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JET PROPULSION LABORATORY

CALIFORNIA INSTITUTE OF TECHNOLOGY

PASADENA, CALIFORNIA

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EXTREME TEMPERATURE REQUIREMENTS FOR SPACECRAFT ELECTRONICS PARTS

FINAL REPORT — PHASE I

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T. Hutchinson

Approved: T. C. Gindorf

T. Gindorf, Supervisor
Thermal-Vacuum Environmental
Requirements Group

for W. Shipley, Section Manager
Environmental Requirements
Section 294

John Small

J. Small, Division Manager
Project Engineering
Division 29

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

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SUMMARY

This study was begun in Fiscal Year 1968 (FY'68) to determine the extreme temperature requirements (ETR) for electronic piece-parts. The requirements are to be used as thermal design criteria for parts development.

Phase I of the study was to be performed in FY'68 and Phase II in FY'69. The objective of Phase I was to determine how (and if) the extreme temperature requirements could be obtained. Phase II is to utilize the method developed in Phase I to produce the temperature requirements and part type information.

The original approach for Phase I was to review existing mission studies and identify missions likely to produce extreme piece-part temperatures. In Phase II the electronic part types were to be identified from the mission study data, and thermal analyses was to be performed on the spacecraft configurations of the studies to determine the part temperatures.

The Phase I efforts have resulted in the determination that insufficient information is supplied by existing mission studies to provide a basis for the Phase II analyses. If Phase II is to be conducted as planned, additional detailed mission studies would need to be undertaken to supply the missing information. Because of the detail required in these mission studies, their inclusion would constitute a gross change in scope for the ETR effort.

Instead of performing the lengthy and expensive mission studies, an alternative to the original approach has been determined. This alternative divides spacecraft into three specific types: flyby, orbiter, and lander. These types will be systematically analyzed to determine the adequacy of current thermal control techniques to provide an acceptable environment for state-of-the-art electronics. The thermal control techniques employed will be constrained to a reasonable percentage of spacecraft weight and power. This "reasonable percentage" will be consistent with the amount of thermal control utilized on previous NASA missions. Extreme temperature requirements will be determined for those piece-parts whose environment is found to be thermally unacceptable.

The first type of spacecraft to be examined will be a flyby, as defined within the context of a "Grand Tour" mission to the outer planets.

SECTION I

INTRODUCTION

1.1 SCOPE

This report presents the results and conclusions from Phase I of the Extreme Temperature Requirements (ETR) Study. Included in this report is the history describing how JPL became involved in the ETR study and how JPL plans to continue the Phase II effort. The report is prepared in such a manner that by reading the first three sections, one should obtain a reasonably complete understanding of Phase I effort as well as of the plans for the remainder of the ETR study. The remaining sections (IV through IX) contain background information developed in Phase I for use in the Phase II effort.

1.2 HISTORY AND OBJECTIVES OF STUDY

In the Spring of 1967, JPL received a request from NASA Headquarters to perform a study of possible extreme temperature requirements for electronics. A preliminary plan was formulated and submitted to NASA Headquarters for approval. The stated objective of this plan was as follows:

> Objective: To determine temperature and life requirements, for survival and operation of electronic components, for use as electronic piece-part development criteria.

It was intended that this study would be used as a guide to determine priorities for development. Each NASA Center that might be affected by the results of the study was contacted and invited to participate in evaluation of the overall plan. Additional NASA Center participation in the effort, through formal or informal reviews, was encouraged. With the approval of NASA Headquarters, discussions were held with Manned Spacecraft Center in Texas, Marshall Spaceflight Center in Alabama, Langley Research Center in Virginia, Goddard Spaceflight Center in Maryland, and Ames Research Center in California. Most of the Centers visited expressed an interest in the results of the study (to be obtained in approximately two years time) but questioned any direct participation. At the time of these visits, the original ETR study plan was reviewed with each of the Centers. Since that time the original plan for Phase II has been significantly modified, as discussed in Section II of this report.

The study was divided into two phases so that at the end of Phase I a decision could be made as to the continuation and/or the method of continuation of the ETR study. That is to say, the primary objective of Phase I was to formulate a basis and a method for the determination of extreme temperature requirements. The objective of Phase II was to actually determine extreme temperature requirements utilizing the information from Phase I.

SECTION II

PROGRAM PLAN

2.1 ORIGINAL APPROACH

The ETR study was originally divided into two phases, Phase I and Phase II. Phase I was to last for the duration of FY'68 and to establish the methodology for the study. Phase II would begin with FY'69 and continue until the completion of the tasks derived from Phase I; completion estimated to be in mid-FY'71.

The objective of Phase I was the determination of how (and indeed if) the extreme requirements could be obtained. This activity resulted in an approach to the problem that would utilize data from past mission studies. The first step was to review these studies and identify those missions likely to produce extreme piece-part temperatures. In the reviewing process the significant details of each report were outlined in accordance with a standard format designed to condense the information (or specific references to the information) necessary for the Phase II activities, and thus to reduce the amount of referral to the original volumes.

Phase II was to utilize the Phase I data to identify the piece-parts and to calculate their temperatures. The part types would be identified either directly from the equipment (component) lists in the studies or indirectly from the stated functional requirements of the missions. Thermal analyses of the respective spacecraft configurations would then determine the piece-part temperatures.

It was hypothesized that the foregoing analyses might not yield the extreme mission results (thermally). In this event, the missions would be "extended", in either time or space, towards more extreme conditions. As an example, in the case of high temperature extremes, the extension might be by going closer to the sun or by lengthening the active life of a lander on a hot planet surface. The magnitude of the extension would be determined primarily by what was desired by the scientist/experimenter of today.

2.1.1 Reasons for Abandoning Original Approach

After examining the studies that were selected for review, it became clear that they did not represent situations where extreme electronic parts temperatures would be encountered. In all cases studied, the maintenance of a nominal range of spacecraft temperatures was a design constraint. The temperature range was dictated by the capability of current parts. The spacecraft configuration, and to some extent the mission profiles, were optimized under that constraint. A thermal control system would be employed that just maintained the parts temperatures within these limits. Parenthetically, it should be noted that for all the studies reviewed, which included near sun probes, the nominal range of electronic equipment temperatures was maintained with a reasonable amount of weight and power allocated to the thermal control subsystem. In all cases it was less than 10% of the spacecraft total weight and power.

It became apparent that the extension of one of those studies into a more severe environment would not be realistic unless the configuration was re-optimized for the extended environment. As an example, it would not be realistic to use a spacecraft designed for a Venus flyby as a model for computing temperatures for a Mercury flyby mission. It is reasonable to assume that at least the spacecraft configuration would be thermally redesigned; i.e., optimized for the Mercury Mission.

It is then apparent that the appropriate way to extend the existing studies (so that Phase II could be performed in the manner originally intended) would be by performing new in-depth mission studies. The series of trade-offs and compromises which occur in determining a spacecraft configuration and functional requirements would then develop normally from the mission profile and objectives.

The task of performing new mission studies was reviewed with NASA Headquarters. It was determined that the dollar amount necessary to conduct the new mission studies (as well as the manpower required within JPL) was prohibitive. At this point it was decided that the original study approach, to determine the extreme temperatures, might require some modification.

2.2 NEW APPROACH

A new approach has been formulated. Essentially, a temperature control type examination of missions will be conducted. The approach will be to determine if a nominally acceptable range of part temperatures can be maintained with a reasonable amount of temperature control for all missions. If this is the case, then there is no temperature problem for that set of electronics which is thermally controlled. The ETR study will then determine thermal requirements for the piece-parts in those assemblies which cannot be adequately thermally controlled. The piece-parts in those assemblies will be identified. It is believed that a reasonable amount of thermal control can maintain acceptable temperatures for <u>bus</u> electronics for any mission presently considered. However, this is probably not true for all of the science and various other sensors that are external to the bus.

The sequence in which the study will be performed involves studying three different types of spacecraft in what is assumed to be an ascending order of thermal control complexity. The first type is flyby spacecraft, which go near the sun or fly by a planet. The second type is planetary orbiter spacecraft, and the third is the landed spacecraft, referred to as "landers".

The study will proceed by first assuming that the spacecraft bus can be thermally decoupled from the solar environment. This is a reasonable assumption for planetary flybys and interplanetary space probes, including solar probes. Later phases of the study will determine the degree to which planetary landers can be decoupled from the surface environment. Once the study has verified this decoupling assumption, thermal control methods can be devised to provide a non-deleterious thermal environment (about room temperature) for the majority of spacecraft electronic components.

Attention will then be focused upon the electronics that are not enclosed in the bus. Sensors (principally attitude control, guidance, and scientific) comprise the majority of such external electronics. Their functional performance is predicated for the most part on an exposure to the space environment. It will be assumed that these sensors can be thermally decoupled from the bus and studied as separate items.

2.3 INITIAL PHASE II EFFORT

The first mission to be studied will be a flyby of the outer planets. (Asteroids, Jupiter, Saturn, Uranus and Neptune; referred to as a Grand Tour Mission.) The results will be representative of all anti-solar flybys.

A Grand Tour has been selected for initial study since it can utilize the information developed in the Outer Planet Advanced System Technology (AST) Project being currently conducted at JPL. The AST Project will provide mission design, spacecraft design, and payload data directly applicable to the ETR purposes. This will be data that could not, under present ETR funding, be otherwise made available.

The outer planet mission, progressing as it does from 1 AU to 30 AU, will provide a vehicle for demonstrating the principle of the decoupling theory, by showing that a spacecraft system can operate successfully independent of the sun. It will also develop temperature information for an extremely cold environment, principally applicable to the sensors.

2.3.1 Task Descriptions (FY'69)

The original Phase II function will be performed for outer planet flyby missions using data generated by the AST study. The principal effort will be a determination of how to maintain the sensors within reasonable temperatures. If such temperatures cannot be maintained, the temperature extremes and piece-parts will be identified (using AST equipment lists).

The AST spacecraft will use a Radioisotope Thermoelectric Generator (RTG) as its source of electric power. The influence of the RTG on the thermal balance of the spacecraft will be examined. The degree to which, and the method by which the spacecraft can be heated by the RTG will be determined. The use of waste heat and electric heaters will be considered.

2.3.2 Task Descriptions (FY'70 and FY'71)

The complete determination of extreme temperature requirements will require the performance of the following tasks listed in preferred order of performance.

1) Examine the sensor problems for near-sun missions (to 0.1 AU). Phase II will be performed for any components whose temperatures fall without present day limits.

2) Verify that the conclusions resulting from the flyby missions are valid for orbiter missions. It is reasonable to suspect that this is true for anti-solar missions. Sunward missions (Venus, Mars) will require further examination. As in the flyby case, Phase II will be performed on components whose temperature appears non-standard.

3) Examine lander missions. These are more difficult to handle and may produce temperature extremes in excess of those for flybys or orbiters.

The limitations on landers covered by existing studies will be examined. One result of this will be a determination of how long a lander should survive the thermal surface environment. The operational duration of a lander is very important. Some thermal control schemes can maintain reasonable temperatures for short duration only, i.e., phase change cooling for hot surfaces and chemical heaters for cold surfaces.

An analysis, similar to that performed on the flybys and orbiters, will be performed to determine if a lander bus can be decoupled from its sensors in both hot and cold environments. The degree to which the lander bus can be held at a nominal temperature will then be determined. If it cannot be held nominal, the piece-parts affected will be enumerated and their extreme temperatures determined by Phase II methodology. (Note: it is possible that lander mission studies might need to be performed.) Finally the temperature of the lander sensors will be determined and Phase II performed on out-of-temperature-tolerance parts.

2.4 SCHEDULE AND RESOURCES FOR ETR STUDY

The remaining efforts in the ETR study are projected in Fig. 2-1. This schedule will need to remain as flexible as possible in order to efficiently complete the effort. Flexibility in the schedule will enable the emphasis on the different aspects of the activities to be appropriately shifted from one phase to the other as warranted by the study results. The FY'69 effort will be devoted

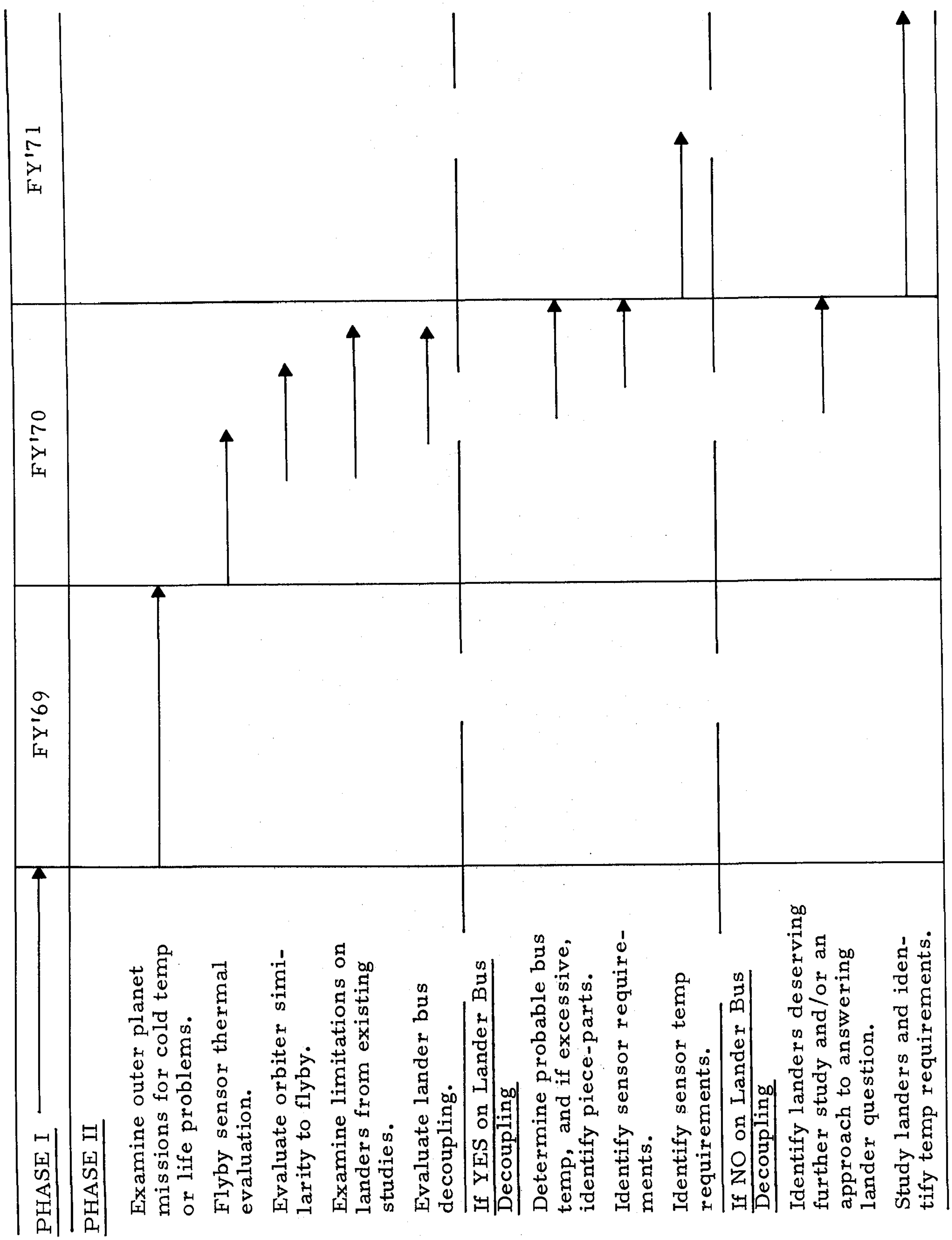

Fig. 2-1. Schedule for remainder of ETR effort

solely to study and analysis of the AST project and the temperature questions associated with outer planet missions. Fig. 2-1 is a realistic estimate at this time of the schedule for the remainder of the ETR effort.

SECTION III

PHASE I RESULTS

3.1 MISSION STUDY REVIEW

Mission studies were selected from the bibliographies that are available through the JPL library system and each of the NASA Centers. These mission studies were the basis for evaluating how complete the advanced mission planning has been in terms of predicting or anticipating extreme temperature requirements. The mission studies review was the principal effort in Phase I.

Approximately 45 mission studies, out of a total of approximately 100 available, have been reviewed in detail. The mission studies selected from the total group for review were those concerned with extreme environments, or representative of specific mission types such as landers, orbiters, etc. The studies included: solar probes in the proximity of the Sun; planetary flybys of Mercury, Venus, and out to Jupiter; and lander systems on Mars and Venus. In addition, the studies by the Illinois Institute of Technology Research Institute (IITRI) on general scientific and engineering objectives for future missions were reviewed.

Based upon the conclusions that have been reached from the studies which were reviewed, it is not necessary to review the remainder. Furthermore, in our opinion, the reviewed group represents those mission studies most suitable to this effort.

A committee comprised of representatives from the various disciplines within JPL formulated a general questionnaire to be answered by the mission studies. The questionnaire consisted of questions which required answering in order to accomplish Phase II of the study -- as Phase II was then envisioned. The studies selected for review were those which were of sufficient depth so that most of the questionnaire could be adequately answered (see Appendix C).

As part of Phase I, it was necessary to formulate additional background information from which one could proceed with Phase II.

3.2 ADDITIONAL PHASE I TASKS

Included were the following:

1) A verification that the spacecraft bus* can be decoupled from the solar thermal radiation and maintained at acceptable temperatures.

2) A summary of the science and sensor requirements for flybys obtained from a review of the mission studies.

3) A tabular listing of planet surface temperatures and their variations.

4) A summary of current lander capability, based upon available mission studies.

The results and conclusions of these efforts are summarized in 3.3, following. The details will be presented in the individual sections discussing each particular effort (Sections V through VIII).

3.3 RESULTS OF ADDITIONAL PHASE I TASKS

3.3.1 Spacecraft Bus Decoupling

It is shown in Section V that a basic system of electronics, such as within a spacecraft bus, can be decoupled, by shielding or insulating, from the solar thermal radiation environment. This decoupling has been studied by several organizations interested in the solar probe type of a mission. Current indications are that a nominal shield weight of less than 10% of total spacecraft weight (actually as low as 3% in one study) would be needed for a 0.1 AU mission, which is one of the most ambitious solar probe missions likely to be attempted in the next 30 years.

*"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 usually is thermally controlled as a unit.

In addition, it has been shown that missions to Jupiter and beyond are not significantly influenced by the solar irradiance. Therefore, for the flyby type of missions to both the extremely hot and cold portions of the solar environment, the decoupled spacecraft system is a realistic approach to be considered for the ETR study.

3.3.2 Flyby Mission Science and Sensor Requirements

Once a flyby spacecraft bus is successfully decoupled from the solar radiation environment, the question of temperature extremes is reduced to experiments and sensors. For all of the flyby mission studies that were reviewed, a tabular listing of the engineering and scientific objectives has been prepared (see Section VI). A summary of these objectives has not been prepared since it is intended that each objective, and the sensors required for them, will be studied further to determine whether or not temperature will present a problem for electronic component survival. The sensor types will be examined later to determine the advisability of excluding certain sensors with common objectives.

3.3.3 Planet Surface Temperatures

If one assumes that a refrigeration system is not available which would provide long term thermal control capabilities, then the conclusion could be drawn that the surface temperature of planets (with atmospheres) would be representative of the minimum temperatures of a spacecraft in which no power was being dissipated. The spacecraft would be warmer than this minimum temperature as a function of power dissipation.

The planetary surface temperatures are presented as an approximation of the lander temperatures for planets with atmospheres. These temperatures are order-of-magnitude representations and are not to be interpreted as being lander temperatures. From this section, one observes that the steady state thermal extremes for landers would occur on Venus (926°F day-time-maximum temperature) and Mercury or Pluto (-436 and -406°F night-time-minimum temperatures, respectively). While the high temperature case is considered certain to occur on Venus, the low temperature case is less certain due to a lack of information on many of the planets and the fact that neither Mercury nor Pluto are believed to have atmospheres. It is, however, obvious that a temperature of -436°F is only ~24° above absolute zero.

The 926°F experienced on Venus and the possible -436°F experienced on the non-sun side of Mercury are considered as the lowest high temperature and lowest low-temperature, since spacecraft power dissipation will cause the system equilibrium temperature to be higher. No bounds are placed on how much higher the maximum spacecraft temperature may be. It should be noted that these planetary cases represent steady state conditions and that effects such as thermal shock ($\partial T/\partial t$) have not yet been considered. The transient cases will be examined later.

3.3.4 Current (Studied) Lander Capability

Martian lander systems which have been studied weighed from 138 to 1186 lb. The survival times ranged from 5 hours to 2 years and nominal temperatures (-140°F to 150°F) were maintained. It should be noted that sterilization temperatures (~257°F) in a non-operating condition probably represent the maximum temperature constraint for Martian landers.

