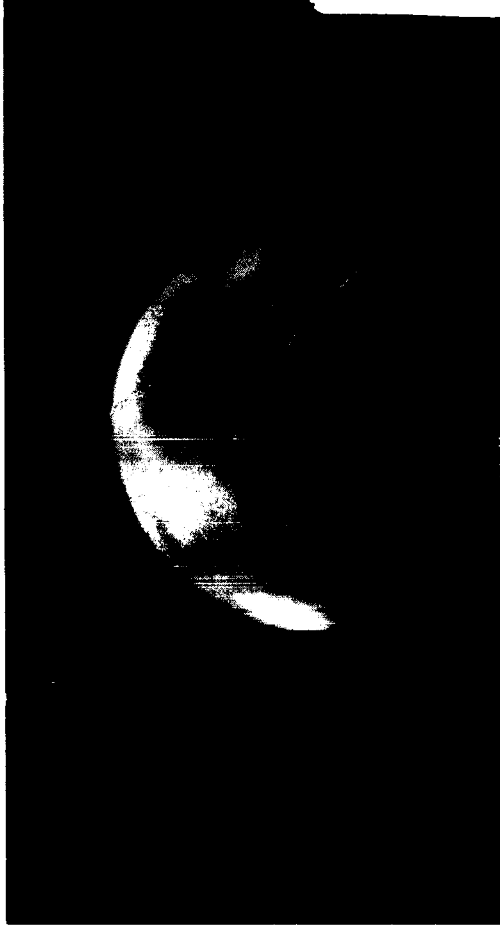




# VOYAGER DESIGN STUDIES

## Volume One: Summary

Prepared Under Contract Number NASw 697 ■ Research and Advanced Development Division ■ Avco Corporation ■ Wilmington, Massachusetts ■ National Aeronautics and Space Administration ■ Avco/RAD ■ TR-63-34 ■ 15 October 1963

## FOREWORD

The Voyager Design Study final report is divided into six volumes, for convenience in handling. A brief description of the contents of each volume is listed below.

Volume I -- Summary

A completely self-contained synopsis of the entire study.

Volume II -- Scientific Mission Analysis

Mission analysis, evolution of the Voyager program, and science payload.

Volume III -- Systems Analysis

Mission and system tradeoff studies; trajectory analysis; orbit and landing site selection; reliability; sterilization

Volume IV -- Orbiter-Bus System Design

Engineering and design details of the orbiter-bus

Volume V -- Lander System Design

Engineering and design details of the lander.

Volume VI -- Development Plan

Proposed development plan, schedules, costs, problem areas.

This document consists of 126 pages,
116 copies, Series A

# VOYAGER DESIGN STUDIES

## Volume I: Summary

Avco RAD-TR-63-34

15 October 1963

Prepared under Contract No. NASw 697 by

RESEARCH AND ADVANCED DEVELOPMENT DIVISION
AVCO CORPORATION
Wilmington, Massachusetts
for
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

TABLE OF CONTENTS

LIST OF TABLES

## LIST OF FIGURES

LIST OF FIGURES (Concl'd)

## ACKNOWLEDGMENT

This report presents the results of a 6-month study effort carried out by a project team of engineers and scientists at the Avco Research and Advanced Development Division. Acknowledgement is made to the many individual contributors. Their large number makes it impractical to list them by name. Subcontractor assistance was provided by the following orgainizations:

Avco Electronics Division -- Communications and Radar Mapping

Avco-Everett Research Laboratory -- Simulation of Radiation Heat Transfer

Bell Aerosystems -- Propulsion

Dynamics Research Corp. -- Guidance Analysis

Geophysics Corporation of America -- Scientific Measurements

Gulton Industries, Inc. -- Power Conditioning Equipment

Hoffman Electronics Corp. -- Solar Cells

Minneapolis-Honeywell Regulator Co. -- Guidance and Control

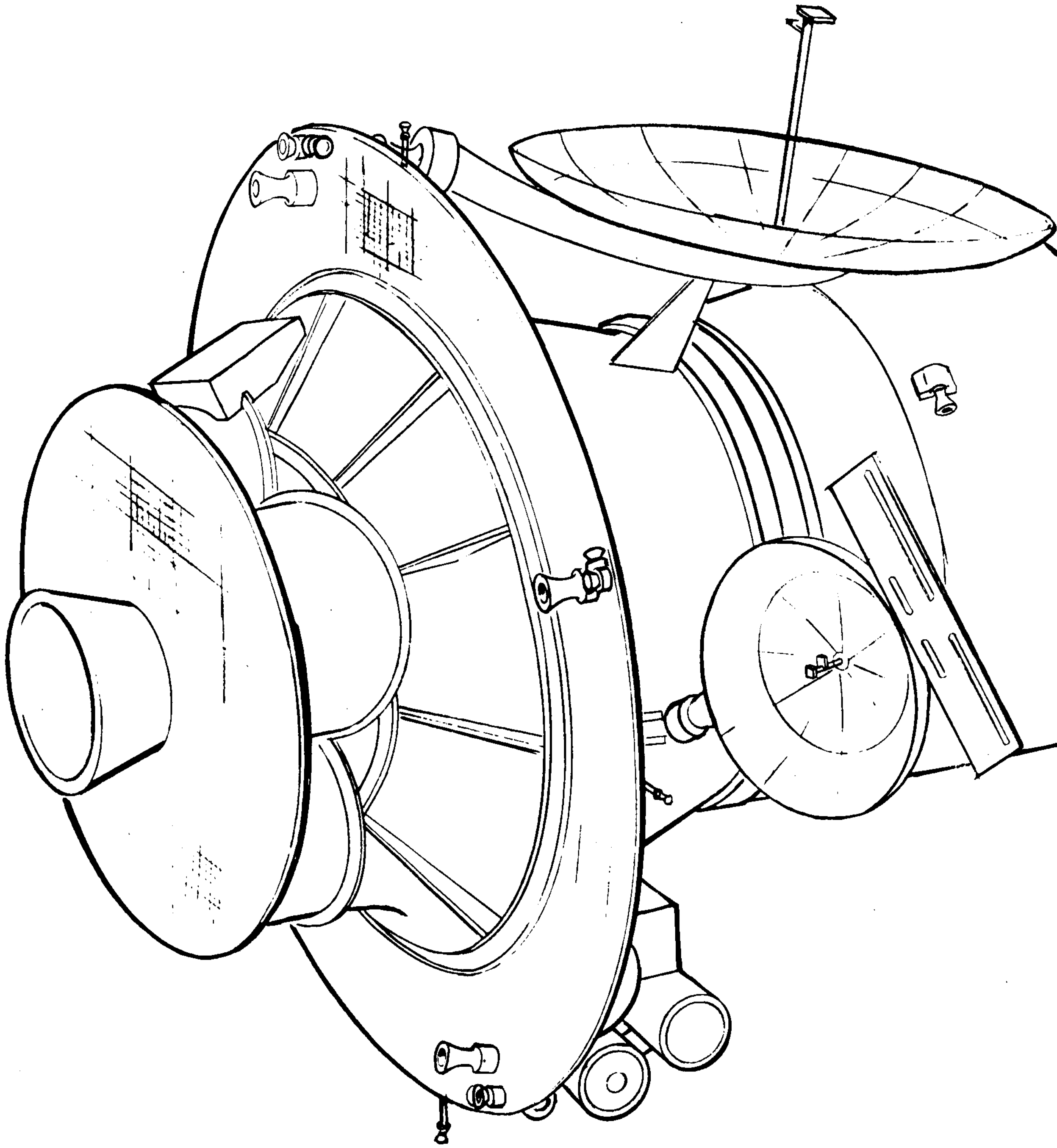
Northrup-Ventura -- Descent System

Raytheon Space Systems Division -- Communications and Television Mapping

Rocketdyne Division, North American Aviation -- Propulsion

Ryan Aeronautical Company -- Solar Panels

Wilmot Castle -- Sterilization



FRONTISEPIECE - ARTIST'S CONCEPTION OF THE MARS
VOYAGER SPACECRAFT
63-7995B

SUMMARY

This report presents the results of a 6-month conceptual design study conducted by Avco Research and Advanced Development Division for the National Aeronautics and Space Administration. The objectives of the study were the synthesis of a conceptual design of an unmanned spacecraft to perform scientific orbiter-lander missions to Mars and Venus during planetary opportunities from 1969 to 1975, and the formulation of a plan delineating the development program leading to first launch during the Mars 1969 opportunity.

The basic approach makes use of a 6000- to 7000-pound orbiter-lander; tradeoff studies were conducted to determine the payload and mission capabilities with smaller and larger spacecraft. The orbiter-lander was selected as yielding the maximum in scientific value short of manned exploration. The lander separates from the orbiter-bus and descends to the planet surface by parachute, where it makes atmospheric and surface measurements and conducts a variety of scientific experiments. The information obtained is relayed to Earth via the orbiter-bus which meanwhile is placed in a planetocentric orbit. The orbiter-bus collects scientific data in transit and maps the planet while in orbit. The lifetime of both orbiter-bus and lander is 6 months for the Mars missions. For Venus, the orbiter life is also 6 months, but the lander life is only 10 to 20 hours because of the hostile environment. A small capsule was designed for Venus, in addition to the lander, to conduct atmospheric measurements after entering from orbit; the capsule does not survive landing. Landers and capsules would be sterilized to avoid contamination of the planets, but the orbiter-bus would be placed on a trajectory which would ensure that it would remain above the sensible atmosphere for at least 50 years; thus, no sterilization would be required. The development plan shows that to obtain the scientific value desired, two spacecraft should be scheduled for each launch opportunity and hardware development should begin in 1964 to meet the 1969 launch date for Mars.

# 1. INTRODUCTION

The United States interplanetary exploration program was successfully initiated on 14 December 1962 when the Mariner II spacecraft passed within 22,000 miles of the planet Venus. As presently envisioned, the interplanetary program will consist of additional Mariner launches using the Atlas-Agena (Mariner C) and Atlas-Centaur (Mariner B) as launch vehicles. These would be followed by larger Voyager spacecraft using the Saturn booster and opening the way for manned exploration of the planets. In order to assist NASA in defining the mission objectives and to provide an orderly development plan for the conduct of the Voyager program, Avco Corporation's Research and Advanced Development Division has performed a comprehensive design study. This work was initiated in April 1962 under company sponsorship and was continued under a 6-month study contract to NASA Headquarters Office of Space Sciences in April 1963. This report presents the results of that study.

The study was carried out in two phases: The first phase included an examination of system tradeoffs and parametric studies so that basic design concepts could be selected. The second phase developed the selected concepts more fully and consisted of preliminary design of the spacecraft and its associated systems. In addition, parallel studies were performed to select the proper launch vehicle, to define the mission evolution, and to prepare a development plan for accomplishment of the mission. The Voyager program, as defined in this study, begins with the Mars launch opportunity in early 1969 and concludes with the 1975 Mars opportunity. In addition, launches to Venus beginning with the opportunity in mid-1970 are included. By NASA direction, the major portion of the study was devoted to the Mars mission because of the greater interest in Mars, the higher probability of biological life on Mars, and the better understanding of the Mars environment. Improved definition of the Venus environment is necessary to facilitate an authoritative design of a versatile Venus lander.

Configurations considered included a single lander, a single orbiter, and a split-payload spacecraft in which a lander is carried on an orbiter-bus. The basic design selected uses the third approach, in which the sterilized lander is separated from the orbiter-bus near the end of its interplanetary journey and proceeds on an impact trajectory. The orbiter-bus, meanwhile, acts as a relay for the lander and also collects scientific information in its own right while in orbit about the planet. Modifications to this approach are followed to make maximum use of the capabilities of the launch vehicle during each launch opportunity. The Mars orbiter-bus has a dry weight of 1849 pounds, including 135 pounds of science payload; it carries between 1500 and 3000 pounds of propellant. The orbiter-bus for Venus is basically the same design as that used for Mars, the major differences being the removal of some of the solar panels, different scientific instrumentation, and different surface coatings for thermal control. It has a dry weight of about 1576 pounds, including 180 pounds of scientific payload; propellant weight is 3710 pounds. The Venus and Mars landers are quite different, due to the significantly different planetary environment. The Mars lander, weighing 1680 pounds, is self-erectable, with a relay capability as well as a high-gain directional antenna, and carries 200 pounds of scientific instruments. For Venus, two entry vehicle concepts have been selected. One, a nonsurvivable atmospheric probe, or capsule, is carried into planetary orbit on the orbiter-bus before entering the atmosphere; it is small, weighing 190 pounds when on the orbiter (including 85 pounds for the de-orbit rocket) and carries 10 pounds of science payload. The second design is a direct-entry lander weighing 1330 pounds with 80 pounds of payload. The capsules will be employed first and are designed to enter the atmosphere from orbit in order to provide an initial conservative approach to the aerodynamic heating problem associated with high-energy direct entry. In addition, the capsules are not designed to survive landing, because there does not exist at present sufficient information on the atmosphere and surface environments to facilitate reliable design of a survivable lander. It is also anticipated that the number of Venus flights planned for the Mariner B program will not be sufficient to provide the required scope of exploration. The appendix to this volume provides a convenient summary of the spacecraft and subsystem design characteristics.

In all cases, the launch vehicle selected is the Saturn I-B with S-VI upper stage, having an injected weight capability of about 6000 to 7000 pounds for most opportunities. The use of a launch vehicle having an injected weight capability of 4000 pounds was considered but rejected as being inadequate for the split-payload missions employing the combined orbiter-bus and lander. The advantages of this split-payload approach were judged to be sufficiently beneficial to justify the use of the S-VI stage. The use of the Saturn V launch vehicle was also considered, but the conclusion reached was that the much greater payload capability would be better used for multiple landers or larger roving landers, rather than a single large stationary lander.

The greater payload capability of the Voyager spacecraft and the great variety of scientific information which can be collected over the lifetime of the program make it a significant step forward after the Mariner explorations. The study shows convincingly the need for a program of at least the magnitude of Voyager to prepare for manned flight to Mars. It shows further that the Voyager program must begin before the middle of 1964 to provide sufficient development and manufacturing time to meet the Mars 1969 launch opportunity. The recommended development plan calls for utilization of all Mars opportunities (1969, 1971, 1973, and 1975), with two spacecraft scheduled for launch on each date, and similar utilization of the Venus opportunities in 1970, 1972, 1973, and 1975. (The cost savings possible by omitting the Venus 1972 and 1975 launches is presented.)

Midway through the study, new data obtained by the Jet Propulsion Laboratory indicated that the atmospheric model of Mars was in question. As part of a change in contract, the effect of the new atmosphere was examined. The lander designed for the earlier (Schilling) model atmosphere is unsuited for use in the revised model, due to the much lower density and high surface-wind velocity. An alternate design was prepared for this atmosphere. Except where stated to the contrary, all discussion herein is based on the early (Schilling) model.

The cost of the Voyager program proposed is $798 million; this includes all design and development, manufacturing, testing, and facilities, but does not include launch vehicles, scientific payload, or other Government-furnished support. The cost of each launch opportunity is between $30 million and $36 million depending on the configuration (not including launch vehicle).

## 2. MISSION EVOLUTION

The mission evolution for Voyager exploration of Mars and Venus consists of three parts. The first is a review of fundamental mission tradeoff investigations and the decisions which resulted, the second is an investigation of the scientific objectives to be obtained from the program, and the third is the spacecraft schedule required to achieve the desired objectives; the third part is of course closely related to the payload capabilities of the booster and to the reliability of both booster and spacecraft. These topics are discussed in this section.

### 2.1 Mission Tradeoffs

To arrive at the Avco concept for the Voyager spacecraft, certain mission tradeoffs were investigated. The results of these tradeoffs were strongly influenced by the Voyager scientific objectives, weight limitations, and schedule boundaries that were furnished. Four major tradeoff areas were considered and influenced the final decisions on the final spacecraft configurations.

The decision was made to split the mission of the spacecraft into an orbiting vehicle and landing vehicle rather than to design a spacecraft that would be used for an orbiter or lander alone. This decision was the most important result of the tradeoff studies and was based on the following:

1. The ability of an orbiter and stationary lander to acquire more information together than either one alone;

2. The ability of the orbiter that would make scientific measurements to serve also as a relay for the lander. If a low-gain antenna is used in the lander, then the rate of transmission of information directly to Earth will be limited to 1 bit/sec. A lander which uses the same antenna system for relay of its information to an orbiter will achieve a bit rate of 4500 bits/sec;

3. The capability of a split payload to maximize the chance of obtaining both an orbiter and lander for a given launch opportunity;

4. The greater utilization of the components of the spacecraft. Since the orbiter will serve as its own bus, it can also serve as a bus for a lander. Alternately, the lander requires a bus, but the bus with added propulsion capability can also serve as an orbiter;

5. The broadening of possible lander sites to include the entire planet, rather than being restricted to only those locations that are visible to Earth during encounter.

The decision to design a split payload led to the selection of a booster that could launch a 7000-pound spacecraft. Studies showed that the split-payload spacecraft weight, necessary to meet program objectives and have a high probability of mission success, was less than 7000 pounds but greatly in excess of 4000 pounds. The results excluded the use of a booster that could launch only a 4000-pound spacecraft. Studies have shown that the spacecraft for Mars would be divided into 1800 pounds for the orbiter, plus 1500 to 3000 pounds of orbiter propellant, to achieve a 1700- by 10,000-km orbit, and 1700 pounds for direct-entry lander. For Venus, the orbiter weight was found to be 1600 pounds, and the weight of three atmospheric capsules which enter from orbit 570 pounds. The weight of the propellant necessary to place the orbiter and capsules into a 1000- by 10,000-km orbit was found to be 3700 pounds.

The decision was made to use a hard lander, i.e., a vehicle capable of completing its mission regardless of its attitude immediately following impact, rather than a soft lander, i.e., a vehicle always capable of maintaining its attitude prior to and immediately following impact. The hard lander was designed with a low-gain antenna system, so that communications would not depend upon its attitude. It was found that a high rate of transmission of information back to Earth could be achieved if the orbiter, designed to perform scientific measurements, were also used as a relay station. A backup capability using a direct link to Earth was also provided. The decision was made to incorporate design features into the spacecraft so that it would be adapted for exploration of both Mars and Venus, could carry more than one lander, and could also be utilized with a 60,000-pound spacecraft. In general, it was found that this adaptability could only be achieved with some degree of spacecraft modification.

## 2.2 Scientific Mission Analysis

1. Introduction. The justification of the Voyager project rests ultimately on the scientific knowledge to be gained from the spacecraft flights. Not only the central question as to the desirability of Voyager, but also the magnitude and timing of the effort can only be properly determined when the scientific objectives are carefully considered along with the many other important questions. An integrated, multidisciplinary study of each of the planets as objects of scientific interest, when carried out in combination with a conceptual design of the spacecraft, can provide the most effective measure of the capabilities of the project. Simultaneously, such consideration of the interactions among scientific measurements, spacecraft, and physical environment will lead to a more effective spacecraft conceptual design.

This analysis of the Voyager scientific program has been guided by three broad objectives established by NASA. In order of priority, these are

a. Exobiological Investigations

b. Geophysical/Geological Measurements

c. Manned Landing Information.

The most exciting possibility of planetary exploration is that of discovering evidence for the existence of life of nonterrestrial origin. The first objective of the Voyager program, exobiological investigations, is to obtain a definitive answer to the question of the existence of lifeforms on Mars and, of lower priority, in the cloud layers of Venus. The second objective of the Voyager mission is to gather geophysical and geological information concerning both Mars and Venus. This mission analysis treats each planet as an aggregate of matter acted on by exogenetic forces (solar, cosmic) and by endogenetic forces (radioactive, seismic), and attempts to determine the essential questions of interest and the relationships of the various measurements intended to answer those questions. The third priority objective of the program is to gather information necessary to make possible future manned landings on Mars. This includes physical information such as temperatures, pressures, and winds; data concerning human physiology, such as cosmic ray intensity at the surface; and information which may permit a degree of coupling of the man/machine spacecraft to the Mars environment.

Finally, the evolution of the Voyager program must be made to contribute to its own success. To the broad NASA objectives, Avco has added for this analysis the requirement that flights to each planet must gather the data necessary to enhance the probability of success of the later flights. In addition, the scientific data obtained on each mission must be effectively assimilated into the body of knowledge of the planets to guide the design of subsequent scientific measurements, and an effective procedure for this is desired.

Each of these scientific objectives contains a broader implication than merely the intellectual satisfaction of answering the specific questions concerning each target planet. This broader value is of course the application of the knowledge so gained towards understanding ourselves, our own planet, and the potential capabilities we have for exploring our universe. The discovery of a life mechanism in any way different from those available for study on Earth will provide a powerful stimulus to the understanding of the reasons for various evolutionary paths and the origin of life. Similarly, new bodies of descriptive geophysical/geological data from other planets will be very helpful in understanding the vast body of data which has been gathered concerning Earth. For example, if the atmospheric circulation of Venus is a simple convective regime, and that of Mars is a pure transient wave regime, then study of these may enable the complex circulatory patterns of our atmosphere to be more easily interpreted. Consequently, Earth's weather may be more easily understood, predicted, and eventually controlled.

Finally, the knowledge we gain to enable a future manned expedition to travel to Mars will also help us define the accessible limits of even more extended space travel. Mars represents a much better opportunity than does the moon for coupling a man/machine ecological loop to an alien environment, and should provide us with a firmer understanding of our potential capabilities in this respect.

The objectives of this expanded study of the scientific mission were considered to be fourfold:

a. To provide a complete analysis of the scientific capabilities of the Voyager project.

b. To determine and specify scientific requirements and to predict the physical environment to be encountered in order to ensure the most effective spacecraft design.

c. To examine the scientific objectives and mission constraints in sufficient detail so as to make possible a precise exploration schedule.

d. To organize the study in such a way that a capability would be provided for future continuing modification and updating of these studies as new information becomes available and as objectives evolve.

In order to achieve these objectives, a rather wide variety of activities was necessary. An outline of each of these is given below, together with a brief summary of the results.

2. Multidisciplinary study. The first stage undertook a study of the target planets as objects of scientific interest. This study consisted first of a review within each scientific discipline of the presently available body of knowledge concerning the planets. In several of the disciplines, this review was followed by original work consisting of the construction of theoretical models or hypotheses, mathematical analysis, and computations in order to provide predictions of planetary conditions. Finally, for each discipline, measurements were recommended which would be most valuable in advancing the state of knowledge of the planets. The details are contained in volume II, Scientific Mission Analysis.

3. Scientific measurements. The second stage of the mission study was an optimization analysis of the scientific measurements to be made. First, a master list of measurements was compiled by combining those suggested by each discipline, and eliminating duplications. Each of the broad objectives specified by NASA was assigned a weighting factor to reflect the relative emphasis given each objective. For each of these objectives, a value rating was assigned to each measurement on a scale of 1 to 10. The net value was finally obtained as the sum of the individual values each weighted according to the emphasis placed on each objective. The most valuable measurements determined by this procedure for the Mars mission are listed in table 1 in order of priority.

TABLE 1

HIGH PRIORITY MEASUREMENTS -- MARS

| Orbiter/Bus | Descent/Lander |
|---|---|
| Television Mapping | Television Mapping |
| Magnetic Fields | Biological Detection |
| Infrared Spectra of Surface | Atmosphere Pressure |
| Infrared Radiometry of Surface | Wind Velocity |
| Spectral Albedo | Atmosphere Temperature |
| Radio Absorption (Lander to Orbiter) | Atmosphere Composition |
| | Solar Optical Absorption |
| | Microscopic Examination of Soil |
| | Magnetic Field |
| | Density of Atmosphere |
| | Chemical Structure of Soil |

It should be pointed out that this scheme of attempting to optimize the scientific gain by means of a systematic analysis should be of continuing value. As more information becomes available from early flights, the value ratings for each measurement will change, and the relative emphasis of the broad objectives will also change. At the very least, this matrix of priorities, updated as desired, should be able to provide a guide and a check for program management, even if other criteria and judgments are subsequently applied to the results.

4. Instrumentation. Once the most important measurements had been selected, the next task was to determine the optimum instrumentation. At the start of the present study contract, NASA provided a list of scientific instruments to be used in the conceptual design studies. From this list, separate packages, or groups of instruments, of increasing weight were designed to accomplish the broad objectives of the Voyager program. Tradeoff studies including power and communications requirements indicated a payload capability equal to the largest of these lander packages, which totaled approximately 200 pounds of instrument weight.

The selection of the scientific instruments for the lander package considered first the utility of each of the possible instruments for the desired measurements. Where instruments were not available on the supplied list, additions were made in order to achieve a complete capability in the package. Considerations taken into account in selecting possible instruments for each desired measurement included weight, power, volume, and bits required, deployment and sample acquisition complexity, especially as it affected the spacecraft design, and utility for several different desired measurements.

Instrument operating cycles were established in order to determine the level of power and bit rates necessary for successful operation of the instruments over the first 24 hours. Schedules were also established for operation of the instruments at less frequent intervals over the succeeding days, weeks, and months to a total 6-month operating life.

5. Landing footprints. The next task was to determine the footprint capabilities of the spacecraft. That is, it was necessary to determine where and when a lander could be placed on the planet so as to study the different and varying conditions. Due to the relative emphasis on Mars, this detailed procedure was carried out only for that planet, and Venus has not yet been considered in this respect.

Several constraints enter into the net determination of the lander footprint:

a. The position of the relative velocity vector of the approaching spacecraft determines the 90-degree entry angle impact point on the planet.

b. The atmospheric entry trajectory determines circular areas (around the 90-degree impact point) achievable with lower entry angles. The minimum entry angle of about 20 degrees below which skipout occurs determines the maximum available landing area.

c. The requirement that orbiter-lander communications be possible during the critical entry and deployment phases restricts the lander to a corridor about the polar orbital plane.

d. The position of the sun determines the portion of the planet which is illuminated so that descent television pictures can be obtained.

e. The position of the Earth determines a maximum latitude below which direct lander-Earth communications are possible for at least some portion of each succeeding day.

The relative positions of the approach asymptote, the Earth, and the sun vary slowly throughout the 30-day launch window, but may be considered as fixed in inertial space for any given day. Their net effect is then to circumscribe an inertially fixed area at the surface of the planet which can be reached by the lander. This inertially fixed footprint, however, determines only latitude limits on the available landing sites. Any desired longitude of landing can be selected by adjusting the time of arrival.

6. Landing sites--Mars. A wealth of observational data is available for Mars, and it has therefore been possible to determine a reasonably detailed schedule of landings. The most obvious and well known phenomena on Mars are the permanent features and the changes in topography with time. Since the Martian pole is tilted relative to its orbital plane almost as is the Earth's, Mars undergoes seasons in its northern and southern hemispheres just as does the Earth. Since the eccentricity of Mar's orbit is somewhat larger than that of the Earth, the difference between its northern and southern seasons is somewhat more marked.

At the autumnal equinox for each hemisphere, clouds begin to form in the polar region. During the fall and winter, this cloud cover grows until the polar regions are completely and continuously covered. Than at spring in each hemisphere, the cloud cover disappears and a white surface polar cap is exposed which begins to recede toward the pole and is continuously bounded by a dark blue collar or band a few hundred kilometers wide. Simultaneously, the dark areas of the planet begin to darken, this darkening progressing as a wave down from the pole to and across the equator. The bright areas remain more or less unaffected by these seasonal changes.

The most popular explanation of these seasonal events suggests that the polar cap is composed of ice or snow which begins to melt in the spring, moistening a narrow band around the edge of the cap. Simultaneously, this moisture evaporates, is carried to the opposite pole, and brings about changes in vegatative life observed as the wave of darkening. The bright areas are interpreted as desert areas devoid of life observable from Earth. Since this theory bears so strongly on the question of the exobiological possibilities, the first mission of the Voyager program must be to land and observe this wave of darkening as it passes the landing spot, thereby making possible a direct observation of the cycle.

A graphical presentation of these phenomena is presented schematically in figure 1. Since these events vary primarily in latitude as time progresses, a plot of latitude versus time through one Martian year is utilized, and the seasons are indicated for each hemisphere. It should be emphasized that this is a somewhat simplified pictorial representation, as these events do not occur uniformly over the entire planet. The cap recedes erratically, some areas proceeding slowly and even temporarily leaving behind detached patches, and the wave of darkening proceeds preferentially along certain paths, all of these phenomena presumably being controlled by surface topography and atmospheric patterns. The width of the individual marks in figure 1 depicting the wave of darkening at each site is an indication of the photometric measure of the darkening in that area as a function of time.

Superimposed on this seasonal pattern are shown the footprint capabilities of the landing capsule. The time dimensions of the footprint are determined by the 30-day launch window and the variable time of flight between planets, and latitude limits are determined as explained earlier. As described elsewhere, in this volume, considerations of reliability dictate the number of attempts necessary to ensure a desired number of successful landings with a prescribed probability. In this section, landing sites are selected for each attempt, rather than only for the expected successes.

In 1969, a unique opportunity occurs to place landers in excellent positions on the planet just ahead of the peak of the wave of darkening. No other opportunity provides such a well placed arrival window, and it is felt that this presents a compelling scientific reason to initiate the Mars program in 1969. Although a similar opportunity occurs in the northern hemisphere in 1973, delays of over 100 days are required between landing and the peak of darkening as compared to about 30 days for the 1969 situation. This becomes a somewhat high-risk attempt for such an important phenomenon, especially if the first flights were to go in 1971 or 1973, and hence it is felt that the 1969 launch is very important.

In 1971, the footprint extends to the south pole, and at the season of arrival the polar cap will be about 300 km in diameter. It will probably not be possible to sit and allow the dark-collar phenomenon to pass the landing site since the lander dispersion requires aiming at the center of the cap and the rate of recession of the cap is only about 4.5 km/day. However, the next opportunity to study details of the cap itself in the north does not occur unitl 1975, and so it is recommended that the second lander on each launch vehicle be directed to the polar region. The first lander on each vehicle should be placed in a dark region where the wave has most recently passed.

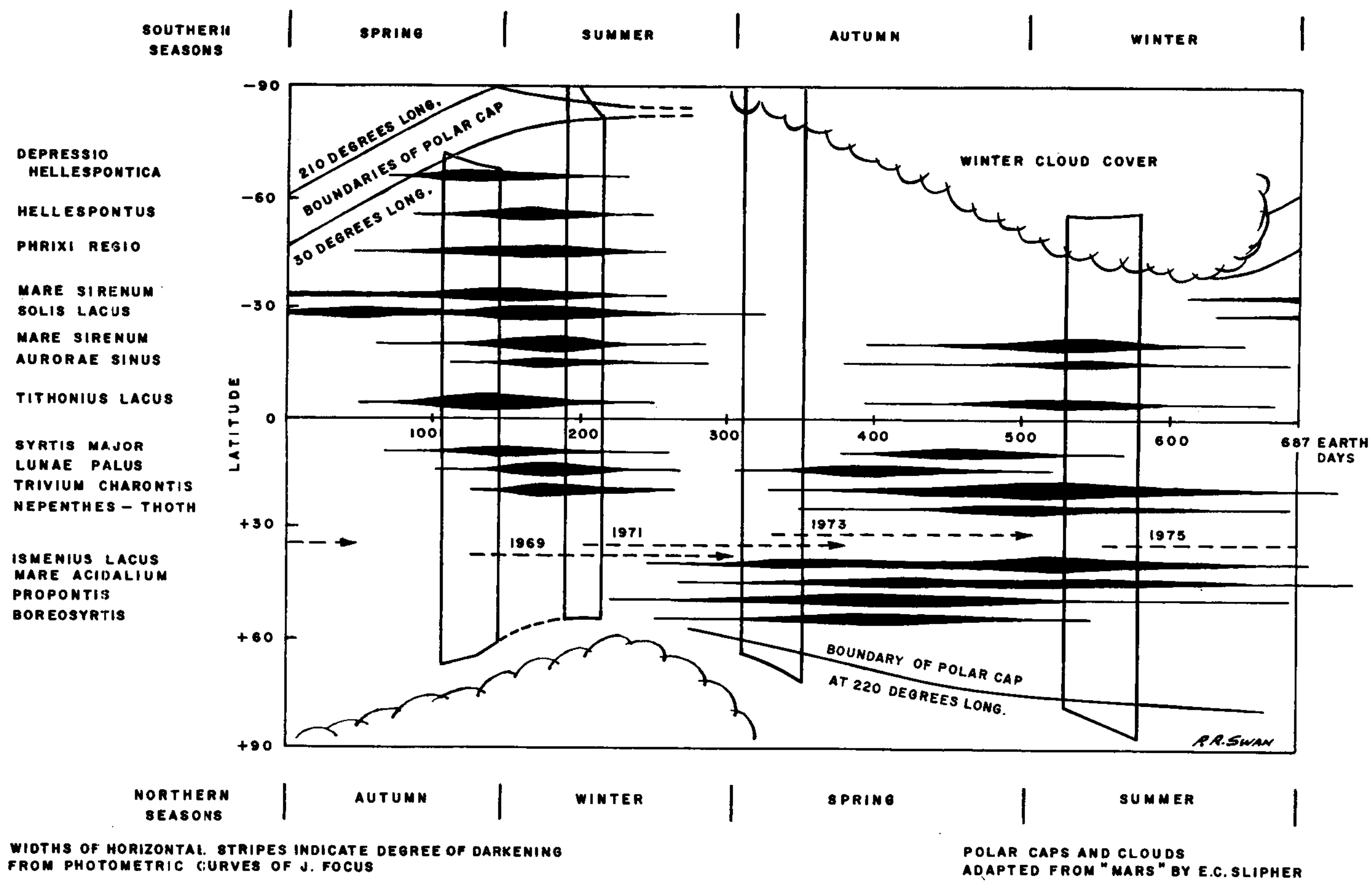



Figure 1 MARTIAN SEASONS AND VOYAGER LANDER FOOTPRINTS
63-9288

The 1973 window should be utilized to study the desert areas and so-called canals. One lander should be directed to the unique "pink" tinted desert regions as it is probable that previous landers in the dark regions will obtain information on the background "ochre" desert material. The lander from the second launch vehicle should be placed in a region where the "canal" and "oasis" phenomena have been observed. In 1975, the footprint extends far enough northward early enough to catch the polar cap, and it is recommended that the lander from one launch vehicle be placed to intercept the dark collar as it follows the receding cap. The second vehicle lander should be considered as being reserved for targets of interest which will undoubtedly be revealed by early Voyager flights.

In table 2, the recommended landing sites are listed for each opportunity, and in figure 2 these recommended landing sites are indicated on Slipher's map of Mars. The diameter of the circle represents the 3-$\sigma$ dispersion in the landing accuracy, which varies with latitude within each launch opportunity. The detailed reasons for the selection of each specific landing site are presented in volume II, and alternate sites are also presented there for use in the event of unforeseen alterations in plans and capabilities.

