STUDY OF MARS AND VENUS ORBITER MISSIONS LAUNCHED BY THE 3-STAGE SATURN C-IB VEHICLE

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PREFACE

A study of advanced planetary missions and spacecraft systems was initiated in May, 1962, by the Planetary Programs Office of the Jet Propulsion Laboratory. A study group was formed consisting of representatives of the several technical disciplines. The objective of the group is the study of planetary missions which might follow the Mariner B missions in the NASA Planetary Exploration Program. The group examines and assesses systems concepts, capabilities and the problem areas of planetary and interplanetary spacecraft. Study results are to be used by the Planetary Program Office to aid in:

1) Formulating long-range Laboratory plans.
2) Determining launch vehicle requirements.
3) Establishing Laboratory Advanced Development Programs.

The first portion of the study effort, from June 1, 1962, to November 1, 1962, was devoted to parametric studies in the several technical areas and to an examination of an advanced planetary orbiter mission. The results of this effort are covered in Volume I of EPD-139.

The second portion of the study, between November 1, 1962, and February 1, 1963, examined limited capability Mars and Venus orbiter missions launched by the three-stage Saturn S-I/SV (C-1B) vehicle. The results of this effort are described in Volume II of EPD-139.

This volume presents the third part of the study, from February 1, 1963, through August 1, 1963, and considers a combination orbiter/landing capsule mission for the 1969 Mars opportunity. A range of both orbiter and lander capabilities has been examined consistent with the performance capability of the Saturn S-I/SV vehicle.

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CHAPTER 1

INTRODUCTION

I. STUDY OBJECTIVES

The broad objective of this study is to examine conceptual designs and accompanying mission capabilities for a 1969 Mars combination orbiter/lander spacecraft system launched by the Saturn S-I/SV (C-1B) vehicle system. Specific objectives are to:

1) Determine the possible range of mission capabilities, scientific and other, for this class of spacecraft system.
2) Establish the functional requirements of the family of spacecraft systems required for performance of combination orbiter/lander missions.
3) Consider alternate technical approaches for accomplishing these functional requirements and illustrate conceptual system designs which embody certain selected technical approaches.
4) Present the mission capability for the selected (reference) system designs and the variation in capability which is realized by systems which differ from the reference designs.
5) Identify the major technical problems of each conceptual spacecraft system and indicate the present and required level of effort by the Laboratory for solution of these problems.

II. GUIDELINES

The study was performed under the following general guidelines:

1) The spacecraft systems are to be launched by a Saturn vehicle consisting of the S-I, SIV-B, and SV stages.
2) A Mars mission is to be performed with launches in spring of 1969.
3) Completion of the study by August 1, 1963.
4) Depth of effort to be sacrificed, if necessary, to perform the study within the allotted time period.

III. STUDY PLAN

The approach selected for this study is outlined in the series of steps given below:

1) Document the study constraints and limitations including:
   a) Launch vehicle characteristics including vehicle performance, environment, and physical constraints such as shroud envelope.
   b) Mission energy requirements.
   c) Launch period and range safety restrictions.
   d) Sterilization requirements.
2) Initiate an examination of mission objectives including:
   a) Scientific objectives for a Mars orbiter and lander and a potentially attractive set of experiments which satisfy these objectives.
   b) Preparation of interface requirements for these experiments.
3) Define the functional requirements and interactions for a combined orbiter and lander spacecraft system.
4) Investigate alternate ways of accomplishing the functional requirements of both orbiter and lander missions.
5) Select one or more conceptual designs for the orbiter and lander systems. Indicate how variations in both orbiter and lander designs provide a range of mission capability from partial to full accomplishment of orbiter and lander objectives, respectively.
6) Show what combinations of orbiter and lander designs are compatible within the launch vehicle performance capability.
7) Provided documentation which describes the several conceptual designs including the following items:
   a) General description
   b) Configuration
   c) Weight breakdown
   d) Power profile
   e) Flight operational sequence
   f) Mission capability
8) Describe the further study efforts required to evaluate the alternate technical approaches and the feasibility of suggested conceptual designs.

CHAPTER 2

SUMMARY OF STUDY RESULTS

I. STUDY SCOPE AND LIMITATIONS

A combination orbiter/landing-capsule mission for the 1969 Mars opportunity has been considered in terms of possible conceptual spacecraft system designs, corresponding mission capabilities, and the associated problem areas. A range of orbiter and lander capabilities has been examined which is consistent with the injection capability of the Saturn I-B/SV vehicle. The missions and systems considered have been found to be scientifically significant, technologically achievable, and potentially attractive as candidates for inclusion in the NASA Planetary Exploration Program, and hence may be worthy of a complete design study effort.

However, it should be noted that the study results are not described in terms of mission feasibility. It is considered of paramount importance that a clear distinction between feasible and "potentially attractive" or "conceptually possible" be made and that the results of this study be presented in terms of this distinction. This point of view is based on the following considerations:

1) Both the total weight available for the spacecraft and the launch opportunity have remained fixed throughout the study.
2) The study has been limited to a short time period; technical depth of effort and a realistic approach to an appropriate reliability analysis have been sacrificed accordingly.
3) The study group has not examined the program resources such as manpower and funding that would be required for mission accomplishment in parallel with the technical requirements of the 1969 orbiter/landing-capsule mission.

Each of these areas must be considered before it can be stated that a specific mission capability or system weight estimate is achievable in 1969, that is, that a specific type of mission is feasible for the 1969 launch opportunity. The purpose of the present conceptual design studies, as given in this study report, is to provide an estimate of the systems performance and mission capability of a Saturn I-B/SV launched planetary spacecraft having simultaneous orbiting and landing capability. In the course of preparing this estimate of system and mission capability, a number of problems both at system and subsystem level have been identified. Those problem areas which warrant further investigation are discussed in Chapter VI of this document.

Using the data developed within this study, it should be possible for the Program Office to estimate the utility of the orbiter/landing-capsule spacecraft in meeting the overall objectives of the Program; and, in doing so, definitize the specific objectives of advanced planetary missions. If the systems investigated here are found to have a high degree of utility within the Program, and interest in them continues, logical next steps would be to:

1) Pursue the problem areas of the orbiter/landing-capsule system and find appropriate solutions for these problems.
2) Initiate an integrated design study/mission-success analysis effort for the purpose of investigating the orbiter/landing-capsule spacecraft design in sufficient depth to establish feasibility and determine the accompanying tradeoffs between probability of mission success and extent of mission capability.

II. SIGNIFICANT ASSUMPTIONS

The study has been performed subject to certain assumptions and constraints which are listed below:

1) Attention is restricted to the 1969 Mars launch opportunity. Varying energy requirements for the period 1967-1972 have been examined previously in EPD-139, Volume II.
2) The Saturn I-B/SV launch vehicle characteristics, including vehicle payload capability, injection accuracy, environment and physical constraints such as a 152 inch diameter shroud envelope.
3) Present range safety requirements which limit the range of launch azimuths from 90 deg to 111 deg East of North. (On rare occassion, waivers have been obtained to launch up to 114 deg East of North.)
4) Two spacecraft systems to be launched within a 30 day launch period, assuming two launch pads are available.
5) A launch in early 1969 of a spacecraft system of this magnitude requires initiation of preliminary design early in 1965; hence the conceptual designs examined here are based on mid-1964 state-of-the-art. In only a few isolated cases has projection or extrapolation of 1964 performance to later years been permitted.
6) Design of a capsule entry system based on Martian atmospheric models described by G. F. Schilling.[1] (If recent observations of a much thinner Martian

atmosphere by L. D. Kaplan and others[2] are confirmed, much of the entry, work done here will have to be re-examined and revised.)

7) Firm sterilization requirements will have to be met; these are discussed in detail in the following section.

A. Sterilization

It is generally accepted that the study of extra-terrestrial life is a dominant scientific objective of the exploration of space. In fact, a great many individuals and scientific groups regard the search for extra-terrestrial life as the primary objective of space exploration. The detection of life forms or elements of life forms on other planets would undoubtedly be considered one of the great discoveries of all times, since it could be a great step toward answering questions concerning the origin of life. To enhance the possibilities of attaining this goal, it is of the utmost importance that no viable earth organism be transported to those planets which may possibly support life. Certainly the unmanned explorations should take every precaution to maintain the natural ecology of the planet, and prevent life detection experiments from detecting terrestrial organisms which may have been delivered to the planet by our own space vehicles.

B. Requirements for Planetary Spacecraft Sterilization

The requirement for maintaining the natural ecology of the planets is well documented in the literature. In fact, some individuals have expanded the requirement to include the prevention of landing any life elements, dead or living, on the planets since future detection methods may be based on determination of certain biochemical components peculiar to life. Thus, the detection of such material deposited on the planet by space vehicles from Earth could lead to erroneous conclusions. Although the probability for the occurrence of this type of detection is remote, it still remains a possibility and points out the advisability of maintaining exceptionally clean spacecraft.

No one knows or can reliably predict the results that may develop from introducing terrestrial organisms upon the surface of other planets. It appears that all of the planets of our solar system have environments which, in general, would be considered hostile to Earth organisms. The planets Venus and Mars have been considered as probable candidates to support life of certain Earth organisms. However, there is now considerable doubt about Venus because of data collected by the Mariner R fly-by. Mars still remains a good prospect and, in fact, several species of micro-organisms have survived and reproduced in a simulated Martian environment. Thus, it appears likely that certain organisms which might be landed on Mars would survive the Martian conditions. Of greater concern than survival is the possibility that the contaminants could reproduce at a rate which would allow them to become generally disseminated over large areas of the planet's surface during the time period between launch opportunities. Such a growth of Earth organisms on Mars would be a scientific catastrophe and every possible precaution must be taken to prevent the occurrence of such a disaster.

C. Sterilization Guidelines

Over the past few years there have been suggested and recommended guidelines designed for the purpose of preventing the contamination of the planets by terrestrial organisms. The latest revised guidelines are as follows:

1) For Mariner buses and booster last-stages, either sterilization must be used or trajectories must be controlled, to insure not over 10^{-4} probability of hitting Mars and not over 10^{-2} probability of hitting Venus.
2) A Voyager entry capsule for Mars should be given recognized and accepted (official) sterilization treatment and handled aseptically thereafter. The goal of these activities should be that there is less than 10^{-4} probability that a single living organism is released on the planet's surface. This figure takes into account the probabilities of sterilization during Mars entry and impact and of releasing organisms from the capsule at the planet.
3) To achieve this probability, capsule sealing and separation mechanisms must be designed to provide a very high degree of assurance that leaks and malfunctions which would affect the capsule cannot occur prior to, or during launch, or at capsule separation.
4) Capsule sterilization should, if possible, be by heat in the final sealed container, with no access permitted or mechanically possible thereafter.
5) If heat sterilization of the entire capsule is impossible, heat sterilization should be used on as large an assembly as possible, and on sterile parts, including fluids, aided by a glove box procedure using ethylene oxide in the box. All packages, components, materials, fluids, and tools must be sterilized. Cognizant engineers should be held responsible for sterility to be achieved and maintained according to procedures specified and supervised by a Control Sterility Group.
6) A Control Sterility Group should be established as a unit within the organization responsible for planetary missions. This unit should be vested with authority appropriate for meeting the following responsibilities:
   a) Designation of certified (official) procedures to be followed in order to achieve and maintain the sterility of space probes.

b) Instruction and training of cognizant engineers in specific procedures designated by the C. S. G.

c) Continuous monitoring and recording of sterilization procedures and sterility maintenance.

d) Reporting progress and results of sterility program directly to Project Management and to NASA headquarters.

e) Recommending research and development work to improve sterility procedures.

7) Assembly, disassembly, repair, or calibration operations on a sterilized capsule should be permitted only under rigid sterility control.

8) A spacecraft sterilization handbook should be assembled that would include the following:

a) Complete directions for effective techniques of sterilizing spacecraft components, subsystem, and assemblies. The various techniques should provide for internal sterilization, aseptic assembly and applicability of clean room facilities, terminal sterilization, and other methods to include emergency provisions.

b) Standard operational procedures for the determination of levels of contamination.

c) An enumeration of key organisms, test materials, and methods to be used for sterility testing.

d) Methods for the preservation and monitoring of sterility.

Appendices would include:

a) Toxicities

b) Temperature vs. Time Curves

c) Physiochemical properties of sterilization agents

d) Handling procedures and provisions

e) Qualified components list

It should be pointed out that these are guidelines and as such, there undoubtedly will be additions, modifications, and deletions as dictated by research and investigations into spacecraft sterilization.

The above guidelines adopted by the Conference on Spacecraft Sterilization may not necessarily represent official policy; however, the official NASA policy regarding the contamination of planets has been stated as: "The probability of landing one or more viable terrestrial micro-organisms, with either spacecraft or final stage booster, shall be less than 10^{-2} for Venus and 10^{-4} for Mars."[(3)] Although there is no reference to any particular type(s) of spacecraft, from the viewpoint of contamination the term spacecraft would be interpreted to mean landers, orbiters, fly-bys, or any fragments thereof which would possibly contact the planet's surface or atmosphere.

To meet the specification of the above policy, planetary spacecraft must either be sterilized or there must be assurance that the spacecraft will never be put on an impact trajectory. For planetary landers and orbiters this means that sterilization (and thus sterilizability) of components and spacecraft becomes a design constraint and, as such, should be given adequate consideration in the development of concepts for planetary spacecraft. Sterilization of spacecraft may be looked upon as an engineering nuisance, but with proper engineering design this constraint should be just as satisfactorily met as have been other constraints such as vacuum, vibration, radiation, etc. It is understood that in the planetary program, sterilization waivers such as those granted in the lunar program will not be allowed.

D. Sterilization Methods

Sterility may be defined as the absence of life. Sterilization may be accomplished by using some means for either the destruction or removal of living organisms, primarily microorganisms, that may inhabit a material. The various methods available for the destruction of microbial life may be generally classified as chemical, radiation, or heat. Chemical sterilization methods employing either liquids, vapors, or gases are primarily restricted to the sterilization of surfaces and thus are not applicable where internal sterilization is also a requirement. Many chemical agents also lack the ability to penetrate into minute cracks and crevices which may harbor microbial life.

Ionizing radiations penetrate materials and therefore are capable of sterilizing the interiors as well as the surfaces of objects. However, the radiation dosages required to kill microorganisms is of such magnitude that it is detrimental to various materials and components. Apparatus and facilities for the administration of radiation are quite large and costly; thus, the use of radiation as a means of sterilization is limited.

The most reliable and most widely used method of sterilization is heat. Pressurized steam is the principle source of heat for sterilization practices in the medical and industrial fields, but, since this method would not be suitable for spacecraft components, dry heat would be substituted.

Because of the reliability of heat to produce sterility when properly used, dry heat has been accepted as the method for the sterilization of planetary spacecraft. Based upon the work performed by Wilmot-Castle Company of Rochester, N.Y., under a NASA contract[4], the sterilization of flight hardware is to be accomplished by exposure to 135°C. in an atmosphere of dry nitrogen for a period of 24 hours after the most thermal resistant mass has equilibrated at the required temperature. For type approval testing, the hardware must be exposed to 145°C in a dry nitrogen atmosphere for 36 hours after proper equilibration. This exposure is to be carried out three different times with stabilization to room conditions between exposures. A specification for testing hardware compatibility to dry heat sterilization is now available.

Although dry heat is being specified as the primary sterilization method, the door should not be closed on the use of other procedures for applications where heat absolutely cannot be used. However, it is hoped that all spacecraft materials can be developed for thermal stability so that the entire spacecraft can be heat sterilized.

III. STUDY RESULTS AND CONCLUSIONS

Both the functional requirements of an orbiter/landing-capsule system and various technical approaches for meeting these functional requirements have been examined within this study. Several conceptual designs have been synthesized for the purpose of illustrating these technical alternatives and the problems associated with such a planetary spacecraft system.

The spacecraft system concepts have been examined in terms of a Mars orbiting photographic mission and a landing biological and photographic mission. The scientific instrument packages included in the system weight estimates contain only those experiments which are operable in orbit or on the surface of the planet.

The nominal orbiter instrument package can acquire the following scientific data at Mars:

1) Stereo photomosaic of approximately 25 percent of Martian surface at 0.5 km resolution plus a thermal profile (IR) map
2) Very high resolution photographs of selected areas within the preceding area coverage, with the final value of resolution determined by the capabilities of the instrument and spacecraft system, as well as the point (altitude) in the orbit from which pictures are taken.
3) Thermodynamic and compositional structure of the Martian atmosphere using complete spectroscopic techniques.
4) Mapping of any planetary field in the orbital plane.
5) Determination of the higher harmonics of the gravitational potential.

6) Mapping of trapped radiation belts in the orbital plane.
7) Survey of micrometeorite density in the vicinity of Mars.
8) Detection and analysis of emission and reflection properties of biological materials, if present, on Mars.
9) A study of the temporal variations of all of the above items, for up to 150 days.

The nominal landing instrument package can acquire the following data:

1) A comprehensive bilogical survey by employing photography, microscopy, light scattering, radioisotope techniques and biological reactions for specific biochemical components. Growth detection for periods in excess of 1 month.
2) Color surface photography of resolution varying from 1 mm at 2 m distance to 5 cm at 100 m distance.
3) A total of approximately 500 surface pictures (both surface photography and micro photography).
4) Geological information through techniques of seismometry, x-ray diffraction, and petrological microscopy.
5) Composition and meteorology of the atmosphere by measurement of thermodynamic parameters, surface conditions, surface wind velocity, and time variations of each of these.
6) A lifetime in excess of 3 months for any or all of the above experiments.

This type of mission capability is based on a combination of the spacecraft system concepts generated within this study, and the 210 ft diameter antenna available within the Deep Space Instrumentation Facility.

Two of the instruments in the nominal Mars package, the magnetometer and the cosmic dust or micrometeoroid detector, would also be useful during cruise to the planet. In addition to these two experiments, there are many other experiments that would investigate the properties of interplanetary space exclusively. These have not been considered in the present study since its primary purpose was to investigate problems associated with a planet orbiting and landing mission. However, the Voyager missions will undoubtedly include certain interplanetary experiments. The interplanetary experiments tend to place less demand on the spacecraft system in terms of required weight, power, and transmission bandwidth than the in-orbit photographic and capsule landing missions considered in the study.

The system concepts outlined in this report result in a typical orbiter weight of 1820 lbs, not including maneuver or orbit injection propulsion weights, and typical landing capsule weights of 1400 lbs to 1700 lbs. The Saturn S-I/SV vehicle injection capability coupled with launch and arrival trajectory restrictions and retro-propulsion requirements allows placing the orbiter system into a Mars orbit of 1800 km x 10,000 km altitude. Considerable contingency weight growth is available for either the orbiter or landing capsule because the orbit apoapsis altitude can be increased to as much as 40,000 km.

Summarizing the points discussed above leads to the following conclusions:

1) A Mars orbiter/landing-capsule mission launched by the Saturn I-B/SV vehicle in 1969 is both potentially attractive and conceptually possible and, hence, warrants further study of possible means of mechanizing such a mission in an adequately reliable manner.
2) The exact mission capabilities of this spacecraft system cannot be specified until integrated design studies and mission success analyses have been completed. However, typical mission capability of the system has been established.
3) Considerable margin in total spacecraft system weight is provided by varying the final orbital parameters of the orbiter portion of the mission.
4) Finalization of mission objectives in terms of orbiter photographic resolution and surface coverage requirements and in terms of lander photographic capability may best be accomplished toward the end of a subsequent "design study - mission success analysis." Such an effort will be required if the mission described herein is to become firmly scheduled within the NASA Planetary Exploration Program.
5) The problem areas uncovered during the present study are discussed in Chapter 6 of this report. Many of these problems must be attacked in the immediate future if the Mars 1969 opportunity is to be seriously considered. The problem areas are not reiterated here, but the reader is asked to review Chapter 6 of this report in detail.

REFERENCES

(1) G. F. Schilling, "Limiting Model Atmospheres of Mars," Rand report No. R-402-JPL, August 1962.

(2) L. D. Kaplan, "An Analysis of the Spectrum of Mars," Astrophysical Journal, January 1, 1964.

(3) "Proceedings of Conference on Spacecraft Sterilization," NASA Technical Note D-1357.

(4) "Sterilization of Space Probe Components," Final Report - NASA Contract No. NAS-r-31, April 19, 1961 - July 31, 1962.

CHAPTER 3

AN ORBITER/SPLIT - CAPSULE MISSION

I. MISSION DESCRIPTION

In this Chapter the orbiter/lander mission will first be described in general terms. The mission objectives will then be described in detail.

For the orbiter/lander mission, launching will occur from the Atlantic Mission Range (AMR) in early 1969. The Saturn S-I/SV will be employed with a standard shroud. Conventional southeast launch azimuths will be employed, and injection will occur (generally) off the east coast of South Africa. The flight time will be in excess of 250 days, and arrival at Mars will occur in 10/69-11/69.

The sequence from injection through midcourse maneuver(s) resembles that of the Mariner series of spacecraft. The spacecraft is separated and the launch vehicle is retarded by retro-rockets. Storage batteries supply power until Sun acquisition occurs, after which the principal axis of the spacecraft is pointed toward the Sun and a solar-energy collection and conversion system supplies power. Spacecraft systems requiring up to 900w of raw power have been considered which, at the distance of Mars, requires up to 360 ft^2 of solar panels.

Following solar acquisition, a Sun-Canopus sensing system is used for full 3-axes attitude control. At some time during the transit phase of the flight, before communication with the spacecraft omni-directional antenna becomes marginal, a 12 ft antenna radiating 25 w at 2300 Mc is pointed toward the Earth.

One or more midcourse maneuvers, based on radio tracking data, will be executed during transit. The midcourse propulsion system will, as in the Mariner systems, use storable liquid propellants; however, the thrust level will be increased to 1500 ft-lb. Configurations using a single motor on the centerline and several motors off the centerline have been examined. It is recognized that space on the spacecraft centerline is at a premium; also, several motors off the centerline can better accommodate variations in the center of gravity.

During the coasting flight, various measurements of the behavior of the spacecraft system and the nature of the interplanetary medium will be transmitted to Earth.

As the spacecraft approaches Mars, the approach guidance system is activated. Maneuvers may be based solely on Earth-based radio-tracking or on radio-tracking in combination with an on-board optical sensing system. In the latter system the angles between Mars and other visible objects such as the Sun and a star are measured, and a corrective maneuver is computed. The magnitude of a typical approach maneuver would be a few tens of meters per second, and it is thus comparable to, or possibly greater than, a typical midcourse maneuver. The computations required are not complex and could be performed in the spacecraft, on the Earth, or in some combination of the two.

The approach guidance system is such that maneuvers made enroute require small velocity increments but yield high dispersions at the target, while the converse holds for maneuvers made close to the target. It is generally economical to make more than one approach maneuver, and for this mission two maneuvers appear adequate. The first maneuver would be performed at a distance of one to two million kilometers from Mars, and the second at about one-half million kilometers. The resultant dispersion at the target would typically be a few hundred kilometers.

It is proposed that the orbiter/landing-capsule combination be placed on a trajectory which, if continued, would miss the planet at a distance corresponding to the periapsis of the nominal post-encounter orbit. The two approach maneuvers would, as required, correct the vehicle to such a trajectory. At some time after the first approach maneuver the landing capsule will be accelerated in a (generally) lateral direction by means of a "separation maneuver" so as to impact the planet at the desired landing area. Approximately 60 hours will elapse between separation and entry.

The capsule, which is completely sterilized, will be passive and will make an uncontrolled, aerodynamic entry into the atmosphere of Mars. The capsule will typically weigh on the order of 1400 to 1700 lb and will consist of structure, ablation and heat-protection material, a parachute for final descent, an impact absorption and erection device, a radio transmitter for establishing communication between capsule and Earth and between capsule and the orbiter/bus, a radioisotope power supply, data handling system, and scientific instruments.

As the capsule enters the atmosphere, its angular orientation will be more or less random and atmospheric forces will act to point the protected side of the capsule toward the surface. An acceleration up to about 100 Earth G's will be experienced briefly, and measurements of this acceleration, together with pressure and temperature, will be relayed to the orbiter/bus. When the capsule has slowed sufficiently, a parachute will be deployed from the rear of the capsule in order to yield a final vertical impact speed at the surface of under 40 ft/sec.

From separation to landing, the capsule is visible to the bus. Shortly after the capsule lands, the spacecraft bus will perform a final maneuver to place itself into a 1800 km x 10,000 km orbit about Mars. A few minutes later the orbiter, as seen from the capsule, will go below the horizon and will appear to set. Several hours later the orbiter will appear to rise and then set again. The exact times of rising and setting are strong functions of the impact point of the capsule and of the trajectory of the spacecraft. The orbiter will receive, store, and retransmit to the Earth the data obtained from the capsule both prior to and after landing. In addition, the landed capsule will transmit data at a low bit-rate directly to Earth.

The orbiter, of course, contains its own instruments for making scientific measurements of the planet. The basic orbiter attitude control will probably be derived from a Sun-star reference system, with inertial sensors used for short-time reference during periods of Sun and/or star occultation. A number of the planetary scientific instruments will be mounted on a gimbaled platform which can be pointed toward the planet independent of the basic orbiter orientation.

The capsule continues to perform measurements on the surface, some for a period of at least several months. Its lifetime is ultimately limited by the half-life of the isotope chosen for power generation. The orbiter continues its measurements for a nominal 150 days in orbit, so that seasonal changes on the surface may be observed. The lifetime limitation for the orbiter is determined by the amount of attitude-control gas which is carried aboard.

II. SPACE SCIENCE OBJECTIVES

A. Introduction

The purpose of this section is to provide scientific objectives and from these objectives, propose experiments for a Voyager spacecraft to be flown to Mars at the 1969 opportunity. The experiments provide a basis from which scientific instrument systems can be developed. This information was generated by a Voyager Scientific Objectives Study Committee of the Space Science Division.

The general scientific objectives for the mission are followed by specific objectives, classified by scientific disciplines or fields. Each objective is accompanied by background information, a statement of the importance of the objective and more definitive future objectives that might be formulated as a result of the initial experiments. The objectives are then summarized and assigned priorities. In the next section experiments are proposed to obtain part or all of the information desired in the objectives. In many cases, several different experiments on this and later spacecraft will be needed to fulfill a scientific objective.

The requirements for typical instrumentation to perform these experiments are presented in Chapter 4, II, B. Whereas different instruments for performing an experiment may vary in their outputs, there is a certain fundamental amount of information desired by each experiment. It can be seen in other sections of this document that the data handling system, the telemetry system and their power supplies constitute a major portion of the spacecraft weight. Therefore, once the principal scientific objectives and their first-order experiments are chosen, it is possible to make a reasonably good estimate of whether the spacecraft is capable of supporting the recommended scientific payload even though the actual flight instruments are not selected.

A preliminary estimate of the spacecraft system requirements needed to accommodate these instruments is included in Chapter 4. II. B as a basis for initiating the spacecraft conceptual design. Optimization of the design of the spacecraft to perform the required experiments within the technical constraints of the launch vehicle, DSIF, etc., will require design iteration and trade-off analysis between various spacecraft sub-systems. This analysis will probably modify the preliminary estimate of the recommended scientific payload included in this document.

B. Scientific Objectives

Three general scientific objectives have been selected for the Voyager missions to Mars. In order of importance, they are the following:

1) Exploration of the surface for evidence of life -- past, present, or precursors of future life;
2) Determination of the geological nature and history of the planet;
3) A continuation of interplanetary space physics research.

The above order was established from both the importance of the objectives and the effect of the landed capsule on experiments. A group of scientists under the auspices of the National Academy of Science (Ref 1) have already established the importance of life detection as part of the planetary exploration program. By its nature Mars offers a unique opportunity for detecting the existence and determining the characteristics of extraterrestrial life within the solar system. Because it is difficult to assure the sterility of the initially landed probes, biological measurements on later missions may not represent indigenous conditions. For these reasons, biological experiments must be given precedence over those concerned with other scientific disciplines.

An understanding of the geology of Mars is a necessary step toward deciphering the history of the solar system. The objectives are to obtain general topographic features as well as detailed geologic information. From data transmission and other standpoints, Voyager is the first spacecraft which will adequately support geological experiments on Mars. The size and detail of geological features necessarily require an extensive program continuing with later missions beyond 1969.

Certain of the interplanetary objectives are similar to those for previous missions in the vicinity of Mars. The proposed experiments are intended to be part of the measurements planned to extend over a full solar cycle. In addition, the larger spacecraft will permit more detailed experiments than in the past.

With the Voyager spacecraft, it is possible to perform more than just larger and more complex experiments. It permits an entirely new class of experiments to be flown because of the great increase in data transmission capacity. The most desirable early objectives for planetary exploration ask broad questions. By necessity, their experiments produce considerable data to be returned to earth. From the results of general exploratory experiments, future experiments can be planned more intelligently. Our understanding of the solar system will proceed more efficiently if the most appropriate experiments can be performed at the right time.

1. Biological Objectives

The biological objectives for Voyager missions to Mars are:

1) To obtain information on the presence or absence of an indigenous biota;
2) To examine the nature of the biota if it exists;
3) To examine the organic chemistry of the planet in conjunction with the detection of life and to obtain information on possible abiotic organic chemistry, in case life does not exist;
4) To obtain detailed information about the physical environment which affects the existence and nature of life.

Biologists are in general agreement that the search for extraterrestrial life is the most important objective for planetary exploration (Ref 1). Its discovery is relevant to the question of whether life can originate on any planet where conditions are favorable (Ref 2), either through an indigenous chemical evolution or through seeding with cosmic spores as hypothesized by Arrhenius (Ref 3). Experimental analysis of extraterrestrial life could reveal basic differences in chemistry, structure, and function and could broaden our fundamental concepts of life. If no basic differences are found, confidence in our present concepts will be augmented, and new knowledge may be gained about the potential variability and adaptability of familiar biochemical systems.

The philosophy for the development of the experimental program is based on concepts of life as derived from terrestrial experiences (Ref 4), current knowledge of the Martian environment, and theoretical ideas about the origin of life on the Earth. Although the environmental conditions on Mars appear far from optimal by Earth standards, seasonal changes in contrast and optical polarization of the maria (Ref 5) and the infrared spectra data of Sinton (Ref 6) strongly suggest biological activity.

Granting the existence of life, it is assumed that an ecological relationship exists within the Martian biosphere in which the trapping and utilization of solar energy are essential processes and the physical and chemical properties of both soil and atmosphere are important biological parameters. It is also assumed that some of the Martian organisms possess the general properties of morphological distinctiveness, metabolic activity, reproductive and growth capacity and a characteristic organic chemistry. However, the detailed properties of Martian organisms can be expected to differ from those of terrestrial organisms. The low atmospheric pressure, the apparent small concentration of water, and the wide diurnal temperature fluctuations suggest that organisms would have developed unusual adaptations to these environmental extremes which could be evidenced as unusual structural and functional properties.

In the search for extraterrestrial life, initial emphasis will be placed on the detection and study of microbiota. By analogy with terrestrial life, Mars should possess an abundant microflora which should be widely if not entirely distributed over the surface of the planet; therefore, the expected probability of recovering microorganisms from any random site on Mars is high. On the other hand, large forms of life are expected to be considerably less abundant and less widely distributed. The adaptability of micro-organisms to environmental extremes is very great and the expectation of finding micro-organisms where low temperatures, low atmospheric pressure and low abundance of water prevail is considerably higher than it is for higher organisms.

From further analogy with the terrestrial biosphere, the probability appears high that micro-organisms play a predominant role in the cycling of the elemental constituents of life. Therefore, it would follow that they exist in great abundance and are widely distributed on the planet.

In addition to the advantages of being highly abundant and widely distributed, micro-organisms will be easier to collect because they can be obtained from soil samples or atmospheric dust. The sampling of large organisms, however, would require complex mechanisms to seek out and capture biological samples.

a. Life Detection

(1) Objectives. The objectives of the life detection experiments are to establish the presence or absence of life by examining the Martian environment for properties typical of living organisms.

(2) Morphological Studies. Morphology, or the study of form and structure, is one approach commonly used by biologists to detect or distinguish biological organisms from inanimate objects. Visual inspection is usually sufficient and sometimes necessary to discern biological organisms which have characteristic symmetry, color, structure, shape and relative size. Although it is considerably less difficult to distinguish between large organisms and large inanimate objects than it is to distinguish between microscopic organisms and inanimate particles, the detection of micro-organisms by visual inspection may have a higher probability of success than the search for organisms larger than microscopic size because of their high abundance and wide distribution. To achieve the maximum probability of detecting life by visual methods, however, both the macro as well as the micro environment of Mars must be examined.

The purpose of morphological studies is to discern biological organisms from inanimate objects by visual inspection and to study the structure, form and color of these organisms.

(3) Metabolic Studies. Metabolism is a complex of integrated chemical reactions within living organisms which results in an exchange of matter and energy with the physical and chemical environment. The two most important metabolic processes of terrestrial life are photosynthesis and respiration. Photosynthesis consists of photochemical and oxidation-reduction reactions whereby carbon dioxide, water and light energy are utilized by living matter to produce molecular oxygen and energy-bearing reduced carbon compounds. Respiration consists of the reverse process in which oxygen serves an electron acceptor in the oxidation of reduced carbon compounds, providing living organisms with the energy required for their various functions. In anaerobic metabolism, or fermentation, where oxygen is not utilized, organic compounds at an intermediate reduced state, such as pyruvate, may serve as the electron acceptor for the oxidation of more highly reduced organic compounds.

Variations of these basic metabolic schemes occur in terrestrial microorganisms. In chemosynthetic species, CO_2 and H_2O may be converted to reduced organic compounds utilizing the chemical energy obtained from the oxidation of various reduced inorganic substances (e.g., ammonia, hydrogen, sulphide, ferrous and manganous ions are oxidized to nitrate, sulphate, ferric and manganic ions. Other micro-organisms are able to carry out these reactions in reverse, so that nitrate, sulphate and ferric ion under anaerobic conditions can be utilized as electron acceptors in oxidation-reduction reactions rather than molecular oxygen.

These examples suggest a range of metabolic variations which should be considered in Martian biological exploration. Because of the presumed low oxygen content of the Martian atmosphere, anaerobic as well as possible aerobic metabolic systems must be studied, including photochemical and chemosynthetic systems. Metabolic studies will investigate the presence of (1) photosynthesis or other photochemical activity in which solar energy is required by the living systems examined; (2) metabolic systems that oxidize or reduce inorganic compounds under aerobic or anaerobic conditions; (3) metabolic systems that oxydize or reduce organic compounds under aerobic or anaerobic conditions.

(4) Reproduction and Growth. Reproduction is the process by which living organisms multiply. Variations include both sexual and asexual reproduction. Sexual reproduction is generally limited to multi-cellular plants and animals, although simple sexual reproduction occurs in lower living forms, including micro-organisms. At the cellular level, asexual reproduction occurs either through the process of fission, as in bacteria, where a single individual divides into two by equi-partition of protoplasm and separation by a pinching-off process or, in more complex cells, by mitosis, whereby identical genetic structures from the nucleus are paired and divided equally before actual division of the cell occurs.

Whenever nutritional and environmental conditions are optimal, all organisms tend to reproduce at relatively constant rates so that their numerical increase is exponential with time. Thus, the basic growth rate equation is: $\frac{dN}{dt} = kN$, where k is the specific growth rate constant and N is the number of organisms. Unfavorable environmental conditions such as temperature or radiation extrmes, lack of water or nutrients or competition with other organisms modify growth rates so that a dynamic equilibrium is established in a particular environment. Such a dynamic equilibrium characterizes the Earth's biosphere.

The process of reproduction is important in the evolution of life because mutations, or changes in the genetic materials of organisms, occur at random resulting in change in structure and function. These changes are transmitted to all subsequent offspring in the species. Such mutation may make the species more or less adaptable to the environment. By natural selection changes which tend toward increased adaptation are preserved in the species. Those which tend toward less adaptation tend to be eliminated. The origin of the millions of diverse terrestrial species has occurred by the processes of mutation and natural selection.

Growth relates to the increase in size and mass of organisms due to their incorporation of structural compounds derived from their metabolic processes. A simple increase in size and mass, per se, is not a sufficient indication of growth (e.g., the swelling of cells due to water ingestion through osmotic pressure differences). Experiments designed to detect growth would normally have to be correlated with data from metabolic experiments for verification.

The detection of reproducing organisms would be one of the most convincing kinds of evidence for the presence of life. The first objective is to examine particles or objects from Martian soil or atmospheric samples for reproductive capacity. If this capability can be demonstrated, later experiments would be designed to determine whether the organisms divided by simple fission or exhibited processes similar to mitosis and whether reproduction was sexual or asexual.

In early experiments, the property of exponential increase may be used as a criterion in the detection of growth, e.g., the exponential accumulation of metabolic product such as carbon dioxide as may occur by culturing Martian micro-organisms.

Examination of samples for several kinds (or species) of organisms would suggest that the laws of mutation and natural selection were operative in Martian biology.

(5) Chemical Analysis. The major elements comprising the compounds in living systems are those which occur in greatest abundance in the cosmic distribution, namely, hydrogen, oxygen, nitrogen and carbon. These elements form a large variety of biochemical compounds of which several major classes are found universally in living systems. These are nucleic acids, proteins, carbohydrates and lipids. They comprise better than

90 percent of the dried material of living cells. The first three are polymerization products of nucleotides, amino acids, and sugars, respectively, while lipids are a complex class of compounds containing glycerols, fatty-acid esters and other hydrophobic compounds. Because these compounds are so universally distributed in terrestrial life, the possibility that they might be constituents in extraterrestrial life seems reasonable.

Polymerized macromolecular structures are essential for life. Practically all the chemical activity comprising living processes are dependent upon catalysis and the catalysts, or enzymes, in living organisms consist of high molecular weight proteins. Therefore, the demonstration of macromolecular structures and enzyme activity in Martian samples will be highly indicative of life. Common classes of enzymes include hydrolyzing enzymes, phosphorylases, dehydrogenases, oxidases, reductases, and isomerases.

A very important chemical property of biochemical compounds is optical activity. In terrestrial organisms, L-amino acids and D-sugars predominate and are completely essential. The antipodes of these isomers cannot be utilized in living processes. It is of great interest to determine if a predominant selectivity for optical isomers in Martian biology exists and if such selectivity is identical to or different from that in terrestrial life. The discovery of optically active organic compounds would be highly suggestive for the existence of life.

Active living protoplasm is approximately 80 percent water. Water is necessary in the metabolic activities of organisms, as a solvent and as a medium through which metabolic materials can be transported between different chemically active structures. It is difficult to imagine a life without water in the liquid phase. Because the abundance of water on Mars is so low, no liquid phase is thought to exist (Ref. 7). Martian organisms would probably require special mechanisms for obtaining water from the solid or gas phase and for storing it as a liquid within their structures.

(6) Biogenic Residues. A considerable fraction of the organic matter in terrestrial soil consists of excretion products of organisms or products derived from their decomposition after death (Ref. 8). Humus, peat, coal and petroleum are presumably the products of biogenic organic material which has undergone various biological and geochemical metamorphosis. Cellulose and lignin represent relatively stable organic residues from plants. Under anaerobic conditions these compounds or their derivatives may persist for long periods of time. In addition, inorganic end-products of organisms sometimes occur, e.g., ferric oxide, metallic iron, sulphur, sulphides, sulphates, silicates, and nitrates.

While it is impossible to predict what biogenic remains might exist on Mars, qualitative organic analysis directed towards identification of functional groups or general classes of organic compounds, e.g., saturated and unsaturated hydrocarbons, aromatics, alicyclic hydrocarbons, etc., would supplement the search for nucleic acids, proteins, lipids, carbohydrates, and porphyrins. Suggestive evidence for life would be obtained through this kind of indirect experiment.

The objectives for the examination of the chemistry of Mars for life detection are as follows:

1) To establish that an organic chemistry is present by determining the presence of organic carbon, nitrogen, oxygen and hydrogen;
2) To determine the presence or absence of the major classes of biochemical compounds;
3) To collect evidence for the existence of macromolecular structures and enzyme activity;
4) To determine the presence of optically active compounds;
5) To measure the water content of biological material, if found;
6) To attempt to detect the presence of organic residues which may be of biogenic origin.

b. Organic Analysis for Possible Abiogenic Compounds

Whether or not life can be recognized in the early missions, preliminary investigation of the organic chemistry of the planet is also important because of its relevance to current ideas of biochemical evolution. Several alternative conditions might determine the organic chemistry of a planet:

1) Chemical compounds which are part of an abiotic chemical evolution may be present;
2) An active biosphere might exist which has included most of the available organic matter;
3) It may retain organic remnants of a once active biosphere;
4) It might contain deposits of carbon compounds of abiogenic origin derived from planetary or extraplanetary sources;
5) It might have biogenic matter of extraplanetary origin;
6) All organic matter might have been oxidized to carbon dioxide at some period in the planet's history;
7) It might have some combination of the above.

According to current ideas on the origin of life (Ref. 9, 10), the relative abundances of hydrogen, nitrogen, oxygen and carbon during the formation of the planets would result in atmospheres consisting essentially of reduced nitrogen, oxygen and carbon (namely, ammonia, water, methane, hydrogen) because hydrogen would be in excess by a factor of about ten thousand. It has been demonstrated that simple biochemical species and possible precursors of complex biochemical compounds can be formed through the interaction of solar ultraviolet light or electric discharges in a reducing atmosphere (Ref. 11). Consequently, such prebiotic materials would be expected to form at some time early in the planet's history. A possible result of further chemical evolution would be the origin of life on the planet.

What alternative chemical evolutionary processes could have occurred on Mars if life did not arise is impossible to predict. The purpose of this phase of the investigation is to collect information on the organic chemistry of Mars relevant to possible abiotic origin. Chemical studies for life detection and abiogenic classes necessarily overlap. Future experiments will be planned on the basis of the results of early missions.

c. Ecological Studies

Ecology is the study of the interactions among organisms and their environment, including both the physical and biological surroundings. The activity of the biosphere consists of a continuous interchange of matter and energy which is expressed by a constant interaction between the biological and physical environment. Organisms have minimum, maximum and optimal tolerances to physical and chemical conditions in their surroundings and will persist or die depending on the types of environmental parameters and their intensities. Living organisms may also profoundly affect the physical environment, producing changes in the lithosphere, atmosphere and hydrosphere which could not occur without their intervention.

In investigating the physical conditions of Mars, factors known to be of greatest significance with respect to biology are to be examined. These are:

1) The prevailing temperatures of the lower atmosphere, soil surface, and immediate soil subsurface, and their diurnal variations;
2) The water content of atmospheres and soil and its physical state within the soil;
3) The intensity and spectral distribution of radiant energy at the surface;
4) The chemical composition and surface pressure of the atmosphere;
5) The mechanical properties of the soil and the distribution of biologically significant ions.

(1) Temperature. Temperature measurements of the soil and atmosphere will define the thermal limits to which any existing organism will be exposed. They will indicate the characteristics of an existing freeze-thaw cycle, which will be important in further defining the physical conditions to which the organisms must adapt. The measurements

will further help establish the probability of survival of contaminating terrestrial micro-organisms, and the data could be utilized in terrestrial laboratories to evaluate the danger of biological contamination with future space probes.

(2) Radiation. Radiation in the visible spectrum (4000-7000Å) activates photochemical reactions that are essential for introducing energy into the biosphere. The amount of solar energy captured is a function of incident energy absorbed by photosynthesizing organisms and their energy conversion efficiency under given environmental conditions.

Radiation in the near infrared range at wavelengths greater than about 7000-8000Å apparently has little or no biologically usable photochemical activity and is not generally absorbed by terrestrial plants. However, Tichoff (Ref. 12) has reported absorption by Martian maria in this region. Lichens found in the arctic regions as well as in cold, high environments also exhibit similar absorption. Tichoff speculates that absorption in the near infrared may be an adaptive phenomenon permitting such organisms to better maintain higher temperatures in cold environments.

Ultraviolet light between 2000-3000Å (Ref. 13) will destroy or injure living organisms at dosages above 10^4 ergs/cm^2. The exact dosages required to produce a given radiation effect will vary over a wide range depending upon the wavelength, species of organism, its metabolic state, and temperature. At wavelength 2537Å, dosages between 10^3-10^5 ergs/cm^2 (Ref. 14) are sufficient to kill most bacteria, while dosages of the order of 10^8-10^9 ergs/cm^2 are required to induce cancer in mice.

(3) Atmospheric Composition. All known terrestrial organisms actively metabolize or generate gases that are absorbed from or liberated into the earth's atmosphere. The present concentration of oxygen is regarded by some to be a result of the photosynthetic activity of the earth's biosphere (Ref. 15). The earth's atmosphere consists of N_2, O_2, H_2O, A, CO_2, Ne, He, CH_4, Kr, N_2O, H_2, O_3, Xe, NO_2, I_2, Rn. Of these gases N_2, O_2, CO_2, CH_4, H_2 are known metabolites for terrestrial life. Hydrogen occurs at concentration of 0.5 ppm; methane, 0.15 ppm; and CO_2, 300 ppm; and although H_2S and NH_3 are metabolites, they are not spectroscopically detectable in the earth's atmosphere.

By analogy with the terrestrial biosphere it would be expected that a qualitative analysis of the Martian atmosphere would suggest the types of metabolism that would be possible in organisms which utilized or evolved gases. Biologically significant gases may not appear in high concentrations; therefore, analysis for gases of expected low abundance is essential as well as the search for major gases. Measurements of both the partial and total pressure should be made.

The most important gases of biological interest for early missions are H_2O, CO_2, O_2, N_2, CH_4, H_2, NH_2, H_2S.

(4) **Soil Moisture, Chemistry and Mechanical Properties.** In soils the capacity to bind water is essentially a function of particle size and the concentration of soluble salts. Soil particles smaller than 1μ down to sizes approaching the size of water molecules can bind water with tensions of from 1 to 10,000 atmospheres. Soil consisting of very fine particles could strongly absorb atmospheric moisture at very low humidities. An interesting question concerns the capacity of Martian soil to absorb and hold atmospheric moisture or to trap moisture as a consequence of the expected freeze-thaw cycle. Therefore, the soil water content is very important in determining the nature of the soil-atmosphere moisture relationships. The mechanical properties, or the particle size distribution in soils is closely related to several biologically relevant environmental conditions. Because the particle size distribution is an important property associated with the moisture absorbing capacity of the soil, good estimation of soil moisture can be obtained if measurements of the mechanical properties are made in addition to the ionic composition, the temperatures of the soil and adjacent atmosphere, and the atmospheric moisture.

An estimation of bulk density and porosity can be derived from which thermal properties of the soil can be estimated, e.g., thermal diffusion rates and heat capacity. Data on moisture evaporation rates can also be derived since this parameter is a function of soil porosity, temperature, salt concentration, soil moisture concentration, the moisture binding capacity and atmospheric moisture concentration and temperature. Porosity will be indicative of gas diffusion rates in the soil which can be of importance to the utilization and exchange of atmospheric gases by soil organisms. For total organic content, see B, a, (2) and B, a, (5). As to inorganic ions, because of the importance of certain soluble ions in terrestrial life processes, the determination of their species and concentration in Martian soil is of considerable biological significance. The ionic properties of soil can impose restrictive environmental conditions on micro-organisms (or more highly evolved plants) due to their effects on pH and osmotic conditions or they may control moisture relationships between the soil and atmosphere. As previously discussed, some ions are utilized by soil micro-organisms in chemosynthetic processes and are also a source of elements which are necessary for the synthesis of important molecular structures such as porphyrins, diphospnopyridine nucleotide and adenosine triphosphate.

The determination of the kinds and concentrations of ions in Martian soil combined with data on mechanical properties, soil and surface temperatures, organic determinations and atmospheric composition and pressure will provide information from which deductions about the biology, or the possible existence of biology, can be made. Initial information of this kind will be used to plan further experimental programs.

In summary, the objectives of ecological studies are:

1) To obtain information on temperature extremes, temperature variations and thermal properties of Martian soil and atmosphere.