Current Venusian landers which have been studied weighed from 5 to 413 lb. The survival times ranged from 0 to 20 hours with a maximum temperature of about 180°F internally. It is noted that the longer duration landers (>20 min) either did not specify maximum temperatures or were completely out of line with the other temperature estimates. The Venusian lander is considered to represent the most severe high temperature environment for a long life lander system. Considerable study will be devoted to the lander temperature questions later in the ETR effort.

SECTION IV

SUMMARY OF REVIEWED MISSION STUDIES

4.1 MISSION STUDY SELECTION

The bulk of the mission studies initially considered for review consisted of studies listed in the NASA STAR index. The ASTIA TAB index of DOD documents was also consulted, but provided few references. Other applicable studies not indexed in STAR were obtained through contacts with NASA Centers, the JPL Future Projects Office, and private industry. The private industry contacts showed no significant in-house (as opposed to NASA funded) study efforts not previously indexed in STAR. Thus, the list of studies contained in Appendix A and B is felt to be sufficiently comprehensive to encompass the total scope of all missions which have been studied. Classified (DOD) studies were not considered. Studies indexed prior to the 1963 STAR were also not considered, to insure that studies used did not represent obsolescent technology.

The studies were too numerous to permit review of all of them within the time period available. Thus, those missions most likely to represent an extreme temperature environment were given primary attention (see Appendix B). Typical of such missions were solar probes, missions to the outer planets, and Venus landers.

Some of the other studies were eliminated from consideration because of lack of sufficient detail to enable the objectives of the review to be met. Because of the large number of Voyager-type Mars lander missions, only the most representative studies were reviewed to avoid needless duplication of effort. Manned missions were not reviewed since it was assumed that, in general, electronics used on such missions would see the same nominal room temperature environment experienced by the crew. (This was confirmed through conversations with the Manned Spacecraft Center.) Some studies were not reviewed because it was felt there was little likelihood of such a mission being performed in the time period considered (1975 to 2000).

The mission study reviews supplied answers to the list of questions shown in Appendix C. The list was prepared by representatives from each of the JPL Technical Divisions and represented specific questions they felt needed to be answered to conduct the Phase II portion of the study. Even though the mission study reviews represent a vast condensation of the original material, the sum total of the reviews still is a formidable amount of paper, as may be seen from the example review included in Appendix D. Therefore, a summary (Tables 4-1 through 4-5) was assembled for this report in order to present the more pertinent conclusions.

4.2 TABULATED SUMMARY

Table 4-1 is a summary of all the studies reviewed. In Table 4-1, where there is a variation in the answers on a topic, the range of values is given on all possibilities listed for a particular planet. For studies concerned with more than one planet (such as multiple planet missions or both Venus and Mars capsules) the results are presented for each planet. Tables 4-2 through 4-5 present the same information as Table 4-1, but information is broken out separately for each destination or planet. This permits identification of the tabulated values with each study reviewed. The mission priorities listed are those recommended by the Space Science Board of the National Academy of Sciences. The priority of each planet is indicated, along with the priority among types of missions to a given planet. (For example, Mars has the highest priority of all planets. A lander would be the most desirable Mars mission, with an orbiter being less desirable and a flyby least desirable.) In general, the number of studies reviewed for each planet reflects the Space Science Board priority. Mars and Venus missions represent nearly 50% of the reviews, with other planets represented in lesser numbers. The only exception was in the case of the solar probes which were not considered on the Space Science Board's priority list, but were included because it was felt that they qualified as extreme temperature mission studies.

An important result of the mission study review should have been identification of the missions already studied for which the thermal control systems could not maintain electronics parts within a suitable temperature range. Instead, it was learned that mission planners have a very optimistic outlook toward spacecraft thermal control systems. In every mission studied, a nominal (room-temperature) thermal environment was assumed (or predicted) for electronics packages. An active thermal control system, for the purposes

Table 4-1. Summary of all missions reviewed

Mission	Number of Existing Studies	Mission Priority	Temperatures within Capability of Electronics?	Thermal Control			Type of Power System	Mission Duration	Launch Opportunities	Launch Vehicles
				% Weight	% Power	Active				
A. Solar Probes	7	-								
0.05 AU	-		-	-	-	-	-	-	-	-
0.09 - 0.11 AU	3	-	Yes	0.09-8	-	No/Yes/Yes	Solar cells	75 days to 1 yr.	1971, 73, 74	Saturn 1B/Centaur
0.18 - 0.24 AU	2	-	Yes	2.5-8	-	Yes	Solar cells	75 to 120 days	-	Atlas/Centaur/Kick, Titan II/ Centaur/X 259
0.28 - 0.30 AU	4	-	Yes	0.9-10	-	2 No, 2 Yes	Solar cells	3 mo. to 1 yr.	-	Atlas/Centaur/X 259 Atlas/ Centaur, Atlas/Agena
0.35 - 0.41 AU	4	-	Yes	0.9-4.5	-	1 No, 3 Yes	Solar cells	3 mo. to 1 yr.	-	Atlas/Agena/X 259, Atlas/ ABL-259
0.5 - 0.65 AU	2	-	Yes	0-4.5	-	No/Yes	Solar cells	4 mo. to 2 yr.	-	Scout, Thor Delta
B. Mercury		6								
Flyby	1 (Venus/Mercury Mission)	A	Yes	2.2	<10	Yes	Hybrid solar thermionic-photocells	154 days	1973	Atlas/Centaur SLV-3C
Orbiter	-	C	-	-	-	-	-	-	-	-
Lander:	-	B	-	-	-	-	-	-	-	-
Hot side	-		-	-	-	-	-	-	-	-
Cold side	-		-	-	-	-	-	-	-	-
C. Venus		2								
Flyby	2	C	Yes	2.2-3.1	<10	Yes	Hybrid-solar cells	102 to 110 days	1970, 73	Atlas/Centaur
Orbiter	2	A	Yes	3.5	-	Yes, No	Solar cell, battery	6 mos. orbit	1969-75	Saturn 1B/SVI, Atlas/Centaur Titan IIIC
Buoyant Station	1	B	Yes	1.2-15*	-	Yes	RTG	7 to 100 days	1972, 73	Atlas/Centaur
Lander	6	B	Yes	2.8-4$^{\Delta}$	-	2 Yes, 4 No	Battery	Impact to 20 days on Surface	1969-75	Atlas/Centaur, Saturn SLB/SVI, Manned Flyby Vehicle
D. Earth										
ATS-4	1		Yes	3.8	-	Yes	Solar cells	2 years	1969, 70	Atlas/Agena, Atlas/Centaur, Titan IIIC

NOTE: Some studies considered more than one planet or mission type
*Includes gondola structure
Δ Includes heat shield

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Table 4-1. Summary of all missions reviewed (contd)

Mission	Number of Existing Studies	Mission Priority	Temperatures within Capability of Electronics?	Thermal Control % Weight	Thermal Control % Power	Thermal Control Active	Type of Power System	Mission Duration	Launch Opportunities	Launch Vehicles
E. <u>Mars</u>		1								
Flyby	1	C	Yes	4.2	-	Yes	Solar cells	202 days	1969, 72	Atlas/Centaur
Orbiter	5	B	Yes	3-3.7	2.5-7	4 Yes	Solar cells-battery	90 to 180 days	1969-75	Atlas/Centaur, Atlas/Centaur/Kick, Saturn CIB, Saturn 1B/SVI, Saturn 1B/Centaur, Titan IIIC
Lander	8	A	Yes	1.0-17Δ	-	6 Yes	RTG or battery	15 min to 2 yrs. surface	1969-79	Atlas/Centaur - 1Kick, Saturn C1B, Saturn V Saturn 1B/Centaur, Saturn 1B/SVI, Manned flyby
F. <u>Comets and Asteroids</u>	1	5	Yes	2.3-4	-	Yes	Solar cells or RTG	160 to 300 days	1967, 70, 73, 74	Atlas/Agena, Atlas/Centaur
G. <u>Jupiter</u>		3								
Flyby	5	B	Yes	1.5-3.0	6.1-11.1	4 Yes	RTG	180 to 900 days	1969-80	Atlas/Agena, Atlas/Centaur/Kick, Saturn V, Saturn V/Centaur, Saturn 1B/Centaur Kick, Titan IIIC/Centaur/Kick
Orbiter	-	A	-	-	-	-	-	-	-	-
Lander	-	C	-	-	-	-	-	-	-	-
H. <u>Saturn</u>		4								
Flyby	1	B	-100°C	N.S.	-	Yes	RTG	1.4 to 3.1 yr.	1976-79	Atlas/Centaur/Kick, Saturn 1B/Centaur/Kick, Saturn V/Centaur
Orbiter	1	A	-100°C	N.S.	-	Yes	RTG	2.4 to 3.8 yr.	1977-79	Saturn 1B/Centaur/Kick, Saturn V/Centaur
Lander	-	C	-	N.S.	-	Yes	-	-	-	-
I. <u>Uranus</u>		4								
Flyby	1	B	-100°C	N.S.	-	Yes	RTG	3.2 to 5.0 yr.	1976-79 1979-85	Same as Saturn Flyby
Orbiter	1	A	-100°C	N.S.	-	Yes	RTG	5.7 to 12 yr.	1977-79	Same as Saturn Orbiter
Lander	-	C	-	-	-	-	-	-	-	-

NOTE: Some studies considered more than one planet or mission type
N.S. Not specified
ΔIncludes heat shield

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Table 4-1. Summary of all missions reviewed (contd)

Mission	Number of Existing Studies	Mission Priority	Temperatures within Capability of Electronics?	Thermal Control			Type of Power System	Mission Duration	Launch Opportunities	Launch Vehicles
				% Weight	% Power	Active				
J. Neptune		4								
Flyby	1	B	-100°C	N.S.	-	Yes	RTG	5.2 to 8.8 yr.	1979-81 1979-85	Saturn 1B/Centaur/Kick, Saturn V/Centaur
Orbiter	1	A	-100°C	N.S.	-	Yes	RTG	9.0 to 20.2 yr.	1977-79	Saturn 1B/Centaur/Kick, Saturn V/Centaur
Lander	-	C	-	-	-	-	-	-	-	-
K. Pluto		7								
Flyby	1	A	-100°C	N.S.	-	Yes	RTG	23.5 to 38 yr.	1979-78	Same as Neptune Flyby
Orbiter	1		-100°C	N.S.	-	Yes	RTG	25.8 to 32 yr.	-	Same as Neptune Orbiter
Lander	-		-	-	-	-	-	-	-	-

NOTE: Some studies considered more than one planet or mission type
N.S. Not specified

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Table 4-2. Summary of individual Venus missions

Venus	Reference	Temperature within Capability of Electronics?	Thermal Control			Power System	Mission Duration	Launch Opportunities	Launch Vehicles
			% Weight	% Power	Active?				
1. Flyby	a) JPL 760-1	yes	2.2	<10	yes	Hybrid solar thermonic-photocells	102 days	1970, 73	Atlas/Centaur
	b) JPL 760-10	yes	3.1	N.S.	yes	Solar cells	110 days	1973	Atlas/Centaur
2. Orbiter	a) AVCO RAD-TR-63-34	yes	-	-	no	Solar cell - battery	116 day transit, 6 month orbit	1969, 71, 73, 75	Saturn 1B with SVI upper stage
	b) Boeing (CR-66302)	yes	3.5	-	yes	Solar cell - battery	104 to 183 day transit, 180 day orbit	1970, 72, 73	Atlas/Centaur, Titan IIIC
3. Buoyant Station	a) Martin (CR-66404)	yes	1.2	-	yes	RTG	7 days	1972, 73	Atlas/Centaur
	b) Martin (CR-66404)	yes	15*		yes	RTG	100 days	Same as Voyager	Voyager Class
4. Lander	a) JPL 760-1	yes	40^{Δ}	-	no	Battery	12 days (separation to impact)	1970, 73	Atlas/Centaur
	b) Bellcomm (7/3/67)	yes	10	-	yes	Battery	One hour	N.S.	N.S. - manned Venus Flyby
	c) Martin (CR-66404)	yes	17	-	no	Battery	One hour (descent to surface)	1972, 73	Atlas/Centaur
	d) JPL 760-10	yes	27.4^{Δ}	0	no	Battery	Survive 18 min. on surface	1973	Atlas/Centaur
	e) AVCO RAD-TR-63-34	yes	2.8	-	yes	Battery - Windmill	10 to 20 hours on surface	1969, 71, 73, 75	Saturn-S-1Bw. S-VI upper stage
	f) Goddard (TN D-1909)	N.S.	N.S.	-	N.S.	N.S.	N.S.	N.S.	N.S.

N.S. Not specified
*Includes gondola structure
$^{\Delta}$Includes heat shield

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Table 4-3. Summary of individual Mars missions

Mars	Reference	Temperature within Capability of Electronics?	Thermal Control			Power System	Mission Duration	Launch Opportunities	Launch Vehicles
			% Weight	% Power	Active?				
1. Flyby	a) AVCO RAD-TR-64-36	yes	4.2	-	yes	Solar cells	282 days	1969, 71	Atlas/Centaur
2. Orbiter	a) JPL, EPD-139	yes	3.3	-	yes	Solar cells	280 days transit	1969	Saturn C-1B
	b) JPL, EPD-250	yes	3.7	-	yes	Solar cell - battery	460 days flight 180 days in orbit	1969	Atlas/Centaur
	c) AVCO RAD-TR-63-34	yes	-	-	no	Solar cell - battery	283 days transit 180 days in orbit	1969, 71, 73, 75	Saturn 1B/with SVI upper stage
	d) TRW 5303-6014-TU-000	yes	3	2.5-7.0	yes	Solar cells or RTG	90 days in orbit	1971, 73, 75	Saturn 1B/Centaur Atlas/Centaur/Kick
	e) Boeing (CR-66302)	yes	3.3	-	yes	Solar cell + battery	180 days in orbit	1971, 73	Atlas/Centaur, Titan IIIC
3. Lander	a) JPL, EPD-139	yes	17[Δ]	-	yes	RTG or batteries	90 days or surface	1969	Saturn C-1B
	b) JPL, EPD-459	yes	1.5	-	yes	RTG or turbine and battery	N. S.	1963, 77, 79	Saturn V
	c) AVCO RAD-TR-63-34	yes	-	-	no	RTG	180 days on surface	1969, 71, 73, 75	Saturn 1B with SVI upper stage
	d) TRW 5305-6014-TU-000	yes	1.0	-	yes	RTG	30 days on surface	1971, 73, 75	Saturn 1B/Centaur Atlas/ Centaur/Kick
	e) JPL TM 33-236	yes	4.3-7	-	yes	Battery	1.5 days, 6 mo	N. S.	Saturn 1B/Centaur
	f) AVCO RAD-TR-64-36	yes	5.5	-	yes	Battery	5 hours on surface	1969, 71	Atlas/Centaur
	g) Aeronutronics U-3237	yes	2.5	-	yes	RTG	2 years on surface	1975	Saturn 1B
4. Sample Return	a) Brown Engineering (CR-61172)	yes	-	-	N. S.	Battery	15 min on surface and return	N. S.	Manned flyby vehicle

N. S. Not specified
[Δ]Includes heat shield

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Table 4-4. Summary of individual Jupiter missions

Jupiter	Reference	Temperature within Capability of Electronics?	Thermal Control			Power System	Mission Duration	Launch Opportunities	Launch Vehicles
			% Weight	% Power	Active?				
1. Flyby	a) LM & SC #M-49-65-1	yes	1.6	0	no	RTG and battery	6 mo to 2.7 years	1969, 72, 74, 76, 78, 80	Atlas/Agena and Atlas/Centaur with 3rd stage
	b) JPL #EPD-358	yes	2.8-3.0	6.1-8.0	yes	RTG	512 to 948 days	1972, 74	?/Centaur
	c) TRW #4547-6006-R000	yes	1.9	11.1	yes	RTG	544 to 910 days	1970-80	Saturn V/Centaur, Saturn V, Saturn 1B/Centaur/ Heks, Titan IIICx/ Centaur, Atlas SLV 3x/Centaur/Heks. Atlas SLV Cx/Centaur/TE-364-3, Titan III Cx/Centaur/TE-364-3
	d) NASA/GSFC #X-701-67-566	yes	3.0	Not indicated	yes	RTG	500 to 600 days to Jupiter, 3.5 yr to 10 AU	1972 (13 mo intervals between future launch)	Atlas/Centaur/TE-364, Atlas SLV-3X/Centaur-70/TE 364-4, Titan IIIC/Centaur
	e) GDC #FZM-4625	yes	1.5-2.1	Not indicated	yes	RTG	421 to 600 days	1973-80	Atlas SLV3/Centaur/Heks, Titan IIICx/Centaur, Saturn V. Saturn 1B/Centaur/Heks

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Table 4-5. Summary of individual solar probe missions

Reference	Is Temperature within Electronics Capability?	Thermal Control			Type of Power System	Mission Duration	Closest Approach To Sun (AU)	Launch Vehicles
		% Weight	% Power	Active?				
1) Stanford, CR 80841 (ICARVS)	Yes	8	N.S.	Yes	Solar Cells, Battery	75-120 days	a) 0.18-0.23 b) 0.09-0.11	a) Atlas/Centaur/Kick b) Saturn 1B/Centaur/Kick
2) G.E., 63 SD 779	Yes	0.9	--	No	Solar Cells	1 Year	a) 0.09 b) 0.29 c) 0.35	a) Saturn 1B b) Atlas/Centaur/ X 259 c) Atlas/Agena/ X 259
3) Minneapolis-Honeywell, CPE 3D-B-35-3	Yes	3.3	--	Yes	Solar Cells, Battery	1Year	a) 0.36 b) 0.28 c) 0.1	a) Atlas/Agena/Solid b) Atlas/Centaur c) Saturn 1B + High Energy
4) Martin Marietta ER 13110	Yes	2.5	--	Yes	Solar Cells, Battery	1 Year (3 mo. to 0.3 AU)	a) 0.37 b) 0.30 c) 0.24	a) Atlas/Agena/ X 259 b) Atlas/Centaur/ X 259 c) Titan II/Centaur/ X 259
5) MIT, N65-32722 (Sunblazer)	Yes	0	0	No	Solar Cells, Capacitors	2 Years	0.5	Scout
6) TRW/STL 4159-6003-RU-000	Yes	4.5	N.S.	Yes	Solar Cells	4 Months	a) 0.65 b) 0.41	a) Thor-Delta b) Atlas/ABL-259
7) NASA/AMES, TM X-54039	Yes	10*	--	No	Solar Cells	1 Year	0.3	Atlas/Agena, Atlas/Centaur

* Includes Vehicle Structure

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of this study, was defined as a system which directly reacts (either mechanically, electrically, or chemically) to varying thermal fluxes to produce an acceptable thermal environment. (By this definition, systems which rely solely on insulation and paint patterns are excluded from consideration as active systems.) Where active thermal control was employed, only a small fraction of total spacecraft weight and power (less than 5% on nearly all missions, and none exceeded 10%) was required to bring about this nominal environment. Active thermal control systems were used mainly for long-duration (greater than 1 day) lander missions, missions beyond Mars, and near-approach solar probes. They ranged in complexity from the melting ice approach of the Bellcom Venus lander to the closely controlled closed-loop cycle of the Automated Biological Laboratory. Most active control systems were variations of the approach used on Mariner IV (louvers used as the active element).

Another tabulated item is the type of power subsystem used. Photovoltaic solar panels are used universally for missions towards the Sun, because of their greater efficiency with increased solar intensity. Solar Panels are also the favored power system for flyby and orbiter missions to Mars. For missions further from the Sun than Mars, other systems become competitive. A Radioisotope thermal generator (RTG) was almost universally considered as the probable main power source for long-duration landers and missions beyond Mars. An interesting point to be noted in the Jovian mission studies is that while all of these studies used an RTG for the power source, none considered waste heat from the RTG as a prime heat source for the thermal control system. This seemed unusual since the efficiency of RTG's is only on the order of 15%, which results in a considerable amount of waste heat available for thermal control purposes. All studies concluded that conventional methods of thermal control would be sufficient, and RTG waste heat would be used only if necessary as a back-up or secondary source. Usually, more consideration was given to shielding the rest of the spacecraft from RTG radiation than in utilizing the RTG in the thermal control system.

The time required for successful completion of a mission was also tabulated. This gives an estimate of the required operational lifetime of electronics exposed to temperature extremes. The missions reviewed will last from 75 days for some solar probes to as much as several decades for the outer reaches of the solar system. Flyby missions to all planets except Uranus, Neptune and Pluto are possible with trip times of 3 years or less. For the highest-priority planets, a one year minimum duration is normal. Multiple planet missions to Jupiter and beyond represent the upper bound of mission durations among those missions most likely to be performed.

The tabulation of launch opportunities reveals the clustering of opportunities to many different planets occurring in the late 1970's. This is further documented by the proposed Space Science Board plan (Tables 4-6 and 4-7)* for planetary exploration which places heavy emphasis on launches in this time period. The launch vehicles used in all studies were types which have already been developed and which use conventional chemical propulsion systems. As a result, all missions use ballistic-type trajectories, with no consideration given to continually-thrusted type missions.