7. Landing sites--Venus. Since the scope of the study did not permit determination of footprints for Venus, the landing sites have been recommended on the assumption that all locations on the surface are available. If, for instance, type I and type II trajectories are required at different launch opportunities, then the subsolar spot and antisubsolar spot may only be accessible at specified times, and the following priority list may have to be modified.

A model for Venus based on observations and analyses carried out in this study is shown in figure 3. The stippled appearance is interpreted as atmospheric convective cells, the yellow subsolar spot is probably due to dust raised in that highly turbulent location, and the ionosphere will be formed primarily on the sunlit side but will diffuse to the dark side of the planet.

For 1970, the three descent capsules should be placed as shown in figure 3, two on the dark side away from the cold pole where conditions may be least turbulent, and one on the sun side to enable a comparison to be made. The second launch vehicle descent capsules should be similarly placed in the northern hemisphere, as shown by the dotted circles. In 1972, two of the descent capsules should be placed near the subsolar spot and one on the antisubsolar spot if the footprint allows this spread. These will thus obtain measurements of the extremes of conditions, thereby supplementing the 1970 information. The second launch vehicle payloads should be oppositely placed as shown.

In 1973 and 1975, survivable lander capsules should be placed near the cold spot discovered by Mariner II and near the hot and cold poles of the planet, but more detailed selections should await the results of the orbital infrared and microwave mapping in 1970 and 1972 flights.

TABLE 2

RECOMMENDED MARTIAN LANDING SITES

| Launch Opportunity | Lander | Landing Site | Longitude | Latitude |
|---|---|---|---|---|
| 1969 | 1 | Solis Lacus | 90° | -28° |
| | 2 | Syrtics Major | 286° | +15° |
| 1971 | 1 | South Polar Cap | 30° | -83° |
| | 2 | Mare Cimmerium | 235° | -18° |
| | 3 | Lunae Palus | 65° | +15° |
| | 4 | Aurorae Sinus | 50° | -15° |
| 1973 | 1 | Propontis | 185° | +45° |
| | 2 | Elysium | 210° | +25° |
| 1975 | 1 | North Polar Cap | 220° | +78° |
| | 2 | Nepenthes-Thoth | 255° | +25° |

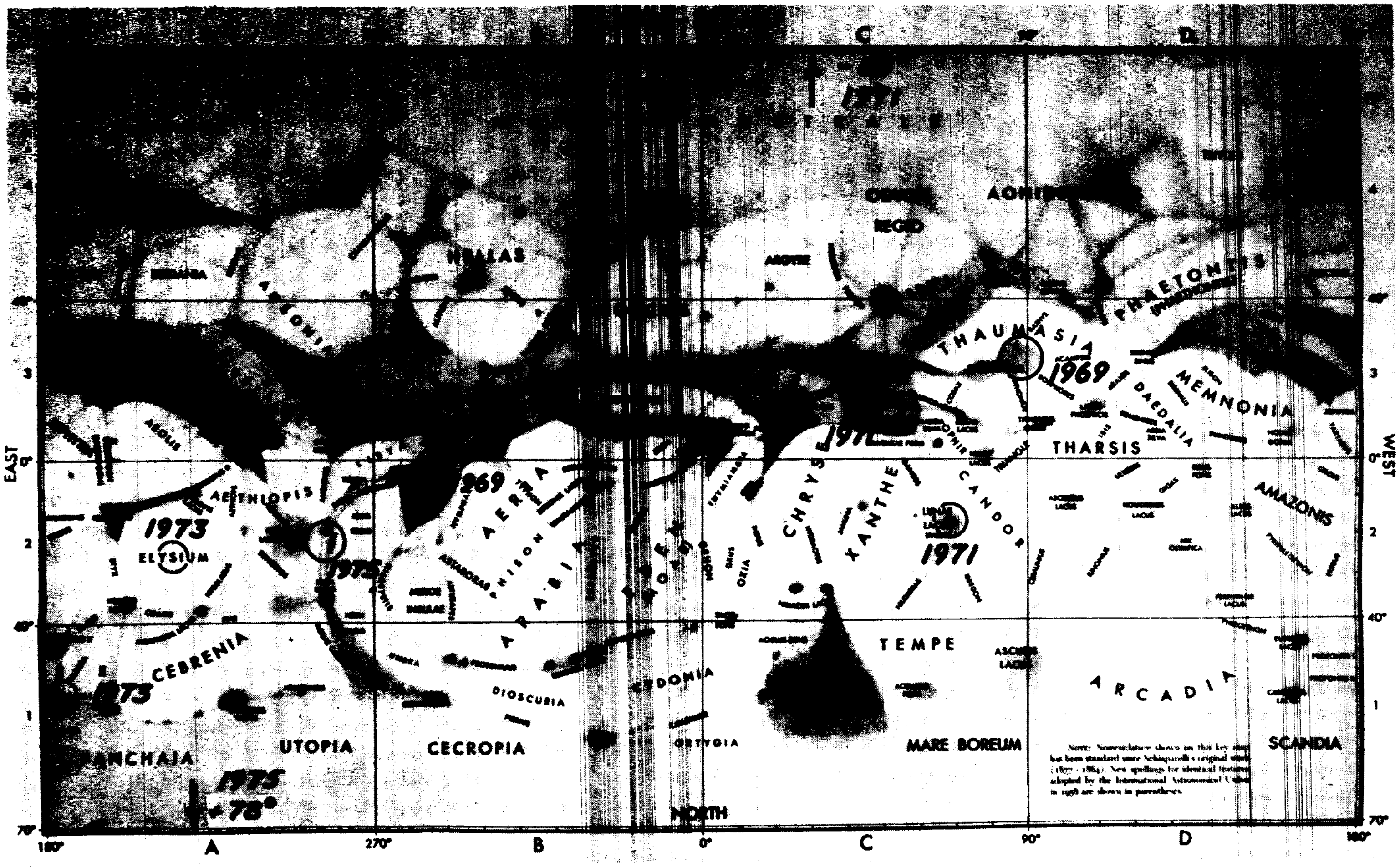



Figure 2 MARTIAN LANDING SITES
63-9976

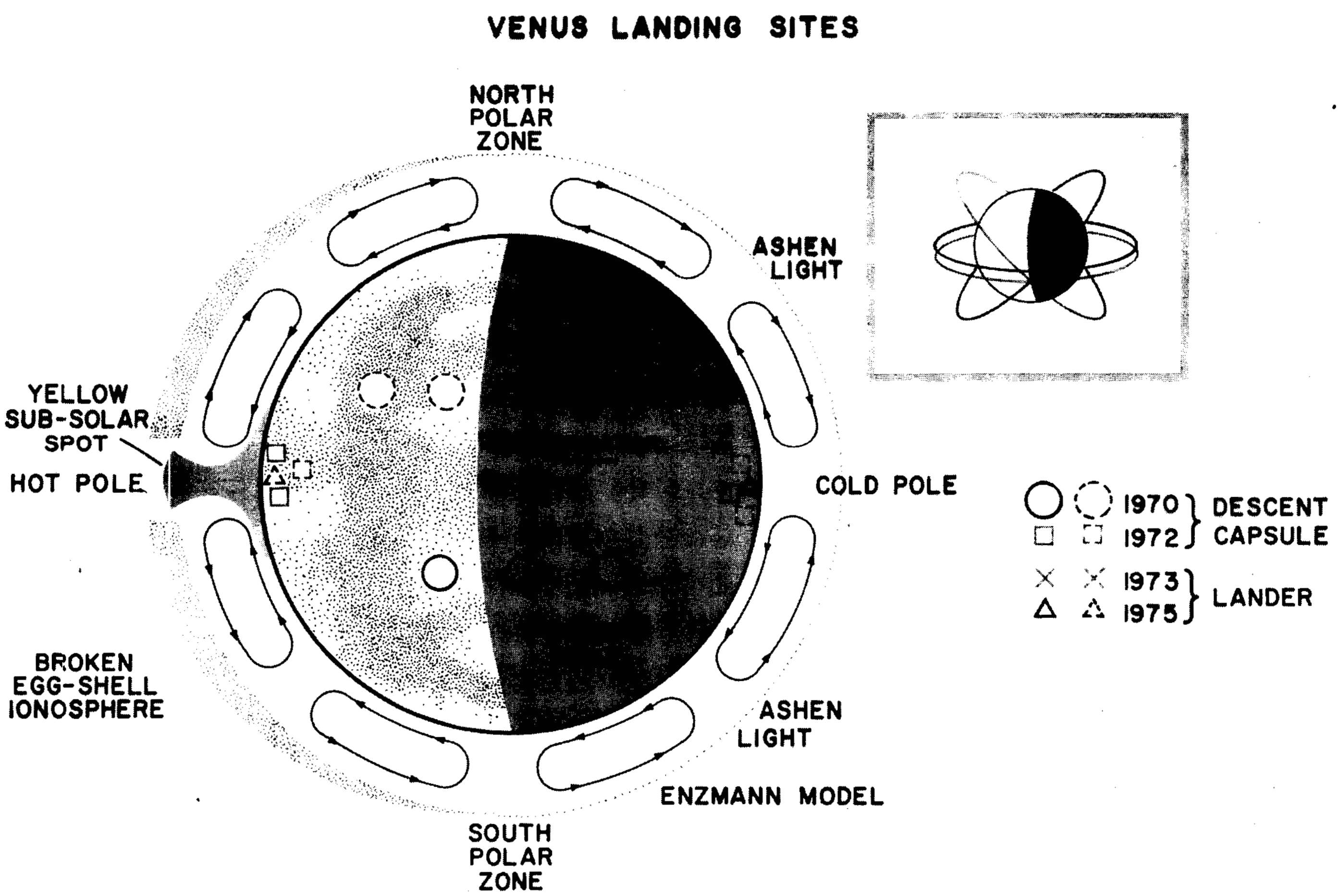



Figure 3 VENUS LANDING SITES
63-8480

## 2.3 Mission Plan

In order to obtain the level of scientific information from the Voyager program indicated in the previous sections, the numbers of successful orbiters, landers, and capsules shown in table 3 are necessary.

TABLE 3

NUMBER OF VOYAGER VEHICLES REQUIRED FOR SUCCESSFUL PROGRAM

| | Mars | Venus |
|---|---|---|
| Orbiters | 2 | 2 |
| Landers | 5 | 2 |
| Capsules | - | 4 |

The two Mars orbiters will permit observation of the planet over different and changing seasons of the year. For Venus, the slow rotation of the planet requires that at least two orbiters be employed to achieve wide-area surface mapping, the two vehicles being in different orbital planes. The Mars landers are selected to observe the several areas of interest. Two are chosen for two different dark areas or more, a third for the edge of the polar cap, the fourth for the bright or desert areas, and the fifth is unassigned, its target to be selected as a result of the features of interest determined by earlier missions. The Venus capsules are flown in the initial opportunities to observe the atmospheric properties in the vicinity of the hot pole and cold pole. (These are defined as the intersections of the sun line with the two hemispheres, the hot pole being the subsolar point. Due to the planet's slow rotation, the surface coordinates of these points change slowly, if at all.) Due to the probability of excessive turbulence in those regions, capsules would also be targeted for regions in the light and dark hemispheres well separated from the hot and cold poles. One of the two landers would be chosen for the hot pole; the second would investigate the cold spot. (The cold spot is the area of low temperature detected by Mariner II; it is not the same as the cold pole.)

Once the number of successful landers, orbiters, and capsules desired is specified, the number of launch attempts which must be made to achieve a given probability of success can be determined. The launch vehicle used for the study has a capability of injecting a spacecraft of 6000 to 7000 pounds on the appropriate interplanetary trajectory. This weight class permits combinations of orbiters, landers, and capsules which take full advantage of the differing launch opportunities. As described in subsequent sections, the orbiter/bus design permits the installation of either one or two landers or three capsules. The selected configurations use the basic orbiter/lander spacecraft in 1969 and 1973 for Mars. In 1971, the more favorable energy requirements permit the orbiter to be used with two landers. In 1975, the two-lander configuration is used again, but this time the orbiter/bus acts only in its bus capacity and does not enter a planetary orbit. A preliminary review of booster and spacecraft reliability indicated that more than one attempt per launch window would be desirable. Other aspects of the reliability study, which is reported in full in volume III, included a survey of booster reliability, the prediction of orbiter and lander expected mission success, and the use of an expectation technique to relate booster/spacecraft reliability to the probabilistic fulfillment of Mars and Venus program objectives. An examination of the reliability of presently available boosters, their relative complexity, and their reliability growth was used to provide an estimate of future Saturn booster reliability which could be assumed for the program. Reliability predictions were made for the various segments of the orbiter and lander missions. The expected success of an orbiter/lander mission was calculated as the sum of the product of the reliability of each mission segment and the contribution of that segment to the overall mission success. For each launch opportunity, the expected mission success was determined on the basis of projected, sequential reliability growth, assuming a full-scale reliability effort. Using the orbiter and lander program objectives and the fraction of total expected mission successes out of the reference launch configuration trials, binomial probabilities of fulfilling program objectives were calculated. This showed that the probability of fulfilling the objective of five landers was 0.90 and of two orbiters 0.85.

The results of this analysis are summarized in table 4, which restates the program objectives for Mars, lists the configurations by date, and gives the probability of fulfilling the objectives with the schedule proposed.

The schedule for Venus makes use of two spacecraft configurations. The 1970 and 1972 launch opportunities utilize the orbiter/bus to transport three small capsules into planetary orbit; from there, the capsules are ejected to enter the atmosphere. In 1973 and 1975, the orbiter/bus would carry only the single direct-entry lander. In all cases, two identical spacecraft would be launched in each opportunity. The probability of achieving the stated Venus objectives is 0.87 for four capsules, 0.78 for two landers, and 0.97 for two orbiters.

An alternate and less ambitious plan was considered as a cost reduction measure, in which the 1972 and 1975 Venus opportunities were omitted. The objectives desired were also more modest. The probability of achieving three successful capsules is 0.43; of one lander, 0.79; and of two orbiters, 0.67.

The Venus program is summarized in table 5.

## TABLE 4

### MARS MISSION EVOLUTION

Mission Objectives: Five Landers, Two Orbiters

Proposed Schedule:

| Date | Spacecraft* |
|---|---|
| 1969 | Orbiter/Lander |
| 1971 | Orbiter/Two Landers |
| 1973 | Orbiter/Lander |
| 1975 | Bus/Two Landers |

Probability of Fulfilling Objective:

| | |
|---|---|
| Five Landers | 0.90 |
| Two Orbiters | 0.85 |

*Two identical spacecraft are launched during each opportunity.

## TABLE 5

### VENUS MISSION EVOLUTION

Mission Objectives: Four Capsules, Two Landers, Two Orbiters

Proposed Schedule:

| Date | Spacecraft[1] Plan I | Spacecraft[1] Plan II |
|---|---|---|
| 1970 | Orbiter/Three Capsules | Orbiter/Three Capsules |
| 1972 | Orbiter/Three Capsules | ---- |
| 1973 | Orbiter/Lander | Orbiter/Lander |
| 1975 | Orbiter/Lander | ---- |

Probability of Fulfilling Objective:

| Plan I | | Plan II | |
|---|---|---|---|
| Four Capsules | 0.87 | Three Capsules | 0.43 |
| Two Landers | 0.78 | One Lander | 0.79 |
| Two Orbiters | 0.97 | Two Orbiters | 0.67 |

[1] Two identical spacecraft are launched during each opportunity.

# 3. SYSTEMS ANALYSIS

The systems analysis consists of a review of the parametric and tradeoff studies which were conducted to determine the interactions between subsystems and design disciplines and which were essential in providing guidelines and requirements for the design studies which they accompanied. This section begins with a brief description of the Voyager mission, from launch until the mission is completed. This is followed by discussion of some of the major system tradeoffs, trajectory and payload optimization, reliability, and sterilization. The details of this work are contained in volume III.

## 3.1 Voyager Mission Profile

1. Mars sequence. The sequence of events in the Mars 1969 mission is [illegible] injection (1)[1] by the Saturn booster into an interplanetary transfer orbit, the antennas and boom-mounted sensors are deployed. The vehicle then goes into the attitude acquisition mode. The roll axis is oriented toward the sun for pitch and yaw control. Roll control is obtained by referencing to the star Canopus. The communication antenna is commanded to the proper angle to acquire Earth. Attitude orientation during cruise is maintained by cold-gas reaction jets controlled by outputs of the sun sensor and Canopus tracker. On DSIF (deep-space instrumentation facility) command, the 35-watt S-band transmitter is turned on and off regularly in transit for tracking and verification of commands and also for transmitting scientific measurements at low bit rates using the 4-foot parabolic antenna. Scientific measurements are made throughout the interplanetary journey.

The first midcourse correction (2) is made approximately 1 week after launch. The digital control unit (DCU) in the guidance and control system is turned on by DSIF command. All the steps in the midcourse correction are then programed by the DCU. The gyros and accelerometer are turned on and nulled. Information as to direction and magnitude of the velocity increment required to correct the vehicle trajectory is received and stored. The spacecraft is reoriented for thrust. The antennas and instruments mounted on booms are stowed during any thrust maneuver because of the stress imposed by the acceleration and also to prevent a shift in the center of gravity. The correct rocket engine burn time is determined by accelerometer measurement. After rocket firing, the vehicle is reoriented to the sun-Canopus reference attitude using rate-integrating gyro information. The acquisition mode is repeated to reacquire the sun, Canopus, and Earth. The DCU is then turned off, and the cruise mode is resumed. A second midcourse correction (3) is made about 2 weeks after

[1]Numbers in parentheses correspond to numbered events on figure 4.

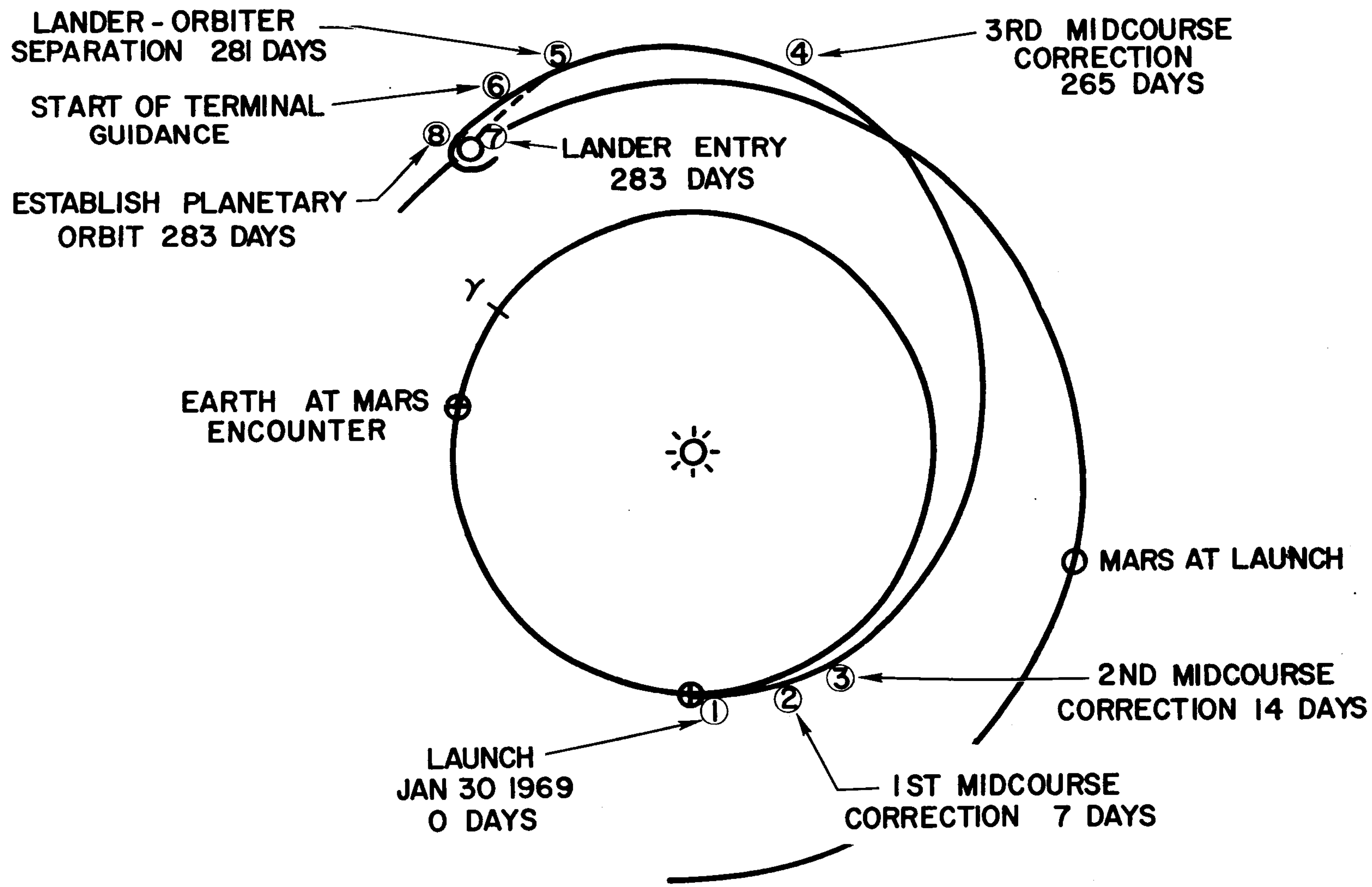



Figure 4 MARS 1969 LAUNCH - TYPE II TRAJECTORY
63-8483

launch. The third correction (4) at $10^6$ km from planet encounter corrects for the effect of uncertainty in solar-radiation pressure.

Approximately $10^6$ km (5) from planet encounter, the DCU is again turned on by DSIF command for programing of lander-orbiter separation operations. In the event of malfunction, DSIF command can be used as emergency backup. The gyros and accelerometer are turned on and nulled. The DCU receives and stores information for a) time and direction of lander launch and magnitude of velocity increment to be imparted, and b) direction and magnitude of orbiter retrothrust. The spacecraft is oriented in the proper direction, and the lander is separated and then pushed away from the orbiter by a spring mechanism. The lander is spun up by four solid rockets and the sterilization shield is jettisoned. After sufficient separation to prevent plume impingement, the lander rocket is fired in a direction normal to the flight path in order to alter the trajectory to achieve planet impact at the predetermined landing point.

The orbiter is now reoriented, and retrothrust is applied so that it will lag behind the lander at planet encounter. This is required in order to have the orbiter in the proper position to relay to Earth the data transmitted by the lander during entry, descent, and impact.

The planet tracker is turned on (6) and lock-on to the planet is obtained. Small corrections are made to the orbiter trajectory, as required, following a procedure similar to that for midcourse corrections. An alternative procedure is to incorporate these corrections into the retropropulsion phase.

The lander vehicle starts collecting data at entry into the planetary atmosphere. (7) Scientific and engineering data from the lander obtained during entry, descent, and impact are relayed to the DSIF via the 35-watt S-band transmitter on the orbiter.

At planet encounter, the orbiter is reoriented, and retrothrust is applied (8) to achieve the proper orbital injection velocity. The desired orbital parameters are 1500 km periapsis altitude and 10,000 km apoapsis altitude. Due to guidance errors, the nominal aiming point is 1700 km periapsis, to ensure that the minimum allowable value of 1500 km is complied with. (This minimum altitude will provide an orbital lifetime of over 50 years for the unsterilized orbiter.) The local vertical is established by a horizon sensor. There is a switch over to the 120-watt S-band transmitter for transmission at high bit rate of the television mapping data and relay of surface scientific measurements made by the lander vehicle. The axis of the planet-oriented instrumentation is aimed along the local vertical, and the planet mapping sequence is initiated. In addition to the television camera, instruments used in orbit are an infrared radiometer and a microwave spectrometer. The same sequence used for midcourse correction can be repeated on DSIF command for orbit corrections if required. Mission lifetime in planetary orbit is designed for 180 days.

Before electrical disconnect of the lander from the orbiter, the computer clock on the lander is started by discrete signal from the orbiter computer. The lander entry transmitter (50-watt VHF omnidirectional antenna) is initiated for transmitting from separation to entry, at low bit rate, engineering status data to the orbiter relay and survival information to the DSIF. At entry, a programed sequence of scientific and engineering measurements is begun. Acceleration, pressure, and altitude measurements are made. The high bit rate output of the VHF transmitter is turned on by an accelerometer switch. Entry measurements and survival data are transmitted in real time. Scientific and engineering data are recorded during the communication-blackout period. The drogue chute is fired at a preset Mach number of 2.5. Base pressure and accelerometer measurements are used to determine Mach numbers. At 15,000 to 20,000 feet altitude, the main chute is opened by radar-altimeter output. A shaped charge is ignited, jettisoning the front and rear heat shields and the drogue chute, and at the same time pulling out the main chute. The VHF entry transmitter in the heat shield is turned off and the S-band 70-watt and VHF relay 50-watt transmitters are turned on, using slot antennas.

Data are obtained and recorded from the instrumentation during descent to determine atmospheric composition and to provide television pictures.

The VHF relay link transmits recorded descent data, TV pictures, and some real-time data to the orbiter relay during a 10-minute period centered about planet impact. The S-band transmitter, which transmits at a very low bit rate directly to the DSIF, is used for emergency backup.

After the lander impacts the planet surface, the parachute is released and a mechanism is actuated to erect the vehicle. The transmitters are turned off and data playback stops 5 minutes after impact. The parabolic antenna of the direct link transmitter (70 watt, S-band) is erected for DSIF communications. The attitude-sensing system and navigation computer are activated to orient the antenna toward the Earth. The programed scientific instrumentation sequences for surface measurements are started.

After approximately 40 hours, scientific data collected on the planet surface are transmitted at regular intervals by the VHF relay transmitter on orbiter relay command and by the direct link S-band transmitter on DSIF command. Data collection and transmission continue for a period of up to 6 months.

2. Venus sequence. The sequence of events in the Venus 1970 mission is shown in figure 5. The Venus mission profile differs from the Mars mission primarily in the following respects:

There are two alternate entry vehicle designs for Venus; these are a capsule and a lander. The capsule design for the Venus mission calls for launch from orbit in order to slow down the vehicle before entering the dense Venus

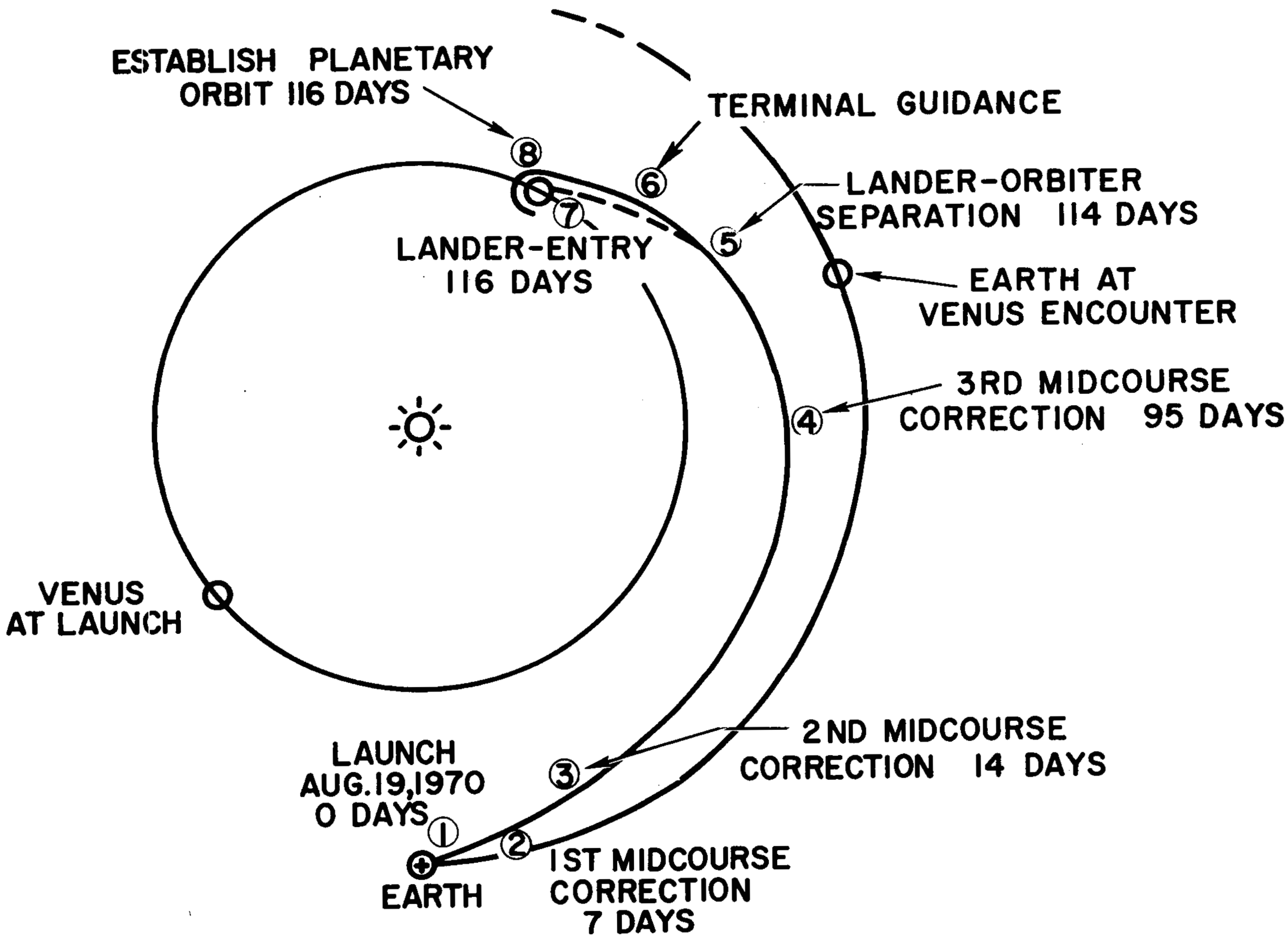



Figure 5 VENUS 1970 LAUNCH - TYPE I TRAJECTORY
63-8484

atmosphere. In this design, only atmospheric measurements are made, and the vehicle does not survive impact. In the lander design, separation and launch procedures are similar to those for the Mars lander, and both atmospheric and surface measurements are made. Lifetime on the surface is limited to 10 to 20 hours by the high-temperature environment.

The orbit will be as near circular as energy requirements and payload will permit. The periapsis altitude will be 1000 km with an apoapsis altitude between 1000 and 10,000 km, depending on the launch date.

Surface mapping of Venus is accomplished by radar and radiometer in the microwave frequency rather than by television.

All transmission from the Venus lander and capsule is via orbiter relay. There is no direct transmission to the DSIF. This eliminates the requirement for a directional antenna as well as the requirement for a navigation system aboard the lander for orienting the antenna.

A drogue chute is not used in either design for the Venus entry vehicle.

The Venus lander vehicle, designed to survive impact, is not erected after impact.

The programed scientific instrumentation sequences for surface measurements will be shorter and simpler due to the limitations imposed by the high surface temperature of Venus.

## 3.2 System Tradeoffs

The principal system tradeoffs include the technique of obtaining proper spacing between lander and orbiter during encounter, choice of lander-orbiter separation range, lander-orbiter communications, spacecraft orientation, and direct versus relay communications to Earth. These questions are considered in this section. Other tradeoff studies pertinent to specific disciplines are discussed in the appropriate design section of this report.

1. Lander-orbiter relay geometry. Engineering and scientific measurements made by instruments on the lander during atmosphere entry and descent are recorded for later playback. This data is transmitted to the orbiter relay during a 5-minute period before planet impact. Also, 5 minutes are required after impact for transmission of data in real time. To obtain the necessary 10-minute communication time, the lander must lead the orbiter so that the orbiter remains within the lander antenna beam during the communication period. The zone of possible lander-to-orbiter communication must be large enough to

allow for descent time uncertainty (approximately 10 minutes), uncertainty in orbiter position, and uncertainty in lander beam location due to lander dispersion and vertical alignment error of the lander antenna.

An analysis was performed of orbiter-lander geometry at planet encounter using approach velocities ranging from 3 to 6 km/sec. (See figure 6.) The results show that a lander transmitter of 15,000-km range and beamwidth of 120 degrees with sidelobes up to 150 degrees will provide adequate communication for landing sites corresponding to entry angles from 30 to 90 degrees. There is insufficient communication time at entry angles shallower than 30 degrees, especially at the higher approach velocities of 5 and 6 km/sec. Lander-orbiter communication requirements restrict landing sites to an area within approximately 30 degrees central angle of the orbiter trajectory plane.

2. Lead-time requirements. The required lander lead time is also determined from the geometric analysis illustrated in figure 6. It is defined as the difference in time between nominal orbiter periapsis passage and lander atmospheric entry. The amount of lead time required to provide the necessary communication time depends on the entry angle and approach velocity, and varies between 30 and 70 minutes. The required lead time can be achieved by accelerating the lander or by slowing down the orbiter. The magnitude of velocity change required along the flight path is a function of the lead time required, the separation range, and the approach velocity.

