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EXTREME TEMPERATURE REQUIREMENTS FOR SPACECRAFT ELECTRONICS PARTS

FY 69 REPORT

THE LOW TEMPERATURE CASE

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I. SUMMARY

The Extreme Temperature Requirements (ETR) study purpose is the determination of the extremes of temperature which will be experienced by electronics components on future spacecraft. This information will be utilized 1.) to evaluate the capability of existing parts and 2.) as design criteria for new piece parts. The case considered during FY 69 was "Thermoelectric Outer Planets Spacecraft Advanced System Technology, Document TOPS-2-100," a spacecraft planned for use in the Grand Tour Mission of the Outer Planets. This was chosen because it represented the low-temperature extreme for any planetary flyby mission planned in this century.

Preliminary studies indicated that temperature extremes would be encountered at locations remote from the main spacecraft bus, i.e., the science experiments and attitude control sensors. The types of electronic parts contained in units at these remote locations were identified. Thermal studies were performed to determine how severe the thermal environments would be and to help evaluate the constraints imposed on the mission by not having parts with an extremely low-temperature (-200°C) survival capability.

II. CONCLUSIONS AND RECOMMENDATIONS

Development of special low temperature electronic parts is not necessary to insure a successful TOPS mission. The mission can be performed with currently existing parts types, but the price that must be paid for this is: increased power and weight requirements for thermal control of scientific experiments, tighter thermal design, and perhaps some constraints on mission operations.

These constraints could be eased considerably by use of electronics parts types capable of surviving low temperatures of -100 to -125°C (about 50°C lower than the quoted capability of most current parts). With such a low temperature capability for all electronic parts used at sensor locations, the power required for thermal control of non-bus hardware could be reduced almost 50%, saving about 10 watts of power and 5 pounds of weight.

If one chose to develop new parts types rather than pay the price of thermal control power and weight requirements, attempts should first be made to qualify existing parts types for operation at lower temperature levels. Perhaps the problem is not really development but understanding existing capabilities. The capability evaluations should include expected life effects as well as expected reliability at low temperatures.

The minimum electronics likely to be mounted at a non-bus location would consist of a detector element and pre-amplifier (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 temperatures proves fruitless, development of new parts should begin with these types.

III. APPROACH

The Extreme Temperature Requirements study is being performed in several phases. The first phase (concluded last year) dealt with possible methods of determining what temperature extremes were 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 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) as:

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 closer distances from the surface.

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

In Phase I, it was shown that the spacecraft bus* could be thermally decoupled from the solar environment for flyby missions. Once the bus is successfully decoupled, a reasonable amount (normally less than 4% 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 the following tasks:

A. PARTS IDENTIFICATION.

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

B. THERMAL STUDIES.

This 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 estimates of the weight and power required for thermal control of each subsystem and the temperatures which would be reached if no thermal control were used.

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?

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* "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, and is usually thermally controlled as a unit.

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?

The first mission selected to provide answers to these questions was JPL's Thermoelectric Outer Planets Spacecraft (TOPS). 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 the limiting cold-temperature case for any flyby missions planned for this century.

IV. TOPS MISSION DESCRIPTION

The TOPS mission being studied at JPL is planned to demonstrate the capability to perform missions to the outer planets (specifically the Grand Tour) and to develop understanding of system capabilities and new spacecraft technology necessary for this type mission. The Grand Tour mission will fly by Jupiter, Saturn (inside or outside ring passage), Uranus and Neptune, utilizing a gravity assist technique made possible by the 1976 to 1979 launch opportunity.

The outcome of the TOPS will be the design of a Radioisotope Thermoelectric Generator (RTG) powered, ballistic spacecraft capable of initiating the exploration of the outer planets. Such a spacecraft must be able to operate at great distances from the sun (1 to 30 AU) and for missions lasting an order of magnitude longer than any previously attempted. The first condition dictates the use of an RTG power supply, while the second imposes special reliability needs on spacecraft parts. A complete statement of TOPS Mission Objectives and Design Criteria is included in the Appendix.

In addition to The Grand Tour being the most likely extreme low temperature mission planned, it was believed likely that information of the type required by this study would be readily available from the current TOPS study effort. In actual practice, things turned out differently. At the beginning of this fiscal year, the TOPS mission study was "stretched out" to conserve funds, which meant that not as much information was available in this time period as previously expected. For example, the configuration of the spacecraft, which had been expected to be set by January 1, 1969 will not be finally decided upon until June at the earliest.

This had a significant effect on ETR efforts since the items which were expected to experience temperature extremes were the science instruments located external to the bus. Instruments such as these are generally among the last spacecraft items to have their designs completed. Thermal control for external items is not generally considered until after the bus is under control. After the initial thermal design efforts indicated that temperatures inside the bus could be maintained within a rather nominal range (40 to 80°F), it became evident that temperature extremes, as expected, would be confined to non-bus hardware.

However, the available information about the specific TOPS instrument configurations did not contain enough detail for a rigorous thermal analysis. This was as expected since design of experiments, in particular, is one of the areas which pushes both the state of the art and the project schedules to their limits. Thus, with the variety of likely approaches to measuring scientific knowledge, detailed instrument designs could not be expected this far in advance of the launch date.

What was possible, however, was to assume as a basis the types of instruments used on past missions. This approach is realistic since the types of functions which parts must perform in the planned science objectives for TOPS are very similar to the existing functions performed on past planetary missions. Thus, the conclusions reached represent a reasonable estimate of temperature and parts capability for low temperature missions.

V. PARTS IDENTIFICATION

Although no definite configuration has been chosen for the TOPS spacecraft, those items not contained inside the bus could be readily identified. Most of these are science experiments which require viewing the phenomena being observed and are, therefore, mounted on the scan platform or on a boom. The only non-science electronics which would be directly exposed to the environment were portions of the attitude control system. The Canopus sensor was included in this grouping because possible configurations showed it located outside the bus. (On all previous missions, this sensor has been thermally coupled to the bus.)