The type of launch vehicle also indicates the size of spacecraft contemplated for a mission. The Atlas/Agena and Atlas/Centaur vehicles will accommodate spacecraft of the Mariner class, depending on the mission destination, while Saturn class vehicles are useable for the larger payloads desired for intensive investigations of Mars and Venus. It was also noted that landers as a class underwent more severe thermal environments than flyby or orbiting spacecraft. The present sterilization techniques and aerodynamic heating while entering the planet's atmosphere may cause problems for spacecraft components as severe as hostile planet surface conditions.

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*Taken from "Space Research, Directions for the Future" (1965, Part I), Space Science Board, National Academy of Sciences, National Research Council.

Table 4-6. Space science board recommended program (△)

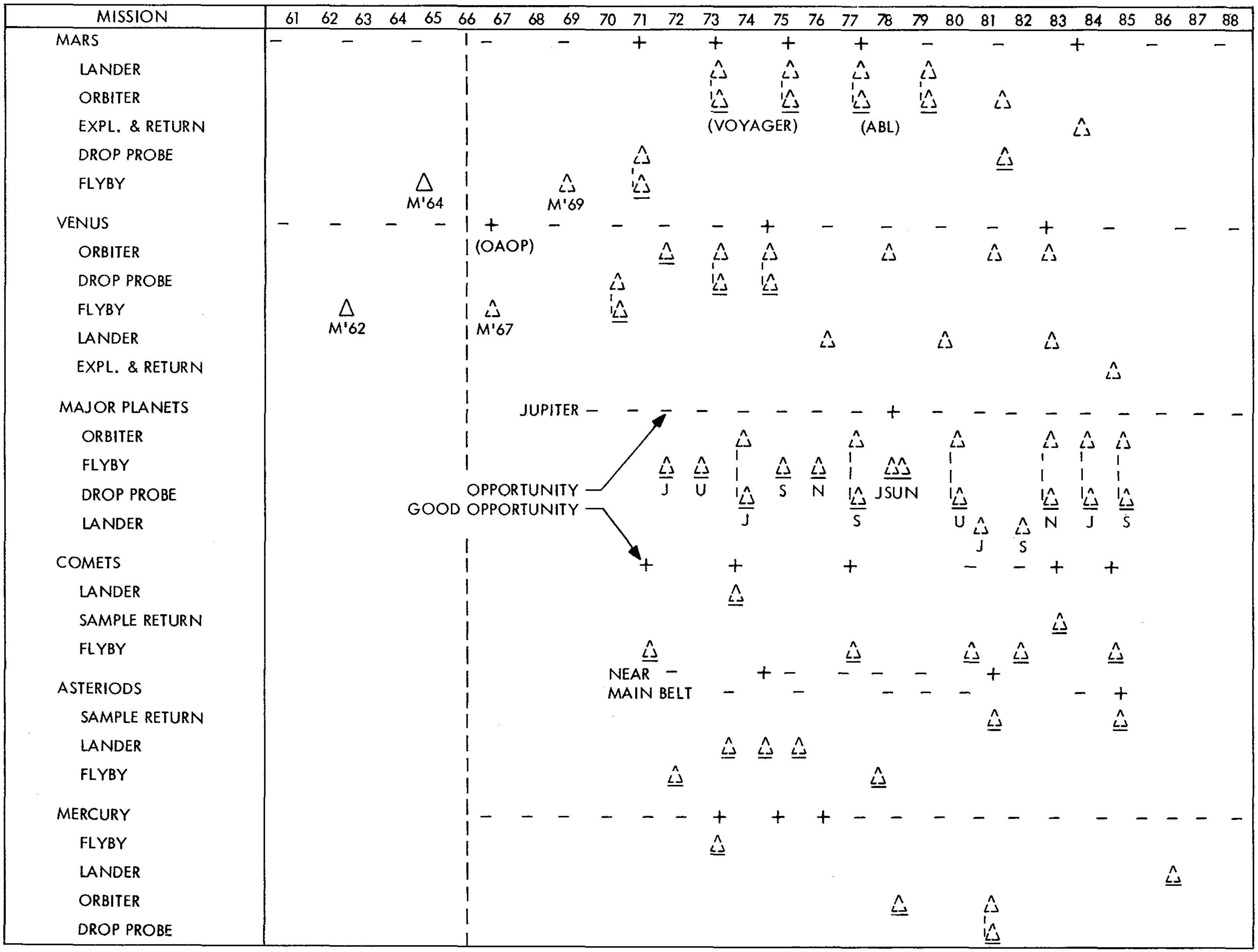

Table 4-7. Calendar launch schedule – program implications of space science board recommendations

PROGRAM	MISSION	67	68	69	70	71	72	73	74	75	76	77	78	79	80	81	82	83	84	85	86	87	88	89	90	91	92
MARINER (600-1000 lbs)	MARS FLYBY			ΔΔ		ΔΔP																					
	VENUS FLYBY	Δ			ΔΔP																						
	COMETS					ΔΔ					ΔΔ				ΔΔ												
	ASTEROIDS						ΔΔ					ΔΔ															
	MERCURY FLYBY							ΔΔV																			
ADVANCED PLANETARY PROBE - APP (500 lbs)	JUPITER FLYBY						ΔΔ																				
	GRAND TOUR*											ΔΔ	ΔΔ														
	SATURN FLYBY									ΔΔ																	
	URANUS FLYBY							ΔΔ																			
	NEPTUNE FLYBY										ΔΔ																
VOYAGER MARS	ORBITER/LANDER							Δ²		Δ²						Δ²P											
	ORBITER/ABL											Δ²		Δ²													
VOYAGER ADVANCED PLANETARY (2000-2500 lbs)	NEPTUNE ORBITER																ΔΔP										
	VENUS ORBITER						Δ²		ΔΔP		ΔΔP	ΔL	Δ²		ΔL	Δ²		Δ+L									
	JUPITER ORBITER									ΔΔP					ΔL			ΔΔP									
	SATURN ORBITER											ΔΔP					ΔL		ΔΔP								
	URANUS ORBITER														ΔΔP												
	COMETS									ΔΔ							ΔΔ	ΔΔ		ΔΔ							
	ASTEROIDS									ΔΔ	ΔΔ	ΔΔ				ΔΔ				ΔΔ							
	MERCURY ORBITER												ΔΔ			ΔΔP					ΔL						

SYMBOLS:
- Δ APPROVED
- Δ PROPOSED
- Δ² 2 S/C LAUNCHED WITH 1 VEHICLE
- ΔΔ 2 LAUNCHES IN 1 OPPORTUNITY

- P - S/C + PROBE
- +L - S/C + LANDER
- L - LANDER ONLY
- V - VENUS GRAVITY ASSIST
- J - JUPITER GRAVITY ASSIST

- S - SATURN GRAVITY ASSIST
- * GRAND TOUR: A GRAVITY-ASSISTED JUPITER, SATURN, URANUS, NEPTUNE MISSION

SECTION V

DECOUPLING OF SPACECRAFT BUS FROM INTERPLANETARY ENVIRONMENT

5.1 INTRODUCTION

A spacecraft bus can be decoupled from the solar intensity in interplanetary space by the proper design of shading and insulating shields, and the minimization of conductive and radiative heat transfer between the bus and other parts of the spacecraft.

5.2 DESCRIPTION OF DECOUPLING

In general, the decoupling approach to temperature control of a spacecraft is to impose a shield between the payload and Sun. This shield, designed to reflect and re-radiate a major portion of the incident solar energy, can provide the payload with an environment that is essentially independent of the solar distance. The ambient spacecraft temperature is then primarily a function of internal power, surface area, and thermal emittance.

The decoupling approach has been investigated by NASA and is reported in TN-D-1209. The study demonstrated that solar-shields could be used effectively to isolate a spacecraft from direct solar radiation. The following paragraph is a synopsis of that report.

Two solar probes were considered which traveled to within 0.1 AU of the Sun. They had identical configurations except that one had a solar-shield. Both spacecraft dissipated 100 w of electrical power. The equilibrium temperatures of the spacecraft were computed as a function of solar distance and are shown in Figs. 5-1 and 5-2. It can be seen that for $\bar{r}$ in the range from 0.1 to 1.0, and solar absorptance of the shield (α_{S1}) equal to 0.2 the capsule temperature change is 26°F for the single-shield configuration and only 2°F for the double-shield configuration. For comparison, the capsule temperature change for the unshielded configuration with $\alpha/\epsilon = 5.0$ is about 1490°F. Even for α/ϵ equal to 0.1, the capsule temperature change is about 360°F. The report concluded that, "the use of solar shields can reduce the capsule temperature variation by at least one to two orders of magnitude from that

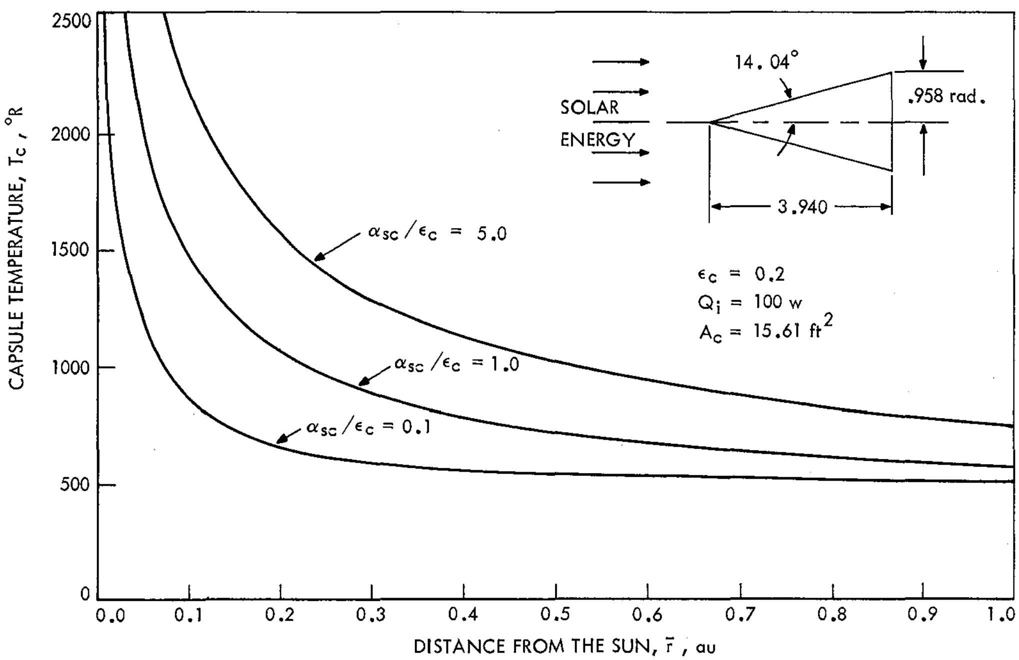

Fig. 5-1. Temperature of an unshielded conical capsule as a function of distance from the Sun

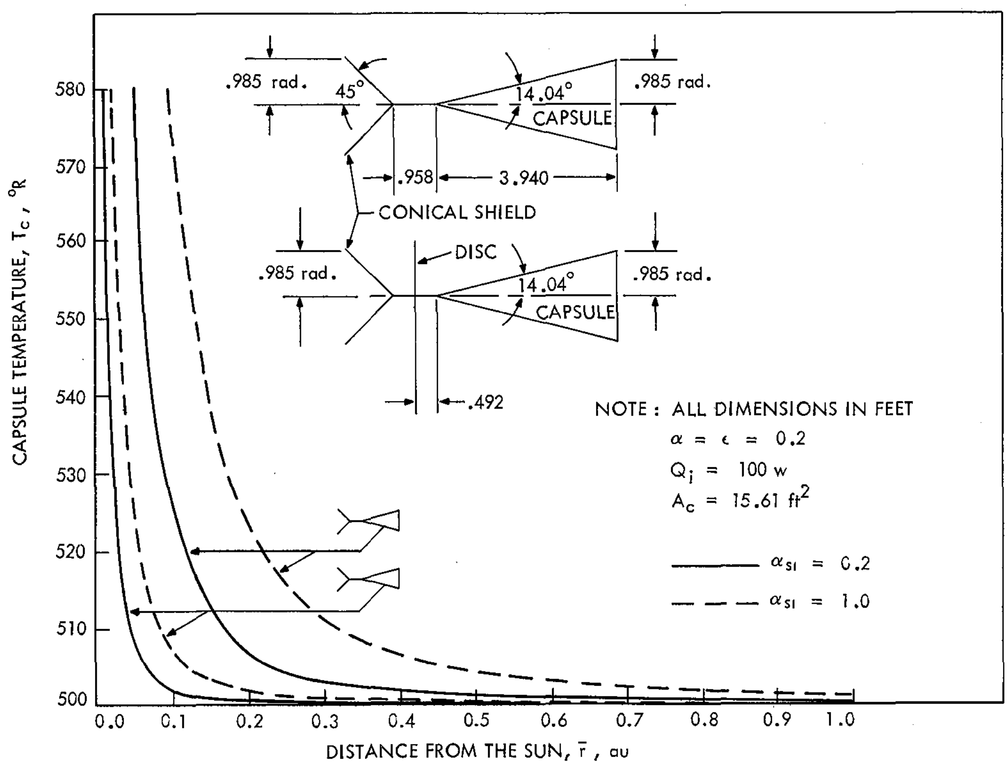

Fig. 5-2. Capsule temperature for the shielded solar probe configurations as a function of distance from the Sun

attainable with an unshielded capsule. Moreover, these small temperature variations can be achieved by conventional materials with no unusual emissive or absorptive properties."

Isolation from solar radiation makes the temperature of the spacecraft bus primarily dependent upon internal electrical power, and the effectiveness of the primary radiating surfaces[1] (Fig. 5-3) in dissipating this power. Because electrical power dissipation varies as a function of mission phase, and because complete solar isolation is not possible, (due to conduction through structural members, etc.), the radiating capability of the primary radiators is modulated by use of temperature actuated louvers. Thus, a relatively constant bus temperature can be maintained over wide ranges of solar intensities and electrical power levels.

A typical bus temperature curve is shown in Fig. 5-4, and is taken from Mariner V Temperature Control Model (TCM) data. The steep portion of the curve at the left is the characteristic temperature response with the louvers completely closed; the steep portion at the right is the response with the louvers fully open. As can be seen, the spacecraft temperatures were somewhat biased toward the cooler control limit to allow greater high temperature margin at encounter with Venus.

5.3 LIMITATIONS

Current Mariner spacecraft have demonstrated the capability, in ground tests, to maintain acceptable bus temperatures over a wide range of solar intensity. Mariner V maintained temperatures between 40 and 80°F at solar intensities of zero and 250 w/ft^2 (Venus intensity), respectively. Mariner '69 has maintained the same temperature range for solar intensities of zero and 133 w/ft^2, respectively. In both cases midcourse maneuver occurs near earth. Heating, due to off-sun orientation during the maneuver, is absorbed as temperature rise within the bus.

[1] On the Mariner Spacecraft, these surfaces are located behind the thermal control louvers shown in Fig. 5-3.

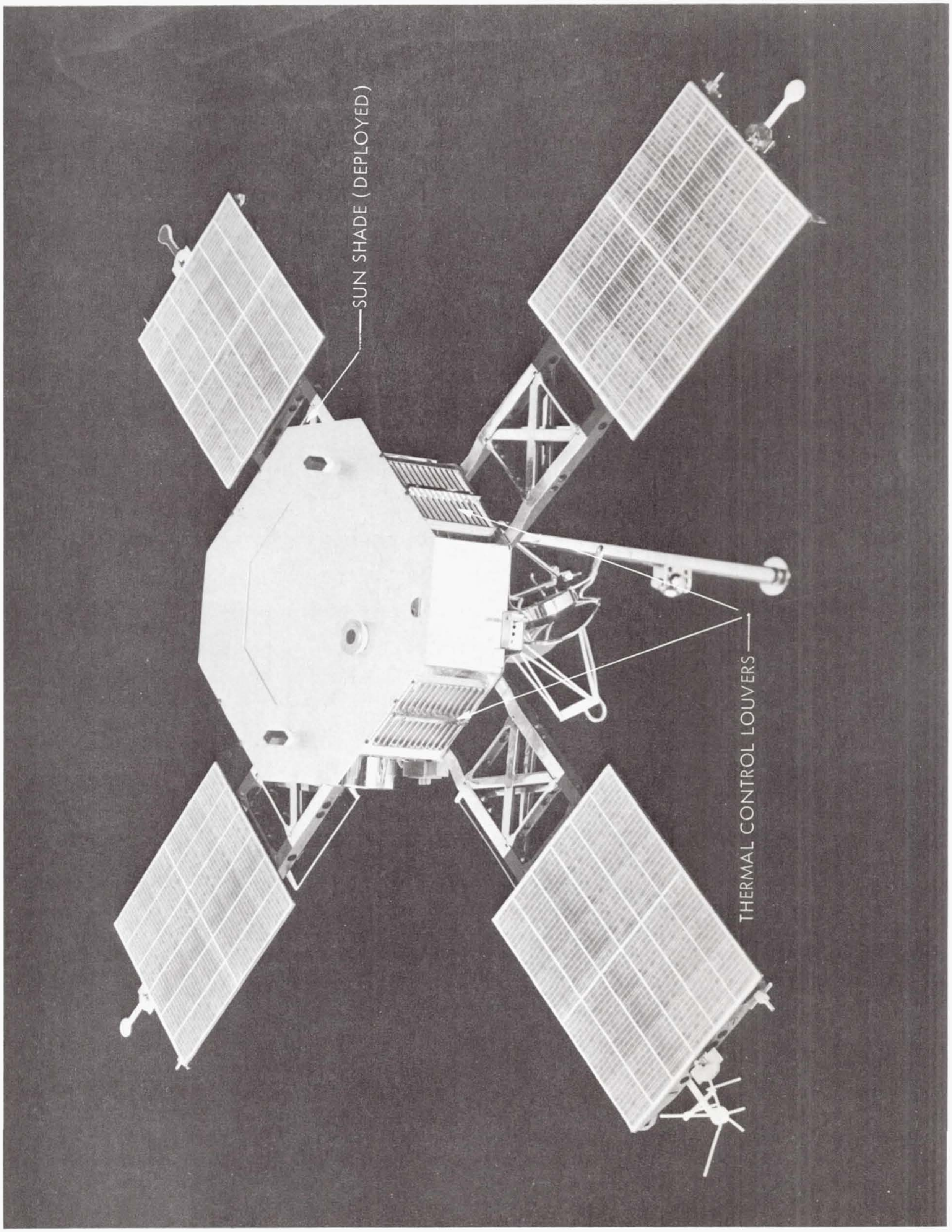

Fig. 5-3. Mariner V sunward

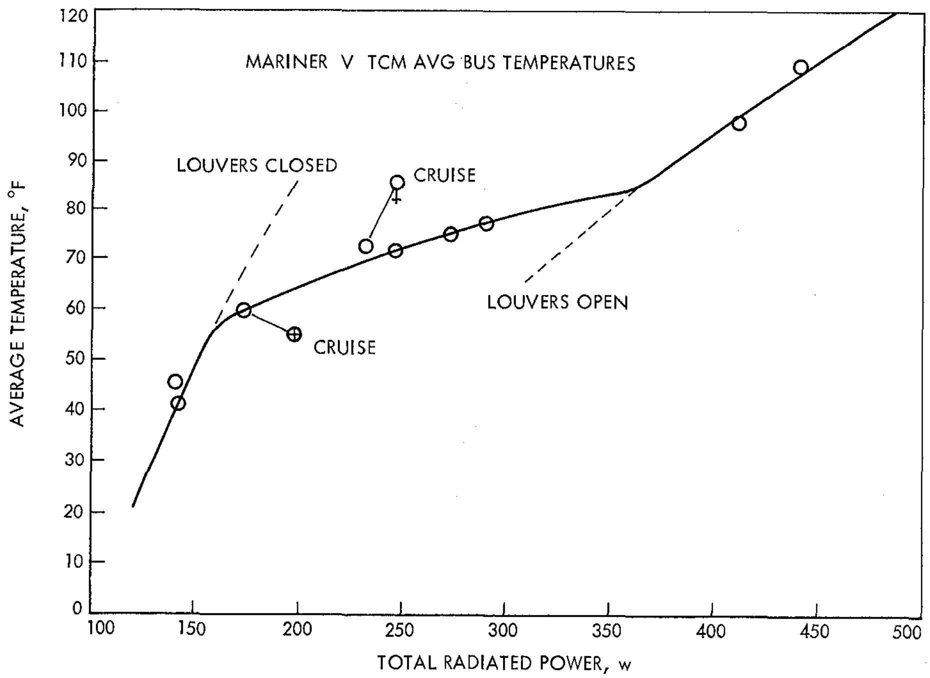

Fig. 5-4. Average bus temperature versus total power

The following types of situations reduce the effectiveness of any isolation scheme:

1) Large solar panels which radiate to the louvered faces of the bus,

2) Structural members which support equipment outboard of the bus,

3) Guidance or scientific sensors which must penetrate the isolation, or

4) A mission requirement to operate in a non-sun oriented condition (such as during a trajectory change maneuver) for long periods of time or near the Sun.