The method selected for obtaining lead time was to slow down the orbiter and impart a velocity increment to the lander normal to the flight path in order to change it from a fly-by to an impact trajectory. The following factors were considered in making the selection:

a. Accuracy of achieving desired landing site. Outside of the uncertainty in vehicle position at separation due to guidance error, the most significant source of lander dispersion is the error in magnitude and direction (launch angle) of the velocity increment imparted to the lander. The dispersion due to uncertainty in the magnitude of the velocity increment is a function of the normal component of the velocity increment and is therefore independent of the method of obtaining lead time. The dispersion due to launch-angle error is a function of the total velocity increment and the cosine of the launch angle. Since the required velocity increment along the flight path is much larger than the normal component, the launch angle would be close to 0 degree in case of lander speedup. This would maximize the effect of launch-angle error. If no velocity increment is applied to the lander in the direction of the flight path but is applied only normal to the flight path, then the launch angle is 90 degrees and the dispersion due to launch angle error is reduced. In order to demonstrate the significance of the launch-angle error, typical design parameters and an operational sequence were assumed for the case of lander speedup. Using a separation range of $10^6$ km, an approach velocity of 3 km/sec, a rocket action time of

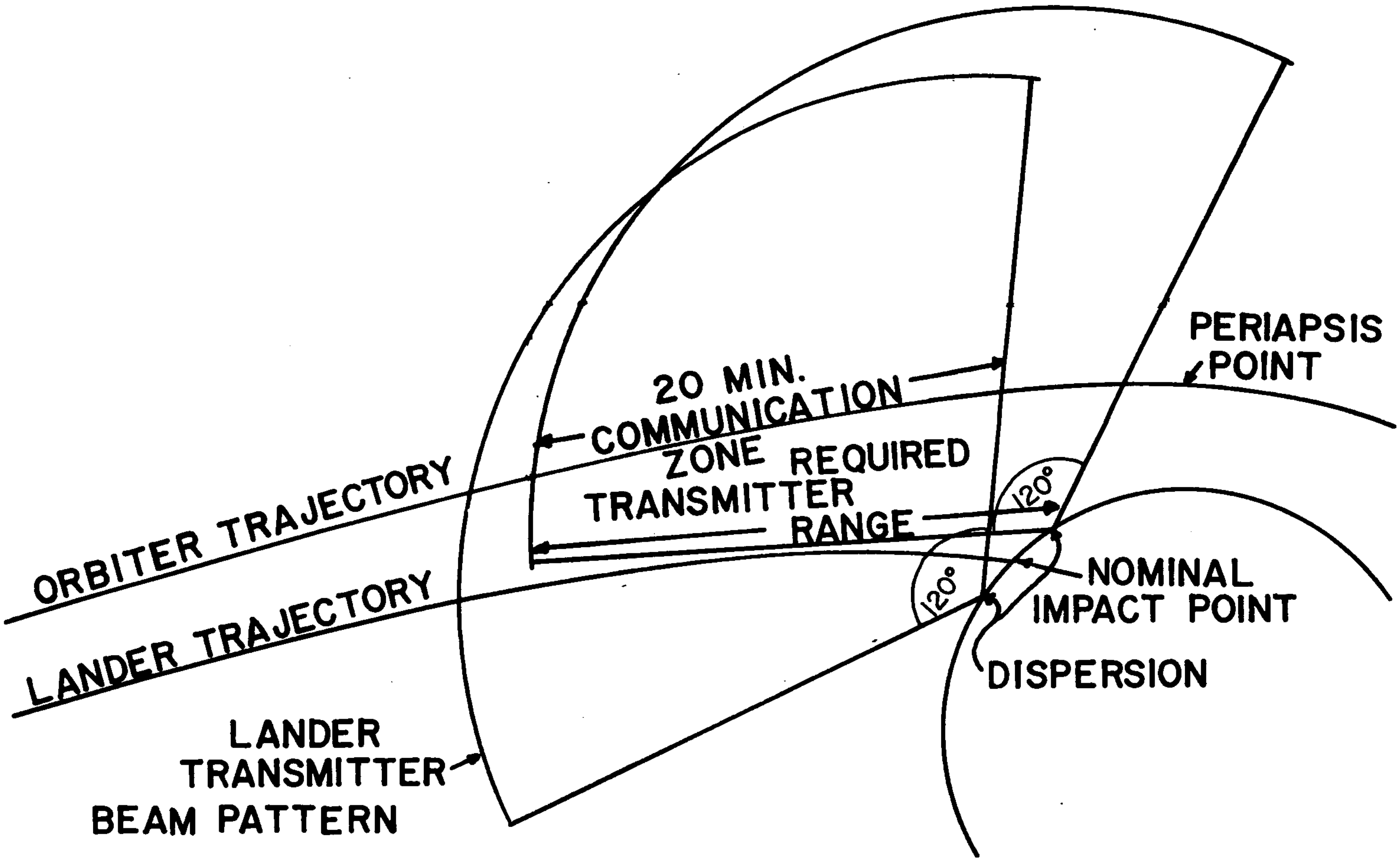



Figure 6 LANDER-ORBITER RELAY GEOMETRY
63-8482

10 seconds, and a spin rate of 1 rad/sec, the dispersion due to launch-angle error ranged from 220 to 520 km ($1\sigma$) for entry angles from 30 to 90 degrees. As the entry angle approaches zero, the dispersion due to launch-angle error rapidly increases. The lander launch angle error can be decreased by increasing spin rate or rocket action time, but the alternative choice was made to slow down the orbiter in lieu of speeding up the lander.

b. Accuracy of establishing orbit. Dispersion in orbiter periapsis altitude could result from retrovelocity errors similar to those causing lander dispersion. Since the orbiter is provided with a precise attitude-control system, accelerometer, and a restartable engine, the major contributions to error can be eliminated. Further, errors produced in the orbiter trajectory by retrothrust can be detected by DSIF guidance or terminal guidance and corrected prior to or during orbit injection.

c. Effect on weight of lander and orbiter. The weight of lander propulsion which would be required to produce an additional 500 ft/sec along the trajectory is about 92 pounds. However, retropropulsion of the orbiter after separation does not add significantly to the orbiter propellant weight since it reduces the required orbital injection velocity at planet encounter. Therefore, deceleration of the orbiter results in a weight saving which can be used to provide additional payload on either orbiter or lander.

d. Effect on sterilization requirements. Applying a velocity change to the orbiter may increase the probability of the unsterilized orbiter impacting on the planet. However, an unlikely sequence of events must occur to cause orbiter impact. The malfunction must be undetected prior to rocket firing. The velocity change due to the malfunction must be in the proper direction. The DSIF command to correct the trajectory error must fail to be carried out. If the probability of these events occurring is shown to be unacceptably high, the velocity change could be applied in smaller increments, allowing time between impulses to ensure by DSIF tracking that the retrothrust maneuver is being performed correctly.

e. Complexity of design. Slowdown of the orbiter does not increase system complexity, since it makes use of equipment already present. Speedup of the lander would impose high spin rates for stabilization. This would lead to structural and mechanical design problems, resulting in increased weight. Also, a despin device would be required prior to entry into the atmosphere. Acceleration of the lander would result in higher entry velocities with a consequent increase in heating and loads.

3. Lander orbiter separation range. Selection of the separation range is largely a tradeoff between propulsion weight required to impart velocity changes and lander vehicle dispersion. The propulsion weight decreases with separation range while the lander dispersion tends to increase. Both propulsion weight and

dispersion increase with approach velocity. If the separation range is greater than $0.5 \times 10^6$ km, the required velocity increment is small, and the effect of propulsion weight on payload is minor.

The dispersion as a function of range is predominantly influenced by the type of guidance system employed. The DSIF is able to determine the position of a vehicle at Mars distance to an accuracy of 150 km ($1\sigma$). The use of this type of guidance alone results in essentially constant error regardless of separation range.

The recommended guidance technique utilizes self-contained terminal guidance based on a planet tracker and star tracker to supplement the DSIF. In this case, the accuracy in determining position and velocity improves as the vehicle approaches the planet.

4. Spacecraft orientation. In making a selection of the reference attitude of the spacecraft, consideration must be given to the equipment on the craft which must be pointed in various directions and to the sources available to provide reference directions. The selection of solar energy as the source of power immediately identifies the solar panels as the primary object which must be pointed toward the sun. The advantages obtained in a rigid structure by not deploying the panels, and the real engineering problems involved in mounting the panels on gimbals due to their large size, led to the selection of a sun-oriented configuration. The second reference direction is provided by the star Canopus which is chosen because of its brightness and because of its location near the south ecliptic pole. Knowledge of the spacecraft position obtained by DSIF tracking together with ephemeris data permits the gimbal angles to be computed for pointing the Earth-oriented communication antennas. The Canopus tracker is oriented so that a single gimballed mirror is all that is required for pointing, as the angle between the sun line and the star line changes during the interplanetary voyage. Thus, the gimballing of the antenna and star tracker pose no difficulties with the sun-Canopus orientation. When in planetocentric orbit, additional equipment must be pointed toward the planet. All planet-oriented science is mounted on a single gimballed platform which is attached to the periphery of the solar panels. This allows convenient pointing of television cameras, radar, and other experiments as the spacecraft holds its fixed orientation and the platform turns about two axes at the orbital rate. Pointing is accomplished by a horizon sensor mounted directly on the platform itself. Since the spacecraft is in an elliptical orbit, the mapping gimbal must rotate at a varying rate. This causes some expenditure of attitude control system fuel, but the amount is not excessive. Smooth drive of the platform over its limited dynamic range is provided by a servo motor on each axis which is coupled to the platform axis through a speed-reducing unit.

5. Direct versus relay communications. The large quantity of scientific information obtained by the lander requires a high-capacity information channel to Earth. This can be provided with reasonable power either by a direct link

using a high-gain antenna or by relay through the orbiter using a low-gain lander antenna and the orbiter high-gain antenna. The disadvantage of the direct link is the difficulty in designing a lander which can be erected after impact so that the large antenna can be pointed toward Earth. The design selected for the Mars lander utilizes a direct-link antenna in which the lander is designed to reerect itself after landing. The present lack of knowledge of the terrain of Mars makes it difficult to be sure that the reerection mechanism will function under all possible circumstances. However, the benefit to be gained by eliminating the lander dependence on the orbiter makes the attempt worthwhile. To minimize the risk, a capability is provided for using a relay link as well. The additional weight penalty which is incurred is relatively small, and it permits the more desirable direct-link technique to be developed for use in later flights which may not utilize an orbiter. The Venus lander and capsule utilize only a relay link; no high-gain antenna is considered feasible for the lander because of the complete lack of knowledge of the surface conditions.

## 3.3 Payload Capabilities

An integral part of a preliminary design study for an interplanetary mission entails analysis of the various trajectory characteristics associated with each launch opportunity if overall mission payload performance is to be maximized subject to a variety of engineering constraints. A very significant constraint arises due to range-safety limitations. The maximum Earth parking orbit inclination that can be achieved for Atlantic Missile Range (AMR) launches is approximately 34 degrees. On some dates, the minimum-energy interplanetary trajectory requires that the Earth parking orbit from which it departs have an inclination greater than that achievable within range-safety constraints. In such cases, a dogleg or plane change maneuver, with significant payload reductions, is required. Therefore (for this analysis), only those trajectories not requiring such a maneuver were analyzed. For booster vehicles under development, a launch period of several months duration occurs every 19.2 months for Venus and every 25.6 months for Mars. Since the target-planet orbits are neither circular nor coplanar with the ecliptic plane, the injection energy requirements vary as a function of the launch opportunity. However, there is a cyclic recurrence of the energy requirements resulting from a repetition of approximately the same absolute space-fixed geometry every 8 years or 5 synodic periods for Venus and approximately every 15 years or 7 synodic periods for Mars. Within these cycles, the absolute minimum injection energy requirements occur in 1967 and 1975 for Venus. For Mars, the date occurs in 1971 for type I transfer trajectories, and in 1969 for type II transfers. (Type I trajectories are those which traverse a heliocentric angle less than 180 degrees between departure and encounter; type II trajectories exceed the 180-degree angle and hence involve longer flight times.) Examination of the trajectory characteristics for the 1962 and 1970 Venus launch opportunities indicates that with minor differences, the present Venus trajectory information for the 1962-1970 period is for preliminary design purposes, applicable to the 1970-1978 period.

In conducting a mission payload analysis there are two distinct approaches that can be employed to determine initial payload estimates: the first is to determine the maximum payload that can be injected into a given planetocentric orbit; the second is to determine the minimum-energy orbit (lowest altitude) for a fixed scientific payload and propulsion system. Once these initial values have been determined, optimizing techniques such as off-loading propellant if the vehicle is heavy or increasing the departure velocity to achieve a reduced approach velocity if the vehicle is light can be employed. Utilization of minimum departure velocity maximizes the weight injected into the heliocentric transfer orbit. However, this method does not maximize the weight injected into a planetocentric orbit, since this occurs when the hyperbolic approach velocity is a minimum, for a given departure velocity.

Since the minimum approach velocity is, not in general associated with the minimum departure velocity, an analysis was undertaken for each launch opportunity to determine the departure-and arrival-velocity characteristics associated with the daily payload maximization. For the all-orbiter as well as the split-payload orbiter/lander mission, the daily payload calculations indicate a peak in the vicinity of those trajectories for which the sum of the departure and arrival velocities are minimized . In general, this analysis yields payload increases of 25 to 75 pounds over the corresponding payloads associated with the minimum departure energy requirements. Due to departure geometry, energy requirements, and time of flight, one or the other of the two types of transfer trajectories (types I or II) is more desirable than the other. In 1969 and 1975, type II trajectories to Mars can be achieved within the range-safety launch azimuth constraint, and in 1971 and 1973, type I trajectories yield larger payloads in addition to having acceptable departure geometry. For Venus, type I trajectories were selected for the 1968-1969, 1970, and 1975 launch opportunities, and type II for 1972 and 1973. For the mission payload analysis, the pertinent parameters for both Mars and Venus are summarized in table 6. For Mars, it was desired to obtain the maximum payload for an orbit having a periapsis altitude of 1700 km and an apoapsis altitude of 10,000 km, while for Venus the object was to obtain the lowest energy orbit for a given payload. Also, in the case of Venus, the results are shown for two configurations, the first in which three capsules are carried into orbit and subsequently deployed from the orbiter, and second in which a lander separates before encounter and makes a direct entry, as in the case of Mars. The weights used in the trajectory analysis do not in all cases agree with final design weights, since the trajectory studies were completed before final weight numbers were available. The results are generally conservative, however, in that lander, and capsule weights used in the analysis are higher than the final design values.

## TABLE 6

## PERTINENT PARAMETERS FOR PAYLOAD ANALYSIS

| Mars and Venus Direct Entry | Venus Capsule Entry |
|---|---|
| 1. midcourse and time of arrival correction: $\Delta V = 0.125$ km/sec ($3\sigma$) | 1. same |
| 2. lander ejection: a. one 1880-pound vehicle in 1969, 1973, 1975 (Mars) b. two 1880-pound vehicles in 1971 (Mars) c. one 1340-pound vehicle for Venus | 2. not applicable |
| 3. orbiter slowdown: $\Delta V = 0.052$ km/sec | 3. not applicable |
| 4. terminal correction: $\Delta V = 0.030$ km/sec ($3\sigma$) | 4. same |
| 5. orbit establishment: $\Delta V$ as required | 5. same |
| | 6. capsule ejection from orbit: three 200-pound vehicles |
| Propulsion system: Specific impulse = 327 seconds Propellant mass fraction = 0.88 | |

For Mars, the maximum mission payload with the associated propulsion system requirements is presented in table 7 for the four launch opportunities, based on the Saturn I B launch vehicle with the S VI upper stage. In 1969, the orbiter-bus propulsion system weight is approximately 150 to 200 pounds higher than that required for the other opportunities. Therefore, this additional weight may be considered as the penalty associated with propulsion system commonality for the entire Mars mission evolution. However, even with this penalty the payload at the extremity of the best 30-day window varies from a low of 1580 in 1971 (using an orbiter-bus with two landers) to a high of 2180 in 1975. If a propulsion system weight of 260 pounds rather than 403 pounds is selected, the payload capability is increased by 150 to 200 pounds for the launch opportunities between 1971 and 1975 and reduced by 400 pounds for 1969, as shown in table 8. Tables 7 and 8 were computed for the 1975 configuration as an orbiter-bus plus lander. If the 1975 Mars mission utilizes the orbiter as a bus for two landers, the best 30-day window payload weight of the bus (exclusive of propulsion-system weights) is 2210 pounds and increases to 2370 pounds in the middle of the window. In this case, the orbiter-bus does not enter a planetocentric orbit.

For Venus, a nominal elliptic orbit was selected having a periapsis altitude of 1000 km and an apoapsis altitude of 10,000 km to determine the maximum orbiter playload where three 200-pound vehicles (85-pound nonsurvivable capsules) are to be ejected after establishment of the desired planetocentric orbit. The payload weight (which in this case includes the capsules carried into orbit) and propulsion system requirements for this mission are presented in table 9 for the 1970 and 1972 launch opportunities. For a reference orbiter design of 1300 pounds and a three-capsule mission, the best orbit (minimum apoapsis altitude for a 1000-km periapsis altitude) is computed for a 460- and also for a 630-pound dry propulsion system weight. The 460-pound engine is employed to show the effect of using one common propulsion system for both Mars and Venus while the 630-pound engine is employed to show the effect of a common engine for only the Venus opportunities. The penalty is difficult to assess for the 460-pound engine since the vehicle is light at launch, and the optimum window does not occur within the 60-day launch period under investigation; however, it appears to be on the order of several thousand kilometers. The apoapsis altitudes for these dry propulsion system weights and fixed orbiter/capsule weights are summarized in table 10.

For the Venus direct-entry lander in 1973 and 1975, the analysis was performed for both the 460- and 630- pound engines and is presented in table 11. For this mission, apoapsis altitudes below approximately 3000 km are achievable with each engine for both opportunities with only a slight penalty being incurred for the small engine.

TABLE 7

MARTIAN PAYLOAD FOR FIXED ORBIT

| Date | Trajectory Type | Time of Flight (days) | Injected Weight (pounds) | Propellant Weight (pounds) | Dry Propulsion[1] System Weight (pounds) | Payload[2] (pounds) | Payload[3] (pounds) |
|---|---|---|---|---|---|---|---|
| 1/15/69 | II | 272 | 6592 | 2398 | | 1911 | 1911 |
| 1/17/69 | | 272 | 6637 | 2427 | | 1927 | 1927 |
| 1/25/69 | | 274 | 6805 | 2561 | | 1961 | 1961 |
| 1/31/69 | | 274 | 6897 | 2652 | | 1962 | 1962 |
| 2/16/69 | | 280 | 7135 | 2957 | 403 | 1895 | 1895 |
| 4/24/71 | I | 202 | 6598 | 1355 | | 1264 | 1080 |
| 5/10/71 | | 210 | 7327 | 1539 | | 1809 | 1625 |
| 5/18/71 | | 208 | 7426 | 1574 | | 1873 | 1689 |
| 5/24/71 | | 206 | 7502 | 1607 | 219 | 1916 | 1732 |
| 6/1/71 | | 202 | 7377 | 1565 | | 1833 | 1649 |
| 6/9/71 | | 200 | 7179 | 1503 | | 1697 | 1513 |
| 6/25/71 | | 206 | 6438 | 1266 | | 1193 | 1009 |
| 7/6/73 | I | 204 | 5658 | 1779 | | 1749 | 1596 |
| 7/22/73 | | 208 | 6164 | 1830 | 250 | 2204 | 2051 |
| 7/30/73 | | 208 | 6249 | 1791 | | 2328 | 2175 |
| 8/7/73 | | 210 | 6136 | 1678 | | 2328 | 2175 |
| 8/15/73 | | 212 | 5827 | 1513 | | 2184 | 2031 |
| 8/23/73 | | 212 | 5348 | 1315 | | 1903 | 1750 |
| 8/9/75 | II | 330 | 5254 | 1263 | | 1852 | 1708 |
| 8/25/75 | | 340 | 6030 | 1575 | | 2316 | 2172 |
| 9/1/75 | | 346 | 6265 | 1691 | | 2435 | 2291 |
| 9/8/75 | | 350 | 6408 | 1782 | | 2487 | 2343 |
| 9/15/75 | | 354 | 6453 | 1846 | | 2468 | 2324 |
| 9/22/75 | | 360 | 6417 | 1896 | 259 | 2382 | 2238 |
| 10/8/75 | | 370 | 5965 | 1875 | | 1951 | 1807 |

1 Based upon maximum propellant requirement for launch opportunity
2 Based upon maximum dry propulsion system weight for launch opportunity
3 Based upon maximum dry propulsion system weight for the four opportunities
Note: Payload consists of weight in Martian orbit exculsive of dry propulsion system.

TABLE 8

MARTIAN PAYLOAD COMPARISON FOR TWO PROPULSION SYSTEM WEIGHTS

| Launch Date | Trajectory Type | Time of Flight (days) | Dry Propulsion System Weight (pounds) | | | |
|---|---|---|---|---|---|---|
| | | | 403 -pound | | 260 -pound | |
| | | | Injected Weight | Payload | Injected Weight | Payload |
| 1/15/69 | II | 272 | 6592 | 1911 | 5609 | 1564 |
| 1/17/69 | | 272 | 6637 | 1927 | 5598 | 1553 |
| 1/25/69 | | 274 | 6805 | 1961 | 5526 | 1481 |
| 1/31/69 | | 274 | 6897 | 1962 | 5465 | 1420 |
| 2/16/69 | | 280 | 7135 | 1895 | 5245 | 1200 |

Note: Payload consists of weight in Martian orbit exclusive of dry propulsion system.

TABLE 9

MAXIMUM PAYLOAD WEIGHT FOR FIXED VENUSIAN ORBIT

| Date [1] | Trajectory Type | Time of Flight (days) | Injected Weight (pounds) | Propellant Weight (pounds) | Dry Propulsion [2] System Weight (pounds) | Payload [3] (pounds) | Payload [4] (pounds) |
|---|---|---|---|---|---|---|---|
| 7-20-70 | I | 140 | 6836 | 4431 | | 1782 | 1776 |
| 8-5-70 | | 130 | 7181 | 4566 | 623 | 1992 | 1986 |
| 8-13-70 | | 124 | 7282 | 4564 | | 2095 | 2089 |
| 8-19-70 | | 120 | 7285 | 4504 | | 2158 | 2152 |
| 8-25-70 | | 116 | 7217 | 4395 | | 2199 | 2193 |
| 9-4-70 | | 110 | 6965 | 4062 | | 2180 | 2174 |
| 9-18-70 | | 100 | 5868 | 3354 | | 1891 | 1885 |
| 3-1-64 | II | 186 | 6615 | 3868 | | 2118 | 2118 |
| 3-15-64 | | 180 | 7065 | 4240 | | 2196 | 2196 |
| 3-29-64 | | 172 | 7330 | 4489 | | 2212 | 2212 |
| 4-13-64 | | 164 | 7365 | 4611 | 629 | 2125 | 2125 |
| 4-30-64 | | 154 | 6925 | 4461 | | 1835 | 1835 |

1 Trajectory analysis for 1964-1970 period is applicable to 1972-1978 period with at most a 5-day shift in the window.

2 Based upon maximum propellant requirement for launch opportunity

3 Based upon maximum dry propulsion system weight for launch opportunity

4 Based upon maximum dry propulsion system weight for the five opportunities

Note: Payload consists of weight in Venusian orbit exclusive of dry propulsion system plus three 200-pound capsules carried into orbit.

TABLE 10

MINIMUM-ENERGY VENUSIAN ORBIT WITH FIXED PAYLOAD AND PROPULSION SYSTEM WEIGHTS

| Launch Date | Time of Flight (days) | Dry Propulsion System Weight | | | |
|---|---|---|---|---|---|
| | | 460-pound | | 630-pound | |
| | | Injected Weight (pounds) | Apoapsis Altitude (km) | Injected Weight (pounds) | Apoapsis Altitude (km) |
| 7-20-70 | 140 | 6070 | 15365 | 6836 | 12417 |
| 8-5-70 | 130 | 6070 | 13219 | 7150 | 8834 |
| 8-13-70 | 124 | 6070 | 11902 | 7150 | 7940 |
| 8-19-70 | 120 | 6070 | 10816 | 7150 | 7191 |
| 8-25-70 | 116 | 6070 | 9757 | 7150 | 6448 |
| 9-4-70 | 110 | 6070 | 8101 | 6865 | 6378 |
| 9-18-70 | 100 | 5868 | 7611 | 5868 | 10342 |
| 3-1-64 | 186 | 6070 | 7522 | 6615 | 6990 |
| 3-15-64 | 180 | 6070 | 8865 | 7065 | 6149 |
| 3-29-64 | 172 | 6070 | 10140 | 7150 | 6718 |
| 4-13-64 | 164 | 6070 | 11814 | 7150 | 7892 |
| 4-30-64 | 154 | 6070 | 14596 | 6925 | 11176 |

Note: Payload consists of a 1300-pound orbiter exclusive of dry propulsion system, plus three 200-pound capsules carried into orbit.

TABLE 11

MINIMUM VENUSIAN APOAPSIS ALTITUDE
ORBITER/DIRECT LANDER MISSION

| Launch Date | Trajectory Type | Time of Flight (days) | 460 Pound Engine | | 630 Pound Engine | |
|---|---|---|---|---|---|---|
| | | | Injected Weight (pounds) | Apoapsis Altitude (km) | Injected Weight (pounds) | Apoapsis Altitude (km) |
| 10/16/65 | (1973) II | 174 | 6470 | 3844 | 7105 | 3316 |
| 10/31/65 | | 164 | 6470 | 3552 | 7468 | 1965 |
| 11/8/65 | | 158 | 6470 | 3398 | 7550 | 1634 |
| 11/15/65 | | 154 | 6470 | 3219 | 7530 | 1546 |
| 11/22/65 | | 148 | 6470 | 3095 | 7410 | 1751 |
| 11/28/65 | | 142 | 6470 | 3017 | 7194 | 2280 |
| 5/27/67 | (1975) I | 138 | 6470 | 2282 | 6655 | 3315 |
| 6/1/67 | | 138 | 6470 | 1523 | 6900 | 1689 |
| 6/6/67 | | 140 | 6470 | 1004 | 7140 | 1000 [1] |
| 6/11/67 | | 140 | 6470 | 1093 | 7550 | 1000 [1] |
| 6/16/67 | | 133 | 6470 | 1228 | 7490 | 1000 [1] |
| 6/21/67 | | 128 | 6470 | 1416 | 7345 | 1000 [1] |
| 6/26/67 | | 122 | 6470 | 1549 | 7055 | 1300 |
| 7/1/67 | | 117 | 6470 | 1719 | 6670 | 2590 |
| 7/6/67 | | 112 | 6218 | 2790 | 6218 | 4722 |

[1] Propellant remaining after establishment of circular orbit

## 3.4 Reliability

Reliability goals were allocated to the various subsystems to identify those portions of the spacecraft which are potential reliability weak links. In the absence of a numerical spacecraft reliability requirement, a tentative spacecraft goal of 0.833 was established. This goal was determined on the basis of a mission success criterion of 0.50 for the first Mars launch and an assumed 0.60 reliability for the booster. The allocation of subsystem reliability goals was initially accomplished by a qualitative evaluation of such pertinent mission-design factors as relative complexity, mission time, state of the art, and environmental hazard. This reliability apportionment was updated by a quantitative assessment of design reliability parameters associated with the failure contribution of each subsystem. Table 12 presents a comparison between the latter allocated reliability goals and predicted reliability estimates for the various spacecraft subsystems.

The relative reliability improvement effort needed to meed these subsystem reliability goals is also indicated in the table. Preliminary guidelines were prepared for the types of effort required to realize the necessary reliability improvements. The type of effort was dependent upon whether or not the incorportion of redundancy was feasible within a given subsystem. When feasible, the number of redundant elements required to achieve the specified subsystem reliability goal was determined. For those subsystems which do not lend themselves to the use of redundancy, general guidelines were suggested for achieving the reliability goals.

## 3.5 Sterilization

The basic requirements for sterility confidence were specified by NASA such that the probability of landing one or more viable terrestrial microorganisms should be less than $10^{-2}$ for Venus and $10^{-4}$ for Mars. There is some evidence that these requirements may be satisfied by dry heat sterilization of 135°C for 24 hours, provided a low burden of contamination has been maintained throughout manufacturing of all equipments. Dry heat appears to be the most favorable technique of sterilization from the standpoint of reliability (of sterilization) and ease of implementation, provided that heat-sensitive components can be eliminated from the system. Statistical experimental support for the adequacy of this technique is necessary, and a pilot-plant development program is recommended. (Such a program might be part of the Mariner B project.)

A slightly more conservative procedure is recommended herein, in which components and subassemblies are exposed to an additional preliminary sterilization cycle before assembly into the vehicle. This additional cycle might be an

TABLE 12

VOYAGER MARS PREDICTED AND ALLOCATED SUBSYSTEM RELIABILITY VALUES

| Subsystem | Predicted Reliability Estimates | Allocated Reliability Goal | Relative Reliability Improvement Effort (percent) |
|---|---|---|---|
| Orbiter-bus propulsion | 0.901 | 0.973 | 15 |
| Orbiter-bus guidance | 0.850 | 0.958 | 24 |
| Orbiter-bus attitude control | 0.966 | 0.991 | 5 |
| Orbiter-bus power source | 0.954 | 0.988 | 7 |
| Orbiter-bus communications | 0.904 | 0.974 | 15 |
| Orbiter-bus thermal control | 0.995 | 0.999 | <1 |
| Orbiter-bus structure | 0.997 | >.999 | <1 |
| Orbiter-bus separation (booster) | >.999 | >.999 | <1 |
| Lander-bus separation | >.999 | >.999 | <1 |
| Lander structure | 0.997 | >.999 | <1 |
| Lander crushup | 0.999 | >.999 | <1 |
| Lander thermal control | 0.995 | 0.999 | <1 |
| Lander erection | 0.999 | >.999 | <1 |
| Lander power source | 0.924 | 0.979 | 12 |
| Lander heat shield | 0.999 | >.999 | <1 |
| Lander communications (direct) | 0.920 | 0.978 | 12 |
| Lander propulsion | 0.972 | 0.992 | 4 |
| Lander attitude control | 0.998 | 0.999 | <1 |
| Lander spinup | >.999 | >.999 | <1 |
| Lander descent equipment | 0.995 | 0.999 | <1 |
| Lander sterilization | >.999 | >.999 | <1 |
| Lander scientific payload | 1.0 | 1.0 | <1 |
| Lander deployment mechanism | 0.991 | 0.998 | 1 |
| Lander engineering instrumentation | 0.995 | 0.999 | <1 |
| Orbiter scientific payload | 1.0 | 1.0 | <1 |
| Orbiter engineering instrumentation | 0.995 | 0.999 | <1 |
| Lander-orbiter relay communications | 0.975 | 0.993 | 4 |
| Joint expectation of orbiter and lander success | ~0.50 | ~0.83 | |

ultimate requirement and was considered as the reference approach in order to provide a more conservative impact of sterilization on the development plan and cost estimates.

Sterilization of only the lander is recommended; maintaining the orbiter on a biased, noncollision trajectory at all times can satisfy the contamination probabilities. The lander is encapsulated in a rigid envelope before sterilization and remains within this envelope until separation of the lander from the orbiter at planet encounter. The encapsulation approach is described in greater detail elsewhere in this volume.

The complete sterilization technique is summarized below, and one of its most important considerations is the location of acceptance testing within the overall sterilization cycle. The technique recommended is to conduct a major part of the acceptance testing following the preliminary sterilization cycle, but prior to the terminal sterilization. Following terminal sterilization, the lander undergoes an abbreviated cycle of acceptance testing to ensure that the high-temperature environment has not degraded system performance.

An alternate approach would delay all acceptance testing until completion of the sterilization cycle. The risk in this procedure is the exposure of the lander to accidental contamination without detection during the rigors of the testing procedure. Of course, the disadvantage of the recommended procedure is the possibility that performance degradation caused by terminal sterilization may be undetected by an abbreviated final checkout.

The recommended sterilization procedure is as follows:

1. Low burden of contamination (microorganisms and detritus) components and subassemblies are classified according to their abilities to undergo sterilization.

2. The components and subassemblies are cleaned, monitored microbiologically, packaged, and stored.

3. The components and subassemblies are then sterilized by dry heat, steam, ethylene oxide, radiation, chemicals (methanol-formalin, etc.), radiation, and heat according to their sterilization classification. After sterilization, a microbiological monitoring of the components, subassemblies, mockups, or models is performed.

4. The components and subassemblies are then assembled, checked out, and monitored microbiologically. The assembly area is of a white-type area where not more than 100 microorganisms/ft$^2$ will settle out from the air in 1 hour.

5. The assembled lander is then packaged and placed in a combination gas/ dry heat sterilizer (terminal sterilization), and components which are thermo-labile will be removed and sterilized by other techniques, such as gas (ethylene oxide), radiation, chemicals, etc.

6. The packaged lander is moved from the sterilizer by means of a sterile lock system into the sterile assembly area. The thermo-labile components and subassemblies are then reinstalled in the lander. The lander is checked out and monitored microbiologically. Individuals who work in these areas are to be completely enclosed in barrier-type suits and enter only through a sterile maintaining lock.

7. Terminal sterilization times and temperatures are as follows:

a. Mars, 24 hours at 135°C.

b. Venus, 21 hours using Hobby's modification of the Schmidt equation for thermal resistance of microorganisms, or 18 hours using the original Schmidt equation at 135°C.

# 4. SYSTEMS DESIGN

The scientific objectives and mission evolution have been described, and the tradeoff studies which set the guidelines for the system design concepts discussed. This section will provide a brief description of the design features of each vehicle and its subsystems. These vehicles consist of the Mars orbiter-bus, the Mars lander, and the modified lander designed for the revised Mars atmosphere. Next, the Venus orbiter-bus, the Venus capsule, and the Venus lander are described. The details of these designs are contained in volumes IV and V.

## 4.1 Mars Orbiter-Bus

The design of the orbiter-bus for a Mars mission has been influenced greatly by the desire to achieve a flexible spacecraft concept. A serious attempt has been made to provide a design which will have the greatest degree of versatility for one lander, two landers, and different launch windows and look angles. The design also possesses simple convertibility to a Venus mission. The frontispiece is an artist's conception of the spacecraft in its interplanetary trajectory. Figure 7 is an outline of the design layout. (More detailed drawings are shown in volume IV.)

A significant help in the design was the decision to use a rigid (nondeployable) solar array. The large permissible payload diameter and volume of the S-VI stage permitted this approach. In this manner, the large disk acts like a boom for the mounting of communication antennas and the scientific payload gimbal; the angular location (clockangle) of this equipment can be altered to suit the varying look angles that will be required by the different launch windows and mission profiles. Changes of this nature have a relatively minor effect on the configuration and structural design. The rigid solar panels also remove the requirement for deployment in flight.