The cognizant JPL Technical Divisions prepared lists of electronic parts types likely to be used on the Grand Tour mission for each of the non-bus subsystems. The parts lists also contained estimates of the temperature capability of each unit. Tables 1 to 3 are a summary of the inputs received from the Technical Divisions. Basically, each of these units can be considered to consist of a sensing element or detector, a signal amplifier, and electronics to convert the signal into a data format suitable for transmitting information about the observed event back to earth. The detector is the one part of the unit which must be exposed to the ambient environment in order to obtain data. The rest of the items may be located at the same physical location as the detector, or may be partially included in the bus hardware. The minimum amount of electronics likely to be exposed to the environment might consist of the sensor element and a pre-amplifier to carry the signal to the bus, where the rest of the data-handling could take place.

Electronics problems on preceding interplanetary probes have been mainly concerned with exceeding the upper temperature limits. For the TOPS mission, the priorities are reversed, with low temperature requirements becoming more important. Some of the electronics items considered (such as photomultiplier tubes) actually perform better at temperatures considerably lower than present operating levels, and therefore, would not be affected by low temperature extremes.

Table 1. Parts Type Breakdown - Science Experiments

(All Parts have Temperature Capability of -55°C to +125°C, Except as Noted.)

Instrument	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/Transformer	Crystals (Quartz)	Bipolar/MOS ICs	Bipolar/MOS MSI ICs	Bipolar/MOS LSI ICs	Silicon Solid-State Radiation Detectors (4)	Lithium Cores (5)
TV	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X		
IR Radiometer	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X		
UV Photometer	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X		
Cosmic Ray & EP Detector	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	
Solar Plasma Probe	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	
Micro-Meteoroid Detector	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X		X
D. C. Magnetometer	X	X	X	X	X	X	X	X	X	X	X		X	X	X		
A. C. Magnetometer	X	X	X	X	X	X	X	X	X	X	X		X	X	X		
Pre-Amps (2)		X	X		X (3)	X		X			X						

Notes:

(1) Includes Transistors, Diodes, FETs, Photodiodes, GaAs Diodes, Zeners, Etc.

(2) Minimum Electronics Likely to be Located at Detector, on Boom, Etc.

(3) Also Wire-Wound Resistors

(4) Temperature Limits are -40°C to +50°C

(5) Temperature Limits are -20°C to +90°C

Table 2. Potential Science Detectors for Outer Planet Missions

(Suggested by JPL Space Science Division as Most Likely Current Candidates for Future Mission Use)

Detector	Current Estimation of Temperature Capability (°C)		Possible Instruments
	Lower	Upper	
Image/Sound/Orthocon	-40	+20	TV
Germanium Bolometer	-150	+50	IR Radiometer
Lead Sulfide Cell	-250	+50	Planet Sensor, Approach Guidance
Doped Detectors (for IR Work in 8-14 μ Range and Beyond)	-50	+50	IR Radiometer
Photomultipliers with Si, SiO_2, etc., surfaces	-100	+15	Optical Sensors, Radiation Instruments, TV
Magnetometers (Ferrite core, Helium Cells)	-50	+100	DC & AC Magnetometers
Solid State Particle Detectors (Si, Ge)	-40	+50	Cosmic Ray, Solar Plasma, Trapped Radiation, Low Energy Radiation
Plasma Detectors (Curved Plate Spectrometer, Faraday Cup)	-100	+100	Solar Plasma Probe
Electromagnetic Antenna	-150	+100	Radio Analysis, Ionization Analysis of Planetary Atmosphere
Electrometers (vibrating Reed)	-50	+75	UV, IR Radiometer/Spectrometer Magnetometer

Table 2. Potential Science Detectors for Outer Planet Missions (Continued)

Detector	Current Estimation of Temperature Capability (°C)		Possible Instruments
	Lower	Upper	
Electron Multipliers			
(a) Multistage secondary Electron Simulator	-100	+20	UV, Mass Spectrometer
(b) Secondary electron conductor	-100	+20	UV, Mass Spectrometer
(c) Electrostatic and Electromagnetic Chance Multiplier			
Selenium Compound Vidicons	-40	+20	TV
Silicon Sensors	-65	-10	TV
Photo Cathode Devices	-65	20	TV

Table 3. Parts Type Breakdown - Attitude Control System

Unit (Operating Temperature Limits, in Parenthesis)	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	Other
Cruise Sun Sensor (-45°C to +65°C)	X[4]	X[2]		X[3]		X[1]	X									
Acquisition Sun Sensor and Sun Gate (-45°C to +65°C)		X[2]		X[3]												X[5]
Digital Position Encoder (Optical) (-55°C to +85°C)						X										X[6]
Approach Guidance Tracker - Optics and Vidicon Assembly (-20°C to +30°C)	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
Canopus Tracker (-15°C to +75°C)		X[2]		X	X	X	X		X	X	X					

Notes:

(1) Includes Transistors, Diodes, FETs, Photodiodes, GaAs Diodes, Zeners, Thermistors

(2) Operational Amplifier

(3) Also Wire-Wound Precision Resistors

(4) Resistor

(5) CdS Detector

(6) GaAs Light Source and Silicon Detector

Most of the electronics parts likely to be used on the science experiments (aside from the detector element) have a temperature capability within limits of -55°C to 125°C corresponding to the temperature limits in MIL-E-5400lC and MIL-E-8189C. One exception is the lithium cores used in the science data system. These operate only in the range -20°C to +90°C, but this is not a significant constraint since the cores would be installed in the bus where a more favorable temperature environment exists. There is a possibility of using lithium cores for memory on a micrometeoroid detector if the expected impact rate while crossing the asteroid belt would be too great for the science bit rate, but again, these could be installed in the bus.

In general, temperature considerations will play a large part in determining the type of detector element to be used in a particular experiment. (The same holds true, but to a lesser degree, for electronics used in pre-amps at the detector location.) The detector elements with temperature problems seem to be the silicon solid state radiation detectors, (which can only operate between limits of -40°C to +50°C) and selenium compound vidicons (-40°C to +20°C). In both cases, alternate detectors are available for the experiments likely to use such parts.