Each particular mission profile and complement of vehicle equipment must be evaluated as to the degree and types of isolation required.

In general, isolating the bus of a deep space probe is easier than a near solar probe because of the inverse square relationship of solar intensity with distance from the Sun. As the vehicle moves away from the Sun, the temperature of sunlit components drops. Beyond roughly 2 AU, the conductive or radiative loss to the component from the bus does not change significantly as the solar intensity drops. Hence, a spacecraft for 30 AU is not significantly different from one for 2 AU.

Going toward the Sun is distinctly different. Associated with the rapidly increasing solar intensity is a corresponding high temperature on the sunlit components. The high temperatures drastically increase conductive and radiative heat transfer to the bus. In addition, the high temperatures and high solar intensity tend to accelerate materials degradation and to reduce structural capability. For extended duration exposure new thermal control materials may be required to withstand the high temperatures and high solar fluxes. Non-sun-oriented maneuvers, if required near the Sun, will be severely restricted as to the time or direction of turn so as to minimize heat addition to the bus.

Ames Research Center in their advanced Pioneer studies have indicated that the decoupling of primary bus electronics is feasible to distances as close to the Sun as 0.1 AU if proper design techniques are applied.

Swingby trajectories, such as those proposed to swingby Jupiter to get to Mercury faster than a direct trajectory, present special problems. On the Jupiter leg of the trajectory, the antenna must point toward Earth on the sun-side of the vehicle while planet sensing instruments must generally point in the opposite direction. As the vehicle passes 1 AU on the way to Mercury, all of the antenna and planet sensing instruments reverse their look directions relative to the direction of the Sun. Shielding and isolation techniques must accommodate these changes in look direction together with the large changes in solar intensity. It is a difficult task, but is not considered impossible.

5.4 WEIGHT SUMMARY

Current temperature control subsystem weights are running roughly 3 to 5% of total spacecraft weight. A weight estimate as a function of perihelion distance for a compact multilayer shield presented in a solar probe mission study by Minneapolis-Honeywell (Report No. CPE 3D-B-35-3) shown in Fig. 5-5 replotted in percent of total spacecraft weight. As can be seen, the heat shield weight required to maintain the payload at 90°F at 0.1 AU is 3.4% of the total spacecraft weight. Combined with the weight required for louvers, etc., this fraction may go up to 10% for close solar probes, but probably not more.

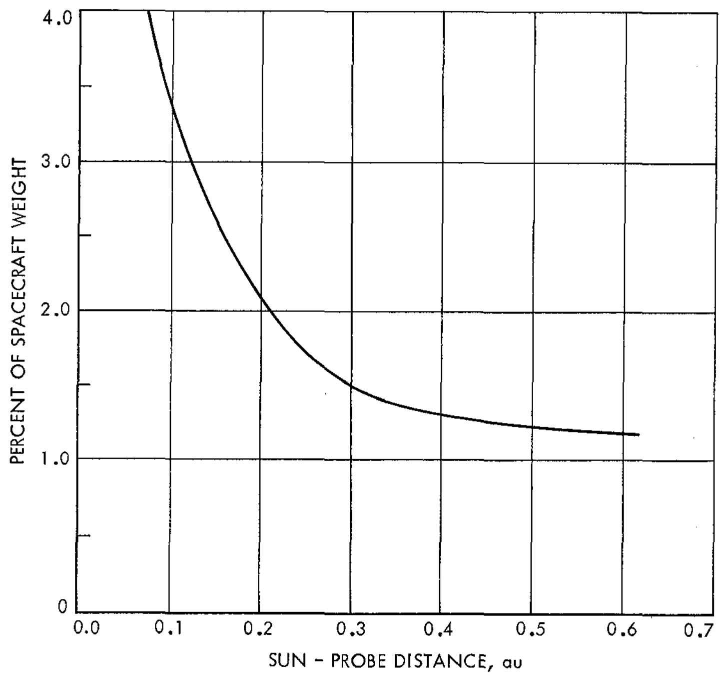

Fig. 5-5. Heat shield weight required to maintain payload at 90°F

SECTION VI

OBJECTIVES AND SENSOR REQUIREMENTS FOR FLYBY MISSIONS

6.1 DEFINITION OF A SENSOR

For the purposes of this study, a sensor is defined as a spacecraft component consisting of a sensing element used to measure phenomena for scientific or engineering purposes. It is exposed directly to the environment being measured and includes all auxiliary data processing equipment at the immediate sensor location.

Three types of sensors will be considered. First, those sensors used on science experiments. Second, those used as part of the attitude control system. Third, those which do not strictly fit the above definition, but are of sufficient importance to warrant consideration and do meet the criteria of direct exposure to the environment, i.e., solar panels or electronics located at an antenna. Scientific experiments represent the greatest variety of sensor types because of the many different kinds of phenomena to be investigated and the variety of missions to be performed. The other sensor types are neither as numerous, nor do they vary as greatly from mission to mission.

6.2 TYPES OF SENSORS REVIEWED

The IITRI studies dealing with scientific objectives for space investigations (Appendix B) were not given the same review as other mission studies, since they did not deal with a specific system design, but only with objectives and instrumentation. Instead, a list of recommended science instruments for space missions to all parts of the solar system (aside from the Moon and Mars) has been abstracted from these IITRI studies and is presented in Table 6-1.

Some of these instruments (such as the aerodynamic-type sensors, gas chromatographs, surface analyzers, seismometers, and life detectors) are only required for atmospheric entry and/or lander missions. The rest are likely candidates for flyby and orbiter missions.

Table 6-1. ITTRI-recommended instrumentation for interplanetary exploration objectives mission

Instrumentation	Comet Missions		Venus		Jupiter	Outer Planets (Multi-Missions to Jupiter, Saturn, Uranus, Neptune, Pluto)		Asteroids		Inter-planetary
	(1)	(2)	Flyby & Orbiter	Probe/ Lander		Flyby	Probe	Flyby	Lander	
Magnetometer										
Rubidium Vapor	X	X	X	X	X	X		X	X	X
Fluxgate	X	X				X		X	X	X
Rotating Coil					X					
Helium Vapor			X							
Micrometeorite Detector										
Acoustic	X	X			X			X	X	
Foil					X					X
Crystal Flash					X					
Pressure Cell					X					
Plasma Probe										
Energy Spectrum	X	X	X							X
Direction	X	X	X							X
Cosmic Ray Telescope					X					X
Solar Proton Detector			X		X					X
Ionization Chamber					X					X
Spectrophotometer		X								

(1) Particles and Fields Only
(2) Full Experimental Payload

Table 6-1. ITTRI-recommended instrumentation for interplanetary exploration objectives mission (contd)

Instrumentation	Comet Missions		Venus		Jupiter	Outer Planets (Multi-Missions to Jupiter, Saturn, Uranus, Neptune, Pluto)		Asteroids		Inter-planetary
	(1)	(2)	Flyby & Orbiter	Probe/ Lander		Flyby	Probe	Flyby	Lander	
IR Spectrometer			X		X	X				
IR Radiometer			X							
Visible Spectrometer			X		X	X				
Visible Polarimeter					X	X				
U.V. Spectrometer			X		X	X				
U.V. Polarimeter					X	X				
Photometer				X				X	X	
Multicolor Photometer			X							
Microwave Radiometer			X		X	X		X	X	
Radar			X		X			X	X	
Television		X	X	X	X	X		X	X	
Aerodynamic-Type Sensors										
Pressure				X						
Density				X						
Temperature				X						
Water Vapor				X						

(1) Particles and Fields Only
(2) Full Experimental Payload

Table 6-1. ITTRI-recommended instrumentation for interplanetary exploration objectives mission (contd)

Instrumentation	Comet Missions		Venus		Jupiter	Outer Planets (Multi-Missions to Jupiter, Saturn, Uranus, Neptune, Pluto)		Asteroids		Inter-planetary
	(1)	(2)	Flyby & Orbiter	Probe/ Lander		Flyby	Probe	Flyby	Lander	
Mass Spectrometer	X	X		X			X			
Gas Chromatograph				X					X	
Neutron Capture Gamma Ray Analyzer									X	
Seismic Experiment				X					X	
Penetrometer				X						
Impactometer				X						
Life Detector							X			
Video Microscope				X						

(1) Particles and Fields Only
(2) Full Experimental Payload

There is no sharp line of demarcation between science experiments for flyby and orbiter missions. Flyby encounter experiments may be flown on orbiters to obtain larger data samples or scan larger surface areas. Those experiments used only for interplanetary cruise measurements would be the same for both flybys and orbiters.

Tables 6-2 through 6-7 list the science experiments proposed for the flyby missions. The listing includes all possible payloads considered, even as alternates, in order that as comprehensive a selection of instruments as possible would be considered, at least initially, in Phase II. Also included are instruments used for cruise science on orbiter and lander missions. All of the experiments from the IITRI list are included, with the exception of those which are only applicable to lander missions.

Flyby science experiments have been grouped in the following categories:

1) Magnetometers
2) Micrometeoroid or Cosmic Dust Detectors
3) Radiation Detectors
4) Optical Instruments
5) Television
6) Miscellaneous

The radiation detectors have a variety of instrument types, so they were further divided according to the sources of the radiation measured (galactic cosmic rays, solar cosmic rays, trapped radiation, and solar wind plasma). The optical instruments are also divided into radiometers/photometers, spectrometers/interferometers and polarimeters, according to whether the electromagnetic radiation has been filtered, dispersed or polarized in its path from source to detector.

The particular requirements selected for tabulation were the weight, power, volume, data, and temperature requirements for each experiment. The requirements were felt to be the most influential in determining the design of an instrument, and hence the most valuable inputs to the Phase II study.

Weight and power requirements are the minimum design requirements for any instrument, and nearly all the studies included estimates of these. Volume information was not as universally stated, but the available information has been presented either as overall dimensions or total volume.

The temperature requirements shown are those identified as limits of each instrument for a specific mission. In some cases these may be absolute limits for the instrument. In other cases, these limits are the estimated temperatures predicted for the electronic compartments, which may not approach the instrument's actual limits.

Data requirements have been expressed as bits/second or bits/sample, since some instruments are event-dependent in their data requirements and do not require continuous transmission of data.

No attempt will be made to evaluate the contents of Tables 6-2 through 6-7 at this time, since they are raw data to be used as the input to the Phase II study effort. The output of this particular Phase II portion would be the following:

1) Identification of equipment required to perform mission objectives within stated functional requirements

2) Listing of currently available equipment meeting the functional requirements

3) Summary of temperature limits, life expectancy, weight, materials, power, configuration, flight qualification status, and other mechanical properties of the equipment.

Table 6-2. Flyby mission science experiments and requirements

MAGNETOMETERS

Mission	Reference see (Appendix B)	Weight (lbs)	Power (Watts)	Data Rate (Bits/Sec)	Volume (in. or in.^{3})	Temperature (^{o}F)	Remarks
Solar Probe	9	4.5	0.5	-------	--------	70 - 80	0.5 to 400 gamma
	20	0.7 (S)* 4 (E)	6.0	-------	3 X 2 dia. (S) 3 cube (E)	32 - 104	Flux Gate, 0 to 128 gamma
	20	2.63	5.5	-------	13 sphere	32 -104 (Gas Cell, 86-122)	Rubidium Vapor, 0-64 gamma
	22	5.0	5.0	0.6	100	150 - 180	Helium
	23	-------	------	-------	--------	---------	Tri-axial
	28	0.75 (S) 5.0 (E)	3.5	-------	6 X 4 dia. (S) 2.5 X 11 X 6 (E)	27 - 207 (S) 27 - 242 (E)	Flux-Gate
	30	1.0 (S) 3.7 (E)	------	-------	---------	77-140 (E)	-------
Venus/Mercury Flyby	15	2.0	------	------	---------	-40 to 130	-------
Mars/Venus Orbiter	13	------	------	------	---------	-22 to 212 (Operating)	Non-operating: -58 to 212 (S) -40 to 212 (E)
Mars Flyby/Lander	3	5.0	5.0	------	---------	0 to 120^{o}F	
Mars Orbiter	12	14.5	5.0	4	4.0	14 to 131 (Operating)	Non-Operating: -58 to 176
Comet/Asteroid	27	6.1	7.0	-------	2 - 33.3	-65 to 130(S), 0 to 150 (E)	
Jupiter Flyby	7	8	7	300	--------	-13 to 167 (Gas Cell, 32 to 122)	Rubidium Vapor
	16	5	4	4	--------	--------	----------
	18	5 2	5 1	21 7	4 X 4 X 6 3 X 1 dia	-30 to 160 -30 to 160	Helium, 0 to 100 gamma Flux Gate, 10^2 to 10^7 gamma
	25	5	4		--------	57 - 86	
	29	5.5	4.0	24	--------	to 200 (S), -40 to 140(E)	
Outer Planets	10 a & b	2.0 6.0	0.1 5.0	3 0.5	-------- --------	-148 to -184 -148 to -180	Flux Gate, 0.1 - 10 gamma Helium, 0.1 - 100 gamma 10^2 - 10^5 gamma

*(S) - Sensor
(E) - Electronics

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Table 6-3. Flyby mission science experiments and requirements

COSMIC DUST/MICROMETEOROID DETECTORS

Mission	Reference (Appendix B)	Weight (lbs)	Power (Watts)	Data Rate (Bits/Sec)	Volume (in. or in^3)	Temperature (^oF)	Remarks
Solar Probe	9	4.0	1.0	-------	-------	40-100	Mass > 10^{-10}g @ 50-100 km/sec
	22	7.0	0.2	0.05	240	150-180	10^2 Dynamic Range
	23	---	---	-------	-------	-------	Acoustic Sensor
	30	1.5	---	-------	-------	74-140	-------
Venus/Mercury Flyby	15	5.0	---	-------	-------	-100 to 250 (S)* 0 to 40 (E)	-------
Mars/Venus Orbiter	13	8	1	20 bits/impact	-------	-58 to 212 (S) -40 to 212 (E) } Operating	Non-operating: -103 to 212 (S) - 58 to 212 (E)
Mars Flyby/Lander	3	2.5	0.2	-------	-------	0-12	Cosmic Dust
		8.0	0.5	-------	-------	0-120	Micrometeroid
Mars Orbiter	12	5.0	0.5	3	-------	14 to 131 (Operating)	Non-operating: -58 to 176
Comet/Asteroid	27	2.3	0.2	0.2 - 5	-------	-40 to 160	--------
Jupiter Flyby	7	2.5	0.2	0.1	-------	-148 to 392 (S), -40 to 212 (E)	--------
	16	5	1	3	-------	--------	--------
	18	3	---	7 bits/sample	2 X 3 dia	-30 to 160	High Sensitivity
		6	---	7	2 X 8 dia	-30 to 160	Low Sensitivity
		30	5	38	12 X 18 X 24	-30 to 160	Meteoroid Monitor
		10	2	320	10 X 4.5 dia	-30 to 160	Optical Meteoroid Detector
	25	5	1.5	Cruise: 10-100(1) Encounter:100-500	---	57-86	-------
	29	4.0	1.0	14	-------	-40 to 140	-------
Outer Planets	10 a & b	5.0	0.1	0.1	-------	-148 to -184	-------

*(S) - Sensor
(E) - Electronics

(1) Entire Payload

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Table 6-4(a). Flyby mission science experiments and requirements

RADIATION DETECTORS-GALACTIC COSMIC RAYS

Mission	Reference (Appendix B)	Weight (lbs)	Power (Watts)	Volume (in. or in^3)	Data Rate (Bits/Sec)	Temperature (^{o}F)	Remarks
Solar Probe	9	7.0	1.0	-------	-------	-------	Coincidence Telescope
	22	2.5	0.27	360	0.15 - 0.30	150 - 180^{o}F	Ionization chamber, GM counter for high energy protons (>10^8ev)
Mars/Venus Orbiter	13	---	---	-------	-------	-22 to 122 (operating) -58 to 149 (non-operating)	High energy proton monitor
Jupiter Flyby	7	4.0	0.6	-------	0.1	14 to 122	High energy protons (>1 Bev)
		18.0	2.0	-------	0.2	-22 to 122 (S)*, -22 to 167 (E)	Cosmic Ray spectrum analyzer (1 - 400 Mev)
	18	4	0.5	3 X 4 X 4	42	-30 to 160	High energy proton detector
	25	5 - 7	1.5 - 2	-------	10 - 100 (cruise) (1) 100 - 500 (encounter)	57 - 86	-------
	29	6.0	2.0	-------	5	-30 to 100 (S), -4 to 167 (E)	
Outer Planets	10 a & b	5.0	1.0	-------	0.1	-148 to -184	Cosmic Ray Telescope

*(S) - Sensor
(E) - Electronics

(1) Entire Payload

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Table 6-4(b). Flyby mission science experiments and requirements

RADIATION DETECTORS SOLAR COSMIC RAYS

Mission	Reference (Appendix B)	Weight (lbs)	Power (Watts)	Volume (in. or in^3)	Data Rate (Bits/sec)	Temperature (°F)	Remarks
Solar Probes	9	9.5	1.5	-------	-------	40 to 100	Coincidence Telescope (10-200 Mev)
		3.0	1.0	-------	-------	-------	Solar X-Rays (ionization chamber)
		20.0	1.0	-------	-------	-------	Neutron Detector (1-10 Mev)
		2.0	0.5	-------	-------	-------	Energetic Electrons
	20	2.5 (S)* 7.5 (E)	2.0 (S) 1.5 (E)	12 X 2 dia 12 X 6 dia	-------	32-104	Phoswitch & Scintillation telescope
		8.0	0.5	864	-------	32-104	X-Ray Scanner (0 - 0.1 Å)
	22	3.3	0.78	480	0.25 - 0.9	150 - 180	Semi-conductor counter
		---	---	---	0.1	150 - 180	X-Ray ionization chamber (1 Å)
	23	---	---	---	-------	-------	X-Ray, charged particle, & neutron detectors
	28	4.5	1.5	10 X 10 X 10	-------	27 - 242	Cosmic Ray Detector
		3.3	3.5	14 X 7 dia	-------	27 - 242	Phoswitch (1 - 20 Mev)
	30	4.6	---	---	-------	73 - 104	Experimenter: Simpson
		4.8	---	---	-------	79 - 122	Experimenter: McCracken
Venus/Mercury Flyby	15	8.0	---	---	-------	-40 to 122	Cosmic Ray telescope
		5.0	---	---	-------	14 to 122	Low energy proton detector
		5.0	---	---	-------	14 to 122	Fast neutron & gamma ray detector
		2.5	---	---	-------	-22 to 158	Ionization chamber
Mars/Venus Orbiter	13	---	---	---	-------	-22 to 158 (O)** -58 to 194 (N)	Particle flux experiment
		---	---	---	-------	-22 to 257 (O) -58 to 257 (N)	Ionization chamber
		---	---	---	-------	-22 to 122 (O) -58 to 149 (N)	Medium energy proton monitor
		---	---	---	-------	-40 to 122 (O) -58 to 149 (N)	Cosmic Ray spectrum analyzer
Mars Flyby	3	1.3	0.1	---	-------	0 - 120	Ionization chamber
Comet/Asteroid	27	2.6	0.5	---	0.2 - 5	-22 to 158	Ionization chamber

**(O) - Operating
(N) - Non-operating
*(S) - Sensor
(E) - Electronics

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Table 6-4(b). Flyby mission science experiments and requirements (contd)

RADIATION DETECTORS SOLAR COSMIC RAYS

Mission	Reference (Appendix B)	Weight (lbs)	Power (Watts)	Volume (in. or in^3)	Data Rate (Bits/Sec)	Temperature (^oF)	Remarks
Jupiter Flyby	18	1.4	0.1	5 sphere	14 bits/sample	-30 to 160	Ion chamber
		2.6	0.35	4 X 5 X 6	42 bits/sample	-30 to 160	Particle Flux meter
		3.0	1.0	4 X 5 X 5	84 bits/sample	-30 to 160	Medium energy proton monitor
		5	3	4 X 5 X 6	14 bits/sample	-30 to 160	X-Ray detector
	25	4.5-5.5	1.5	-------	-------	57 - 86	
	29	6.0	2.0	-------	32	-4 to 104 (S) -4 to 167 (E)	
Outer Planets	10 a & b	5.0	1.0	-------	0.1	-148 to -184	Solar proton detector (100 Kev-10 Mev)
		3.0	0.2	-------	0.1	-148 to -184	Ionization chamber

**(O) - Operating
(N) - Non-operating
*(S) - Sensor
(E) - Electronics

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Table 6-4(c). Flyby mission science experiments and requirements