The orbiter bus is located inside the spacecraft adapter and attached to the adapter at a mounting flange which also serves as a tie for the lander adapter. Although the spacecraft is cradled in the adapter, the design can tolerate the expected tipoff disturbances at spacecraft-booster separation. The solar cell array is arranged in two tiers; one is a 17-foot-outside-diameter and the other is a 9-foot-outside-diameter. This arrangement permits the lander to be located opposite the solar cells so as to allow for a more uniform temperature distribution. The orbiter-bus receives energy for temperature control from the sun, and the lander receives its energy from its radioactive thermoelectric generator. If the lander were on the same side as the solar cells, then the lander would have a good deal of energy available, and the orbiter-bus less, because of the blockage due to the lander. The lander is supported by a conical adapter and attached

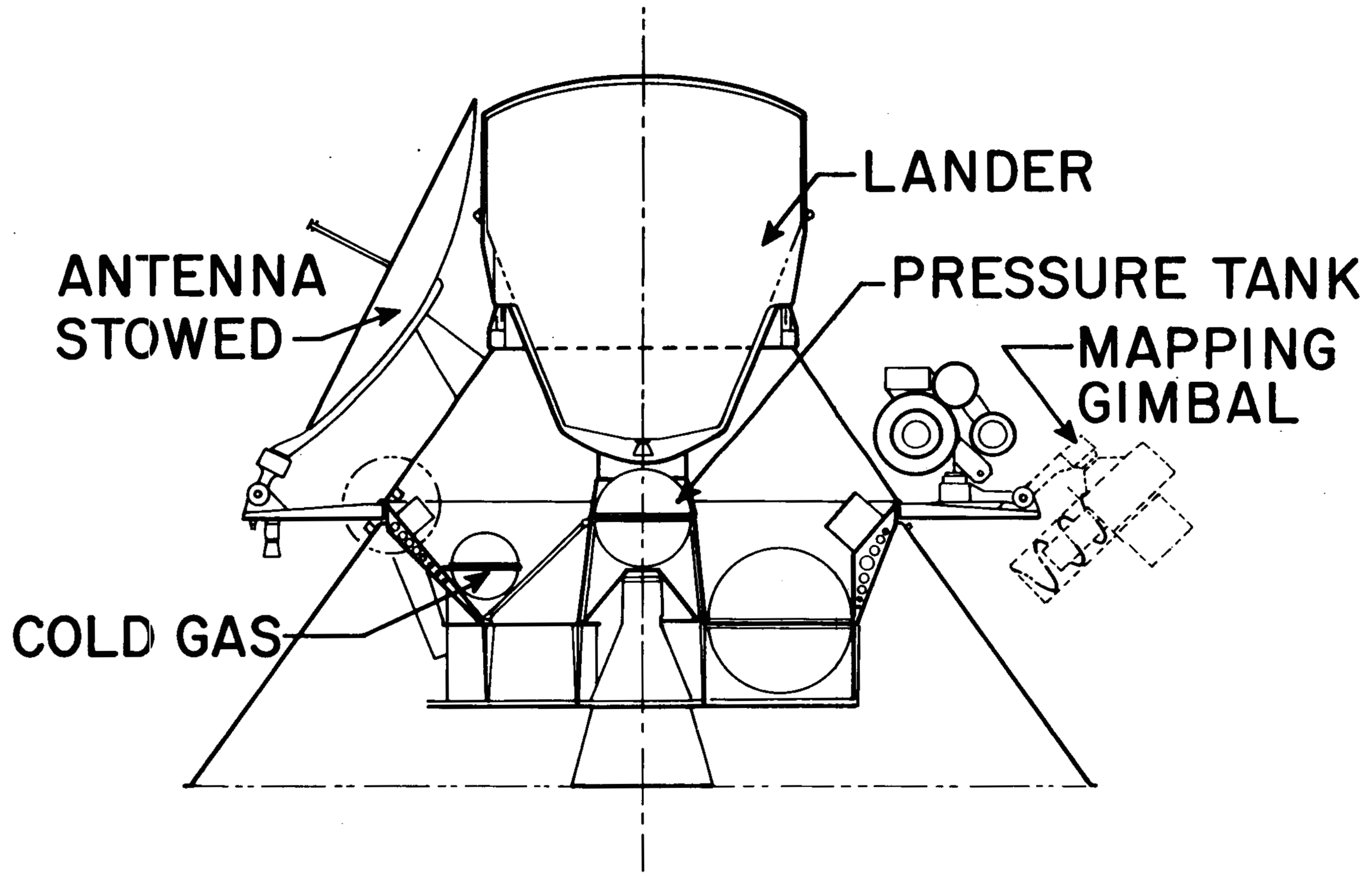



Figure 7 LAYOUT OF THE MARS ORBITER-BUS WITH A SINGLE LANDER
63-8517-2

at three points with a ball-lock separation system. This arrangement permits the lander launch loads to be transmitted directly to the spacecraft adapter without passing through the orbiter-bus. This design feature, plus mounting the orbiter-bus on the bottom so that it will experience tension loads during launch allows for a minimum-weight structural design of the orbiter-bus. This weight saving in orbiter-bus structure is magnified by the saving in propulsion for orbital injection. Furthermore, the modular design of the lander and orbiter permits clean separation and lander changes without alterations in the orbiter.

The separation sequences throughout the mission are as follows: The spacecraft is separated from the booster at or near the main mounting flange by a shaped charge after it is injected into a heliocentric orbit. The lander is separated from orbiter by the actuation of three ball-lock joints. The lander in its sterilization can is pushed away by a separation spring with simultaneous firing of the spin rockets (located on the outside of the can). After the proper separation distance is established, the can is split into four quadrants by a shaped charge and the lander propulsion system is activated. The lander adapter is subsequently jettisoned from the spacecraft to decrease the retropropulsion requirements for planetary capture. Figure 8 shows the orbiter in orbit about Mars.

The general arrangement of the orbiter-bus consists of a conical structure supporting two oxidizer tanks and two fuel tanks in cradled mountings and staggered locations. In this manner, the principal axes will remain unchanged as propellant is depleted. The attitude-control jets and thrust-vector-control rockets are located in line with the tanks, on the principal axes, at the extremities of the larger solar panel. The main thrust chamber and nozzle are centrally located with the ablative portion of the nozzle in the spacecraft interior and the radiation skirt exposed. The electronic and guidance packages are attached to the conical skirt suitably arranged to permit the installation of twin-landers. Antennas, mapping equipment, science payload, and scanners are attached to the rim of the large solar panel as previously discussed. The two-lander configuration is shown in figure 9.

1. Structure (Mars orbiter-bus). The reference conceptual design employs the use of a semimonocoque structure for the major members with longerons and tubular struts to redistribute localized concentrated loads. Use of a monocoque-type rather than truss-type structure is employed because of its efficient capability in load redistribution and its high stiffness-to-weight ratios in reducing vibrational problems. These advantages are in addition to reduced thermal control problems and meteoroid hazards.

A structural analysis was performed for the reference concepts, single lander, and double lander, which indicated a structural weight of 286 and 324 pounds, respectively (excluding propellant tank structure and solar panel structure). This analysis utilized aluminum alloy material. Another analysis was conducted considering the use of magnesium alloy as a possible material. The

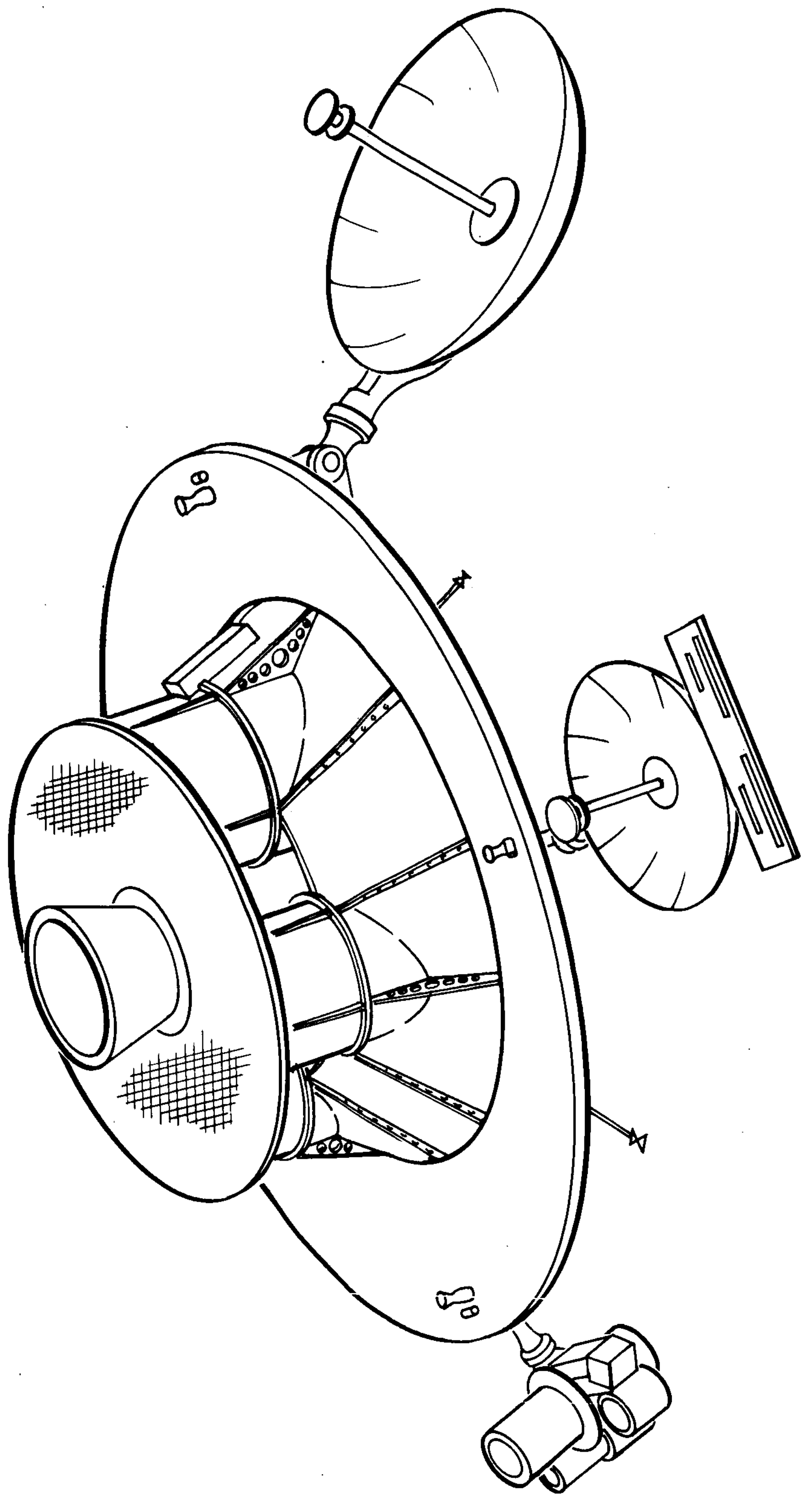


Figure 8 ARTIST'S CONCEPTION OF THE ORBITER-BUS IN ORBIT ABOUT MARS
63-8517-1

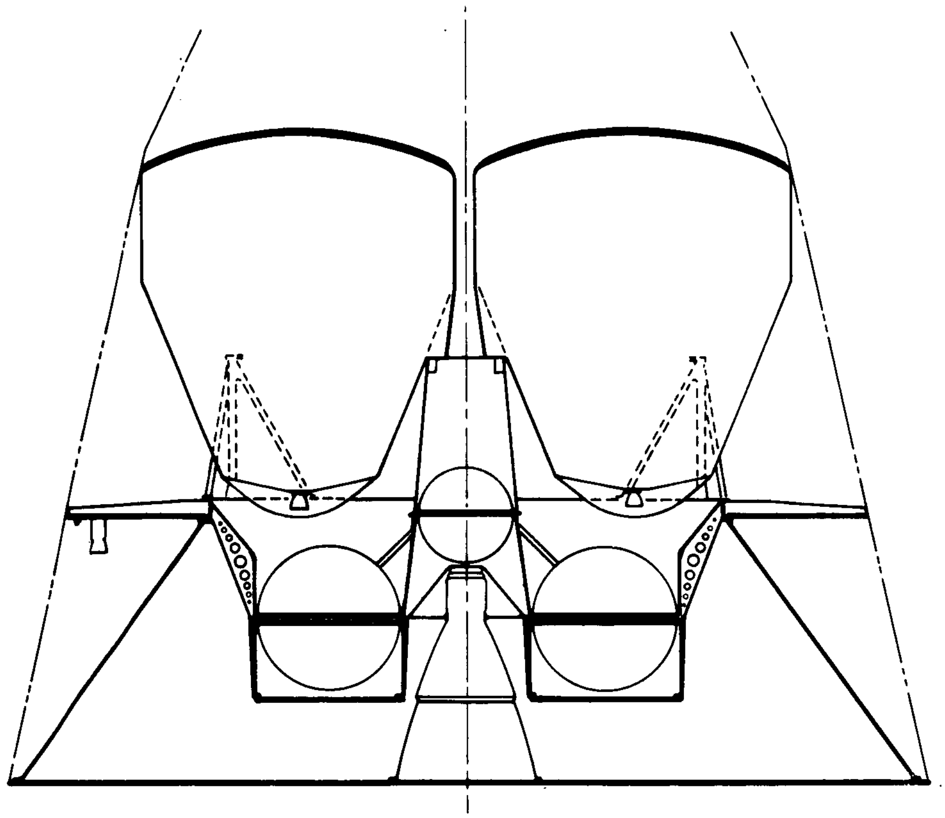


Figure 9 LAYOUT OF THE MARS ORBITER-BUS WITH TWO LANDERS
63-8518

analysis showed a total weight saving for the single and double lander concept of 80 and 88 pounds, respectively. For this reason, magnesium could be selected if further detailed design reveals a need for additional weight reduction.

The critical structural design environment for the orbiter-bus is the 5-g axial acceleration and associated vibrations during launch conditions. However, the cold soak temperature environment during interplanetary travel could prove in the final analysis to be a more critical condition.

The solar panel design is a large annulus around the separation plane of the lander-adapter/spacecraft-adapter interface. These panels are rigidly supported by beams cantelevered from the orbiter-bus main structure. The beams not only support the solar panels but also provide the major load path for the TV camera, antennas, and other equipment mounted at the periphery of the solar panel array. Shock loads of 35 g were considered to be the critical design environment in the structural analysis.

Protection of critical components, such as propellant tanks and instruments, from the hazards presented by meteoroids will be accomplished by using the "Meteor bumper" concept. During much of the interplanetary journey, this double-layer protection is afforded automatically by the structural design. After lander separation and when in orbit, some components will be exposed; these will be protected by a 1 inch layer (0.4 to 0.5 $lb/ft^2$) of sprayable, case-hardening, polyurethane foam.

Protection of critical components such as propellant tanks and instruments from the hazards presented by meteoroids will be accomplished by using the semimonocoque structure as meteor bumper and outer covering of the equipment compartment as the secondary structure. This protection system is far from satisfactory; however, it does represent a seemingly valid protection system. A more detailed and thorough design of the meteroid protection system must await a) a description of the meteoroid environment as to size, number and frequency and b) a model for the penetration of high-velocity particles.

2. Materials (Mars orbiter-bus). The micrometeroid shield concept utilizes an outer skin (the bumper) of some relatively dense material backed up by an absorbing foam (the spacer). It has been shown that the double-layer concept represents a weight saving over the single wall shield. The outer skin must be sufficiently strong to shatter the impacting micrometeroid, and the foam must be able to effectively stop the fragments from seriously damaging the substructure, as well as support the bumper. Avco RAD has conducted tests on a series of outer-skin materials backed by rigid polyurethane foam. The materials were silicone, stainless steel, aluminum, and an epoxy. The shields were impacted with glass pellets at velocities near 20,000 ft/sec. The results indicated that several different combinations of bumpers and spacers can be used as effective shields if the results can be extrapolated to the meteoroid velocity range. Plastic bumper materials have the advantage of low weight, large energy absorption during

particle fragmentation, and ease of application. They are not as stable to a space environment as metallic bumpers, but silicones may be sufficiently stable to provide an effective shield. These studies represent a possible approach being studied at Avco, but it is premature to recommend as a qualified design approach.

The thermal control coatings will be of the same general nature as those used on the Mars lander, described later. Stabilized, pigmented silicone or acrylic coatings, or ceramic coatings, will be utilized wherever possible.

Lubricants that are effective during long-term exposure to the space environment must be selected. The effect of high vacuum is probably the most important consideration, as the lubricant should be well shielded from particle and corpuscular radiation. Presently, molybdenum disulfide appears to be one of the most effective lubricants in the space environment.

3. Telecommunications (Mars orbiter-bus).

a. System requirements. The Mars orbiter-bus has the following telecommunications functions to accomplish.

1) Transmit intransit scientific data to Earth,

2) Provide the transponding function for range and range rate information.

3) Provide the receiving terminal for Earth-to-spacecraft commands.

4) Transmit scientific data (principally TV mapping) during the inorbit phase.

5) Act as a relay between the lander and the DSIF.

Of the above functions, the most important in establishing the system characteristics is that of transmission of mapping data during the in-orbit phase. To meet the mapping requirements, it is necessary to transmit approximately $9 \times 10^7$ bits per orbital period. The principal constraints imposed on the telecommunications system are

1) The aforementioned high data rate

2) The extreme Earth-Mars separation distance

3) Physical limitations on the allowable area for a solar panel

4) Weight penalty necessary to achieve the desired system reliability

5) Limitations on antenna aperture size, based on maintenance of tolerances, pointing accuracy, and shadowing problems.

b. System description. There are two major data-transmission systems onboard the Mars orbiter. They are designated the "in-transit system" and the "orbital system." Both transmit at S-band and utilize PCM/PSK/PM multiplexing and modulation techniques.

1) Orbital system. The orbital system is used to transmit scientific data collected during the 180 days of orbiter life. It is a high-data-rate system (4500 bits/sec) which utilizes an 8-foot parabolic antenna in conjunction with a 120-watt transmitter. This system contains three high-data-capacity tape recorders (approximately $10^8$ bits). Two tape recorders will be utilized to store and play out mapping data on alternate orbits. The third recorder will be used to collect and play out information from the lander.

Because of the long in-transit period followed by a 180-day desired orbit life, this system will be formant during the in-transit phase. A high degree of redundance is incorporated to ensure reliable operation.

2) In-transit system. As the name implies, the in-transit system will be utilized to periodically transmit the scientific data collected in transit. A secondary function is that of providing redundancy for the orbital system to allow the continuation of a degraded mission in the event that a system failure rendered the high-data-rate system inoperative. A 35-watt transmitter used in conjunction with a 4-foot parabolic antenna results in a data rate of approximately 300 bits/sec. At this bit rate, approximately 30 days would be required to relay lander data back to Earth. This agrees well with the time required by the lander direct link to perform the same task.

3) Command system. There will be the capability of command reception through a system of bicone antennas or through the directional S-band antennas. Cross connection at the video level of command receivers will provide redundancy and avoid severe interference nulls in the resulting antenna patterns. The general features of the system are shown in figure 10.

4. Mapping television system (Mars orbiter-bus.

a. Introduction. The purpose of the Voyager mapping TV system is to obtain a set of high-quality picture of sections of the Martian surface. These are eventually to be compiled into a map. The design of the TV system is intended to carry out this objective with high reliability, with a degree of resolution consistent with mission requirements, and with minimum weight, power, and volume.

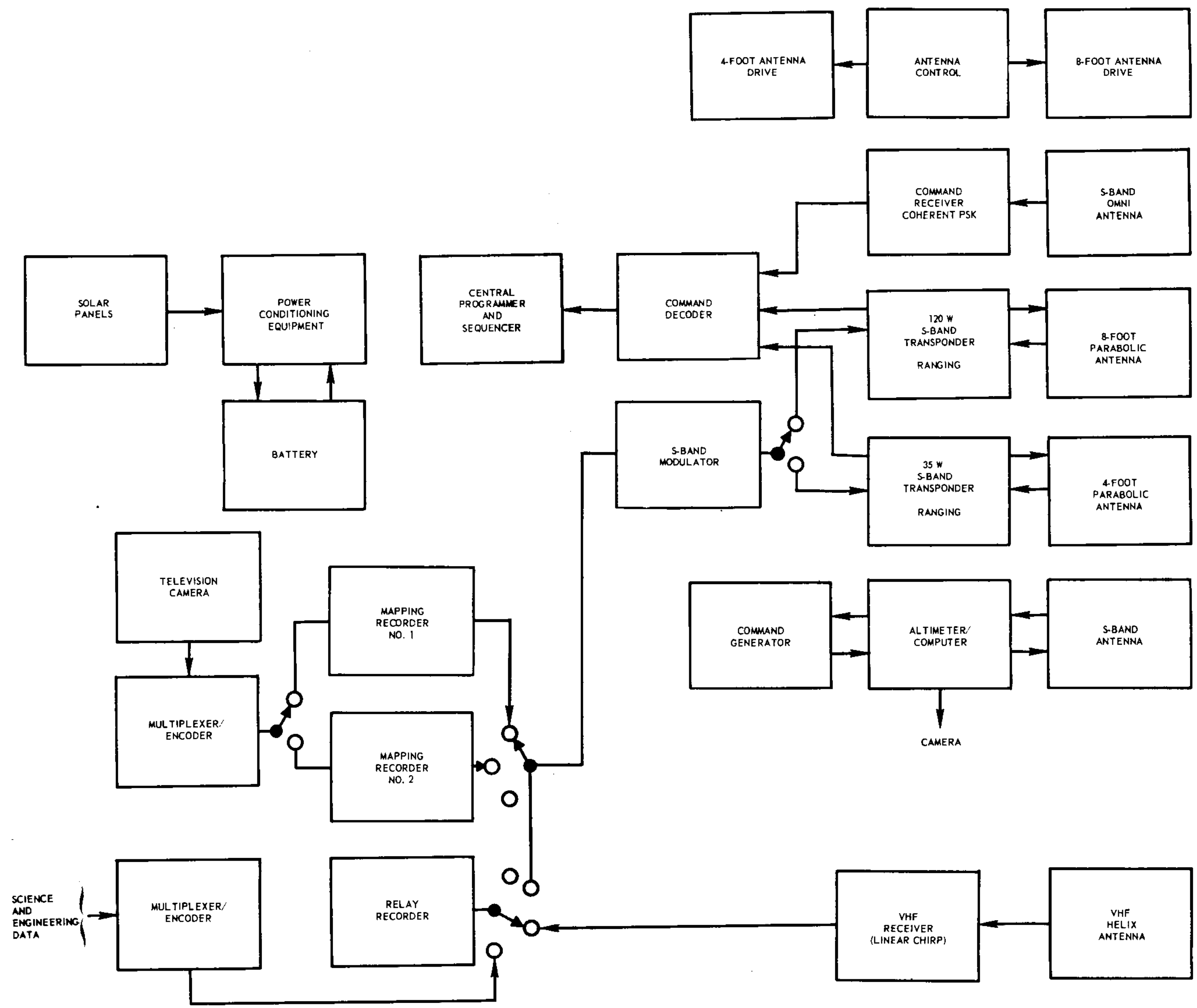



Figure 10 MARS ORBITER-BUS COMMUNICATIONS SYSTEM
BLOCK DIAGRAM
63-9673

b. Basic decisions. The orbital parameters selected involve a suborbital velocity of 2.5 km/sec at periapis. To produce a blurring of no greater than 25 meters requires exposure times no greater than 10 msec. This, combined with the desire to take pictures at the terminator, necessitates the use of the image orthicon camera tube. This choice need not be entirely a liability on other grounds, however. In particular, the high image quality, relative temperature independence (compared to vidicons), and facility for electronic shuttering are employed to good advantage in the design.

A two-camera system with tape buffering is planned. The two cameras with different focal-length objectives permit comparable quality images at all altitudes between 1500 and 10,000 km. In case of failure, one camera can carry out the entire mission at somewhat reduced efficiency or resolution (depending on which camera fails). In addition, as an alternative technique, very high resolution (about 40 meters) images of small areas can be obtained. The tape buffering permits nearly continuous transmission to Earch even though pictures are taken only in daylight. It also permits the use of a technique in which continuous scanning is maintained at a much higher rate than permitted by the transmission link to facilitate certain automatic camera adjustments.

The modulation method is 4-bit/element PCM, employing pseudorandom noise to obtain a continuous-tone scale. Experimental results with this technique have shown very high quality pictures comparable to 6-bit/element PCM. The signal-to-noise ratio is about 48:1 in amplitude or 34 db and the performance in the presence of channel noise is very good. The digital technique permits multiplexing of telemetry data on the same transmitter, and also permits an emergency mode of very slow transmission with no adjustments to the TV system other than change of clock frequency. It also allows a change in the TV system parameters during the development period without circuit design changes.

The picture-taking operation is entirely automatic. As long as the illumination is adequate, pictures will be taken with about 50 percent overlap, under control of the onboard altimeter and clock. Raster dimensions are set, and a camera is selected to produce constant-scale images at all altitudes. No commands are needed unless it is desired to override some of the automatic controls, or a camera or image scale different from that originally planned is desired.

c. Scanning the image orthicon.

1) Geometrical considerations. At a data rate of 4500 bits/sec with an assumed duty cycle of 75 percent, about $9 \times 10^7$ bits are transmitted per orbit, giving the brightness of 22 million picture elements. With a 233-line raster (for reasons to be given), the swath is 84,000 picture elements in length. Since with 50 percent overload each element is transmitted twice, but only half the circumference is covered, these elements divide into 21,000 km, giving the desired 0.25-km resolution and a swath width of approximately 60 km. A wider

swath could be covered, reducing the mission duration; however, this would call either for a higher-resolution camera tube, or for a more complicated camera-pointing routine so that side-by-side as well as tandem pictures could be taken.

The choice of 233 scanning lines, while somewhat arbitrary, is based on the desire that the quality of the final maps be determined by the data rate and not the camera-tube performance. While image orthicons have been reported to have resolved some thousands of lines, the response of even the very high quality 4-1/2-inch studio types such as the 9389a is only some 56 percent at 400 lines in the 4-by-3 format (465 lines in square format) under carefully controlled conditions. In space applications, it would be unreasonable to expect better performance than that. It is well known that image quality is largely a function of the amplitude response at the middle line frequencies, rather than simply the resolution, or cutoff line frequency. Indeed, this is one of the principal reasons for the superior quality of image orthicon pictures compared to vidicon pictures, since the two have comparable resolution limits.

The amplitude response can be improved somewhat by equalization, but this is easily done only in the horizontal direction, and then only at some sacrifice in signal-to-noise ratio. Thus, it seems highly desirable to limit the line density to a value at which the response is not too small at the upper end of the video band. The value selected is 600 lines in square format.

The 233-line raster comes about from the choice of two cameras with lens focal lengths differing by 10,000/1500 or 2.58. To preserve the map scale, and for other reasons, the raster size is made a function of altitude. At periapsis, the raster is full size, but at 3873 km, the raster will be shrunk to the point where the line density is 600 per target inch. Note that even with this quite conservative line density, the quality of the pictures taken at 1500 km will be somewhat better than those at the upper limit of the short focal length lens, and the same for the pictures taken at the extremes of operation of the long lens.

2) Time sequence considerations. Once the decision to use tape buffering of the video data has been made, considerable flexibility is available in the choice of the scanning rate. At one extreme, one might scan at the slowest possible rate, so that recording on the tape would be essentially continuous. For an average rate of 3300 bits/sec, this would be 825 picture elements/sec. On the other hand, very high speed scanning is also possible. The rate chosen is 25,000 picture elements/sec, a rate which is low enough so that low-power sweeps suffice, but high enough so that at perispsis, only about 1 in 10 pictures will be recorded. At apoapsis, where the suborbital valocity is much lower, about 1 frame in 70 will be used. The reason for this procedure is so that automatic correction of exposure and focus can be achieved with time constants several frames in length, thus keeping these adjustments essentially constant during each frame that is recorded, but permitting smooth control, especially during transit of the terminator.

The procedure for selecting frames to be recorded is to integrate a smoothed analog voltage proportional to suborbital velocity. The next frame following the time when a preset level is exceeded will be recorded, and the integrator discharged. This will result, at worst, in a variation of overlap from 45 to 55 percent.

5. Stabilization and control system (Mars orbiter-bus). The stabilization and control system (SCS) is required to null initial body rates after separation of the spacecraft from the launch vehicle, hold the sun-Canopus reference attitude throughout the mission, reorient when required for velocity changes and lander separation, and respond to steering commands during orbit injection.

An analysis of the disturbing torques showed that the total impulse requrements could be met without a great deal of weight difference by either a cold-gas or a hot-gas system. The cold-gas system was selected on the basis of past performance and reliability. However, if the effect of micrometeroid impact is much greater than estimated, or if significant improvements in hot-gas systems occur, this choice would be reexamined.

The primary components used in the SCS are:

a. Sun and Canopus sensors. These sensors are used to provde the attitude reference in the normal spacecraft attitude-hold configuration. They are positioned in the spacecraft to provide control of the roll axis toward the sun and to hold the preselected spacecraft plane in the spacecraft-sun-Canopus plane. These sensors include the following:

1) Sun sensors. It was necessary to use two sensors to accomplish the incompatible requirements of (1) sun acquisition from any random orientation, and (2) very accurate sun line pointing accuracy. To accomplish these requirements, a coarseacquisition sun sensor with 360-degree of view in pitch and yaw was selected for acquiring the sun, and a limit cycle sun sensor with a limited field of view, but high accuracy was used for sun-line hold.

2) Coarse acquisition sun sensor. The coarse acquisition sun sensor consists of four silicon detectors in each spacecraft axis properly mounted on the spacecraft and connected in a bridge. They provide electrical outputs in pitch and yaw indicative of the direction to which the spacecraft must be commanded, to bring the sun within the field of view of the limit cycle sun sensor. The accuracy of this sensor is approximately ± 1 degree.

3) Limit cycle sun sensor. The limit cycle sun sensor is a passive electrooptical device in the pitch and yaw axes which provides an electrical signal indicative of the direction and magnitude of the sun's deviation from a null. The optical configuration employed provides a very sharp null and good scale factor at the expense of a limited field of view. The field of view is ± 5 degrees, and the accuracy is approximately ± 0.01 degree.

4) Canopus star tracker. The Canopus star tracker is an electro-optical device which provides electrical signals indicative of the magnitude and direction of the deviation of the star Canopus from a null. An image dissector photomultiplier tube is used as a detector. The mirror of the optics is gimballed to accommodate the apparent motion of Canopus during the Voyager mission. The requirements placed upon this tracker are primarily the accuracy requirements of the guidance subsystem. The tracker has an accuracy of approximately 20 arc-seconds and a field of view of ± 1.5 degrees (± 18 degrees in pitch with gimballing).

b. Cold gas reaction system. A mass expulsion system is necessary to provide torques on the spacecraft for maneuvering and overcoming disturbance torques during nonthrusting phases of the mission. The system chosen on the basis of weight and reliability considerations is a nitrogen cold-gas jet system. Four jets of 0.03-pound force provide a torque couple for each of three axes, pitch, yaw, and roll rotation in either direction. The system comprises the 12 jets, 2 nitrogen storage tanks, a pressure regulator, pressure transducers, and other necessary plumbing.

c. Gyro/electronics package. The gyro/electronics package includes the following:

1) Three floated single-degree-of-freedom gyros. These three gyros are used as (a) rate sensors in three axes for damping purposes, and (b) attitude sensors in three axes for attitude hold purposes during spacecraft maneuvering and during Martian orbit when either the sun or Canopus is occulted. The GG159 MIG gas bearing gyro was selected for use on the basis of reliability, long life, and accuracy. These gyros feature a ceramic gas bearing motor and a frictionless hydrostatic fluid gimbal suspension.

2) Evaluation electronics. During the extended duration of the Voyager mission, it is possible for unexpected gyro drift rates to accumulate. The evaluation electronics allow an inflight evaluation of these drifts so that they may be compensated for during commanded spacecraft maneuvers. The short-term random drift of a gyro cannot be predicted and compensated by this method, but the overall technique will yield an error input from the gyros of only ± 0.1 degree during a maneuvering period of 1 hour.

A current-pulse generator, pulse rebalance electronics, and a torquer switch bridge are used for gyro evaluation. To evaluate drift, the gyros are caged by these circuits while the spacecraft is attitude stabilized to the sun and Canopus sensor inputs. Any gyro drift during this phase is apparent as torquing pulses, and these are sent to the digital computer unit where they are subsequently used to bias maneuvering commands. By this means, gyro drift is effectively compensated, and very accurate maneuvering can be accomplished even during extended missions.

6. Guidance system (Mars orbiter-bus). The primary mode of guidance throughout the mission is provided by the DSIF, and in fact it is the only mode of guidance until approach to the planet. Midcourse corrections are made by DSIF command. The SCS holds the spacecraft in the proper attitude while the rocket engine is fired; thrust cutoff is determined by the accelerometer aligned with the thrust axis. On-board optical guidance begins near encounter, just before and after lander separation. Position fixes made by measuring planet-star angles and disk angle measurements are used to supplement DSIF tracking information. Injection into planetocentric orbit is accomplished by following a programed turn, using the three-body mounted accelerometers and DCU to provide steering and cutoff signals. While in orbit, determination of the orbital elements is performed by DSIF. The on-board optical system can made simultaneous measurements to provide an independent orbit determination.

The guidance system consists of a computer, star tracker, planet tracker-scanner, horizon scanner, and an accelerometer package. The descriptions of the components are given below.

a. Computer. The computer is a compact and lightweight Honeywell subminiature computer designed for advanced aerospace applications. Its solid-state biax memory consists of 8192 words which are divided into two categories: 1024 24-bit words capable of being altered under program control, and 7168 24-bit words capable of being altered only by external control. The 7168 words provide a maximum memory capacity of 10,752 16-bit instruction words.

The estimated memory requirements are 4000 words for midapproach navigation, 1500 words for orbit determination (assuming use of the same subroutines as required in midapproach navigation), and 2600 words for miscellaneous functions (gyro evaluation, orientations, acquisition phases, mode control, etc).

In addition to satisfying the memory requirements, this computer met all other computational objectives for this program, and a breadboard model has already been built and operated.

b. Auxiliary star tracker. The proposed configuration of the auxiliary star tracker is a gimbal-mounted image-dissector photomultiplier tube and associated electronics designed to track first-magnitude stars and reject less bright stars. The instantaneous field of view is ± 1.5 degrees in each of two orthogonal axes, and the gimballed field of view is ± 40 degrees in one axis, by ± 60 degrees in an orthogonal axis of rotation. Accuracy is ± 20 arc-seconds over the gimbal field of view. The tracker is a modification of the Canopus star tracker, using practically identical electronics and detector, with only slight difference in mechanical configuration. The star tracker is used to obtain a second star-planet angle for use in midapproach and inorbit navigation.

c. Planet tracker-scanner. The sensor element of the proposed planet tracker-scanner configuration is an image orthicon tube in which the planet image is electronically scanned to measure image size, which is a measure of apparent diameter. Scan conversion by fiber optics and variable focal length are used to increase dynamic range. The possibility exists that the television mapping system can be employed for this purpose, thus eliminating a separate planet tracker-scanner.

Outputs of the planet tracker are binary numbers proportional to the polar coordinates $(r,\theta)$ of either the planet limb or terminator. The determination of limb or terminator is made using the guidance computer. In the process, the position of the center and the diameter of the image will be determined. The computer will then generate gimbal drive signals to center the image in the orthicon field of view.

The planet tracker is required to provide planet range and direction during midapproach navigation. This configuration was conceived and proposed to satisfy the midapproach accuracy requirements. No components of this type were found to exist at the present time.

d. Planet horizon scanner. The planet horizon scanner is used during orbit determination to provide a vertical reference. The proposed instrument is being developed by Barnes Engineering. Its accuracy is 0.5 degree within the expected orbit altitudes of Voyager. The concept uses a multielement thermopile detector array in the focal plane of the objective optics with each detector being sequentially sampled to determine planet horizon position.

e. Accelerometers. The accelerometers proposed are the Honeywell GG177 type. These are miniature hinged-pendulum accelerometers which combine high accuracy with compact size and high reliability. They are in production now and were selected for this application because they satisfy the availability, cost, and accuracy objective of the program.

The only area in which some uncertainty as to the time and cost of development exists is the planet tracker-scanner. This type of sensor has not received a great deal of development work to date and the concept suggested for the Voyager mission is a new and novel approach although it embodies techniques and equipment that have been used in other instrument developments.