2) To examine the relationship between soil moisture and atmospheric moisture;

3) To determine the radiation environment of the surface;

4) To examine the atmosphere to determine its possible relationship to the biosphere;

5) To examine the soil chemistry to determine its possible relationship to the biosphere.

2. Geological Objectives

The specific objectives leading toward an understanding of the nature and history of Mars are discussed in the following paragraphs.

a. Topography

The chief objectives of a topographic survey by surface photography are the following:

1) The mapping of the geometry of surface features (relative to a surface spheroid) with detail down to about a meter. Surface geometry consists of the abundance, shape, and relief of features as a function of area and scale. Further, it describes the continuity of these features and the gross fabric they form relative to one another and to the rotational poles of Mars.

2) The division of the surface of Mars into units which appear to be homogeneous with respect to surface geometry, total and spectral reflectivity, size, and shape and which differ in one or more of these properties from adjacent units.

Another objective is to observe in much finer detail the features on the Martian surface now resolvable with telescopes. Some questions we can currently ask are what are the differences between light and dark areas and what is the geometry of the areas which undergo progressive reflectivity changes during Martian spring?

Photographs of the Martian surface will be interpreted for:

1) Modes and intensity of crustal deformation and associated inferences of internal activity.
2) Erosive and depositional processes that are currently modifying the Martian surface and, to a degree, an estimate of these processes in the past.
3) Intensity and types of volcanic activity.
4) Effects of impact on Martian surface.
5) Sequences of formation of surface features and consequent establishment of Martian time scale.

Of greater significance at this stage of exploration is the fact that surface photography will allow us to define the planetological problems of Mars. The present telescopic resolution of 80 Km. is insufficient for us to understand the Martian surface well enough to pose valid problems of the evolution of the planet. It is worth noting that the extensive theories and histories postulated for the moon by many investigators are based chiefly on our detailed knowledge of the moon's surface features and its mass.

b. Some Requirements for Photography

1) Wide areal coverage is much preferred over good resolution for initial surveys. Areal resolutions of 1/2 Km. are satisfactory for these flights. Higher resolution photography (2-10 meter) should be for coverage of small areas where critical relations are known to exist.
2) Stereo coverage is most important at all scales of areal resolution.
3) Concomitant measurement of spectral reflectivity would be valuable in characterizing and differentiating surface units on the photographs.
4) The most critical areas for photographic examination based on present knowledge are the bright-dark area contacts at latitudes around 30° S.

c. **Internal Structure and Activity**

The internal structure of Mars is basic in interpretation of the thermal history of the planet and in understanding the processes that have formed the surface features. Further knowledge of the internal structure is fundamental in interpretation of the shape and moments of inertia of the planet.

The density of Mars suggests that 10-15 percent free metal phases may exist, therefore, a basic problem is whether a metallic core exists in Mars or whether the interior is homogeneous and the density gradient is only a compressional one. For a heterogeneous interior, the depth of discontinuities should be determined and the average elastic wave velocities for each layer measured. The latter will suggest by comparison the kind of material composing the layer. It should be determined whether this is a core and if one exists, whether it is liquid or solid.

Variations in depth of any crustal layer should be examined. The correlation of this depth with surface topography and petrologic nature of surface rocks will lead to an understanding of isostatic adjustments in Mars.

A second major objective is to determine the degree of internal (seismic) activity in Mars. The frequency, depth, and areal distribution of Mars quakes should be determined as a function of energy release. Correlation of Mars quake distribution and the surface topography may provide clues to crustal structure and may suggest kinds of deformation occurring in the crust of Mars.

d. Petrologic Investigations

These investigations consist of working out the petrology of the units defined by photographic and other mapping techniques. This information will indicate the chemical heterogeneity of the surface and the types of rock-forming processes active on Mars, but most important, it will be our best information on the history of Mars.

Five types of petrologic measurements are discussed below, and are given in a priority based on requirements of prior knowledge for clear interpretation of a set of data. That is, knowledge of measurement 1 is required before measurement 2 is useful and knowledge of 1, 2 and 3 is required before 4 and 5 should be done. Use of these 5 measurements concomitantly will define the nature of Martian materials rather completely.

If not all these petrologic measurements can be made simultaneously, however, the higher priority ones should be done first; the later items should be held until they can be used on a Martian unit which had previously been examined.

(1) Mineral assemblage. Identification of the phases composing the specimen, their relative abundance, and their precise composition will define the composition of the system, and allow estimates of the thermodynamic variables under which the system evolved. The number of phases and types of phases may suggest whether the assemblage was in equilibrium during its formation and whether it is in equilibrium with the environment from which the specimen was taken.

(2) Texture. The geometric relations of the mineral phases in the assemblage indicate conditions during crystallization of the system and supply critical evidence on the sequence of formation of the phases and disequilibrium relations between phases.

(3) Composition. The abundance of major and minor elements in the assemblage is required to establish the minerology.

(4) Oxidation state. Determination of the ratios of $Fe^{0}/Fe^{+2}/Fe^{+3}$ and, if possible, Ti^{+3}/Ti^{+4}, and Mn^{+2}/Mn^{+3}. The oxygen pressure and associated water pressure are of critical importance in determining the course of differentiation in rock melts. Knowledge of the oxidation state will also suggest the approach to equilibrium of materials on the surface with the surface environment.

(5) Isotopic abundances. Consists of determination of abundance of stable and radioactive isotopes. This sort of experiment would be of most value once the gross nature of the material is well defined. The isotopic composition of certain elements may differ in general from those for the corresponding elements at the surface of the earth. These isotopic variations would be important clues to the origin of the solar system. In age determinations, however, it is especially important to have as much geologic information as possible to supplement and to facilitate the interpretation of the isotope data.

e. Structural and Temporal Relations of Rock Units

This consists of determining the structural relations of adjacent petrologic units as a measure of the way in which each unit originated and from this, further information on the nature of Martian crustal deformation, volcanicity, and modes of surface deposition. Of great importance is the sequence of formation of petrologic units during the history of Mars. When coupled with petrologic knowledge of each unit the temporal relations will allow interpretation of variance in external and internal processes on Mars during time and the variance in the surface environment. It will allow us to develop a Martian time scale.

Structural analysis will be chiefly by visual examination on scales from a few mm. to a km. Temporal analysis will be visual observation of sequences of layered rocks, by superimposed relations, and by use of radioactive dating. These temporal methods are subject to some uncertainties unless something of the petrology and structure of the unit is known.

f. Surface and Atmospheric Environment

This objective includes measurements of atmospheric composition (atomic, molecular, and isotopic), surface thermodynamic parameters, meteorological parameters, surface insolation, surface physical parameters and the dynamics of each of these. The general information derived from such measurements is an estimate of the conditions under which the planet has evolved. Some indication of weathering processes on the surface can be obtained from the pressure, temperature cycling, and chemical equilibrium calculations and then verified by direct observation. The presence and intensity of winds and dust storms also would allow a prediction of surface characteristics over large areas of the planet.

g. Shape

Because the mass of Mars is known to relatively high accuracy, a measurement of the physical dimensions will give a better measurement of average density. The shape of Mars is particularly interesting because of the apparent anomaly between the optical and the dynamical flattening. As yet, geologists have been unable to explain this discrepancy by reasonable density distributions for the planetary interior. A new determination of the optical flattening, using modern techniques, is warranted to determine whether the anomaly is real. Improved terrestrial observations are desirable. If the anomaly turns out to be real, then the determination of the shape would be a very high priority item. If new optical observations resolve the anomaly, then the determination of the shape would have a much lower priority.

h. Magnetic Field

The present theory of terrestrial magnetism ascribes the planetary magnetic field to the self exciting dynamo effect produced by a rotating liquid core of the planet. A measurement of the magnetic field of Mars combined with shape and seismic measurements will provide new data on the mechanism for generation of the secular magnetic field of planetary bodies.

A detailed, three-dimensional contour map of the planetary field should indicate internal and surface distribution of any ferromagnetic masses. In particular, if a magnetic field exists, it will be interesting to see the relationship between the magnetic and rotational polar axes.

i. Satellites of Mars

It would be of interest to determine the shape and density of these small bodies whose mass is unknown. These measurements would indicate whether the bodies are iron or silicate. Such observations would be our first contact with a small body in the solar system. It is important to determine the origin of these satellites and, specifically, whether or not they are captured asteroids. Interpretation of satellite observations would be considerably enhanced if one could also look at some asteroids.

j. Distribution of Heat Flux

The thermal conditions of the interior of the planet indicate its state of evolution and probable history. Measurement of the heat conducted out from the interior, in conjunction with composition and physical measurements of the surface and seismic studies of the interior, will produce a better understanding of the planet.

3. Interplanetary Research Objectives

The investigation of the interplanetary and interstellar medium is a large and interesting field of scientific research in itself. Interplanetary research is actually research on the nature and behavior of the sun, which has influence on terrestial weather, food and communications. In some instances it should be and, in fact, must be pursued by means of specifically interplanetary spacecraft, unconnected with the planetary program. At present, however, in the very early stages of the program, there is much that can be learned from experiments conducted on spacecraft which are constrained to go into the vicinity of some particular planet, and it has thus been decided that a considerable fraction of the interplanetary research program will be done in this way. There is an advantage in that some of the instruments will continue on an orbiter in the vicinity of the planet. A knowledge of fields and particles in the vicinity of Mars may have important consequences for interpretation of atmospheric measurements.

a. Magnetic Fields

The measurement of magnetic fields by magnetometers carried on or released from spacecraft can be expected to continue almost indefinitely into the future. There are two major problems--the planetary fields and the solar field.

The objectives of planetary magnetic field research are (1) to understand the origin and the mechanism of production of these fields in general, and of the geomagnetic field in particular, and (2) to understand the details of the interaction between the solar plasma and the magnetosphere. Measurements will be made of the intensity, orientation, and multipole character of the fields, and of the extent, configuration, and temporal variations of the magnetospheres.

The objectives of interplanetary magnetic measurements are:

1) To understand the nature of the general solar field by making frequent and protracted measurements of it throughout the inner solar system at all latitudes, all longitudes, and all times during the cycle of solar activity.
2) To elucidate the nature of magnetic disturbances on the solar surface and of the mechanism of accelerating and temporarily storing charged particles of high energy. Measurements of temporal and spatial variations of the field and their correlation with fluxes of charged particles or of electromagnetic radiation and with detectible solar phenomena are of interest here.
3) To determine the extent of the solar field, the nature of its boundary with the galactic field, and the variation of both of these with the solar activity cycle. This problem may require a whole series of extremely long-distance probes carrying extremely sensitive magnetometers.

b. Interplanetary Plasma

Because of the inherent nature of plasma physics, the study of the charged particles that make up the interplanetary plasma proper and the study of the magnetic fields that are associated with it, are really inseparable parts of the same problem. The two main goals are the investigation of the physical processes in the solar corona and of the basic plasma physics of the interplanetary medium. For the former, we shall wish to measure the intensity, extent, chemical composition, and temporal variations of solar-plasma streams, and to identify their sources. Measurements at all heliocentric latitudes and longitudes will eventually be required. For the latter, we shall look for interactions of the plasma with magnetic fields (both planetary and interplanetary), with solid bodies, with comet tails, and with other clouds of plasma. Wave motion and plasma instabilities will also be of interest. The position and nature of the transition between "supersonic" and "subsonic" flow in the solar wind, and the phenomena occurring at the boundary between the solar and galactic fields should be investigated. Other questions associated with the physics of the plasmas are the source of the Van Allen belt particles, the nature of the mechanism of their injection into the magnetosphere, and the detailed nature of geomagnetic disturbances.

c. Galactic Cosmic Rays

In the last few years, it has become clear that the cosmic rays that come near the earth are not a truly representative sample of those that exist in interstellar space. Thus, the major problems of modern cosmic-ray research are:

1) What are the flux, energy spectrum, composition, and anisotrophy of the interstellar cosmic rays
2) What are the nature and the location of the modulating mechanisms which alter the cosmic-ray fluxes at the earth in response to solar flares and to the general level of solar activity

To answer these questions will probably require fairly detailed measurements of cosmic rays out to great distances from the sun over a period of years. The measurements should include total flux, directional flux, energy spectra, and composition.

d. Solar Cosmic Rays

As has been mentioned already in more than one context, the nature of the accelerating and trapping mechanisms of high-energy particles is one of the major problems of solar physics. Detailed measurements of solar-accelerated particles, similar to those outlined above for galactic cosmic rays, are required. An important objective in this connection is to develop the capability to predict large solar-proton events with at least some degree of certainty. The nature of the phenomenon which causes the solar proton fluxes to become isotropic is the major facet of the problem.

e. Trapped Radiation

The magnetically trapped charged particles around other bodies in the solar system are of interest, both in connection with the possibility of sending men to them, and for the additional understanding that they will provide concerning our own Van Allen belts. As is now being done with earth satellites, we shall wish to determine intensity, extent, composition, temporal variations, and relations to solar events.

f. Neutral Gas

The non-ionized portion of the interplanetary medium is believed to be a small fraction of the total, at least in the inner solar system. Determination of its density and directional flux will give an important confirmation of our picture of the solar wind.

g. Dust

The study of particulate matter in space is a major objective of the program because of the light which it should shed on the problem of solar-system evolution. Although our early experiments in this field can afford to be (and of necessity must be) rather simple, with the primary objective of determining the effect of the dust on the spacecraft, later experiments will include size distribution, spatial distribution, orbital parameters, density, composition, and electric charge.

4. Summary and Priority

At a fully staffed meeting of the Voyager Scientific Objectives Study Committee of the Space Science Division, there was a thorough discussion of science objectives priorities. All of the scientific disciplines involved in the preparation of this section were present with Division 32 representatives who understood the capabilities and limitations of Voyager. At this meeting, the following list of priorities was prepared (see Table 6-1).

Type of Experiment		Priority
Biologic		
Detection of Life		1
1. Morphology		
2. Metabolism		
3. Reproduction and growth		
4. Characteristic chemistry		
Biosphere Examination		2
Geologic		
Topography (Surface Photography)		3
Structure of the Interior Seismic Activity		4
Identification of Rock Units (Petrologic Investigations)		
1. Mineral assemblages	Textural	5
2. Textural relations		
3. Composition (major and minor elements)		
4. Oxidation states		
5. Isotopic abundances		
Structural and Temporal Relations of Rock Units		6
1. Stratigraphy (photography)		
2. Isotopic dating		
Surface and Atmospheric Environment		
Shape		
Magnetic Field		
Satellites		
Distribution of Heat Flux		
Interplanetary Physics		7
Magnetic Fields		
Interplanetary Plasma		
Galactic Cosmic Rays		
Solar Cosmic Rays		
Trapped Radiation		
Neutral Gas		
Dust		

C. Experiments

The following types of experiments are appropriate for accomplishing the scientific objectives outlined in part 3. The experimental goals, techniques, and requirements are specified and typical instrumentation that is presently in some stage of development is mentioned. For particularly complex experiments whose interactions with the entire spacecraft system are large, such as orbiting stereophotography, a more extensive discussion is made of several possible ways to mechanize the experimental system. The discussion outline is broken down to separate capsule and orbiter experiments and then, under each of these, a further breakdown indicates the major scientific discipline to which the experiment applies. A few experiments, such as surface photography, are applicable to two or more disciplines and these are discussed under a common general heading. More experiments are presented than can be flown on a single Voyager mission.

1. Capsule Experiments

a. Biology

The experimental program presented for biology covers missions considerably beyond 1969. Although it may be very difficult to design instruments to perform some of these experiments, they have been presented as methods of achieving the scientific objectives. The selected experiments for 1969 are listed in part D, 1, a of this section.

b. Surface Photography and Photomicrography

Visual scan from macroscopic to microscopic ranges to discriminate between living and non-living objects will be accomplished. Size, form, symmetry, pigmentation, optical density, surface characteristics, internal and external structural complexity, as well as changes due to movement and growth will be observed. Two optical systems will be used, one to examine the landscape for objects larger than one millimeter in diameter, the other to examine sub-millimeter particles by microscopy.

1) Surface photography
   - a) Vertical coverage: Immediate vicinity of the capsule to the horizon; ranges 0-5 meters (resolution 1 mm); 5-25 meters (resolution 0.5 cm); 25-100 meters (resolution 5 cm); 100 meters - horizon.
   - b) Horizontal coverage: Coverage of each vertical range towards north and towards south and either east or west, depending upon the position of the Sun. Coverage of the surface in each direction with the Sun at the zenith. Horizontal angle to match the vertical (see a).

c) Color filters: Red, green, blue and near infra-red (0.7-1 micron).

2) Photomicrography. Photographs of microscopic particles retrieved from the surface and processed.

a) Size ranges: one millimeter to 100 microns diameter (resolution 10 microns); 100 to 20 microns (resolution 2 microns); 20 to 1 microns (resolution 0.5 micron). The particles will be processed and the size ranges separated for observation.

b) Illumination: Phase contract, ultra violet (2600Å-3000Å) and dark field microfluorescence.

c) Chemical and biological stains to discriminate among specific parts of organisms.

c. Metabolic Activity

Experiments to detect basic metabolic processes of respiration, photosynthesis and fermentation will be performed on soil samples from the surface and particles from the atmosphere. Samples collected will be placed in reaction vessels and monitored with suitable detectors.

1) Detection of overall complex of metabolic reactions. Accumulation of end-products or the disappearance of nutrients will be determined. Variations will be introduced into these experiments to examine aerobic and anaerobic metabolish, optical isomeric specificity, and optimal activity by adjustments in pH, temperature and concentration of substrates. These experiments will indicate the existence of a metabolic pathway without fully characterizing it, e.g., detection of CO_2 resulting from respiration by viable organisms.

2) Detection of changes in oxidation-reduction potential, hydrogen ion concentration and osmotic pressure of a nutrient solution resulting from biological activity.

3) Detection of unique and specific biochemical reactions. Fractionated soil samples will be tested for enzymatic activity using prepared substrates and concentrating on the more enzymatic reactions. In other kinds of experiments, immunological substances, co-factors and hormones will be detected by specified biological reactions.

4) Detection of accumulating intermediary substances of metabolic pathways using specific metabolic inhibitors.

d. Reproduction and Growth

Experiments will be performed to detect the increase of biological mass or an increase in the number of microorganisms taken from a soil sample. Populations will be grown under suitable conditions.

1) Exponential increase of metabolic activity or optical absorbency. These will indicate an increase in the numbers of individuals, an increase in the mass due to the growth of individuals, or a combination of these. The mass or numbers will be monitored for a period (2-15 days) sufficient to observe detectable changes. Depending upon the initial number of organisms or the initial mass, six or more population doublings will be required for detection. Based upon this data, the growth rates of the organisms could then be determined.
2) Determination of net synthesis of biochemical substances. Proteins and/or nucleic acids (organisms need not be intact for analysis) will be determined.
3) Direct observation of individual cell division. Cells will be observed for kind of division and for species differences. This could only be done during a long period of observation.

e. Characteristic Chemistry

Since the chemistry of biological organisms is a specific kind of organic chemistry, suitable chemical analyses will be carried out on materials taken from the Martian surface and atmosphere. These analyses will constitute an extremely sensitive detection method and also permit some reconstruction of the nature of the organisms. Chemical analyses will also be extended to investigate the abiogenic planetary chemistry with emphasis on organic substances.

1) Detection of organic matter. An important first-order experiment would be the detection of organic matter.
2) Detection of water. Quantitative analysis of biologically trapped water and water existing within inanimate substances will be carried out.
3) Determination of biologically important elements. These will be determined quantitatively including carbon, hydrogen, oxygen, nitrogen, sulfur and phosphorus. Analyses will be carried out on fractionated samples.
4) Characterization of biologically important classes of compounds.
   a) Analyses for compounds among four major classes of biochemically active substances of low molecular weight, viz., carbohydrates, amino acids, nucleosides and lipids, will be performed.
   b) Detection of high molecular weight macromolecules will be accomplished. Chemical and functional tests as well as analysis of degradation products will be carried out. Examples include susceptibility to hydrolysis, fluorescent activity, activity in electric or magnetic fields, chromatographic behavior, optical rotatory dispersion characteristics, and nuclear magnetic resonance behavior.
5) Detection of end-products of reactions. These include oxidation, reduction, pyrolysis and hydrolysis products.

6) Functional group analysis will be made for specific chemical groupings. Groups such as carbonyl, amino, alefin, aryl, thio and phosphoryl will be included in the tests.
7) Special tests to detect the presence of particularly stable compounds found as residues or in the structural components of organisms. These may include lignin, cellulose, chitin, and various mucopolysaccharides, mucoproteins, lipoproteins, etc.

f. Physical Parameters

Organisms are intimately associated with their environment. Insight into the existence and nature of Martian biology will be gained by the determination of the kinds of environment indigenous to the planet and available to the organisms. Five important parameters will be measured: temperature, soil moisture, availability of essential elements, surface radiation and the physical characteristics of the soil.

1) Temperature. Measurements will be made on the surface, 50 cm below the surface, and 2 meters above the surface. Diurnal variations will be determined by observing hourly measurements for prolonged periods. This will be done for the surface, subsurface and suprasurface measurements.
2) Soil moisture. Measurements will be made of both the biologically available and nonavailable water by performing water binding capacity experiments. These measurements will be made on the surface and 50 cm below the surface. Measurements will also be carried out at hourly intervals to establish diurnal variations.
3) Radiation flux. Photometric measurements will be made hourly of the radiation flux of wavelengths between 2000Å and 3000Å. Intensities of from 10^4 ergs/cm^2 down to 1 erg/cm^2 will be measured.
4) Atmospheric Composition. Determination of the major gaseous components of the atmosphere and their partial pressures will be carried out. Analyses for 0_2, CO_2, H_2O, N_2, H_2, CH_4, NH_3, and H_2S will be performed. Required limits of accuracy will be ±20 percent.
5) Physical characteristics of the soil. Measurements of salt concentrations, pH and Eh values will be made. Determinations of the mineral character, size distribution, and porosity of the soil particles will be made.

2. Geology

a. Surface Photography

A photographic system on the capsule should have the following capabilities:

1) Resolution near the landing spot of a few mm.
2) 360° lateral coverage and vertical angle capability of (-50°) to (+10°) relative to the horizon.
3) Knowledge of the local vertical.
4) Stereocoverage of at least 1/4 of the observed area so that a size and distance scale can be assigned to observed features.

The object of capsule photography is to observe the structural and temporal relations of the rock unit upon which the capsule has landed and is performing petrological experiments. Further, photos will allow an understanding of constructive and distinctive surface processes on Mars.

b. Seismology

Three general types of seismic experiments can be used. The ability of each type to investigate Martian internal structure and activity is given below in order of increasing complexity.

1) Short-period single axis seismometer
   a) Determine presence of crust (analogous to Earth's crust)
   b) Estimates of seismic energy release.
   c) Presence of deep or shallow focus quakes.
   d) Presence or absence of a solid or liquid core.*
   e) Correlation of quakes with topographic or geologic features.*
   f) Determine velocity-depth function for planet and hence infer density distribution with depth.
2) Long-period three-axis seismometer
   a) Same as above with greater degree of success because azimuth of approach of seismic waves can be easily determined, hence locations determined and velocity determined since travel time is known.
   b) Longer period surface waves will be recorded which "feel" a greater depth into the interior. Surface waves effectively "feel" or sample of a depth equal to 1/3 their wavelength. $V = f\lambda$

   $$VT = \lambda$$

   A 300 second Rayleight wave traveling 5 Km/Sec on the Moon would sample 1/3 x 5 x 300 or 500 Km so increasing the long-period response of a seismometer enables it to record surface waves which have sampled deeper.

*Dependent on whether enough disturbances at different quake to recorder distances will be recorded.

c) Possible recording of free oscillations of a planet.

3) More than one three-axis seismometer

a) Everything above can easily and definitely be determined because we are effectively using triangulation to determine locations and so velocity-depth relation can easily be determined.

C. Petrology

The study of the nature and evolution of rock units. Typical instruments are:

1) Mineral Assemblages: X-ray Diffraction: Infra-red Spectroscopy: Microscopy.

2) Textural Analysis: Visual microscopic examination of mounts of granulated material (or, if possible, thin sections of rocks).

3) Composition: X-ray spectroscopy; Visible emission and absorption spect.; Mass. spec.; α ; proton; neutron scattering; Neutron, proton activation; wet chemical analysis and others.

4) Oxidation States: Chemical analysis of $Fe^{\circ}/Fe^{2}Fe^{3}$.

5) Isotopes: Mass spectroscopy α, β, γ spectroscopy.

Study of clouds by direct measurement and spectral transmission.

1) Meteorological and climatic studies by long-time surface measurements.

1. Atmosphere

1) Study of the height profiles of thermodynamic variables by direct measurement.

2) Study of molecular, atomic, and isotopic composition by direct measurement measurement.
3) Study of clouds by direct measurement and spectral transmission.
4) Study of suspended particulate matter by direct measurement.
5) Meteorological and climatic studies by long time surface measurements.

2. Orbiter Experiments

a. Atmospheric

1) Study of molecular, atomic, and isotopic composition and of thermal structure by infrared absorption and emission.
2) Study of atomic and molecular composition by ultraviolet absorption and emission.
3) Study of cloud patterns using visual, infrared, and ultraviolet photography.

b. Geology and Environment

(1) Topographic studies using photography. It is necessary to have some idea of vertical relief in order to extract useful information from pictures of a scene for which no prior knowledge is available. Some features of the Martian terrain will be similar to terrestrial topography and to some degree can be understood using monoscopic techniques; however, stereophotography will provide the only means to adequately answer critical topographical questions. Several ways of producing the stereophotographic effect are possible using an orbiter. Shadow stereo using a monoscopic system can be done only if there is a way to distinguish shadows from surface coloration and if the geometry of the sun, camera, and object are known. A single framing camera system can produce a stereo through scene overlap either by successive pictures or by pictures from successive orbits. The best vertical resolution occurs when the base height ratio is unity. A system using two cameras looking forward and aft along the suborbital path probably is the optimum system. To assure the success of this experiment the whole orbiter system must be designed with the requirements of the photographic experiments being paramount considerations, since so many subsystems affect the performance directly. The orbital parameters are critical and, generally, pre-specified values must be met for orbit altitude, orbit inclination, circularity, and sun-orbit angle. Attitude control of the spacecraft must provide pointing to the planet vertical and roll control about this axis in addition to Earth and possible Sun acquisition. The communication system is usually sized to fit the information requirements of the visual system since requirements for non-visual information is usually negligible in comparison.

The experiment suggested here is stereo coverage of as large an area of the planetary surface as possible to a resolution of 500 meters. Eventually it will be desirable to increase stereo resolution by at least an order of magnitude for selected areas of the planetary surface. The latter experiment is not suggested at this time because spacecraft capability is inadequate to accomplish it. In lieu of this experiment, a series of nested photographs should be taken from the descending lander. Surface temperature mapping using microwave and infrared radiometry. Microwave and infrared emissivity studies. Surface composition studies using selective reflection, polarimetry, and photometry.

c. Biological

1) Search for organic bonds using infrared absorption and emission spectra.
2) Photographic search for evidence of life.

d. Field and Particles

1) Three-dimensional magnetic survey.
2) Investigation of potential trapped radiation belts.
3) Investigation of increased concentration of cosmic dust near the planet.

e. Satellites

Photographic investigation of shape and size.

5. Interplanetary experiments

The interplanetary experiments are discussed with their objectives.

D. Recommended Scientific Payload

The following experiments are recommended for inclusion in the conceptual spacecraft design of a Voyager capsule/orbiter mission to Mars presently being undertaken by the Advanced Planetary Spacecraft Study Committee. This recommendation is based upon a study made by the Voyager Scientific Objectives Study Committee of the Space Sciences Division whose conclusions are summarized in part 2 of this document. The experiments are chosen from the list outlined in part 3 and are consistent with the priorities shown in paragraph B, 4.

1. Capsule Experiments

a. Biology

Use of surface photography, microscopy, light scattering, radioisotope techniques, and biological reactions for specific biochemical compounds is recommended as a means for direct life detection. Organic chemistry of the surface surroundings should be done using gas chromatographic techniques. The surface ultraviolet flux should be measured by an integrating photometer.

b. Geology

Three specifically geological experiments are recommended. These use the techniques of seismometry, x-ray diffraction, and petrological microscopy. Photographic surface surveillance is also included and will contribute geologic information.

c. Atmospheric

The composition and meteorology of the atmosphere should be studied. Meteorology includes profiles of thermodynamic parameters, surface conditions, surface wind velocity, and time variations of each of these.

2. Orbiter Experiments

a. Atmospheric

Study of the composition and thermal structure of the atmosphere and its variations over the planetary sphere is recommended using infrared and ultraviolet experiments. Atmospheric phenomena of aurora and dayglow using ultraviolet techniques should be included.

b. Surface

A topographic map of most of the planetary surface should be made at 500 meters resolution. The highest possible resolution at selected points should be done stereoscopically. A thermal map of the surface will be obtained.

c. Orbit Altitude Area

The payload should include a magnetometer, a micrometeoroid detector and instrumentation to investigate trapped radiation.

3. Interplanetary Experiments

Instruments appropriate for investigating plasma, magnetic fields, and cosmic rays between the Earth and Mars should be included.

E. Capsule Landing Location

It is presently believed that dark areas provide the best chance for detecting Martian life. These areas change in shape and darkness with the Martian seasons. It is conceivable that such changes could be caused by active life on the surface. Among the dark areas the best region for landing appears to be Syrtis Major. This is a large, constantly dark region slightly above the equator. An undesirable feature is that there are light areas on either side to the east and west, and these remain light at all times. There is a considerably larger band of dark regions near Syrtis Major but south of the equator, which should be almost as suitable for landing. Here there are more changes in shading with the seasons than within Syrtis Major itself. For the initial landings, either of these regions would be satisfactory for both the life detection and geological experiments. Eventually, a more specific landing region will be desired and can be chosen from information gained in the early orbital surveys. For instance, geologists presently would like to land at a bright-dark boundary region such as that between Syrtis Major and Aeria.

The aiming points for the areas selected will be at their centers. Syrtis Major is a triangular-shaped area with corners at 15 deg N, 283 deg E, 10 deg S, 283 deg E, and 10 deg S, 308 deg E. The larger band of dark regions also recommended as a landing location is between 0 deg and 30 deg S latitude and between 220 deg and 310 deg E longitude.

F. Capsule Lifetime Requirements

The scientific objectives of biology and seismology, plus the requirement to measure diurnal variations in other experiment categories, will determine the lifetime requirements of a Voyager capsule. The communication capability, which affects the length of time necessary to transmit sufficient data, will also exert a strong influence.

1. Biology

Certain biological experiments require long periods of continuous operation. With the exception of growth or metabolic experiments, the biological life detection experiments can be performed within an hour or so after capsule landing. The morphological, chemical analytical and physical experiments are dependent only on the time of the operative procedures. The metabolic experiment is dependent on the time required for adequate biochemical or physical changes to take place to permit detection. The growth experiment is dependent on the time required for sufficient reproduction of the extraterrestrial organisms to permit their detection. In both cases the organisms must be maintained in controlled environments during the entire course of the experiment.

Using terrestrial organisms as the model, two factors affect the duration:

1) Supplying a growth environment that will permit the organisms to metabolize and to reproduce rapidly.
2) Using a sensitive system to detect the metabolism or reproduction with as few cell divisions as possible.

Considering the possibility of major advancements in detection techniques and optimum conditions, a collection of tens of terrestrial organisms might be expected to be detected after six or seven divisions (10^3 to 10^4 organisms). Some rapidly dividing organisms might then be detected in 6 to 12 hours; however, even under suitable conditions, the majority of species would require several days for detection. The uncertainty about optimum constituents and conditions of the growth media and of the growth rates of extraterrestrial life suggests that considerably longer periods of time are likely to be required for a successful extraterrestrial growth experiment. In order to draw suitable conclusions of the success or failure of a growth experiment, it is reasonable to prepare the growth experiment for a duration of at least two to three weeks. A failure after this period would indicate that the media is incorrect, the biology is quite different, or the division rate is considerably slower than was anticipated.

2. Diurnal Variations

The tentative payload consists of experiments whose primary purposes are to measure the diurnal variations on the Martian surface of temperaure, pressure, wind velocity, sounds, and ultra-violet flux. The length of time required to perform each of these experiments is different. Immediately after landing and sensor activation, one point on the curve can be drawn for each phenomenon of interest. The value of these measurements lies in two areas; the short term value and the average value. Therefore, in order to obtain satisfactory statistical information, the phenomena should be measured for as long a period as possible.

3. Seismology

Within the scope of the present knowledge and assumptions concerning the activity of the Martian crust and core, it can be stated that the operating time required for a seismograph will be minimum of one month. This instrument will be sampled continuously during this period.

III. SPACECRAFT TECHNOLOGY OBJECTIVES

The primary technological objective is the accomplishment of the above scientific objectives with a high probability of success. Secondary objectives include the development of orbit, entry, landing, and post-landing operational techniques in preparation particularly for the large lander missions expected in the advanced Voyager projects and, also, for the techniques expected to be involved in future manned flights to the planets.

A discussion of the possible steps that might be involved in the technological progression towards these goals is included in Section III-B, C, and D of Volume II of this report.

REFERENCES

1. A review of Space Research, Chap. 9, NAS-NRC, Publ. No. 1079.
2. Horowitz, N., Fed. Proc. 21, 4, part 1, August 1962.
3. Arrhenius, S., Worlds in the Making, New York: Harper, 1908.
4. Lederberg, J. Science, 132:3424, 393-400, August 12, 1960.
5. Dollfus, A., Proc. Lunar and Planet. Colloq., II (3), North American Aviation, Downey, California, August 15, 1961.
6. Sinton, W.M., Science 130:1234, 1959.
7. Dollfus, A., in Planets and Satellites, The Solar System, Vol. III, p. 388, ed. by G. Kuiper, University of Chicago Press, 1961.
8. Waksman, S.A., Soil Microbiology, Chap. 4, John Wiley & Sons, Inc., New York.
9. Oparin, A.I., The Origin of Life on the Earth, Academic Press, 1957.
10. Urey, H.C., The Planets, Yale University Press: New Haven, 1952.
11. Miller, S. & Urey, H.C., Science 130, 245, 1959.
12. Tichoff, G.A., in Principal Works of Astrobotany and Astrophysics, 1912-1957., p. 24-134. OTS: 60-11, 367, March 11, 1960.
13. Zelle, M.R. & Hollaender, A., in Radiation Biology, Vol. II Chap. 10, ed. by A. Hollaender, McGraw-Hill: New York, 1955.
14. Rentschler, H.C., Nagy, R. & Mouromseeff, G.J., Bact., 41:6, 745-774, June 1941.
15. Hutchinson, G.E., in The Earth as a Planet, The Solar System, Vol. II, Chap. 8, ed. by G. Kuiper, University of Chicago Press, 1961.

CHAPTER 4

SPACECRAFT SYSTEMS

SECTION I

MISSION PHASES

The Mars orbiter/split-capsule missions consist of six phases: acquisition and extension; cruise; trajectory correction maneuvers; capsule separation, atmosphere entry, landing, and surface operation; planetary orbit injection; and planetary orbit.

A. ACQUISITION AND EXTENSION

This phase begins with separation of the spacecraft from the booster and continues until the spacecraft is operating in the normal cruise mode. Spacecraft members such as solar panels, radiators, directional antennas, etc., are extended or erected and attitude references are acquired. Range-rate and range data are obtained by tracking the spacecraft coherent transponder.

B. CRUISE

The cruise phase begins after acquisition and articulation of the spacecraft and continues for a period of months until the planet is approached. Short interruptions of this phase occur for performance of trajectory correction maneuvers. After each maneuver, cruise is initiated by reacquisition of the attitude reference bodies. During cruise both tracking and telemetry data are recorded to provide updating of the spacecraft trajectory predictions and engineering data on the operation of the spacecraft system. The process of making guidance observations is performed during the cruise phase.

C. TRAJECTORY CORRECTION MANEUVERS

The correction of injection inaccuracies is required in order to secure the desired arrival conditions at the planet. The desired accuracy may be achieved by several different combinations of midcourse, approach, and orbital trim maneuvers performed at various times during the mission. Thus, this phase is of relatively short duration and occurs more than once during the mission. Each correction maneuver includes the following events:

1) During the cruise phase preceding the maneuver, a set of guidance observations are made by Earth-based radio tracking devices and/or onboard optical and radio devices.

2) These observations are processed in a computer, located either on the Earth or aboard the spacecraft, to determine a set of maneuver commands which will cause a rocket motor to be fired in the desired direction for the correct change in velocity.
3) The computed set of commands are relayed to the spacecraft sequencing equipment for storage and sequential execution.
4) The spacecraft reorients itself and fires the correction motor according to the stored set of commands.
5) The spacecraft begins reacquisition of its attitude references and resumes the cruise mode.

D. CAPSULE SEPARATION, ATMOSPHERIC ENTRY, LANDING, AND SURFACE OPERATION

At capsule separation the spacecraft performs an orientation maneuver as described in the preceding paragraph. After separation from the bus, the capsule is given a velocity increment to place it on an impact trajectory with the planet. Ballistic atmospheric entry of the capsule occurs several days later, final retardation occurs by deploying several parachutes which allow the capsule to descend slowly to the surface. Experiments are performed during this descent. Following landing on the surface, the lander is erected to a preferred attitude for operation of its equipment for a period of a few days up to many months.

E. PLANETARY ORBIT INJECTION

The events which occur during this mission phase are not grossly different from those occurring during a trajectory correction maneuver, i.e., observations are made, appropriate maneuver commands are computed and stored, and the bus reorients itself and fires a rocket motor according to the stored commands. The velocity increment supplied by the motor is planned so that injection into the desired planetary orbit will occur. This maneuver is identified as a separate mission phase because (1) the velocity increment required is several orders of magnitude greater than that of the trajectory correction maneuvers, and (2) the time at which the maneuver must be performed is relatively fixed.

F. PLANETARY ORBIT

During this mission phase the spacecraft is in orbit about the target planet. Planetary measurements are made and the resulting data is telemetered to Earth. Tracking and engineering telemetry giving data on the orbit and the status of the orbiter are continued during this portion of the mission. The duration of this phase may be a few days or many months, depending on the detailed objectives of the mission and the capability of the orbiter system.

SECTION II

TECHNICAL AREAS

A. INTRODUCTION

The functional requirements of the orbiter/split-capsule mission may be met by a number of alternate technical approaches. In the technical areas which follow, many alternatives are described, their relative advantages and disadvantages examined, and tentative recommendations made.

B. TRANSIT TRAJECTORIES

1. Introduction

In EPD-139, Volume II, allowable orbiter masses are given for missions from 1966-1972 to Mars and Venus for a periapsis altitude of 1000 kilometers (km) above the target planet using the Saturn S-I/SV as the launch booster. Also given are probable constraints which will limit the use of optimum trajectories for maximum orbiter weights. These constraints included the restriction that the declination of the outgoing geocentric asymptote lie between about ±33 deg (Atlantic Missile Range range safety considerations without dog-leg maneuvers), and that the angle between the hyperbolic excess velocity at the planet and the Mars-Sun vector be less than 60 deg or greater than 120 deg to provide proper lighting for scientific experiments (referred to as the lighting constraint). This approach constraint applies mainly to the high-inclination (greater than 60 deg) orbits which are currently considered the most desirable.

In the subsequent paragraphs, allowable orbiter and capsule masses will be given for the Mars 1969 mission for an 1800 km periapsis altitude using the Saturn S-I/SV as the launch booster. Various trajectory plots will be given for both Mars 1969 Type I and Type II trajectories, with emphasis placed on the Type II transfers. Type I trajectories are defined as those transfers having heliocentric central angles less than 180 deg from launch to encounter. Type II trajectories are defined as those transfers having heliocentric central angles greater than 180 deg but less than 360 deg from launch to encounter.

2. Orbiter Capability

a. Type II Trajectories

For the Mars launching in 1969, optimum trajectories for maximum orbiter mass at the planet are Type II trajectories. Figure 4-1 shows a plot of orbiter mass vs firing period for an 1800 km periapsis by 10,000 km apoapsis altitude orbit. For all Type II plots presented herein, (such as Figure 4-1) values are presented which satisfy both the declination and lighting constraint. Approximately 100 lb of additional orbiter mass may be gained if the lighting constraint is disregarded. In Figure 4-1 orbiter masses are given for both variable and fixed spacecraft propellant loadings. For a given firing period, the variable propellant loading technique will provide greater orbiter capability as compared to the fixed loading. The spacecraft propellant mass is continually varied over the firing period for the variable loading techniques but is fixed during the firing period for the fixed loading technique. Figure 4-2 shows a plot of orbiter mass vs apoapsis altitude for a fixed periapsis altitude of 1800 km and a firing period of 30 days. Note that for an apoapsis altitude of 10,000 km, an orbiter mass of about 2320 lb is available.

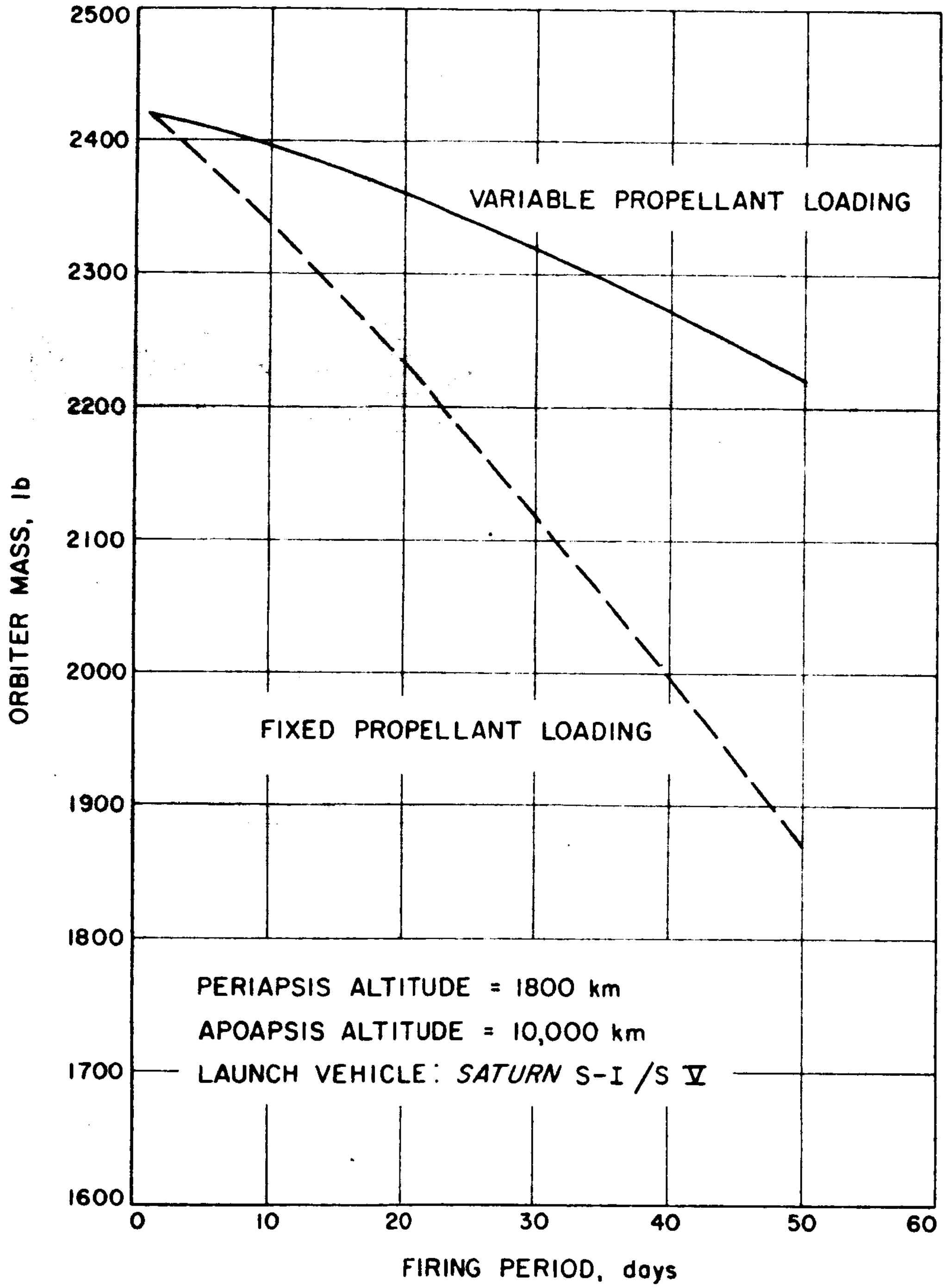


Figure 4-1. Orbiter Mass versus Firing Period for Mars 1969 Type II Trajectories

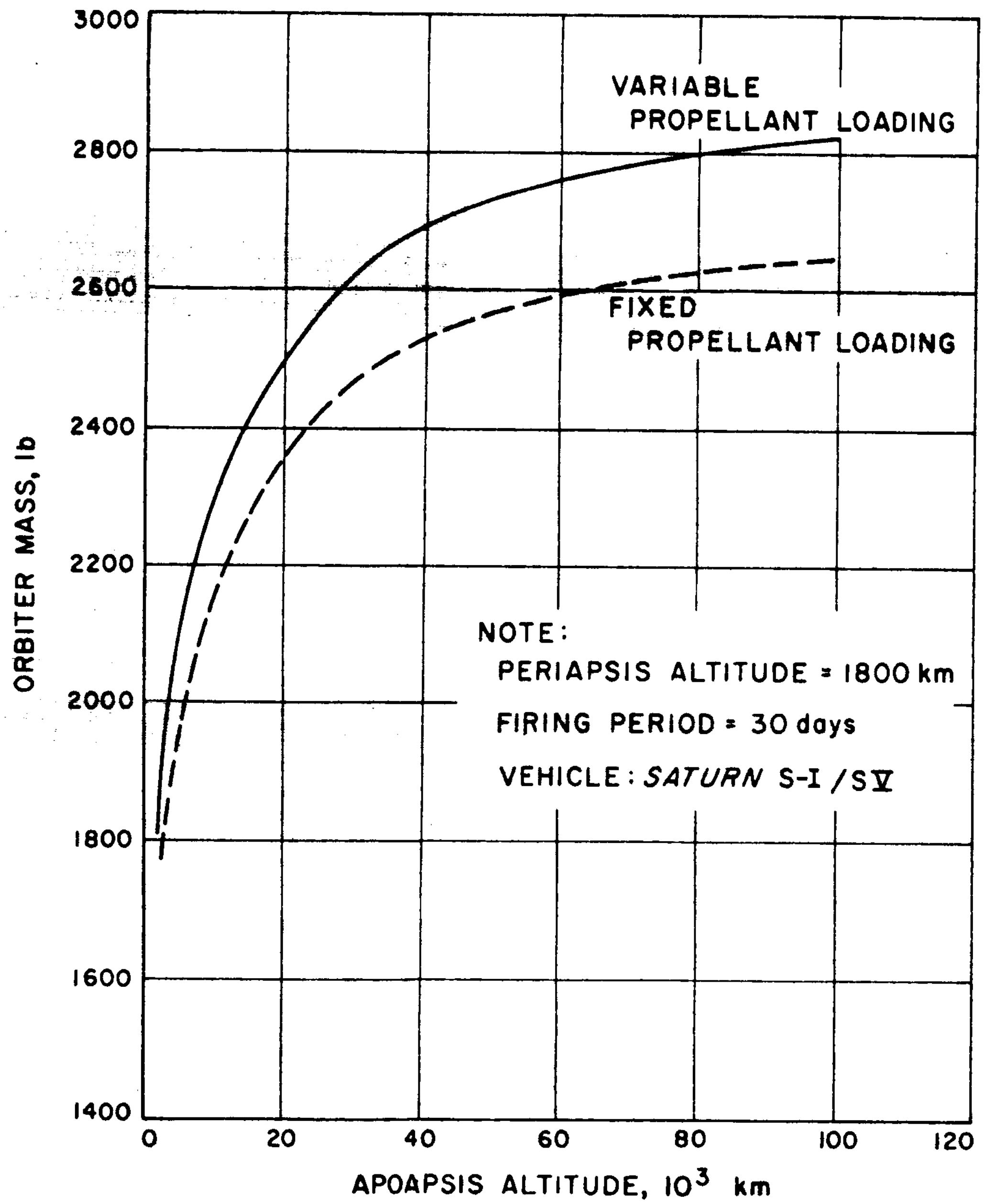


Figure 4-2. Orbiter Mass versus Apoapsis Altitude for Mars 1969 Type II Trajectories

b. Type I Trajectories

For the Type I trajectories, constraints on the declination of the outgoing geocentric asymptote greatly restrict the use of optimum trajectories. For maximum orbiter mass, large negative asymptote declinations are needed for the 1969 Type I transfers. Figure 4-3 shows a plot of orbiter mass vs firing period for an 1800 by 10,000 km orbit when the value of the asymptote declination is restricted to be greater than -50 deg. Figure 4-4 shows the orbiter mass vs firing period when the value of the asymptote declination is restricted to be greater than -36 deg. Note that a 500 lb loss in orbiter mass results in going from the -50 deg to the -36 deg asymptote declination restriction for a 30-day firing period. Utilizing a trajectory with a -50 deg asymptote declination requires the use of launch azimuths from AMR less than 47 deg or greater than 133 deg east of north (without dog-leg maneuvers); these launch azimuths would not be allowed by range safety. The -36 deg asymptotic declination corresponds to utilizing launch azimuths greater than 113.4 deg or less than 66.5 deg east of north (without dog-leg maneuver); these launch azimuths might be acceptable by range safety if negotiated.