TRAPPED RADIATION DETECTORS

Mission	Reference (Appendix B)	Weight (lbs)	Power (Watts)	Volume (in. or in^3)	Data Rate (Bits/Sec)	Temperature (^oF)	Remarks
Solar Probe	9	2.0	0.5	---	---	---	Semi-conductor detector
Venus/Mercury Flyby	15	2.6	---	---	---	14 to 122	Trapped radiation detector
Mars/Venus Orbiter	13	---	---	---	---	-22 to 149 (O) -58 to 149 (N)	---
Mars Flyby	3	2.5	0.35	---	---	0-120	Particle Flux Detector
Mars Orbiter	12	2.5	0.5	---	2	14 to 131 (O) -22 to 140 (N)	---
Comet/Asteroid	27	2.0	0.4	---	0.2 - 5	---	Geiger - Muller Tube
	7	2.5	0.2	---	0.1	14 to 122	Energetic Particle Detector
	16	8.0	3.0	---	6	---	Particle counters: Low energy electrons, relativistic electrons, energetic protons
Jupiter Flyby	18	4.0	0.7	4 X 5 X 5	28 bits/sample	-30 to 160	Trapped radiation analyzer
	25	6.0	1.5	---	---	57 - 86	Jovian X-Rays
		4-6	1.0	---	---	57 - 86	Trapped radiation
	29	6.0	1.0	---	24	-40 to 140	Trapped radiation
		5.0	2.0	---	30	-30 to 100 (S) -30 to 140 (E)	Auroral detector

*(S) - Sensor
(E) - Electronics
**(O) - Operating
(N) - Non-operating

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Table 6-4(d). Flyby mission science experiments and requirements

SOLAR WIND (PLASMA) INSTRUMENTS

Mission	Reference (Appendix B)	Weight (lbs)	Power (Watts)	Volume (in. or in^3)	Data Rate (Bits/Sec.)	Temperature ($^{\circ}$F)	Remarks
Solar Probe	9	4.5	0.5	---	---	-328 to 482	Faraday Cup
		8.0	3.0	---	---	40 to 100	Electrostatic Analyzer
	20	4.5	1.0	250	0.4	140 to 392 (S)* 77 to 248 (E)	Plasma Probe
	22	19.2	4.0	1120	2.4 - 3.6	150 to 180	Electrostatic Analyzer (Protons: 50 to 10^4 ev Electrons: 10 to 100 ev)
	23	---	---	---	---	---	Wide and Narrow Angle Plasma detectors
	27	4.5	1.0	10 X 10 X 10	---	27 to 242	Plasma Probe
	30	4.0	---	---	---	84 to 122	Experimenter: Wolfe
		4.5	---	---	---	203 to 302 (S) 79 to 165 (E)	Experimenter: Bridge
Venus/Mercury Flyby	15	---	---	---	---	14 to 176	Solar Plasma Probe
Mars/Venus Orbiter	13	---	---	---	---	-4 to 176 (O)** -67 to 212 (N)	Electrostatic Analyzer
Mars Orbiter	12	10.0	12.0	---	1.0	14 to 131 (O) -58 to 176 (N)	---
Comet/Asteroid	27	7.0	3.5	---	0.8 to 45	14 to 175	---
Jupiter Flyby	7	7.0	2.5	---	0.1	14 to 176	---
	18	7.0	7	---	---	-30 to 160	Low Energy Plasma Analyzer
	25	5-8	1.5-4	---	---	57 - 86	Plasma Probe
		4	1.0-2.0	---	---	57 - 86	Thermal Plasma Detector
	29	5.5	1.5	---	24	5 to 122	---
Outer Planets	10 a & b	4.0	1.0	---	1.5	-148 to -184	Faraday Cups

*(S) - Sensor
(E) - Electronics
**(O) - Operating
(N) - Non-operating

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Table 6-5(a). Flyby science experiments and requirements

OPTICAL EXPERIMENTS - RADIOMETERS/PHOTOMETERS/POLARIMETERS

Instrument	Reference (Appendix B)	Weight (lbs)	Power (Watts)	Volume (in. or in^3)	Data Rate (Bits/Sec.)	Temperature (^oF)	Remarks
Microwave Radiometer	7	28.0	6.0	---	1.3	-20 to 150 (O) -58 to 212 (N)	Juipter Flyby -(4, 8, 13 & 20µ wave lengths)
	16	15.0	5.0	---	160 - 240	---	Jupiter Flyby (21 cm to 4 mm)
	18	18.0	4.0	8 X 8 X 14	70 bits/sample	-30 to 160	Jupiter Flyby (1 to 5 cm)
	10 a & b	10.0	1.0	---	1.0	-148 to -184	Outer Planets (1 to 10 cm)
IR Radiometer	7	5.0	3.0	---	0.7	-4 to 104 (O) -58 to 212 (N)	Jupiter Flyby (10 to 20µ)
	18	3.0	3.0	4 X 5 X 7	28 bits/sample	-30 to 160	Jupiter Flyby (1 - 10µ)
	29	3.0	3.0	---	28	-30 to 140	Jupiter Flyby
Scanning Radiometer	25	5.0 - 6.0	4.0	---	---	57 to 86	Jupiter Flyby
Photometer	10 a & b	5.0	1.0	---	5.0	-148 to -184	Outer Planets
Visible Photometer	7	2.0	1.5	---	0.2	-4 to 104 (O) -58 to 212 (N)	Jupiter Flyby (5000 Å)
White Light Photometer	23	---	---	---	---	---	Solar Probe (Z odiacal Light)
Lyman-Alpha (UV) Photometer	22	10.0	1.0	540	---	150 to 180	Solar Probe
	23	---	---	---	---	---	Solar Probe
	27	3.0	3.0	---	1.0	0 to 1222	Comet/Asteroids
UV Photometer	16	5.0	1.0	---	3.0	---	Jupiter Flyby (1216 Å, 3888 Å, 6402 Å)
	27	3.0	3.0	1.0	---	0 to 120	Comet/Asteroids
White Light Corona Meter	20	5.0	0.5	20 X 2 dia	---	32 to 104	Solar Probe (Polarimeter)
Photometer/Polarimeter	18	6.0	5.0	5 X 6 X 7	63	-30 to 160	Jupiter Flyby

*(S) - Sensor
(E) - Electronics
**(O) - Operating
(N) - Non-operating

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Table 6-5(b). Flyby science experiments and requirements

OPTICAL EXPERIMENTS - SPECTROMETERS/INTERFEROMETERS

Instrument	Reference (Appendix B)	Weight (lbs.)	Power (Watts)	Volume (in. or in^3)	Data Rate (Bits/Sec.)	Temperature (^oF)	Remarks
Microwave Spectrometer	15	50.0	---	---	3.3	14 to 122	Venus/Mercury Flyby (3-30 mm, 40 mm)
IR Spectrometer	3	29	---	---	---	0 to 120	Mars Flyby
	7	16.0	5.0	---	33.3	-4 to 104 (O), -58 to 176 (N)	Jupiter Flyby (5-30μ, 1μ resolution)
	10 a & b	10.0	10.0	---	5.0	-140 to -184	Outer Planets (2-50μ, 1μ resolution)
	12	26.0	5.0	3.0	7.0	Operating: -121 to 50 (S) -110 to 50 (E) Non-operating: -110 to 140	Mars Orbiter
	18	29.0	7.0	15 X 12 dia	420	-30 to 160	Jupiter Flyby (1 - 10μ)
IR Interferometer	12	26.0	8.0	4.0	1730	18 to 23 (O), -58 to 140 (N)	Mars Orbiter
Mars Scanner (IR)	12	13.0	9.4	3.0	3500	-40 to 4 (O), -58 to 158 (N)	Mars Orbiter
Visual Spectrometer	18	22.0	---	---	---	-30 to 160	Jupiter Flyby
UV-Visible Spectrometer	7	20.0	10.0	---	66.7	-4 to 104 (O), -40 to 257 (N)	Jupiter Flyby (1000-6000 Å, 10 Å resolution)
	10 a & b	20.0	10.0	---	5.0	-148 to -184	Outer Planets (1000 - 10,000 Å)
UV Spectrometer	12	24.0	12.0	4.0	2000	14 to 104 (O), -22 to 140 (N)	Mars Orbiter
	15	23.0	---	---	3.3	14 to 104	Venus/Mercury Flyby (1100 - 4500 Å)
	27	22.0	12.0	---	33.3	0 to 120	Comet/Asteroid

*(S) - Sensor
(E) - Electronics
**(O) - Operating
(N) - Non-operating

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Table 6-6. Flyby science experiments and requirements

TELEVISION SYSTEMS

Mission	Reference (Appendix B)	Weight (lbs)	Power (Watts)	Volume (in. or in^3)	Data Rate (Bits/Sec.)	Temperature ($^{\circ}$F)	Remarks
Solar Probe	9	10.0	12.0	---	---	---	Alternate Experiment for mission
Venus/Mercury Flyby	15	31.0	---	---	100,000	14 to 122	High and Low Resolution Cameras
Mars Flyby	3	20.0	15.0	---	---	0 to 120	---
Mars Orbiter	12	23.0	10.0	7.5	8000 - 64,000	14 to 131	---
Comet/Asteroid	27	35.0	16.0	---	1000	0 to 120	---
Jupiter Flyby	7	15.0	10.0	---	20,000	Optics: 14 to 158 (O), -4 to 257 (N) Electronics: -4 to 104 (O), -58 to 212 (N)	1000 lines/frame, 10° view angle, 32 shades of grey
	18	6.0	15.0	2 X 3 X 4	2 X 10^5 bits/sample	-30 to 160	Low Resolution
		30.0	15.0	15 X 7 dia	1.8 X 10^6 bits/sample	-30 to 160	High Resolution
		10.0	6.0	3 X 3 X 5	9.6 X 10^5 bits/sample	-30 to 160	Infrared
	29	10.0	10.0	---	250	-270 to 212 (S), -30 to 140 (E)	---
Outer Planets	10 a & b	10.0	10.0	---	20	-148 to -184	Mariner Type, 240,000 Bits/frame

*(S) - Sensor
(E) - Electronics
**(O) - Operating
(N) - Non-operating

Table 6-7. Flyby science experiments and requirements

MISCELLANEOUS EXPERIMENTS

Experiment	Reference (Appendix B)	Weight (lbs.)	Power (Watts)	Volume (in. or in^3)	Data Rate (Bits/Sec.)	Temperature ($^{\circ}$F)	Remarks
Radio Noise Receiver	9	5.0	2.0	---	---	40 to 100	Solar Probe-Coronal Electron Density
		8.0	0.5	---	---	---	VLF Radio Noise - Antenna
		2.0	1.0	---	---	---	Electron Density - VLF Antenna Impedance
	20	10.0	1.5	250	---	32 to 104	Solar Probe - Coronal Electron Density
	22	18.0	23.0	720	---	150 - 180	Coronal Electron Density
	23	---	---	---	---	---	Solar Probe - Integrated Electron Density
	28	4.5	2.0	16	---	---	Solar Probe - Electron Density
	30	5.3	---	---	---	77-117	Solar Probe - Radio Propagation Receiver
	12	6.5	2.0	3.0	1.0	14 to 131 (O), -58 to 176 (N)	Mars Orbiter - RF Noise Detector
	7	5.0	2.0	---	0.6	-20 to 150 (O), -40 to 147 (N)	Jupiter Flyby - Locate 10 meter wavelength source
		5.0	2.0	---	0.3	-20 to 150 (O), -40 to 157 (N)	Null Radio Seeker
	18	3.0	2.0	4 X 5 X 6	70 bits/sample	-30 to 160	Jupiter Flyby -(1 to 20 Mc)
	25	6.0	1.5	---	---	57 - 86	Jupiter Flyby - Radio Propagation Experiment
	29	6.5	1.5	---	---	to 200	Jupiter Flyby - Radio Propagation
Radio Occulatation Experiment	12	5.5	1.6	---	2	14 to 131 (O), -58 to 176 (N)	Mars Orbiter - Occulation Beacon
	16	included in S/C radio	---	---	---	---	Jupiter Flyby - Atmospheric Scale Height
	25	---	---	---	---	---	Jupiter Flyby - Uses S/C Radio System
	29	---	---	---	72	---	Jupiter Flyby
Bi-Static Radar	13	---	---	---	---	-4 to 176 (O),-67 to 212 (N)	Interplanetary Electron Density
	7	15.0	8.0	---	0.3	-20 to 150 (O),-40 to 157 (N)	Radar Scattering at 40MC
	18	5	---	---	---	-30 to 160	Jupiter Flyby

*(S) - Sensor
(E) - Electronics
**(O) - Operating
(N) - Non-operating

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Table 6-7. Flyby science experiments and requirements (contd)

Experiment	Reference (Appendix B)	Weight (lbs)	Power (Watts)	Volume (in. or in^3)	Data Rate (Bits/Sec.)	Temperature (^{o}F)	Remarks
Radar Altimeter	7	25.0	10.0	---	0.3	-20 to 150 (O),-40 to 157 (N)	0.5 to 1 meter wavelength
Top-side Sounder	18	25.0	10.0	5 X 5 X 14	440	-30 to 160	Jupiter Flyby
Resonance Sounder	25	5.0	0.5	---	---	57 to 86	Jupiter Flyby
Radio Astronomy	25	4.0 - 5.0	2.0 - 3.0	---	---	57 to 86	Jupiter Flyby
Ion-Electron Trap	27	8.0	2.0	---	0.2 - 45	---	Comet/Asteroid Mission
Mass Spectrometer	9	5.0	5.0	---	---	---	RF M. S. - Composition of Interplanetary Gas
	20	15.0	20.0	9 X 20 X 2	0.4	32 to 104	Mass + Flux of Interplanetary Gas
	23	---	---	---	---	---	Ion composition of plasma
	27	8.0	8.0	---	150	0 to 120	Ion-mass spectrometer
	18	40.0	5.0	24 X 24 dia	540	-30 to 160	Impact Mass Flash Spectrometer (Particle Composition)

SECTION VII

PLANETARY LANDER STEADY STATE TEMPERATURE CONSIDERATIONS

7.1 ENERGY EXCHANGE

In general, a lander will exchange energy with the Sun, the sky, the planet's atmosphere and the planet's surface. The energy balance may be expressed as:

$$q_e + q_i = C_{th} \frac{dT}{d\theta} \quad (7\text{-}1)$$

where

q_e = Net external energy flux absorbed/emitted as defined below

q_i = Internal heat generated

C_{th} = Thermal capacity of the lander

T = Lander temperature

θ = Time

The term $\left[C_{th} \frac{dT}{d\theta}\right]$ is the rate of change in sensible heat of the lander and accounts for stored energy during transient condition. This term goes to zero at steady state.

The net energy exchange with the external environment is described by the relation:

$$q_e = A_S F_S S + A_P F_P P + A_{CV} h\,(T_\alpha - T) + A_{cd} C(T_P - T) - A\sigma\epsilon T^4 \quad (7\text{-}2)$$

where

A_S = Surface area interacting with solar energy

F_S = Solar radiation interchange factor

S = Solar energy flux

A_P = Surface area interacting with planet emission

F_P = Planet emission radiation interchange factor

P = Planetary emission flux

A_{CV} = Surface area interacting with heat conduction with the planet's atmosphere

h = Convection heat exchange factor

T_α = Planet's atmospheric temperature

A_{cd} = Surface area interacting with heat conduction with the planet's surface

C = Thermal conductance

T_P = Planet's surface temperature

ϵ = Emittance of the lander at temperature T

A = Emitting surface area

σ = Stefan-Boltzmann Constant

Equation 7-2 represents the summation of the absorbed solar radiant energy both direct and reflected (albedo), the absorbed planetary emitted radiant energy including both the sky and surface radiation, the energy exchanged with the atmosphere by convection, the energy exchanged with the planet's surface by conduction and the energy emitted by the lander.

7.1.1 Simple Lander

Considering a simple lander, in which internal power dissipation is negligible, equation 7-1 reduces to:

$$q_e = C_{th} \frac{dT}{d\theta} \quad (7\text{-}3)$$

Therefore, the net energy exchange, expressed by equation 7-2, may be written as:

$$A_S F_S S + A_P F_P P + A_{CV} h(T_\alpha - T) + A_{cd} C(T_P - T) = \epsilon A \sigma T^4 + C_{th} \frac{dT}{d\theta} \quad (7\text{-}4)$$

The near surface (6 to 10 ft) atmospheric temperature (T_α) may be approximated by the planet surface temperature (T_P). This is a realistic first order assumption since there is a close relationship between ground temperature and near-ground atmospheric temperature. Compared to the total range of a planet's temperature, the difference expected between them is small.

The simplified equation resulting from this assumption is:

$$A\epsilon\sigma T^4 + C_{th}\frac{dT}{d\theta} = A_S F_S S + A_P F_P P + A_{cv} h(T_p - T) + A_{cd} C(T_p - T) \qquad (7\text{-}5)$$

The term describing convection heat transfer $\left[A_{cv}h(T_p - T)\right]$ is the most significant term in equation 7-5 for planets with atmospheres, since it tends to "force" the lander to the planet temperature. For example, if the lander temperature was initially below the planet temperature, heat would be added to the lander by convection from the atmosphere causing the lander temperature to approach the planet temperature.

Similarly, the conduction heat transfer, described by the term $\left[A_{cd}C(T_P - T)\right]$, will also "force" the lander to the planet temperature. Because the conductance between the lander and the planet is low for most expected planet surface conditions, heat conduction between the planet and the lander is probably negligible compared to the heat transferred by convection. However, this would not be the case if, for example, the planet surface was fluid and made intimate contact with the lander.

The absorbed planetary thermal radiation $\left[A_P F_P P\right]$ will also tend to cause the lander to attain the planet surface temperature. However, because the source of this energy is the planet's temperature, it would not cause the lander temperature to be higher than the planet temperature. This term is considered to be less significant than convection.

The incident solar energy absorbed $\left[A_S F_S S\right]$ could cause variations from the planet surface temperature. As on Earth, dark objects will get warmer than the atmosphere when subjected to solar radiation. The magnitude of this variation is determined by the configuration and surface thermal properties of the lander. Naturally selection of the properties for use on a lander will be directed towards minimizing this effect for the high temperature cases (Venus and Mercury). This term will be negligible for the outer planets (Jupiter and beyond) due to the small value of S.

It may be concluded then, that when steady state conditions are finally attained, a lander, dissipating small amounts of power, will be roughly at the same temperature as the planet. This implies that for planets with atmospheres (Venus, Mars, etc.) the lower temperature limit for a lander will likely be the planet surface temperature.

For planets without atmospheres the lander temperature is only very grossly related to the planet surface temperature by the absorbed planetary thermal radiation and conduction. Because of this, the planet surface temperature, at best, only very generally describes the lander temperature. The lander temperature depends to a large extent on the thermal properties and configuration selected for the lander.

7.2 PLANET SURFACE THERMAL PARAMETERS

Since a non-power dissipating lander temperature is approximately the same as the planet, the maximum and minimum planet surface temperatures define the expected lowest high and lowest low lander temperatures for planets with atmospheres. Therefore, a listing of the temperature extremes for all of the planets is an approximate indication of the expected temperature extremes for the corresponding lander.

Table 7-1 presents a list of estimates of the surface temperatures for each of the planets and their major satellites. The range of temperatures, such as 477 to 666°F for the maximum surface temperature of Mercury, indicates the range of uncertainty in present data. The rotation period is given for each planet to indicate the time between temperature extremes. The atmospheric density is listed to indicate whether convective heat transfer can take place. In the case of planets without atmospheres, the planet surface temperature does not define the expected upper and lower lander temperatures since extreme variations from the planet surface temperatures are possible depending on the temperature control system used.

7.3 EXPECTED LANDER TEMPERATURES

From the data contained in Table 7-1 it is obvious that the Mercury and Venus lander missions, with maximum possible surface temperatures of 666 and 926°F respectively, would present high temperature thermal problems. The temperature of a lander will be even higher than those shown because of electrical power dissipation within the system. Low temperature problems could occur for lander missions to Jupiter and beyond, with planet temperature decreasing with increased solar distance, due to the decreasing solar flux.

The actual situation is not as dismal as it appears in Table 7-1. Generally, the lander thermal properties can be selected to reduce the possible temperature extremes. By using spectrally selective surface coatings, RTG waste heat, insulation and thermal shields, and controlling internal heat dissipation, it is possible to exercise some control over these temperature extremes.

A temperature control system for a Mercury mission may isolate the lander from the sun and the high temperature planet surface and depend on heat dissipation to interplanetary space to maintain internal electronics temperatures below the +666°F indicated in Table 7-1. This is possible because of the absence of an atmosphere on Mercury.

In the case of cold missions (Jupiter and beyond) RTG waste heat can be used most effectively to maintain lander temperatures warmer than the planet surface temperatures. It is felt that RTG heating combined with internal power dissipation could provide acceptable room temperatures for all internal electronics. These cold lander missions probably present no greater thermal problems than flyby missions to the same planet.

The techniques suggested so far either have been successfully flown or are a reasonable consequence of the lander design (such as using RTG heating for missions in which RTG power supplies are required). An exception to the temperature control approach appears to exist for Venus lander missions. A long term Venus lander temperature control system would require a refrigeration system to maintain reasonable temperatures. Refrigeration systems for landers would require extensive development and could cause an exorbitant increase in the temperature control weight and power consumption. In this case, high temperature electronic components would be necessary if a practical thermal control system were not developed.