The attitude control sensors use parts types similar to those contained in the science units. For example, the approach guidance tracker has what is essentially a TV camera as its sensor with electronics located in the bus. The 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.

A. BUS HARDWARE PARTS CAPABILITY

Even though a nominal temperature environment is predicted for the bus electronics, it was felt that some investigation should be made of the temperature capability of bus electronics parts, in the event that some of these might be used on sensors. (This would also provide an independent check on the inputs received for the sensor electronics.)

The JPL Electronic Parts Engineering Section has prepared a list of candidates for the TOPS Recommended Parts list. Parts selected for this list were chosen mainly on the basis of susceptibility to radiation damage. (This is a significant TOPS constraint, not within the scope of this study, since parts may be subjected to heavy radiation bombardment from the RTG and the radiation environment around Jupiter.) The appropriate parts specialists were contacted to determine what they considered to be the current temperature limits of the parts types on the list. This was taken to mean the currently specified test levels used by JPL in qualification of individual parts types. The following summarizes the information received on each parts type:

1. Capacitors are generally rated for the temperature range -55°C to +125°C. Some may need to be derated to a high temperature of +85°C in order to extend lifetime. Ceramic capacitors have a slightly increased high temperature capability of +150°C. When Surveyor capacitors were tested to determine lunar night survival capability, some defects were revealed in all capacitors after exposure to a -125°C temperature.

2. Resistors are currently rated for the -65°C to +125°C temperature range. Some metal film resistors have survived exposures to -185°C.

3. Diodes, Silicon Controlled Rectifiers, and Silicon Controlled Switches are all rated for a -55°C to +150°C temperature range.

4. Transistors have a survival capability of -198°C to +200°C. The operational characteristics depend on the type used. Bipolar transistors can be operated in the -65°C to +200°C range if proper power derating at the high end is used. (A few types cannot be used at higher than 175°C, but can be stored at 200°C.) Field Effect Transistors (FET's) have an operating range of -100°C to +200°C (with power derating at the high end).

5. Some Relays have been qualified at the sterilization temperature of 145°C, but most are rated for the -65°C to +125°C temperature range.

6. The only Switch recommended for TOPS is qualified for the -184°C to +260°C temperature range; therefore, no temperature difficulties are expected, especially since most of its functions are done at separation.

7. Fuses are specially produced for JPL and qualified for the -55°C to +85°C temperature range.

8. Crystals are rated from -55°C to the sterilization temperature of +145°C.

9. Photocells are rated at -50°C to +175°C, but their electrical characteristics change drastically with temperature and may require temperature control within narrow limits (or temperature compensated circuits). For example, a cell which is normally operated at 25°C may have a conductance of 114% this value at -25°C and 37% at 75°C.

If groups of these parts are formed corresponding to the low-end qualification temperature, we get the results shown in Table 4. Obviously, few parts are presently qualified at temperatures lower than -55°C.

Since previous missions have required more attention to the upper qualification temperature limits, it is felt that the low temperature limits are more a reflection of current MIL-Specification performance guaranteed by the manufacturer 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, but 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, what is needed is not necessarily development of new electronics parts, but qualification of present parts for lower temperature operation.

Table 4. Low-End Qualification Temperature Results

Low Qualification Temperature	Parts Types
-184°C	Switches, special Metal Film Resistors
-100°C	FETs
- 65°C	Resistors, Bipolar Transistors, Relays
- 55°C	Capacitors, Diodes, Silicon Controlled Rectifiers, Silicon Controlled Switches, Fuses, Crystals
- 50°C	Photocells

VI. THERMAL STUDIES

Tables 5 and 6 show the results of an initial thermal study on the TOPS sensors. For this study, the method of thermal control was assumed similar to that on MM'69. Resistance heaters wired in parallel to each instrument were used to dissipate enough power when the unit was not operating to keep the temperatures within the operating range. Therefore, the power required for thermal control, as listed in Table 5, is the specified operating power for the instrument and represents the estimated maximum "replacement" thermal control power available. The temperature limits listed are those specified by the project engineers for each unit based on current technology.

This type thermal control can have its effectiveness compromised if the cruise science experiments are turned off to permit the encounter science to operate near each planet. In this case, the heater power required for the non-operating units, when added to the power required to operate the planetary instruments, may cause a power deficit. If the RTG is sized for operation of both science packages simultaneously, power is wasted for most of the mission. Or if no cruise science is placed on board because of power requirements, a unique scientific opportunity would be missed; therefore, the amount of heater power required for the mission should be minimized for an optimum mission.

Table 6 lists the temperatures reached by each instrument if no heater power is applied. In short, these tables indicate that thermal control is possible for the sensors, and it is felt that the operating temperatures listed can be achieved with proper thermal design.

The specified low-end operating temperature of some instruments may cause problems during environmental testing. Typically, the thermal control system is sized to just maintain the specified operational levels, with little margin for unexpected deviation in predicted temperatures. With a mission lasting as long as the Grand Tour, test philosophy must express a large degree of conservatism to ensure reliable operation of all spacecraft subsystems. Therefore, test levels must be significantly more severe than the expected environment. When the appropriate margins for thermal-vacuum

Table 5. TOPS Non-Bus Hardware (Sensors)

	Power Required For Temperature Control (Watts)	Temperature Limits In Centigrade Degrees	
		Operating	Storage
Science			
A. Encounter On Scan Platform			
(1) Television	9.0	-60 to +25	-60 to +35
(1) IR Radiometer	5.0	-40 to +25	-40 to +25
(1) UV Photometer	1.0	-30 to +25	-30 to +35
B. Cruise			
(2) Cosmic Ray and EP Experiments	4.0	-30 to +40	-30 to +50
(1) Solar Plasma Probe	8.0	-25 to +35	-25 to +35
Trapped Radiation Detector	0.5	-20 to +30	-30 to +40
Micro-Meteoroid Detector	1.0	-30 to +25	-30 to +35
(3) DC Magnetometers	3.0	-45 to +65	-45 to +70
(3) AC Magnetometer	2.0	-50 to +50	-50 to +50
Non-Science Sensors			
Primary Sun Sensors	3.8	-45 to +65	-50 to +70
Secondary Sun Sensors	3.8	-45 to +65	-50 to +70
Canopus Sensor*	4.0	+ 4 to +21	-30 to +50
Approach Guidance Tracker	5.0	-60 to +25	-60 to +35