7. Thermal design (Mars orbiter-bus). Studies have indicated that a passive thermal-control system can be utilized for the Mars orbiter-bus as well as for the Venus orbiter-bus. The tradeoffs between these systems have involved selection of temperature excursions that the equipment can withstand and noting whether a passive thermal-control system can keep the temperature within this allowable excursion. The alternatives to a passive thermal-control system are to allow a narrower temperature excursion, which would then require (a) an active thermal-control system, in which a fluid is pumped, or (b) a hybrid thermal-control system, in which the characteristics of the surface coatings are altered by means of mechanical shutters. The arguments for the system that is superior

must consider (a) the poorer reliability of equipment which operates over a wide temperature range, but which also includes the superior reliability of a static control system, versus (b) the expected superior reliability of equipment which operates over a narrower temperature range, but which includes the poor reliability of dynamic control system. Studies have indicated that allowable equipment temperature excursions are compatible with a passive thermal-control system.

The attitude-controlled orientation of the spacecraft assures a simple and reliable passive thermal design. With the low sun intensity and large surface area, the vehicle will have a tendency to run cold. The solar input variation between Mars and Earth would result in a temperature ratio of 1.23. This corresponds to approximately 100°F difference if the area and surface characteristics remain constant. Actually, the situation is more favorable with the separation mechanism used. The large cone cover on the rear has a higher emissivity ($\epsilon$ = 0.5) than the remaining part ($\epsilon$ = 0.14). Up to time of separation, this cover helps to radiate heat. After the lander has separated, the cover is disposed of and a smaller total area with a low emissivity remains.

The electronic equipment is placed in the midsection region. It is not exposed to direct sunlight, and the relative location of individual packages is of little importance. This gives adequate freedom of movement of each package (e.g., for center-of-gravity adjustments) without disturbing the heat balance. Gyros, the image orthicon, and some of the boom-mounted instruments may require separate heating to maintain proper temperature limits. Because the sun-oriented surfaces are covered with solar cells, the spacecraft is not subject to $\alpha/\epsilon$ degradation effects on the surface coatings.

The radiation from the rocket-propulsion nozzle will not seriously influence the the adjacent equipment. The exit plane is placed sufficiently far beyond the panels so that the expanded plume will not sweep the surfaces with hot exhaust gases. The ablative nozzle has sufficient heat capacity to prevent overheating of adjacent structure within the design burning time.

The large exposed area of 520 $ft^2$ will be sufficient to dispose of the incident solar energy. With the surface emissivity of $\epsilon$ = 0.14 (iridized), except for the rear cone with $\epsilon$ = 0.5, the operating levels of equipment, tanks, etc., will be approximately 80 to 120°F near Earth and 0 to 20°F near Mars.

8. Power sources (Mars orbiter-bus).

a. Selected system. The power source selected for the Mars orbiter is a solar cell array consisting of two flat-plate annular rings with a total area of approximately 200 $ft^2$ with an effective area of 182 $ft^2$. Energy requirements during the shadow phase will be provided by 112 pounds of nickel cadmium batteries. For reasons of economy, it is likely that a concentrator type or sawtooth array would be substituted for the conventional flat plate provided a suitable reflecting material is developed. Present materials, such as Alzac, degrade in a hard vacuum and ultraviolet environment.

The reasons on which selection of a solar cell/NiCad power system was based relate to the fact that both have accumulated a significant history of successful long-life operation in the space environment, and adequate technology already exists to ensure high confidence in the reliable operation of the power system in the orbiter.

The only competitive system, and one which received considerable attention, is a radioisotope power system. This system is not burdened by the weight penalty of batteries and is therefore particularly attractive by virtue of the nearly 135-pound weight advantage for the overall system. However, this assumes the optimum isotope, Curium 244 which will not likely be available in quantity great enough to meet the Voyager demand. In any case, insufficient operating data exists on which to establish this system as preferred.

b. Power requirements and system description. The peak load requirement for the 1500-km by 10,000-km orbit is 770 watts for the condition of periapsis mapping. This power level represents the raw, unregulated power requirement. Solar cells with a nominal efficiency of 9 percent at 25 °C will be used. It is not anticipated that glass covers will be employed, since these are primarily used for thermal reasons and the solar flux is low enough so that reasonably low temperatures can be maintained by simply coating the back side of the panel with a material of suitable emissivity. At Mars aphelion, 4 watts/ft$^2$ is available, while 5.2 watts/ft$^2$ will be available at perihelion. The power system block diagram is shown in figure 11 and the load profile is shown in figure 12. It should be noted that the mapping portion of the mission will be met only every other orbit. This is due to the fact that 200 ft$^2$ is the maximum which may be used for the vehicle configuration selected and all available power is needed to meet the load demand with insufficient excess for battery power recharge; the alternate orbits will be used for recharge purposes.

9. Propulsion (Mars orbiter-bus). Several propulsion system concepts were investigated and subjected to detailed analysis. These systems included gimbal concepts, multiengines (two fixed and one gimballed), and solid propellant engines.

The following description sets forth what appears to be the most promising propulsion system to meet the proposed Voyager mission profile. This system represents a conservative approach and reflects current attainable values. In addition, the propulsion system has been integrated with the spacecraft structure.

During the course of the investigation, three general conclusions were reached:

a. The use of solid-propellant motors for the orbiter and/or lander was not compatible with restart and high-propellant performance requirements for the mission profile and, in addition, the propellant sterilization requirements presented a singular problem for the lander.

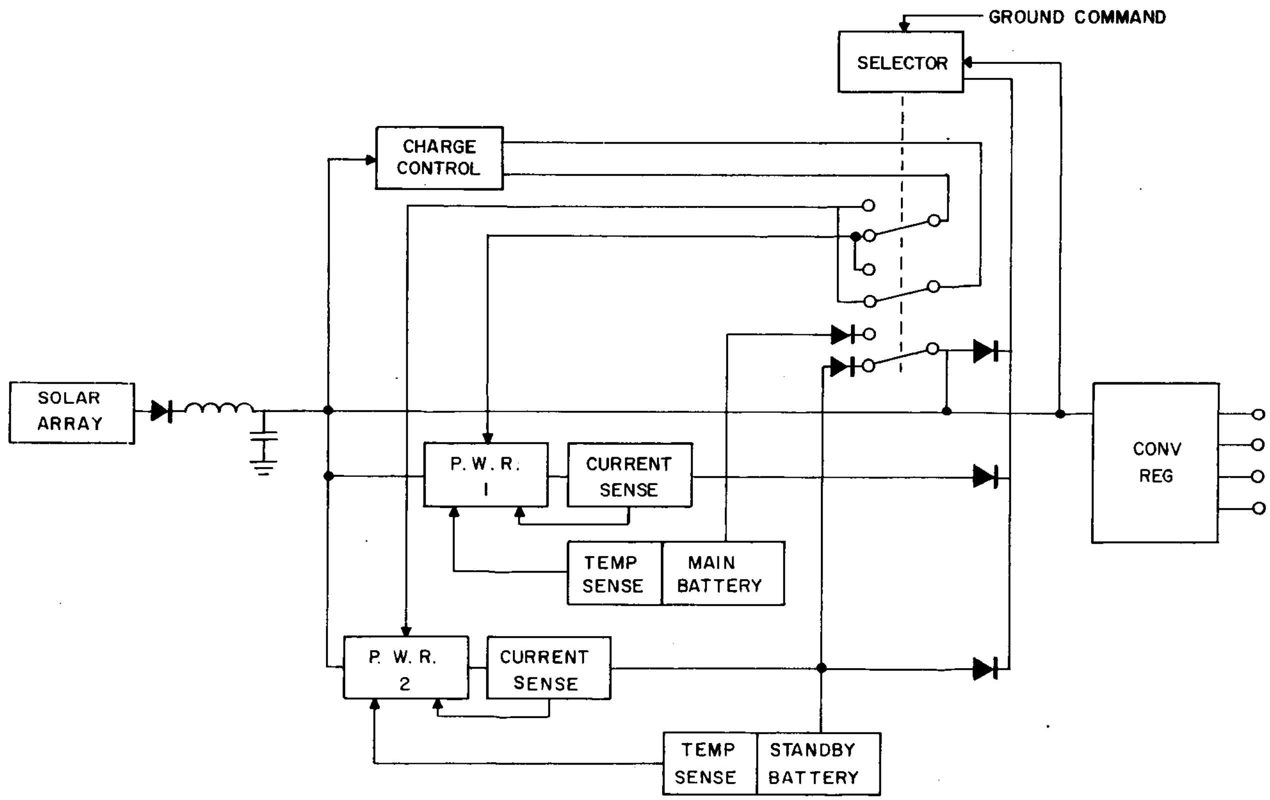



Figure 11 MARS ORBITER-BUS POWER SYSTEM BLOCK DIAGRAM
63-8436

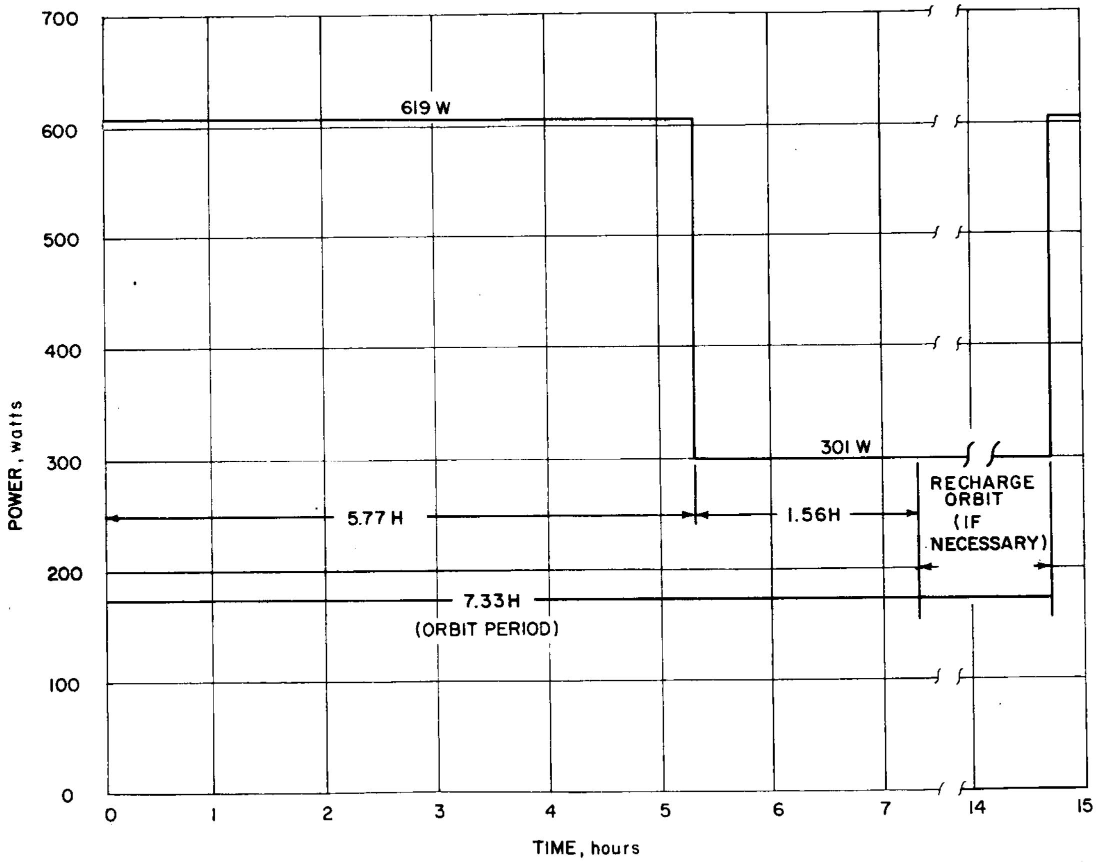



Figure 12 MARS ORBITER-BUS LOAD PROFILE
63-9194

b. Pump-fed liquid-propulsion systems were considered inferior to expulsion-fed systems.

c. Propulsion systems can be designed with growth potential for change over to higher performance propellants.

The orbiter-bus propulsion system, figure 13, is a pressure-fed, storable, hypergolic bipropellant system with a delivered total impulse capability of $1.2 \times 10^6$ lb-sec. The system propellants are mixed oxides of nitrogen (MON) composed of 85 percent nitrogen tetroxide and 15 percent nitric oxide and a eutectic blended fuel (EMHF, eutectic mixed hydrazine fuels) compound of 88 percent monomethylhydrazine and 12 percent hydrazine. The system, table 13, is specifically designed with a variable total impulse capability and provides for complete system sealing during inoperative periods.

The main thrust chamber assembly is rigidly mounted, ablatively cooled, and controlled by solenoid pilot-actuated linked bipropellant valves. A radiation cooled skirt is used on the expansion nozzle between a station 20 inches back of the nozzle exit and the exit plane for increased performance with minimum weight. The chamber is designed to operate at chamber pressures between 125 and 75 psia, as propellant tank pressures are varied between 275 and 130 psia delivering a vacuum thrust between 2500 and 1500 pounds. The thrust variation is obtained by careful control of the chamber inlet pressure, as the injector is a fixed-orifice invariant-geometry assembly.

Pitch and yaw disturbing torques are corrected by four remote positioned thrust-vector-control chambers. These are rigidly mounted, ablatively cooled, and controlled by solenoid-actuated propellant valves. Radiation skirts are used on the expansion nozzles to satisfy minimum-weight requirements. The chambers are designed to operate at pressure variations equal to that of the main chamber, 125 to 75 psia, and deliver vacuum thrust levels of 60 to 36 pounds for restoring torque moments. The chambers are designed for variable-time-interval operation rather than by pulse-frequency modulation. Negligible impulse penalties are incurred due to pitch and yaw corrections by the alignment of the pitch and yaw chamber thrust vectors parallel and unidirectional with the main thrust chamber.

Restoring roll torque is provided by pure couple positioning of four roll-control chambers. The all-ablative chambers may be controlled by time-interval operation of solenoid-actuated propellant valves. Vacuum thrust is varied from 2.5 to 1.5 pounds as a function of inlet pressure.

The propellant is contained in two equal-volume oxidizer tanks and two equal-volume fuel tanks balanced in opposition about the system longitudinal axis. The tanks incorporate metallic positive expulsion diaphragms and are designed to permit variable propellant loading and system prepressurization.

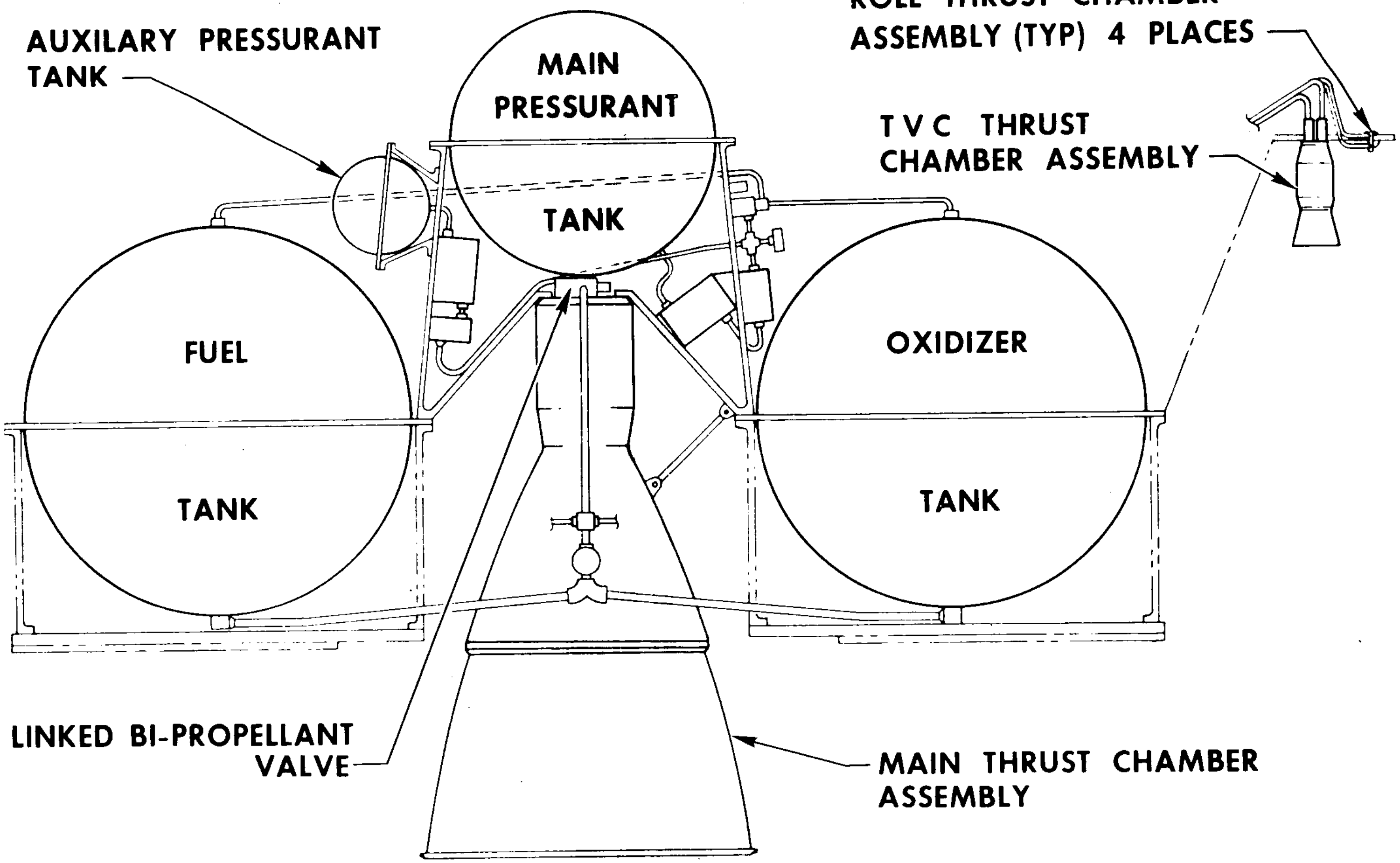



Figure 13 LAYOUT OF THE ORBITER-BUS PROPULSION SYSTEM
309-188-343

The pressurization system is of special design to meet mission requirements. Pressurant gas is provided in the propellant tanks for early mission maneuvers. A small and a large pressurant tank satisfy subsequent expulsion requirements and the propellant tanks remain pressurized for operation to mission completion. The pressurization system is completely sealed from external leakage at all times. This feature will allow use of helium as a pressurant gas with no penalty in reliability.

Lines and controls are provided for proper system functioning. All welded and brazed construction is employed for system assembly to prevent gas or propellant external leakage. An added feature incorporated in the control system is a low-pressure bypass pressurant regulator which permits a significant system weight reduction to be effected by complete utilization of the major portion of the helium pressurant.

System-pressure monitoring is provided for telemetry surveillance during the mission. Temperature monitoring should be provided on the vehicle external to the propulsion system.

TABLE 13

SUMMARY OF SYSTEM PERFORMANCE PARAMETERS

| | Main TCA [1] | Pitch-and-Yaw TCA | Roll TCA |
|---|---|---|---|
| Thrust (pounds) | 2500 - 1500 | 60 - 36 | 2.5 - 1.5 |
| Chamber pressure (psia) | 125 - 75 | 125 - 75 | 125 - 75 |
| Mixture ratio, o/f | 2.15 | 2.15 | 2.15 |
| Characteristic velocity ($c^*$) (ft/sec) | 5351 - 5324 | 5295 - 5268 | 5295 - 5268 |
| Thrust coefficient, $C_F$ | 1.9667 - 1.9737 | 1.9357 - 1.9426 | 1.641 |
| Specific impulse (seconds) | 327.1 - 326.6 | 318.5 - 317.9 | 270 [2] |
| $C^*$ efficiency | 0.950 | 0.940 | 0.94 |
| $C_F$ efficiency | 0.985 | 0.980 | 0.92 |
| Throat area (in.$^2$) | 10.17 | 0.248 | 0.0122 |
| Expansion area ratio | 80 | 60 | 10 |
| Fuel flowrate (lb/sec) | 2.426 - 1.458 | 0.0598 - 0.0359 | 0.00294 - 0.00177 |
| Oxidizer flowrate (lb/sec) | 5.217 - 3.135 | 0.1286 - 0.0773 | 0.00632 - 0.00379 |

[1] Thrust chamber assembly

[2] The roll thrust chambers will operate for brief pulses. The pulse duration will approximate 20 msec, with a resultant pulse efficiency of 0.80. The resultant specific impulse would be 216 seconds.

10. Scientific instrumentation (Mars orbiter-bus). The scientific instrumentation carried aboard the orbiter-bus is designed to accomplish two goals: a) to obtain interplanetary data during flight in addition to that previously obtained by Mariner B flights, and b) to obtain planetary data while the vehicle is orbiting.

The prime objective of the orbiting vehicle is to make maps of the planet. This will be accomplished in several ways: visually as discussed earlier under the mapping television system, by temperature gradients using an infrared radiometer , and magnetically with a magnetometer. The location of radioactive belts will be determined by various flux and particle detectors. Infrared instruments will operate continuously so that mapping of the planet surface will not be confined to the sunlit side of the planet. Therefore, coverage of the planet surface will be obtained in a short period of time.

Information about the atmosphere of the planet will be derived from measurements made by the bi-static radar experiment and the infrared spectrometer.

Scientific instrumentation on the orbiter-bus is summarized as follows:

a. Television for surface mapping

b. Ion chamber to measure the total ionizing radiation (cosmic rays)

c. Particle flux detector to monitor the energetic particle and photon radiation

d. Cosmic dust detector, to measure the flux of cosmic dust particles as a function of direction and distance from the sun

e. Micrometeoroid detector, to directly measure the velocity and cumulative mass distribution of the cosmic dust in the zodiacal cloud and in the vicinity of the planets

f. Bistatic radar, to obtain information on the absolute electron density in interplanetary space

g. Magnetometer, to measure magnetic field intensity

h. Infrared radiometer, for surface mapping

i. Infrared spectrometer, for atmospheric and surface measurements.

## 4.2 Mars Lander

The primary aim in the design of the Mars lander has been to provide a vehicle consistent in size with the Voyager capability which can perform a broad variety of scientific measurements in the atmosphere and on the surface. It should be reliable, have a long lifetime, and yet be of simple design. Furthermore, it should be suitable for landing on a terrain surface of unknown characteristics. While it is expected that early landers will operate in conjunction with an orbiting spacecraft, the lander design should possess the growth capability which would permit it ultimately to be independent of successful functioning of an orbiter. These requirements are to some extent contradictory, and a variety of lander designs were studied to arrive at a suitable compromise. The basic ingredient in all configurations was the scientific package, and parametric studies were conducted to determine the weight of the lander as a function of the number of bits of data collected and transmitted to Earth. Two other major factors in the evaluation of the designs were the lander power supply and the mode of communication. The use of a radioisotope thermoelectric generator (RTG) provided the long lifetime desired for the scientific mission but presented problems in the thermal design of the lander. The enormous weight penalty of batteries alone plus the problem of sterilization ruled out that choice for a long life mission, and the RTG was selected for the reference design. The desire for a lander with growth capability almost forces the choice to a design which can either land erect or be erected after impact. The uncertain knowledge of the terrain favored the latter approach. The reerectable lander makes feasible the use of a high-gain antenna, an antenna which can be directed to Earth and provide a direct communication link. The addition of a relay-communication capability can be provided for small additional cost in weight, and was judged to be essential for the early flights, until the practicality of the reerectable design is demonstrated. These characteristics tended to set a minimum size on the lander; the maximum size was set by the amount of instrumentation which could provide useful information from a stationary observatory. The external configuration of the lander was dictated by the decision to employ direct entry at entry angles varying between 20 and 90 degrees below the horizontal, by the need to provide sufficient deceleration at high altitude to permit atmospheric measurements, and by the necessity of deploying a parachute to ensure a relatively soft impact. Further details on all these points and others are contained in the following pages. All the material presented is based on the Schilling model atmosphere; in a later section are discussed the changes in Mars lander design which result from the more recent JPL model.

1. <u>Mechanical design (Mars lander)</u>. Figure 14 shows a layout of the reference design. Detail drawings are contained in the lander design volume, volume V.

While attached to the orbiter, the lander is encased in a sterilization can which also acts as a micrometeoroid shield. At lander-orbiter separation, the

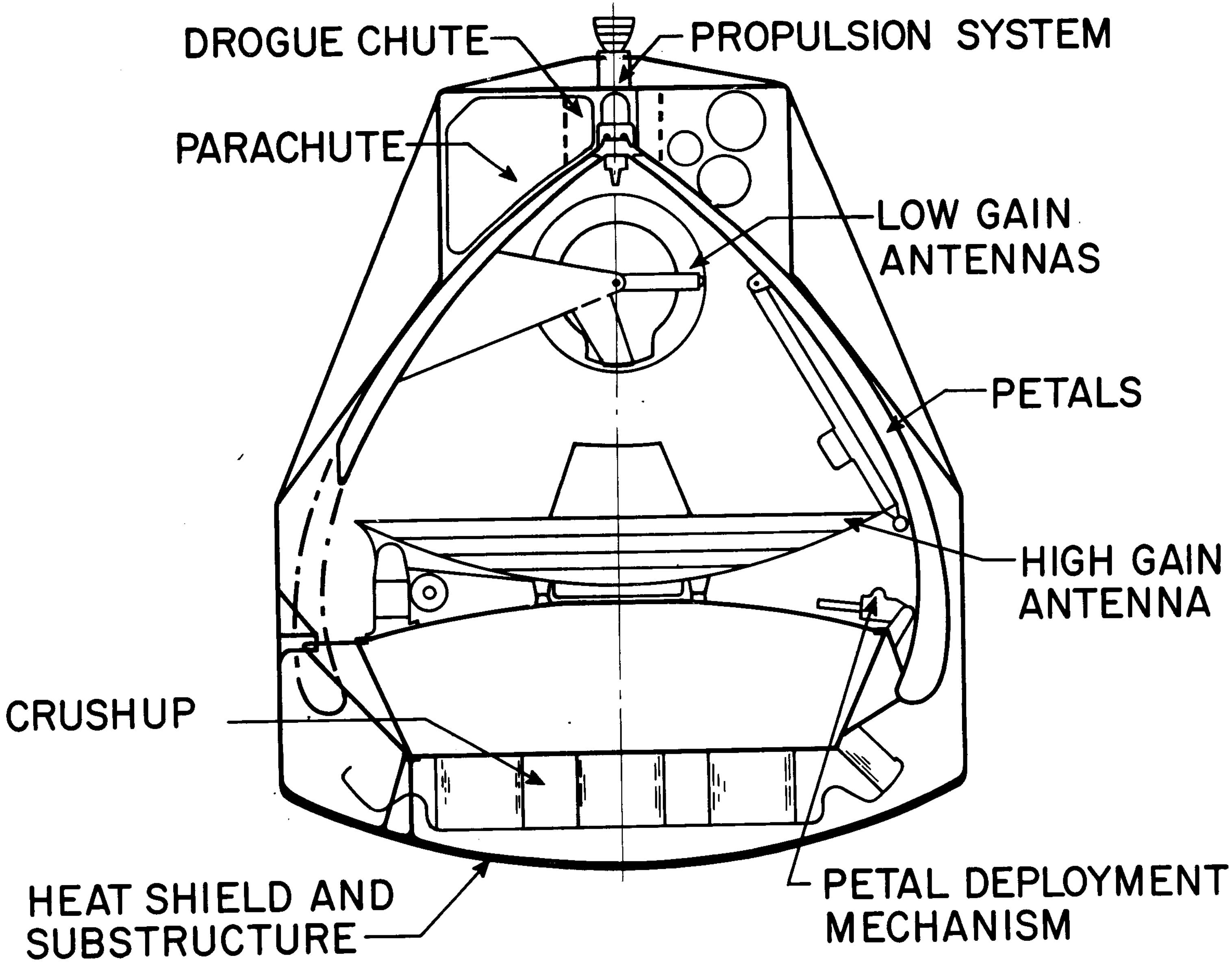



Figure 14 LAYOUT OF THE MARS LANDER
63-8516

lander is spun up by two small solid-propellant rockets, located on the sterilization can. The can is then jettisioned and the bipropellant liquid propulsion system places the lander on a planetary impact course. As the vehicle enters the Martian atmosphere, it is decelerated aerodynamically until Mach 2.5 is reached at which time a drogue chute is deployed through a mortar-type ejection system, as shown in figure 15. At Mach 0.5 (15,000 feet minimum altitude), the heat shield is ejected by means of a shaped charge which severs the rear portion of the reentry vehicle from the forward portion. The drogue chute force then pulls the rear reentry vehicle portion backward, and in so doing, deploys the main parachute. During the main parachute phase, TV pictures and atmospheric data are recorded and the data played back to the orbiter relay via the hemiomnidirectional slot antenna located in the gimbal at the top of the vehicle. The vehicle impact velocity is 40 ft/sec.

At impact, the vehicle is protected by aluminum spiral grid crushup pads on the bottom, and the sides and top of the lander are encased in an acorn-shaped shell which is covered with crushup material. No attempt is made to keep the vehicle erect, and it is allowed to roll and tumble until a final stationary equilibrium position is reached. The external shape of the shell and c.g. position are so designed that the vehicle will right itself automatically on a reasonably smooth and level surface. Rough terrain may prevent self-erection. The shell is formed by six petals which are deployed by means of a motor and gear train for each petal. The opening of these petals erects the lander and raises it off the ground slightly for thermal control requirements. As the petals are opened, the instrument boom, seismograph, the gimballed antenna are automatically deployed. Figure 16 shows the lander before and after deployment.

If the petals are unable to deploy because of some severe terrain feature, means are nevertheless provided for collection of surface soil samples. Data transmission may then be made through the internal gimballed hemiomnidirectional antenna via orbiter relay. The gimbal is self-leveling and dielectric material is used for petal fabrication to facilitate transmission if the petals fail to deploy.

After the vehicle is erected, a 5-foot-diameter, high-gain parabolic antenna for DSIF communication is oriented towards Earth by means of a navigation computer. The programed scientific instrumentation sequences for atmospheric and surface measurements are conducted and the data transmitted to Earth via the high-gain antenna and also via the orbiter relay link.

2. Aerodynamics (Mars lander).

a. Atmospheric model. The Mars-entry study has utilized an atmospheric model as specified by Schilling's model II in reference 1. The limits of the model are used for design of the vehicle and are meant to account for atmospheric uncertainties.

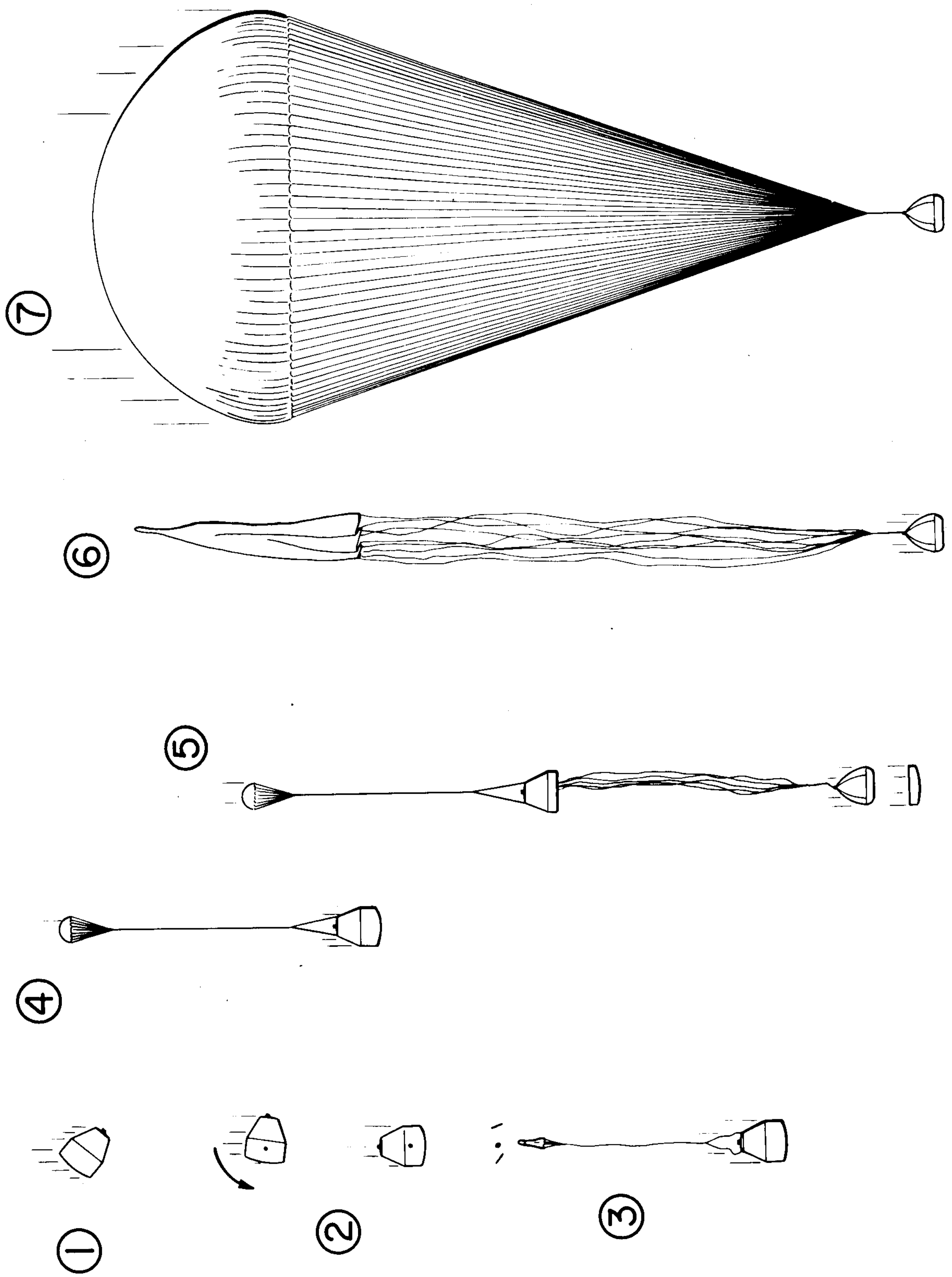



Figure 15 MARS DESCENT SEQUENCE
63-8515

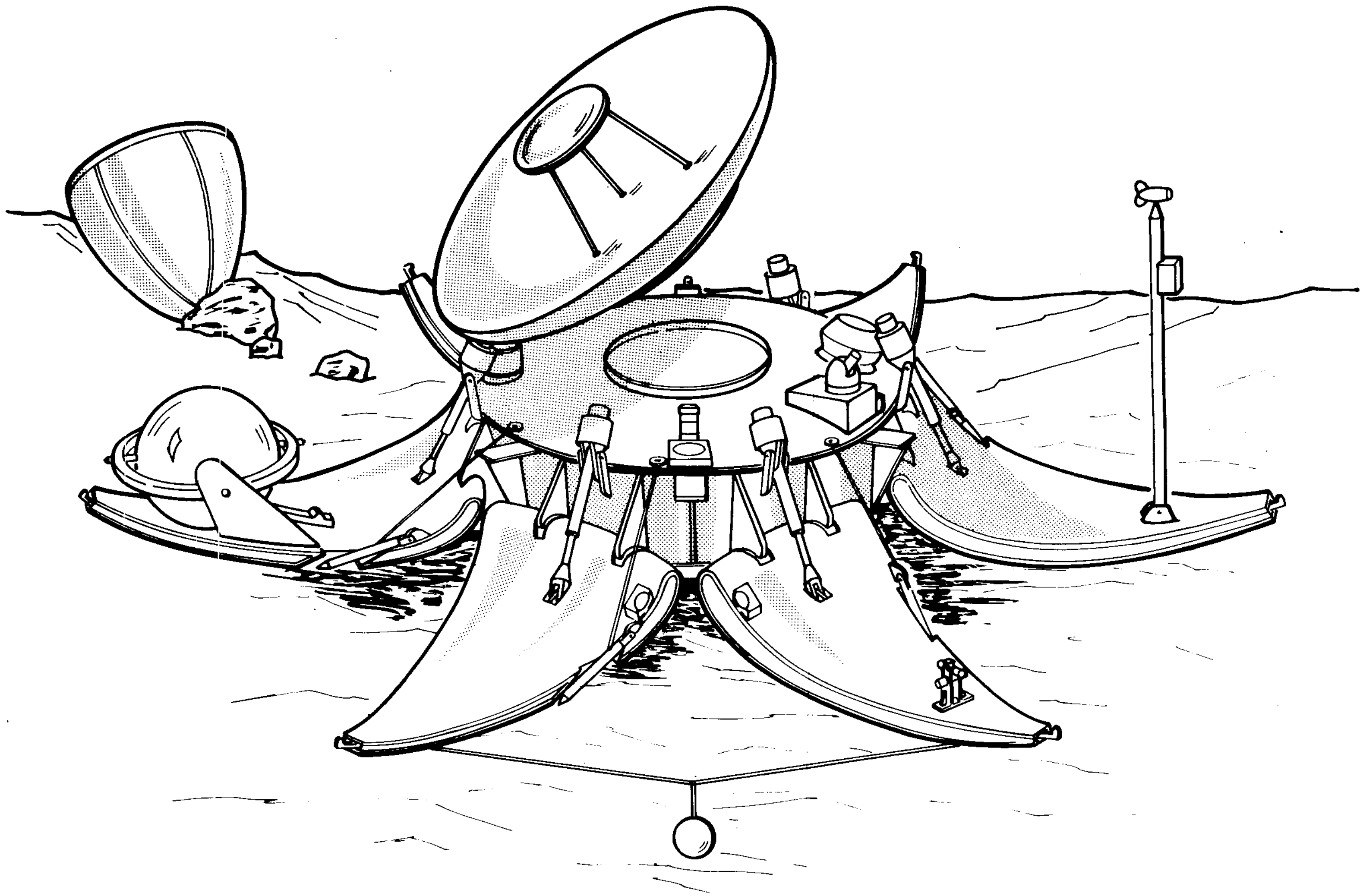


Figure 16 ARTIST'S CONCEPTION OF THE MARS LANDER BEFORE AND AFTER PETAL DEPLOYMENT
63-7996B

b. Vehicle configuration. The configuration selected for entry into the Mars atmosphere is the Avco E-5 shape which has been designated the Voyager V-2 vehicle. The blunt configuration is necessary to provide adequate deceleration for atmospheric measurements and a successful landing. The reference design ballistic coefficient ($M/C_DA$) is 0.9 and was determined by a tradeoff between descent system requirements and available payload weights. The aerodynamic characteristics utilized for the V-2 shape are from wind tunnel, shock tube, and ballistic range test programs.

c. Vehicle dynamics. Six-degree-of-freedom trajectories have been generated to determine dynamic histories for the V-2 vehicle as a function of entry conditions, model atmosphere, and vehicle parameters. Angle-of-attack effects on heating and loads have been determined and are included in vehicle weight calculations.