Table 7-1. Planet surface thermal parameters*

Planet	Surface Temperature (°F)		Rotation Period	Atmosphere	Ref.
	Day Max.	Night Min.			
Mercury	477 to 666	-136 to -436	88 days	none	1, 2
Venus	512 to 926	512 to 926	243 days	dense	3
Mars	8 to 120	-64 to -153	24.62 hours	thin	7
Jupiter	-230 to 86	-240 to -308	9.93 hours	dense	1, 2, 4
Moons					
Io	-217	-	-	none	4
Europa	-206	-	-	none	4
Ganymede	-181	-	-	none	4
Callisto	-157	-	-	none	4
Saturn	-202 to -229	-	10.23 hours	dense	2, 5
Moon					
Titan	-230	-	-	moderate	6
Uranus	-274	-	10.82	dense	5, 2
Neptune	-264	-	15 hours	dense	2, 1
Pluto	(-346) to (-325)	-406	6.4 days	none	1, 6

Ref. 1 Weil, Lunar and Planetary Surf. Cond., Ap. J., Vol. 149, Sept, 1967
2 Leaflet 345, Astron. Soc. of the Pacific, 1958
3 Venus Atmos. Design Criteria Monograph
4 NASA SP-3031
5 Icarus, Vol. 5, P-478, 1966
6 Allen, Astrophysical Quantities
7 Voyager Environ. Standards

*Temperatures are infrared and equilibrium values, parentheses indicate radiometric values.

SECTION VIII

CURRENT PLANETARY LANDER CAPABILITY

Mission studies that were reviewed envisioned spacecraft landings only on Mars and Venus. Tables 8-1 and 8-2 summarize the capabilities of these proposed landers in terms of operational lifetime and the spacecraft constraints limiting this lifetime.

Operational lifetimes on the Martian surface varied from less than a day to two years. The lifetime was more a function of the complexity of the mission objectives than of any environmental constraints. The short duration missions were of the small survival capsule type concerned mainly with entry atmospheric measurements. Landed weights ranged from 138 to 1186 lb, with the shorter-lived landers weighing less. The thermal control and power systems also varied with mission duration. The longer operational times were associated with active thermal controls and an RTG power source. The survival sphere described in JPL TM 33-236 was designed for more extreme temperature requirements (-140 to +150°F) than the other Mars landers, which tended more to a room-temperature environment. The Automated Biological Laboratory had the most severe constraints on internal temperature (4°C ±3°C) because of the chemical reactions in the experiment package, and the most complex thermal control system of any lander.

By way of a comparison of lander systems, Surveyor, a successful unmanned lunar lander, had a landed weight of 600 lb, operated for 40 days, and maintained temperatures inside the thermally controlled compartments of between 40° and 130°F. Thus, in these features, it is the same as a "nominal" Mars lander from Table 8-1.

Because of the expected high surface temperature, operational lifetime and complexity of Venus missions were not nearly as great as for Mars missions. Most of the Venus landers were capsules designed to make atmospheric measurements during the descent to the surface, with survival after impact not a major objective.

The AVCO study referenced here (RAD-TR-63-34) was done in 1963 and shows an optimism about surviving Venus surface operations that is not present in studies done later with better information about the planet. (An example of these later studies is documented in JPL IOM VMOS67-68 which was completed too recently to be reviewed.)

Weights of Venus landers varied from 413 lb for the AVCO study to 5 lb for probes dropped from the Buoyant Venus station. Thermal control systems relied mainly on insulation, but phase-change active systems were proposed for two landers. Estimated temperature ranges were similar to Mars landers, but the period of time for which these limits could be maintained was much shorter. It is clear that long-life Venusian landers are currently beyond the state-of-the-art and should be studied extensively to determine the likely thermal requirements for electronics.

It should be noted that factors limiting the operational life of planetary landers are not due entirely to the planet's surface temperature. Such factors as mission trip time, sterilization requirements, entry heating, surface winds, and atmospheric heat transfer parameters may be as significant.

In particular, sterilization and entry heating pose their own peculiar temperature requirements. Heat sterilization methods now in use require a temperature soak for about 24 hours (depending upon the size of the item to be sterilized) at 125°C. This is in a non-operating condition.

Entry heating produces high temperatures, but for a short duration and under operating conditions. Because of its greater density, the Venus atmosphere produces more severe entry heating conditions than those encountered in Martian entry. This condition is likely to result in severe mission constraints on Venus entry vehicles.

Table 8-1. Current lander capacity

Mars

Reference	Survival Time on Surface	Landed Weight (LB)	Type of Thermal Control System	Temperature Limits	Constraints on Operation
a) JPL, EPD-139	3 months	771-940	Heaters	Not Specified	Sterilization
b) JPL, EPD-459	Not Specified	735	Waste Heat from RTG	+40 to 110° F	Battery operating temperature range, Sterilization
c) TRW 5305-6014-TU-000	30 days	300	Heaters	-20° to 100° F	Sterilization
d) AVCO RAD-TR-63-34	180 days	961	Passive	-10° to 130° F	Battery must be at 50° F minimum, day heat sterilization
e) JPL TM 33-236	1.5 days (Survival sphere) 6 mo. (Parachute landing)	138-150 (does not include impact limiter)	Radioisotope	-140° to +150° F	Mars surface conditions Surface winds (330 fps)
f) AVCO RAD-TR-64-36	5 hours	258-272	Insulation, Surface coating heaters	40° to 100° F	Impact shock (6000 g, 20 msec) Heat sterilization Entry Heating Surface winds (200 fps)
g) Aeronutronics U-3237	2 years	1186	Single phase circulating coolant plus heat from RTG	4° C ± 3° C	Temperature of reagents, Sterilization Martian surface conditions

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Table 8-2. Current lander capability

Venus

Reference	Survival Time on Surface	Landed Weight (LB)	Type of Thermal Control System	Temperature Limits	Constraints on Operation
a) JPL as 760-10	18 min.	212	insulation	180° F (internal)	External temperature (890° F), Entry heating, Heat sterilization.
b) AVCO RAD-TR-63-34	10-20 hr.	413	ammonia boil-off	100° F (internal)	Hard lander, Heat sterilization, Entry heating (10,000 btu/sec.ft^2), Venus surface temperature
c) JPL 760-1	Separation to impact only (12 days)	119	insulation	0-160° F (internal) 300° F (antenna)	Sterilization, Entry heating
d) BELLCOMM (7-3-67)	1 hour	150	Melting ice	not specified	Sterilization, Entry heating, Venus atmosphere and surface temperatures
e) Martin-Marietta, CR-66404	1 hour (descent to surface)	5-28	insulation	not specified	Sterilization, Venus atmosphere and surface
f) Goddard, TN D-1909	N.S.	N.S.	N.S.	N.S.	N.S.
g) JPL IOM VMOS 67-68	None (15-60 min. descent time)	180	insulation	-	Sterilization, Venus atmosphere

APPENDIX A

STUDIES CONSIDERED, BUT NOT REVIEWED

APPENDIX A

STUDIES CONSIDERED, BUT NOT REVIEWED

1) Aeronutronic

   a) "The EMPIRE Dual Planet Flyby Mission" (N64-10907)

   b) "A Study of Early Manned Interplanetary Missions (EMPIRE)" (U-1951, 1963)

2) AVCO

   a) "Manned Mars Surface Operations-Final Report" (RAD-TR-65-26, 1965)

   b) "Mars-Venus Capsule Parameter Study" (RAD-TR-64-1, 1964)

   c) "Comparative Studies of Conceptual Design and Qualification Procedures for a Mars Probe/Lander" (AVSSD-0006-66-R, 1966)

3) BELLCOMM

   a) "Experimental Payloads for a Manned Mars Flyby Mission" (TR-67-233-1, 1967)

4) Cutler-Hammer

   a) "Study of Topside Sounder for Mars & Venus Ionospheres from Mariner Spacecraft" (1844-1, 1963)

5) Douglas Aircraft

   a) "Handbook of the Physical Properties of the Planet Jupiter" (SP-3031, 1967)

   b) "Handbook of the Physical Properties of the Planet Venus" (SP-3029, 1967)

   c) "Manned Mars Exploration in the Unfavorable (1975-85) Time Period" (Jan. 1964)

6) Electro-Optical Systems

   "Feasibility Study of an Ion-propelled Mars Orbiter/Lander Spacecraft with Solar Photovoltaic Power" (AD-804109, 1966)

7) Fairchild-Hiller

"ATS-4 Study Program Final Report" (SSD 102.3, 1967)

8) General Dynamics/Ft. Worth

   a) "A Study of Mission Requirements for Manned Mars and Venus Exploration" (FZM-4366)

   b) "A Study of Manned Interplanetary Missions" (AOK 64006, 1964)

9) General Electric

   a) "Venus-Mars Capsule Study" (NASA CR-50811, 1963)

   b) "Study of Low Acceleration Transportation Systems - Phase I Study Effort" (65SD4315, 1965)

   c) "Navigator Study of Electric Propulsion for Unmanned Scientific Missions" (65SD 4296, 1965)

   d) "Electrically Propelled Cargo Vehicle for Sustained Lunar Supply Operations" (66SD 2019, 1966)

   e) "Study of Low Acceleration Space Transportation Systems - Phase 2 Study Report" (66SD 2026, 1966)

   f) "Voyager Design Study" (63SD801, Oct. 1963)

10) Hughes Aircraft Company

"Solar-Powered Electric Propulsion-Program Summary Report" (SSD 60374R)

11) JPL

   a) "Venus: Preliminary Science Objective & Experiments for Use in Advanced Mission Studies" (EPD-328, 1965)

   b) "Voyager Standardized Soft Lander Study Report" (VPE-14, EPD-459)

   c) "Venus/Mercury Swingby with Venus Capsule: Preliminary Science Objectives & Experiments for Use in Advanced Mission Studies" (TM 33-332, 1967)

12) Lockheed-California

"Advanced Mission Analysis Study" (LR 17358, 1963)

13) Lockheed-Missiles & Space

"Advance Study of an Applications Technology Satellite (ATS-4) Mission" (NASA CR-81765, 1966)

14) NASA/Goddard

"Experiments from a Small Probe which Enters the Atmosphere of Mars" (TN D-1899, 1963)

15) NASA/Lewis

a) "Solar-electric Probes for Exploring the Solar System" (TM-x-52318, 1967)

b) "Manned Venus Orbiting Mission" (TM-x-52311)

c) "A Parametric Study of Constant Thrust, Electrically Propelled Mars & Venus Orbiting Probes" (TN D-2154, 1964)

16) NASA/Marshall

a) "Manned Planetary Reconnaissance Mission Study: Venus/Mars Flyby Technical Summary Report" (TM X-53204, 1965)

b) "Multiple Planet Flyby Missions to Venus and Mars in the 1975 to 1980 Time Period" (NASA TMX-54511)

17) North American Aviation, S&ID

"Mariner B/Voyager Entry Capsule Preliminary Design Study" (SID 63-1293, 1963)

18) RAND Corp.

a) "The Physics of Balloons & Their Feasibility as Exploration Vehicles on Mars" (R-421-JPL, 1963)

b) "Mars Environmental Measurements in Support of Future Manned Landing Expeditions" (RM-4437-NASA, 1965)

19) United Aircraft

"Study of Low-Acceleration Space Transportation Systems" (D-91026 2-3, 1965 and E-91026 2-6, 1966)

20) USAF Cambridge Research Labs

"Bibliography of Lunar & Planetary Research 1960-64" (AF CRL-SR-40, 1966)

APPENDIX B

REVIEWS COMPLETED

APPENDIX B

REVIEWS COMPLETED

1) Aeronutronics, "Final Report: Study of the Automated Biological Laboratory - Project Definition" (U-3237, 1965)
2) AVCO, "Voyager Design Studies" (RAD-TR-63-34, Oct. 1963)
3) AVCO, "Conceptual Design Studies of an Advanced Mariner Spacecraft" (RAD-TR-64-36, 1964)
4) BELLCOMM, "Venus Lander Probe for Manned Planetary Missions" (Case 223, 1967)
5) Boeing, "Study of Applicability of Lunar Orbiter Subsystems to Planetary Orbiters" (D-100710-2, 1967)
6) Brown Engineering, "Retrievable Mars Probe Definition" (NG7-26635, 1967)
7) General Dynamics/Ft. Worth, "A Study of Jupiter Flyby Missions" (FZM-4625, 1966)
8) General Electric, "An Advanced Study of an ATS-4 Mission" (1966)
9) General Electric Co., "Solar Probe Study" (63 SD 779, Sept. 1963)
10) IITRI/ASC
    a) M-11, "A Survey of Missions to Saturn, Uranus, Neptune and Pluto"
    b) M-14, "Digest Report: Missions to the Outer Planets"

Other IITRI Studies

    c) "The Scientific Objectives of Deep Space Investigations - Jupiter" (P-1, 1964)
    d) "... - the Satellites of Jupiter" (P-2)
    e) "... - Comets" (P-3)
    f) "... - the Asteroids" (P-4)
    g) "... - Interplanetary Space Beyond 1 AU" (P-5)
    h) "... - Saturn, Uranus, Neptune, & Pluto" (P-11, 1966)

i) "... - Venus" (P-7, 1966)

j) "... - the Origin & Evolution of the Solar Systems" (P-18, 1966)

k) "A Survey of Missions to the Asteroids" (M-3, 1963)

l) "Survey of a Jovian Mission" (M-1)

m) "Scientific Objectives for Mercury Missions" (1964)

n) "Summary of Flight Missions to Jupiter" (M-4, 1964)

o) "A Survey of Comet Missions" (M-7)

p) "A Study of Interplanetary Space Missions" (M-6, 1965)

q) "Critical Measurements on Early Missions to Jupiter (P-10, 1965)

r) "Missions to the Comets" (M-9, 1965)

s) "Asteroid Fly-through Mission" (S-2, 1966)

t) "A Study of Multiple Missions Using Gravity - Assisted Trajectories" (M-12, 1966)

u) "Mission Requirements for Exobiological Measurements on Venus" (P-16, 1966)

v) "Low Thrust Trajectory Capabilities for Exploration of the Solar System Using Nuclear Engine Propulsion" (T-17, 1966)

w) "Preliminary Payload Analysis of Automated Mars Sample Return Mission" (M-13)

11) JPL, "Mars Entry and Landing Capsule" (LTM-33-236, 1965)

12) JPL, "Mariner Mars 1969 Orbiter Technical Feasibility Study" (EPD 250, 1965)

13) JPL, "Study of Mars and Venus Orbiter Missions Launched by the 3-stage Saturn C-1B Vehicle" (EPD 139, 1963)

14) JPL, "Study of a 1973 Venus Capsule/Lander Mission" (AS 760-10, 1967)

15) JPL, "Study of a 1973 Venus-Mercury Mission with a Venus Entry Probe" (Document 760-1, 1967)

16) JPL, "Advanced Planetary Probe - Jupiter Flyby Application" (EPD-358, 1966)

17) JPL, "Voyager Standardized Soft Lander Study Report" (EPD-459, 1967)

18) Lockheed, "Asteroid Belt and Jupiter Flyby Mission Study" (JPL Reorder No. 65-153, 1965)

19) Martin Marietta Corp., "Final Report, Buoyant Venus Station Feasibility Study" (NASA CR-66407, 1967)

20) Martin Marietta Corp., "Solar Probe Study" (ER 13110, 1963)

21) MIT Center for Space Research, "Study of a Small Solar Probe (Sunblazer)", (PR 5255-5, 1965)

22) Minneapolis-Honeywell, "Solar Probe Report" (CPE-B35-2, 1963)

23) NASA/Ames, "The NASA Advanced Pioneer Mission" (TMX-54039, 1964)

24) NASA/Goddard, "Exploration of the Atmosphere of Venus by a Simple Capsule" (NASA TN D-1909, 1964)

25) NASA/Goddard, "Phase A Report, Galactic Jupiter Probe" (Report X-701-67-566, 1967)

26) Philco/WDL, "Solar Probe Study" (WDL-TR 2133)

27) Philco/WDL, "Comet and Close Approach Asteroid Mission Study" (WDL-TR2366, 1965)

28) Stanford Univ., "ICARVS: Interplanetary Craft for Advanced Research in the Vicinity of the Sun" (NASA CR 80840, 1966)

29) TRW Systems, "Advanced Planetary Probe Study" (#4547-6004-R0000, 1966)

30) TRW Systems, "Extended Pioneer Study" (1964)

31) TRW/STL, "Study of Unmanned Systems to Evaluate the Martian Environment (Report 5303-6014-TU 000, 1965)

APPENDIX C

ETR MISSION STUDY REVIEW FORMAT

APPENDIX C

ETR MISSION STUDY REVIEW FORMAT

I. IDENTIFICATION

1. Corporate Source
2. Title
3. Report Number
4. Date of Study
5. Mission Classification
    a. Destination
    b. Type of Mission (Flyby, Orbiter, Lander, Probe, etc.)
    c. Manned or Unmanned

II. GENERAL

1. What are mission primary objectives?
2. What are mission secondary objectives?
3. What were the constraints and assumptions used in this study?
4. What were types of launch vehicles to be used or considered?
5. Which launch opportunities beyond 1973 are listed? What is the closest planetary approach for each?
6. How long is mission duration?
    a. Spacecraft bus or orbiter
    b. Entry Probe
7. Describe briefly mission sequence (list only major events, such as midcourse maneuvers, planetary encounter, etc.). How long after launch does each occur?
8. List the science (spacecraft bus and probe) instruments, measurements taken by each, and when it is operating.
9. List any alternate science payloads considered.
10. Describe any special devices and packaging techniques proposed.

11. Attach drawing showing recommended spacecraft and probe configuration.

12. Describe non-thermal radiation environments considered.

III. MISSION-UNIQUE THERMAL ENVIRONMENTAL FACTORS

1. What is the expected solar radiation level at the closest or furthest point from the sun?

2. List the materials degradation factors considered (such as UV, micro-meteoroids, sand and dust storms, solar flares) and the expected intensity or frequency of each.

3. Does the vehicle geometry make the spacecraft collimation sensitive?

4. What was considered for the following planetary atmospheric environments?

   a. Pressure
   b. Winds
   c. Convection Heat Transfer

5. What are the maximum aerodynamic heating rates for launch and re-entry?

6. How is the albedo of the planet considered?

7. Is thermal radiation from the planet a significant environmental factor?

8. What is the frequency and severity of environment or mission-caused thermal shocks?

9. Describe any other mission-unique thermal environmental factors considered in the study.

IV. WEIGHT AND POWER SUMMARY

	Spacecraft Weight	Spacecraft Power	Probe Weight	Probe Power
1. Structure				
a. Basic Bus				
b. Appendage or Experiment Supports				
c. Other (Landing Devices, Etc.)				
2. Thermal Control				
3. Power				
a. Solar Panel(s)				
b. Battery				
c. Conversion Equipment				
d. Other				
4. Guidance & Control				
a. Altitude Stabilization System				
b. Optical Sensors				
c. Inertial Sensors				
d. Computer				
e. Sequencer				
5. Telecommunications				
a. Antenna(s)				
b. Radio				
c. Telemetry				
d. Command				
e. Bulk Storage				
6. Propulsion				
a. Solid				
b. Liquid				
7. Science				
(List Individual Instruments)				
TOTAL				

V. THERMAL CONTROL

1. Are active temperature control devices used? What kind? Were they specifically developed for this mission?

2. Was temperature a major constraint in the choice of any of the spacecraft equipment? Which equipment?

3. List the expected temperatures, temperature limits, and most critical mission phase described for each major piece of spacecraft equipment in this study.

VI. TELECOMMUNICATIONS SUBSYSTEM (SPACECRAFT AND PROBE)

1. What is the maximum communication range?

2. What is communication channel bit rate?

3. How much transmitted power is available?

4. What is the maximum doppler rate?

5. What is the Earth vector (antenna cone and clock angles)?

6. List antennas used. Include number, size, type, and whether fixed or pointable.

7. What telemetry and command modulation methods are used?

8. List the number and type of commands (DC, QC, other).

9. What is the number of telemetry measurements?

10. How much data storage is required?

11. What is the science and TV bit rates?

12. For an entry probe or orbiter mission, is the communications system used direct or relay?

13. For an entry probe, how much atmospheric RF attenuation is expected?

14. For an entry probe or lander, how long will communications blackout last? How far above the surface or how long before landing will this occur?

VII. POWER SUBSYSTEM

1. What criteria led to choice of power source?

2. How many charge/discharge cycles are assumed if battery power is used?

3. What are maximum, minimum and average power demands for each power source.

VIII. GUIDANCE AND CONTROL SUBSYSTEM

1. What are the celestial references?

2. Is need for approach guidance anticipated (is it a self-contained on-board system).

3. Is on-board computing capability included?

APPENDIX D

EXAMPLE STUDY SUMMARY OF BUOYANT VENUS STATION FEASIBILITY STUDY-FINAL REPORT

I. IDENTIFICATION

Martin-Marietta Corp., "Buoyant Venus Station Feasibility Study - Final Report," NASA CR-66404, 1967.