For almost all of the firing periods of Figures 4-3 and 4-4, trajectories would have to be used which do not satisfy the lighting constraint mentioned in 1. Introduction. Type I transfers over a 30-day firing period which do satisfy the lighting constraint result in an additional orbiter mass loss of several hundred pounds and, thus, are not presented herein. Figure 4-5 shows a plot of orbiter mass vs apoapsis altitude for a periapsis altitude of 1800 km over a 30-day firing period which satisfies only the -36 deg asymptotic declination constraint. Note that an orbiter mass of 1860 lb is available for the variable propellant technique for a 10,000 km apoapsis altitude.

c. Orbiter Capability Comparison

Type II trajectories have about a 560 lb orbiter mass advantage over the Type I transfers for an 1800 by 10,000 km orbit over a 30-day firing period. For the Type II transfers about 2320 lb of orbiter mass is available when both the declination and lighting constraints are satisfied. Without the lighting constraint about 2420 lb would be available. For the Type I transfers, 1860 lb of orbiter mass is available with the -36 deg asymptotic declination restriction and without the lighting constraint, making a difference of 560 lb between Type I and Type II.

In computing orbiter masses at the planet, a total velocity increment of 280 m/sec was assumed for trajectory corrections, gravity burning-time losses, and alteration of bus flight time.

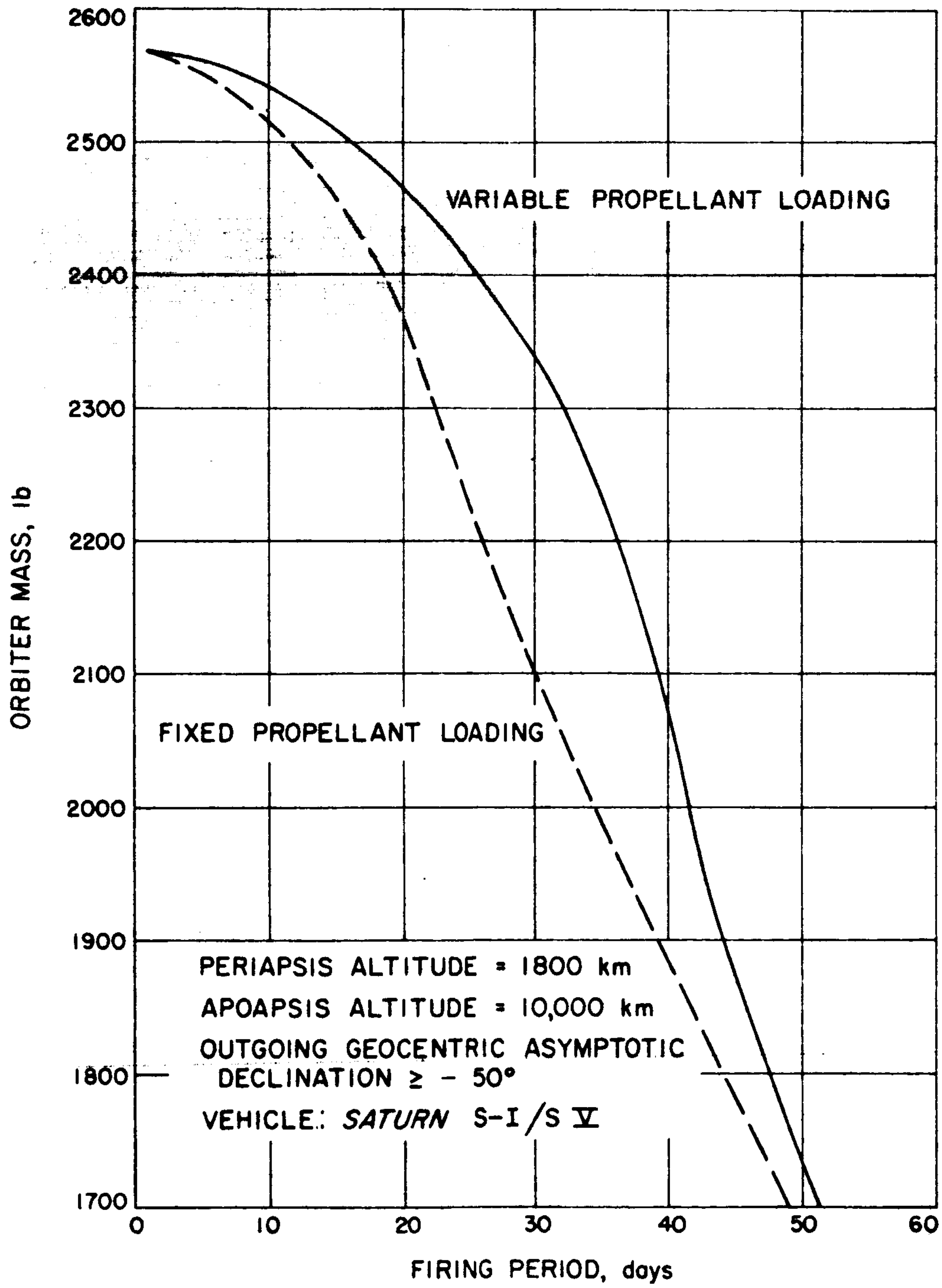


Figure 4-3. Orbiter Mass vs Firing Period for Mars 1969 Type I Trajectories

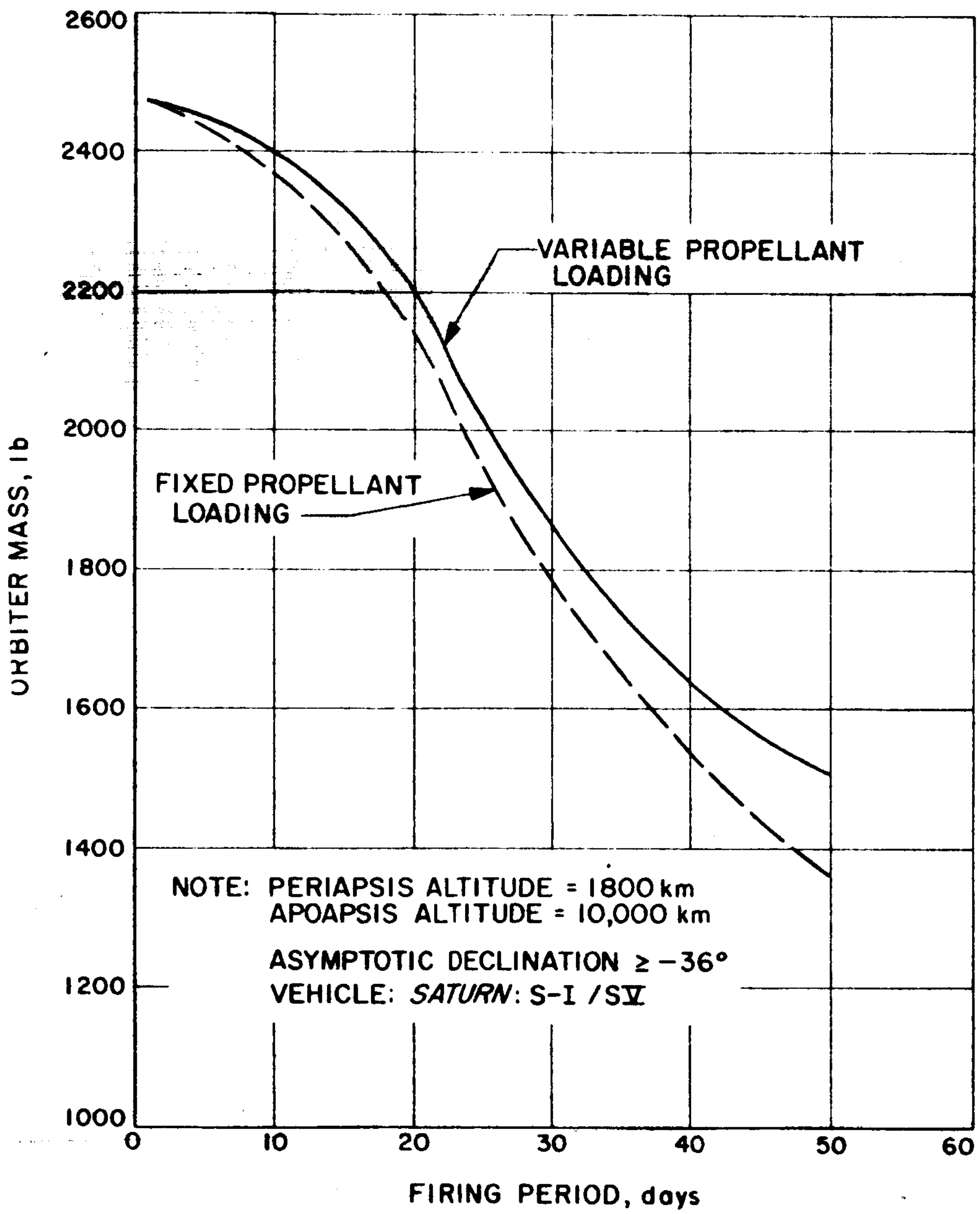


Figure 4-4. Orbiter Mass vs Firing Period for Mars 1969 Type I Trajectories

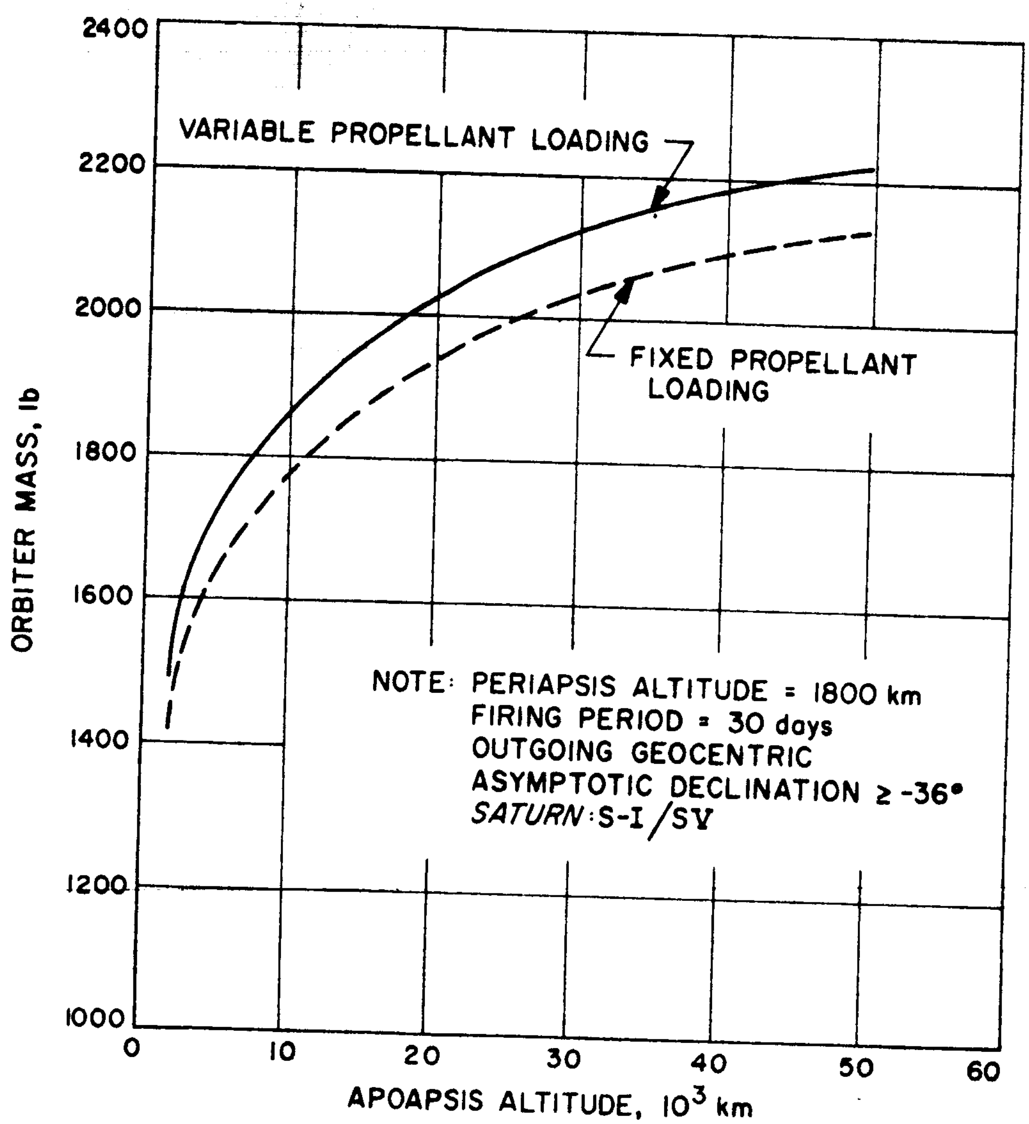


Figure 4-5. Orbiter Mass vs Apoapsis Altitude for Mars 1969 Type I Trajectories

3. Capsule Capability

For a given launch date, arrival date, periapsis altitude, apoapsis altitude, and orbiter mass, the capsule mass capability depends on whether the capsule is to be separated before or after bus retro into planetary orbit. The technique of separating the capsule after the bus is in a planetary orbit rather than before retro yields less capsule mass capability for a given orbiter mass, because more bus retro fuel is required. It is found that only one-third to one-half as much capsule mass capability is available for separation after retro as compared to separation before retro.

a. Type II Trajectories

Figures 4-6 and 4-7 show Type II capsule mass capability vs apoapsis altitude for a fixed periapsis altitude of 1800 km, firing period of 30 days, and variable propellant loading, assuming orbiter masses of 1650 lb and 2050 lb. Figures 4-8 and 4-9 show the capsule mass capability for the same assumption but with a 1000 km periapsis. Approximately two to three times more capsule mass is available if separation occurs before rather than after orbit, and about 150 lb to 350 lb more capsule mass (separated before retro) is available for the 1000 km periapsis as compared to the 1800 km periapsis depending on the orbiter mass and the apoapsis altitude. Figures 4-10 and 4-11 show capsule mass capability separated before orbit as a function of orbiter mass for variable and fixed propellant loading assuming a fixed periapsis altitude of 1800 km and a firing period of 30 days. From Figure 4-10 it is seen that, for an orbiter mass of 1650 lb and an apoapsis altitude of 10,000 km, about 1820 lb of capsule mass (separated before retro) is available. The same mass can be extracted from Figure 4-6. Figure 4-12 shows a plot of firing period vs apoapsis altitude for various orbiter and capsule masses (separated before retro) assuming a periapsis altitude of 1800 km with variable loading. Note that for a 1850 lb orbiter and a 1500 lb capsule (ejected before retro), an apoapsis altitude of 15,000 km must be utilized in order to have a 20-day firing period.

b. Type I Trajectories

Figure 4-13 shows a plot of Type I capsule mass capability vs apoapsis altitude for a periapsis altitude of 1800 km and an orbiter mass of 2050 lb, assuming variable propellant loading over a 30-day firing period with a -36 deg asymptotic declination restriction. Figure 4-14 shows a plot of capsule mass (separated before retro) vs orbiter mass with the same assumptions as Figure 4-13. From Figure 4-14 it is seen that, for a 1650 lb orbiter and a 10,000 km apoapsis altitude, about 600 lb of capsule mass is allowable. Figure 4-15 shows firing period vs apoapsis altitude for various orbiter and capsule masses (separated before retro). For an orbiter mass of 1850 lb, a capsule mass separated before retro of 1500 lb, and a firing period of 20 days, an apoapsis altitude of 40,000 km is required.

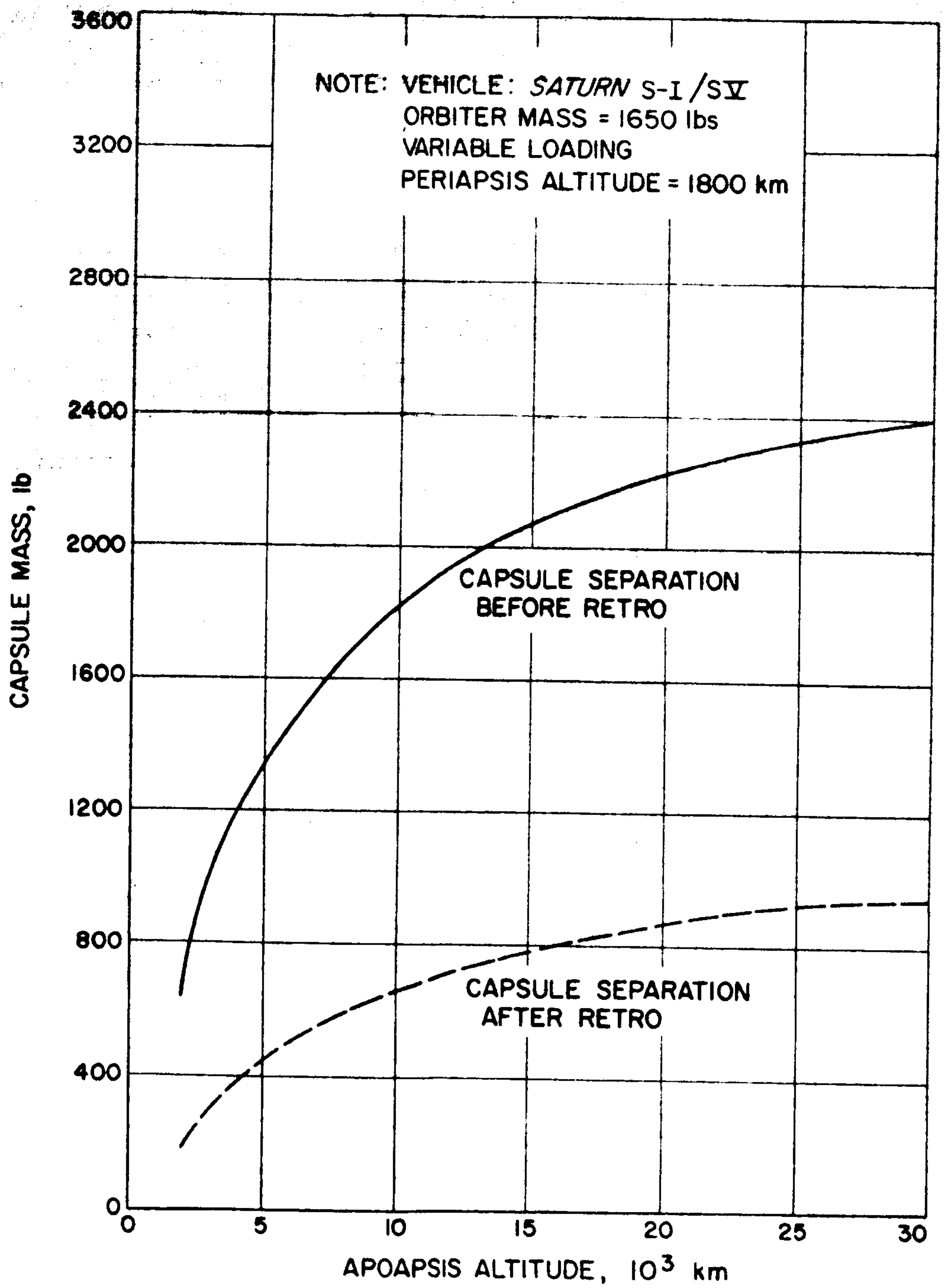


Figure 4-6. Capsule Mass vs Orbiter Apoapsis Altitude for Mars 1969 Type II Trajectories

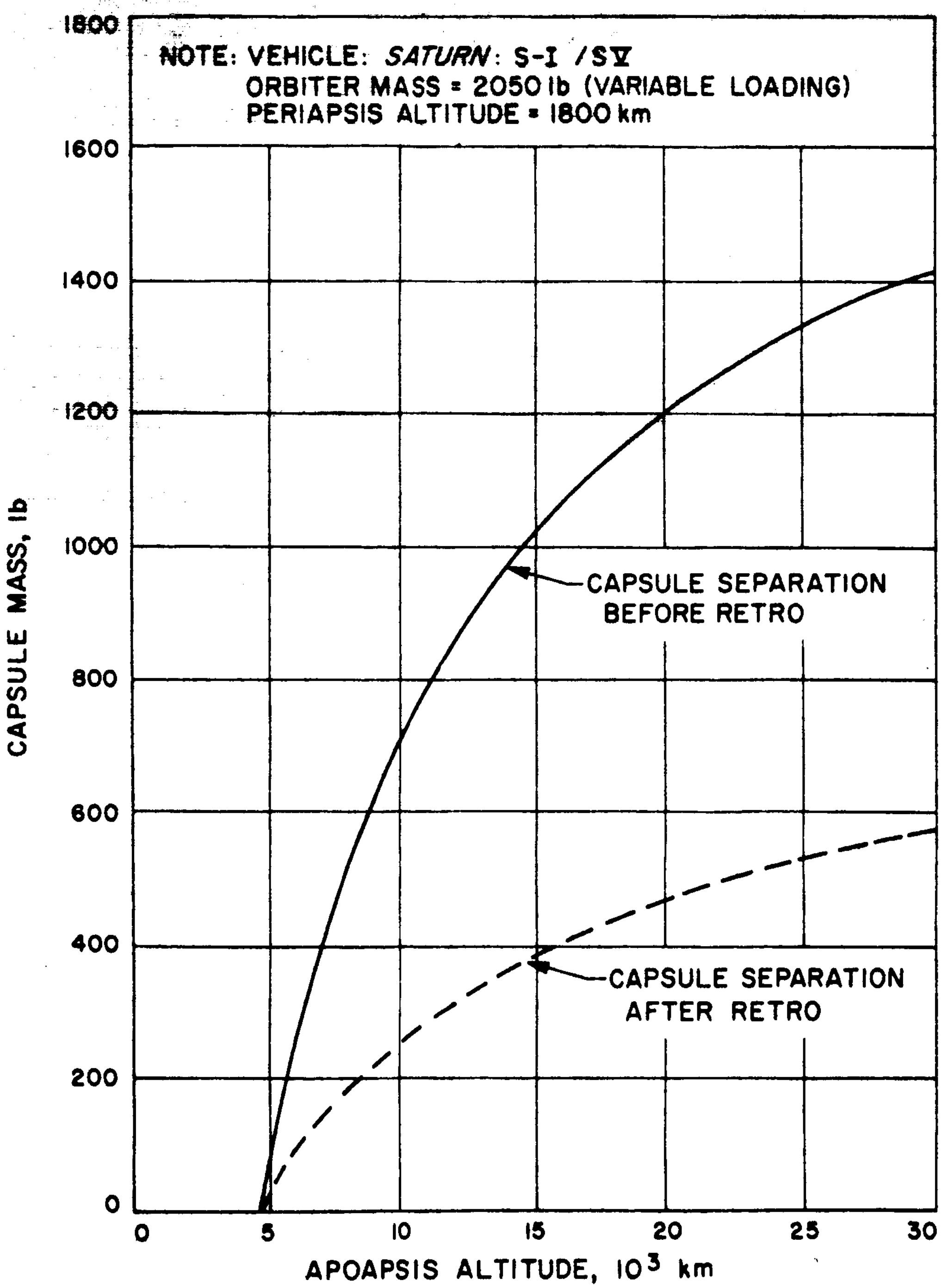


Figure 4-7. Capsule Mass vs Orbiter Apoapsis Altitude for Mars 1969 Type II Trajectories

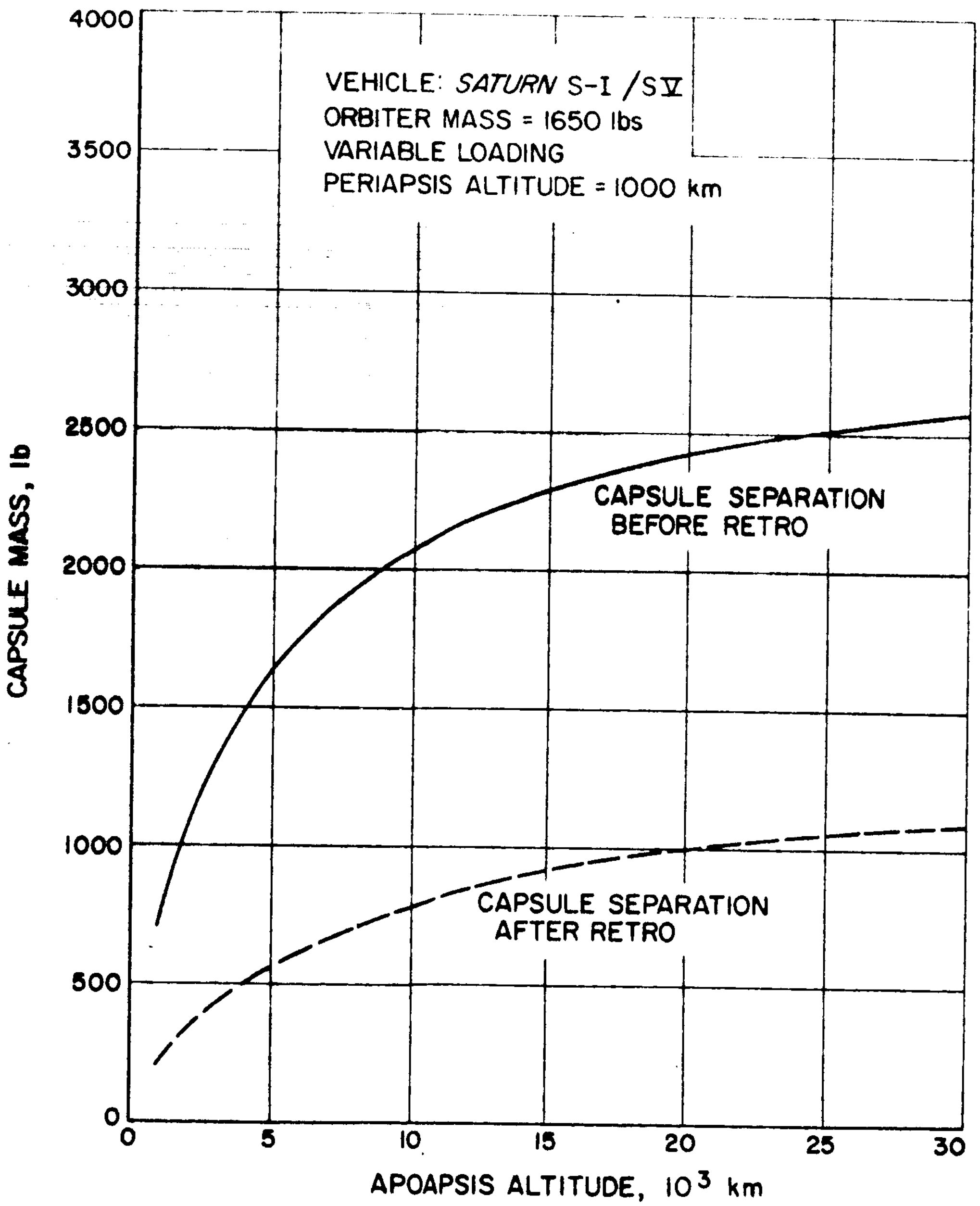


Figure 4-8. Capsule Mars vs Orbiter Apoapsis Altitude for Mars 1969 Type II Trajectories

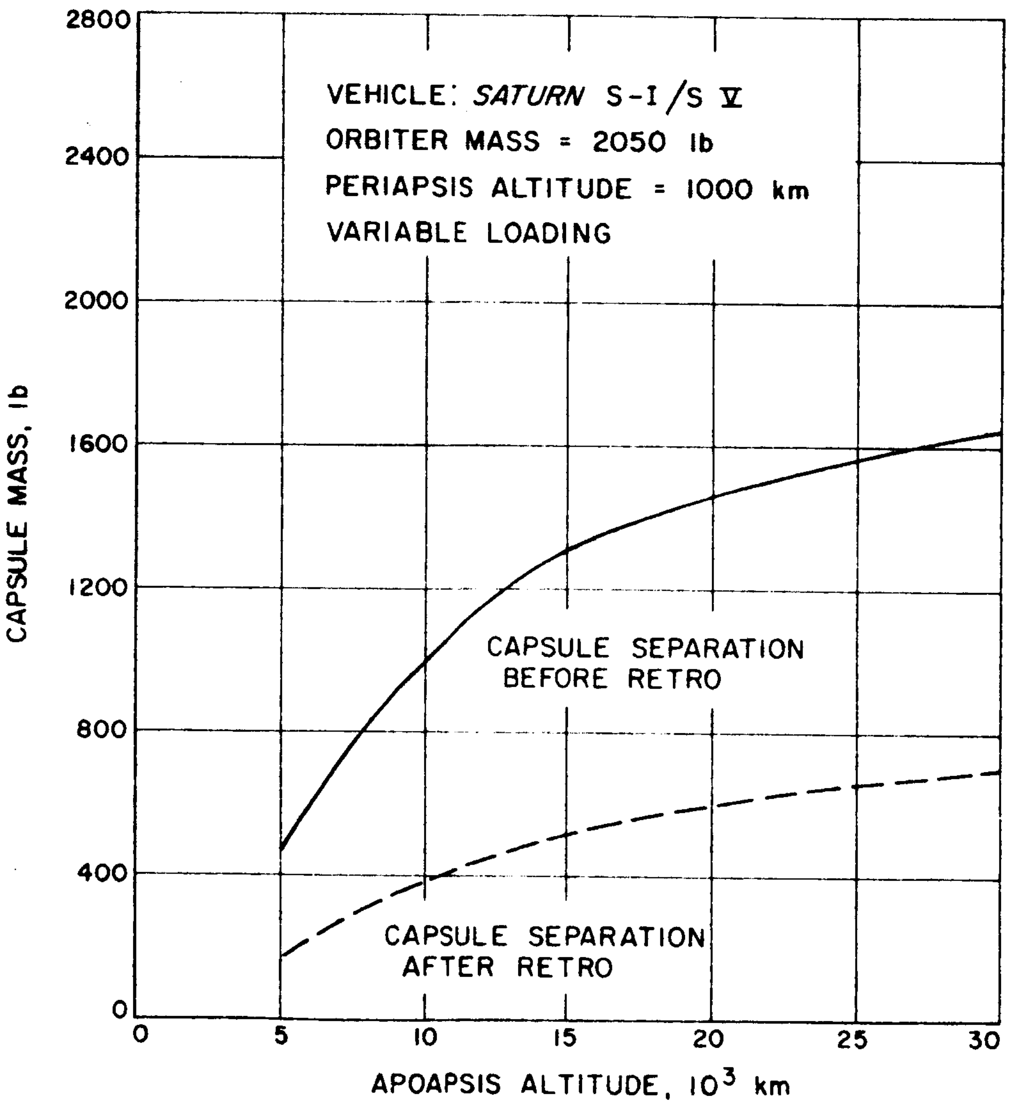


Figure 4-9. Capsule Mass vs Orbiter Apoapsis Altitude for Mars 1969 Type II Trajectories

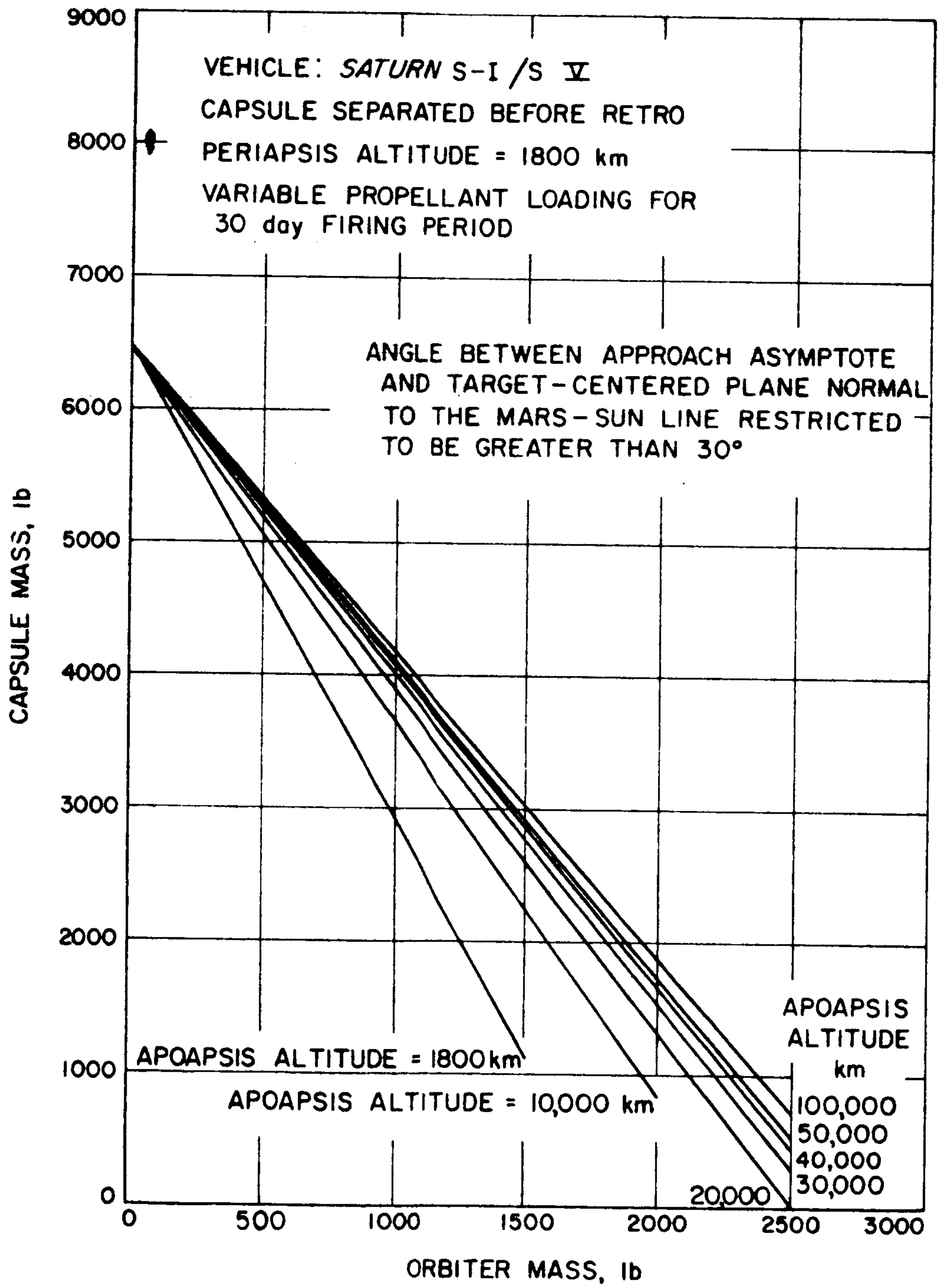


Figure 4-10. Capsule Mass vs Orbiter Mass for Mars 1969 Type II Trajectories

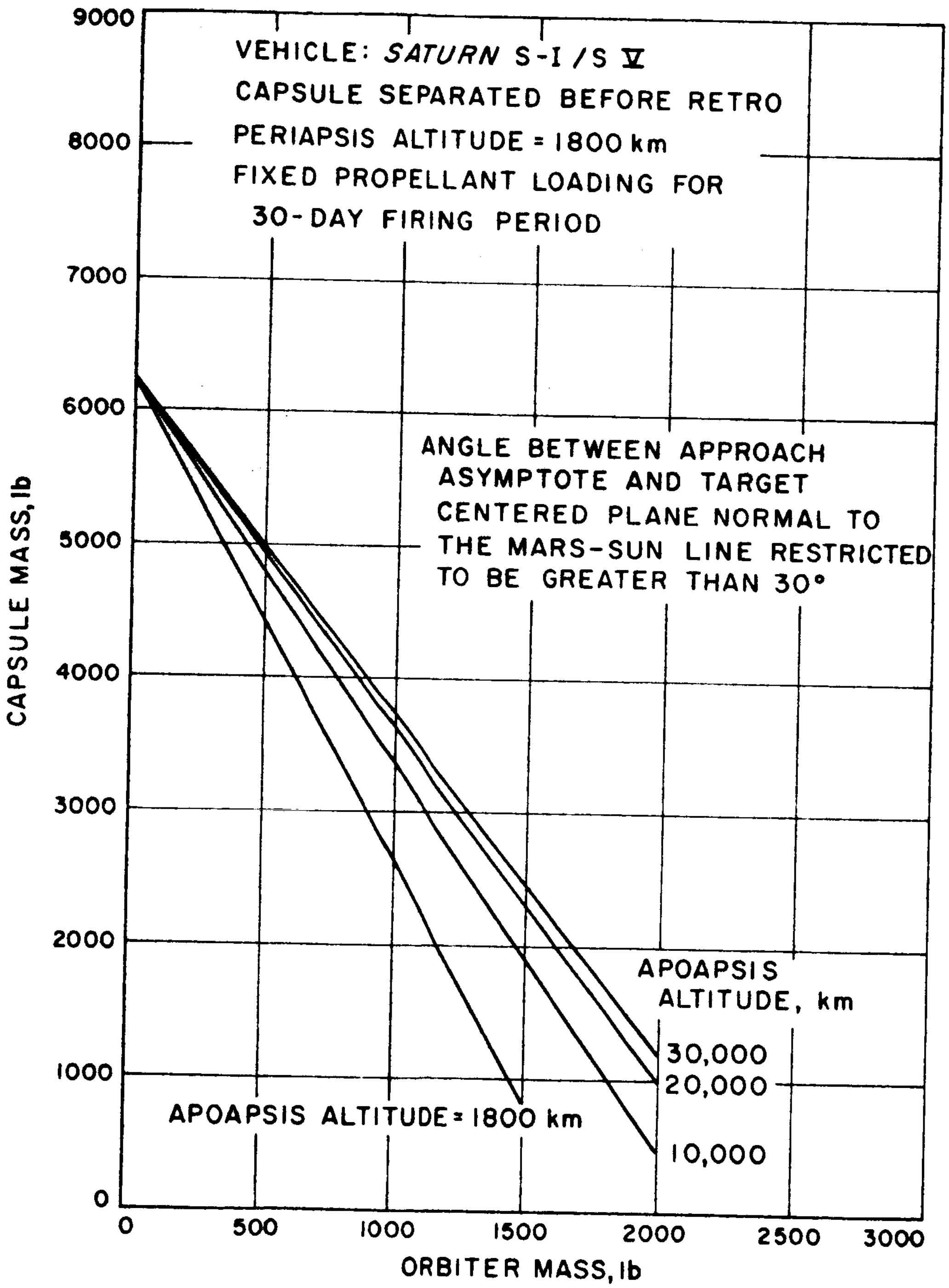


Figure 4-11. Capsule Mass vs Orbiter Mass for Mars 1969 Type II Trajectories

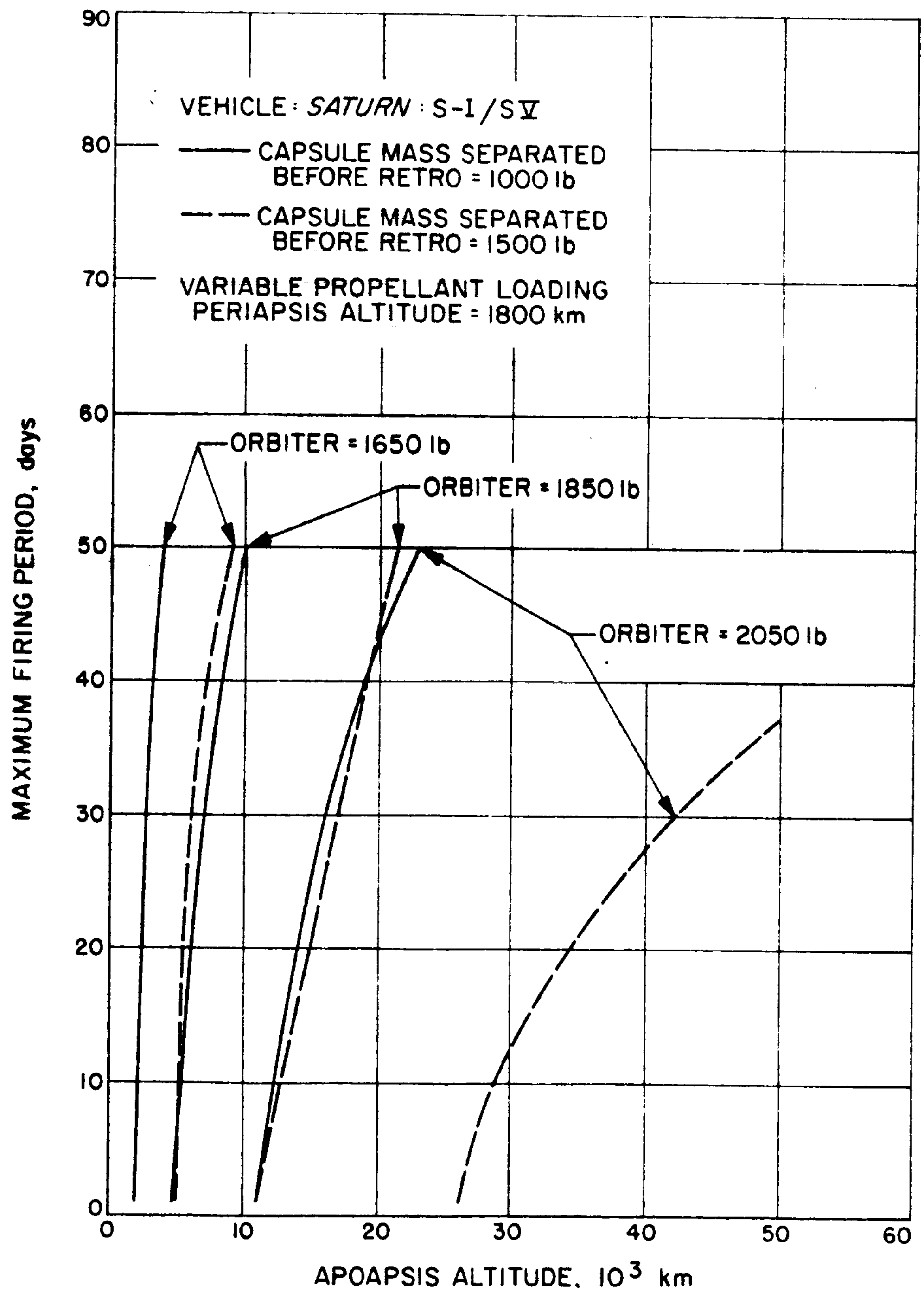


Figure 4-12. Firing Period vs Orbiter Apoapsis Altitude for Mars 1969 Type II Trajectories

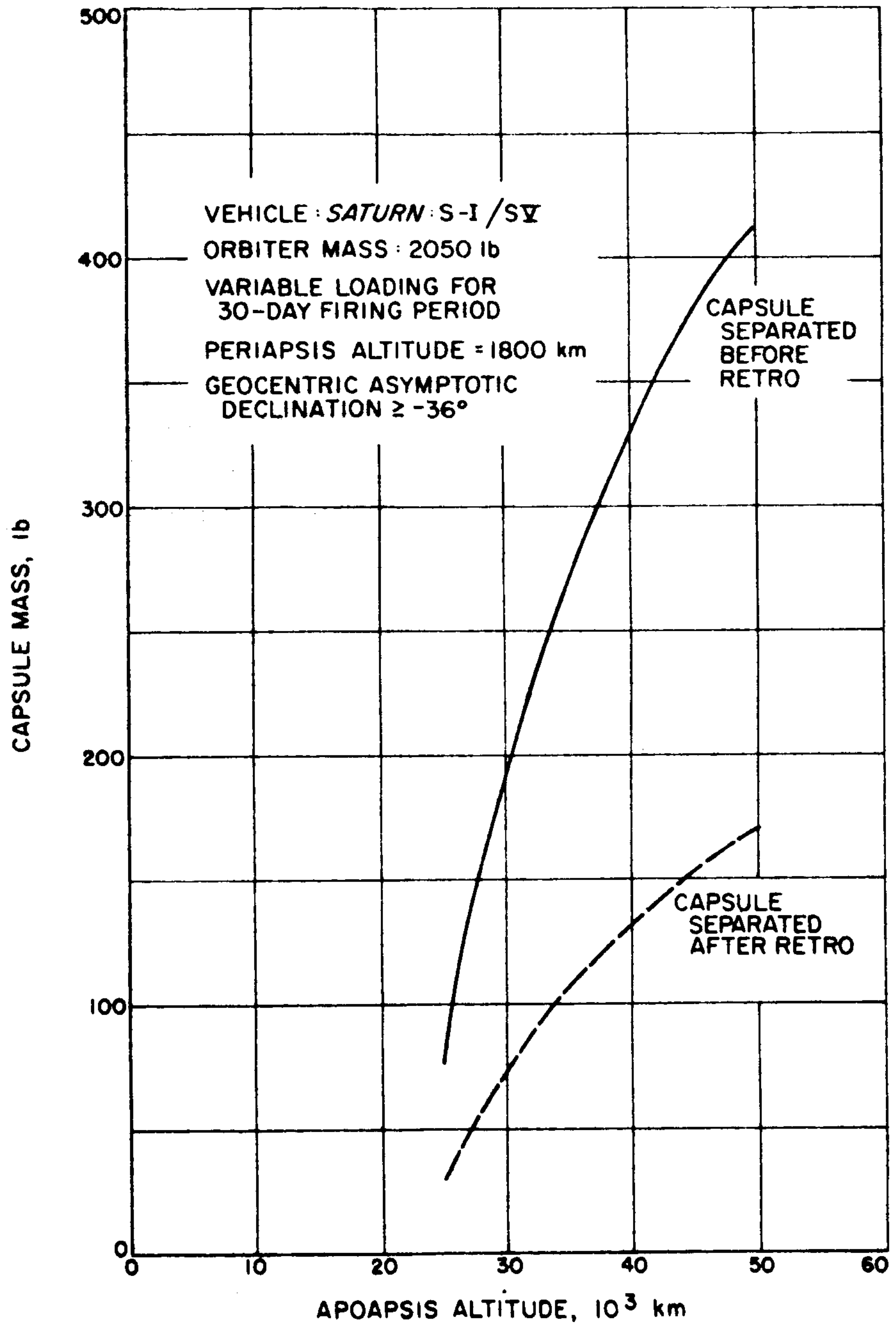


Figure 4-13. Capsule Mass vs Orbiter Apoapsis Altitude for Mars 1969 Type I Trajectories