The center of gravity location is such that the vehicle has a single trim point at zero angle of attack and would be satisfactory for a tumbling entry. Trajectory results indicate a possible maximum angle of attack of 30 degrees in the region of peak heating and peak loads. A drogue parachute, which is deployed at $M = 2.5$ to provide sufficient descent time, also improves dynamic characteristics in the low supersonic range.

d. Entry heating and loads. Convective and radiative heating pulses have been obtained as a function of entry conditions and atmospheric model. The design conditions for the heat shield yield radiative heating which is relatively small compared to the convective heating. The blunt nose under these heating conditions does not compromise the design and, in fact, is beneficial. The maximum axial deceleration for entry is approximately 180 Earth g's.

3. Thermal design (Mars lander). The extended mission life made possible by the use of an RTG requires special consideration inthe thermal design of the lander for its transit journey as well as for operation on the planet.

During transit, the lander with its combined sterilized can and meteoroid protection is placed in the shade. This allows good radiation to space which is maintained (except for relatively short periods during midcourse corrections) up to the time of separation at a range of 1,000,000 km from Mars. After shedding the can, the lander is subjected to a sun input the intensity of which is very much lower than near Earth.

On the planet the heat from the RTG is advantageous in keeping the instrument box warm in the extremely cold environments. The main design problem thus occurs during transit, namely, the requirement to distribute the heat from the RTG over the surface area of the lander. This is accomplished by using a ventilating fan blowing over the RTG as well as conduction along the top of the instrument box. The design is based on pure radiation so that loss of pressure and/or fan does not cause catastrophic failure.

An added feature is the mechanical design arrangement which permits extended petal movement to lift the bottom of the instrument box above the ground, thus preventing close contact with a cold surface.

The temperature range of the instruments will be from -10 to 130°F, with the exception of the battery which will be kept at 50°F minimum by means of thermostatically controlled electric heating pads.

4. Structures (Mars lander). The structural configuration of the lander can be categorized into four major classifications: (a) the external structure which is the aerodynamic load bearing portion of the thermal protection system, (b) the internal structure which serves as the load carrying members for the payload, (c) the petal structure which provides protection if toppling occurs after impact (also used for re-erection), and (d) the impact attenuation structure which is used to limit loads transmitted to the internal structure during impact.

The structural weights have been generated from design criteria formulated with regard to the following definitions. Instability modes of failure are referenced to loads which are 1.25 times the operational limit load experienced under the specified environmental conditions. Modes of failure which are governed by the yield strength are referenced to limit loads.

The load-bearing structure of the entry vehicle is of sandwich construction which is designed by two modes of failure: (a) instability of orthotropic shell and yielding due to planetary entry aerodynamic, and (b) inertial loads. A number of external shapes were investigated along with a study of different materials and types of construction. The structural weight of the entry vehicle becomes extremely significant or even prohibitive for low values of $M/C_DA$.

The internal structure is of stiffened monocoque construction. The primary mode of failure is yielding due to the axial and lateral inertial loads during entry and impact. The structure is stiffened aluminum sheet.

The petal structure is constructed with an aluminum box beam central rib. This is the load-carrying member for the main parachute load and load imposed on the lander during re-erection. The top portion of the petal structure is fiberglass which is used because of the necessity of communicating through the structure. The remainder of the petal is an aluminum sandwich construction that surrounds the central rib. This part of the petal protects the payload if the landed package does not remain erect at impact.

The impact attenuation system consists of pads of spiral-wound, corrugated, aluminum foil, which dissipate the impact energy in crushing. A fiberglass, load-distribution plate serves to protect the payload from small-scale irregularities in the topography of the planetary surface. The impact attenuation system is designed to limit the impact loads imposed on the internal structure and payload to less than those experienced during entry.

The sterilization can is not part of the lander structure; however, it does serve as protection for the lander during interplanetary transit. The can is constructed as a sandwich with a foamed aluminum core. From the standpoint of the meteoroid hazard, this sandwich construction is superior to a monocoque construction in that it provides added protection for the ablative heat shield, and also reduces the possibility of a complete puncture of the can, which would compromise the sterility of the lander.

5. Materials (Mars lander). The major material consideration on the Mars lander is the heat shield. The reference material upon which the thermal design was based is the Apollo ablator. This material is a random mixture of silica fibers, epoxy resin, and phenolic microballoons pressed into a fiberglass honeycomb, giving a continuous, one-piece, heat shield surface over the structure. The outstanding features of this material are its low density and thermal conductivity, coupled with excellent ablative behavior under both convective and radiant heating conditions. Table 14 summarizes some typical physical properties of this material. Figure 17 illustrates the char formed when exposed to convective heating at low heat fluxes. The char layer contains pyrolytic graphite, which increases the char strength, and consequently, its resistance to erosion. A stable char layer is also formed under pure radiant heating conditions, at low heat fluxes (less than 500 Btu/ft$^2$-sec). Instrumented tests at higher radiant fluxes have not been conducted. If the heat flux and enthalpy are too low to effect appreciable ablation, then the low thermal conductivity and thermal stability make this material an excellent pure insulator. Tests have shown negligible change in thermal and optical properties after exposure of more than 300 hours to simulated solar irradiation at 0.14 w/cm$^2$ under a vacuum of less than $10^{-5}$ torr. Exposure of the material to 300°F in argon or vacuum for 4 days resulted in negligible weight loss or change in tensile strength.

Another material of concern is the thermal control coating. Many thermal control coatings have been developed in the past few years, and are available with $\alpha/\epsilon$ ratios of less than 0.2 to near 14. Organic coatings are desirable from the viewpoint of ease of application, and variability of the $\alpha/\epsilon$ ratio by changing fillers and/or resin binder. Organic coatings suffer from the adverse effect of space exposure, especially ultraviolet light, which tends to increase absorptivity. The most stable organic coatings are filled silicones and acrylics. With variation of these materials, $\alpha/\epsilon$ ratios of 0.2 to 1.0 can be readily obtained. The addition of ultraviolet absorbers increases coating stability. Ceramic-type coatings, such as the silicates, provide stable coatings to a space environment, while metallic surfaces, such as silver, gold, and platinum platings, are the most stable. Micrometeroid impact can change the absorptivity and emissivity values because of increased surface roughness. Recent studies indicate that for gold, platinum and aluminum $\alpha/\epsilon$ ratios decrease after simulated micrometeroid impact. The individual $\alpha$ and $\epsilon$ values increase, but the $\epsilon$ increases faster than the $\alpha$. The $\alpha/\epsilon$ ratio of stainless steel 304 is unaffected by particle impact.

TABLE 14

MARS LANDER MATERIALS PROPERTIES

| | | |
|---|---|---|
| Density ($gm/cm^3$) | 0.55 | |
| Thermal conductivity at 250°F ($\frac{Btu}{hr\ ft\ °F}$) | 0.065 | |
| Cp (Btu/lb-°F) | 0.37 | |
| Coefficient of thermal expansion (°F) | 17-30 x $10^{-6}$ | |
| Ultimate strength (psi) vs. temp. | Parallel to Honeycomb | Perpendicular to Honeycomb |
| -100°F | 1410 | 950 |
| 78°F | 1250 | 860 |
| +350°F | 200 | 130 |
| Percent Total Strain vs. Temp. | | |
| -100°F | 0.74 | 0.58 |
| 78°F | 1.02 | 0.86 |
| +350°F | 0.7 | 0.65 |
| Modulus ($Ex10^{-6}$ psi) vs. Temp. | | |
| -100°F | 0.21 | 0.18 |
| 78°F | 0.17 | 0.13 |
| +350°F | 0.04 | 0.025 |

Figure 17 APOLLO ABLATOR AFTER TESTING IN OVERS ARC FACILITY AT TWO LOW HEAT FLUXES (5026-39 3/8 HC FROM GUNNED PANEL)
P 9664B

| | | |
|---|---|---|
| AP 1341-A2 | AP 1341-A2 | AP 1341-A3 |
| H = 20,000 Btu/lb | H = 10,000 Btu/lb | H = 18,400 to 9400 Btu/lb |
| $q$ = 270 Btu/ft$^2$-sec | $\dot{q}$ = 36 Btu/ft$^2$-sec | $\dot{q}$ = 270 to 36 Btu/ft$^2$-sec |
| t = 360 seconds | t = 120 seconds | t = 60 -(36) -300 seconds |

The reference material for the parachute is HT-1, a newly developed aromatic polyamide. This material retains more than 80 percent of its strength after 24 hours at 400° F, while nylon loses all its strength at 370° F. The sterilization cycle of 135° C for 24 hours will not affect the properties significantly, and the material can be readily made into fabrics similar to nylon parachute cloth.

6. Telecommunications (Mars lander).

a. General system requirements. The function of the Mars lander is to perform a number of scientific experiments and to transmit the acquired data to the DSIF. Because of the nature of the ensemble of experiments, the initial 48 hours of lander life is of greatest importance. The quantity of data taken during this period exceeds that taken during the remainder of the lander's 180 days of desired life. In addition to the postlanding data requirements, it is extremely desirable to obtain information during lander entry and parachute descent. These data are of great importance for obtaining some scientific return in the event of postlanding failure. The short duration involved in the entry-descent phase, coupled with the high data collection requirements, strongly dictates a relay requirement. As will be shown in a later section, incorporation of this capability does not create a prohibitive weight penalty.

b. System description. The lander has two high data rate transmission systems: an S-band direct link system and a VHF relay system. Although analysis of the problem indicates that a relay link would provide a minimum communication system weight for a given scientific mission, uncertainty associated with the ability to achieve an orbit, coupled with the long range desire to utilize multiple landers without orbiter, made it advisable to incorporate an S-band direct transmission system as well as a relay system in the lander.

c. Relay link. If the assignment of a primary communications system were to be made, this would be the relay link.

1) Frequency selection. To preclude the necessity of controlled spatial acquisition between the orbiter and the lander, it was decided to utilize relatively broad beamwidths at each link terminal. For the lander, a vertically orientated antenna with a cardioid radiation pattern with a beamwidth of approximately 140 degrees was selected. For the orbiter, a planetocentric orientation on a 65-degree beamwidth antenna was selected. With antenna characteristics determined by geometric considerations, the communications system performance is a function of the inverse square of transmission frequency. This dictated the use of a relatively low frequency. The VHF band was selected as the lower limit for antenna-size considerations.

2) Modulation technique. The choice of modulation technique is a compromise between the low energy per bit achievable with phase-coherent systems and the insensitivity to multipath problems achievable with certain

noncoherent systems. Because of the gross uncertainty associated with the terrain likely to be encountered, it was decided to utilize a noncoherent system. A short-duration-pulse modulation was considered, but the resulting high peak power requirements were considered inconsistent with the low-pressure Martian atmosphere. A linear chirp (pulse compression technique) was selected to obtain the characteristic short-pulse amplitude spectrum, while eliminating the high peak power requirements.

3) Performance characteristics. A bit rate of 10, 000 fps is achieved using a 50-watt transmitter and limiting playout to the periapsis region. With the orbit geometry selected, transmission will be possible approximately every fourth pass. This results in approximately 23 orbiter passes to play out the results of the first 48 hours of the lander scientific mission.

d. Direct link. The direct transmission system utilizes a steerable 5-foot parabolic antenna guided by an Earth-seeking and tracking navigator. This operates in conjunction with a 70-watt transmitter to achieve a bit rate of approximately 1500 bps. PCM/PSK/PM modulation is used in this link which should be devoid of multipath transmission problems. The power system design is such that the full lander scientific mission can be handled by the direct link. In addition to the high-gain direct link, there is a low bit-rate (1 bps) system capable of providing engineering status information in the event of gross system failures in the lander.

e. Command system. There will be a capability of receiving commands from Earth through a gimballed S-band slot antenna and a 5-foot parabolic antenna. In addition, a command receiver connected to an S-band antenna will be capable of receiving commands via the orbiter. Commands from Earth will use PCM/PSK/PM modulation, while those via the orbiter will be PCM/AM. The general features of the system are shown in figure 18.

7. Power sources.

a. Selected system. A radioisotope (Plutonium 238) thermoelectric generator and nickel cadmium batteries were selected as the reference design for the power system in the Mars lander. The decision on the system best suited for the lander application was essentially made based on the requirements that (1) all components be heat-sterilized and (2) no exhaust be permitted into the environment which could compromise the scientific mission. Although numerous power sources were examined, only three devices remain qualified after application of the above criteria: RTG, NiCd batteries, and solar cells.

An extensive analysis was undertaken to determine the performance of a fixed horizontal, flat solar panel at an arbitrary landing site. The conclusion, as expected. is that the solar panel is unsuitable under ideal conditions. Qualitative considerations such as the uncertainties of landing attitude, latitude, atmospheric attenuation, cloud cover, and restriction to day-time operation aggravate the situation and, alone, are sufficient to eliminate the solar panels' candidacy.

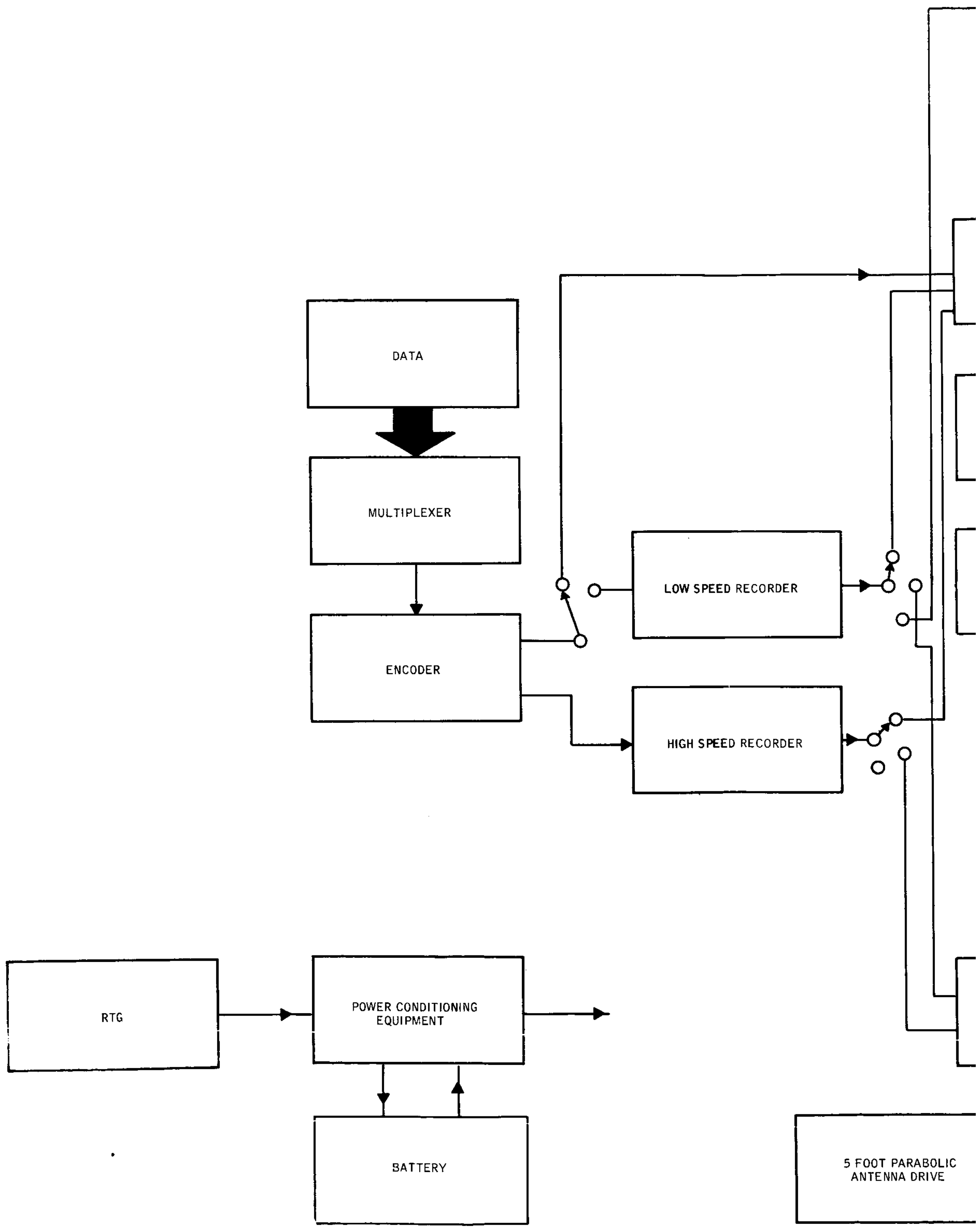
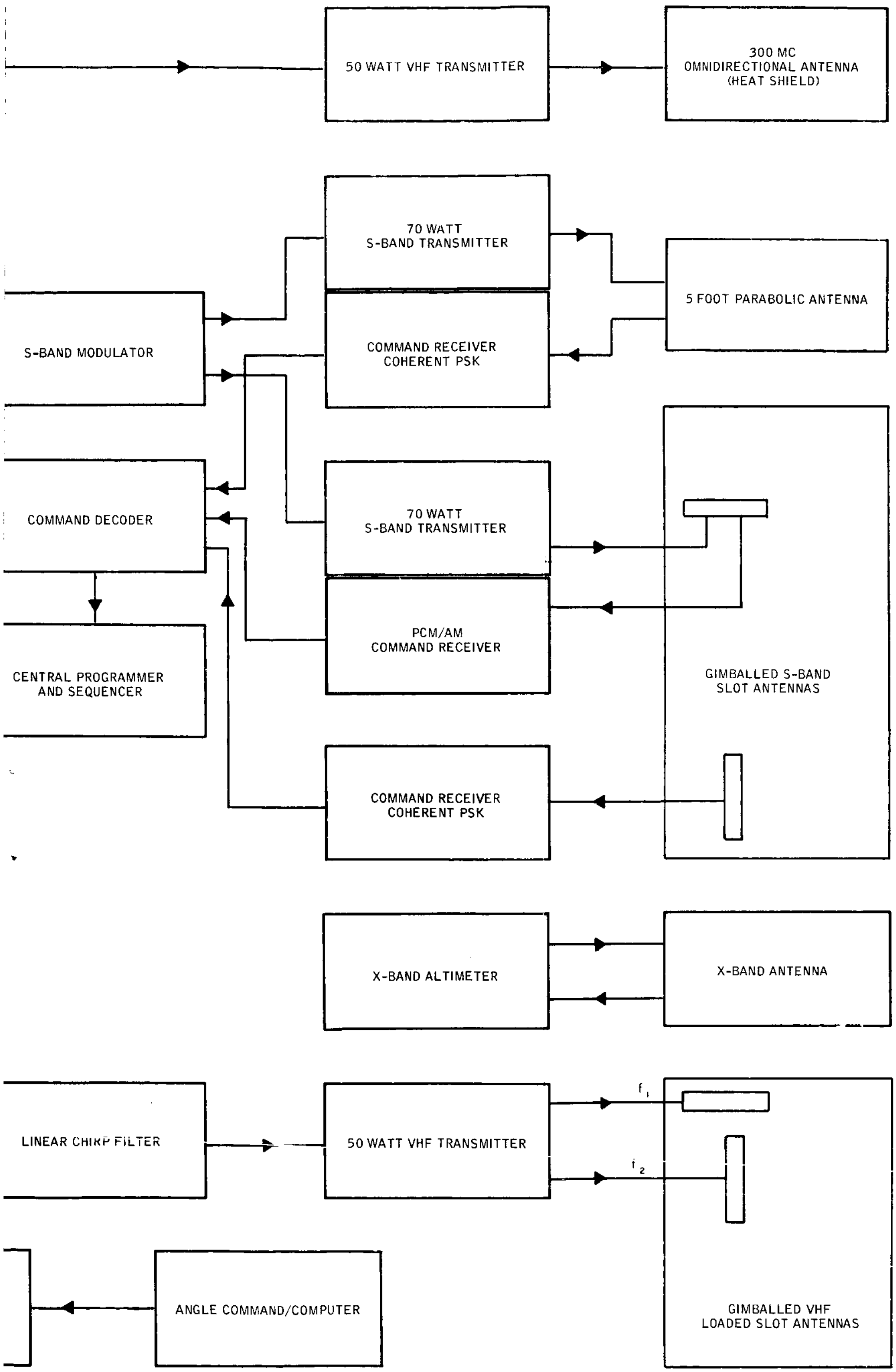



Figure 18 MARS LANDER COMMUNICATIC
63-9241

#1




NS SYSTEM BLOCK DIAGRAM

#2

As for the prospects of using high-energy density, silver-zinc batteries, the problem associated with development of a heat sterilizable battery are severe, the mode of failure is fundamental - the positive plate sheds material and the dynel separator disintegrates - and it is difficult to predict when and whether a system tolerant to heat sterilization without large capacity loss will be available. Problems may be simpler with the automatically activated system since the electrolyte is separated from the battery proper, but it is impossible to check out such a battery. Apart from sterilization considerations, further complications arise because of the need for self-heating of the batteries to maintain operating temperature of not less than 30°F. Since the Martian surface temperature is expected to be much lower, considerable oversizing of the battery will be required to accommodate the self-heating penalty.

b. Power requirements and system description. The RTG shown for the lander case will provide 110 watts. The RTG will be packaged so as to assure a high degree of confidence in the survival of an intact system to prevent radioactive contamination of the planet.

The power profile for the mission is shown in figure 19. Note the scale change on the abscissa. As indicated, the greater burden of the mission occurs during the first 55 hours after landing; thereafter, the instrumentation load reduces to 10 watts and, depending on the communication mode of relay or direct link, pulses of 50 watts for 10 minutes or 251 watts for 4 hours will be required.

The worst case energy storage requirement is established by the data transmission in the direct link mode. This results in a need for 146 pounds of nickel-cadmium batteries. As indicated in the previous section, sterilizable cells of this type are apparently tolerant to heat sterilization, although a 5 to 10 percent capacity loss has been evidenced. The present design involves using the waste heat from the RTG for heating of the batteries and thereby largely reduces the self-heating requirement.

8. Navigation system (Mars lander). A method of determining location on a planet, essentially similar to "shooting" the sun with a sextant, is employed in conjunction with a mechanization of the scheme which allows position to be determined automatically and also provides antenna pointing information for communication with Earth. A sun seeker tracks the sun using a two-gimbal drive. Azimuth and elevation of the sun are determined with respect to the base using angle pickoffs located on the gimbals. The orientation is determined by means of bubble levels located on the sensor. A computer, with stored data on the locations of the sun and Earth relative to Mars, and a clock complete the list of major components. The system is self-contained, simple, reliable, and accurate.

The sterilized sun sensor is a gimballed detector package with binary outputs accurate to ±1 arc minute over ±90 degrees in each of two orthogonal axes of rotation.

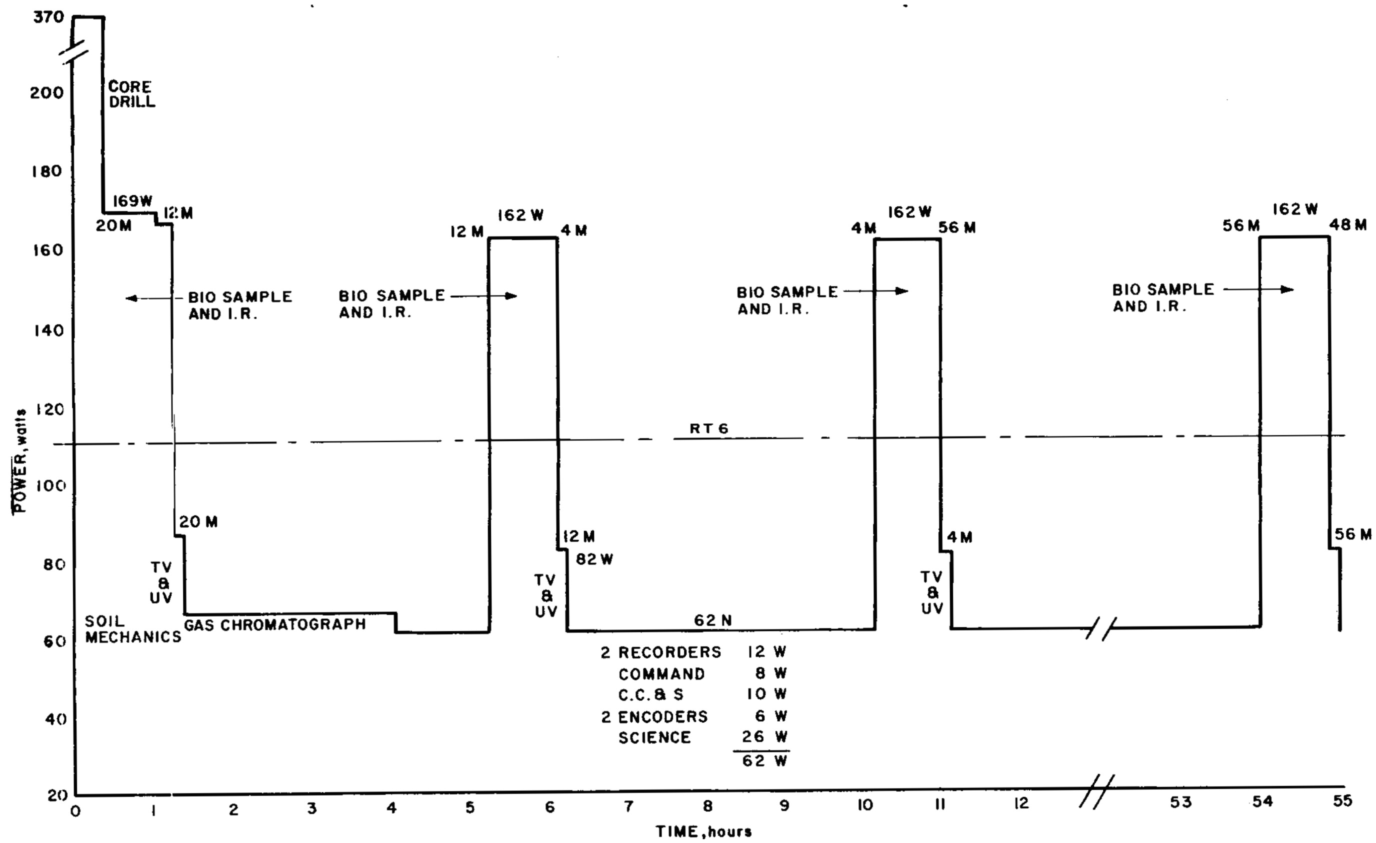



Figure 19 MARS LANDER LOAD PROFILE FOR SCIENTIFIC MISSION
63-9242

The detector package consists of four detectors and a shadow plate in each axis. The detectors are connected in a bridge and fed into a low impedance load so as to behave like current sources. The output current is proportional to detector area covered by sunlight and the level of incident flare density. The detector bridge outputs are fed to amplifiers and d-c motors which drive the gimbals until the bridge output is zero.

The proposed lander computer has one-thousand seven-hundred and ninety-two 24-bit words of permanent memory and two-hundred and fifty-six 24-bit words of alterable memory. The memory is a wired-core fixed memory for program and a coincident current core scratch pad for fast access data.

The computer will be designed to meet the requirements of sterilization and planetary atmospheric entry.

9. Propulsion (Mars lander). Mars lander propulsion system (figure 20) is a pressure-fed, storable, hypergolic, bipropellant engine with a total delivered impulse of 5150 lb-sec. The system propellants are mixed oxides of nitrogen (MON) composed of 85 percent nitrogen tetroxide and 15 percent nitric oxide and an eutectic blended fuel (EMHF) compound of 88 percent monomethylhydrazine and 12 percent hydrazine. An all-welded configuration will be used to prevent leakage. The system is prepackaged with propellants and the fill and vent connections will be welded after filling. This system (table 15) is capable of being sterilized in the prepackaged condition at a temperature of 275° F for periods of 24 hours or more.

The main thrust chamber is rigidly mounted, ablatively cooled, and is controlled for the one period of operation by normally closed and normally open squib valves in series. The chamber is designed to produce 40 pounds of thrust at a chamber pressure of 100 psia. Propellant tank pressure will be regulated to 140 psia. Thrust-vector alignment tolerances are 0.01 inch for lateral displacement and 0.26 degree angular misalignment. The effect of these inaccuracies upon the velocity vector is reduced to an acceptable value by spinning the lander at 10 rpm. Spinup is accomplished by a pair of solid-propellant rockets attached to the exterior of the lander sterile container.

The propellant is contained in two equal-volume oxidizer tanks and two equal-volume fuel tanks. The tanks are packaged as nearly as possible in a dynamically balanced arrangement within the space allowed. The lander spin rate is utilized as a means of propellant orientation to give a dependable means of propellant expulsion. Tank outlets are positioned to allow maximum propellant utilization.