*Bus Dependent

Table 6. TOPS Non-Bus Hardware (Sensors)

	Temperature Range With No Thermal Control Power In Degrees Centigrade At Various Planets				
	Earth	Jupiter	Saturn	Uranus	Neptune
Science					
A. Encounter	+7 to +60	-150 to -125	-185 to -165	-210 to -197	-222 to -212
(1) Television					
(1) IR Radiometer					
(1) UV Photometer					
B. Cruise					
(2) Cosmic Ray and EP Experiments					
(1) Solar Plasma Probe					
Trapped Radiation Detector					
Micro-Meteoroid Detector					
(3) DC Magnetometers					
(3) AC Magnetometer					
Non-Science Sensors					
Primary Sun Sensors					
Secondary Sun Sensors					
Canopus Sensor*					
Approach Guidance Tracker					

*Not Bus Dependent

testing are added onto the listed operational temperatures, the resulting low-end temperature will exceed the current low-temperature capability of most electronic parts types. Therefore, some of the instruments may require a higher low-end temperature than shown, which would require additional thermal control weight and power.

For this reason, an examination of the power vs. temperature characteristics of some of the science experiments assumed to be flown on TOPS was performed. The Infrared Radiometer and TV Camera B were selected for these studies. Both of these instruments are presently being flown on Mariner '69, and similar type instruments are presently being considered for use on TOPS. The parameters required for thermal studies of this type are size, dimensions, weight, power, construction methods, aperature selection, electronics, etc. After consultation with the Space Science Division Representative, it was decided to use the Mariner '69 instruments as a basis for the study, since no closer approximation to the TOPS configuration could be found and weight and power requirements were similar for the instruments on both missions. By making certain assumptions and using Mariner '69 flight data, it was possible to estimate instrument temperatures at various heater power levels for the extreme outer planet of the TOPS mission.

The following assumptions were made:

1. The Mariner '69 Infrared Radiometer and TV Camera B would comprise a good cross section of the type of electronic parts to be used on the TOPS science instruments.

2. The science instruments will be mounted on a boom or platform so that some portion of each instrument will be illuminated by solar energy.

3. The Mariner '69 science instruments were beneath the spacecraft bus so that solar energy did not add an additional heat load to the instruments.

4. The Mariner '69 Infrared Radiometer consisted of a rectangular box having sides 3.5" x 7" x 7". Two lens apertures exist. One is located on the 3.5" x 7" side and the other on the 7" x 7" side. Each aperture has an area of 2 inches square and an

effective infrared emittance of 0.9. Normal operating power is 2.5 to 3 watts. Temperature control power is 2.0 watts and is used only when the instrument is not operating. Flight data showed the instrument to have an equilibrium temperature of -13.33°C under nominal operating power near earth.

5. The Mariner '69 TV Camera B consists of a rectangular box having sides 10" x 10" x 20". A lens aperture exists on the side having the 10" x 10" dimensions. The aperture has a diameter of 10" and a special temperature control coating so that the effective infrared emissivity is 0.1. Normal operating power is 4.5 watts. Temperature control power consists of a continuous 5.5 watts in the optics and 3.5 watts replacement power in the Vidicon.

A. RESULTS AND DISCUSSION

For the IR Radiometer, the results of this study are shown in Figure 1. The ordinate is the bulk or average temperature of the instrument. The instrument was assumed to be a lumped mass node. In reality, this is not true, and certain components of the instrument will have temperatures which exceed this value and some whose true temperatures are less than this value. Such temperature deviations are real and are brought about by variations in electrical power dissipation densities within the instrument. To estimate the corrections of the numbers furnished by this study would mean that a detailed thermal analysis be made for the instrument. From such an analysis, each component temperature would be determined and an error analysis made. Then, a statistical deviation or variance could be determined using as a base the temperature or temperatures given by this study. With that information, a comparison could be made between the approximate method of analysis used in this study and the actual situation. However, it is believed that the temperatures of this study (using a lumped mass node) would not vary by more than 15% from the true average temperature determined by the more exact method.