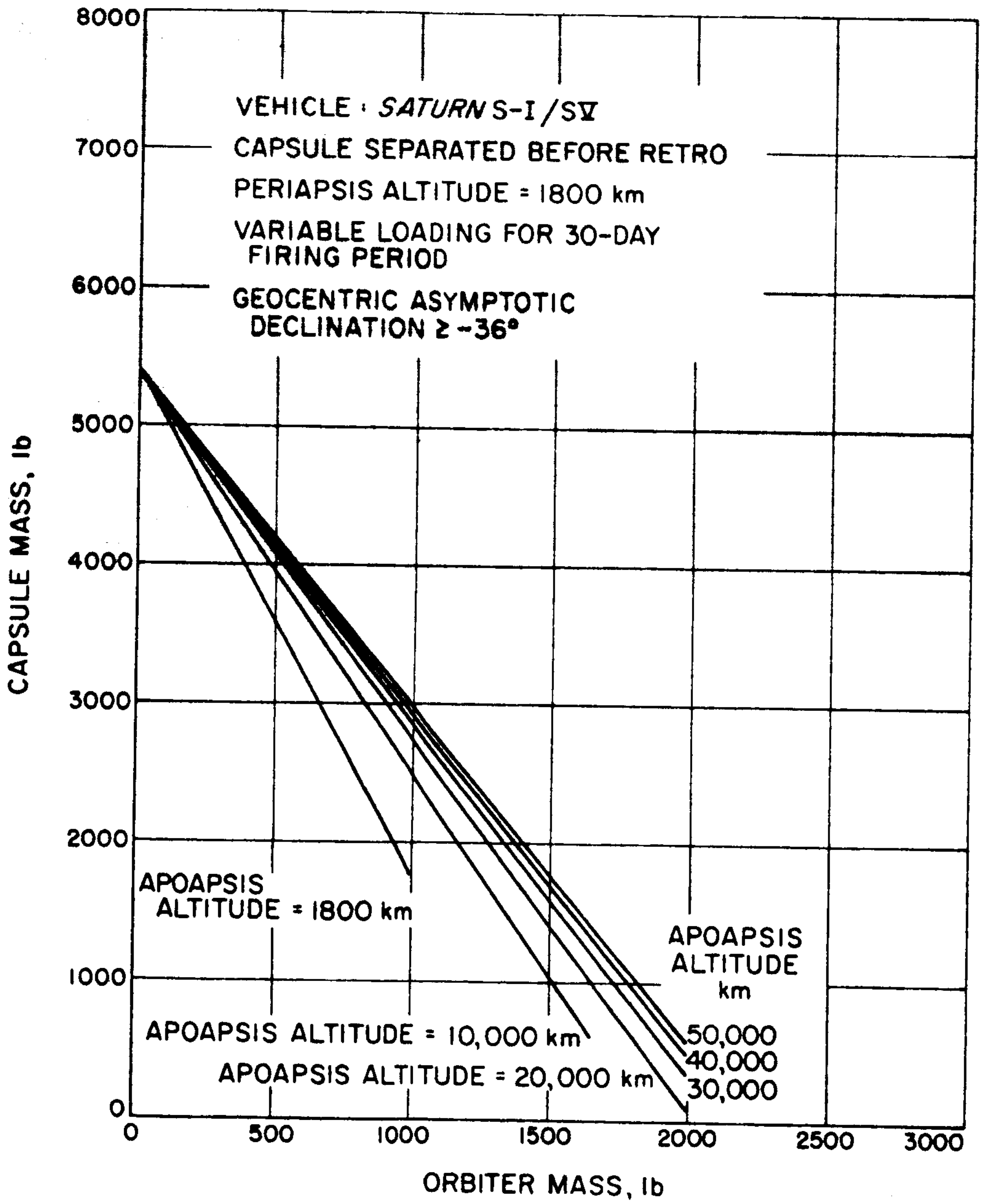


Figure 4-14. Capsule Mass vs Orbiter Mass for Mars 1969 Type I Trajectories

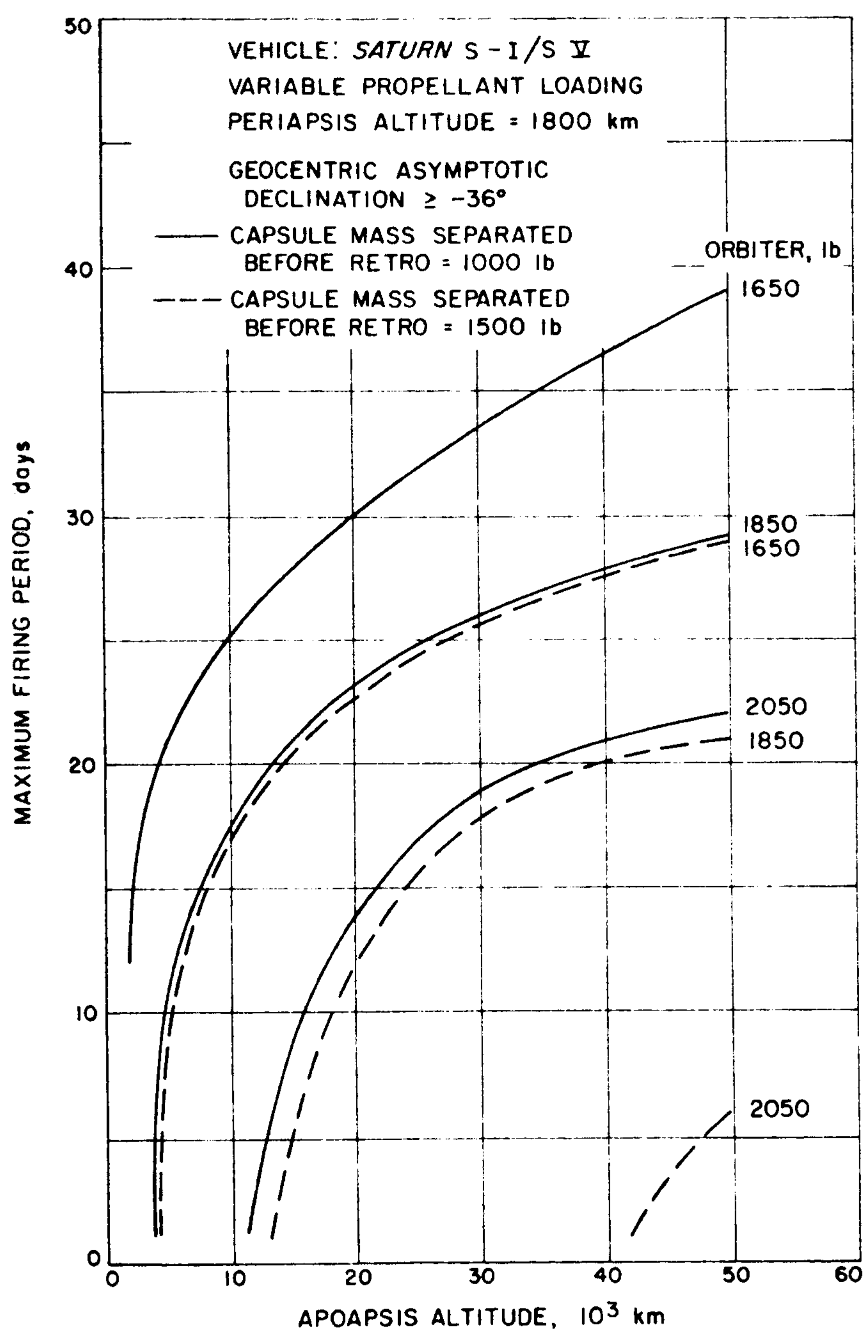


Figure 4-15. Firing Period vs Orbiter Apoapsis Altitude for Mars 1969 Type I Trajectories

4. Trajectory Characteristics

a. Near-Earth

(1) Type II Trajectories. For the Type II trajectories, the outgoing geocentric asymptotic declination varies from about -5 deg to +25 deg over a 30-day firing period. Daily firing windows of 2.7 to 4.2 hours will be available respectively for this declination range assuming launchings of 90 deg to 114 deg east of north from AMR. Figure 4-16 shows the range of injection loci for the Type II trajectories over a 30-day firing period.

(2) Type I Trajectories. As mentioned previously, the most negative outgoing asymptotic declination of Type I trajectories will have to be restricted to a value of about -36 deg or greater in order to launch from AMR at a launch azimuth less than 114 deg without dog-leg maneuvers during powered flight. For an asymptotic declination of -36 deg, launch azimuths greater than 113.5 deg or less than 66.5 deg east of north must be used. The limiting launch azimuths are given by the expression $\Sigma_L = \text{arc sin} \left(\frac{\cos \Phi S}{\cos \Phi L}\right)$, where Σ_L is the launch azimuth measured east of north, Φ_L is the launch site latitude, and Φ_S is the declination of the outgoing asymptote. Figure 4-17 shows the launch azimuth vs asymptotic declination for launching from AMR.

For the large asymptotic declinations whose absolute magnitudes are in excess of 28.3 deg (latitude of AMR), moderate firing windows are available for small variations in launch azimuth. For instance, assuming a launch azimuth spread of 113.5 deg to 114.0 deg east of north for a -36 deg asymptotic declination, a 1.3 hour window is available. Figure 4-18 shows a plot of relative launch time vs asymptotic declination for various launch azimuths. From this plot the daily firing windows may be extracted, given the desired asymptotic declination and maximum launch azimuth to be used; for example, for the -36 deg declination and the maximum launch azimuth of 114 deg, the firing window is equal to 2.3 hours less 1.0 hour or 1.3 hour. During this daily firing window the launch azimuth must be varied from 114 deg (at the beginning of window) to 113.5 deg (at the middle of the window) and back to 114 deg (at the end of the window). It is also seen from Figure 4-18 that for the -36 deg asymptotic declination, a 1.3 hour window will be available for the equivalent northeast launch azimuth range of 66 deg to 66.5 deg east of north.

For these northeast launchings, short parking orbit coast times could be used with injections located over the northwest tip of Africa. For the southeast launchings, long parking orbits in the neighborhood of 80 to 100 minutes duration must be used with injections located over Mexico and the Gulf of Mexico. Figure 4-19 shows a typical Earth track for a Type I 114 deg launch azimuth trajectory. It should be noted that the spacecraft injects over Mexico (0 hour), rises above the horizon of Johannesburg at .45 hours after injection,

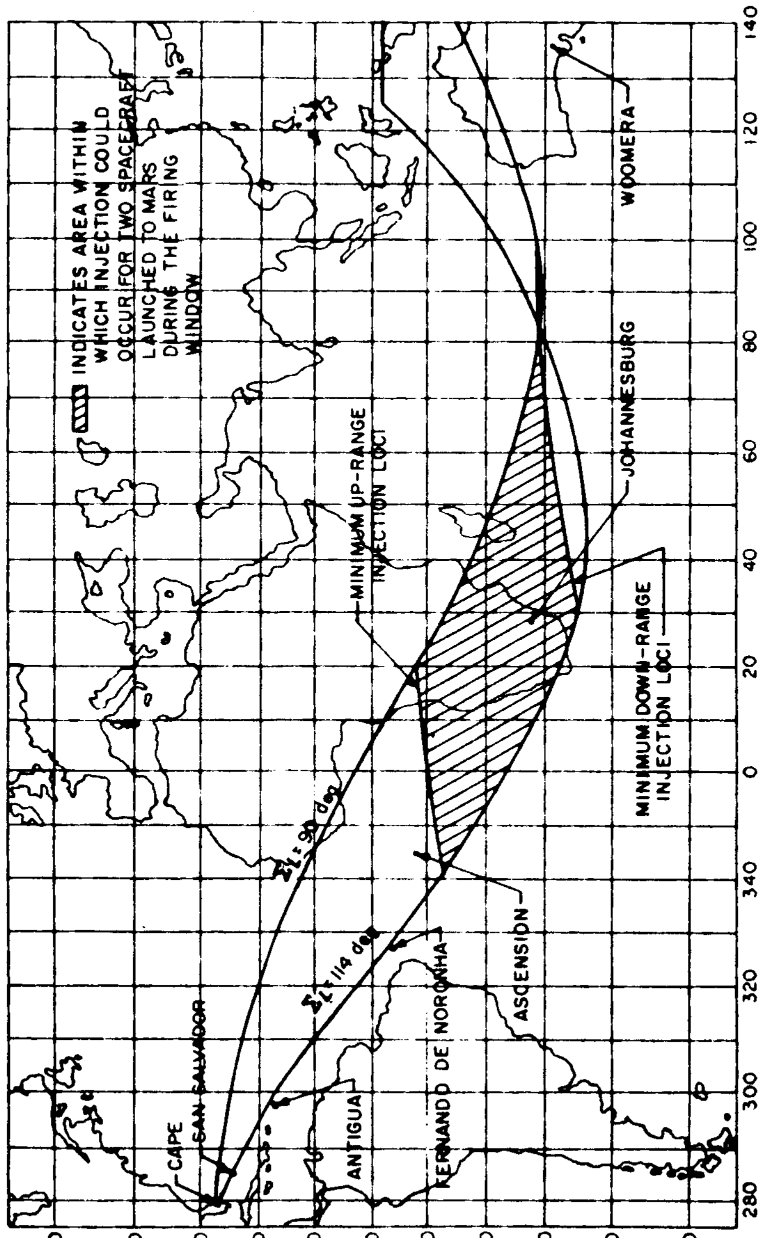


Figure 4-16. Mars 1969 Injection Locii, January-February Type II Trajectory

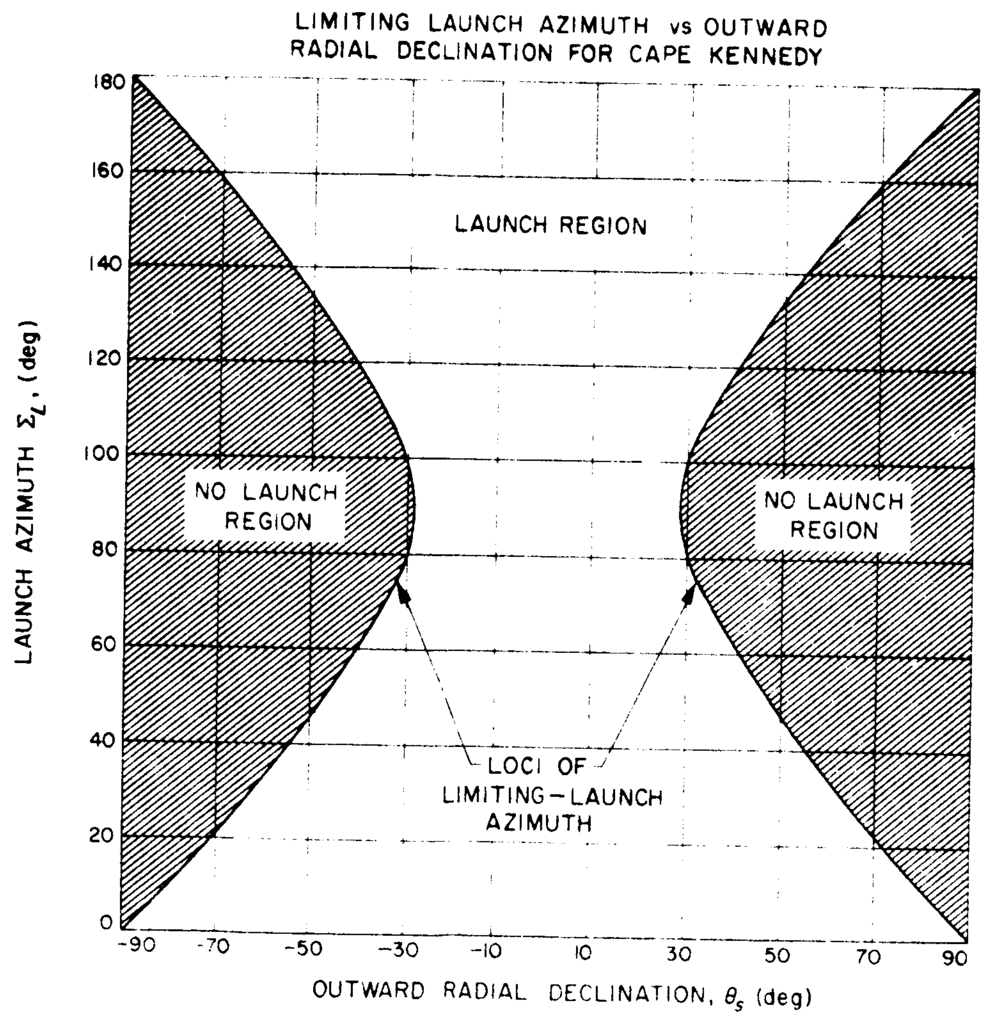


Figure 4-17. Limiting Launch Azimuth vs Outward Radial Declination for Cape Canaveral

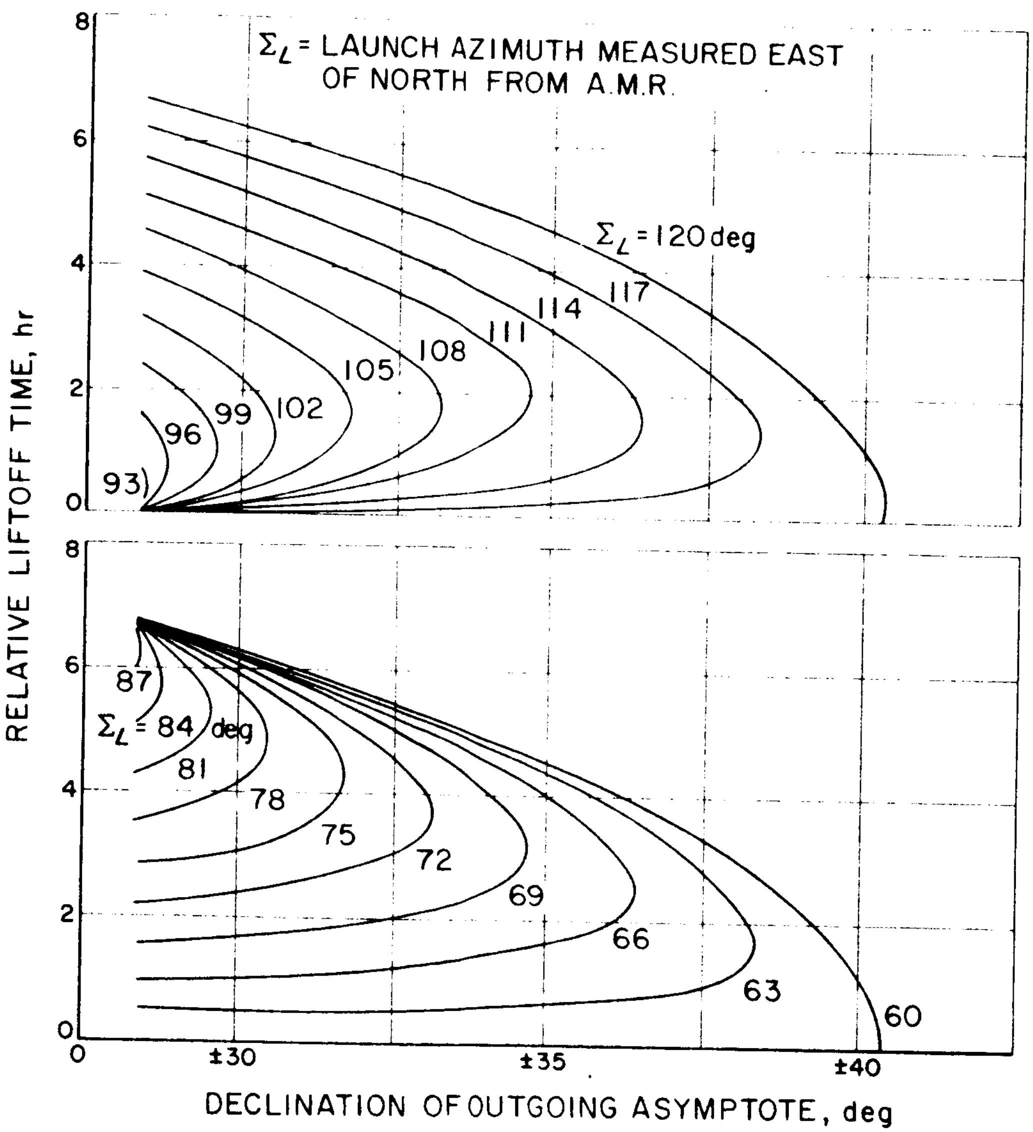


Figure 4-18. Relative Firing Window

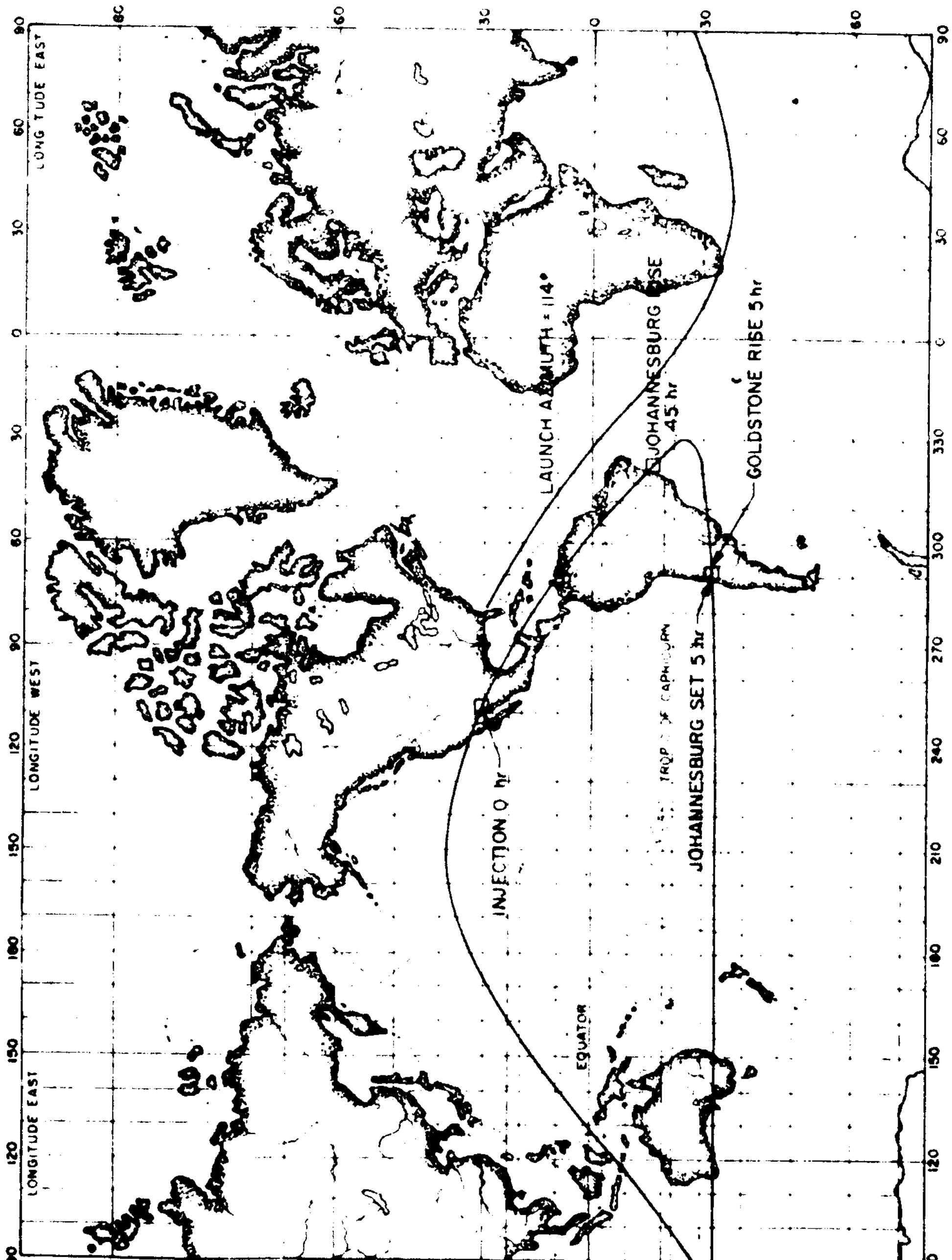


Figure 4-19. Earth Track of Typical Type I 1969 Mars Trajectory, Launch Azimuth = 114 deg

injection, and remains in view from Johannesburg until 5 hours after injection at which time it becomes visible from Goldstone.

As previously mentioned, an outgoing asymptotic declination of -50 deg will increase the payload capability of Type I trajectories. However, launch azimuths greater than 133 deg or less than 47 deg east of north would have to be utilized. It is unlikely that range safety will allow these extreme northeast or southeast launch azimuths to be used. The feasibility of dog-legging from launch azimuths of 110 deg or 114 deg to 133 deg is currently being investigated to determine associated payload losses, tracking and range safety problems.

b. Heliocentric

Tables 4-1 and 4-2 give general characteristics of Mars 1969 Type I and II trajectories for a 30 day firing period. Note that flight times for Type II trajectories range from 275 - 290 days, while those of the Type I (outgoing asymptotic declination ≥ -36 deg) are 165 - 210, a difference of 80 - 125 days. Also, the Type II trajectories transit perihelion subsequent to launch and transit aphelion prior to encounter.

Figure 4-20 shows a typical heliocentric view of a Mars 1969 Type II transfer. The inclinations of the heliocentric transfer orbits to the ecliptic are about the same for both Type I and II trajectories, ranging from 2 deg - 5 deg (see Tables 4-1 and 4-2).

Figure 4-21 shows arrival dates and communication distances at encounter versus launch date for the Type I and Type II trajectories using variable propellant loading over one month firing periods. The two Type II contours, "a" and "b", drawn represent the range of arrival dates permissible if no lighting constraint is observed, and if the lighting constraint is satisfied, respectively. The contour "c" drawn for the Type I trajectories represents the range of arrival dates allowable if the absolute value of the geocentric asymptotic declination is restricted to be less than 36 deg. Note that trajectories inside of this contour violate the lighting constraint. The range of arrival dates per launch date can be increased by decreasing payload capability.

c. Near-Planet

The near-planet characteristics of Type I and II trajectories in 1969 have gross similarities. The arrival dates and communication distances at encounter are about the same; the hyperbolic excess speeds with respect to Mars have about the same range of 3.7 - 4.5 km/sec, and the spacecraft approaches the planet from the same general direction relative to the Sun and Earth. The encounters differ, however, in that for Type II Trajectories the spacecraft approaches Mars from above the ecliptic with a range in angle between the incoming asymptotic and ecliptic of -15 deg to -30 deg while for Type I

Table 4-1. Trajectory Parameters for Type I Trajectories to Mars in 1969

A. Variable Propellant Loading

Restrictions	1 Month Firing Period (1969)	Flight Time (days)	Communication Distance at Encounter (millions of km)	Arrival Dates (1969)	Geocentric Energies (C_3) at Earth (km^2/sec^2)	Range in Geocentric Asymptotic Declination (deg)	Inclination of Heliocentric Orbital Plane to Ecliptic (deg)
(a)	3 11 - 4 9	170 - 220	135 - 200	9 15 - 11 27	10 - 16	-34 to -50	2.2 - 3.5
(b)	3 23 - 4 21	165 - 210	130 - 190	9 10 - 11 17	12 - 22	-29 to -36	2.0 - 2.7

B. Fixed Propellant Loading

Restrictions	1 Month Firing Period (1969)	Flight Time (days)	Communication Distance at Encounter (millions of km)	Arrival Dates (1969)	Geocentric Energies (C_3) at Earth (km^2/sec^2)	Range in Geocentric Asymptotic Declination (deg)	Inclination of Heliocentric Orbital Plane to Ecliptic (deg)
(a)	3 16 - 4 14	170 - 220	135 - 200	9 15 - 11 27	11 - 18	-34 to -50	2.2 - 3.5
(b)	3 25 - 4 23	165 - 210	130 - 190	9 10 - 11 17	12 - 21	-29 to -36	2.0 - 2.7

Restrictions

(a) Geocentric asymptotic declination (Φ_S) restricted -50 deg $\le \Phi_S \le$ +50 deg

(b) Geocentric asymptotic declination (Φ_S) restricted -36 deg $\le \Phi_S \le$ +36 deg

Table 4-1 (cont)

Angle Between Sun-Mars-Earth at Encounter (deg)	Heliocentric Central Angle (deg)	Hyperbolic Excess Speed with Respect to Mars (km sec)	Range in Angle Between Incoming Asymptote and Ecliptic (deg)	Range in Angle Between Incoming Asymptote and Mars-Sun Directions (deg)
46 deg - 43 deg	125 - 155	3.6 - 4.0	7 - 18	60 - 120
46 deg - 43 deg	115 - 145	3.7 - 4.5	5 - 10	65 - 130
46 deg - 43 deg	125 - 155	3.6 - 3.9	7 - 18	60 - 120
46 deg - 43 deg	115 - 145	3.7 - 4.2	5 - 10	65 - 130

Table 4-2 (cont)

	Geocentric Energies (C_3) at Earth (km^2 sec^2)	Range in Geocentric Asymptotic Declination (deg)	Inclination of Heliocentric Orbital Plane to Ecliptic (deg)	Angle Between Sun-Mars-Earth at Encounter (deg)
(a)	10 - 12	-5 to -25	2 2 - 5 0	46 deg - 43 deg
(b)	9.3 - 12	5 to -15	2 2 - 3 5	44.5 deg - 41.5 deg
(a)	10 - 12	-5 to 20	2 2 - 4 5	46 deg - 43 deg
(b)	9 - 12	-5 to -25	2 2 - 4 5	44 deg - 41.5 deg

Table 4-2. Trajectory Parameters for Type II Trajectories to Mars in 1969

A. Variable Propellant Loading

Restrictions	1 Month Launch Period (1969)	Flight Time (days)	Communication Distances at Encounter (millions of km)	Arrival Dates (1969)
(a)	1 15 - 1 13	250 - 290	150 - 200	10 5 - 11 27
(b)	1 23 - 2 22	275 - 290	180 - 210	11 7 - 12 7

B. Fixed Propellant Loading

Restrictions	1 Month Launch Period (1969)	Flight Time (days)	Communication Distances at Encounter (millions of km)	Arrival Dates (1969)
(a)	1 22 - 2 20	260 - 290	160 - 200	10 16 - 11 27
(b)	1 29 - 2 27	260 - 295	185 - 210	11 12 - 12 7

Restrictions

(a) Optimum Propulsion Trajectories
Geocentric Asymptotic declination (Φ_S) restricted -36 deg $\leq \Phi_S \leq 36$ deg.

(b) Geocentric Asymptotic declination restricted as in b
Angle between Planet-Sun direction and incoming asymptote restricted to be less than 60 deg and greater than 120 deg.

Table 4-2 (cont)

	Heliocentric Central Angle (deg)	Hyperbolic Excess Speed with Respect to Mars (km/sec)	Range in Angle Between Incoming Asymptote and Ecliptic (deg)	Range in Angle Between Incoming Asymptote and Mars-Sun Direction (deg)
(a)	200 - 225	3.8 - 4.2	-15 to -28	60 - 100
(b)	205 - 225	4.0 - 4.4	-15 to -24	50 - 60
(a)	202 - 225	3.8 - 4.3	-15 to -28	60 - 80
(b)	200 - 230	4.2 - 4.5	-15 to -30	50 - 60

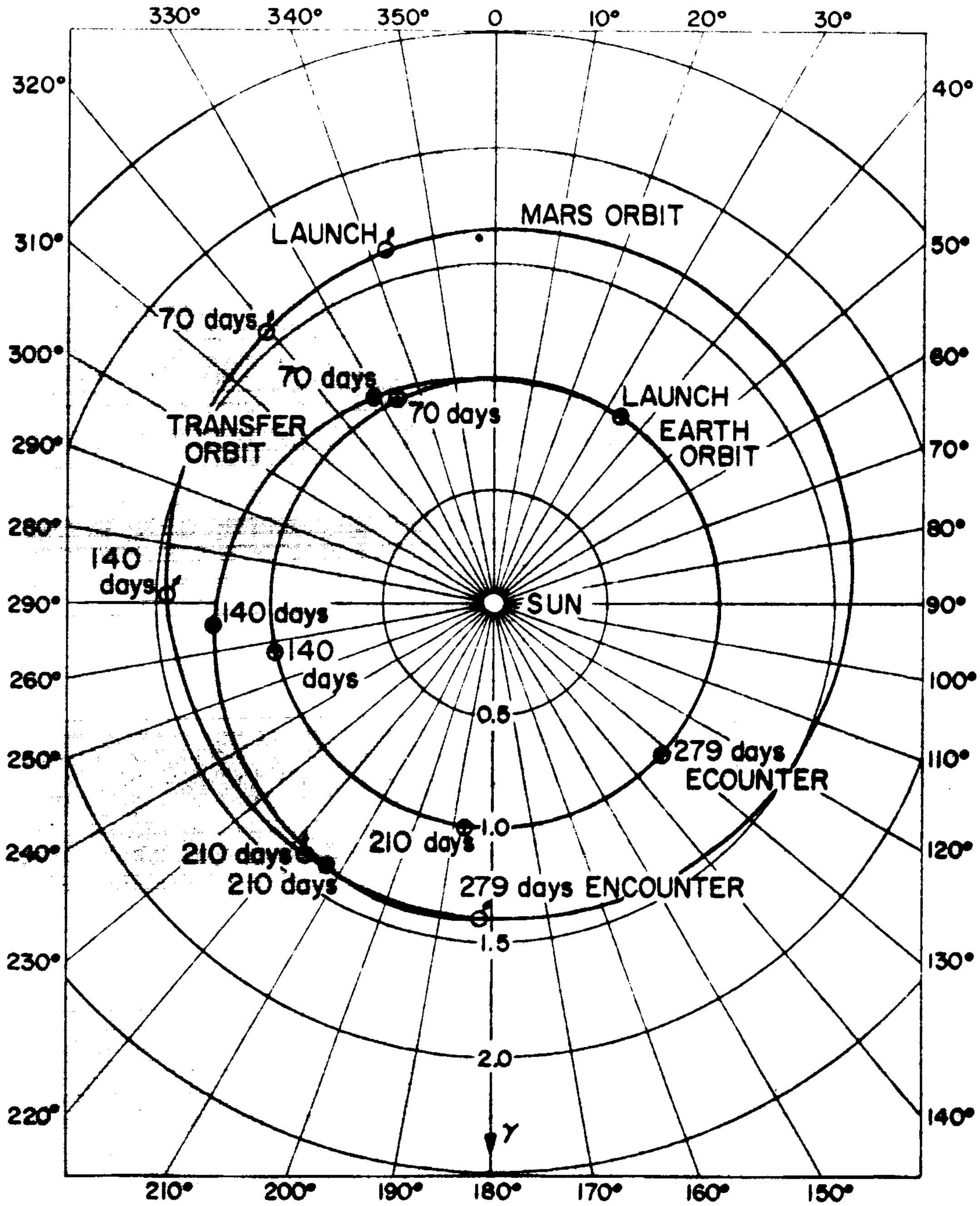


Figure 4-20. Projection of Heliocentric Transfer Orbit Onto Ecliptic Plane, Typical Mars 1969 Type II Trajectory

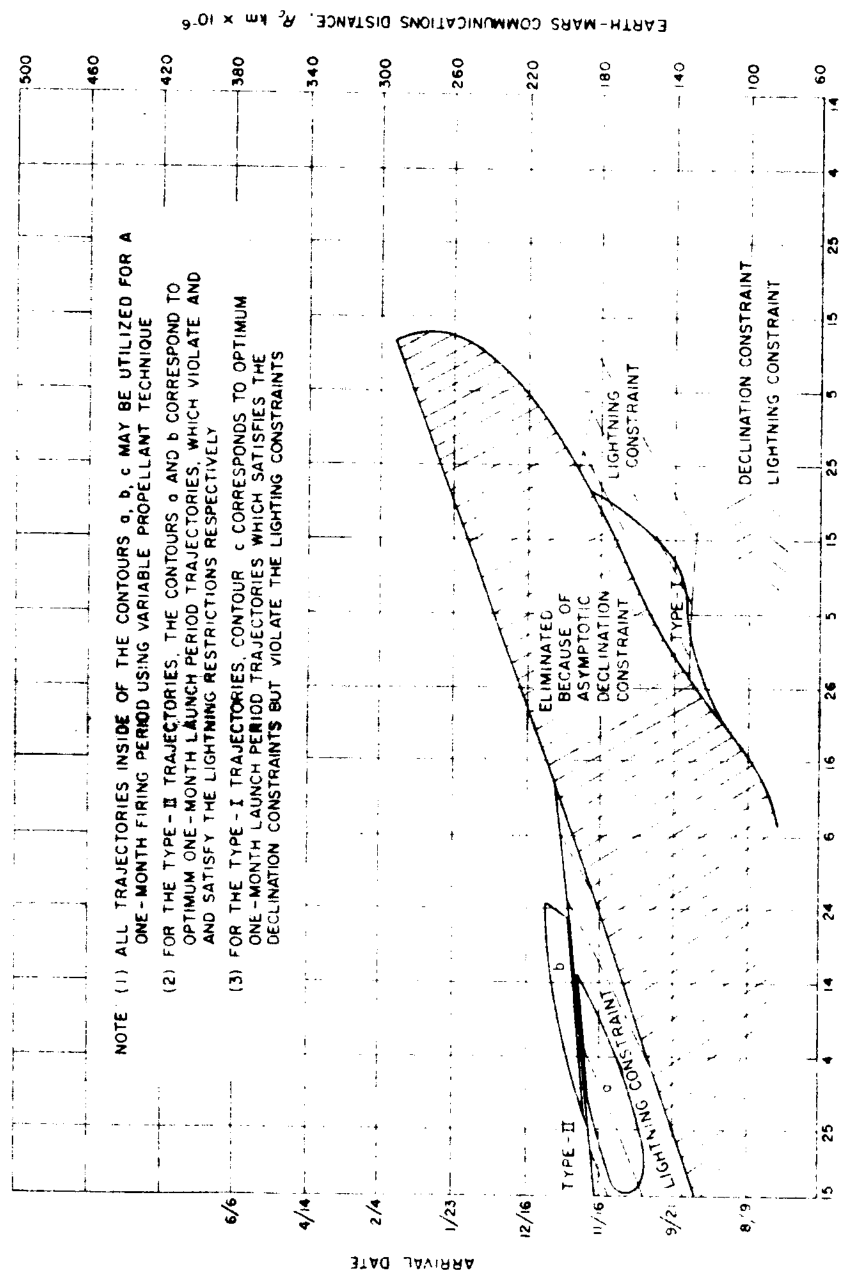


Figure 4-21. Arrival Dates and Launch Dates for Mars 1969 Type I and Type II Trajectories

trajectories the approach is from below the ecliptic at an angle of 5 deg to 20 deg (See Tables 1 and 2). These angles may seem large at first glance, but it should be noted that these approach directions are with respect to Mars and not the Sun. As mentioned previously, Type I Trajectories given in Table 4-1 and Figure 4-21 violate the lighting constraint which restricts the angle between the incoming asymptote and the Mars-Sun direction to be less than 60 deg or greater than 120 deg. Type I Trajectories which satisfy this constraint over a 30-day firing period result in an additional orbiting payload loss of several hundred pounds and, thus, are not listed.

5. Geometrical Properties at Planet

a. Before Encounter

Figures 4-22 and 4-23 show the direction of the hyperbolic excess velocity vector at the planet as viewed from above the ecliptic for optimum transfer trajectories to Mars and Venus during the period 1969 - 1973. For Mars trajectories the spacecraft approaches the planet from the leading edge while for Venus the approach is from the trailing edge of the planet. The angle between the Mars-Sun vector and the Mars-Earth vector at planet encounter remains fairly constant for the trajectories considered, from about 35 deg to 45 deg for both Type I and Type II Trajectories. For Venus trajectories, however, the angular variation is greater; from about 140 deg to 90 deg for Type I Trajectories and from about 90 deg to 50 deg for Type II Trajectories. (This variation is not shown in Figure 4-23).

b. Post-Encounter

Subsequent to planetary encounter, the Earth-Planet-Sun angle decreases for both Mars and Venus trajectories. For the Mars trajectories, encounter usually occurs near the maximum Earth-Probe-Sun angle of 45 deg. Figure 4-24 shows the Earth-Mars-Sun angle from August 17, 1969 through June 23, 1970. Figure 4-25 shows clock angle (Sun-Canopus coordinate system) of the Earth as measured from a spacecraft at Mars. Figure 4-26 shows the definition of clock angle. Figure 4-27 shows Earth-Mars distance as a function of time. The span of arrival dates for Type I and Type II trajectories are marked on the above figures.

For a typical Type II Trajectory to Mars in 1969, encounter takes place on November 20, 1969 with an Earth-Planet-Sun angle of 43 deg, clock angle of 282 deg and Earth-Planet communication distance of 195 $\times 10^6$ km. By April 20, 1970, the Earth-Mars-Sun angle has decreased to a value of 20 deg.