The propellant tanks are filled with the required amount of propellant before sterilization. During sterilization at 275° F, the oxidizer will generate a high vapor pressure. The generated pressure is a function of ullage volume, this

TABLE 15

SUMMARY OF SYSTEM PERFORMANCE PARAMETERS
FOR THE MARS LANDER

| | |
|---|---|
| Thrust (lb) | 40 |
| Chamber pressure (psia) | 100 |
| Mixture ratio | 2.15 |
| Characteristic velocity (C*) (fps) | 5284 |
| Thrust coefficient (Cf) | 1.904 |
| Specific impulse (sec) | 312.7 |
| C* efficiency | 0.94 |
| Cf efficiency | 0.98 |
| Throat area ($in.^2$) | 0.210 |
| Expansion area ratio | 40 |
| Fuel flow rate (lb/sec) | 0.0406 |
| Oxidizer flow rate (lb/sec) | 0.0873 |

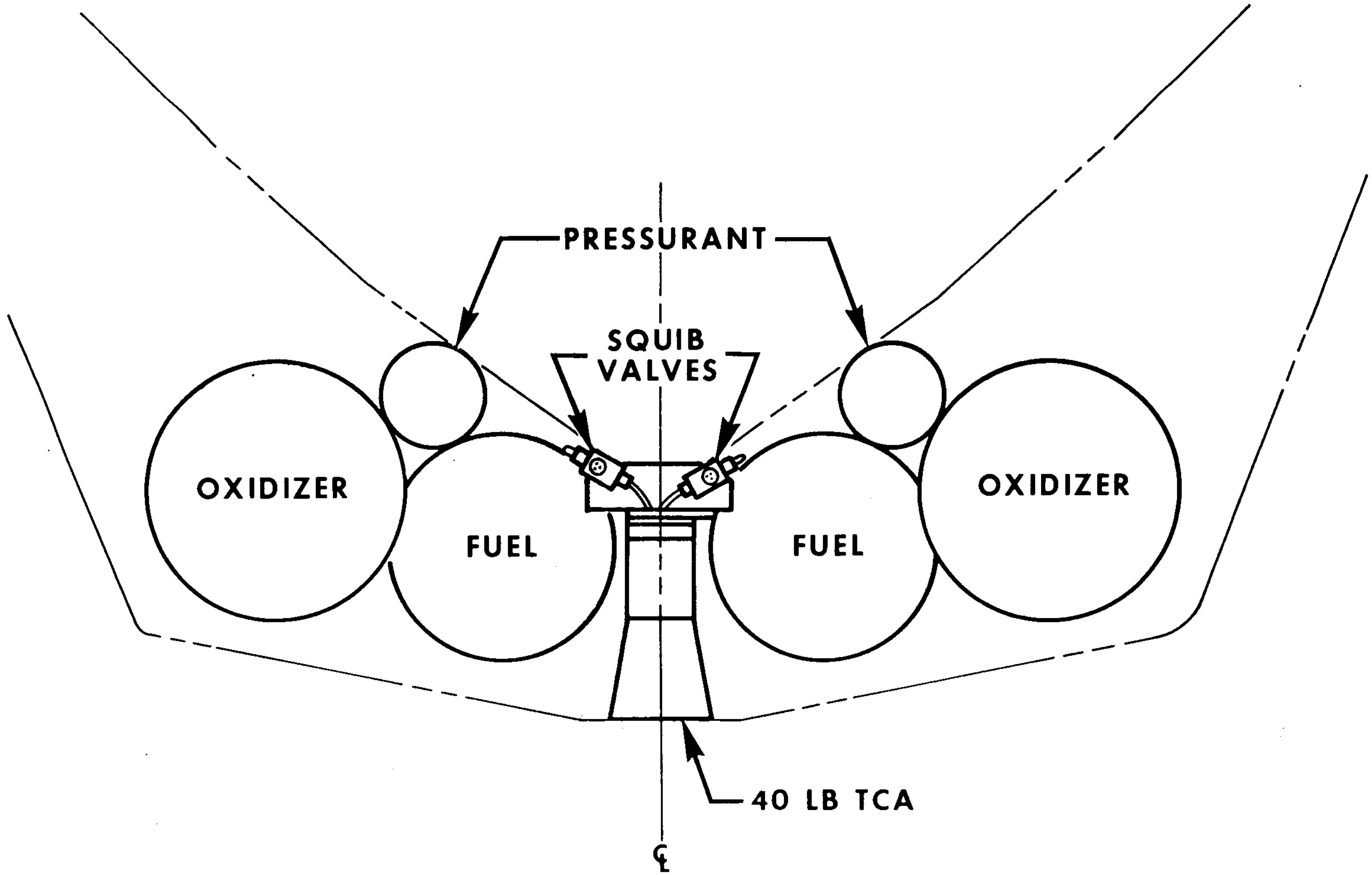



Figure 20 LAYOUT OF THE MARS LANDER PROPULSION SYSTEM
309-188-345

case being designed for a pressure of 1150 psia for a 10 percent ullage volume. A stainless steel, which is compatible with the oxidizer at the elevated temperature, was chosen for the tank material. The steel has such a high strength that manufacturing capability is the governing factor in determining minimum wall thickness. Vapor pressure of the fuel is 53 psia at 275°F which is below the tank design pressure of 140 psia. Aluminum was chosen for the fuel tank material, the wall thickness again being governed by manufacturing capability rather than stress requirements. Normally closed squib valves are used above the tanks rather than check valves because of the absence of diaphragms or bladders in the propellant tanks. The valves insure propellant isolation until system activation. Check valves will then provide assurance of isolating the propellants from one another.

Pressurization is provided by stored helium contained in two equal-volume spheres manifolded together. The two-tank configuration was chosen from packaging and dynamic balance considerations. The pressurant is isolated from the pressure regulator by a normally closed squib valve until the system is activated.

10. Scientific instrumentation (Mars lander). During descent through the atmosphere, the following instruments provide data on the properties of the atmosphere together with television pictures:

a. TV camera - pictures taken at 20,000 to 10,000 feet and immediately before impact.

b. Pressure

c. Density

d. Temperature

e. Velocity of sound

f. Water detector

g. Emission spectrograph for atmospheric composition

Instrumentation for surface scientific measurements includes descent instrumentation plus the following:

a. Wind velocity anenometer

b. Microphone

c. Sun spectrometer

d. Infrared spectrometer, ultraviolet microscope spectrometer, and emission spectrograph to perform biological analysis

e. X-ray diffractometer to identify minerals

f. Three-axis seismograph to measure internal activity of planet

g. Core drill and mill, soil samples are obtained for items 10 through 13

h. Turbidity and ph growth detector

i. X-ray spectrometer

j. Petrographic microscope

k. Mass spectrometer

This choice of instruments carried aboard the lander fulfills not only the primary goal of a biological mission but is also intended to gather data for geophysical and geological studies. The package is so designed that redundancy of information and extended mission life may be achieved. The judicious scheduling of those instruments capable of more than one duty cycle can concievably enable one to attain a 6-month mission with the available power and telemetry.

Emphasis has been placed on obtaining scientific data both during descent and after landing. Those instruments operating during descent are placed so that they can continue to operate after landing.

The instrumentation (202 pounds) is complete enough so that additional weight capability would be best used in multiple landers. This concept would make possible obtaining more valuable information by placing the landers on well chosen landing sites. A heavier single vehicle would not gather significantly more information.

Although the typical list of instrumentation used was assumed (for the purpose of the study) suitable for a Mars mission, it was recognized by NASA and Avco that the instruments, in fact, are not suitable. A major limitation is the obvious inability of many of the instruments to withstand terminal heat sterilization. The design of suitable instrumentation may be a key item in the Voyager timetable.

### 4.3 Mars Lander (JPL Atmospheric Model)

A recently proposed model atmosphere with a significantly lower density profile would require basic changes in lander design. The vehicle ballistic coefficient must be lowered to approximately 0.2 to ensure sufficient descent time. The large diameters associated with a low $M/C_DA$ handicap the V-2 shape because of excessive surface areas and volumes. A blunt shape with conical afterbody would result in significant weight saving; therefore, the Apollo configuration was selected for study of entry into the JPL atmospheric model.

A higher performance descent system is required for the new atmosphere. The parachute deployment would occur at higher velocities to ensure reasonable descent times. The entry vehicle heat shield would be discarded at the time of drogue chute deployment instead of at parachute deployment, thus minimizing the weight of the drogue chute by decreasing the suspended weight at the expense of design complication. Except for these changes, the lander as shown in figure 21 is the same as the reference design. The major impact is a weight increase of about 300 pounds together with the fact that the larger diameter will not permit installation of two landers on the orbiter-bus. The design is only suitable for low wind velocities. An entirely different approach may be required if the wind velocities are as high as 200 ft/sec. Extensive hardening and anchoring features would then have to be considered.

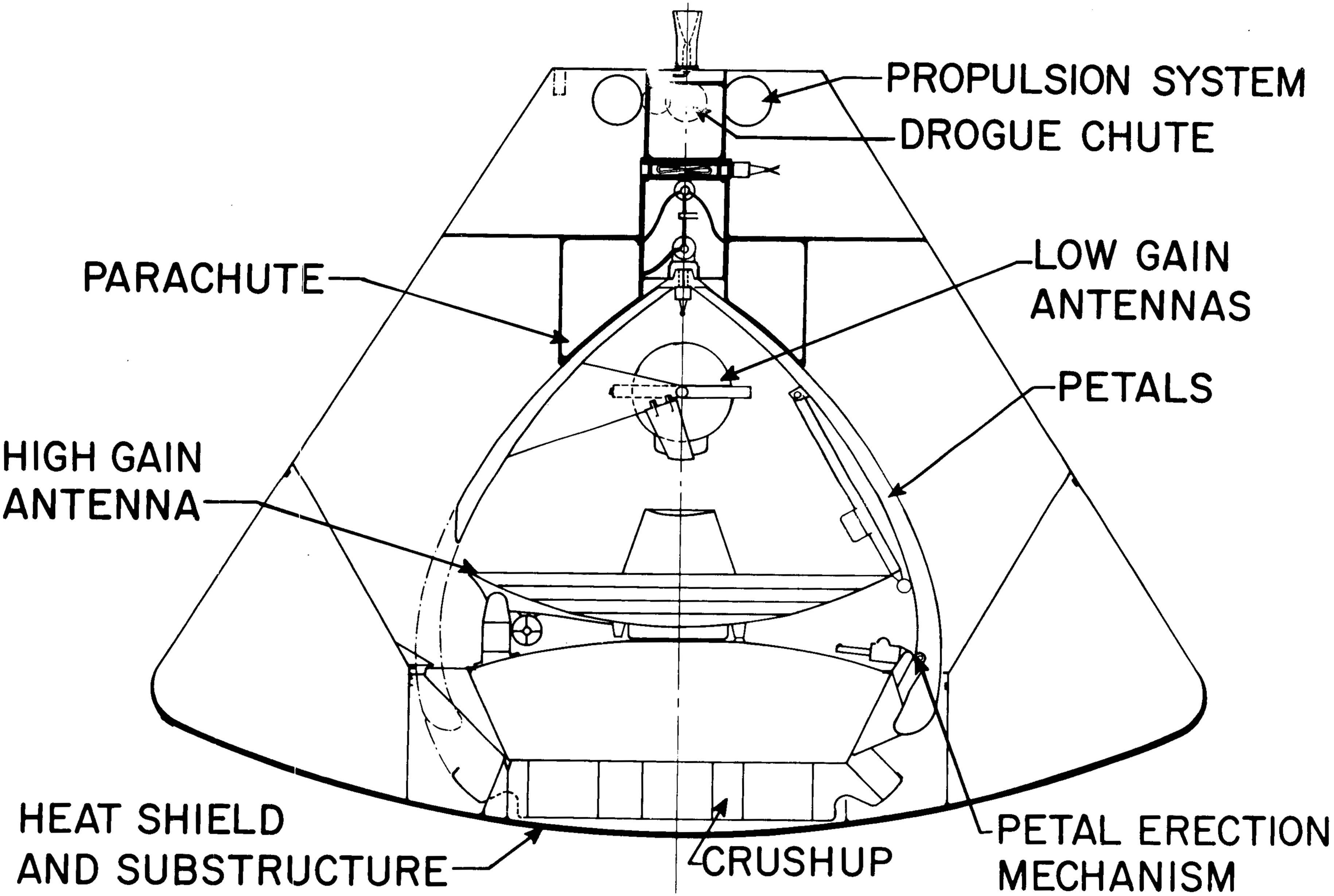



Figure 21 LAYOUT OF THE MARS LANDER DESIGNED FOR THE JPL ATMOSPHERE
63-8545

## 4.4 Venus Orbiter-Bus

The Venus orbiter-bus uses the same basic design that is used for Mars, modified by different solar energy requirements, different mapping instrumentation, and a different lander design.

The solar area requirements for Venus are much smaller than those required for a Mars mission. All cells will be eliminated from the smaller disc and some cells left off the larger disc. However, the substructure of the small disc will be used to support a thermal shield to protect the fuel tanks from the higher solar radiative input and the radiation from the nozzle skirt.

The mapping gimbal will be altered to eliminate visual mapping equipment and 8-foot-diameter and 2-foot diameter-antennas will be added for radar mapping and microwave mapping.

The Venus capsule, because of its small size, will be supported by a truss structure joined to the spacecraft at the central cone structure (above the large pressurant tank). The design can be modified to support two or more landers by extending a suitable truss structure from the main support flange, as shown in figure 22. The Venus lander is similar in dimensions to the Mars lander and requires only a modified adapter section to be installed on the Venus orbiter-bus. The orbiter-bus structure, materials, guidance, stabilization and control system, propulsion, and scientific instrumentation (except in-orbit mapping) are the same as for the Mars orbiter-bus. The telecommunications, mapping, power source, and thermal design are different and will be discussed in turn.

1. Telecommunications (Venus orbiter).

a. General system requirements. The system requirements for the Venus orbiter are similar in nature to those of the previously described Mars orbiter. Data collected in transit and in orbit, as well as data relayed from the lander, will be transmitted along with range and range-rate information. The basic difference lies in transmission range requirements. In an effort to obtain emission and reflection characteristics of the Venusian surface, a radiometer and radar-mapping system will be incorporated. The data from these are instrumental in dictating the communications systems requirements.

b. Orbital system. This system has a transmission capability similar to that of the Mars orbiter, utilizing an 8-foot diameter parabolic antenna in conjunction with a 70-watt transmitter. There are essentially no differences in the data handling, multiplexing, and modulation system.

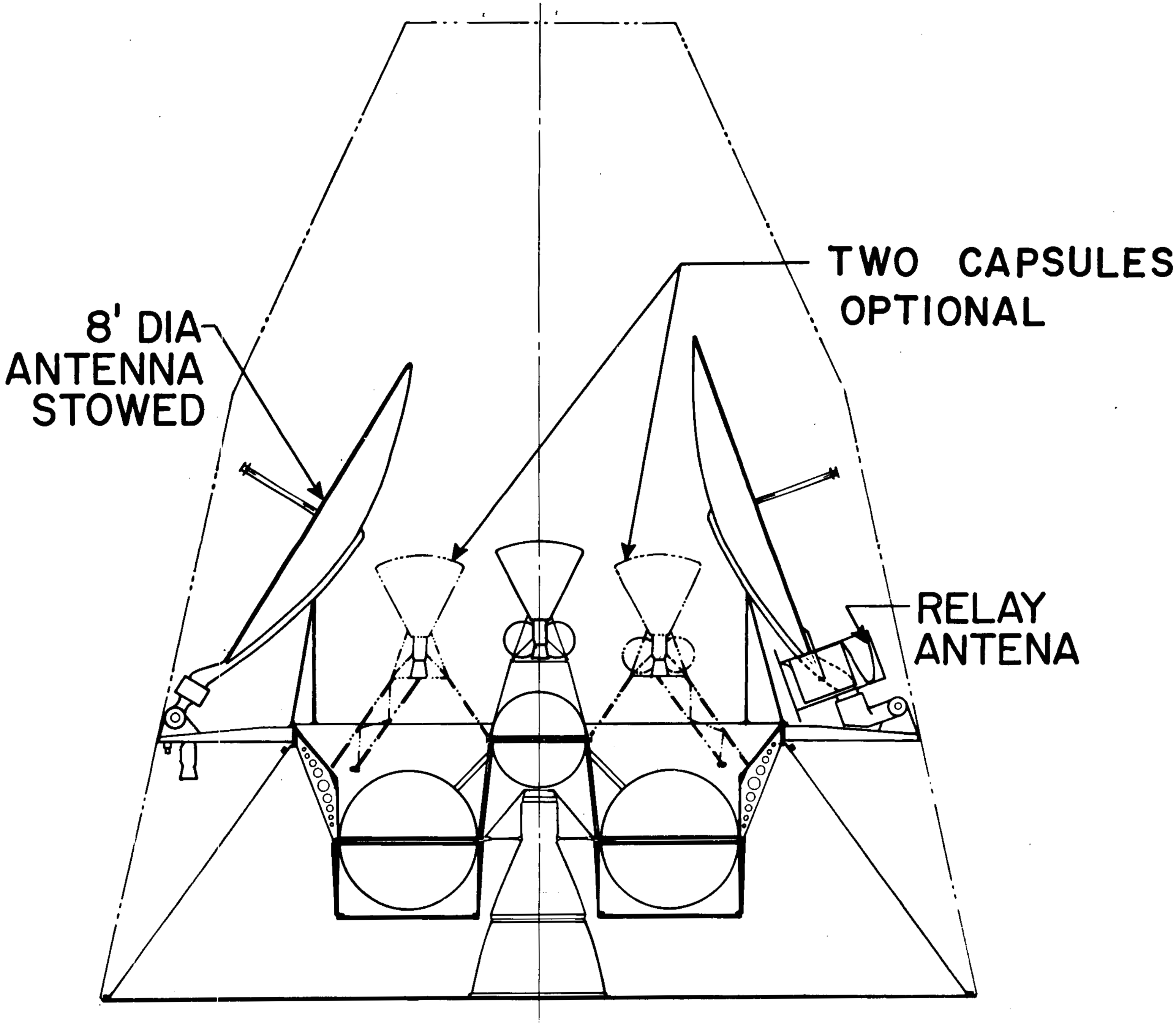



Figure 22 LAYOUT OF THE VENUS ORBITER-BUS WITH THREE CAPSULES
63-8519

c. In-transit system. The in-transit system will have a bit rate of approximately 400 bps, utilizing the same antenna and transmitter as the Mars orbiter. It also will be capable of handling a degraded mission.

The general features of the system are shown in figure 23.

2. Microwave mapping system (Venus orbiter-bus). The radar system proposed here is an X-band pulsed radar capable of operating simultaneously through a single 8-foot-diameter antenna with a fixed-frequency radiometer operating at X-band. The radar operates at a nominal 9500 mc; the X-band radiometer operates at a nominal 8500 mc. X-band was selected for the radar on the basis of available information on the atmospheric transmission characteristics of the planet Venus. The antenna is scanned in a two-way raster-type pattern (i. e., no flyback) with raster lines normal to the ground track. An additional radiometer operating at $K_u$-band will use a 2-foot-diameter antenna. A 1-$\mu$sec pulsewidth was selected to provide a range resolution capability of approximately 0.15 km (500 feet). A pulse repetition frequency (prf) of 3000 pps was selected to provide a high duty cycle and a correspondingly low ratio of peak-to-average power. A 15-kw peak (45 watts average) output power was used in the performance calculations. A nominal prf of 3000 provides an unambiguous range of 50 km or more than five times the height of Mt. Everest; for comparison, a prf of 30 pps or less must be used to provide unambiguous range at the maximum mapping altitude of 4000 km. The prf may be varied to prevent eclipsing and to resolve range ambiguities.

The proposed radar provides the following measurements. At small scan angles (up to several beamwidths off vertical), data will be obtained on the minimum and maximum ranges, corresponding to the highest and lowest surfaces detectable by the radar, together with data on the signal level of the received pulse integrated over parts of the range interval between the minimum and maximum ranges detected. At angles exceeding several beamwidths off vertical (a beamwidth is slightly less than 1 degree), the range difference between the near and far edges (3-db points) of the beam exceeds the range resolution of 0.15 km, and accurate topographical mapping is no longer realizable. Under these conditions, the signal level of the received pulse is integrated over a range-gated increment of the received pulse. Signal level measurements are quantitized in terms of eight "gray levels" or amplitude levels. Noncoherent video integration is used at all times to achieve adequate S/N values with the assumed transmitter power (15 kw peak, 45 watts average). The system will provide an area resolution of up to 1.5 km.

The proposed radar uses a magnetron output stage operating at a peak power of 15 kw and an average output power of 45 watts; the primary power required is approximately 160 watts.

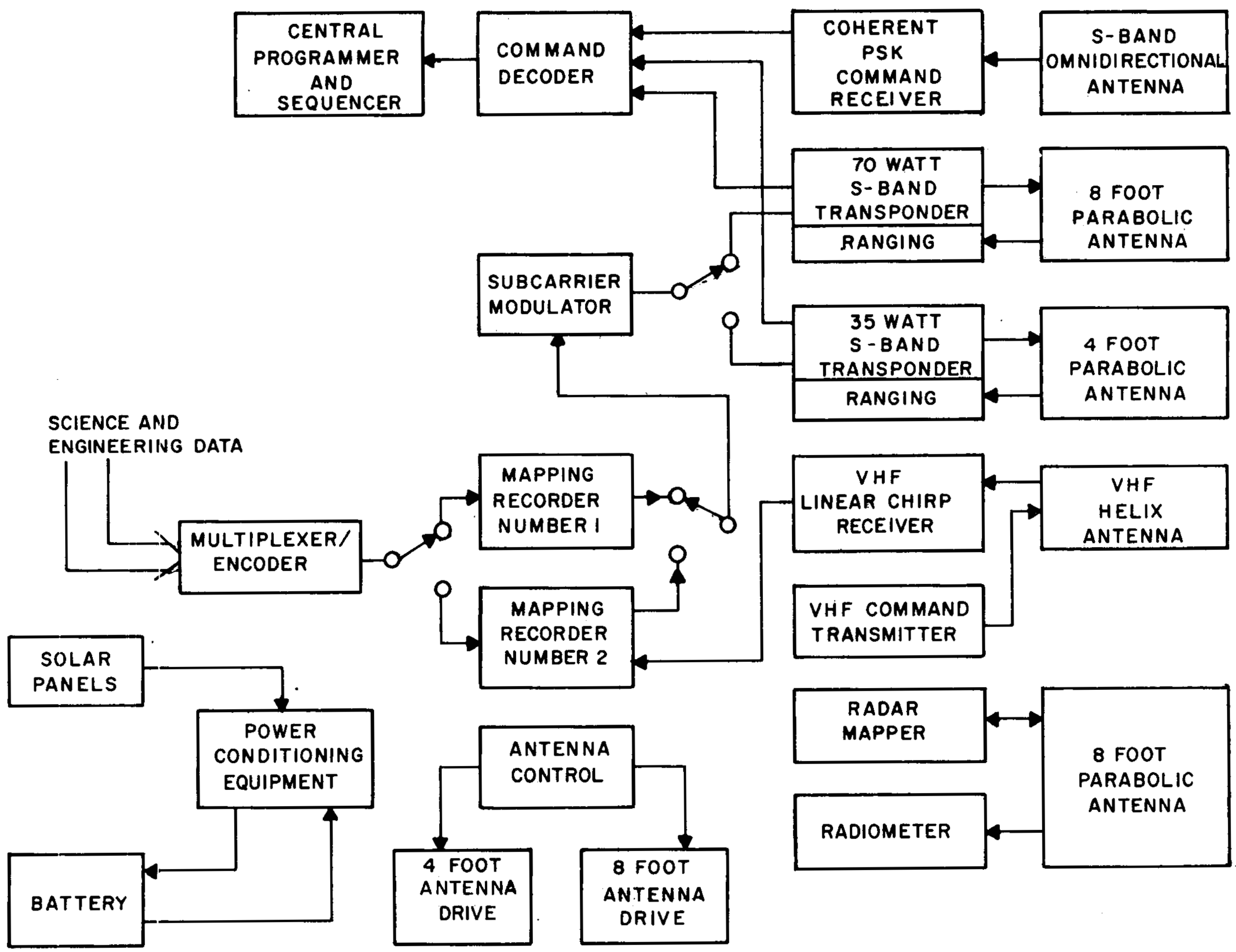



Figure 23 VENUS ORBITER COMMUNICATIONS SYSTEM BLOCK DIAGRAM
63-9240

3. Power source (Venus orbiter-bus).

a. Selected system. The power source selected as reference design for the Venus orbiter is a solar cell array of configuration similar to that used in the Mars case with the exception that (1) the area is reduced to approximately 80 $ft^2$ (2) a conventional flat array will be used, and (3) glass covers for the cells will be required so that reasonable cell temperatures in the higher flux environment may be achieved. Nickel-cadmium batteries are again chosen to provide energy during shadow time and for power-sharing during peak load conditions.

For the primary power source, no candidate other than solar cells can be seriously considered. The RTG is not competitive for the reasons established in the Mars case and for one additional reason, viz, because of the high incident flux, a considerably smaller solar array may be used whereas the RTG is sensibly insensitive to this factor and consequently bears a weight penalty of approximately 130 pounds.

Other devices, such as a solar collector thermoelectric, were not considered because of the many problems associated with their use, e. g., erection mechanism, storage volume, development of reflector materials, and no operational experience. The state of the art for such devices is, in a word, embryronic. Nuclear devices of even the SNAP series are not suitable.

b. System description and power requirements. The raw electrical power obtainable from the solar panel in Venus orbit is 12 $w/ft^2$ for the predicted cell temperature of 200°F. The power system block diagram is identical to that shown in figure 11 for the Mars orbiter and the power profile is shown in figure 24. The smaller orbital power requirement is largely due to elimination of the optical mapping equipment.

4. Thermal design (Venus orbiter-bus). The similarity in the overall configuration between the orbiter-bus for Venus and Mars permits the use of a passive temperature control system of the same characteristics. The sun intensity difference will result in a temperature ratio of 1. 175 to 1. This corresponds to a level of 0°F near Earth with 80°F near Venus as average figures.

The significant difference will be the size of the solar panels. With lower power input requirements and higher energy intensity on the cells, the total solar panel area will be only 80 $ft^2$ on the rim of the large main panel. The top surface of the midsection no longer carries solar panels but is covered with a radiation shield ($\alpha/\epsilon$ = 0. 25). Sun-exposed areas with no cells will be painted white ($\alpha/\epsilon$ = 0. 25) and will thus be resistant to ultraviolet degradation.

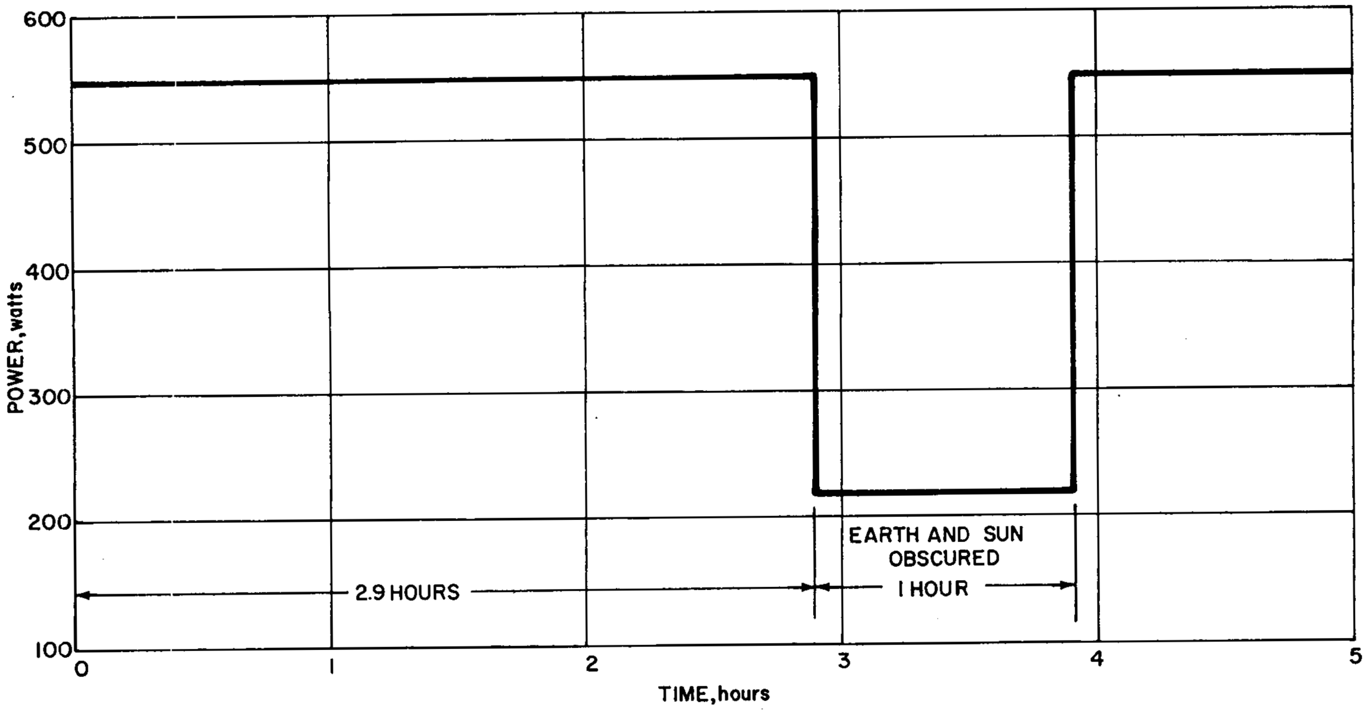



Figure 24 VENUS ORBITER-BUS LOAD PROFILE
63-9324

Temperature levels of 0°F near Earth will rise to 70 to 90°F near Venus. By careful design of electronic equipment capable of operating over this range of temperatures, it is possible to keep all packages within the above design range except for the solar cells. These will be operating at 200°F temperature levels which are acceptable.

## 4.5 Venus Entry Vehicles

Two entry vehicle designs have been considered for Venus, primarily because of the problems associated with direct entry resulting in very high heat fluxes. The first design is a capsule which would not be separated from the orbiter-bus until it was in a planetary orbit. This would permit entry to be made under more favorable conditions. The weight penalty paid in carrying the capsule into orbit and then slowing it to an entry trajectory results in a relatively small vehicle. Although it does not survive impact, it will provide valuable atmospheric information which will permit the design of the direct entry Venus lander to proceed with higher confidence. As discussed earlier, three capsules are carried on the orbiter-bus so that atmospheric data can be taken at several locations on the planet.

The desire to deploy a parachute before entering the Venus cloud layer has been a primary factor in evaluation of the various concepts investigated. The orbital entry and low $M/C_DA$ concepts enable parachute deployment near the top of the cloud layer; however, there are disadvantages. For example, there is a weight penalty associated with injection into orbit. Low $M/C_DA$ vehicles are handicapped by large dimensions for a given total weight.

For direct entry, a high $M/C_DA$ vehicle must pass through the cloud layer while still traveling at high velocities. At this time, little is known about the composition of the Venus cloud layer or heat shield performance while passing through clouds. In addition, this concept would not permit adequate atmospheric measurements of the cloud layer.

The Venus entry studies have used Kaplan's standard and maximum temperature atmospheric models (ref. 2).

1. Venus capsule. The Venus capsule is carried into orbit by the orbiter-bus. In this concept, the capsule is encased in a sterilization can, released, spun up, and the can jettisoned. The capsule is then decelerated sufficiently by the bipropellant liquid propulsion system to achieve planetary entry. The capsule is released from the orbiter near periapsis so the view time of the orbiter is long enough for data playback during capsule descent. The parachute

is deployed near the top of the cloud layer and atmospheric readings are taken every 5000 feet; the capsule does not survive impact. Figure 25 illustrates the basic features of the design.

The orbital entry concept looks promising because of the shallow entry angles and low entry velocities which result in low aerodynamic loads, low heating rates, and parachute deployment near the top of the cloud layer. The radiative heating problem which is significant during direct entry is thus avoided. The Apollo shape was selected for this concept, primarily for the blunt, low M/CDA shape, and the weight-saving conical afterbody.

Since the capsule is designed to operate only during the entry phase and does not survive impact, good insulation to minimize heat transfer from the environment, together with a cold plate for mounting of the telemetry transmitter, will be adequate for the short entry duration.

Because the capsules are placed in the shade of the bus, they will tend to run cold, which is unacceptable to the batteries. The battery electrolyte will therefore be kept at 50°F by thermo-statically controlled heating pads. The power for this heating, as well as for a constant battery trickle charge, will be supplied through the umbilical cord to the orbiter-bus.

The propulsion system uses a thrust chamber to deliver 60 pounds of thrust which is identical to the thrust vector control chamber used on the orbiter-bus.

The system is prepackaged with propellants and the fill and vent connections will be welded after filling. The propellant is contained in two equal-volume oxidizer tanks and two equal-volume fuel tanks balanced in opposition about the system longitudinal axis. The capsule spin rate is utilized as a means of propellant orientation to give a dependable means of propellant expulsion. Tank outlets are positioned to allow maximum propellant utilization (figure 26).

