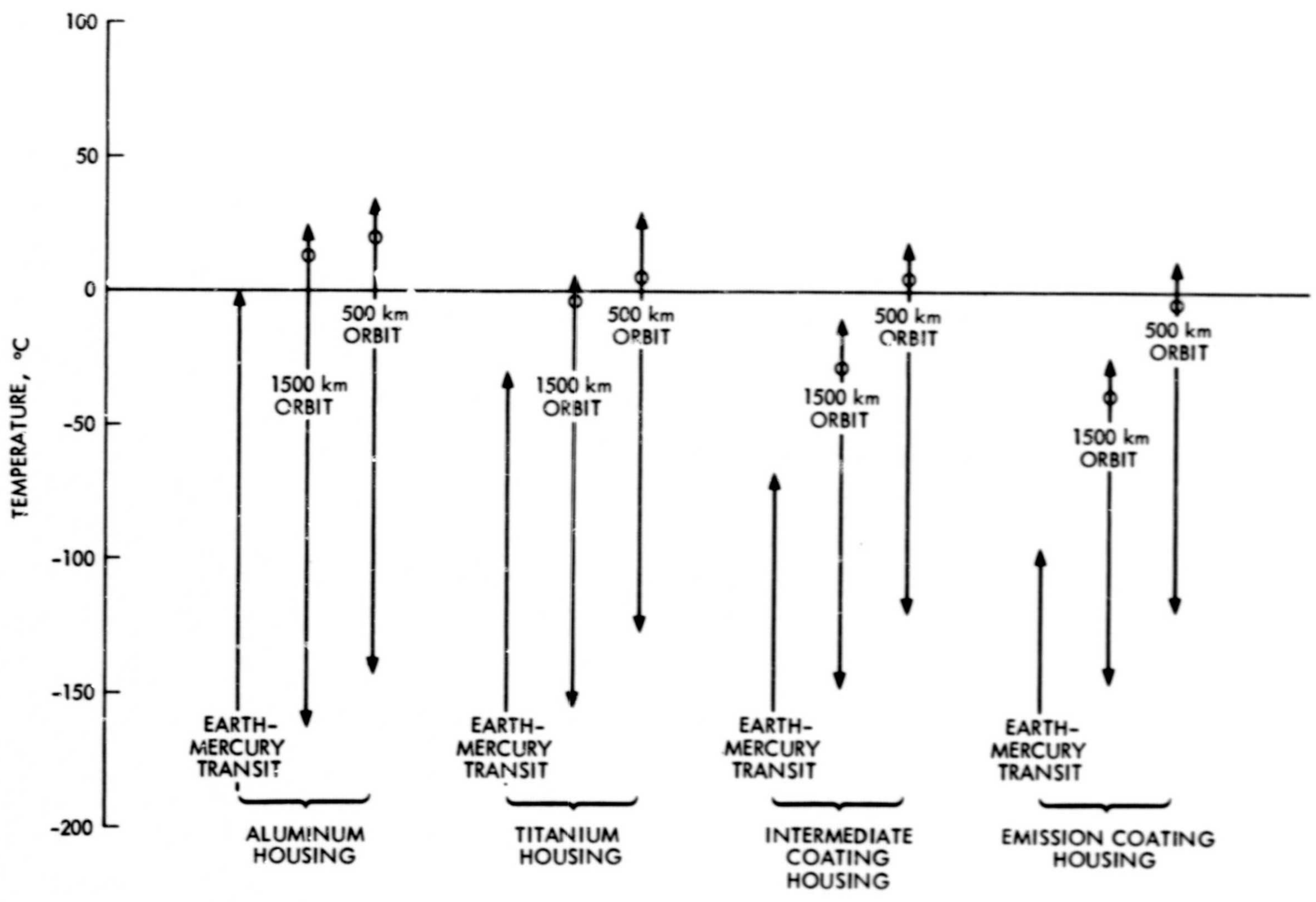
APPENDIX II



NOTES:

1. INSTRUMENT ANALYZED CONSISTED OF A CUBE CONTAINING 0.5 FT^2 TO A SIDE, AND CONFIGURED WITH A SECOND SURFACE MIRROR ON THE SUNLOCKED FACE
2. THE PARAMETERS EXAMINED INCLUDED THE INSTRUMENT SURFACE MATERIAL COMPOSITION AND THE ORBITAL HEIGHT ABOVE THE PLANET
3. ⊖ IDENTIFIES THE STEADY STATE TEMPERATURES ATTAINED BY THE INSTRUMENT DURING A POLAR TERMINATOR ORBIT