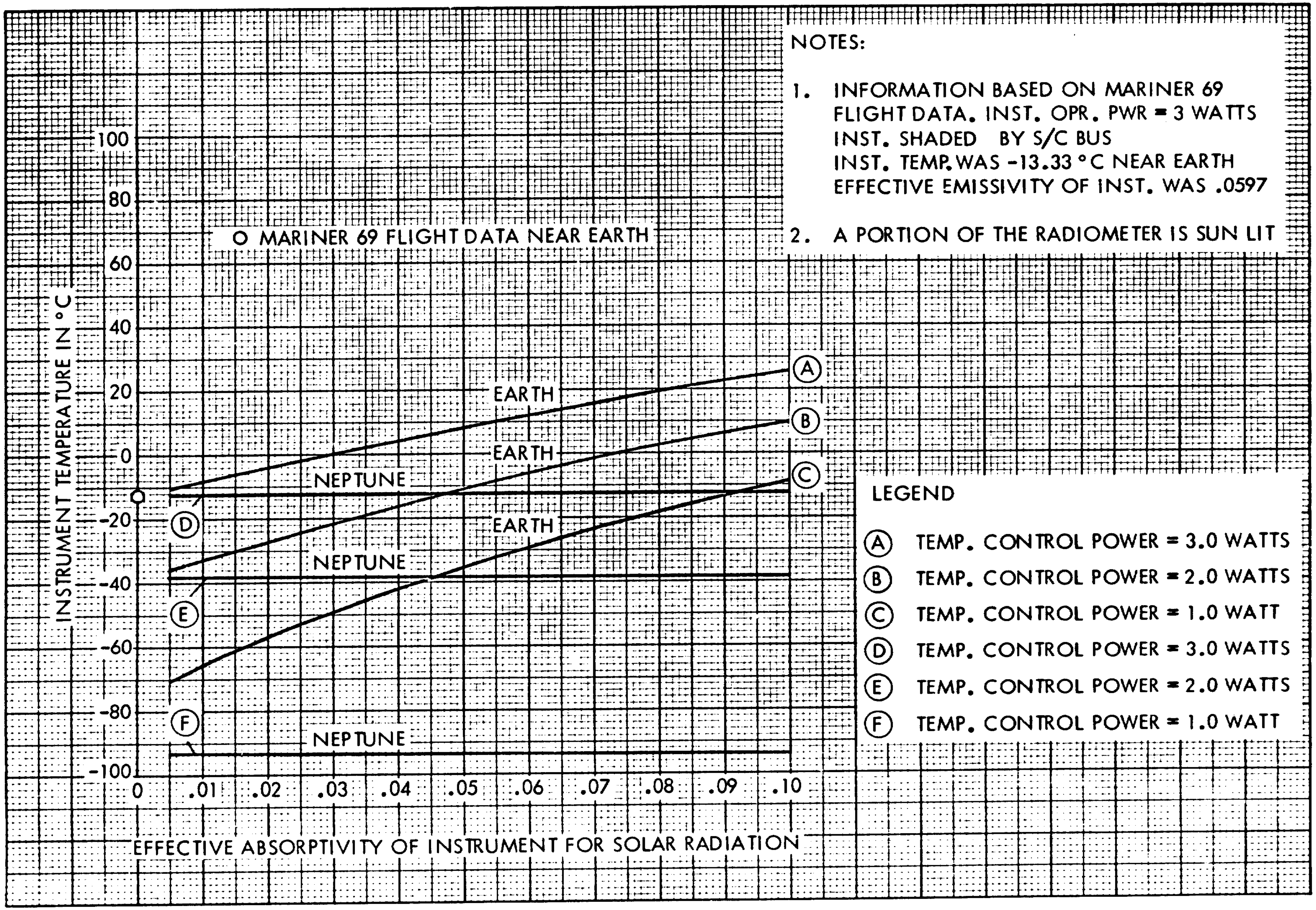

Figure 1. Estimated Infrared Radiometer Temperatures at Earth and Neptune as a Function of Effective Absorptivity and Temperature Control Power

The curves in Figure 1 show the effect of varying the temperature control power and controlling the effective absorptivity of the instrument in terms of instrument temperature. Curves A and D show that when the temperature control power is set at 3 watts (the nominal instrument operating power), and the instrument is shaded by the bus (or its effective abso ptivity for solar radiation is zero), the instrument temperature will be -13.33°C whether the instrument is in a near earth position or at Neptune. Since a portion of the instrument will be sunlit, some solar energy will be absorbed. As the effective absorptivity is increased, the temperature of the instrument will increase at earth but not at the outer planets. This apparent inconsistency is caused by the outer planets solar constant being so small that the absorbed solar heat load is insignificant when compared to the internal heat load (the solar constant at Jupiter is 4% of the value at Earth). It may be noted that instrument temperatures near earth are quite sensitive to the effective absorptivity of the instrument to solar radiation. It follows that the area of the instrument that is sunlit may have increased insulation to reduce the effective absorptivity of the instrument thereby isolating the instrument from the sun. This philosophy reduces the temperature changes that an instrument would normally experience during a TOPS mission. On the other hand, excessive use of insulation will reduce the instrument's ability to reject its internal heat generation caused by normal operating power.

Therefore, the thermal design of an instrument must be carefully detailed so that it will:

1. Not over-heat under normal operating power.
2. Be isolated from the sun as much as possible.
3. Have sufficient insulation so that a minimum of temperature control power will keep the instrument temperature above a defined survival temperature.

Based upon the concept of minimum temperature control power, it is now of interest to investigate the remaining curves of Figure 1. It may be seen that Curves B and E and Curves C and F are for temperature control power levels of 2 watts and 1 watt respectively for near Earth and Neptune positions. The present survival temperature limits for the Infrared Radiometer is -40°C. This temperature suggests that a minimum of 2 watts temperature control power be available for this instrument for the TOPS mission. However, if only 1 watt of temperature control power were available, the instrument temperature would be -94°C at Neptune which is far below the present day survival temperature limit of -40°C.

Just how much power will be available for temperature control of a TOPS spacecraft is still undecided. At present, the design team has allocated approximately 25 watts which may or may not be available continuously throughout the mission. Temperature control has estimated the power requirements to be between 35 and 45 watts depending upon the particular spacecraft and the particular phase of the mission. The additional power must be made up somehow, but this will be expensive, both in weight and power. One approach could be to use larger RTG's, but with the design of most spacecraft, as the schedule closes in on the flight date, power requirements increase rather than decrease. Another alternative would be a better thermal design of the entire spacecraft (of which the science instruments are only a portion). But even so, experimenters prefer individual mounting of their instruments to permit easier removal for calibration or performance checks. For individually mounted instruments, structural dynamics normally requires four through bolts at each corner which serve as a major source for heat losses. Other late design changes (such as substituting copper core wiring harnesses for iron-core wiring to obtain a better signal-to-noise ratio) may serve to negate most of the benefits of a tight thermal design. (In this case, increased conduction heat losses would increase, requiring more heater power.)

Since the TOPS spacecraft is starting out as power-starved (from a temperature control point of view) and since power requirements usually increase rather than decrease, there are definite advantages to having electronics that can survive and operate at temperatures below the present day value of -40°C.