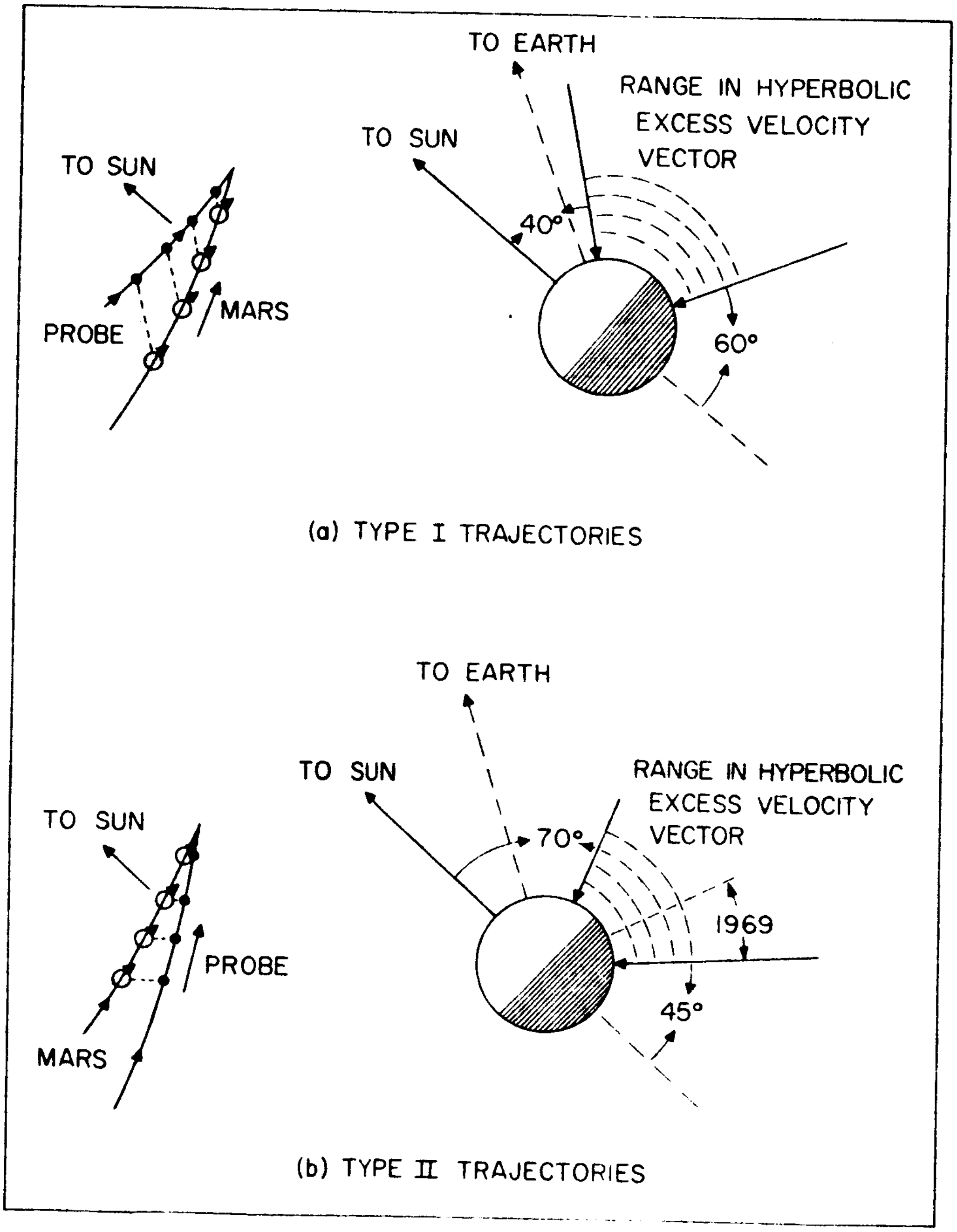


Figure 4-22. Direction of Approach Hyperbolic Excess Velocities for Feasible Mars Trajectories 1969-1975

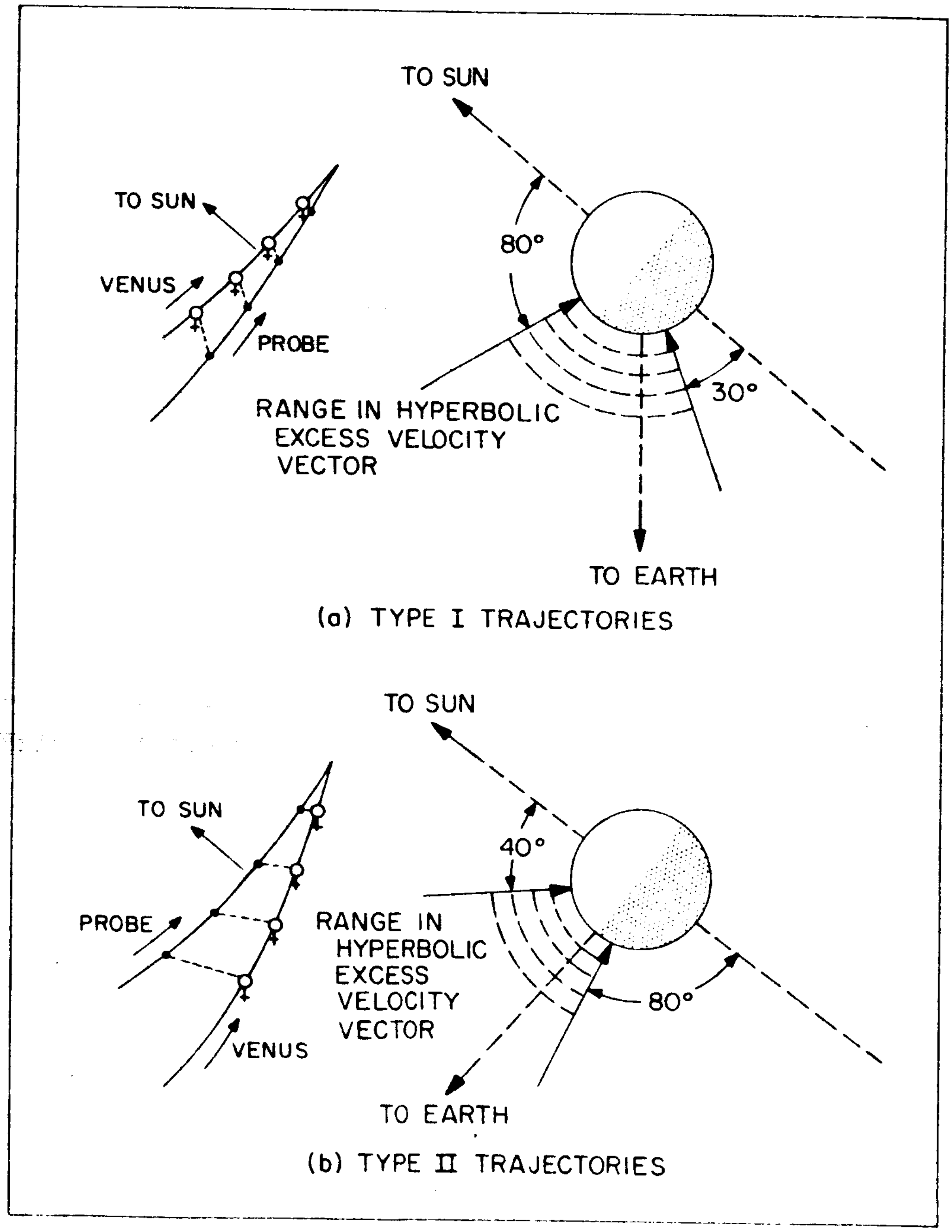


Figure 4-23. Direction of Approach Hyperbolic Excess Velocities for Feasible Venus Trajectories 1969-1973

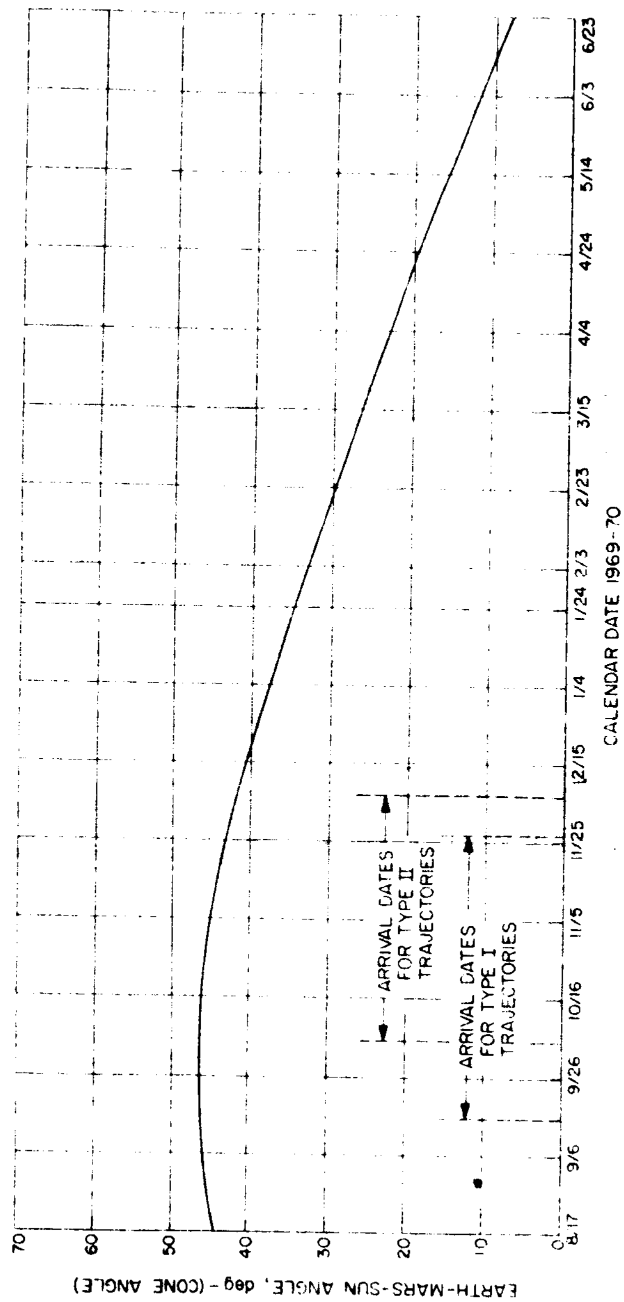


Figure 4-24. Earth-Mars-Sun Angle vs Calendar Date

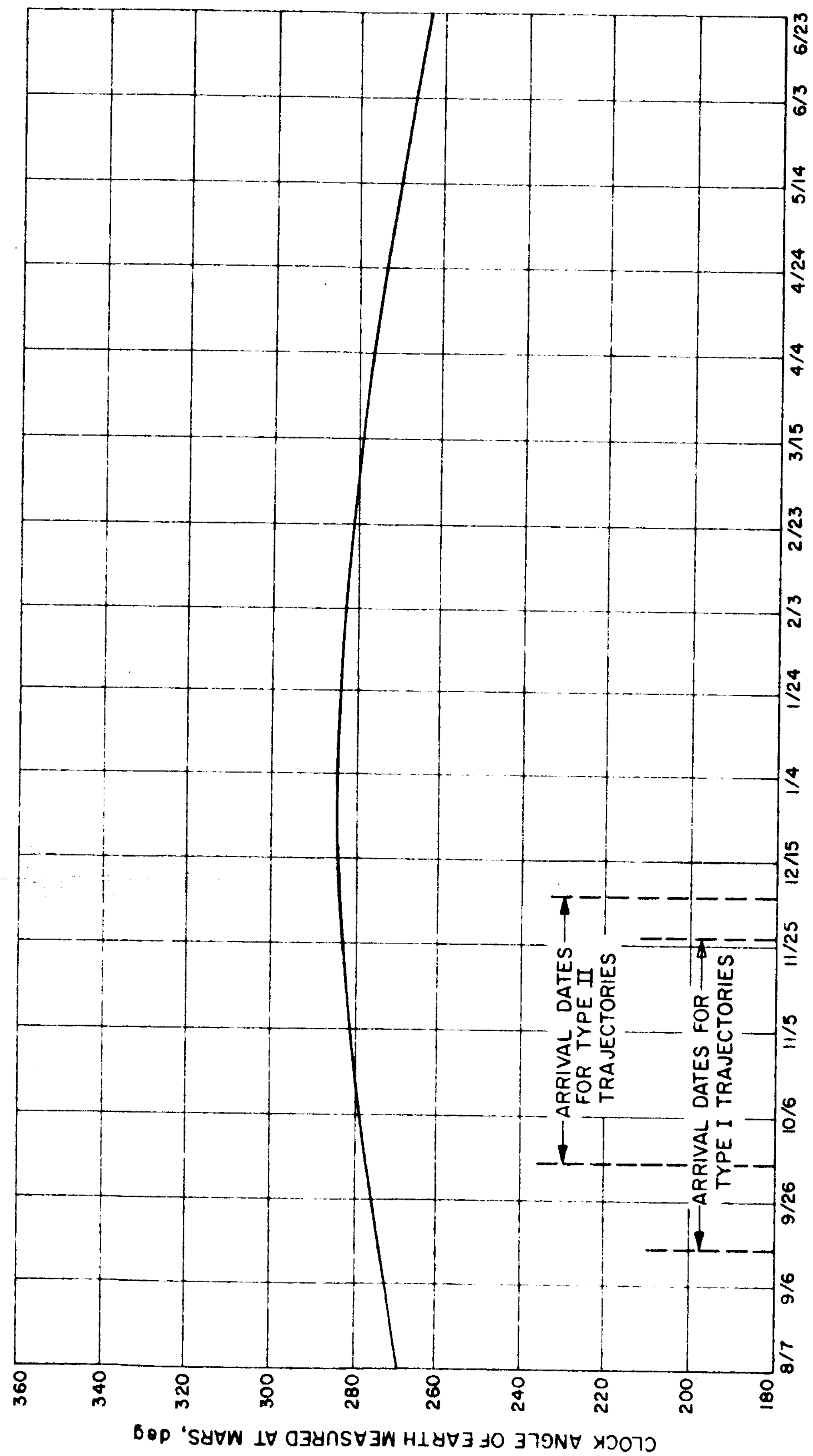
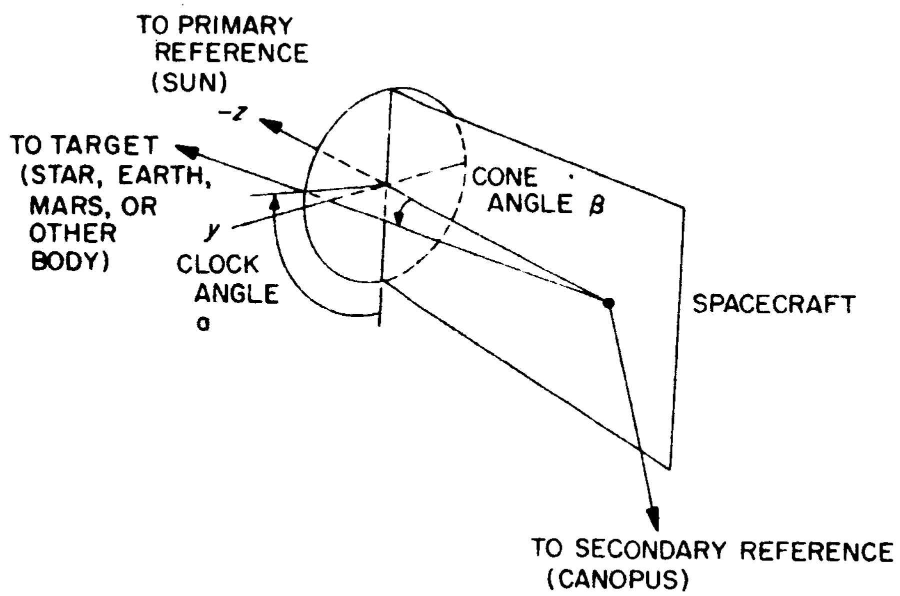


Figure 4-25. Clock Angle of Earth Measured at Mars vs Calendar Date



CONE ANGLE (β)	THE ANGLE, MEASURED AT THE SPACECRAFT, BETWEEN THE SUN LINE AND a LINE THROUGH THE OBJECT OF INTEREST
CLOCK ANGLE (α)	THE DIHEDRAL ANGLE, AT THE SPACECRAFT, FROM THE SUN-SPACECRAFT-CANOPUS PLANE TO THE SUN-SPACECRAFT-OBJECT PLANE MEASURED IN A CLOCKWISE DIRECTION WHEN LOOKING TOWARD THE SUN

Figure 4-26. Attitude Reference Coordinate System for a Sun-Canopus Oriented Spacecraft

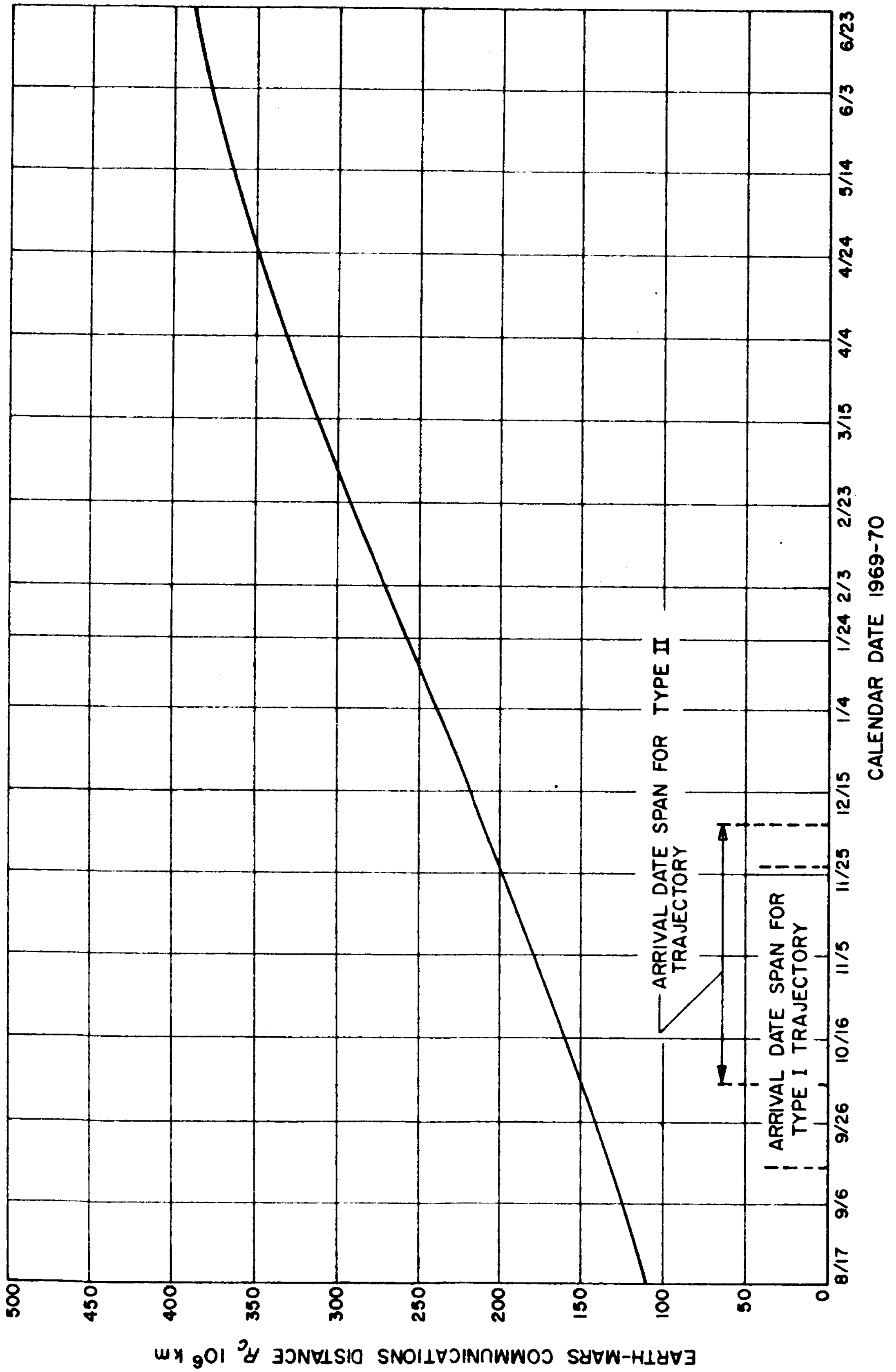


Figure 4-27. Earth-Mars Communications Distance vs Calendar Date

the nomenclature of Section IV-B, EPD-139, Volume I, it can be shown that the time rate of change of the periapsis altitude (due to the above effect) is given approximately by

$$\dot{h}_p = -\frac{15}{8} \frac{aen_3^2}{n} \sqrt{1-e^2} \sin 2\omega \sin^2 I,$$

where n_3 is the mean angular motion of the third body (the Sun) and n is the mean angular motion of the orbiter. It can be seen that $\dot{h}_p < 0$ only when ω is in the first or third quadrant. For typical Mars orbits, however, ω moves very slowly through all quadrants and would not remain fixed in either the first or third quadrant for a long enough period of time to be of concern unless initially placed in one of these two quadrants with an orbit inclination very close to 63.4 deg. For this latter situation, it happens that most initial values of ω tend to lie in the second and fourth quadrants which results in $\dot{h}_p > 0$. If the very worst situation did arise, however, the maximum value of $\dot{h}_p$ would only be about 5-10 meters/day such that, even then, it would require some 25 to 50 years for the periapsis altitude to diminish by 100 km. This third-body effect is of much greater importance for lunar satellite orbits where the effect of the Earth upon a lunar orbiter exceeds the effect of the Sun upon a Mars orbiter by more than two orders of magnitude.

2) During a portion of each orbit, it is desirable that the Sun-orbiter-Planet angle exceed 120 deg in order to satisfy experiment lighting conditions. It is desirable that this condition be maintained for a period of 180 days to allow for a seasonal change beneath a Mars orbiter. The dynamic oblateness of Mars provides a very simple and natural means of satisfying this condition by causing the satellite orbit to process at an "optimum" rate which may be determined, given the guidance errors and orbit period. Generally speaking, Type I orbits (those established from Type I transit trajectories; see Chapter 4, Section II, B) must precess in the same direction as the Sun and at such a rate that the orbit plane actually overtakes the Mars-Sun line, whereas Type II orbits precess more slowly and are overtaken by the Sun or, in some cases, may actually precess in the opposite direction to the Sun's motion in order to improve the lighting conditions at a faster rate and yet still guarantee 180 days of continuously acceptable lighting. The nominal elliptic orbit discussed in paragraph 6 exhibits this latter characteristic. A detailed discussion of the selection of optimum precession rates for Mars orbiters may be found in Section IV-B of EPD-139, Volume I. As many of the figures in that section were based upon a periapsis altitude of 1000 km, Figures 4-28 through 4-31 show a higher periapsis altitude of 2000 km.

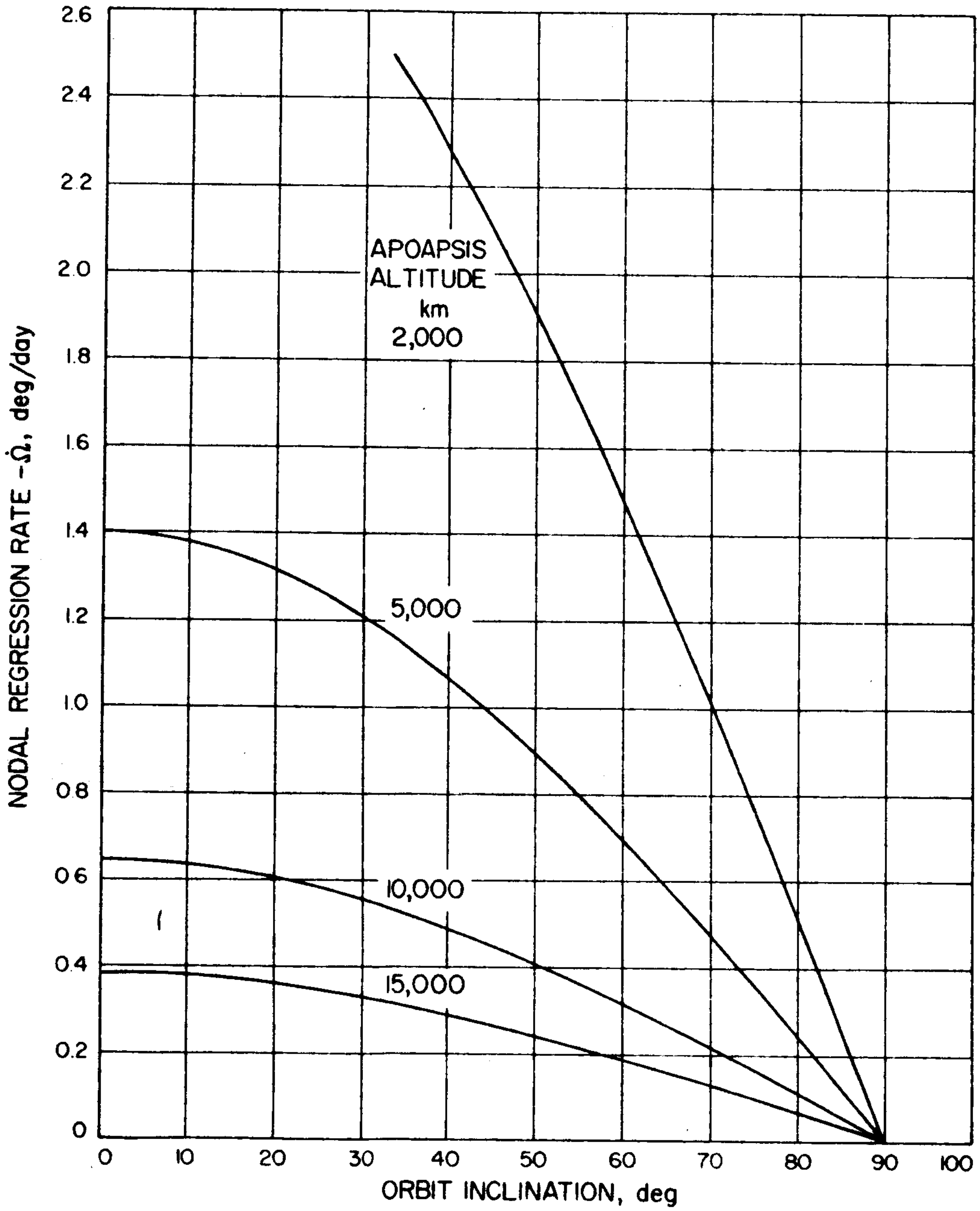


Figure 4-28. Nodal Regression Rate vs Orbit Inclination for a Mars Satellite
Periapsis Altitude 2000 KM

C. ORBITER/CAPSULE TRAJECTORY GEOMETRY

1. Trajectory Constraints

Trajectory constraints which primarily affect the design of the orbiter and capsule trajectories are the subject of this paragraph. Trajectory constraints pertaining to the launch or transit phases of flight were previously covered in Section IV-B-2. Listed first are those conditions which must be satisfied by the bus approach trajectory and subsequent satellite orbit along with a brief description of how each condition may be controlled. The constraints have been listed in a generally descending order of importance.

1) As the orbiter will probably not be sterile, it is essential to maintain a 10^{-4} probability that it does not contaminate Mars. One method of assuring this is to guarantee that its orbit will not decay due to atmospheric drag for some adequate period of time, approximately 50 years, assuming a model of the Mars atmosphere which is ten times more dense than Chamberlain's* model. Lifetime curves are presented in para 5, where it is shown that a circular orbit at an altitude of about 1800 km would decay in 50 years, based upon an effective in-orbit ballistic coefficient ($M/C_D A$) of 0.25 slug/ft^2. This effective value for the ballistic coefficient represents an average value over all possible orientations of the orbiter and may be used for either a solar-powered or an RTG-powered orbiter. Assuming a 3σ guidance error of 500 km (including both orbit uncertainty and execution errors), it would appear unwise at this time to consider any nominal circular orbit altitude lower than approximately 2300 km. In the case of a fairly elliptic orbit ($e \simeq 0.5$), which spends less time passing through the atmospheric fringes, the minimum periapsis altitude which yields a 50-year lifetime can be lowered to about 1300 km. Allowing for the guidance error, this would suggest a minimum nominal periapsis altitude of about 1800 km for an elliptic orbit whose apoapsis altitude is 10,000 km. For planning purposes at this time it seems prudent to assume a total guidance accuracy no better than about 500 km, so that the above minimum nominal periapsis altitudes should only be lowered in the presence of new information which indicates lower densities in the upper Mars atmosphere.

Another effect in addition to atmospheric decay which can cause the periapsis altitude to diminish is the gravitational influence of the Sun upon an elliptic orbit which is oriented in a particular manner about an oblate planet. Using

*See Page 4-82

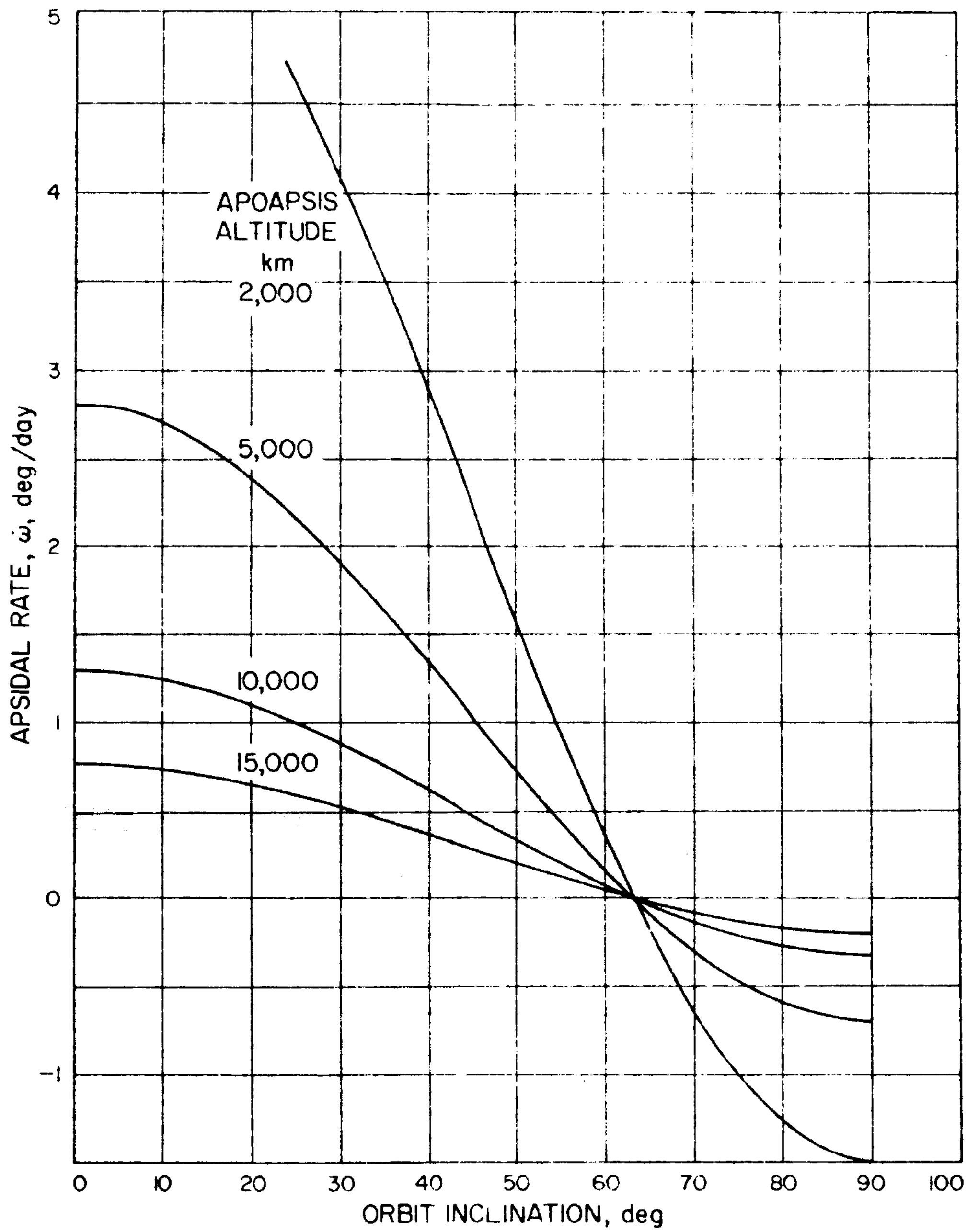


Figure 4-29. Apsidal Rate vs Inclination for Various Satellite Orbits About Mars
Periapsis Altitude - 2000 KM

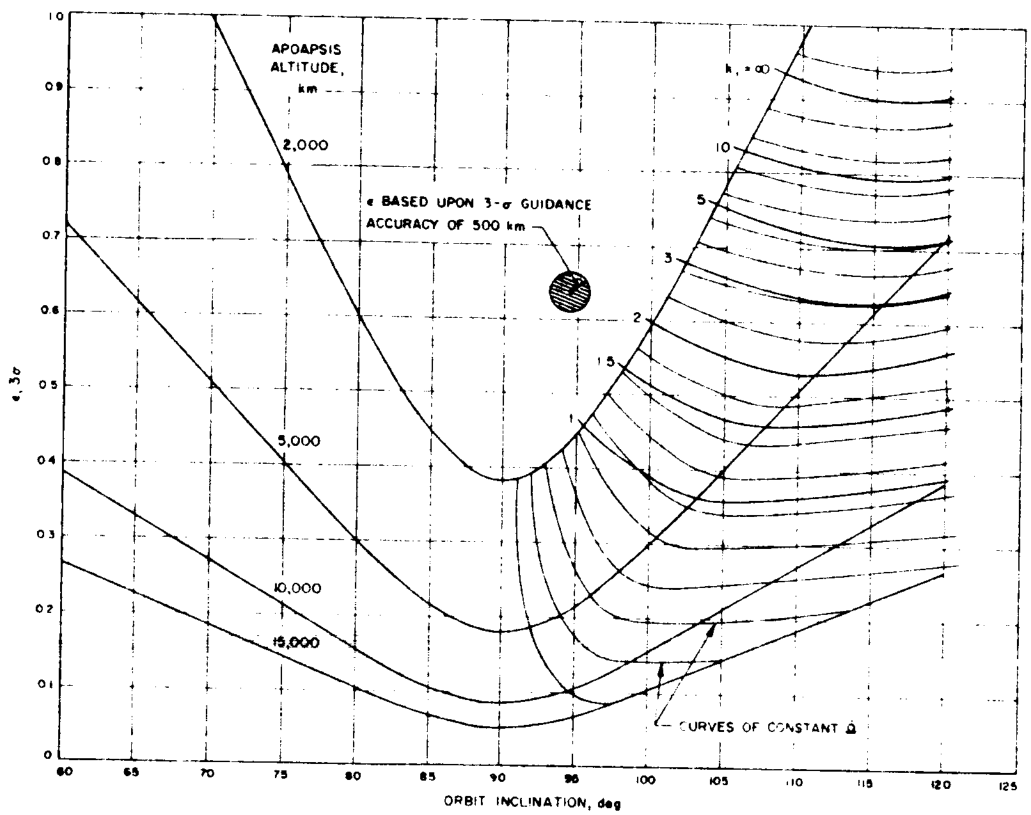


Figure 4-30. Selection of "Optimum" Inclination for Mars Satellite Orbits Periapsis Altitude = 2000 KM

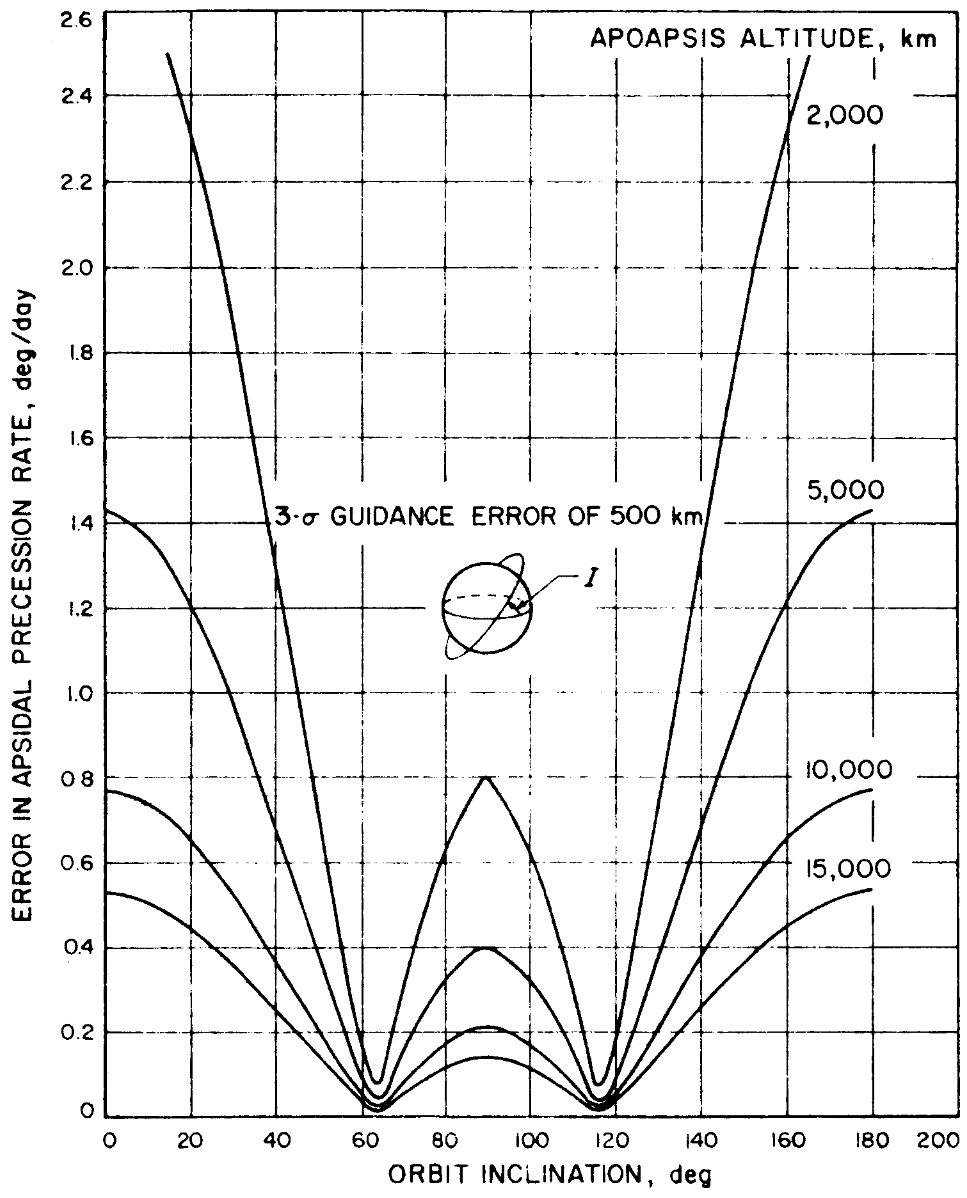


Figure 4-31. 3σ Variation in Apsidal Precession Rate for Mars Orbits with Periapsis Altitudes of 2000 KM

3) It is desirable that the orbit inclination with respect to the Mars equator be maintained above 60 deg in order not to lose too much coverage of the Mars surface. Fortunately, most optimum precession rates may be achieved with inclinations in this region so that a conflict does not exist here. It would be desirable to lower the inclination below 60 deg only if a significant increase in the capsule-to-orbiter data transmission capability could be realized. This consideration is discussed in more detail in the latter portion of paragraph 6.

4) For elliptic orbits, it is desirable to have the subperiapsis point as near the Martian equator as possible. Because planetary approach asymptotes tend to lie within 20 deg of the ecliptic plane, then, for optimum periapsis-to-periapsis transfers, the initial subperiapsis point is usually never closer to the Martian equator than about 30 deg. Due to oblateness, for inclinations above 63.4 deg, the periapsis point normally moves away from the illuminated equator for Type II orbiters and towards the illuminated equator for Type I orbiters, because the line of apsides moves in a direction opposite to that of the orbiter. A slight exception to this rule-of-thumb could occur for the 1969 Mars opportunity, where some of the Type I trajectories have near-Mars approach geometries similar to the Type II trajectories. A reasonable policy would be to select elliptic orbit inclinations reasonably close to 63.4 deg consistent with providing adequate surface coverage, optimum orbit-plane precession rates, reduced primary-body occultation times, and acceptable third-body effects upon orbiter lifetime. An investigation of apsidal advance due to atmospheric drag has not been completed, but it is not expected to be a significant effect at the orbital altitudes which permit lifetimes in excess of 50 years.

5) It is desirable to minimize the time during each orbit when the Sun, Earth, and certain attitude-reference stars (such as Canopus and Vega) are occulted by the planet. If it is more important that the Sun and Earth not be occulted, rather than the reference stars, one would tend to select orbit planes which were oriented in a generally perpendicular rather than parallel manner with respect to the ecliptic plane. As the Martian equatorial plane is inclined only some 25 deg to the ecliptic plane, the above considerations are compatible with previous conditions (2) through (4). It should be noted, however, that the Type II approach geometry results in better view of the Earth from the orbiter than does the Type I geometry.

6) It is desirable to phase the orbital period in such a way with respect to the planet day that, by the end of some prescribed number of days, the loci of orbital tracks will be equally spaced in longitude. It would also be desirable to have this mapping process trace out the planet surface in a reasonably well-ordered sequence. It can be shown that, if the orbital period, p, is related to the planet period, P, by

$$p = (1 \pm KN^{-1})^{-1} P,$$

where I is any integer and K is any integer such that K and N are not both divisible by a common integer, then the orbital tracks will be equally spaced at the completion of exactly N planet days. In the presence of precession of the orbit plane about the planet axis, an effective planet period should actually be used and would be given by

$$P = 2\pi\,(\sigma - \dot{\Omega})^{-1},$$

where σ is the planetary sidereal angular velocity and $\dot{\Omega}$ is the orbit-plane precession rate. Typical circular polar orbits which result in 30 day "even-mapping" may be found at altitudes of 2190, 2300, 2350, 2420, and 2500 km (there are many others). For high-resolution elliptic orbits with 1800 km periapsis altitudes, 30 day even-mapping may be obtained by selecting apoapsis altitudes of 9570, 10070, 10880, 11170, and 12[illegible]0 km, to mention a few. Obviously, guidance errors will prevent these particular orbits from being realized exactly, but it may still be worthwhile to aim for them nominally. The importance of trying to achieve even-mapping orbits generally increases as the orbital period increases and/or as the number of days, N, for completion of the mapping decreases. The determination of the optimum value of K, given N, would be based upon minimizing the mean departure from equally spaced orbital tracks given some probability distribution on the orbital period due to guidance errors. One might feel intuitively that P/p should not be too close to an integer but, beyond that point, the problem of selecting K becomes quite difficult.

7) For accurate reconstruction of the stereo pairs, it is necessary to know the actual orbit altitude to about 1 percent. This degree of orbit determination accuracy can be achieved within a time period of one or two orbits with either Earth-based two-way doppler data or from on-board measurements of planetary angular diameter and Sun-orbiter-Planet angle. By continuing the measurements for several orbits (typically 1 to 3 days), it appears possible to reduce the uncertainty in altitude to as low as 0.01 percent. Satellite orbit determination is discussed in Chapter 4, Section II, D.

8) At this time, it appears that the only constraints which should be placed upon the choice of orbits from the standpoint of Earth based determination of the satellite orbit are: (a) it is desirable that the angular rate of the orbit plane with respect to the Earth-Planet line-of-sight be nonzero, and (b) the orbit inclination with respect to the plane generated by the Earth-Planet line-of-sight should be nonzero (this latter plane is very close to the ecliptic plane). There is a very low probability that either of these situations would occur exactly, and slight departures from the above situations are generally acceptable.

Before discussing the capsule trajectory constraints, it should be noted that the philosophy of releasing the capsule during the approach phase of flight, rather than after orbit establishment, appears necessary in order to yield adequate orbiter and capsule payloads for performing worthwhile scientific missions with the Saturn I-B/SV launch vehicle. A secondary advantage is that, by causing the capsule to arrive at the planet in advance of the spacecraft bus, the atmospheric entry data can be relayed from the capsule through the bus to Earth before the bus is required to perform its retro maneuver. In this way, the entry phase of the mission could still be a success (without direct-link communication) even if (1) the bus fails to go into orbit, or (2) the bus does go into orbit but, for any of several possible reasons, does not thereafter communicate with the capsule. The most important capsule constraints will now be listed in an approximately descending order of importance.

1) The entry capsule must arrive at the planet sufficiently in advance of the approaching bus to guarantee reception of the capsule data acquired during the atmospheric descent. It would be further desirable if communications could continue for some minimum time after capsule landing in order to ensure either (1) redundant transmission of the entry data, (2) verification of the capsule's postlanding operational status, (3) the performance and transmission of some surface measurements, or (4) combinations of the above. If ΔT is defined as the time between bus periapsis passage and capsule main-parachute opening, then it currently appears that a ΔT of 1.5 to 2 hr should be satisfactory. If it is assumed that capsule-to-bus communication is first interrupted at the start of the bus attitude turns prior to the retro maneuver (some 30 to 40 min before periapsis passage), this would then leave 50 to 60 min (ΔT = 1.5 hr) or 80 to 90 min (ΔT = 2 hr) for capsule descent and postlanding surface time. The maximum descent time in the most dense model of the Mars atmosphere would be about 40 to 50 min, so that ΔT = 1.5 hr should be acceptable. In order to be on the safe side, however, ΔT = 2 hr has been selected at this time. Should the Mars atmosphere be less dense than currently assumed, maximum descent times may be considerably reduced, thus suggesting the possibility of selecting a smaller value for ΔT.

   It is desirable that the bus lie within ±45 deg of the capsule antenna axis at a few hours prior to entry so that the capsule subsystems can be monitored prior to the entry phase. This requirement will be met easily if sufficient guidance accuracy can be provided to allow the capsule to be spun-up and fired in a generally forward direction. Studies to date indicate that such a maneuver would provide adequate landing accuracy if the total capsule pointing error does not exceed about 10 mrad (1σ). If this pointing error is degraded by a factor of two, then it would appear wise to fire the capsule only some 30 to 45 deg forward of a lateral separation in order to satisfy both the landing accuracy requirements and the preentry communications geometry.

As this would change the capsule flight time by only 30 min, the remaining portion of the desired $\Delta \Gamma$ would have to be achieved by retarding the bus. In other words, at some time after the capsule velocity increment had been applied, the bus would perform an approach guidance maneuver which would not only correct the bus miss-components but would also change the bus time-of-flight.

2) It is desirable that the capsule be landed between 5 and 16 deg North latitude and between 288 and 300 deg East longitude. This area lies within Syrtis Major, which is of particular interest to the biologist. However, this region is only some 655 by 715 km in size on the surface of Mars and requires a high degree of guidance accuracy (see paragraphs 2 and 3). The actual size of this impact region in guidance space (impact-parameter or massless-target space) is about 810 to 850 km (see Figure 4-36). In order to achieve the maximum guidance accuracy associated with releasing an entry capsule prior to orbit establishment, the capsule should be deflected with the minimum possible velocity (lateral separation) and "extra" velocity should not be applied to change its flight time. For the magnitude of deflection distances (8,000-10,000 km change in the impact parameter of the approach trajectory) typical of a Mars orbiter/capsule mission, the impact dispersions due to capsule execution errors (predominantly pointing errors of the thrust axis as opposed to shutoff errors) do not begin to increase markedly until the flight time is changed by about 30 min or more. Additional information concerning the capsule aiming philosophy, execution errors, etc. may be found in paragraphs 2 and 3.

3) The requirement of maintaining the bus within a certain angle of the capsule antenna axis is discussed in paragraph 2.

4) The capsule will be designed to perform a safe atmospheric transit over the entire entry-angle range from the minimum capture angle to vertical entry at 90 deg. It turns out, however, that in order to land in the desired region and also satisfy the requirement of having the bus direction within ±15 deg of the capsule antenna axis during parachute descent, the entry angles will typically range from 60 to 75 deg. For the Type II 1969 Mars trajectories the Earth lies very near the capsule horizon at the time of landing. If it were desirable to have the capsule land in view of the Earth, this condition could be met by (1) removing the lighting constraint for Type II trajectories and allowing closer approach to the terminator, or (2) utilizing Type I transfer trajectories.

There are several considerations which affect the orbiter and capsule trajectories in an interlocking fashion. For example, it would be desirable to phase the capsule landing-site location, flight-time change, and orbital period in such a way that the first few satellite orbits exhibit the best passes (longest view times at high elevation angles) over the capsule which can be realized from the particular cyclic behavior associated with the landing-site latitude and orbital period. Figures 4-64 and 4-65 illustrate this point, as it can be seen that the best passes over the capsule do in fact occur shortly after encounter and repeat about every five days. As was mentioned earlier, it would be possible to increase the capsule-to-orbiter data capability by lowering the orbiter inclination, but this may be undesirable from the standpoint of the orbiter portion of the mission since (1) there would be reduced latitude coverage of the Mars surface, (2) precession rates would no longer be "optimum", and (3) Sun and Earth occultation times would be increased.

Table 4-3 summarizes some of the advantages and disadvantages of selecting various types of orbits for Type I and Type II trajectories to Mars in 1969. Considerations 2, 4, 5, and 7 are relatively independent of the mission year. Although it is feasible to design a satisfactory mission from either type trajectory, there appears to be a very slight advantage in using the Type II trajectories. Unfortunately, these trajectories exhibit a longer transit time which is a decided disadvantage from an equipment lifetime standpoint. In Table 4-3, the effect of selecting a given orbit upon each of the different constraints has been assigned a value from zero to ten, for the more important class of constraints, and from zero to five, for the less important constraints. A high value means that a given constraint has been satisfied in the best possible manner, while a low rating indicates that the particular constraint has been poorly satisfied or not satisfied at all. The aiming quadrant letters indicate whether the orbit has been established by passing over the northern (N) or southern (S) hemisphere of Mars and whether the orbital motion is direct (D) or retrograde (R). Table 4-3 indicates that, for the assumed 5 and 10-point value categories, the most attractive orbiter missions for Type I transits are high-inclination SR and low-inclination ND whereas, for Type II transits, high-inclination ND, SD, SR, and low-inclination ND are all attractive. The nominal elliptic orbit discussed in paragraph 6 is high-inclination ND and assumes a Type II transit.

2. Capsule and Orbiter Aiming Points

It has been assumed that the main spacecraft is on an approach trajectory whose near-planet plane of motion has the desired orbiter inclination and periapsis distance so that the capsule must be deflected towards the planet in order to achieve the desired impacting trajectory. The justification here is that it would be virtually impossible to guarantee a probability as high as 0.9999 that the unsterile bus could be deflected from an impacting trajectory which it may have been placed upon by one or more successful midcourse corrections performed earlier in the flight when all systems were operating properly. Admittedly, one might decide in view of this to not place the bus upon an impacting trajectory until a few days before encounter such that, if the system were still functioning at that late date, there would be a greater chance (still probably less than 0.9999) of being able to return the bus to a miss trajectory. This alternative would have the disadvantage in that if the capsule had

Table 4-3. Mars Orbit Selection Consideration (Emphasis on 1969 Opportunity)

	Type I transits								Type II transits							
	Low[a] inclinations				High inclinations				Low inclinations				High inclinations			
	ND	NR	SD	SR	ND	NR	SD	SR	ND	NR	SD	SR	ND	NR	SD	SR
Sun and Earth occultation times (10)	6	3	3	4	6	4	5	5	6	3	3	6	8	7	6	9
Surface coverage for stereo photography (10)	5	4	4	5	10	9	9	10	5	4	4	5	10	9	9	10
Initial location of sub-periapsis point (10)	7	3	8	2	5	2	6	5	10	2	9	3	6	2	6	3
Capsule-to-orbiter data transmission capability (5)	4	4	4	4	2	2	2	2	4	4	4	4	2	2	2	2
Motion[b] of sub-periapsis point relative to Mars equator and illuminated area (5)	4	4	3	4	3	3	2	3	3	4	4	4	2	1	3	1
Uninterrupted acceptable lighting for 150 days (5)	1	2	1	2	0	5	0	5	3	2	3	2	4	5	4	5
Earth-based satellite orbit determination conditions (5)	3	1	1	3	3	4	4	3	3	1	1	3	3	4	4	3
Total Rating	30	21	24	24	29	29	28	33	34	20	28	27	35	30	34	33

ND - North Direct, NR North Retrograde, SD South Direct, SR South Retrograde

(a) inclinations less than 60 degrees

(b) due to oblateness effect upon orbit

been designed without a propulsion system, any failure of the bus to achieve its temporary collision course by a late near-planet maneuver would immediately nullify the capsule phase of the mission.

Due to the sterilization constraint, it is evident that the bus should never be placed upon an impacting trajectory. As a matter of fact, even for a miss trajectory, the probability of a bus explosion should be less than 10^{-4}, because it is quite likely that the shotgun effect which such an explosion might exhibit would be almost certain of hitting the planetary atmosphere with at least one unsterile fragment. It is very difficult to estimate the probability that such a fragment might actually contaminate the planet either during its atmospheric passage or after landing (if it were to reach the surface).

As discussed previously, the decision to separate the capsule before going into orbit is based primarily upon maximizing the capsule and orbiter payloads at the planet when performing a joint orbiter/capsule mission with the Saturn I-B/SV. Also, it seems wise to not require the success of the entry mission to depend partially[1] upon the success of the orbiter mission. It should be pointed out, however, that if more injected payload were available, and if a high confidence level in successfully establishing the orbiter could be assumed, it might be wise to consider releasing the capsule from the satellite orbit whose characteristics can be determined quite accurately from Earth-based and/or on-board measurements collected over a period of only a few days following encounter (see Chapter 4, Section II, D). This technique will receive considerable treatment in the Apollo project, where rigid accuracy requirements are essential. Also, by releasing the capsule from orbit, lower entry velocities and better control of the entry angle could be achieved. These factors could lead to a more optimum capsule design and lighter structural weight.

In selecting the bus near-planet trajectory, two basic philosophies may be considered: either (1) the nominal aiming point distance above the Martain surface is chosen upon the basis of performing an approach correction in order to realize the maximum guidance accuracy without an orbit-trim maneuver, or (2) a crude orbit is first established at a conservative distance from the surface and, after determining the satellite orbit characteristics accurately (and leisurely), the orbital periapsis is lowered to a more desirable value. Orbiter aiming points were selected by the first approach. Additional study is in progress on orbit-trim aiming philosophy.