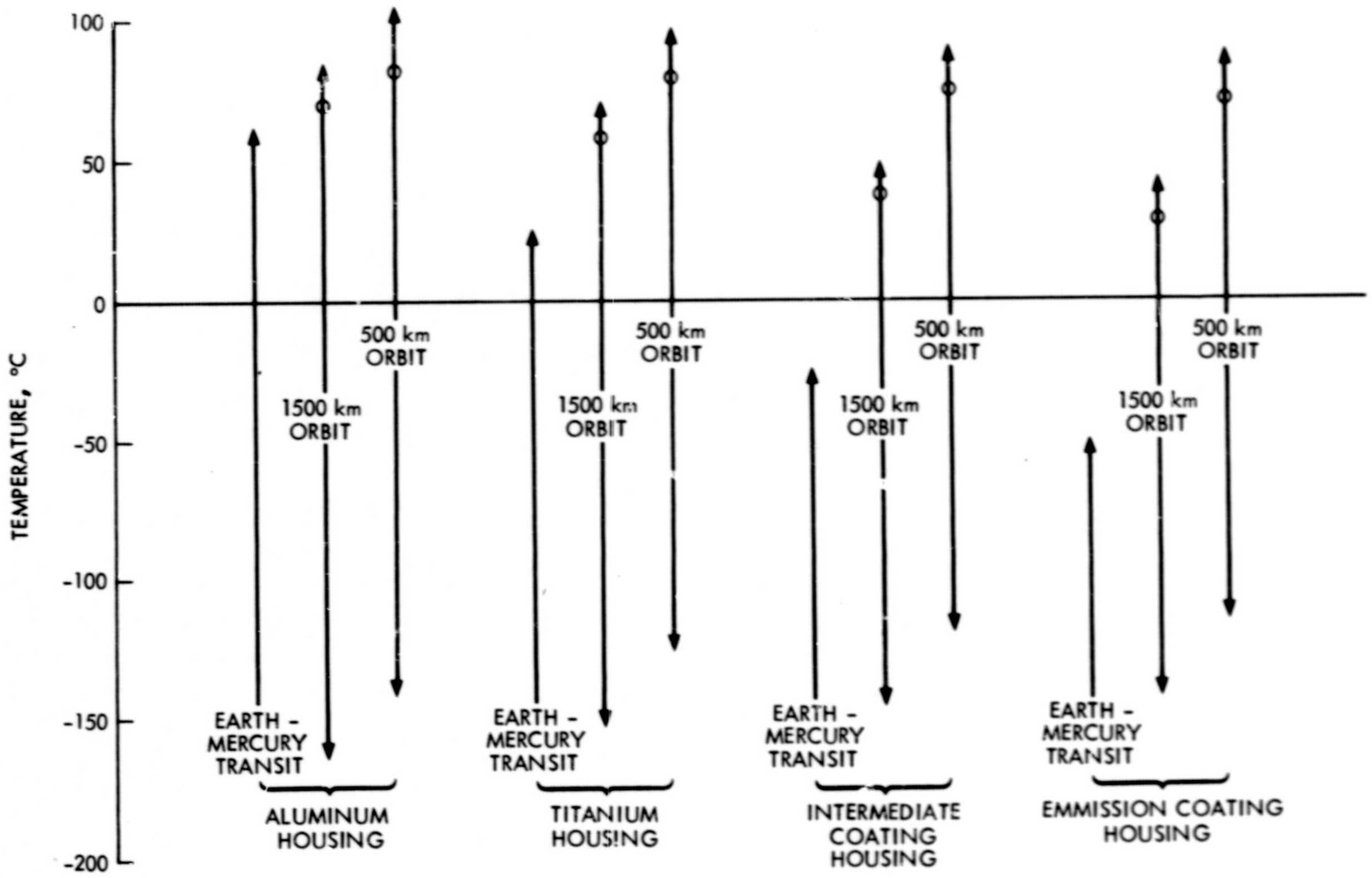
Figure 1. Temperature Extremes and Fluctuations Associated with a Cubical Venus Orbiter Spacecraft Instrument, Dissipating No Power Internally



NOTES:

1. INSTRUMENT ANALYZED CONSISTED OF A CUBE CONTAINING 0.5 FT^2 TO A SIDE, AND CONFIGURED WITH A SECOND SURFACE MIRROR ON THE SUNLOCKED FACE
2. THE PARAMETERS EXAMINED INCLUDE THE INSTRUMENT SURFACE MATERIAL COMPOSITION AND THE ORBITAL HEIGHT ABOVE THE PLANET.
3. ⊙ IDENTIFIES THE STEADY STATE TEMPERATURES ATTAINED BY THE INSTRUMENT DURING A POLAR TERMINATOR ORBIT

Figure 2. Temperature Extremes and Fluctuations Associated with a Cubical Mercury Orbiter Instrument, Dissipating No Power Internally (Mercury at Aphelion)



NOTES:

1. INSTRUMENT ANALYZED CONTAINED 0.5 FT^2 TO A SIDE, AND WAS PHYSICALLY CONFIGURED WITH A SECOND SURFACE MIRROR ON THE SUN LOCKED FACE
2. THE PARAMETERS EXAMINED INCLUDED THE INSTRUMENT SURFACE MATERIAL COMPOSITION, AND THE ORBITAL HEIGHT OF THE INSTRUMENT ABOVE THE PLANET.
3. ⊖ IDENTIFIES THE STEADY STATE TEMPERATURES ATTAINED BY THE INSTRUMENT DURING A POLAR TERMINAL ORBIT

Figure 3. Temperature Extremes and Fluctuations Associated with a Cubical Mercury Orbiter Spacecraft Instrument, Dissipating No Power Internally (Mercury at Perihelion)