One might ask how far below -40°C should the electronics be capable of surviving in order to reduce the magnitude of temperature control power requirements. No definite answer to this question is available. The proposed TV Camera B temperatures are shown in Figure 2, using the same approach as was taken with the Infrared Radiometer. Here it can be seen

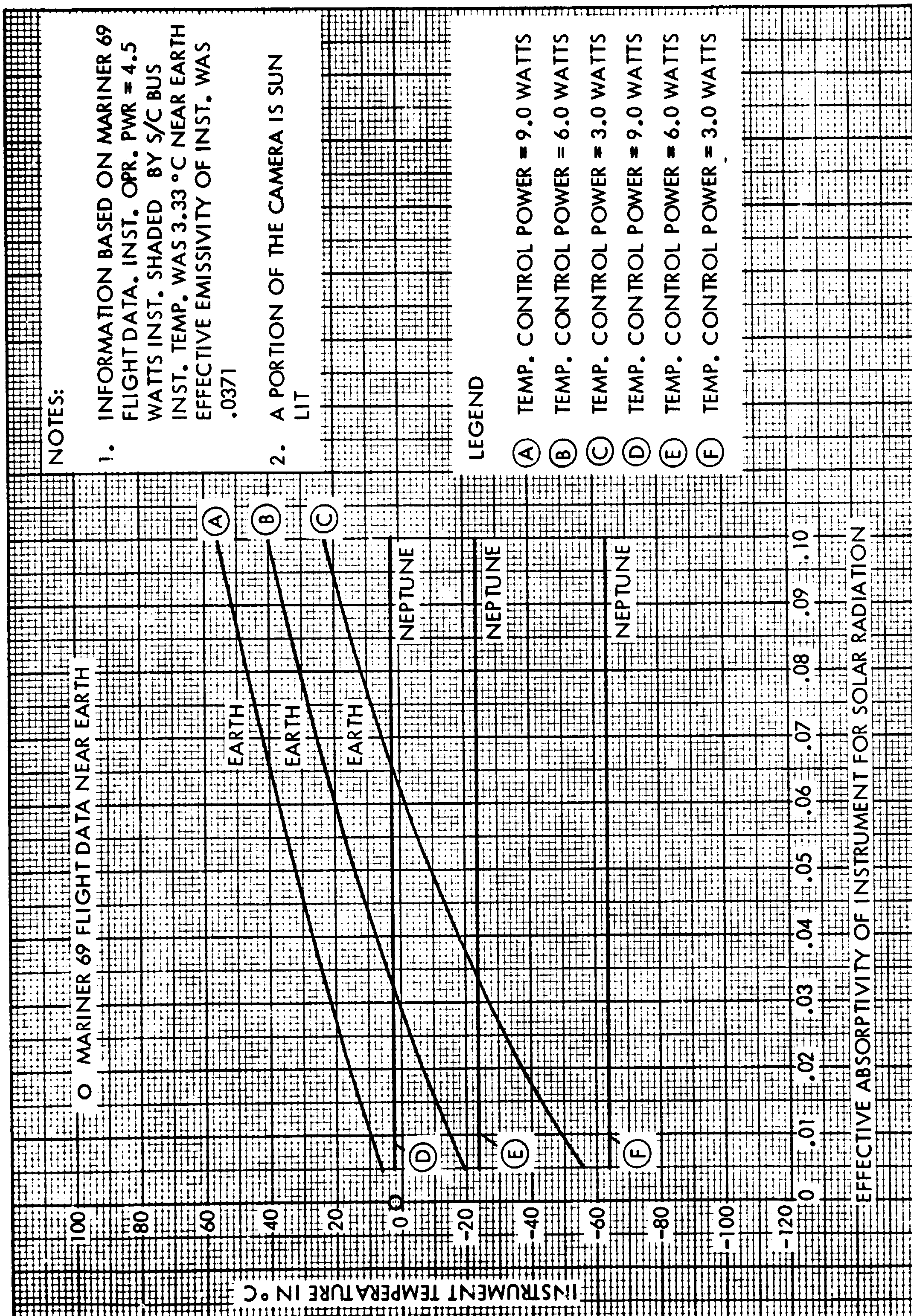

Figure 2. Estimated TV Camera "B" Temperatures at Earth and Neptune as a Function of Effective Instrument Absorptivity and Temperature Control Power

that if electronics could be made or qualified to survive at -65°C rather than -40°C, the TOPS Camera "B" temperature control power can be reduced from 5 watts to approximately 3.0 watts. Such a reduction, if carried straight across the board, would amount to about 8.0 watts for the TOPS program (since there are two sections of two cameras). It may be recalled that the Mariner TV Camera B had a continuous 5.5 watts of TC power and an additional 3.5 watts of TC power when the camera was turned off. This latter power is needed to prevent thermal distortion of the optics. It has been estimated that by selecting better optics for the TOPS camera the thermal control power requirement will be about 5 watts with present day survival temperature limits. In addition, on TOPS there may be a period of 3 to 5 minutes during midcourse motor burn when all temperature control power will be shut off. During this period the camera must depend upon its heat storage capacity to maintain its temperature until temperature control power is resumed.

APPENDIX

The following Thermoelectric Outer Planet Spacecraft Advanced System Technology Mission Objectives and Design Criteria data is provided as a supplement to this document.

Project Manager ______

Spacecraft System Engineer ______

JET PROPULSION LABORATORY

No. TOPS-2-100
17 February 1969

Thermoelectric Outer Planet Spacecraft

Advanced System Technology

MISSION OBJECTIVES AND DESIGN CRITERIA

1.0 INTRODUCTION

1.1 The purpose of this document is to define the criteria to be applied to the design and development of the Thermoelectric Outer Planet Spacecraft - Advanced System Technology (TOPS) Spacecraft System.

2.0 PURPOSE

2.1 To demonstrate the capability to perform missions to the outer planets, specifically the Grand Tour type mission.

2.2 To develop understanding of the necessary system capabilities for this class of mission.

2.3 To provide design and development experience in several new technologies critical to this type of mission.

2.4 To develop an understanding of the required subsystems and their interactions so that realistic performance, reliability, schedule, and cost estimates can be made.

2.5 To emphasize new spacecraft technology.

2.6 The TOPS project will use the Grand Tour mission (Jupiter, inside ring passage at Saturn, Uranus, and Neptune; each with gravity assist) flight project launch in 1977 for design purposes. The design of the system for this mission is intended to provide the basic technical capability for all Jupiter gravity assist outer planet missions in the late 1970's.