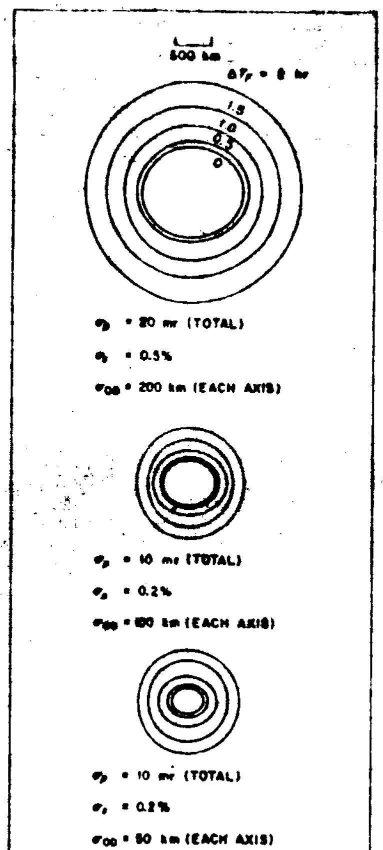
However, the latter approach should result in a more accurate control of the periapsis altitude and could also facilitate the location of the subperiapsis point. This approach could alleviate the initial concern of impacting Mars with an unsterile bus at encounter (due to the

1) Not totally, as there may be a direct capsule-to-Earth link.

greater aiming distance) but would, perhaps, increase the tension when it became time to execute the orbit-trim maneuver. The orbit-trim method would realize less payload in orbit and would require the successful operation of a post-orbit-establishment maneuver. The performance of an approach maneuver might bring to light some important nonstandard effect, the knowledge of which could be a very critical input for the performance of a successful retro maneuver. The above considerations represent some of the potential advantages and disadvantages of the two guidance philosophies.

Figure 4-37 presents Mars as seen in guidance or impact-parameter space. The asymptote to the approach hyperbola is normal to the plane of Figure 4-37, which describes the mapping from massless-planet aiming coordinates to actual surface coordinates on the Martian surface. Due to the gravitational focusing of finite-velocity masses, the effective capture radius of Mars is increased from its actual physical radius of 3415 km to about 5100 km for the nominal Type II approach velocity of 4.237 km/sec. The zero-reference longitude shown in Figure 4-37 is defined only by the vertical-impacting North-South meridian and not by any special topological features. Figure 4-36 represents a transparent overlay which indicates the size of the desired capsule landing zone previously defined in paragraph 1. The landing zone has been arbitrarily centered on various North-South meridians to illustrate its overall size and shrinkage as the aiming point is moved away from the center of Mars towards shallow entry. It should be noted that the desired landing region (Syrtis Major) is not rectangular in appearance, but the zones of Figure 4-36 represent a relatively large rectangular region which has been inscribed within Syrtis Major. The desired landing region is therefore somewhat larger than the zones of Figure 4-36. Figure 4-35 is a transparent overlay which illustrates the effect of various bus-capsule communication constraints upon the selection of the aiming point for the capsule trajectory. Those capsule landing areas located within ±30 deg of the zero-reference longitude line satisfy the constraint that the bus lie within 45 degrees of the capsule antenna axis at the start of parachute descent. After this time, however, as the sub-bus point continues to move into the northern hemisphere of Mars [1], it is no longer possible to satisfy this 45-degree condition. Fortunately, however, the bus-capsule distance is rapidly diminishing so that a 60 to 70 deg off-antenna-axis angle may be acceptable when the communication distance has decreased to 10,000 km. It has been assumed that the capsule antenna axis is directed along the local vertical during the parachute descent phase and after landing. Figure 4-35 shows the locus of those aiming points which will result in an off-antenna-axis angle of 70 deg at the end of the initial communication period which occurs at the start of the bus turns prior to the retro maneuver. This constraint narrows the location of the landing zone to near the -20 deg longitude line. This limitation may be alleviated by (1) using southerly bus aiming points for Type II trajectories, (2) by using northerly bus

1) An elliptical orbit should be established by passing over the northern hemisphere, for Type II trajectories, in order to have periapsis occur on the lighted side, reasonably near the Martian equator.



3 σ (99%) DISPERSION ELLIPSES FOR 1969 MARS MISSION FOR DIFFERENT FLIGHT TIME CHANGES OF CAPSULE

Figure 4-32. 3σ (99 Percent) Dispersion Ellipses for 1969 Mars Mission for Different Flight Time Changes of Capsule

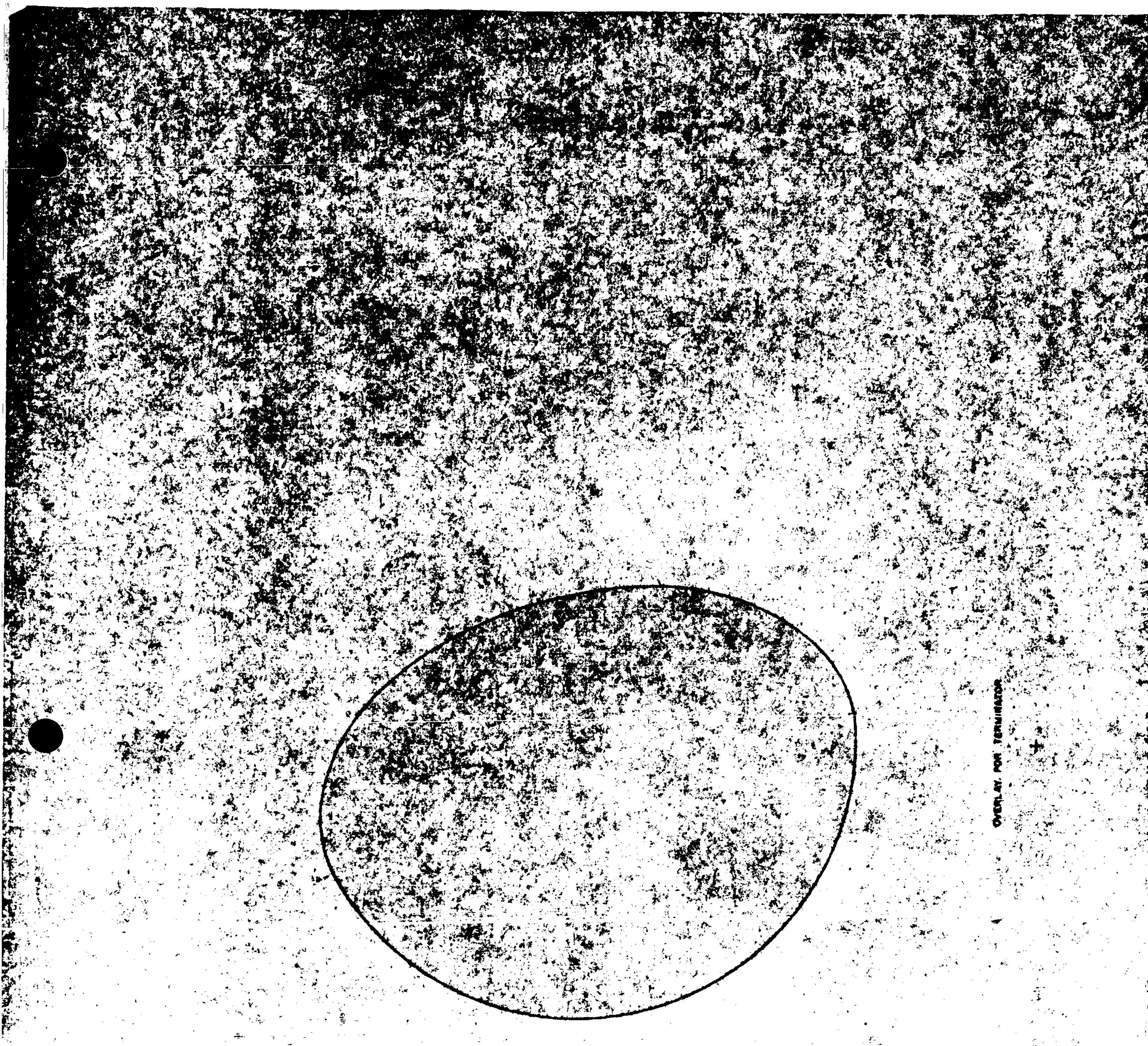


Figure 4-33. Dark Side of Mars

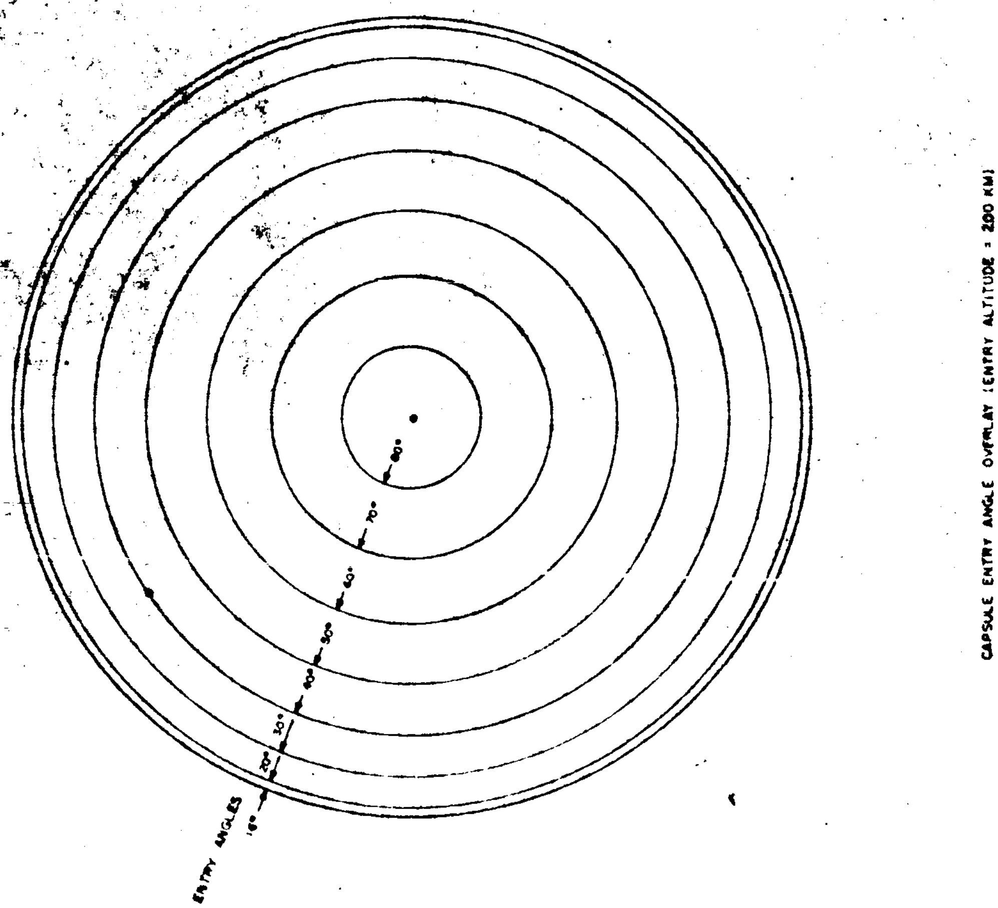


Figure 4-34. Capsule Entry Angle

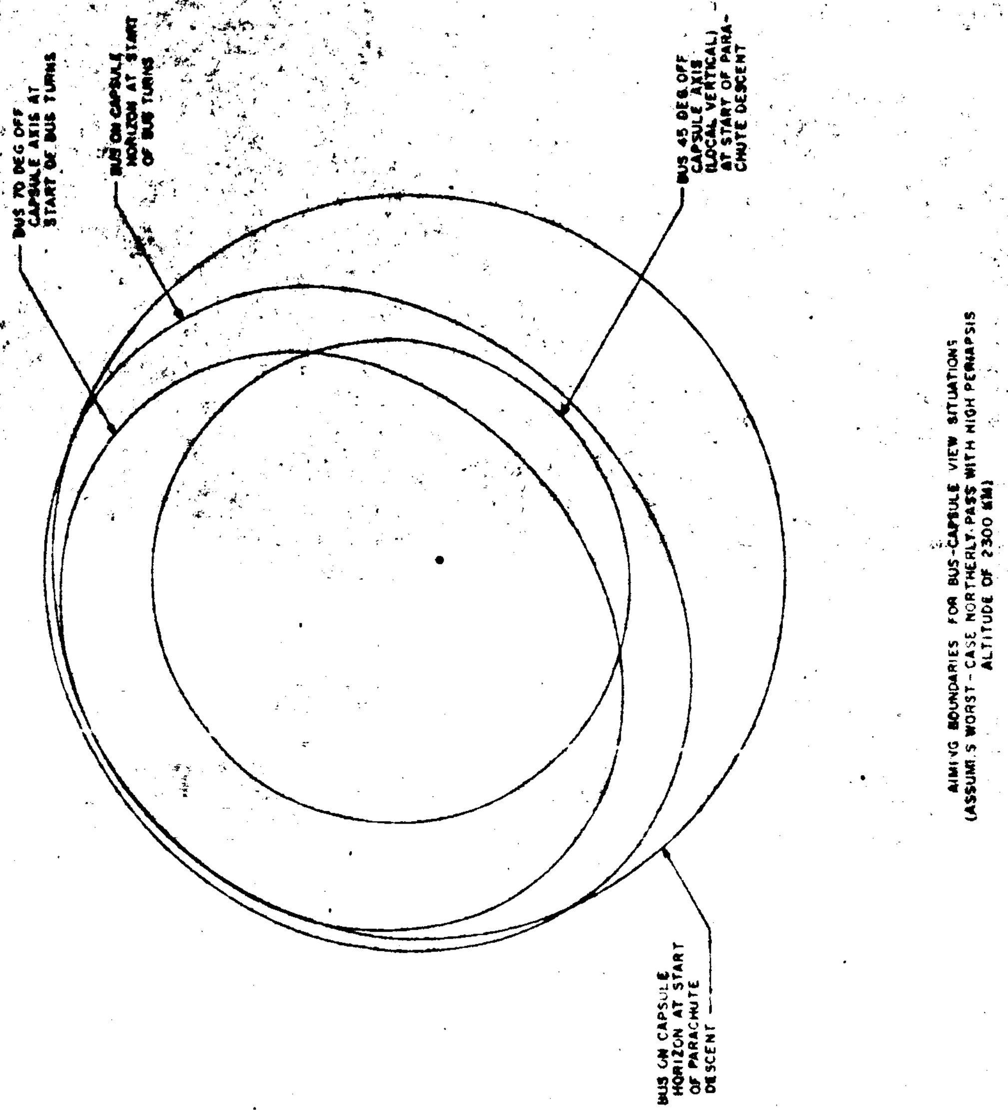


Figure 4-35. Aiming Boundaries for Bus-Capsule

Figure 4-36. Capsule Landing Zones Centered on 20-Deg Spaced Longitude Meridians

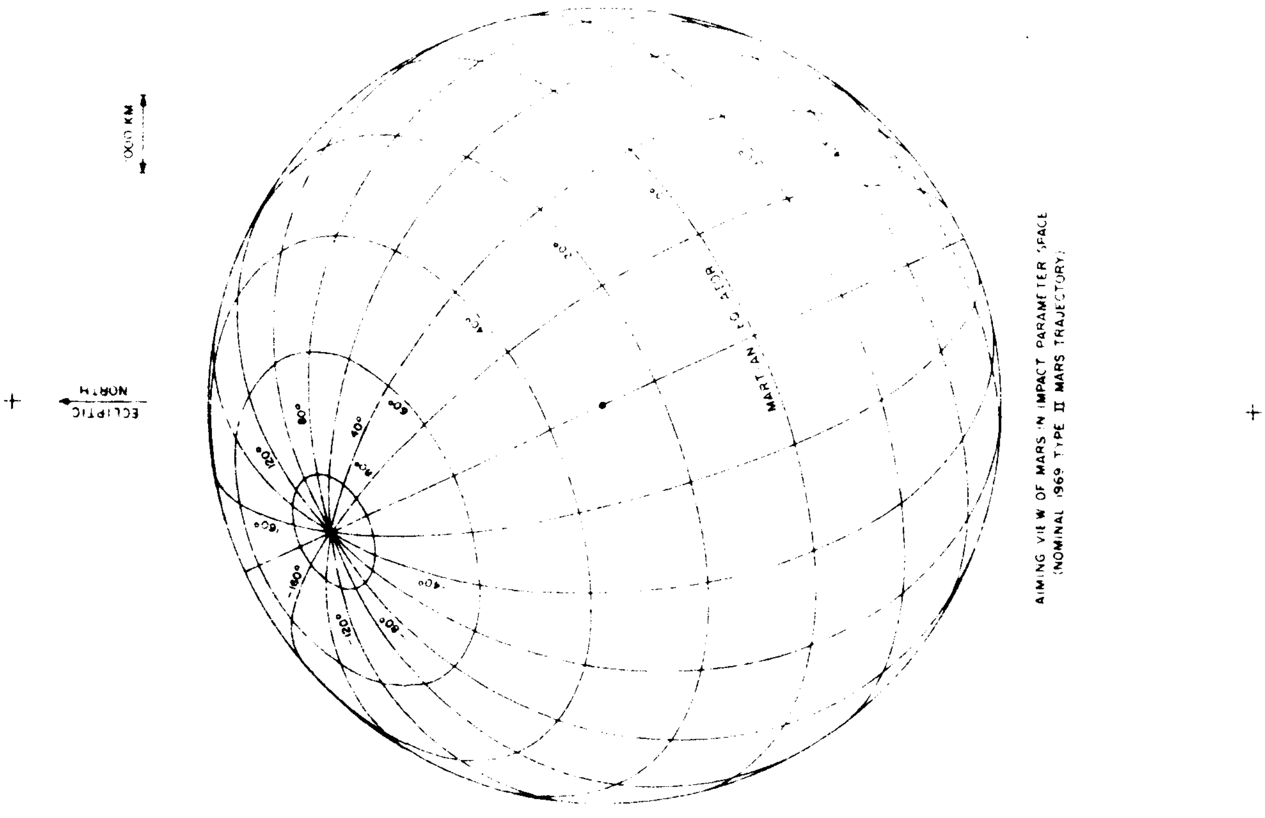


Figure 4-37. Aiming View of Mars in Impact Parameter Space

aiming points for Type I trajectories, (3) by violating the lighting constraint and allowing approach along the terminator, or (4) by selecting low orbit inclinations. Method (1) is feasible for circular or near-circular orbits but not for very elliptic orbits where the sub-periapsis point location becomes important. Figure 4-35 indicates a worst-case situation for satisfying the capsule-to-bus communication geometry. Figure 4-38 illustrates the bus and capsule locations at entry and at the start of bus turns for both a northerly and southerly bus pass. The southerly pass is significantly better from a preorbit capsule-to-bus communications standpoint. Regarding capsule event times, both the terms "entry" and "start of parachute descent" have been used in this report. The term "entry" refers to the time when the capsule begins to experience the presence of the atmosphere. For example, one might select the time when the capsule deceleration reaches some small but measurable value. In this particular section of the report, an entry altitude of 200 km was selected somewhat arbitrarily to represent an upper-bound for all reasonable atmospheric models. The "start of parachute descent" usually occurs at a predetermined Mach number and, although the time of this event depends upon the atmospheric model, it is expected to occur very shortly (within one minute) after entry.

Figure 4-34 illustrates the entry-angle overlay for the various aiming points. In order to satisfy the communication constraint, the capsule entry angles would typically lie in the region of 60 to 75 deg or relatively steep entry. Steep entry is not severe for Mars, however, and it is advantageous to employ rather direct entry both from the standpoint of increasing the chances of landing within Syrtis Major and, in the event of a nonstandard guidance error, of landing at least somewhere on Mars. Figure 4-33 shows the dark side of Mars for the nominal Type II approach geometry. Typically, the capsule would land on the dark side but with the Earth near the Mars horizon (see Figure 4-60). By about 3 hr after landing, capsule sunrise would have occurred, and the Earth would be some 40 deg above the capsule horizon. For most of the Type I trajectories (see Chapter 4, Section II, B) the capsule would land both in the sunlight and in view of the Earth.

The following discussion considers the location of the capsule release maneuver, the capsule velocity requirements necessary to deflect the capsule trajectory and alter its time of arrival and, finally, the target dispersions which can be expected from execution errors in performing the capsule maneuver.

Previous approach-guidance studies (see JPL Technical Memorandum 312-316) have indicated that, for fixed angle biases or rapidly drifting biases, the orbit knowledge does not continue to improve very significantly after 2 to 3 days before encounter until reaching the close vicinity of the planet. This situation results from the use of a planet-center-finding device which exhibits a relatively fixed uncertainty in target miss. In the case of a slowly-drifting angle bias, however, the orbit knowledge is degraded but does continue to improve with decreasing distance from the planet. The TM 312-316 studies assumed a very

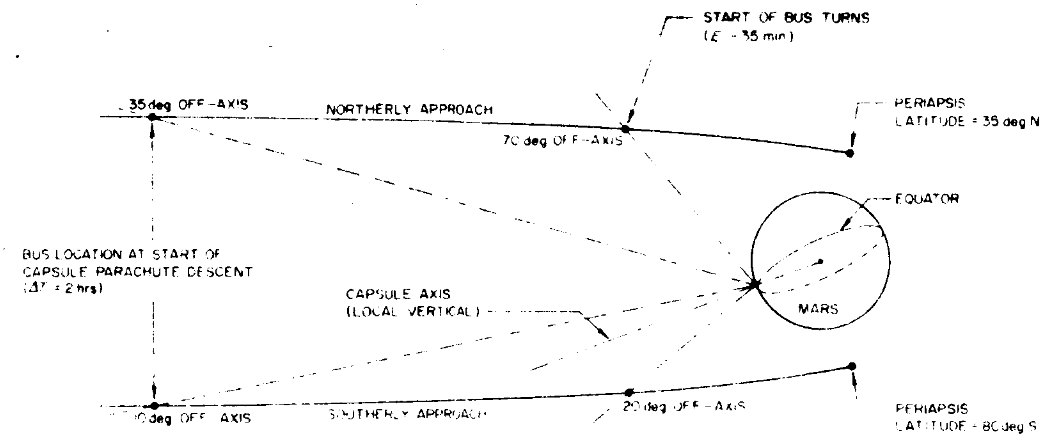


Figure 4-38. Angle from Capsule Antenna Axis to Bus at Start and End of Communications Period for Northerly and Southerly Passes. (Nominal 1969 Mars Type II Trajectory)

pessimistic a priori knowledge of the miss (2500 km rms uncertainty) at the beginning of the approach phase. Referring to Figures 4-73 and 4-74, a more realistic a priori estimate of the miss would be about 100-200 km, so that the improvement in orbit knowledge for a slowly-drifting bias error would not be as marked. It would, therefore, probably be wise to release the capsule at a reasonable distance from the planet because (1) a smaller capsule velocity increment would be required and (2) a better balance between orbit-determination errors and capsule maneuver execution errors would be realized. In other words, regarding (2), there would be little sense in releasing the capsule very near the planet where the orbit uncertainty was, for example, 50 km while the execution errors were expected to result in a 200 km uncertainty in target miss. All things considered, it currently seems appropriate to perform the capsule maneuver in the vicinity of one million kilometers from Mars. This occurs 2 to 3 days before encounter, depending upon the approach speed.

Figures 4-39 through 4-41 are based upon performing the capsule maneuver at 10^6 km but may be suitably scaled for a different release distance, if desired. Figure 4-39 illustrates the capsule velocity requirements necessary to achieve various arrival time differences, ΔT, between the bus and capsule trajectories. For a ΔT of 2 hr typical capsule velocity requirements range from about 100 to 150 m/sec, with a maximum requirement of 180 m/sec. As a fixed velocity increment will probably be used, this implies a variable release distance from the planet (due to the variation in approach speed). From Figure 4-40 the maximum bus-capsule communication distance would be 56,000 km based upon a ΔT of 2 hr and a maximum approach velocity of 5 km/sec. By the start of the bus turns (prior to retro), the bus-capsule communication distance will have diminished to about 10,000 km. Finally, Figure 4-41 illustrates the elevation angle of the bus measured with respect to the capsule local horizontal plane at entry. The elevation angle of the bus is simply the complement of the off-capsule-antenna-axis angle. It is important to realize that Figure 4-41 represents the largest departure of the bus from the capsule antenna axis and occurs for high northerly passes by only a small set of the Type II trajectories. However, as these passes result in attractive elliptic orbiter missions, the capsule-to-bus communications and data-handling system should be designed to handle these conditions.

It should be noted that it is not generally possible to select the capsule entry and landing conditions arbitrarily. In other words, such parameters as entry angle, entry velocity, release point, and landing-site latitude must be compatible. The basic entry geometry is illustrated in Figure 4-42, where it may be seen that two landing points can each satisfy the entry conditions. The "entry circle" represents the locus of equal entry angles and is symmetric about the subrelease-point which is the vertical entry or 90 deg entry angle location. However, it may be seen from Figure 4-42 that, had a landing-site latitude of less than about 10 deg South or greater than 80 deg North been selected, it would not have been possible to land at such a site location and still maintain the particular entry angle associated with the entry circle of Figure 4-42. An automatic trajectory computation program[1] has been used

1) Planetary Orbiter/Split-Capsule Study Program, described in JPL TM 312-271, dated 2/4/65.

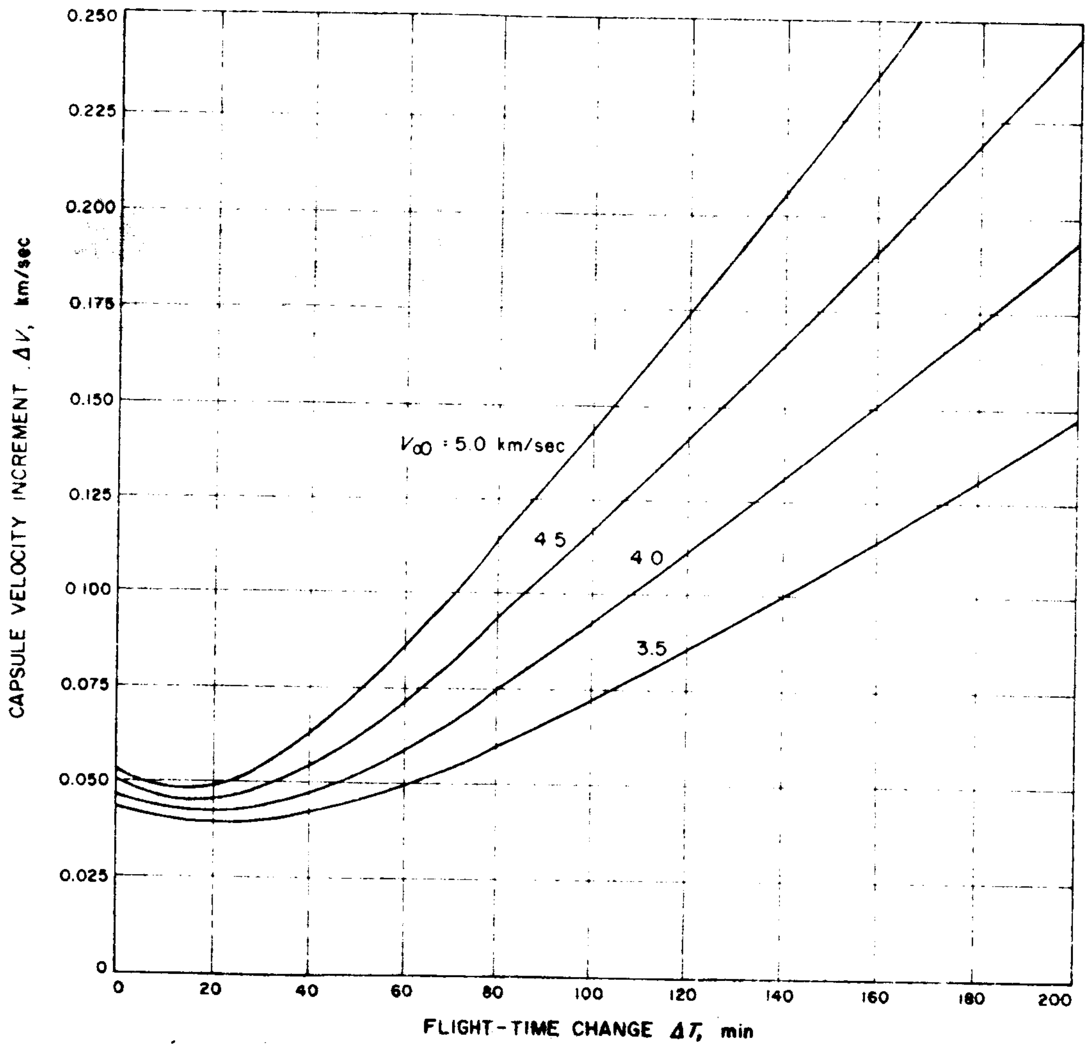


Figure 4-39. Capsule Velocity Increment Necessary to Achieve Flight-Time Change

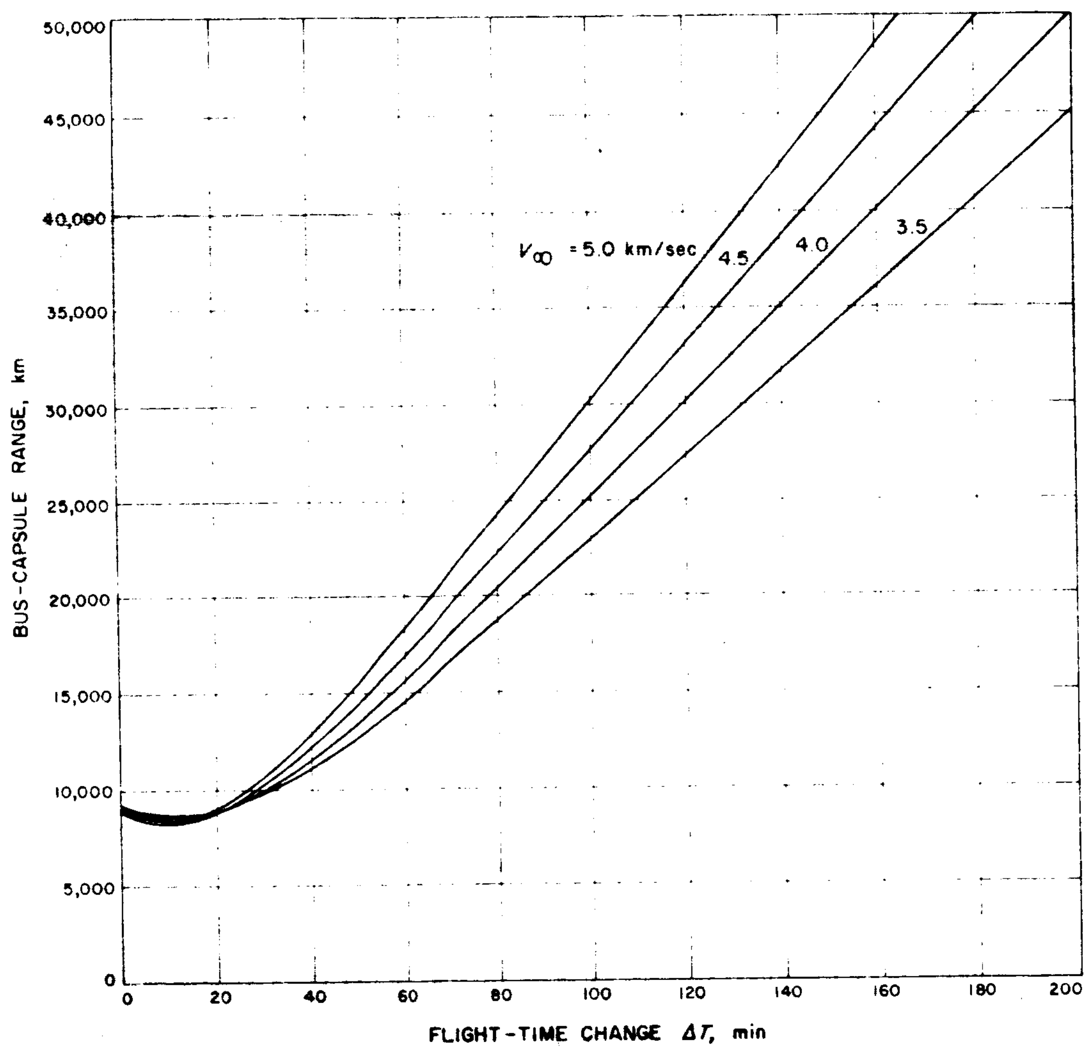


Figure 4-40. Bus-Capsule Range at Entry vs Flight-Time Difference

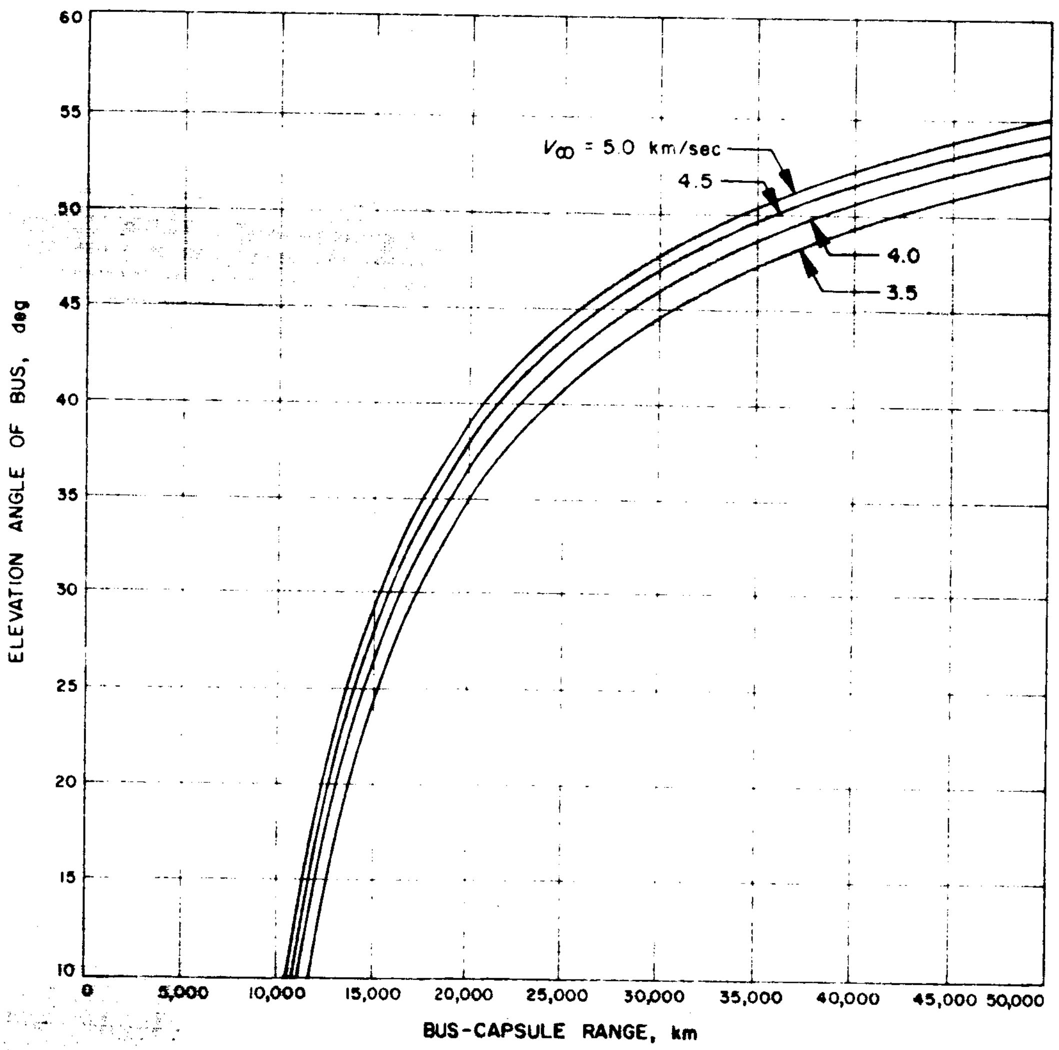


Figure 4-41. Elevation Angle of Bus at Capsule Entry vs Bus-Capsule Range

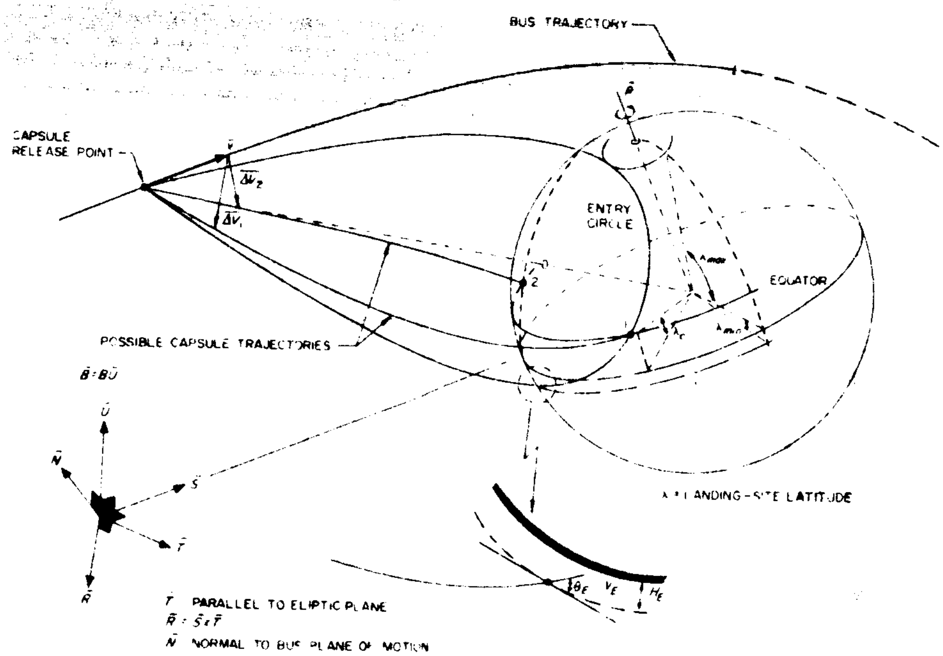


Figure 4-42. Basic Planetary Entry Geometry

which makes allowances for incompatible entry conditions by changing one or more of these conditions to satisfy the desired situation as closely as possible. Figure 4-43 illustrates the minimum possible capsule velocity increment (lateral separation) necessary to land at 10 deg North latitude as a function of the desired entry angle. The geometric impossibility of landing at this latitude for steep entry angles may be understood by referring to Figures 4-36, 4-37, and 4-34.

3. Effects of Guidance Accuracy Upon Capsule Entry and Landing Conditions

This discussion considers the guidance capability which may be expected for the capsule trajectory. It has been estimated that the direction of the capsule velocity increment can be controlled to an accuracy of about 10 mrad (combined rms pointing error), the magnitude of the velocity increment can be controlled to about 0.2% (1σ), and that the orbit determination uncertainty will be approximately 50-100 km (1σ about each of two orthogonal miss directions). Figure 4-32 shows the dispersion ellipses associated with three different error models and is used as an overlay to Figures 4-36 and 4-37. The worst error model, which assumes σ_p = 20 mrad, σ_s = 0.5%, and σ_{OD} = 200 km, is felt to be conservative while the other two models may be more realistic when it is realized that Voyager missions probably will not occur before 1969. Figure 4-32 is overlayed in such a manner that the minor axis of the dispersion ellipse is aligned with the orbit-plane edge. For example, in the case of a polar orbit, the major capsule dispersions would be in longitude, if the aiming point were the landing area centered on the zero-longitude meridian. Figure 4-44 illustrates the probability of impacting the desired landing area as a function of the change in capsule flight time, for the three assumed guidance-accuracy models. As Syrtis Major is larger in latitude spread than longitude, Figure 4-44 also shows the probabilities associated with trying to impact a region of twice the latitude dimension than the prescribed area. Conclusions which can be drawn from Figure 4-44 are that (1) if either of the two accurate error models may be confidently anticipated, then the capsule should provide the additional velocity required to change its arrival time by the desired two hours, or (2) if the conservative error model becomes more "realistic", then the capsule should be fired between [illegible] and 45 deg forward of a lateral separation such that its flight time would be altered by only 0.5 hr, the remaining time separation being achieved by retarding the bus.

If it is assumed that the total guidance error may be considered as an equivalent uncertainty in the impact parameter of the capsule trajectory, then Figure 4-45 gives the shallowest nominal entry angle which may be selected as a function of the trajectory error, ΔB. In other words, if ΔB = 500 km, the nominal entry angle would have to be at least 29 deg in order to guarantee atmospheric capture in the presence of a 500 km outer dispersion of the entry trajectory. For a very pessimistic guidance accuracy of 2000 km, the nominal entry angle would have to be at least [illegible] deg. Even for a ΔB of 5000 km, it would still be possible to guarantee impacting Mars by choosing an entry angle above [illegible] deg (essentially a

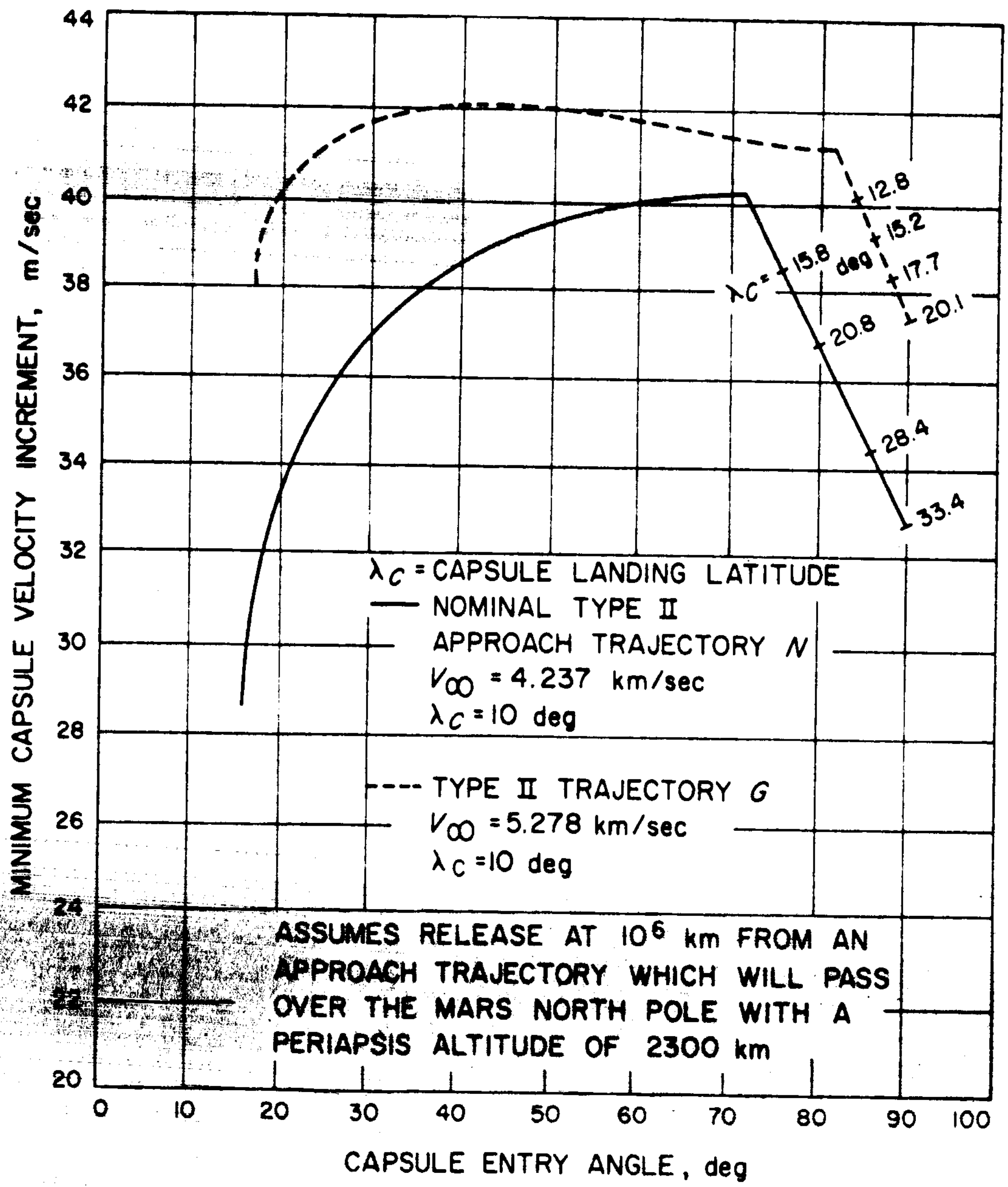
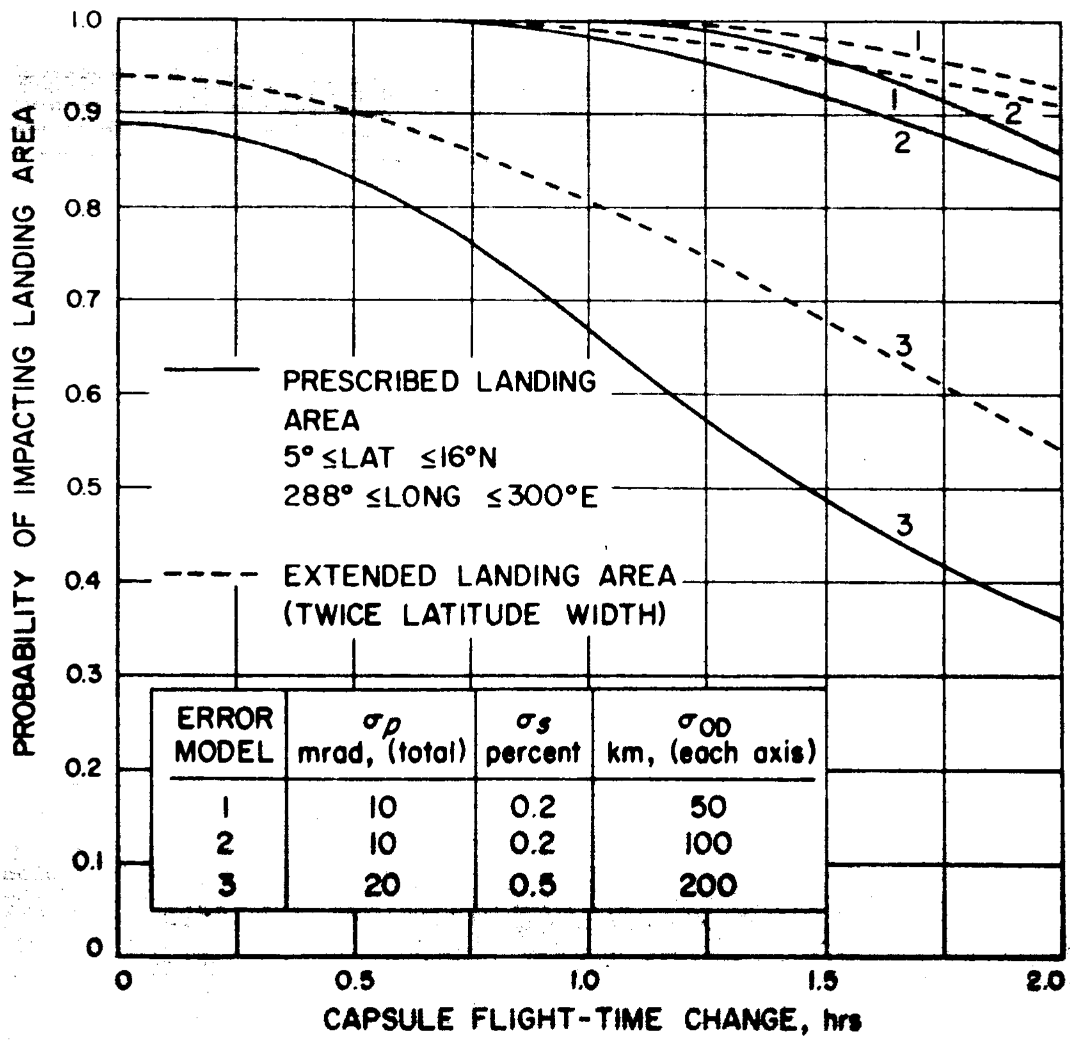


Figure 4-43. Minimum Capsule Velocity Increment to Land as Near λ_C = 10 Deg as Possible



ERROR MODEL	σ_p mrad, (total)	σ_s percent	σ_{OD} km, (each axis)
1	10	0.2	50
2	10	0.2	100
3	20	0.5	200

Figure 4-44. Probability of Impacting Landing Area within Syrtis Major vs Flight-Time Change of Capsule, for Three Different Error Models

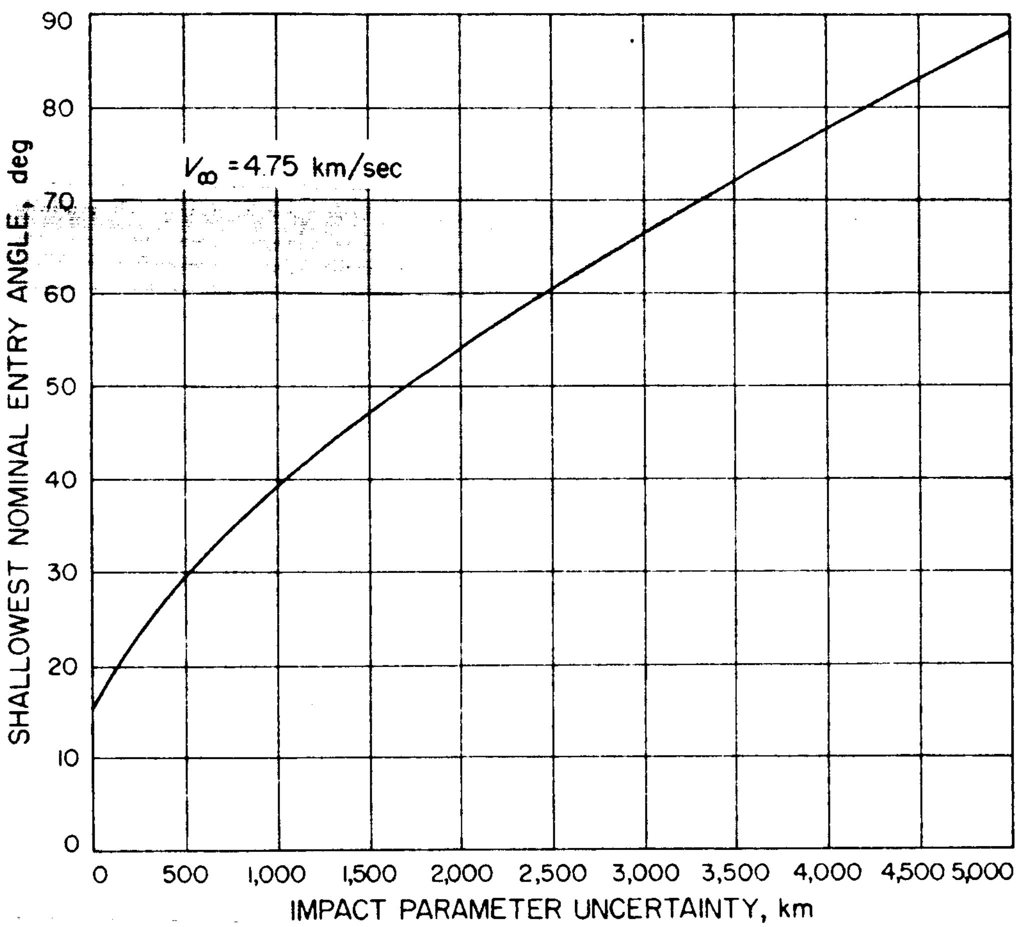


Figure 4-45. Shallowest Nominal Entry Angle Which Guarantees Capture in the Presence of an Uncertainty in the Impact Parameter of the Capsule Trajectory

direct vertical entry). Figure 4-46 illustrates the maximum surface dispersions which would result from guidance errors of 500 and 2000 km, as a function of the nominal entry angle. It is of interest to note that there exists an entry angle for which the surface error is the same as the impact-parameter error. As a matter of fact, for vertical entry, the surface dispersion is quite a bit less than the guidance uncertainty, due to the gravitational focusing of the capsule trajectory onto a more direct (as opposed to sloping) target surface.