Of primary interest is the composition, temperature, wind velocity, and pressure distribution through and below the cloud layer. Ample time for analysis is provided during parachute descent. The descent period will cover approximately 100 miles; sampling will be done at intervals of 10 miles for a total of 10 samples. The following instruments will be used to obtain simultaneous analyses during time of descent:

a. Emission spectrograph for the analysis of the molecular composition of the atmosphere and elemental composition of the cloud particles

b. Temperature sensor for temperature measurements

c. Barometric sensor for barometric pressure.

d. Light scattering photometer

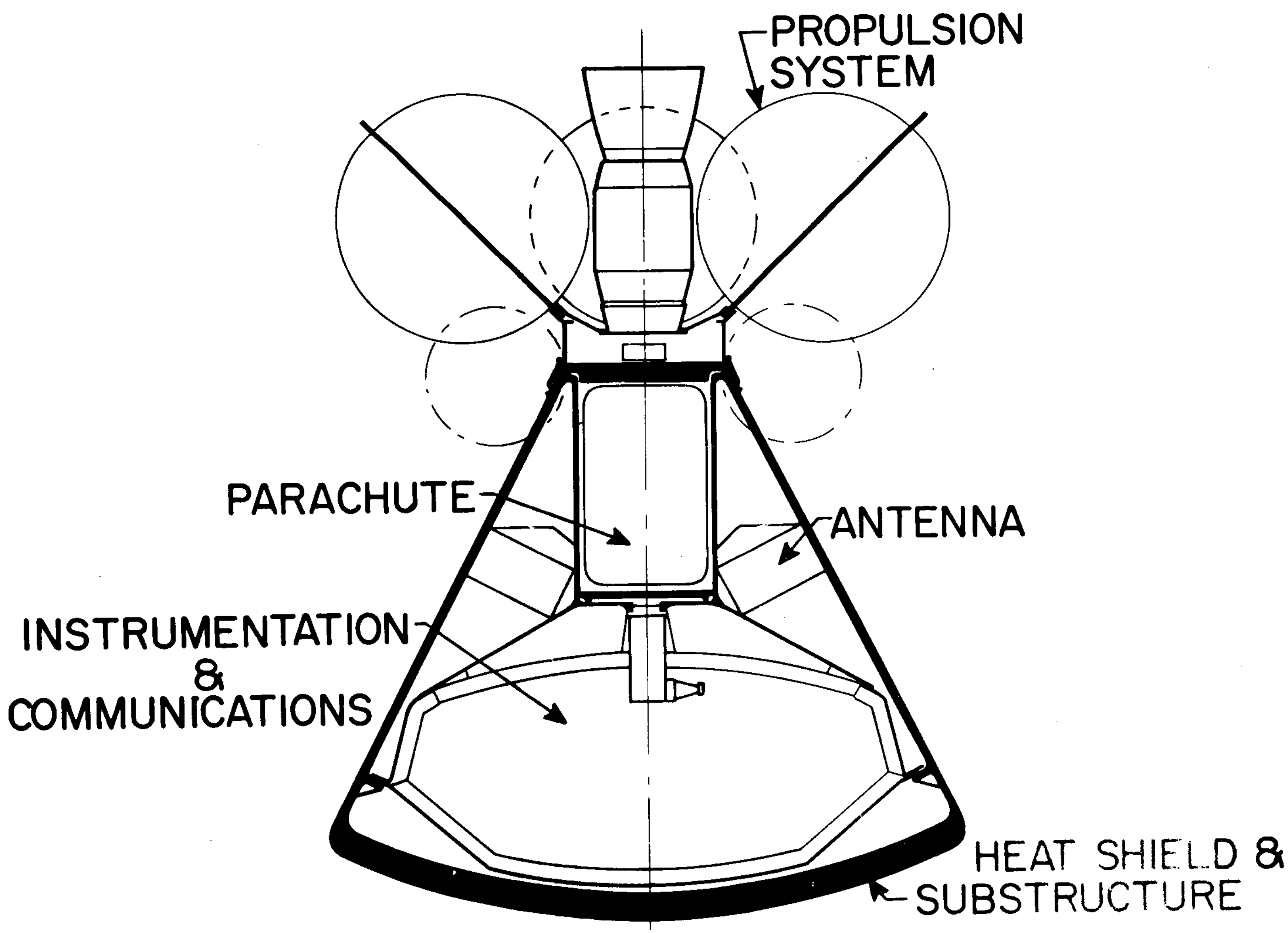



Figure 25 LAYOUT OF THE VENUS CAPSULE
63-8514

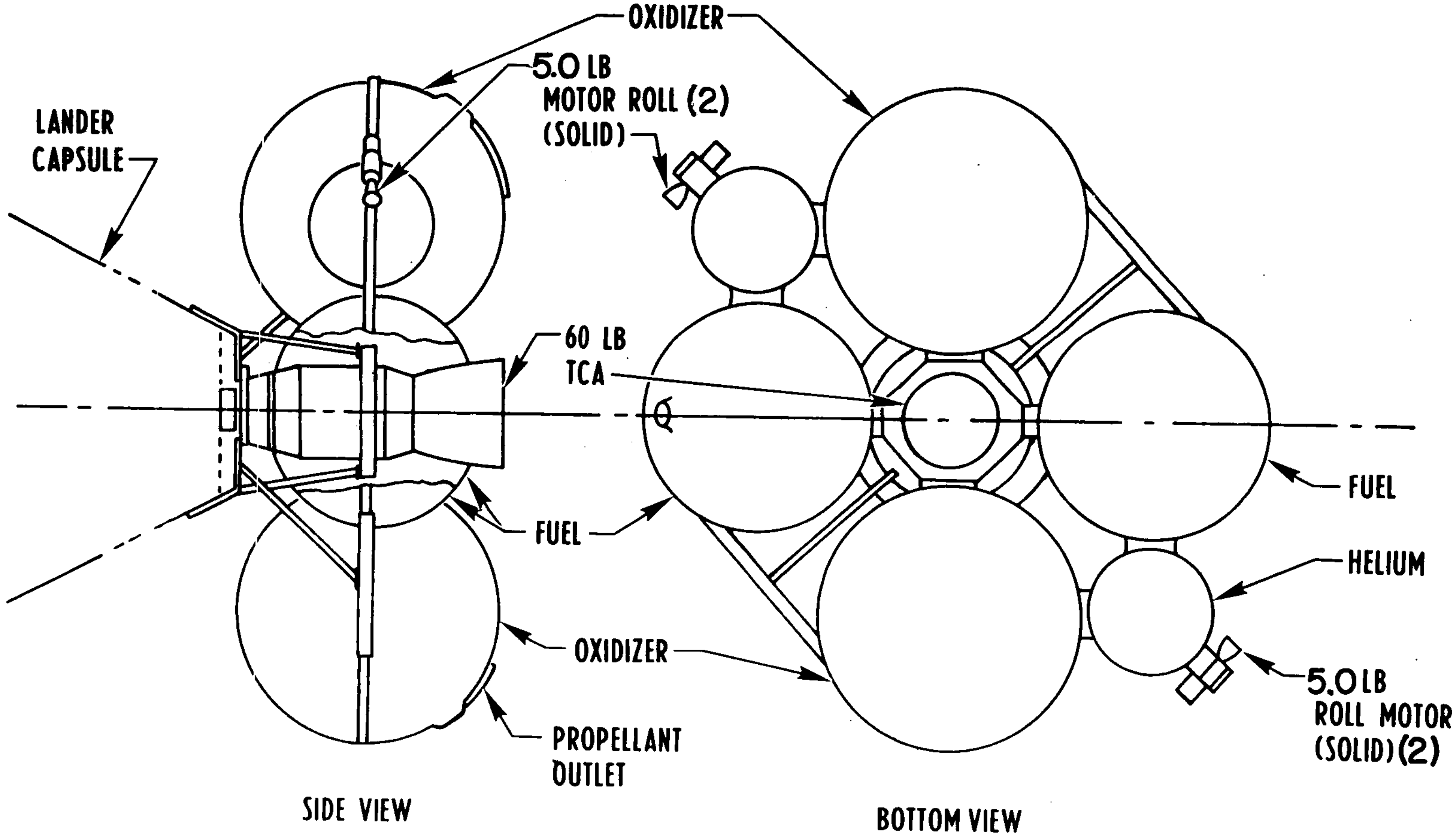



Figure 26 LAYOUT OF THE PROPULSION SYSTEM FOR THE VENUS CAPSULE
309-188-344

In addition, supplementary data will be obtained by a three-axis accelerometer and a radar altimeter for engineering use.

Communication will be relayed via the orbiter during capsule descent through the atmosphere, using a VHF link similar to that employed for the Mars lander. It will have a wide antenna beamwidth and will also employ a linear chirp modulation. The transmitter will operate at approximately 300 mc and 50 watts.

Atmospheric measurements during entry will be recorded and played out shortly before impact. The total bit content during this phase is on the order of 1,000,000 bits. The 50-watt VHF system is capable of relaying these data to the orbiter shortly before impact. Power will be supplied by nickel-cadmium batteries.

2. Venus lander. The direct lander (figure 27) is separated from the orbiter before the orbital phase, similar to the Mars lander. The $M/C_DA$ of this lander is 0.6, a value which was selected as a result of a trade-off between lander size (limited by the ascent shroud), payload capacity, and maximum altitude for parachute deployment. Parachute deployment near the top of the cloud layer is desired to meet the scientific objectives of atmospheric sampling.

The lander installation on the orbiter-bus, and the separation and entry sequence are identical to the Mars lander except that only one parachute is employed.

As the vehicle enters the Venusian atmosphere, it is decelerated aerodynamically until Mach 2.5 is reached where the parachute is deployed through a mortar-type ejection system. The heat shield is cut by means of a shaped charge and is discarded during parachute deployment. At impact, the vehicle is protected by stainless-steel crushup which completely covers the disk-shaped lander. No attempt is made to keep the vehicle erect, and it is allowed to roll and tumble until a final stationery equilibrium position is reached. The disk shape comes to rest on one of two sides. Antennas will be situated such that data will be played out through the surface facing up. The disk shape provides a good ground plane for the antennas, thus minimizing lobing. Communications will be made by relay, with the same VHF system described for the capsule. Atmospheric information will be transmitted as for the capsule during descent. After impact, the lander will acquire one-soil sample and record the data. During the next two orbital periods, the lander will relay the recorded data to the orbiter at a bit rate of 10,000 bps, when the orbiter is at periapsis.

The transmitter will operate at approximately 300 mc and at 50 watts. This will adequately satisfy the lander mission bit content which is in the order of 5000 bits per soil sample. The communications system block diagram is

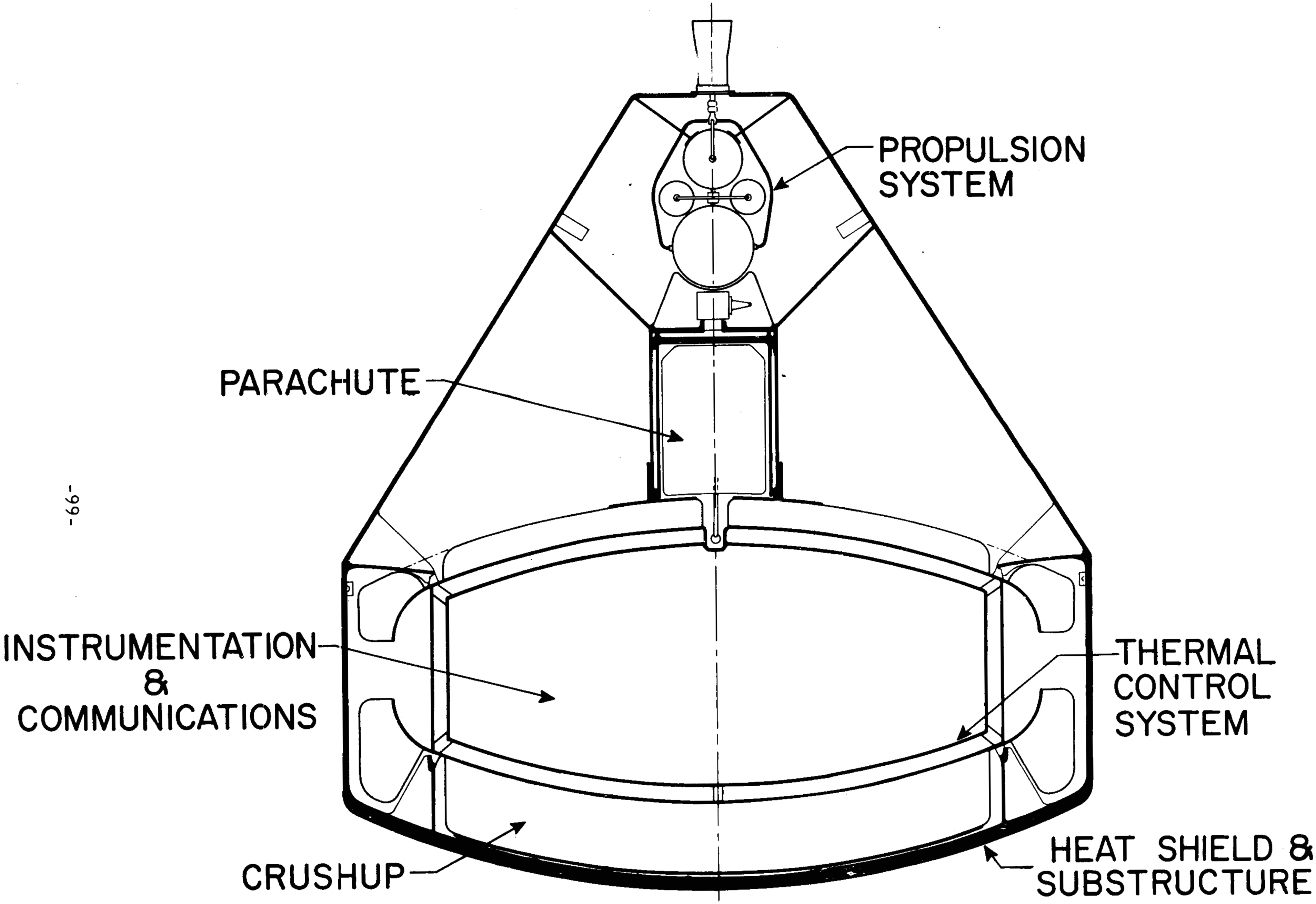



Figure 27 LAYOUT OF THE VENUS LANDER
63-8513

shown in figure 28.

Materials selection for the Venus lander is considerably more difficult than for the Mars lander because of the large atmospheric uncertainties and expected high atmospheric temperatures near 800°F.

The Venus lander heat shield material might encounter heat fluxes over 10,000 Btu/ft$^2$-sec during entry, depending on the trajectory and atmosphere. However, peak heat fluxes of less than 10,000 Btu/ft$^2$-sec are associated with the reference design.

This heat pulse will be primarily radiant, but with appreciable convective heating and shear forces present. Although no current facility can simulate possible Venus entry conditions, estimates of material behavior can be made under less severe heating conditions. The Avco-RAD 10-Mw arc facility is capable of convective heat fluxes up to 3000 Btu/ft$^2$-sec with a constricted type of tube specimen. Tests indicate that graphite-filled epoxy and phenolic show considerable promise as a high heat flux material. Figure 29 shows a typical graphite-based material after an 8 to 10-second exposure to convective heat fluxes of 1000 to 1500 Btu/ft$^2$ sec. This material exhibited a low erosion rate and excellent char stability, both prerequisites for Venus entry. The delaminations noticed are caused by bond-line separation. Heats of ablation for these graphite-based materials are significantly higher than for more conventional heat shield materials, such as Astrolite and OTWR, especially at the higher heat fluxes and enthalpies. A few tests performed under radiant heat fluxes also indicate that the graphite-based materials are superior to silica-based materials. A series of tests conducted in the Avco-RAD solar furnace showed that graphite based materials exhibited little dimensional change after up to 7 to 8 minutes of exposure at a heat flux of 2900 Btu/ft$^2$-sec, while silica-based materials eroded considerably. More extensive arc facilities are required to generate the very high radiant-convective heat pulses encountered by the Venus lander. Avco RAD and the Avco-Everett Research Laboratory have started to develop these facilities; preliminary testing is underway.

Another major material problem in the Venus lander is the parachute material. If the atmospheric temperature is near 800°F, the HT-1 fabric is a marginal material. Recent developments have indicated that polybenzimidazole (PBI) fibers may possess considerable stability at 800°F, but more development is required. The most readily available high-temperature decelerator materials are the metal filaments that can be woven into cloth. These fabrics can withstand temperatures over 1500°F, and may be coated with high temperature polymers (such as HT-1, PBI, or silicones) to increase the drag coefficients upon initial parachute deployment. Problems associated with these metallic fabrics are weight and flexibility for inflight storage.

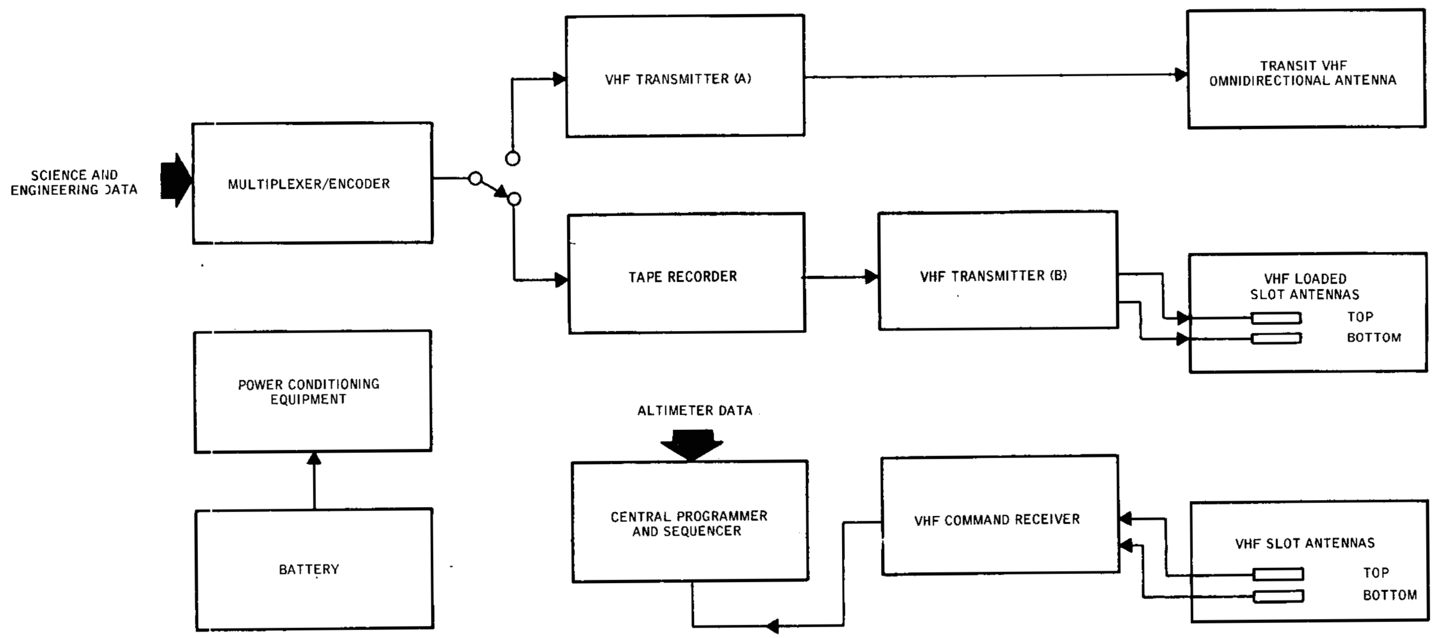



Figure 28 VENUS LANDER COMMUNICATION SYSTEM BLOCK DIAGRAM
63-9239

FLOW

2 3 4

| X-6023 | | |
|---|---|---|
| RUN NO. | 3962 | |
| ENTHALPY | 336 | |
| HEAT FLUX | 1420 | $Btu/ft^2$ sec. |
| SHEAR | 14.3 | $lb/ft^2$ |
| TIME | 8.35 | sec. |

| X-6026 | | |
|---|---|---|
| RUN NO. | 3963 | |
| ENTHALPY | 270 | |
| HEAT FLUX | 1180 | $Btu/ft^2$ sec. |
| SHEAR | 14.4 | $lb/ft^2$ |
| TIME | 9.43 | sec. |

Figure 29 TWO GRAPHITE-BASED SPECIMENS AFTER TESTING IN THE 10-MW ARC FACILITY
P9669A

Thermal control coatings will be similar to that used on the Mars lander. Some variation will be required because of the different in-flight time and the probable requirement for a lower $\alpha/\epsilon$ ratio because of greater proximity to the sun.

The aerodynamic load-bearing structure for the Venus lander is of aluminum sandwich construction which minimizes the structural design and development problems.

The internal structure of the Venus vehicle is of the same construction and has the same load paths as the Mars lander. The Mars structure has a minimum amount of modification due to the thermal control system that must be provided for a survivable Venus lander. The lander is further modified by the fact that there is no re-erection system; consequently the internal structure is surrounded by crush-up material to provide overall protection for toppling after impact. The parachute loads are introduced directly into the internal structure.

The lander has a boiloff type of temperature control unit (ammonia) with a mission time of 10 to 20 hours. By means of a vitreous fiber-type, high-temperature insulation (which also acts as crush-up protection), in conjunction with an evacuated section of multilayer super insulation (aluminized Mylar), the temperature in the instrument and battery enclosure can be kept at about 100°F. Pressurized feed from a small $N_2$-bottle has been selected to obtain reliable flow of evaporant. During the entry phase, the discharge of ammonia will be made into the wake so as not to contaminate gas samples admitted for analysis from the forward section. The tendency of ammonia to decompose at relatively low temperature would result in $N_2$ and $H_2$ species being added to the local atmosphere. This contamination problem cannot be avoided with any type of boiloff system. However replacing the evaporant with a heat-of-fusion type heat-sink material avoids completely the contamination aspects, but shortens considerably the total operating time with a given weight by a factor of approximately 8.

## 5. DEVELOPMENT PLAN

A major result of the Voyager design study was the development plan for accomplishing the program, an estimate of the costs involved, and identification of problem areas, particularly those which would require long lead initiation of facilities or development necessary to meet the first Mars opportunity in 1969. This section presents a summary of these studies; the details are contained in Volume VI, Development Plan.

### 5.1 Schedule

Figure 30 indicates a summary schedule for meeting the launch date in January 1969. The key date is the start of a hardware contract in late 1964. This assumes further that the present conceptual design study is followed by a complete preliminary design which precedes the hardware contract. The schedule allows sufficient time for the results of component and evaluation tests and qualification tests to be incorporated into the design without causing undue changes, excessive costs, and program delays in the manufacturing process. While it is not a leisurely program, there is still room for compression by causing overlap of the development and qualification tests and manufacturing. This would permit a later start at the beginning of the program without sacrificing the 1969 Mars opportunity. Since it would represent an undesirable overlap of activities, it would result in added cost as well.

Two other key activities which could pace the program are the development of sterilization procedures and preparation of the science payload. Unless the sterilization technique is developed and suitable facilities planned and built for other interplanetary programs, this activity must begin in early 1964 with a pilot plant study. The problems of developing the scientific instrumentation, which can be sterilized and which will have the required reliability for the exteneded Voyager mission, are formidable, and must be started promptly to meet the interface dates indicated.

### 5.2 Costs

The total program cost is $798 million, $540 million of which is funded for development. These costs are summarized in table 16 which shows the cost per fiscal year as well as the cost of each launch opportunity. The cost is broken down in table 17 in terms of the major items. The costs included herein are the contractor costs together with facilities. They do not include government costs for the launch vehicle, range support, operation of DSIF, and other

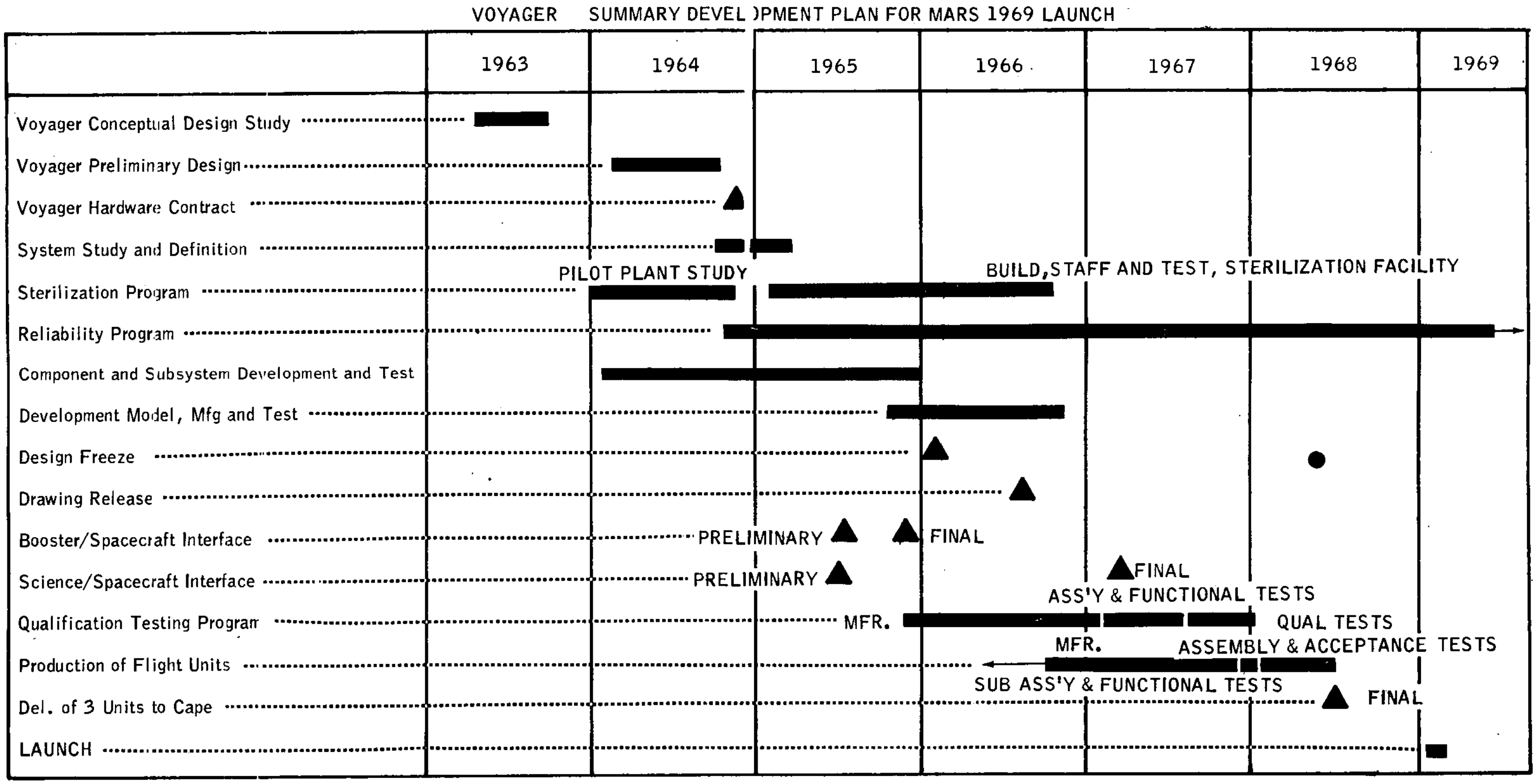



Figure 30 VOYAGER SUMMARY DEVELOPMENT PLAN FOR MARS
1969 LAUNCH
63-8597

TABLE 16

OVERALL SPACECRAFT COST SCHEDULE

| | Start Hardware | | | | M | V | M | V | MV | | VM | | | Launch Opportunities |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| FY | 65 | 66 | 67 | 68 | 69 | 70 | 71 | 72 | 73 | 74 | 75 | 76 | 77 | Total |
| Millions of Dollars | 68.6 | 124.4 | 105.3 | 74.2 | 67.8 | 57.4 | 60.6 | 55.0 | 59.8 | 50.2 | 48.6 | 26.1 | 0.6 | $798.6 |

Cost per Launch Opportunity

| | | Mars | | | | Venus | | | |
|---|---|---|---|---|---|---|---|---|---|
| CY | Dev. | 69 | 71 | 73 | 75 | 70 | 72 | 73 | 75 |
| Millions of Dollars | 540.0 | 30.8 | 36.3 | 30.8 | 34.1 | 32.1 | 32.1 | 31.2 | 31.2 |

TABLE 17

MAJOR ITEM COST SUMMARY

| Program Breakdown | Cost ($\$10^6$) |
|---|---|
| Program Management | 6 |
| System Analysis and Integration | 11 |
| Reliability and Quality Assurance | 79 |
| Communications and Power | 41 |
| Guidance and Control | 32 |
| Propulsion | 47 |
| Heat Shield and Structure | 72 |
| Optical and Radar Mapping | 11 |
| Thermal Control | 15 |
| Spacecraft Design | 40 |
| Ground Support Equipment | 10 |
| Sterilization | 10 |
| Flight Test Support | 34 |
| Manufacturing and Quality Control | 357 |
| Facilities | 15 |
| Miscellaneous | 18 |
| Total | 798 |

NASA activities such a program management and technical direction.

It is assumed that all aspects of the science are government-furnished and are not included in the costs; this includes design, development, and fabrication of the scientific instrumentation in addition to analysis of the scientific data received. Contractor flight test support and analysis of engineering data are included, however.

## 5.3 Problem Areas

The Voyager design study has revealed a number of design and development areas which represent potential problem areas in carrying out the Voyager program. The most important of these are listed and briefly discussed.

1. Sterilization. To comply with the requirement that Mars be kept free from contamination with the probability of 1 in 10,000 requires development of techniques which are beyond the present capabilities of clean room manufacture and assembly. Not only does the requirement for sterilization impose real problems in the reliability of equipment, but also results in major difficulties in demonstration that the desired degree of freedom from contamination has been attained.

2. Surface topography of Mars. One of the purposes of the Mars spacecraft is to obtain more information about the surface topography. Yet in the absence of this information, the lander must be designed so that it has a capability of surviving and communicating with Earth. The proposed solution to this problem is to use a lander with a re-erection capability together with a relay link. More topographical information would clearly increase the confidence in this or any design.

3. Communications. There are several difficulties in the design of the communication systems. First the signal level received by the lander command link with the DSIF is marginal. Second, uncertainty in the surface terrain may result in multipath transmission from the lander. Third, the possibility of voltage breakdown on the lander antennas limits the transmitted power which can be used.

4. Reliability. The extremely long mission lifetime imposes new challenges to the reliability of all components of the system. The flight time for Type II trajectories to Mars is approximately 1 year in 1975. This, together with the 6-month mission duration after encounter, results in a total mission life of a year and a half.

5. Heat shield. The design of the heat shield for Mars imposes no particular difficulties. The heat shield design for Venus, however, represents development problems because of the extremely high heat fluxes which will be encountered during direct entry. Adequate facilities for the simulation of these fluxes do not exist and must be developed.

6. Atmospheric variation of Mars and Venus. The uncertainty in the atmospheric model for Mars and Venus and, in particular, the diurnal and annual variations in the atmosphere require entry vehicle designs which are adaptable to a wide variety of conditions.

7. Instrumentation design. The primary difficulty in scientific instrumentation is the achievement of the high reliability and long life required with instruments which can sustain terminal heat sterilization.

8. Radioisotope thermoelectric power supply for the lander. Development of an RTG power supply capable of delivering 110 watts and having a reasonable weight is one of the pacing items for the program. This activity should be started at once.

9. In-transit thermal control. The same basic orbiter design is planned for both Mars and Venus. The thermal control technique proposed is completely passive and will be achieved by the use of different surface coatings for the Mars and Venus missions. Much more detailed design is required before this approach can be proposed with confidence.

10. Surface environment of Venus. The present lack of information about the surface environment of Venus makes the design of a lander difficult and the design of a direct link capability extremely questionable. The high surface temperature also poses severe problems in designing a lander with long mission life.

11. Mapping of Venus. The cloud cover of Venus makes a visual mapping impossible and its slow rate of rotation makes mapping of any kind difficult. To achieve wide area mapping, either extremely long lifetime in a polar orbit or mapping from different orbital planes is required.

12. Space environment. Adequate knowledge of space environment does not exist at present to design adequate meteroid protection or to design for the effect of other cosmic particles which may cause sputtering and degradation of materials.

The success of the Voyager program will be dependent in part on the ability to find solutions to these problems. An important part of the Mariner program will be undertaken to help provide these answers.

## REFERENCES


1. Schilling, G. F.: Limiting Model Atmosphere of Mars. The Rand Corporation, Rep. R-402-JPL, August 1962.

2. Kaplan, L. D.: A Preliminary Model of the Venus Atmosphere. JPL, Technical Report No. 32-379, 12 December 1962.

# APPENDIX

## TABLE OF CHARACTERISTICS

WEIGHTS (pounds)

| | Mars Spacecraft (single lander) | Venus Spacecraft (single lander) | Venus Spacecraft (three capsules) |
|---|---|---|---|
| Orbiter-Bus | 1849 | 1576 | 1576 |
| Structure and Thermal Control | 324 | 324 | 324 |
| Guidance and Control | 186 | 186 | 186 |
| Communications | 283 | 232 | 230 |
| Power Supply | 461 | 194 | 194 |
| Scientific Payload | 135 | 180 | 180 |
| Propulsion System | 460 | 460 | 460 |
| Lander or Capsule | 1680 | 1330 | 190 x 3 = 570 |
| Adapter, Sterile Can Propulsion | 210 | 220 | 100 |
| Heat Shield, Structure Parachute | 450 | 690 | 40 |
| Touchdown and Deployment | 320 | 160 | -- |
| Scientific Payload | 200 | 80 | 10 |
| Communications | 200 | 60 | 20 |
| Power Supply | 300 | 120 | 20 |
| Orbiter-Bus Propellant | 3000[1] | 3710 | 3710 |
| All-up Weight | 6529 | 6616 | 5856 |

1Amount of propulsion is as low as 1500 pounds on some Mars launch dates.

ORBITER BUS CHARACTERISTICS

| System | Mars Orbiter-Bus | Venus Orbiter-Bus |
|---|---|---|
| Structure | Aluminum monocoque Diameter: 17 feet, length: 15 feet | Same |
| Propulsion | Hypergolic mixed oxides of nitrogen and mixed hydrazine fuels, pressure fed with helium. 1500 to 2500- lb thrust. Auxiliary nozzles for thrust vector control. | Same |
| Mapping System | Television, two sets of optics 85 and 33-inch focal length. Resolution: 250 meters mapping, 40 meters discrete pictures. | X-band radar, 8 foot parabolic antenna. Range resolution: 150 meters; area resolution: 1500 meters.<br><br>X-band radiometer, 8 foot antenna $K_u$ - band radiometer, 2-foot parabolic antenna. |
| Communications | In transit, 35 Watts, S-Band, 4-foot parabolic antenna. In orbit, 120 watts, S-Band, 8-foot parabolic antenna VHF command receiver, helix antenna. S-band command receiver, omni-antenna. | Same except 70-watt transmitter for in-orbit system. |
| Power Supply | Solar cells, 182 $ft.^2$ effective area; nickel-cadmium batteries. | Solar cells, 68 $ft.^2$ effective area; glass covers, nickel-cadmium batteries. |
| Stabilization and Control | Sun-Canopus reference, nitrogen cold gas, limit cycle ± 0.1 degree, using sun sensor, star tracker, and gas-bearing gyros. | Same |
| Guidance | DSIF plus optical-inertial using accelerometers, planet tracker, horizon scanner, and computer. | Same |
| Scientific Instrument (other than mapping) | Particle flux detector, ion chamber, cosmic dust detector, bistatic radar, magnetometer, IR radiometer, infrared spectrometer and micrometeoroid detector. | Same plus microwave spectrometer |

LANDER CHARACTERISTICS

| System | Mars Lander | Venus Lander | Venus Capsule |
|---|---|---|---|
| Lifetime after entry<br>$M/C_DA$<br>Entry Weight (lb)<br>Entry Diameter (feet) | 180 days<br>0.9<br>1470<br>6.4 | 10 to 20 hrs.<br>0.6<br>1110<br>7.0 | 1 hour nonsurvivable landing<br>0.5<br>90<br>2.1 |
| Heat Shield and Substructure | Charring ablator (Avco 5026), 0.65-inch on nose cap, aluminum honeycomb substructure | Same, only 1.8-inch heat shield | Same, only 0.52-inch heat shield |
| Propulsion | Hypergolic (same mixture as orbiter), 40-lb thrust; $\Delta V$ = 100 ft/sec. | Same | Same, only 60-lb thrust; $\Delta V$ = 5000 ft/sec. |
| Communications | . VHF descent telemetry, 50 watts, omin-antenna.<br>. VHF relay, 50 watts, slot antenna, 10,000 bps.<br>. S-band direct link, 70 watts 5-foot parabolic antenna, 1500 bps.<br>. S-band direct link, 70 watts, slot antenna, 1 bps. | . VHF descent telemetry, 50 watts, omni-antenna<br>. VHF relay, 50 watts, slot antenna | VHF relay, 50 watts, slot antenna |
| Power Supply | RTG, 110 watts, Pu-238; nickel cadmium batteries, 185 pounds | Nickel-cadmium batteries, 20 pounds | Nickel-cadmium batteries, 6 pounds |
| Stabilization and Control | Spin-stabilized | Spin-stabilized | Spin-stabilized |
| Guidance | Sun-tracking position location system. | None | None |
| Scientific Instruments | [1] TV Camera<br>[1] Pressure<br>[1] Temperature<br>[1] Sound Velocity<br>[1] Water Detector<br>IR Spectrometer<br>UV Microscope/Spectrometer<br>[1] Emission Spectrograph<br>Sun Spectrometer<br>Anemometer<br>Microphone<br>X-Ray Diffractometer<br>Three-Axis Seismograph<br>Turbidity + pH growth<br>X-Ray Spectrometer<br>Core Drill and Mill<br>Petro Microscope<br>Mass Spectometer<br>[1] Density | [1] TV Camera<br>[1] Density<br>[1] Pressure<br>[1] Temperature<br>[1] Mass Spectrometer<br>[1] Emission Spectrograph<br>Surface Sample<br>Acquisition System<br>[1] Light-scattering photometer | [1] Emission Spectrograph<br>[1] Temperature<br>[1] Pressure<br>[1] Density<br>[1] Light scattering photometer |

[1] Measurements made during descent

## DISTRIBUTION

| Addressee | No. of Copies |
|---|---|
| NASA<br>Director, Lunar and Planetary Programs<br>400 Maryland Avenue SW<br>Washington 25, D. C.<br>Attn: D. P. Hearth, Code SL (+1 reproducible) | 10 |
| Central Files | 1 |
| Document Control | 5 |
| Research Library | 100 |