3.0 MISSION OBJECTIVES

3.1 Grand Tour Mission Objectives

3.1.1 <u>Primary Objective</u>

To initiate the exploration of the outer planets (Jupiter, Saturn, Uranus, and Neptune) by conducting flyby missions of all of these planets. To make exploratory investigations which will pave the way for future missions and experiments -- particularly those relevant to the understanding of the origin of the solar system.

3.1.2 <u>Secondary Objective</u>

To develop the technology necessary for the further detailed investigations of outer planet missions that require operation at great distances from the sun.

3.2 TOPS Project Objectives

3.2.1 Design a Grand Tour mission which is compatible with a Radioisotope Thermoelectric Generator (RTG) powered ballistic spacecraft (S/C) for the 1976-1979 opportunity.

3.2.2 Understand the technology requirements unique to these missions.

3.2.3 Develop a spacecraft system design concept which emphasizes the long life, environment immunity, and emergency adaptability characteristics required for such missions. The system design is to represent a proven technology for the 1976-1979 period. To permit flight testing of such a system, this technology should be limited to that appropriate for a developmental (high risk) flight in 1974, a Jupiter fly-by, intended as a Grand Tour precursor.

3.2.4 Develop techniques for assuring compatibility between the RTG radiation fields and the remainder of the mission systems and subsystems; investigate interactions between the S/C and science instruments.

3.2.5 Develop advanced designs as required; including the design of the thermoelectric generator of an RTG power subsystem based on system requirements.

3.2.6 Demonstrate, by analysis and test, the validity of design concepts that transcend previous experience or represent major technical innovations; a combination of these results and currents experience should establish feasibility of accomplishment of this mission with this type of S/C.

3.2.7 Develop an understanding of, and possible solutions for nuclear safety problems associated with an RTG powered S/C. Establish design rules to assure compatibility between launch vehicle (L/V) and S/C, so that operations can be designed to satisfy the Atomic Energy Commission (AEC), National Aeronautics and Space Administration (NASA), and Air Force Eastern Test Range (AFETR) constraints.

4.0 MISSION RESTRAINTS

4.1 At least one Titan III D/Centaur/Burner II with the standard 10-foot diameter shroud or the 14-foot diameter Viking shroud will be provided for the 1977 opportunity.

4.2 As a basis for planning, a launch period of 18 days shall be required.

4.3 The design of the spacecraft and all other variable elements of the project should be such as to provide a very high probability of successfully achieving the mission objectives with each launch.

4.4 The design of the spacecraft will be compatible with the use of a 210-foot receiving antenna at each major site (3) operating at S-band or X-band for telemetry. The design of the spacecraft will be compatible with the use of a 210-foot receiving antenna at each major site (3) operating at S-band, X-band, or both S and X-bands for normal tracking. The design of the spacecraft will be compatible with the use of a 210-foot transmitting antenna using 400 KW transmitting at each major site (3) operating at S-band for command. The design of the spacecraft will be compatible with utilizing additional ground capability to the extent of two 210's in phased array or up to 400-foot antennas with up to 2000 KW transmitters at each major site (3).

4.5 Planetary Quarantine and Contamination

4.5.1 Planet contamination constraints remain to be established. However, the spacecraft will not be sterilized.

4.5.2 In view of the long life requirement, and the number of optical instruments involved in the spacecraft operation, the spacecraft design will emphasize employment of concepts that minimize particulent contamination.

4.6 A test launch in 1974 (Grand Tour Precursor) may be provided to obtain scientific knowledge of the Astroid Belt and the Jupiter radiation environment at Grand Tour flyby distances. This flight would also provide engineering developmental and flight experience with the advanced technology hardware. This experience would be incorporated into the Grand Tour spacecraft.

4.7 All the launchs will be conducted from AFETR and as such will face certain launch azimuth and window restraints. The availability of 90° to 114° firing aximuth sector will be requested.

4.8 The primary mission objective can be considered achieved after the complete recovery of the encounter data from the last planet, Neptune

4.9 The TOPS spacecraft will be designed for a completely ballistic "gravity assist" Grand Tour mission with a launch in 1977 for a flyby at Jupiter, inside ring passage flyby at Saturn, flyby at Uranus, and flyby of Neptune. The TOPS spacecraft design will retain the capability to perform other outer planet missions with minor modifications to the basic spacecraft.

5.0 DESIGN CRITERIA

5.1 Design Approach

5.1.1 The TOPS spacecraft design approach shall be a large technology step forward, such that the future missions can use this knowledge as a means of achieving the mission objectives, increasing the system reliability,and reducing costs.

5.1.2 The spacecraft shall be designed, developed, and fabricated such that future requirements of similar missions can be met with minimum additional design development and fabrication effort at that time.

5.1.3 The spacecraft shall be automatic; that is, it shall be capable of operation from post planet manuever through pre-planet manuever without the use of ground commands.

The automatic capability shall be advanced after completing a planetary flyby to cover the period to the next planet. The extremely long communication distances make it imperative that the spacecraft be automatically adaptive to changes. The spacecraft must be capable of detecting errors and out of tolerance conditions and initiating the appropriate action.

5.1.4 Ground command capability shall be provided to backup onboard functions. It shall be used to provide the interplanetary trajectory correction maneuver parameters and to alter and adjust the flight sequence of events if such adjustments are required to correct trajectory dispersions or would improve the results over what would be obtainable with the stored flight sequence. In addition, it shall be used to modify the automatic adaptive parameters as appropriate.

5.1.5 The spacecraft design shall be such that elements can be tested in the Earth's 1 "g" environment.

5.1.6 On-pad tests and operations shall be limited to the minimum required to turn the spacecraft on, condition it for launch and verify its flight-readiness. Only system loop checks or system diagnostic tests shall be performed after the spacecraft has left the final assembly area. No provisions shall be made for component or subsystem testing from the blockhouse.