Another important consideration is the accuracy with which the capsule arrival time can be controlled. When the spacecraft has reached the capsule release point in the vicinity of 10^6 km from Mars, the arrival time will be known to about one minute (1σ). Capsule execution errors will further contribute to the arrival-time error. A 1 m/sec velocity error (worst-case, 1σ) in the direction of flight would result in an arrival time error of about one minute. For a combined 3σ flight-time error of 5 min, the equivalent longitude error would only be 75 km which is still considerably less than the 3σ lateral dispersions. Therefore, the control of arrival time should not be a significant problem.

4. Bus-Capsule Relationships During Entry

This section presents bus-capsule relationships from capsule separation (at 10^6 km from Mars) through entry, landing, and a short time after landing, for a typical 1969 Mars Type II trajectory. For the example studied, a 1.8 m/sec velocity increment was applied to the capsule at separation such that entry would occur about 2 hr prior to bus periapsis passage. A nominal entry angle of 68 deg (below the local horizontal) was chosen along with a landing point at 10 deg North latitude on the left side (as seen from the approaching capsule) of Mars. The bus trajectory had a periapsis altitude of 1800 km and an inclination of 70 deg to the Mars equator.

A value of 1.76 slugs/ft^2 was assumed for the capsule ballistic coefficient, M/C_DA, and the JPL model atmosphere B (see JPL TR 32-455) was used for this simulation. In this model, the surface temperature is 410°R and decreases linearly to 320°R at an altitude of 13.44 km. Above this altitude the temperature is assumed constant at 320°R. A surface density of 0.00014 gm/cm^3 was assumed. Above 13.44 km an exponential density profile was employed with a reference density of 0.00022 gm/cm^3 and a scale height of 12.7 km.

A 15-ft diameter supersonic parachute is deployed when the capsule has decelerated to Mach 3. After this chute has slowed the system to Mach 0.9 (about one second later), the main 77-ft diameter chute is reefed to a diameter of about 4 ft. The full chute is deployed approximately seven seconds after reefing. The parachutes are so designed that the opening shocks will not exceed the peak ballistic deceleration. As can be seen from Table 4-4, the first chute is deployed at an altitude of 12.5 km and the main chute is opened fully at about 11 km. Capsule descent time is approximately 38 min with a final touchdown velocity of 4.8

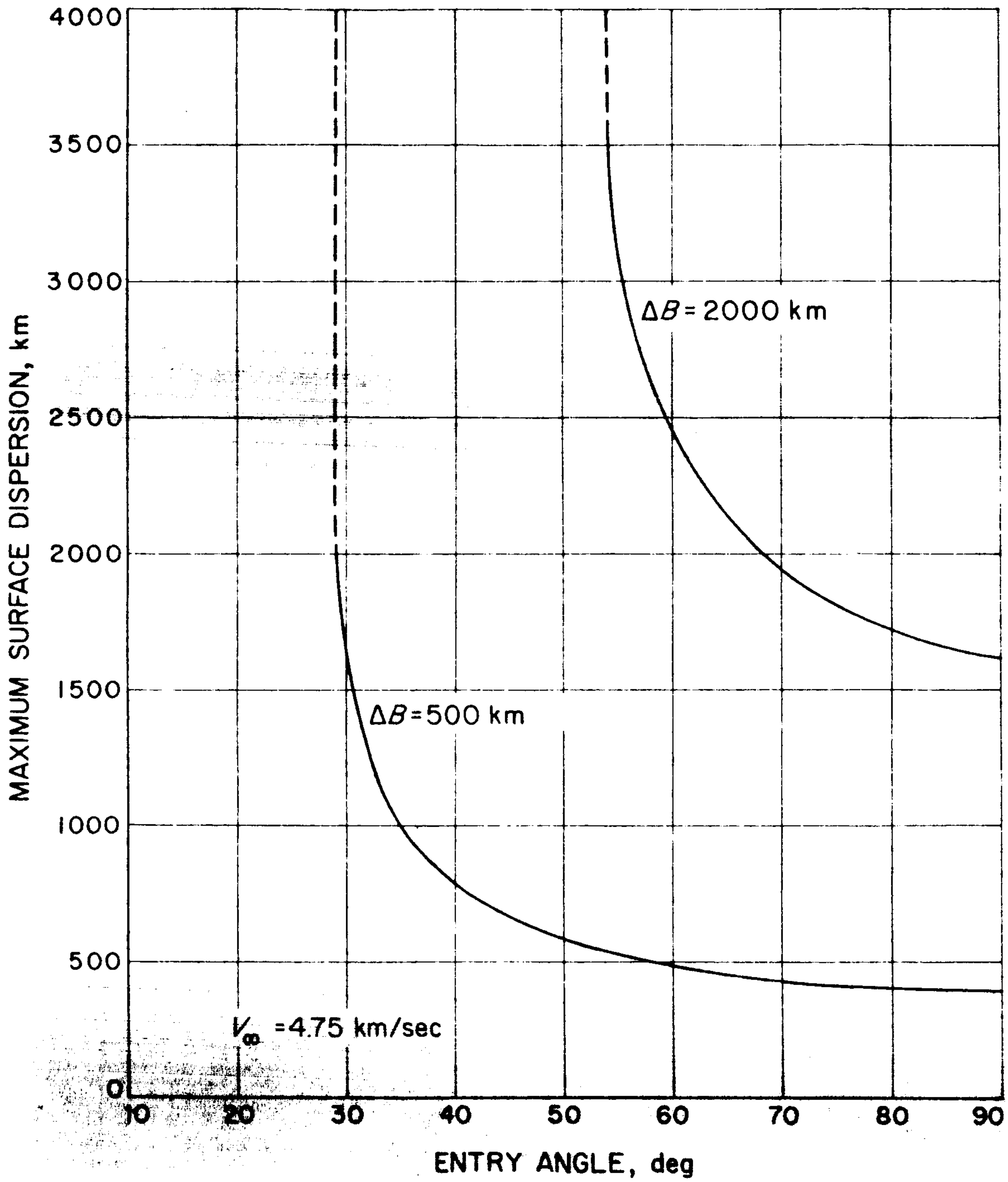


Figure 4-46. Maximum Surface Dispersion vs Entry Angle for Impact-Parameter Errors of 500 & 2000 KM

Table 4-4. Significant Events During Capsule Entry

Event	Time before impact (sec)	Altitude (km)
Entry altitude 200 km Entry angle 68 Entry speed 6.53 km sec Range to bus 29,690 km Elevation angle of bus - 48	2328 1	200.00
Maximum laminar stagnation point heating rate of 346 Btu ft^2 sec (corresponding radiation equilibrium temperature of 5180 R)	2301 6	42 95
Peak capsule ballistic deceleration of 60 Earth g's Bus capsule doppler acceleration of 552 m sec^2	2298 6	29 79
15 ft parachute deployment at Mach 3	2289 2	12 52
77 ft parachute reefed at Mach 0 9	2288 4	12 12
77 ft parachute fully opened seven seconds later	2281 4	10.97
Minimum bus capsule range rate of - 4 428 km sec	2274 6	10 91
Impact speed 4 5 meters sec Impact latitude - 10 N Range to bus 19,664 km Elevation angle of bus - 40 Total heat absorbed at laminar stagnation point - 3636 Btu ft^2	0	0
Start of bus attitude turns (probable end of initial communications period)	**Impact + 40 minutes**	**0**
Bus at 10 elevation above capsule horizon	**Impact + 45 minutes**	**0**

meters/sec (neglecting surface winds). It should be mentioned that, had a capsule ballistic coefficient of 1.2 slugs/ft^2 been used instead of 1.76 slugs/ft^2, the first chute would have opened at about 14 km, but this would have increased the descent time by only 4 min.

Figures 4-47 through 4-49 illustrate the time behavior of various capsule entry and capsule-to-bus parameters of interest. From Figure 4-47, it can be seen that the main portion of the ballistic deceleration phase lasts for only some 20 sec and occurs within an altitude band from about 80 km down to 15 km. Figure 4-50 illustrates the behavior of bus-capsule range and two off-antenna-axis look angles for a worst-case communications situation. This worst-case geometry occurs when the bus trajectory experiences a 500 km high dispersion and the capsule lands lower than intended at 5 deg North latitude. The axis of the bus antenna has been assumed normal to the roll axis of the bus, which derives its attitude reference from the directions to the Sun (primary reference) and Canopus (secondary reference). The discontinuity in the off-capsule-antenna-axis curve occurs when the spin-stabilized inertial orientation which the capsule was given at separation is changed to the local vertical after the parachutes have opened shortly after entry.

5. The Effect of Atmospheric Drag on Orbiter Lifetime

The problem of predicting the lifetimes of terrestrial orbiters has been comprehensively treated in the literature, and various solutions have been described.[1), 2), 3)] Most of these deal only with the influence of aerodynamic drag, although the effects of atmospheric oblateness, rotation, and radiation pressure have been individually treated.

In contrast, the problem of predicting the lifetimes of spacecraft orbiting other planets has not received much consideration. The reasons for this are many, however, it appears that the lack of accurate upper atmosphere models for Mars and Venus has been a major obstacle.

The planning of orbiter missions to Mars makes it necessary to estimate orbital lifetime in the atmospheres of planets other than the Earth. In the case of a Mars orbiter mission, the orbits should be designed to provide sufficient time for biological investigation of the Martian surface before possible contamination by an unsterilized orbiter spacecraft. It is presently thought that a lifetime of not less than fifty years is required.

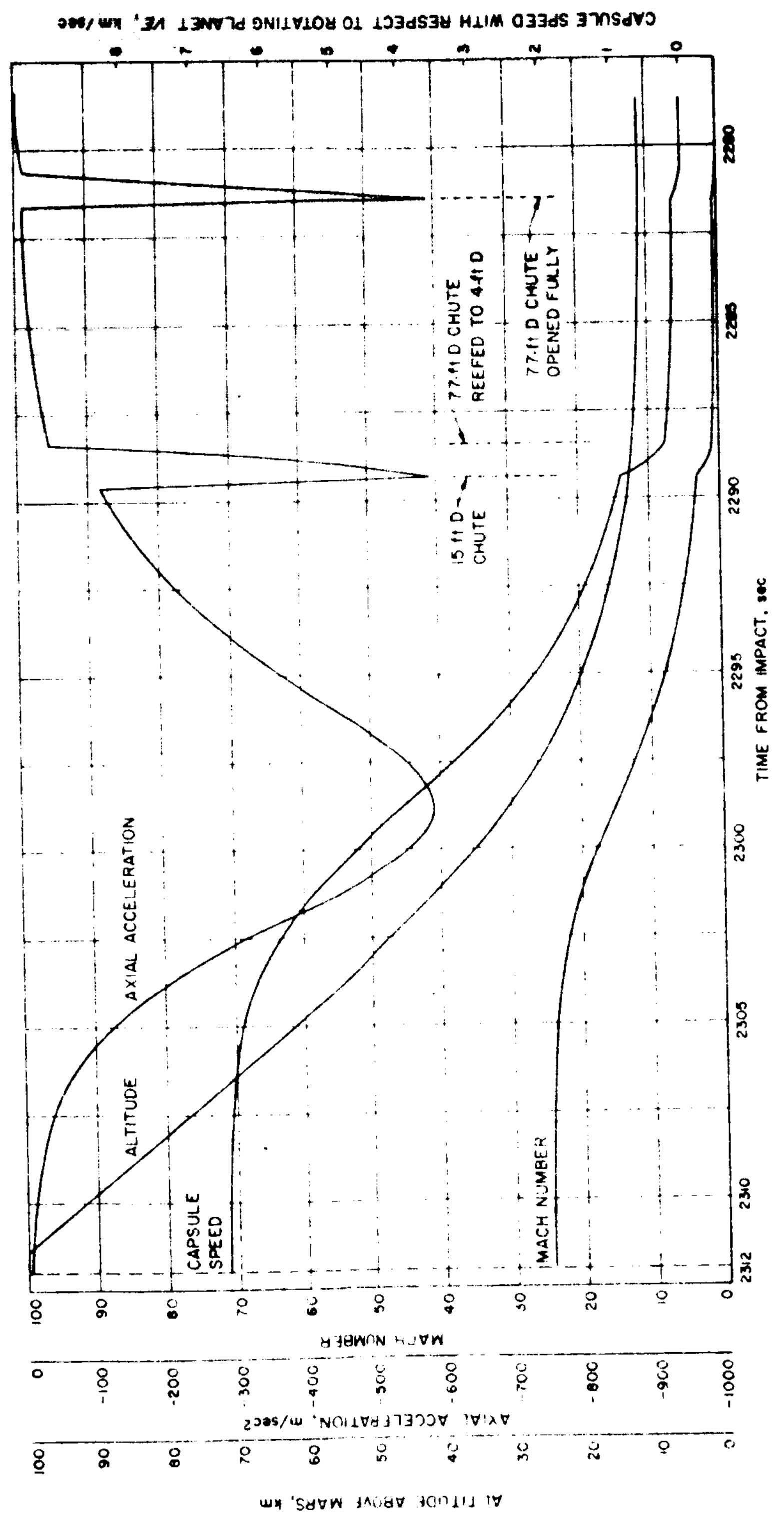


Figure 4-47. Capsule Mach Number, Axial Acceleration, Altitude and Speed vs Time from Impact for Altitude Range 0-100 km

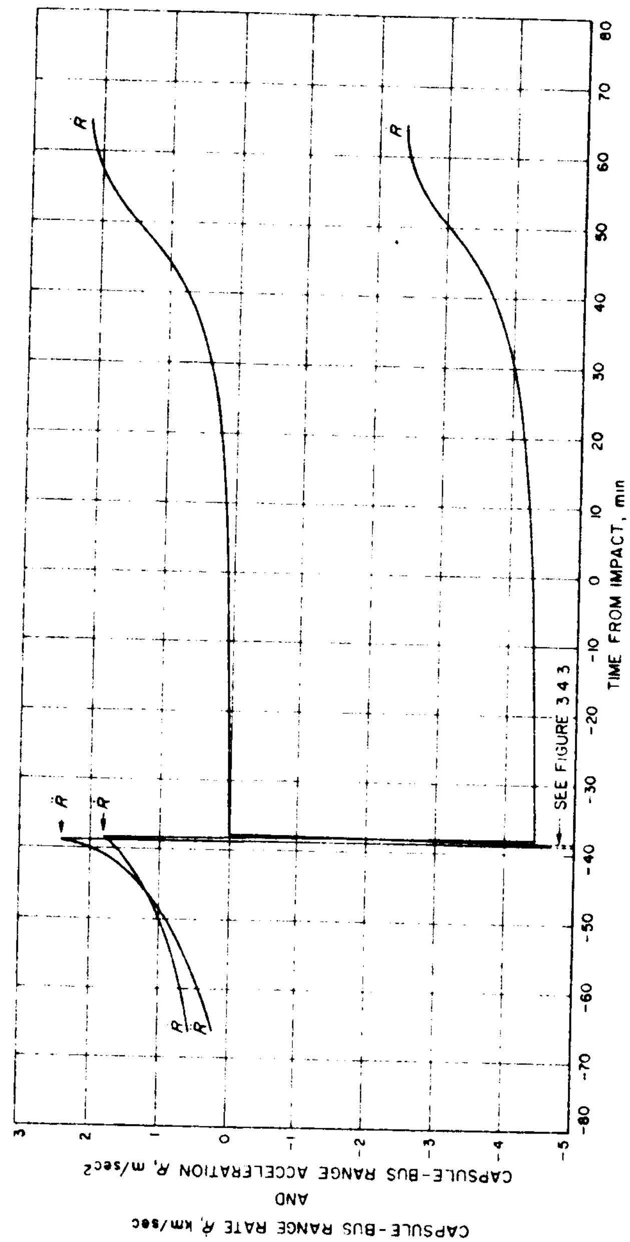


Figure 4-48. Capsule-Bus Range Rate and Range Acceleration vs Time from Impact

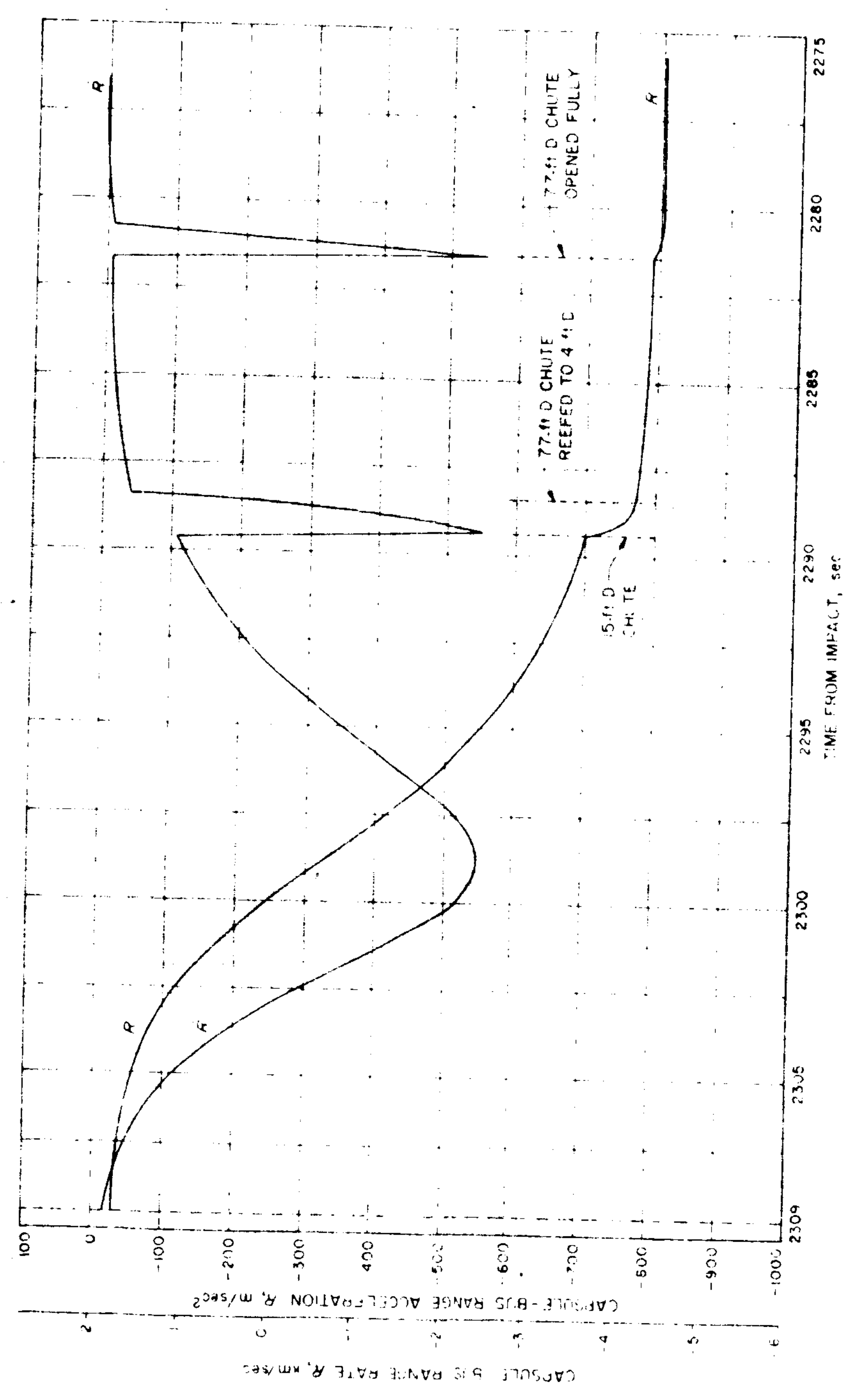


Figure 4-49. Capsule-Bus Range Rate and Range Acceleration vs Time from Impact

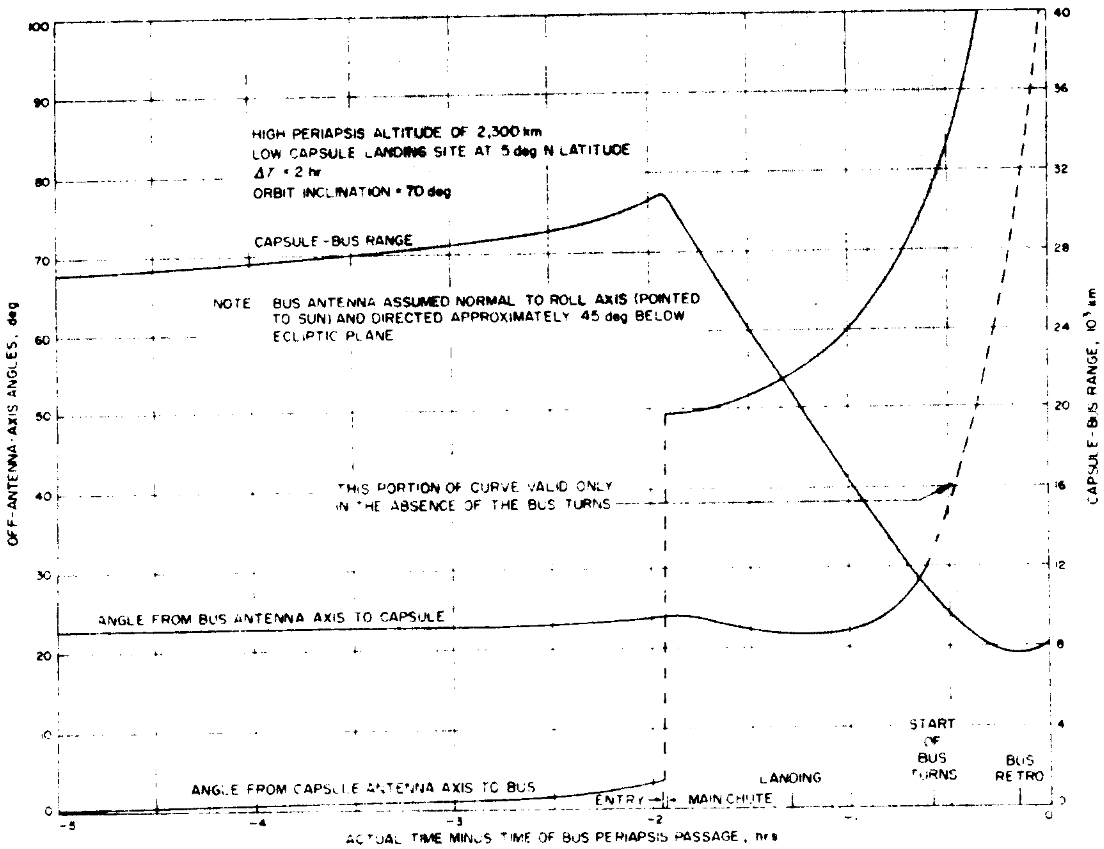


Figure 4-50. Communications Parameters for Worst-Case 1969 Mars Type II Approach

Thus, the orbital altitudes must be as high as necessary to obtain the required lifetime, and as low as possible to obtain maximum information from planet science experiments. For this reason, a study of satellite lifetimes in planetary atmospheres was undertaken, and the results are presented in this section.[4)] For the sake of simplicity, only the effect of drag is considered as it is felt to represent the dominant effect. When considering lifetimes of the order of fifty years, however, it is possible that perturbations due to higher order gravitational harmonics, radiation pressure and density changes due to solar activity may be as important as the effect of atmospheric drag, and these effects justify further investigation.

It is obvious that any results describing the effects of atmospheric drag on the lifetime of an orbit are highly dependent upon the assumed atmospheric properties. In the case of Mars, the available atmospheric data is very limited. For altitudes from 0-80 km, which are important to investigators of atmospheric entry problems, the physical properties of the atmosphere of Mars have been estimated with some degree of precision. (For example, the density at 80 km of one atmosphere model[5)] is known to within two orders of magnitude.)

However, in order to investigate the decay of 50-year orbits, the density of the atmosphere at altitudes up to 2000 km must be known.

In this range, there exist only two formulations of an atmosphere model.[6), 7)] Chamberlain[6)] presents a model based on the photo-chemistry and thermodynamics of an atmosphere composed primarily of nitrogen. This model is held in esteem by investigators in the field of aeronomy; however, the density model includes only one point above 320 km (at 1500 km). Thus, accuracy of the upper position of the atmosphere model depends upon the validity of this one point.

Vachon[7)] presents a model based on the existence of a troposphere, stratosphere and thermosphere in the atmosphere of Mars, based on knowledge of the Earth's atmosphere. These models, together with Chamberlain's, are shown in Figure 4-51. Models B-1 and C-1 are based on troposphere heights of 13 km and 24 km respectively, with a 1°K/km thermosphere, while models B-2 and C-2 represent models with 3° K/km thermospheres. It can be seen that at 1800 km the density difference between the extreme models is some five orders of magnitude. However, Vachon prefers model B-1, which is within an order of magnitude of Chamberlain's model.

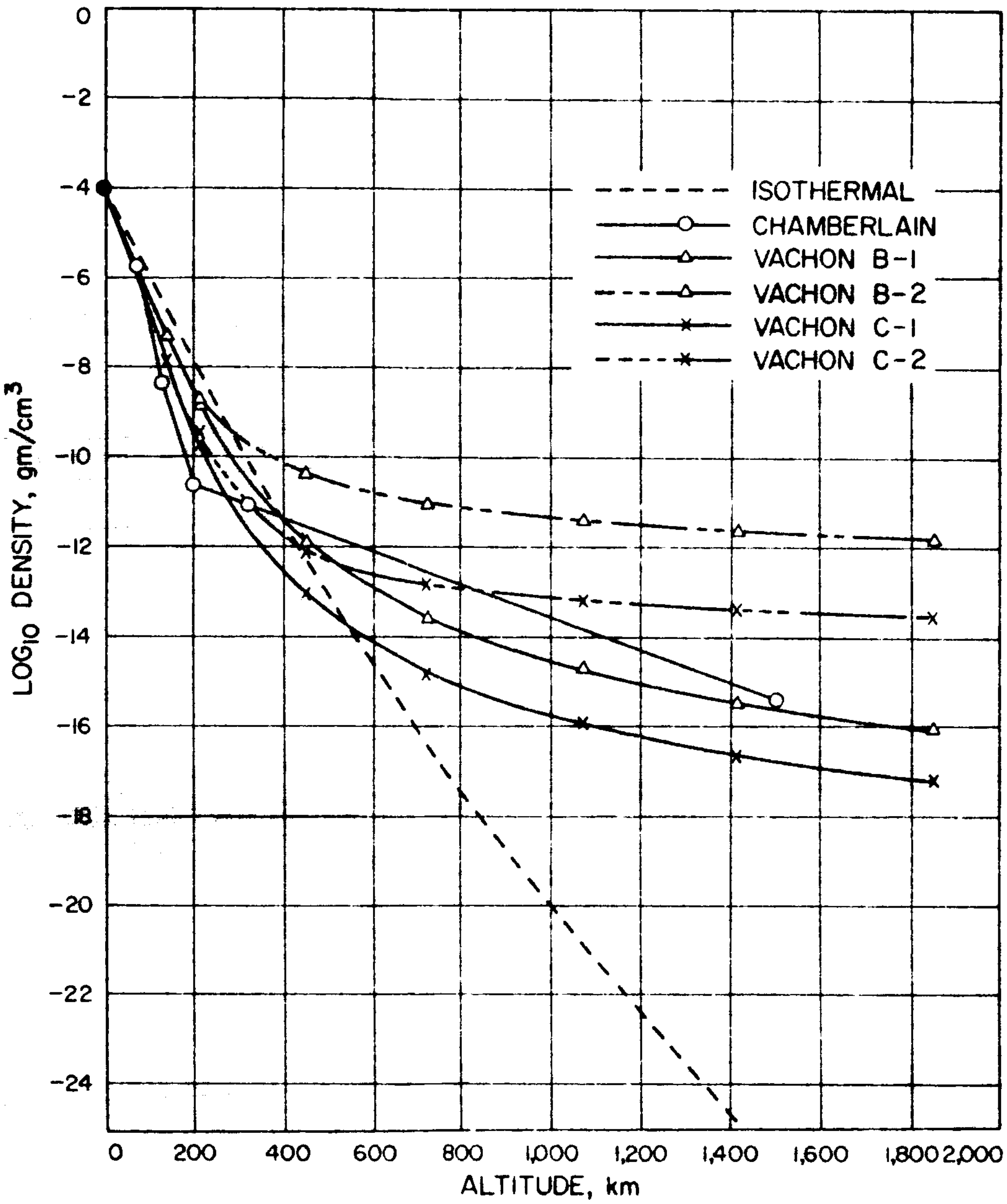


Figure 4-51. Models of the Upper Atmosphere of Mars

Three [illegible] atmospheres were selected for this study. The first is based on Chamberlain's model, assuming an exponential characteristic above 120 km. The second is obtained by assuming [illegible] to represent a density higher by an order of magnitude than in Chamberlain's [illegible] model, and the third is an atmosphere that is ten times denser than Chamberlain's model [illegible] altitudes. These atmospheres are referred to in this discussion [illegible] I, II and III, [illegible].

For circular orbits [illegible] was computed by numerically integrating the [illegible] by equating the orbital energy expression to the energy [illegible].[1]

These lifetime vs. perigee altitude relationships are plotted as part of Figures 4-52 through 4-54 for atmospheres I, II and III. The ballistic coefficient M/C_DA, was assumed to be equal to 1 slug/ft^2.

In the case of elliptic orbits, the situation is considerably more complicated. In a spherical atmosphere surrounding a homogeneous spherical planet, the only orbital parameters of interest in the orbital decay problem are a, the semimajor axis, and e, the eccentricity. Atmospheric drag also causes rotation of the line of apsides, but this is important [illegible] the atmosphere is unusually spherically asymmetric.

[illegible] for elliptical orbits were obtained from an approximate solution of the radial and transverse differential equations of motion perturbed by drag deceleration.[3] An exponential [illegible] was [illegible]. The resulting integral expression was evaluated numerically for the three atmosphere models, using a range of initial eccentricities from 0.[illegible] to 0.0[illegible]. The result [illegible] the time required for the orbit to decay from the initial eccentricity to an eccentricity of zero. The lifetime of the resulting circular orbit is then computed and added to the previous result. The total represents the lifetime of the initial orbit, and this quantity is plotted vs perigee altitude in Figures 4-52 through 4-54.

While the curves of Figures 4-52 through 4-54 essentially supply all the necessary information, they are obtained for only one value of M/C_DA, the ballistic coefficient (1 slug/ft^2). However, since the ballistic coefficient appears as a multiplicative constant in the lifetime expression, these results can be easily extended to any arbitrary ballistic coefficient. Thus, in Figures 4-55 through 4-57 the initial apoapsis altitude necessary to provide a lifetime of fifty years is plotted vs initial periapsis altitude for eight values of the ballistic coefficient ranging from 0.125 to 4.0 slugs/ft^2.

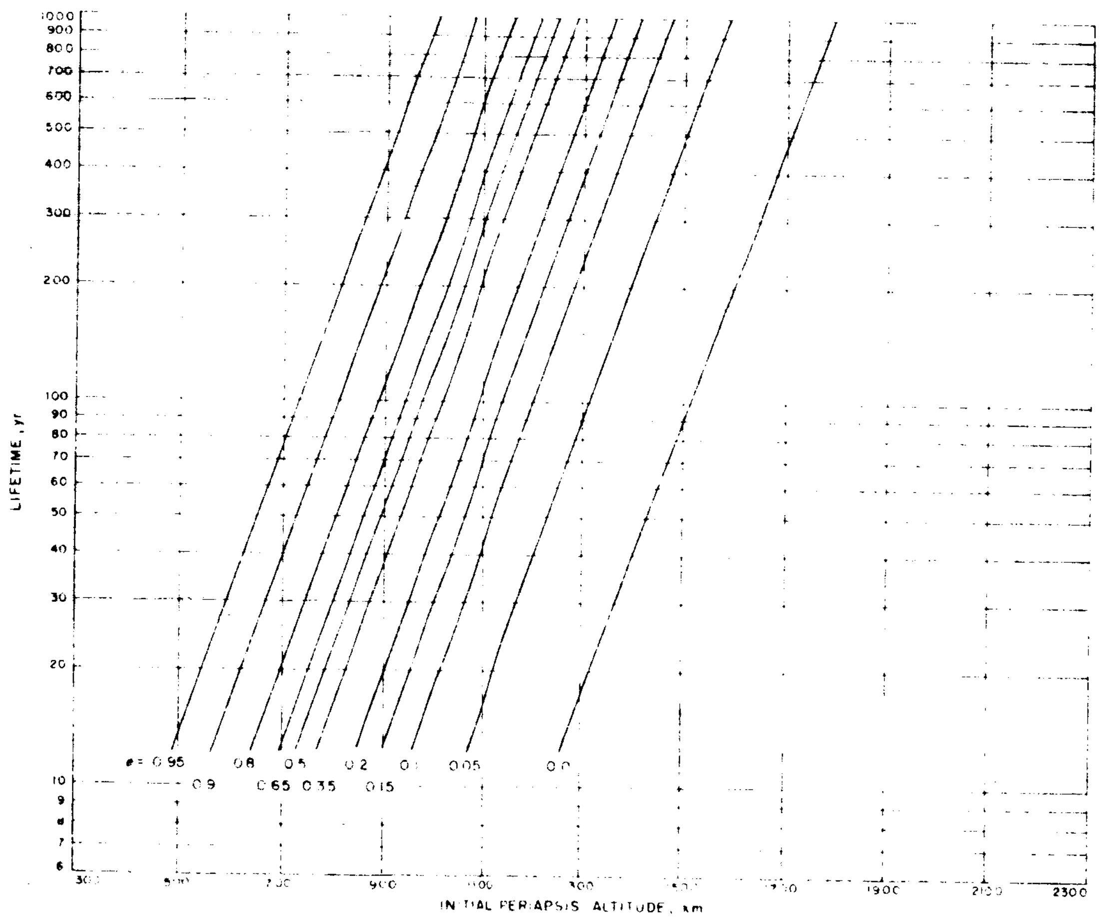


Figure 4-82. Lifetime vs Periapsis Altitude, Atmosphere I ($M/C_DA = 1$ slug/ft^2)

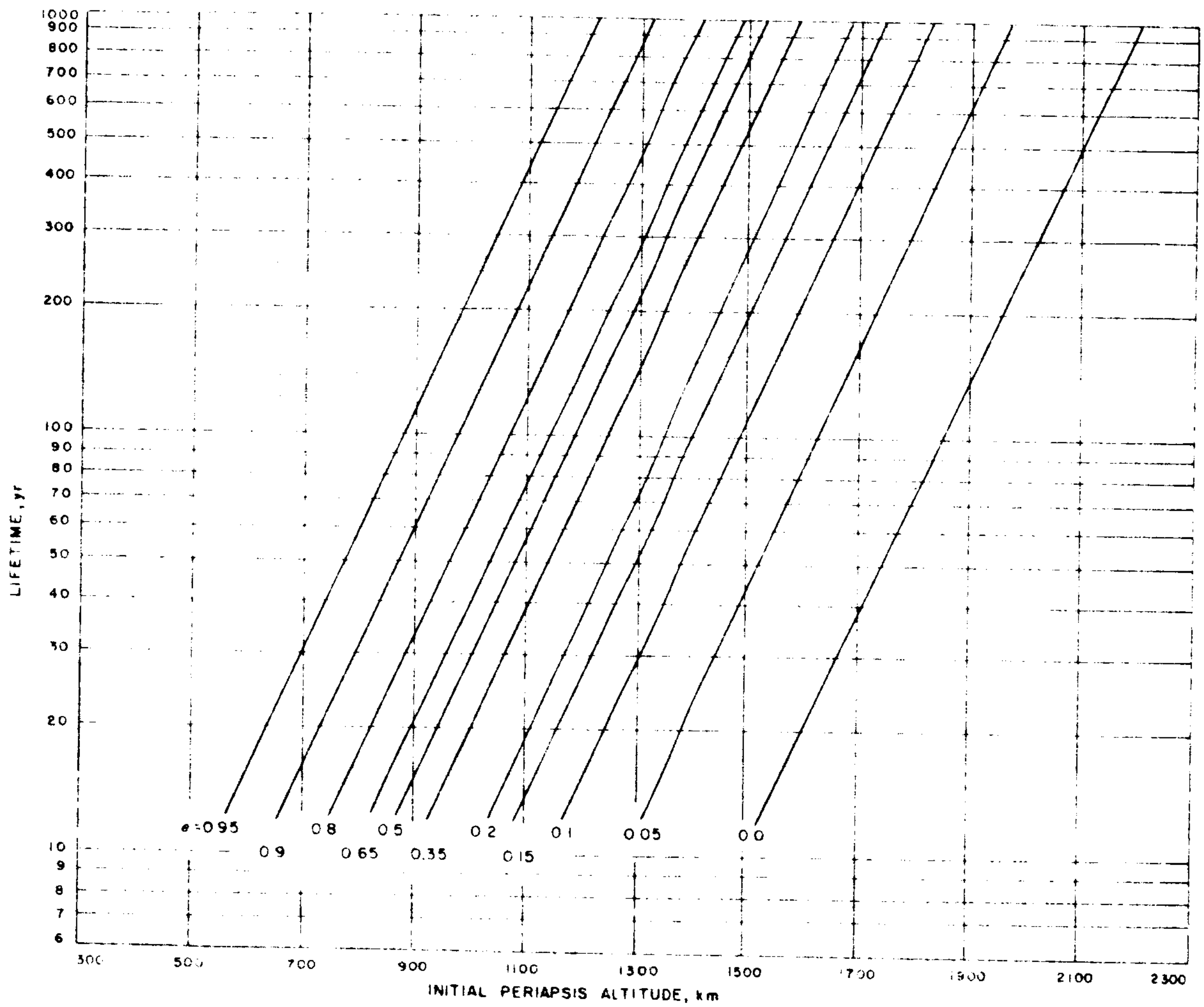


Figure 4-53. Lifetime vs Periapsis Altitude, Atmosphere II (M/C_DA = 1 slug/ft^2)

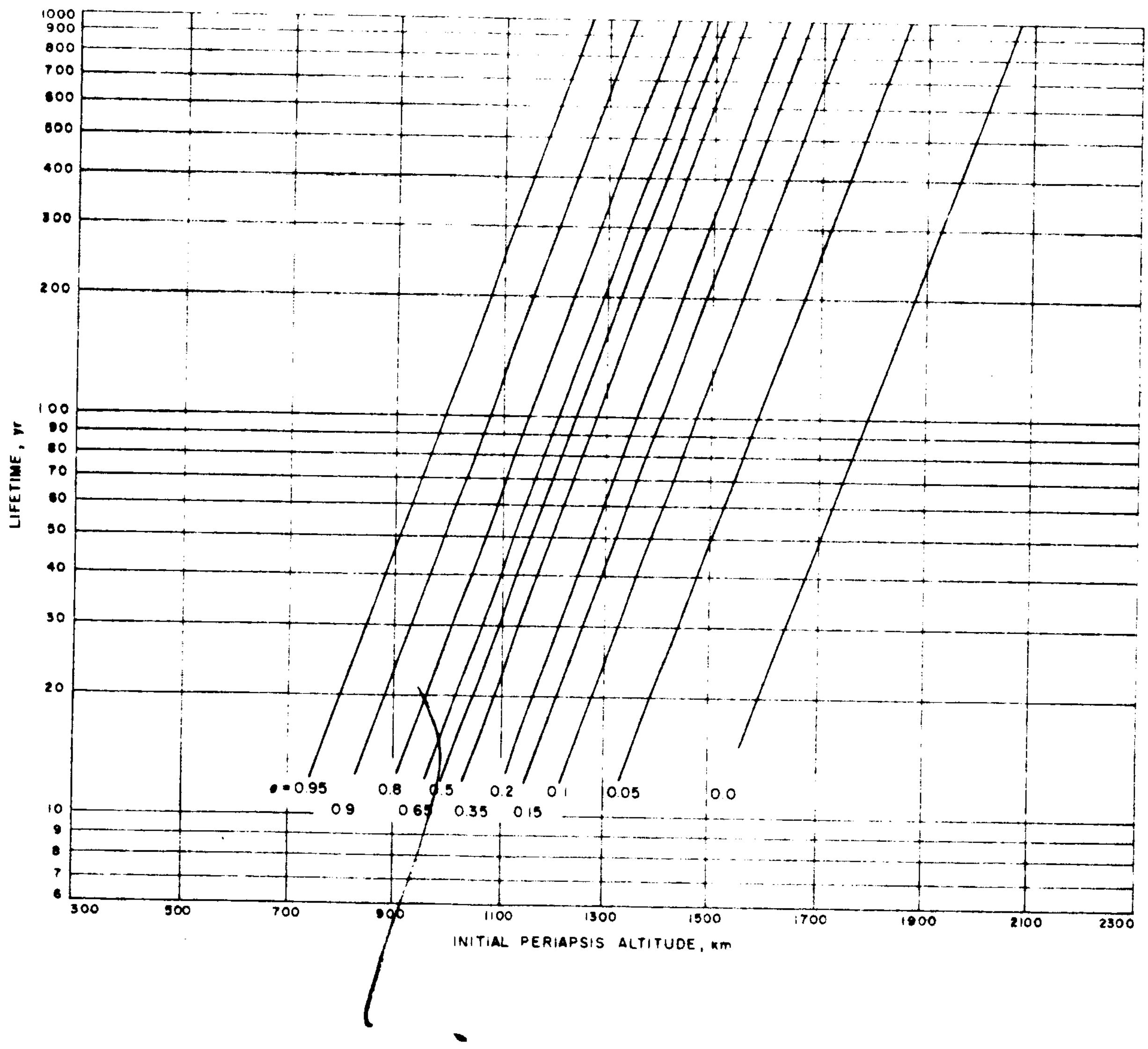


Figure 4-54. Lifetime vs Periapsis Altitude, Atmosphere III (M/C_DA = 1 slug/ft^2)

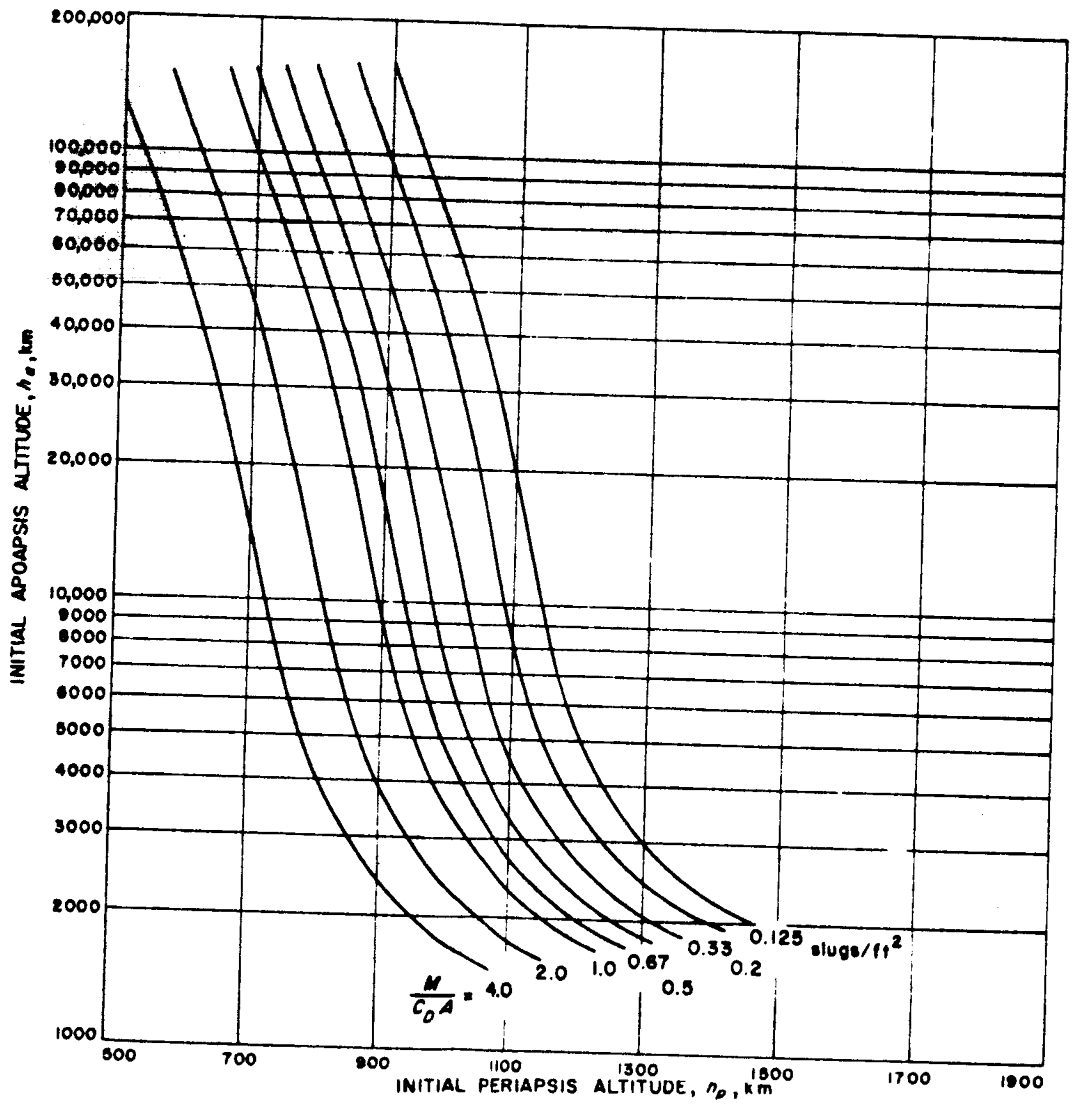


Figure 4-55. h_a vs h_p for 50 Yr. Lifetime (Atmosphere I)