Mission Description: Unmanned buoyant station (BVS) which remains in Venusian atmosphere for relatively long periods of time, releasing drop probes to surface and relaying data to orbiting spacecraft bus. Two cases were considered: 200-lb and 2000-lb stations.

II. BENERAL

1. Primary Objectives

Conduct experiments aimed at removing present uncertainties in knowledge of the surface and subsurface, clouds, atmosphere, and life on Venus (Vol. 3, p. 3).

2. Secondary Objectives

Not stated.

3. Constraints and Assumptions

a. Orbiter spacecraft will serve as relay to transmit data to Earth and assumed to possess all required receiving, storage and transmitting capabilities.

b. Nominal orbit assumed to have periapsis altitude of 1000 km and apoapsis altitude of 10,000 km.

c. NASA SP-3016 Venus model atmospheres (3) assumed.

d. Initial conditions at inflation assumed to be consistent with subsonic velocity above visible cloud layer for all 3 atmospheres.

e. Only buoyant station mission modes considered.

f. Mission modes investigated to be consistent with sampling times and coverage required for experiments; communication time to transmit data; and limitations of instruments, materials and systems.

g. No consideration of survivable landing of buoyant station on Venusian surface.

h. Buoyant station may serve as mobile platform for release of probes to investigate lower atmosphere and surface conditions.

i. Total station(s) weight at inflation initial conditions not to exceed 5000 lbs.

j. Station to include engineering instrumentation to monitor all significant events and operational status throughout mission.

k. At a minimum, station measurements to include following classes and quantities:

   1. Position: Altitude above terrain, absolute altitude, and horizontal position.

   2. Ambient Environment: Pressure, temperature, density, composition, particles, electromagnetic fields, gravitation, radiation, and winds.

   3. Below Station: Radiation from surface characteristics, cloud top height(s), cloud particles.

l. Station to orbiter communications system assumed not to require directional orientation of antennas or high gain antennas.

m. Mission is either the first to enter the planet's atmosphere or has been preceded only by a simple atmosphere entry probe or flyby vehicle.

n. Not concerned with entry conditions (Vol. IV, p. 59).

4. Launch Vehicles

   a. 200-lb station - Atlas/Centaur.

   b. 2000-lb & 5000-lb stations - Voyager class mission.

5. Launch Opportunities Beyond 1973

   a. 200-lb station - 1972 and 1973.

   b. 2000 and 5000-lb stations - launch opportunities not stated, but would coincide with Voyager opportunities.

   Closest approach to planet - orbiter: 1000 km
   Buoyant station - in planetary atmosphere
   Drop probe - land on surface

6. Mission Duration

   200-lb station - 7 days
   2000-lb station - 100 days

   Drop probe - 1 hour descent (Table 13, p. 31, Vol. V)

Special Constraints for 200-lb Station

a. 225-lb launch vehicle compatibility.
b. Sterilization considerations
c. NASA SP-3016 atmosphere
d. Float at cloud tops

e. Relay communications, no directional antennas
f. 1000 by 10,000 km orbit
g. Probe to surface with 2 sondes and final descent
h. Science measurements: temperature, pressure, density, composition, and horizontal position.
i. Onboard sequencing with command as backup
j. 7-day mission
k. Transmission during deployment
l. Adaptability to environment.

Special Constraints for Large Station

a. 5000-lb limit
b. Sterilization considerations
c. NASA SP-3016 atmospheres
d. 1000 by 10,000 km orbit
e. Relay communications, no directional antennas
f. Will not be first probe to enter Venus atmosphere, thus more will be known about atmosphere than assumed

7. Mission Sequence

a. 200-lb station (Table 5, Vol. 5)

1. Deployment (~ 55° S lat, 15° E long) Time = 0 (Decelerate in atmosphere, eject chute, release aeroshell, open cover, chute deploys ballon, and begin balloon inflation)

2. Measure pressure, temperature, density, composition at altitude

3. Release drop sonde - on command ~ 1 day after deployment

4. Measure pressure, temperature, density and composition at altitude

5. Release 2nd drop sonde- on command near pole ~ 6-7 days after deployment

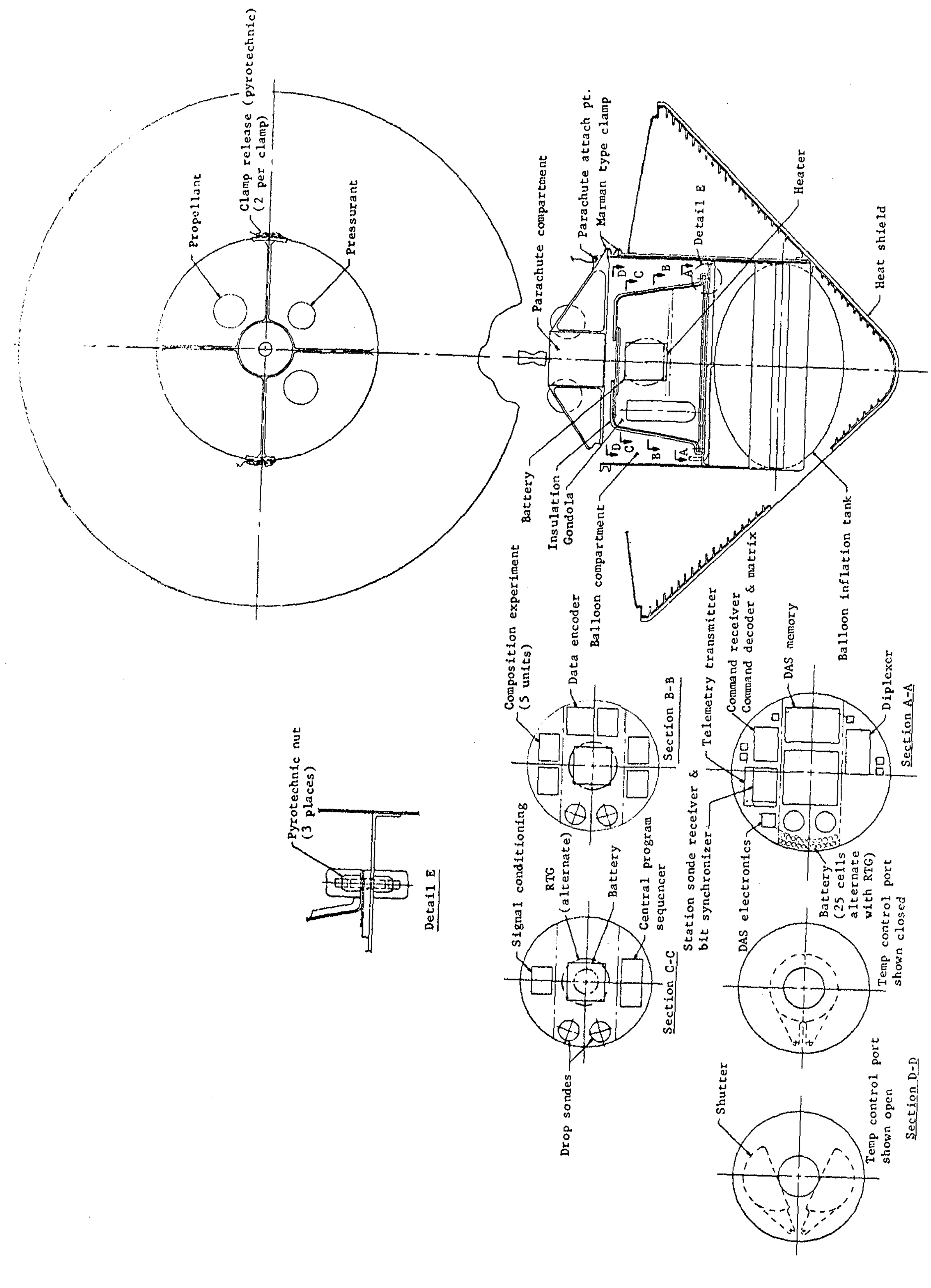
6. Make measurements at altitude

7. Measure pressure, etc., during station descent > 7 days after deployment

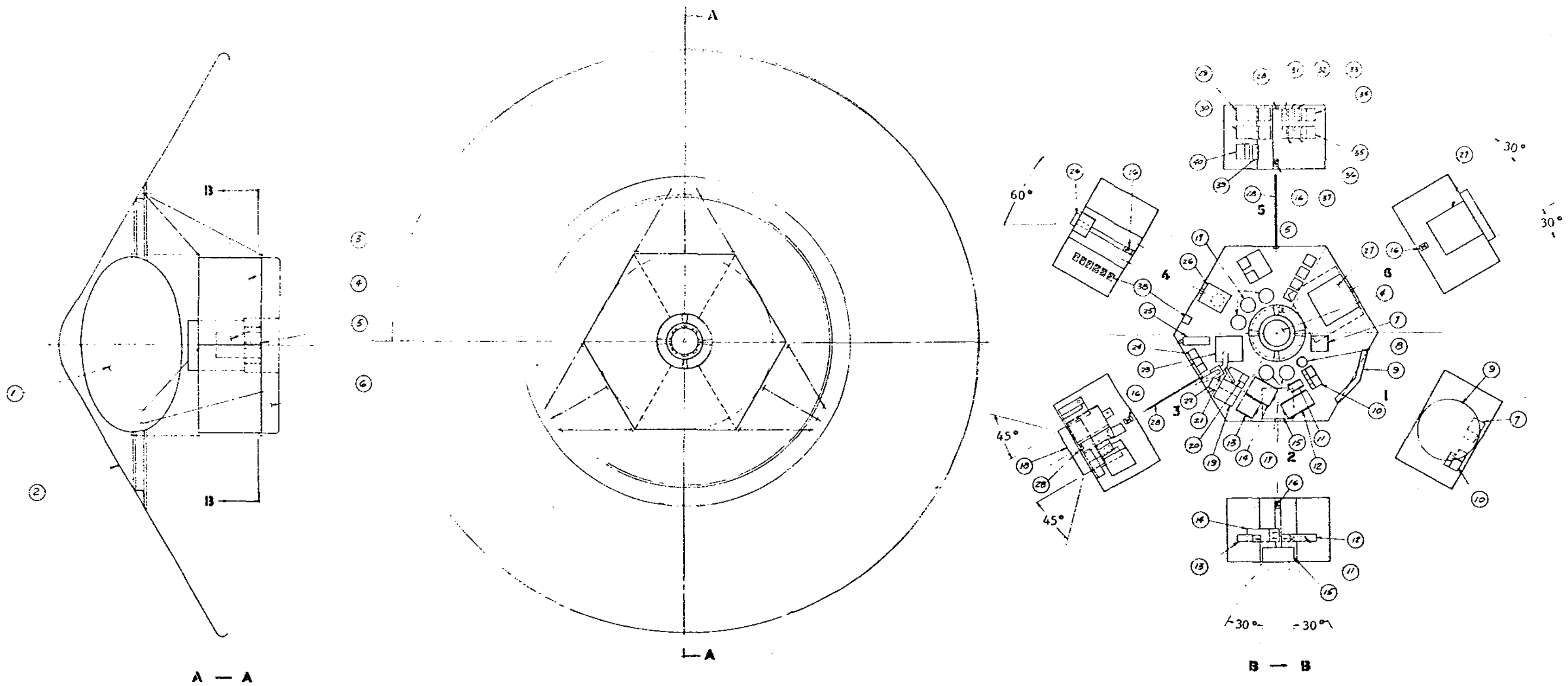
b. 2000-lb BVS

1. Noncyclic station (Fig. 11, Vol. 1)

Separation & deployment
(Aeroshell descent mission)
Inflation
Tankage Drop
Mission at equilibrium (see cyclic station, below)
Drop sonde missions (4)
Descent mission



Buoyant Venus station, general arrangement (200 lb)



Legend:

1	Inflation tank	10	Junction box, switches, and relays	20	Vidicon microscope	30	Group IIA memory
2	Voyager Mars aeroshell (19-ft diam)	11	Diplexer	21	Dust-sampler (for vidicon microscope)	31	Data package 1
3	Gondola	12	Transmitter	22	Sampler (atmosphere and clouds)	32	Data package 2
4	Stowed parachute	13	Command receiver	23	Micro-biology laboratory)	33	Data package 3
5	Phased array antenna	14	Command decoder	24	Mass spectrometer	34	Data selector
6	Stowed balloon	15	Radar altimeter	25	Acoustic densitometer	35	Programer/sequencer
7	Battery	16	Pressure sensor (total 10 req)	26	IR scanner/spectrometer	36	Group I memory
8	Balloon inflation hose	17	Drop sonde tubes (total 5 req)	27	Microwave spectrometer/scanner	37	Group II memory
9	RTG power supply	18	Drop sonde antenna	28	Temperature sensors-boom mounted (2 req)	38	Aerosol detectors
		19	Gas chromatograph	29	Group III/DS memory	39	Drop sonde receiver
						40	[illegible]

2000-lb buoyant Venus station

7. b. 2000-lb BVS (continued)

2. Altitude Cycling Station (Fig. 14, Vol. 1)

Separation and Deployment (Aeroshell Descent Mission)
Balloon inflation
Tankage and parachute release
Mission at equilibrium

a. Group I measurements 6 times every orbit. Mass spectrometer makes 1 analysis. Dust collectors started. Small drop sonde released.
b. Group II measurements made while data transmitted and analyzed from 1st orbit.
c. Group II and III measurements. Group III made every 3rd orbit unless drop sonde release command is given.

d. At beginning of 6th cycling, Group IIA and II B measurements commanded if enough dust is collected. Bio-lab measurements made once per orbit for 35 consequtive orbits. Measurements continue until Cycle command is given.

Altitude Cycling (3 minimum)
Descend to 10 km altitude
During descent, make Group IV measurements
During ascent, make only Group I measurements
Resume other measurements at original equilibrium altitude.

Release drop sonde when terminator is crossed.
Final descent on command.
Group IV measurements made til impact.

8. Science Payload

a. 200-lb Station (Vol. 5, Tables 2 & 3)

	Instrument	Measurement	When Operating
1)	4 platinum resistors temperature sensors	Temperature of atmosphere	After Deployment
2)	6 pressure sensors	Atmospheric pressure	After Deployment
3)	Single gas detectors for H_2O, N_2, O_2, A, CO_2	Atmospheric composition	After Deployment
4)	Acoustic transmission line densitometer	Atmospheric density, speed of sound	After Deployment
5)	Drop sondes (2)		
	a) Platinum resistance temperature sensor	Atmospheric temperature profile (lower altitudes)	During Descent from BVS

	Instrument	Measurement	When Operating
5)	(continued)		
	b) Pressure sensors (2)	Atmospheric pressure profile (lower altitude)	During Descent from BVS
	c) Water vapor detector	Atmospheric composition	During Descent from BVS

b. 2000-lb Station (Vol. VI, pp. 6-8)

	Experiment	Measurement	When Operating
1.	Temperature sensors (4)	Temperature	Group I, IV
2.	Pressure sensors (10)	Pressure	Group I, IV
3.	Acoustic transmission	Speed of sound, density	Group I, IV
4.	Mass spectrometer	Atmospheric composition	Group II, IV
5.	Pyrolysis/gas chromatograph/mass spectrometer	Cloud, dust composition	Group II
6.	Vidicon microscope	TV pictures of dust and biota	Group II (1 picture) Group IIA(17 pictures
7.	Minimum biolab	Life detection	Group IIB
8.	Ion chamber & geiger tube	Incoming ionizing radiation	Group I, IV
9.	UV radiation flux	Nature of clouds, presence of ozone	Group I, IV
10.	Visible/near IR flux		Group I, IV
11.	Altimeter/radar scatterometer	Altitude, scattering & electrical properties of surface	Group I, Group III (scatterometer), Group IV
12.	Microwave scanner/ spectrometer	Thermal map of surface	Group III
13.	IR Scanner/spectrometer	Clouds & atmosphere, thermal map of surface	Group III, IV
14.	Light backscatter from aerosols	Particulates in atmospheres	Group IV

	Experiment	Measurement	When Operating
15.	Large sondes (4)		
	a) Temperature sensors	Atmospheric temperature profile	During Descent
	b) Pressure sensors (4 static, 1 impact)	Pressure profile	During Descent
	c) Impactometer	Senses impact	At Impact Only
	d) Photometers & filters looking up	CO_2, H_2O absorption bands	
	e) Mass spectrometer	H_2O, N_2, O_2, A, CO_2 presence	
	f) Cloud sampler	Cloud composition	
16.	Small sonde		

Same as 200-lb BVS sonde

Unless otherwise specified, BVS instruments operated after deployment, and sonde instruments during descent.

9. Alternate Science Payloads (Not Used On Either 200 or 2000-lb BVS)

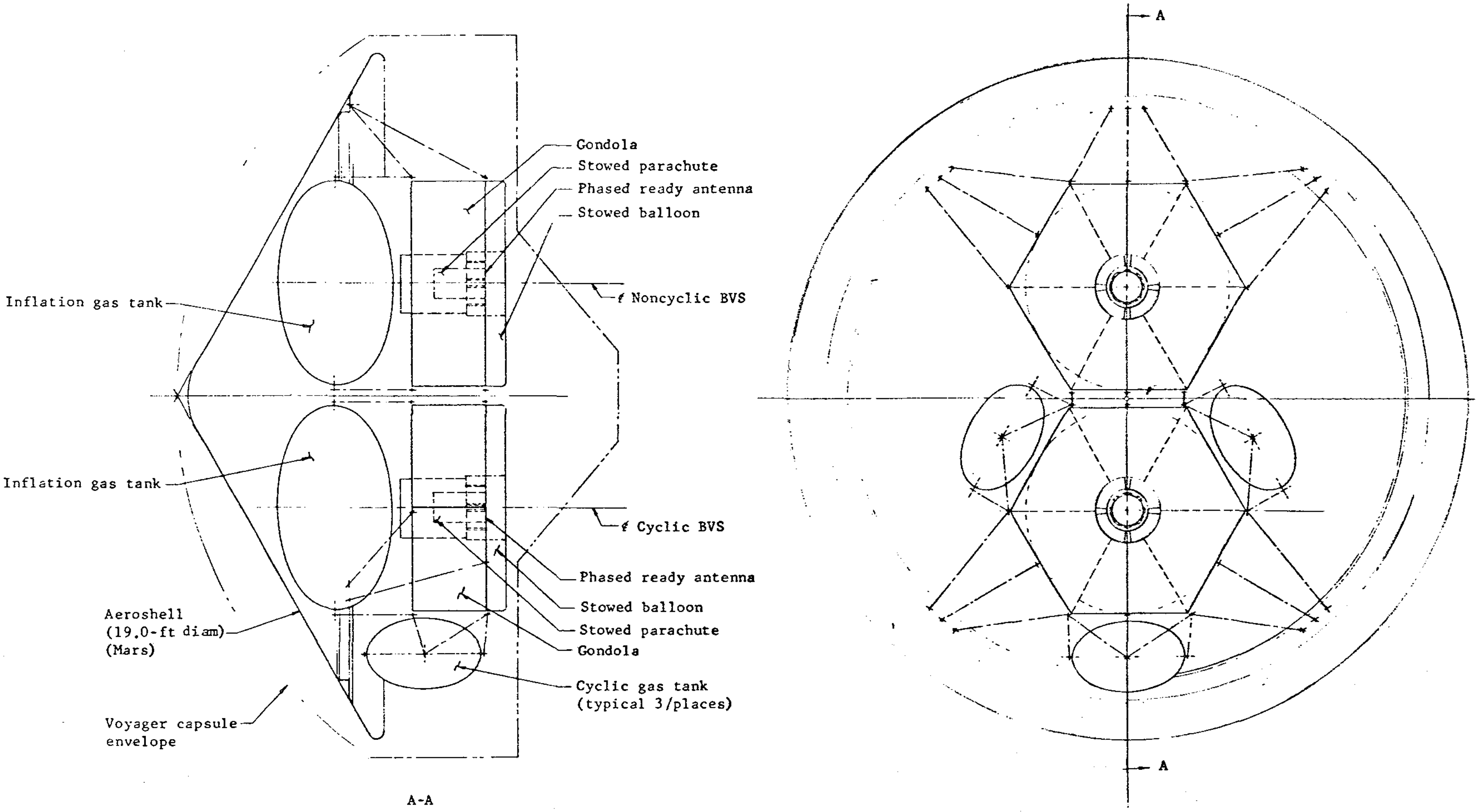
Measurement	Instrument
Priority 3:	
Gravity	Gravimeter
Winds - small scale	Drop sonde accelerometers on BVS
Precipitation	Aerosol detector, vidicon microscope
Priority 4:	
Seismic activity	Seismometer on drop sonde
Volcanic activity	Thermal mapping, seismometer
Surface raioactivity	Geiger tube on drop sonde

All Priority 1 and 2 measurements made on either 200 or 2000-lb BVS.