5.1.7 Weight shall be used to increase the probability of the achievement of the mission objectives (increase reliability) in preference to increasing mission capability to meet low priority objectives.

5.1.8 The spacecraft shall be instrumented, to the maximum extent practical, to obtain diagnostic telemetry in the event of failures, whether the failures are caused by system malfunction or by the encountering of environments beyond those accounted for in the design.

5.1.9 To the maximum extent possible, the design shall use technologies that appear appropriate for the 1977 Grand Tour spacecraft design. The design shall take advantage of the previous Mariner program technologies, design concepts, and philosphies wherever applicable.

5.1.10 It shall be a design goal to achieve independent backup capability for every critical event, "backup" here meaning from a functionally different source, not identical block redundancy.

5.1.11 All scientific instruments shall be designed to be as functionally independent of one another as is practical to increase the assurance that a failure in one instrument, or in equipment common to several instruments, will have a minimum affect on the total data received.

5.1.12 During the design and fabrication of the spacecraft, special effort shall be made to provide scientific instruments with an environment which will minimize their integration problems and maximize their potential scientific data gathering capability.

5.1.13 The spacecraft subsytems shall be designed such that no single part, component, or subassembly shall cause a catastrophic mission failure, but shall allow a degraded subsystem performance which in turn will at most only reduce the total mission capability.

5.1.14 The system shall be designed to survive environments at severities consistent with required probabilities of system survival.

5.1.15 The TOPS spacecraft will be fully attitude stablized using the sun and the star, Canopus, as the basic attitude references (the Earth may be used with a closed RF loop for an additional attitude reference) and containing the following elements:

a. A three-axes attitude control system utilizing momentum storage with mass expulsion.

b. Temperature control equipment.

c. Radioisotope Thermelectric Generators and power conversion equipment.

d. Two-way communication and command equipment based upon the use of low-gain antennas, a steerable medium-gain antenna, and a fixed high-gain antenna.

e. Trajectory correction maneuver equipment with the capability of making at least nine maneuvers.

f. On-board programming and sequencing equipment.

g. Planetary and interplanetary scientific instruments including scan equipment for the planetary experiments.

h. Instrumentation, instrument control, data handling, and data storage equipment.

i. Approach guidance equipment.

5.2 Reliability Criteria

5.2.1 It is recognized that one of the most difficult aspects of this mission is the reliability of operations during the long flight to Neptune. The spacecraft design shall be based on a lifetime requirement of 12 years for expendables and a high probability of surviving 12 years for all other equipment.

5.2.2 The design should,therefore, take advantage of equipment and experience gained in successful spaceflight missions and military programs. Advantage shall also be taken of recent industrial developments which fit this application.

5.2.3 Wherever the weight, cost, and schedule risk penalties are not prohibitive, functional or alternate mode redundancy should be employed such that no single failure mode of any electronic or electro-mechanical component could cause catastrophic effect on the mission.

5.2.4 Efforts shall be taken to reduce the functional interdependence between elements of the spacecraft. Where dependency must exist, every attempt should be made to achieve a degraded performance from the dependent element in spite of a failure in the element upon which it depends.

5.2.5 All scientific instruments shall be designed to be as functionally independent of one another as is practical so as to increase the assurance that a failure in one instrument or in equipment common to several instruments will have a minimum effect on the total data received.

5.2.6 Particular emphasis (for the 1977 Grand Tour time period) shall be placed upon simple and conservative design such that the full reliability can be achieved when a flight project is initiated with a complete program of component, subsystem, and system testing and analysis.

5.2.7 All spacecraft equipment shall have fail safe overload protection.

5.2.8 Simplicity in design approach and mechanization of hardware shall be emphasized in all subsystems.

5.3 Schedule Criteria

5.3.1 Since the mission objectives involve the 1977 Grand Tour (possibly 1974 Jupiter Precursor) opportunity, it obviously follows that all designs, techniques and components must be compatible with the project development time schedule, including all intermediate milestone objectives leading up to the launchings. The TOPS program shall develop the most critical advanced equipment which will be required for the 1977 Grand Tour mission.

5.4 Spacecraft Weight

5.4.1 The launch weight of the separated spacecraft must be no more than 1300 pounds.

5.5 Scientific Experiments

5.5.1 In order to provide a firm basis for design, a representative science payload has been selected as follows:

Instrument or Experiment

a. Trapped Radiation Detector (TRD)

b. Plasma Probe (PP)

c. DC-Magnetometer (DC-MAG)

d. AC-Magnetometer (AC-MAG)

e. "Radio Occultation"

f. Ultraviolet Photometer (UV)

g. "Celestial Mechanics"

h. Micrometeoroid Detector (MD)

i. Television (TV)

j. Infrared Radiometer (IRR)

k. Cosmic Ray (CR)

l. "Charged Partical and RF Propogation"

5.6 Competing Characteristics

5.6.1 In case of design conflicts, where modest compromise is required, emphasis should be given the following functions in the order listed:

a. Arriving at the planets within the prescribed accuracy and being able to communicate telemetry during the encounter periods and for the required scientific data playback interval thereafter. This total function requires the following specific functions:

(1) Continuous proper sun-line biased (and earth-line) attitude orientation,

(2) Continuous proper temperature control,

(3) Proper functioning of the RTG power equipment,

(4) Proper roll attitude control during encounter and for required scientific data playback periods after the encounters,

(5) Proper operation of the communication equipment during encounter and for required scientific data playback periods after the encounters,

(6) Proper operation of the trajectory correction maneuvers.

b. Proper operations of the planetary instruments and the capability of these instruments to observe the planets during the encounter modes.

c. Proper operation of the data storage and data handling equipment.

d. Telemetry communication capability during the cruise phase. This function requires roll attitude control when the high-gain antenna is required.

e. Adequate operation of the interplanetary science equipment.

f. Continuous return of science and engineering data from the spacecraft following the Neptune encounter to the limit of reasonable radiation effects buildup, degradation of the RTG, and communication capability.

g. Operation of the planetary scan in the desired fashion.

h. Compatibility of the design with other outer planet mission requirements.