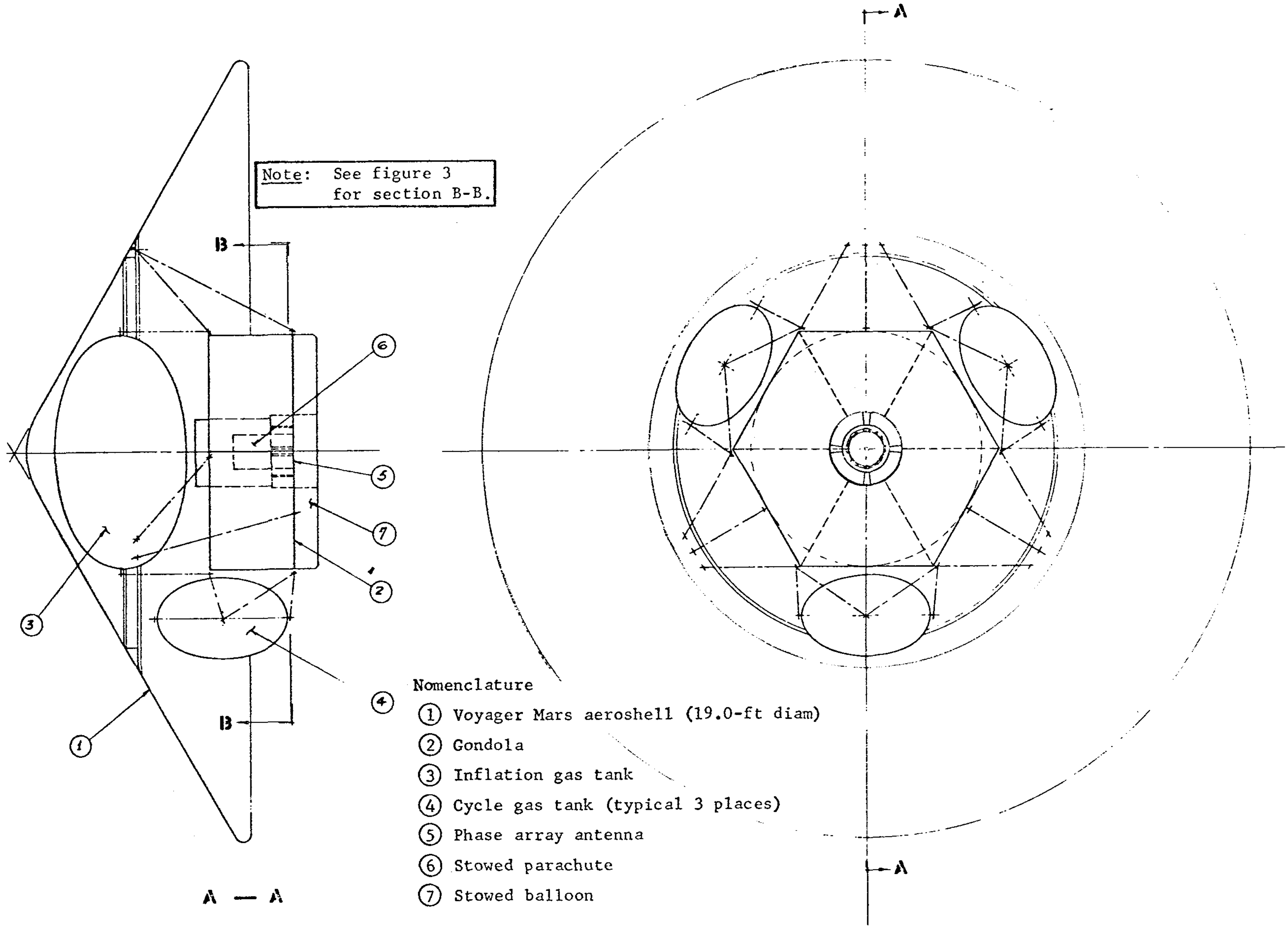
10. Special Devices & Packaging Techniques

a. Gondola could be made smaller to decrease insulation weight for 2000-lb and 5000-lb BVS, necessitating repackaging of assemblies. Moderate packing density of 20 lb/cu. ft. on 200-lb BVS (p. 4, Vol. 5).

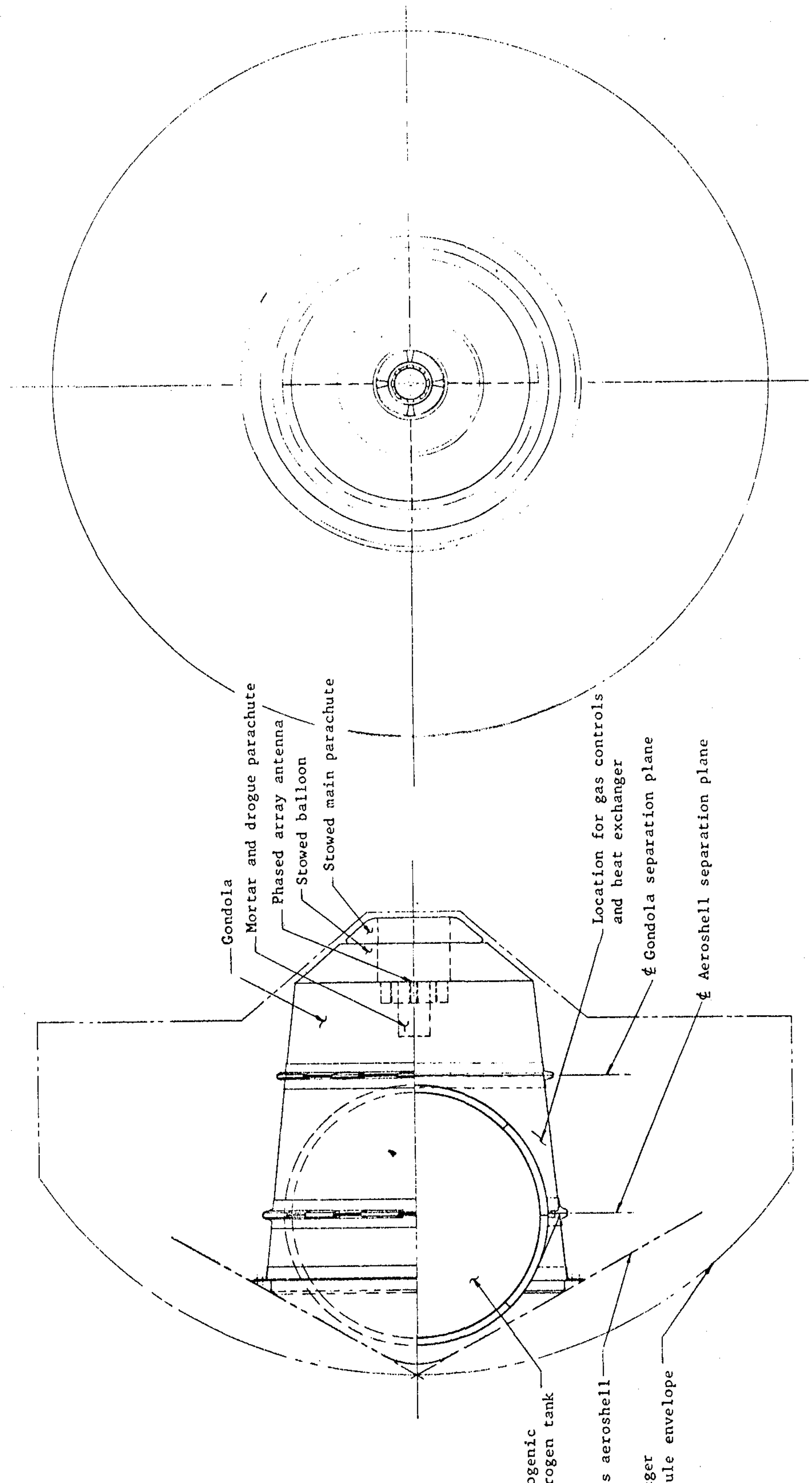
b. Hydrogen for balloon inflation is stored in nickel-lined tank because nickel is not attached by hydrogen (p. 10, Vol. 5).



Two 2000-lb stations in Voyager capsule



Cyclic 2000-lb buoyant Venus Station



5000-lb Noncyclic buoyant Venus station

11. Non-Thermal Radiation

Normal deep space (SP-3016) plus RTG radiation of 3 rad/hr maximum.

III.

1. Expected Solar Radiation Level at Closest Point to Sun:

0.718 AU - 0.728 AU (2710 to 2640 w/m^2) (p. 51, Vol. IV)

2. Materials Degradation Factors

Atmospheric temperature:

Above cloud tops - 195 - 287°K
At Bottom of cyclic mission - 675°K maximum

3. Vehicle geometry not collimation sensitive.

4. Planetary atmospheric environments as per NASA SP-3016.

5. Maximum Aerodynamic Heating Rates

a. Launch not considered in this study.
b. Re-entry - Same as Voyager; uses same aeroshell.

6. Albedo of planet - as in NASA SP-3016.

7. Thermal radiation from planet not a significant environmental factor.

8. Frequency and severity of thermal shocks not specified.

IV. WEIGHT AND POWER SUMMARY

200-Lb BVS	SPACECRAFT		DROP SONDE	
	Weight (lb)	Power (w)	Weight (lb)	Power (w)
1. Structure			0.5	
a. Parachute	12	---		
Balloon system	23.8	---		
Hydrogen system	94.8	---		
b. Gondola	10	---		
2. Thermal Control (Insulation)	2.4		0.85	
3. Power				
a. Solar Panels	---	---	---	---
b. Battery	1.5		0.25	

		SPACECRAFT		DROP SONDE	
		Weight (lb)	Power (w)	Weight (lb)	Power (w)
3.	Power (continued)				
	c. Conversion equipment	2.5			
	d. Battery charger	1.0			
	RTG	5.0			
	Wiring, etc.	2.5			
4.	Guidance & Control				
	a, b, c	----------------NA----------------			
	d, e. Central programmer/ sequencer	1.3	0.3	---	---
5.	Telecommunications				
	a. Antennas (2)	3.9	---	---	---
	b. Radio (command receiver & bit synchronizer)	4.9	0.47	---	---
	c. Telemetry (telemetry transmitter, data encoder & signal conditioning)	5.0	13.56	1.5	2.0
	d. Command (command decoder & matrix, diplexer	3.0	2.0	---	---
	e. Bulk storage (science DAS, electronics & memory)	2.4	.375	---	---
6.	Propulsion	----------------NA----------------			
7.	Science				
	a. Temperature sensors	1.0	0.8	0.25	0.2
	b. Pressure sensors	3.0	0.6	0.75	0.2
	c. Composition				
	H_2O	1.5	1.0	0.5	0.5
	N_2	1.0	1.0	---	---
	O_2	1.5	1.0	---	---
	A	1.5	1.0	---	---
	CO_2	1.0	1.0	---	---

	SPACECRAFT Weight (lb)	SPACECRAFT Power (w)	DROP SONDE Weight (lb)	DROP SONDE Power (w)
7. Science (continued)				
d. Acoustic transmission line	3.0	4.0		
TOTAL	200.0	27.3	5.0	2.9

Non-Cyclic 2000-Lb BVS

	SPACECRAFT Weight (lb)	SPACECRAFT Power (w)	DROP SONDE Weight (lb)	DROP SONDE Power (w)
1. Structure				
Parachute system	106	---		
Balloon system	274	---		
Hydrogen system	850	---		
2. Thermal Control (Includes Gondola Structure)	307			
3. Power				
a) Solar Panels	NA			
b) Battery	17.0			
c) Conversion equipment	2.0			
d) RTG	40			
Battery Charger	3.7			
Wiring, etc.	14.0			
4. Guidance & Control				
a) Altitude stabilization	NA			
b) Optical sensors	NA			
c) Inertial sensors	NA			
d) Programmer/sequencer	1.0	1.0		
5. Telecommunications				
a) Antennas (2)	14	---		
b) Radio (command receiver, sonde data receiver & bit synchronizer)	6.2	2.17		

		SPACECRAFT Weight (lb)	SPACECRAFT Power (w)	DROP SONDE Weight (lb)	DROP SONDE Power (w)
5.	Telecommunications (continued)				
	c) Telemetry (transmitter, data packages 1-3, data selector)	22.8	105		
	d) Command (command decoder, diplexer)	8	2		
	e) Bulk storage (4 memory cores)	34	1.8		
6.	Propulsion	----------NA----------			
7.	Science				
	a) Temperature sensors	2	.8	2.5	0.5
	b) Pressure sensors	5	1.0	1.0	0.4
	c) Acoustic transmission	3	4.0	3.0	4.0
	d) Mass spectrometer	10	10.0	10.0	10.0
	e) Pyrolysis/gas chromatograph/mass spectrometer/dust cloud particle collector	2	10	---	---
	f) Vidicon microscope	15	8	---	---
	g) Minimum biolab	20	10	---	---
	Dust collector for (f) and (g)	2	.5	---	---
	h) Ion chamber & geiger tube	3	.5	---	---
	i) UV radiation flux	2	1.5	---	---
	j) Visible/near IR flux	3.0	2.3	---	---
	k) Altimeter/Radar scatterometer	15.0	30.0	---	---
	l) Microwave scanner/spectometer	25		---	---
	m) IR scanner/spectrometer	10	4	---	---
	n) Light backscatter from aerosols	5	5	2.5	2.5
	o) Photometers/filters (looking up)	---	---	3	2.3
	p) IR radiometer	---	---	5	3.0
	q) Electrometer	---	---	1	1
	TOTAL	1837	215	28	23.7

NOTES:

1) 2000-lb BVS includes 4 25-lb drop sondes (with payloads listed above) and 1 5-lb sonde (same payload as probe on 200-lb BVS).

2) Cyclic 2000-lb BVS is same as non-cyclic BVS listed above, except as follows:

NOTES (continued)

Cycle system	236 lb
Gondola structure and Thermal control	84 lb

V. THERMAL CONTROL SUBSYSTEM

1. Active thermal control devices used in both 200-lb and 2000-lb BVS.

   a. 200-lb BVS

      1. 5-w RTG or
      2. Combustion of lithium or beryllium in Venusian atmosphere (burners must be developed for this mission)
      3. Temperature control port

   b. 2000-lb BVS

      1. 40-w RTG
      2. Phase change material (Eicosane) used on cyclic station for cooling at lowest altitude.

      Thermal Control Assumptions (Vol. VI, p. 32)

      1. Adiabatic surface between compartments
      2. Compartment external surface temperature equal to ambient temperature
      3. Ambient equilibrium temperature of 225°K
      4. Cycle ambient temperatures increasing linearly to 620°K and decreasing linearly to 225°K in 4.25 hrs
      5. Minimum allowable compartment temperature of 288°K (mainly for batteries)
      6. Phase change material is "Eicosane"; melting point - 310°K, density - 53.3 lb/ft^3, latent heat - 106 BTU/lb.
      7. Insulation is micro-quartz with conductivity of 0.02 BTU/hr-ft°R and density of 6 lb/ft^3.
      8. Compartment equipment heat of 25 w.
      9. Thermal energy of RTG not used in this analysis.
      10. Mean model atmosphere assumed.

2. Temperature Constraints (Major) on Choice of Spacecraft Equipment

   a. PBI film chosen for balloon material on cyclic BVS to withstand temperatures at low point of cycle.

   b. Battery must be maintained at minimum temperature of 278°K (200-lb BVS); thus, entire gondola requires active thermal control.

   c. Heat sterilization required.

3. Temperatures, Limits, Critical Mission Phases

a. 200-lb BVS

	Expected Temperatures	Temperature Range	Critical Mission Phase
Balloon (mylar)	195-287°K	213-423°K	After Sterilization
Hydrogen System & Inflation Hardware	195-300°K	----	After Sterilization
Science Telecommunications Power	(Inside Gondola) 278°K	----	After Deployment
Drop Sonde	Venus Atmosphere & Surface		Descent

b. 2000-lb BVS

Balloon			
Non-cyclic (mylar)	195-287°K	213-423°K	After Deployment
Cyclic (PBI)	195-675°K	4-723°K	At 10 km Altitude
Hydrogen System	Same as balloon		Sterilization
Science	Thermally controlled		
Telecommunications	Compartments @ 98°F maximum for cyclic mission		
Power	Battery requires minimum temperature of 40°F		

VI. TELECOMMUNICATIONS SUBSYSTEM

1. Maximum communication range

14000 km (orbiter/station link) Vol. 3, p. 6
100 km (Drop sonde/station link)

2. Communication channel bit rates

200-lb BVS (Vol. V)

p. 16	Station/orbiter	30 BPS (Data transmission)
p. 16	Command link	30 BPS
p. 17	Ranging orbiter to station	18,750 BPS
p. 19	Drop sonde	1 BPS

2000-lb BVS (Vol. VI)

p. 23	Station/orbiter	1000 BPS
p. 24	Command link	50 BPS
p. 25	Ranging orbiter to station	18,750 BPS
p. 26	Drop sonde	25 BPS

3. Transmitted power

	200-lb BVS (Vol. V)	2000-lb BVS (Vol. VI)
Station/orbiter	p. 16- 5w (200 MHz)	p. 23 40w (400 MHz)
Command (Orbiter)	p. 16 10w (230 MHz)	p. 24 20w* (370 MHz)
Drop Sonde	p. 19 10 mw (300 MHz)	p. 26 12 mw (300 MHz)

* For ranging, 40w is required

4. Maximum Doppler Rate

43.8 Hz/sec ? (Vol. IV, p. 20)

5. Earth Vector - Not considered since study did not deal with orbiter.

6. Antennas

200-lb BVS (Vol. V)

p. 18	Station/orbiter link	Crossed dipole mounted above ground plane. Centered at wavelength of 215 MHz over range of ±15 MHz. Fixed.
p. 19	Station antenna for station-sonde link	Not critical, can be conical crossed dipole or deployable helix (compressed like spring and released). Circularly polarized.
p. 20	Sonde antenna	4 monopoles fed to provide circular polarization.

2000-lb BVS (Vol. VI)

p. 29	Station/orbiter	Phased array of 4 cavity-backed slots.
	Station antenna for drop sonde link	Cavity-backed cross slot.
	Sonde antenna	Type not stated but should be same type as for drop sonde on 200-lb BVS (see Vol. III, p. 55)

7. Modulation Methods

200-lb BVS (Vol. V)

p. 16	Station/orbiter	Phase modulated by sum of two coherent subcarriers.
p. 16	Command link	Same as above.
p. 19	Drop sonde	Frequency shift key (FSK)

7. (continued)

2000-lb BVS

Same as 200-lb BVS

8. Number & Type of Commands

200-lb BVS:	9 commands, all direct
2000-lb BVS:	31 direct commands (noncyclic) 36 direct commands (cyclic),

9. Number of Telemetry Measurements

200-lb BVS (Vol. V, Fig. 21, p. 50)

Deployment/Engineering Data	16 measurements
Engineering Data	16
Science Data	16
Drop Sonde Data	16

2000-lb BVS

Engineering Data	64 (cyclic) 56 (noncyclic)
Science	
Group I	11 measurements
Group II, IIA, IIB	4
Group III	3
Group IV	3
Drop Sondes	
Per Sonde	13

10. Data Storage Requirements

200-lb BVS	Core memory of uncertain capacity (10,000 bits?)
2000-lb BVS	4 core memory systems of these capacities: Core A - 10,000 bits Core B - 10,000 bits Core C - 90,000 bits Core D - 50,000 bits

11. Science and TV Bit Rates

200-lb BVS	
Station Science	30 BPS
Drop Sonde	1 BPS
2000-lb BVS	
Station	1000 BPS
Drop Sonde	25 BPS

12. Communications system - relay

13. Expected atmospheric RF attenuation at 3000 MHZ, 1×10^{-3} db/km (positive slope with frequency). Vol. IV, p. 7

14. No communications blackout considered.

VII. POWER SUBSYSTEM

1. Criteria for Choice of Power Source

200-lb BVS

a. Ability to survive sterilization.
b. Battery chosen to provide peak demands.
c. RTG selected for continuous loads and to help in thermal control.
d. Combination RTG-battery system gives minimum weight.

2000-lb BVS

a. Sterilization requirements.
b. RTG has weight advantage for long mission duration (greater than 5 days)
c. Battery used for high peak demands

2. Number of Charge/Discharge Cycles Assumed:

Not explicitly stated, but appears to be continuous.

3. Power Demands

200-lb BVS (Power profile shown in Fig. 22, Vol. V)

RTG:	Maintain Battery	1.6 w	(peak loads)
	Converter Regulator	2.3 w	(continuous)
	Total	3.9 w	(5 w RTG used)

2000-lb BVS (Power profile shown in Fig. 31, Vol. VI)

RTG:	Maintain Battery	27.2 w	(peak)
	Converter Regulator	9.4 w	(continuous)
	Total	36.5 w	(40 w RTG used)

VIII. GUIDANCE & CONTROL SUBSYSTEM

Questions not applicable, since study did not concern orbiter.