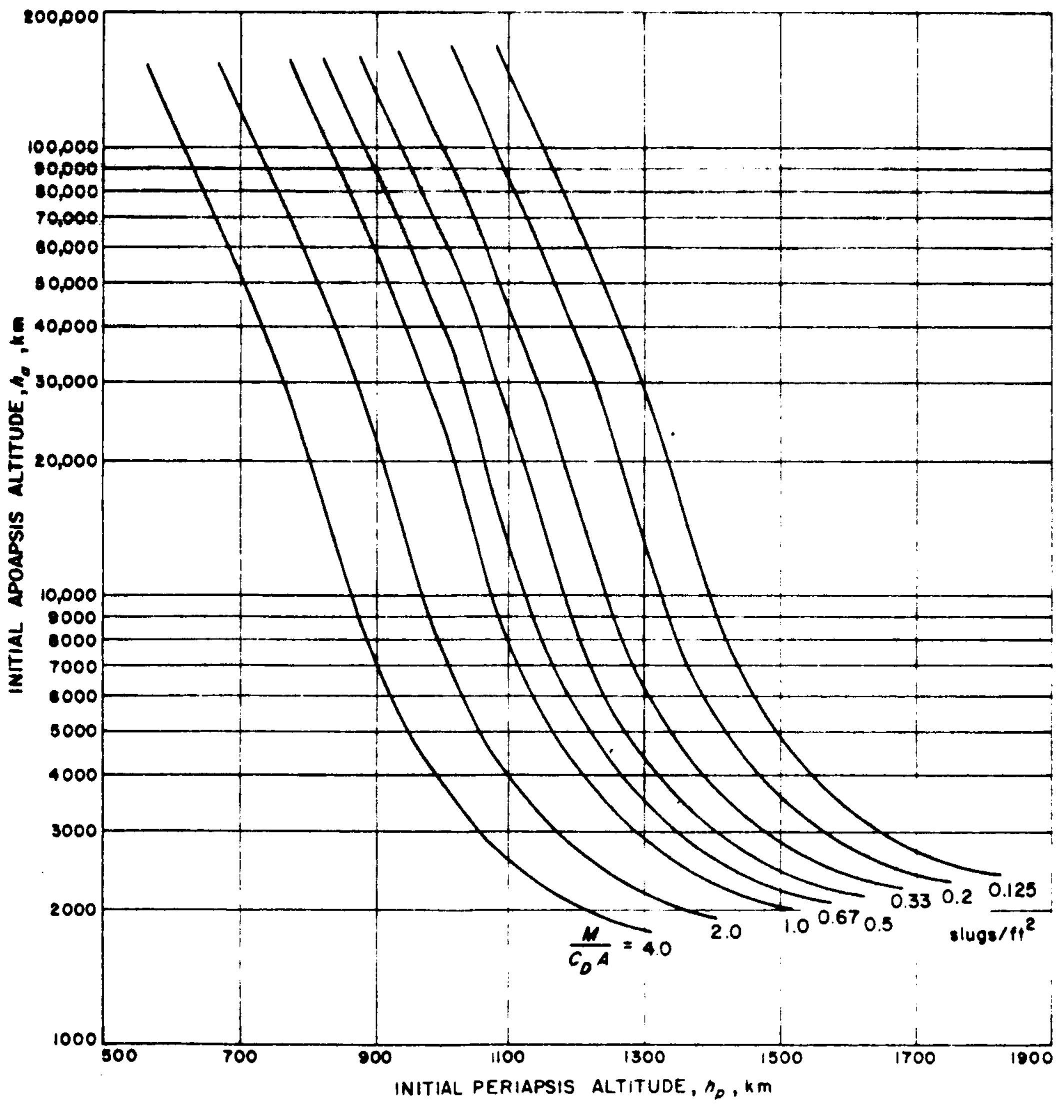


Figure 4-56. h_a vs h_p for 50 Yr. Lifetime (Atmosphere II)

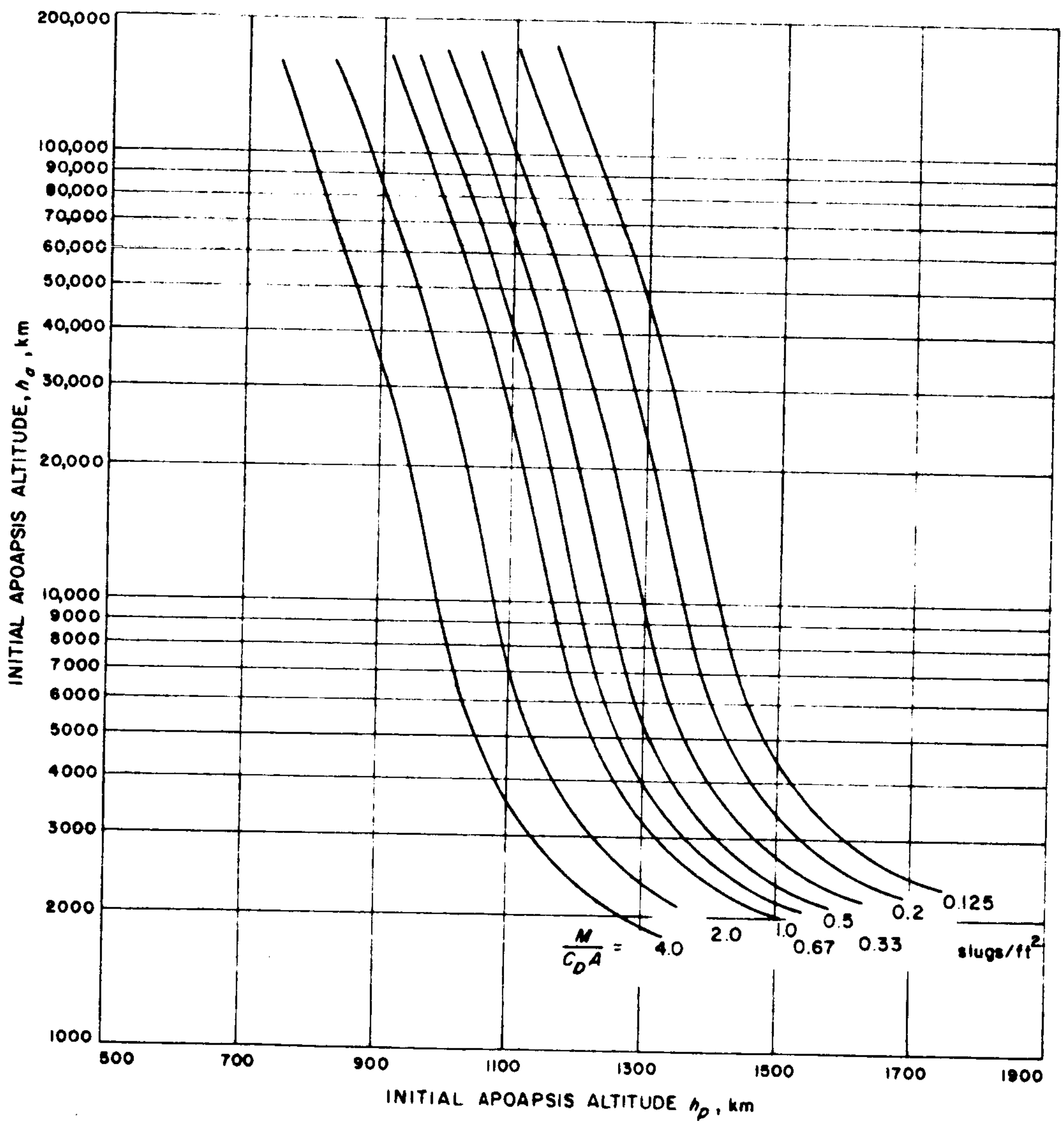


Figure 4-57. h_a vs h_p for 50 Yr. Lifetime (Atmosphere III)

The use of these curves is illustrated by the following example: Suppose that the spacecraft is assumed to have a ballistic coefficient of 0.5 slugs/ft^2. Then, using the worst-case atmosphere III, corresponding to Figure 4-57, a periapsis altitude of approximately 1,250 km is permissible for an apoapsis altitude of 10,000 km. However, if the ballistic coefficient were as high as 4.0, then an apoapsis altitude of 10,000 km could lead to an allowable periapsis altitude of slightly less than 1000 km.

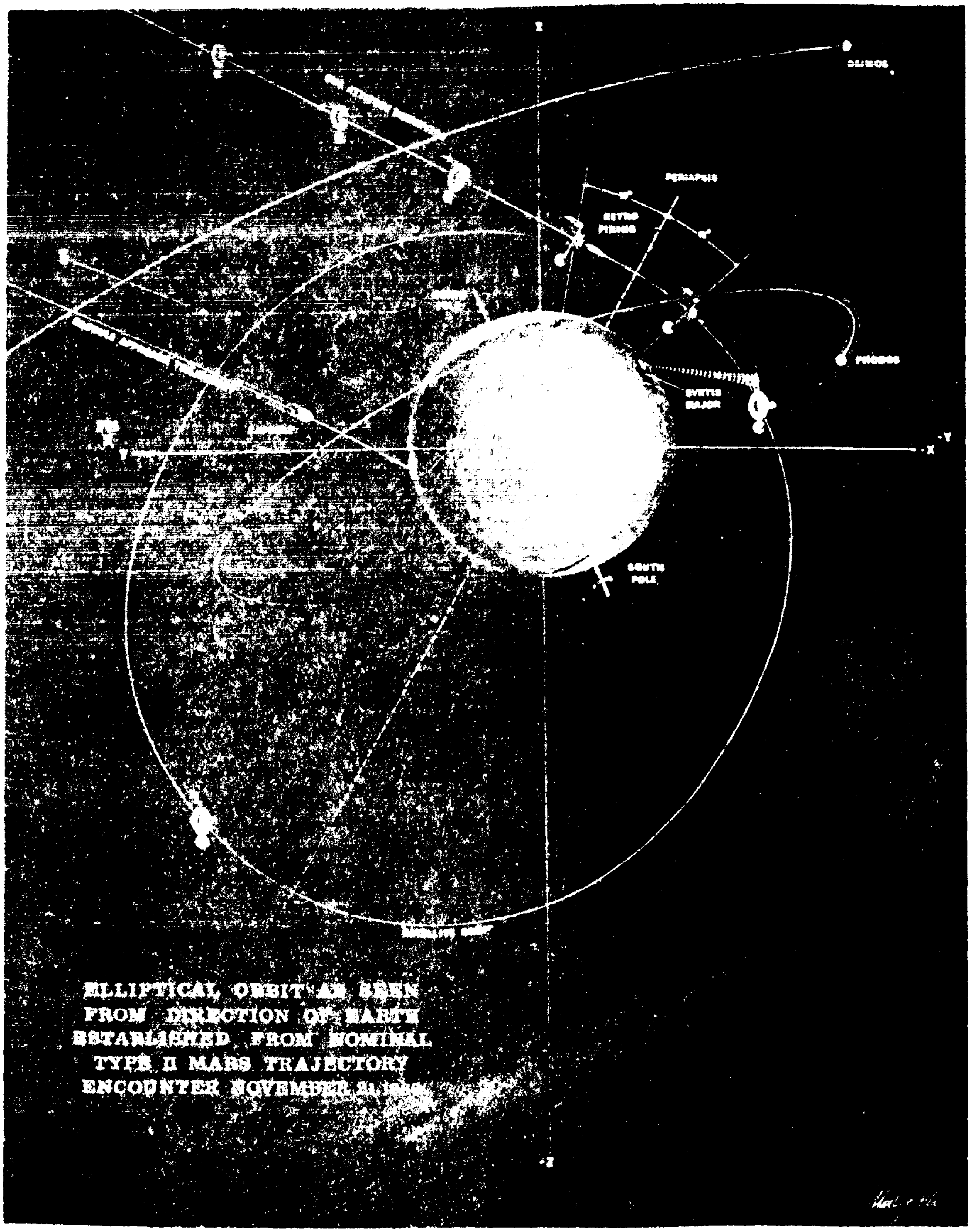
6. Attractive Orbiter/Capsule Missions

Based upon the assumption that Type II 1969 Mars trajectories will be employed, this paragraph describes two Martian satellite orbits which appear to satisfy most of the constraints previously outlined. The first is an elliptic orbit with a periapsis altitude of 1800 km, an apoapsis altitude of 10,070 km (closest "even-mapping" altitude to 10,000 km; see paragraph 3.1, constraint 6), and an inclination to the Mars equator of 70 deg. The second is a circular polar orbit with a nominal altitude of 2300 km. Figures 4-58 through 4-60 illustrate the orientation of these two orbits as well as the directions to the Sun, Earth, Canopus, and Vega for a typical Type II encounter in November of 1969.

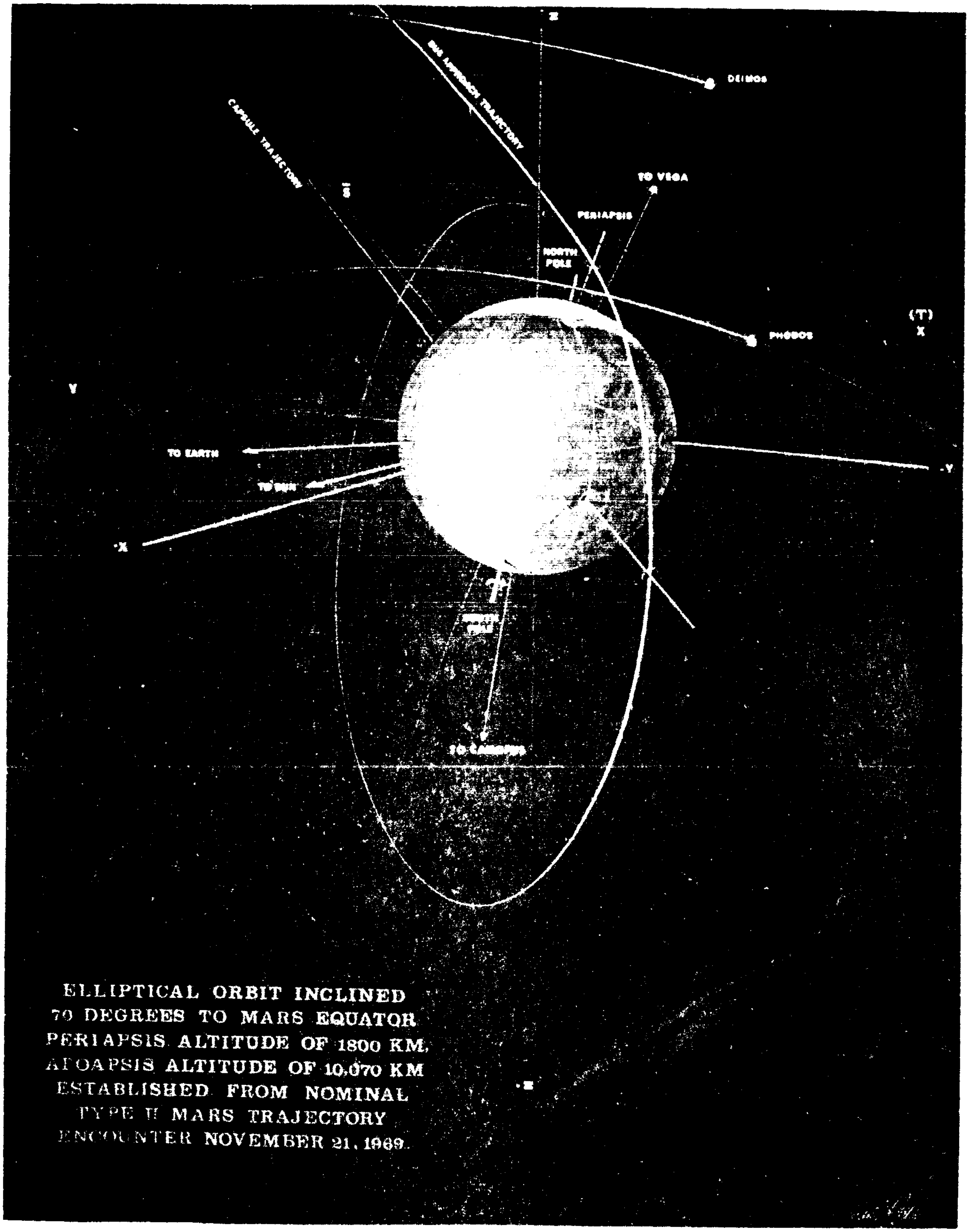
Figure 4-58 shows the elliptic orbit as seen from the Earth and displays the true size and orientation of the ellipse. Also shown are the proper bus locations at the beginning, during, and at completion of the attitude turns which precede the retro maneuver. The start of the retro burn is indicated by the exhaust plume, while thrust termination occurs some 12 deg past periapsis (this number varies with the approach speed and thrust level). The entry capsule has been shown for illustrative purposes but is not phased properly in time with respect to the bus. More specifically, capsule entry occurs when the bus is still some 30,000 to 35,000 km from Mars. The small Martian Moons may be seen quite clearly in Figure 4-58. The inner Moon, Phobos, has a diameter of about 15 km and is travelling in a nearly-circular (e = 0.021) equatorial orbit at a distance of 9350 km from the center of Mars. The outer Moon, Deimos, has a smaller diameter of about 8 km and is also travelling in a nearly-circular (e = 0.003) equatorial orbit but at a distance of 23,500 km from the center of Mars. The Martian light-dark areas, polar caps, seasonal changes, daily rotation at nearly Earth's rate, and the two Moons make Mars a very appealing planet to explore.

Figure 4-59 shows the elliptic orbit viewed from a different angle. By passing over the northern hemisphere, periapsis occurs on the lighted side of Mars at about 55 deg North latitude. Although oblateness effects will cause the subperiapsis point to move away from the equator, it is estimated that, even in the presence of a 3 σ guidance error in establishing the orbit, the northerly motion of the subperiapsis point will not exceed a maximum of 0.2 deg/day (see Figures 4-29 and 4-31). A southerly pass would place periapsis (for a natural transfer) above the dark surface at 85 deg South latitude (see also Figure 4-35).

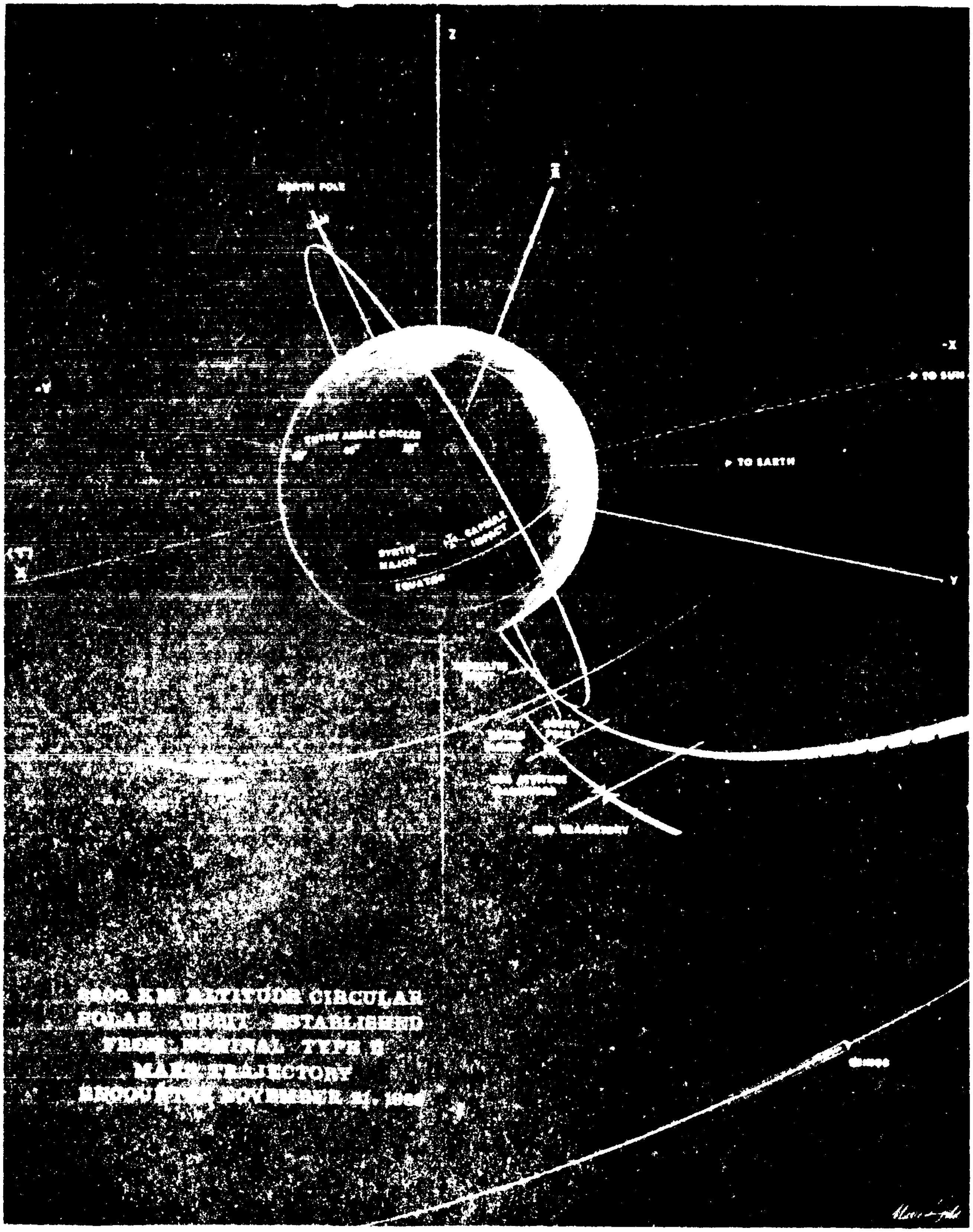
Figure 4-60 shows a circular orbit which has been established from a southerly-pass approach trajectory, since the peripasis location is no longer important and a better preorbit capsule-to-bus communication geometry can be achieved for Type II southerly bus approaches. Also shown in Figure 4-60 are some of the capsule entry-angle circles for the approach-phase capsule release. The approximate size and location of the desired landing area Syrtis Major is also indicated in this figure.



ELLIPTICAL ORBIT AS SEEN FROM DIRECTION OF EARTH ESTABLISHED FROM NOMINAL TYPE II MARS TRAJECTORY ENCOUNTER NOVEMBER 21, [illegible]



ELLIPTICAL ORBIT INCLINED
70 DEGREES TO MARS EQUATOR
PERIAPSIS ALTITUDE OF 1800 KM,
APOAPSIS ALTITUDE OF 10,070 KM
ESTABLISHED FROM NOMINAL
TYPE II MARS TRAJECTORY
ENCOUNTER NOVEMBER 21, 1969.



Z
NORTH POLE
-X
TO SUN
TO EARTH
CAPSULE IMPACT
EQUATOR
Y
X
CIRCULAR
ESTABLISHED
TYPE

As can be seen from Figures 4-58 through 4-60, the Type II approach geometry permits an uninterrupted view of both the Sun and the Earth for a considerable period of time after encounter. As a matter of fact, the orbiter is continuously in the sunlight for about 45 days, while planetary occultation of the Earth does not occur until 90 days after encounter. Table 4-5 gives the occultation times of various bodies during a single orbit at 20 day intervals after encounter. The circular orbit has a period of 3.64 hr (218 min), while the elliptic orbit period is 7.61 hr (457 min). The retro velocity increment necessary to establish the circular orbit is 3.002 km/sec (based upon V_∞ = 4.237 km/sec), the approach aiming point is at $\vec{B}\cdot\vec{R}$ = 7015 km and $\vec{B}\cdot\vec{T}$ = 3294 km (see Figure 4-44 for presentation of $\vec{B}$, $\vec{R}$, and $\vec{T}$), circular velocity is 2.74 km/sec, angular velocity is 0.48 mrad/sec, and the planetary angular diameter as seen from the orbiter is 75.4 deg. The elliptical orbit requires a retro velocity of 2.42 km/sec, has aiming coordinates of $\vec{B}\cdot\vec{R}$ = -7221 km and $\vec{B}\cdot\vec{T}$ = -438 km, orbital velocity varies from 1.55 to 3.45 km/sec, angular velocity ranges from about 0.10 to 0.66 mrad/sec, and the planetary angular diameter varies from 29 to 82 deg.

Figures 4-61 through 4-63 illustrate the behavior of the lighting angle (Sun-orbiter-Planet angle) with orbiter latitude during a single orbit at 20 day intervals after encounter. During the first month after encounter, lighting conditions are best for photographing the southern hemisphere of Mars. Fortunately, the area near 30 deg South latitude is currently considered to contain important areas for photographic examination.

Figure 4-64 shows the interesting cyclical behavior of the elevation angle of the orbiter (as seen from the capsule) vs time after encounter. The orbiter spends an average time of 200 min/day above the capsule horizon and about 55 min/day above 30 deg elevation. Figure 4-65 shows the capsule-to-orbiter range patterns with time. For the circular orbit, the capsule-to-orbiter range is nominally 4580 km at zero elevation and 3200 km at 30 deg elevation. It should be noted that Figure 4-65 illustrates continuously the distance between the orbiter and capsule for negative as well as positive orbiter elevation angles. Figure 4-66 illustrates the elevation-angle history of the elliptic orbiter which spends an average time of 390 min/day above the capsule horizon and 150 min/day above 30 deg elevation. Figure 4-67 shows the surface coverage (on the lighted side of Mars) which would be completed by the elliptic orbiter during the first 30 days after encounter.

It is important to emphasize that the orbital altitudes discussed throughout this section and depicted in Figures 4-58 through 4-60 are selected on the basis of a Mars atmosphere model ten times more dense than Chamberlain's data and a 3σ approach-guidance error of 500 km. However, if (1) the Mars upper atmosphere is much thinner than assumed and/or (2) the control of periapsis altitude is improved through the use of orbit trim, then it should be possible to select periapsis altitudes which are much closer to the Martian surface.

Table 4-5. Occultation Times During a Single Orbit

Days after encounter	Sun	Earth	Canopus	Vega
0	0[a], 0[b]	0, 0	41, 35	44, 0
20	0, 0	0, 0	41, 35	44, 0
40	0, 0	0, 0	41, 3[illegible]	44, 0
60	30, 55	0, 0	42, 3[illegible]	44, 0
80	40, 55	0, 0	42, 32	44, 0
100	44, 20	22, 55	42, 32	44, 0
120	45, 0	38, 50	43, 31	44, 0
140	45, 0	44, 0	43, 30	44, 0

(a) time in minutes for circular orbit
(b) time in minutes for elliptical orbit

Figures 4-61 through 4-67 were generated by the Planetary Orbiter/Split-Capsule Trajectory Program, which has been completely described in JPL TM 312-171. Many of the results related to the Mars orbiter missions may be found in JPL Technical Memoranda 312-2[illegible], 312-291, and 312-302.

c. Low Inclination Orbits

Since the desired capsule landing point lies in an area between 5 deg and 10 deg North latitude, orbits which have a low inclination to the Martian equator will give longer view periods and, [illegible] occultation times than high inclination orbits. Low inclination orbits also provide that neither Canopus nor Vega are occulted during the desired lifetime of the orbiter (150 days [illegible]). [illegible] low inclinations [illegible] orbits have the undesirable feature that a smaller surface area will be in view of the orbiter, and the Earth and Sun are always occulted [illegible] during each orbit. Since a tradeoff will have to be made between the [illegible] and other effects to ultimately determine the value of a particular inclination, quantitative studies of the above parameters must be made for orbits of various inclinations. As a starting point, studies originally were [illegible]

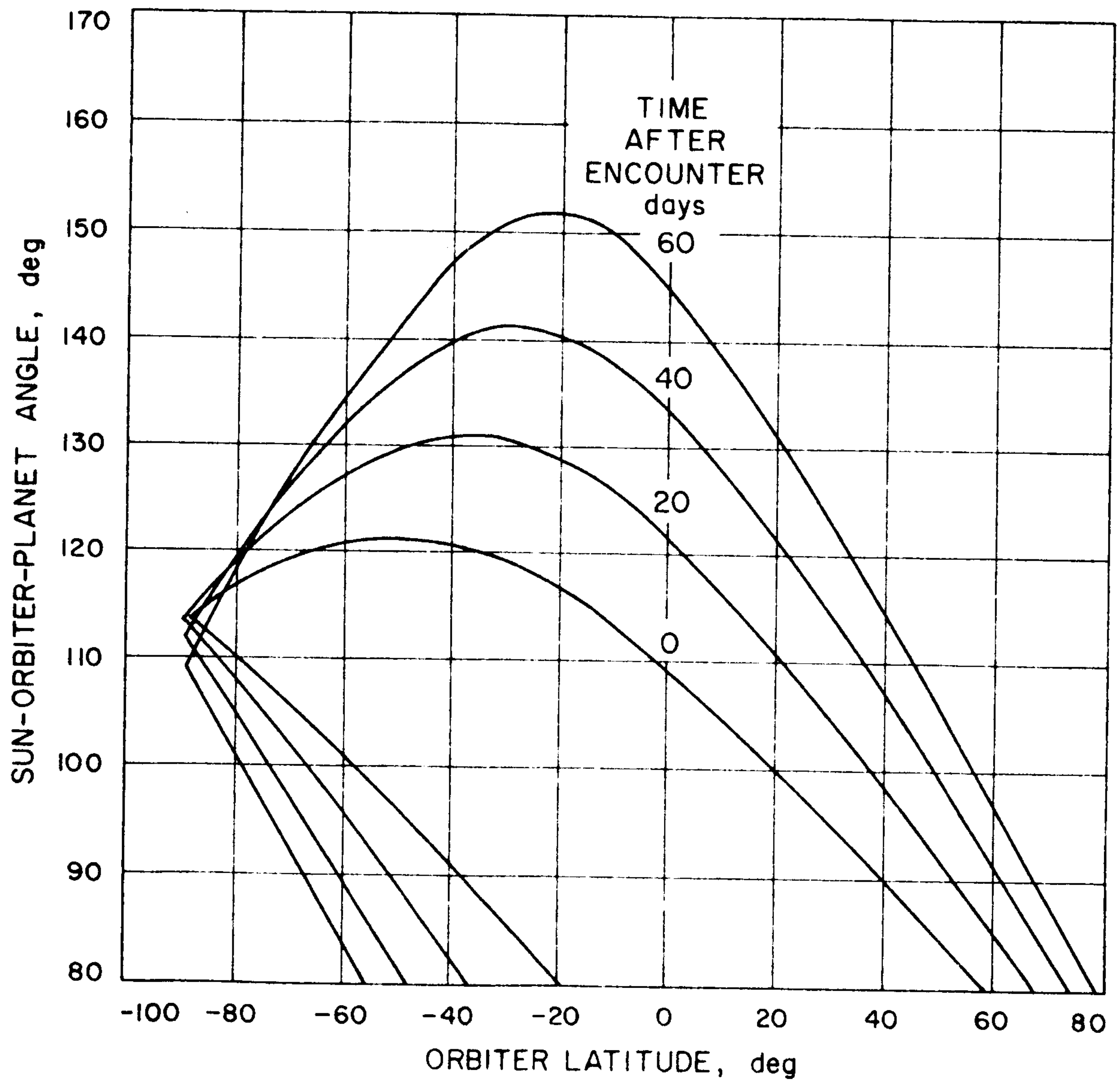


Figure 4-61. Sun-Orbiter-Planet Angle vs Latitude During Single Orbit at 0, 20, 40, 60 Days (2300-km Circular Polar Orbit)

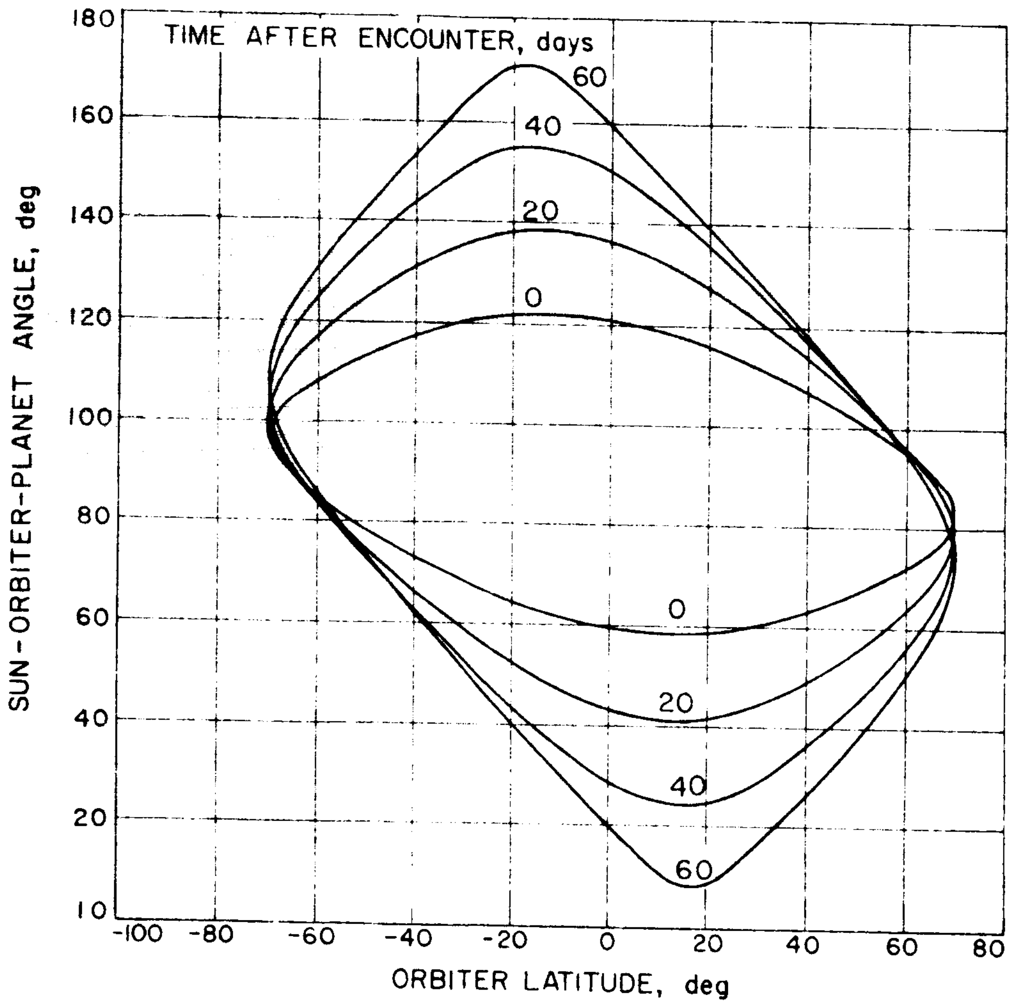


Figure 4-62 Sun-Orbiter-Planet Angle vs Latitude During Single Orbit at 0, 20, 40, 60 Days (Elliptical Orbit)

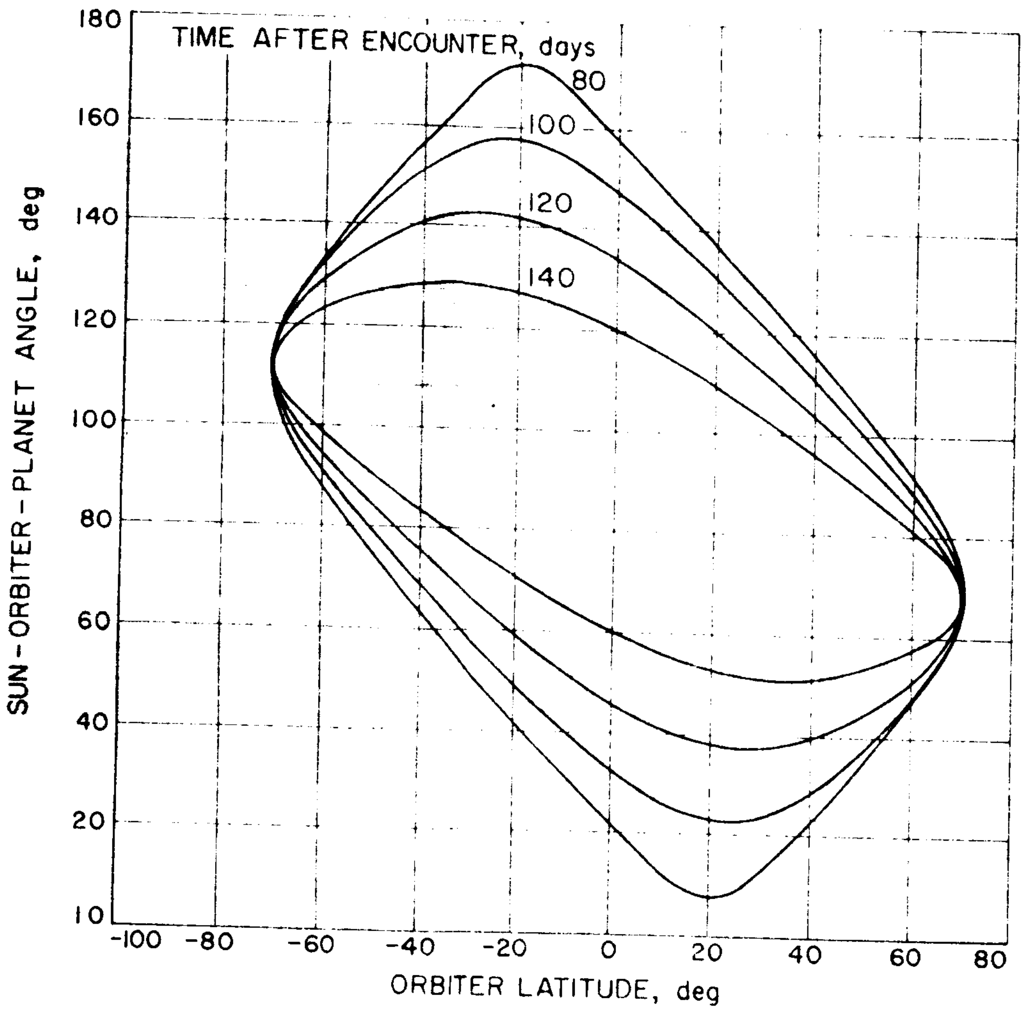


Figure 4-65. Sun-Orbiter-Planet Angle vs Latitude During Single Orbit at 80, 100, 120, 140 Days (Elliptical Orbit)

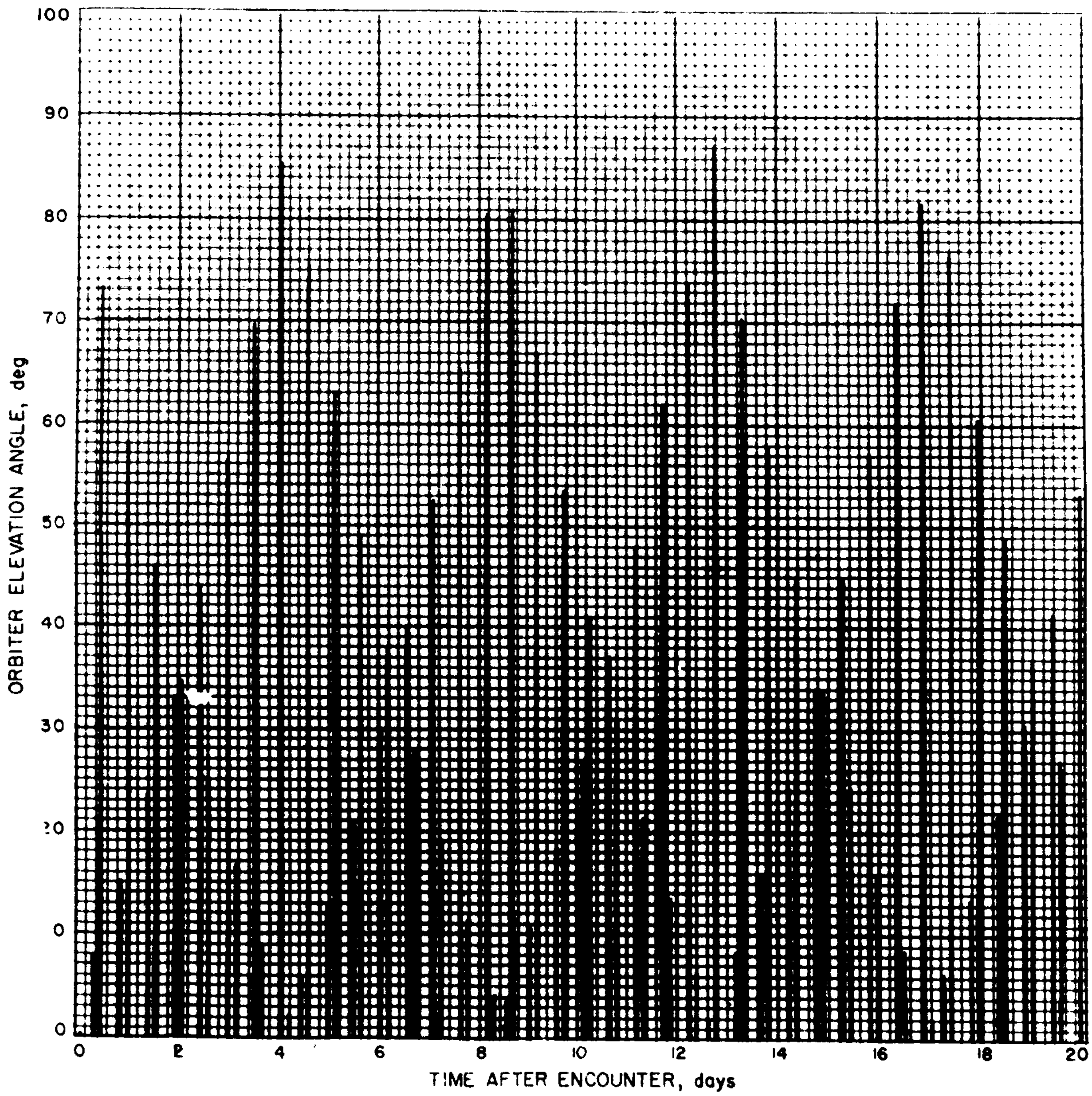


Figure 4-64. Elevation Angle of Orbiter During First 20 Days
(2300-km Circular Polar Orbit)

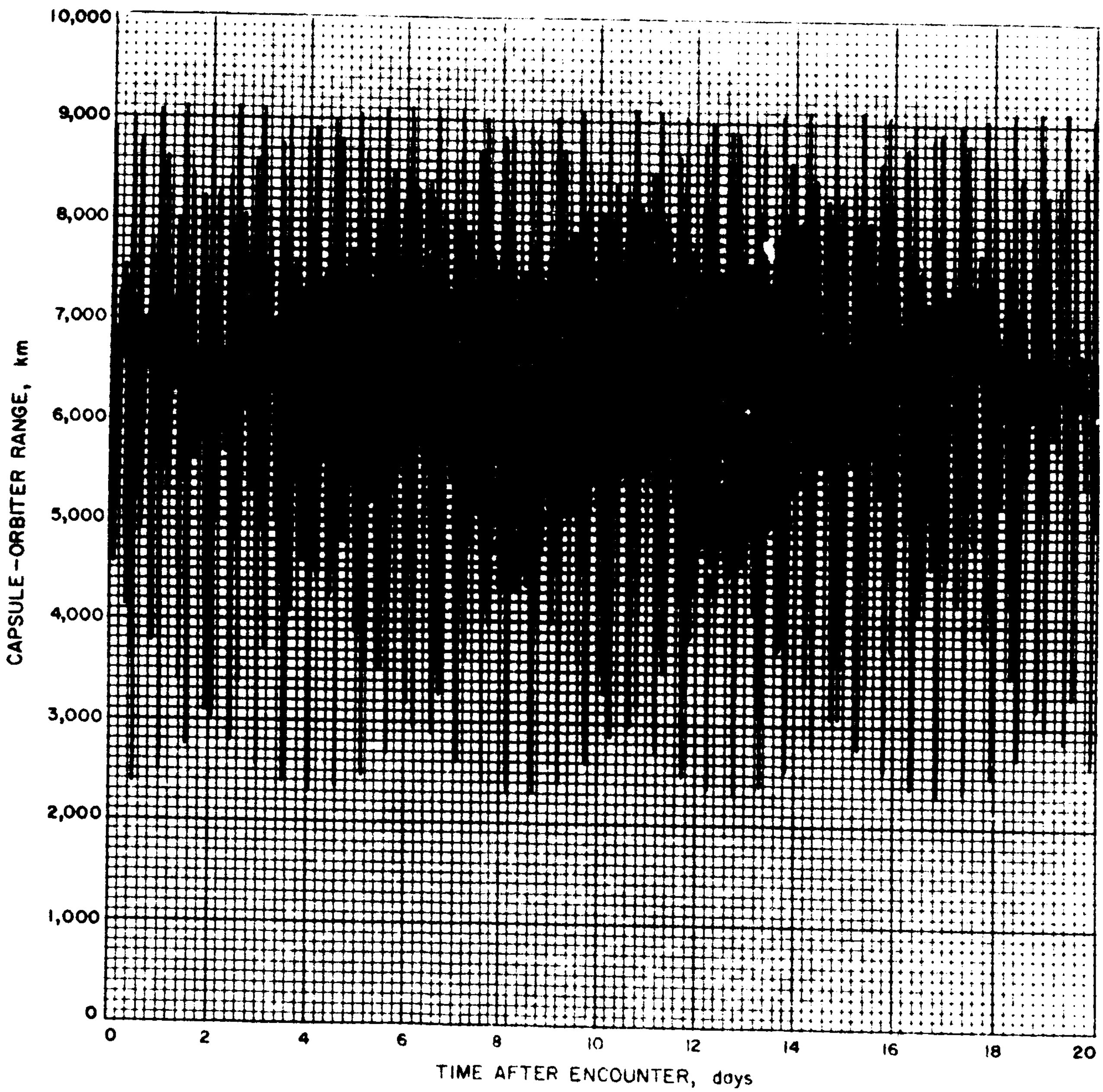


Figure 4-6[illegible]. Capsule-Orbiter Range [illegible] During [illegible] 20 Days
[illegible]

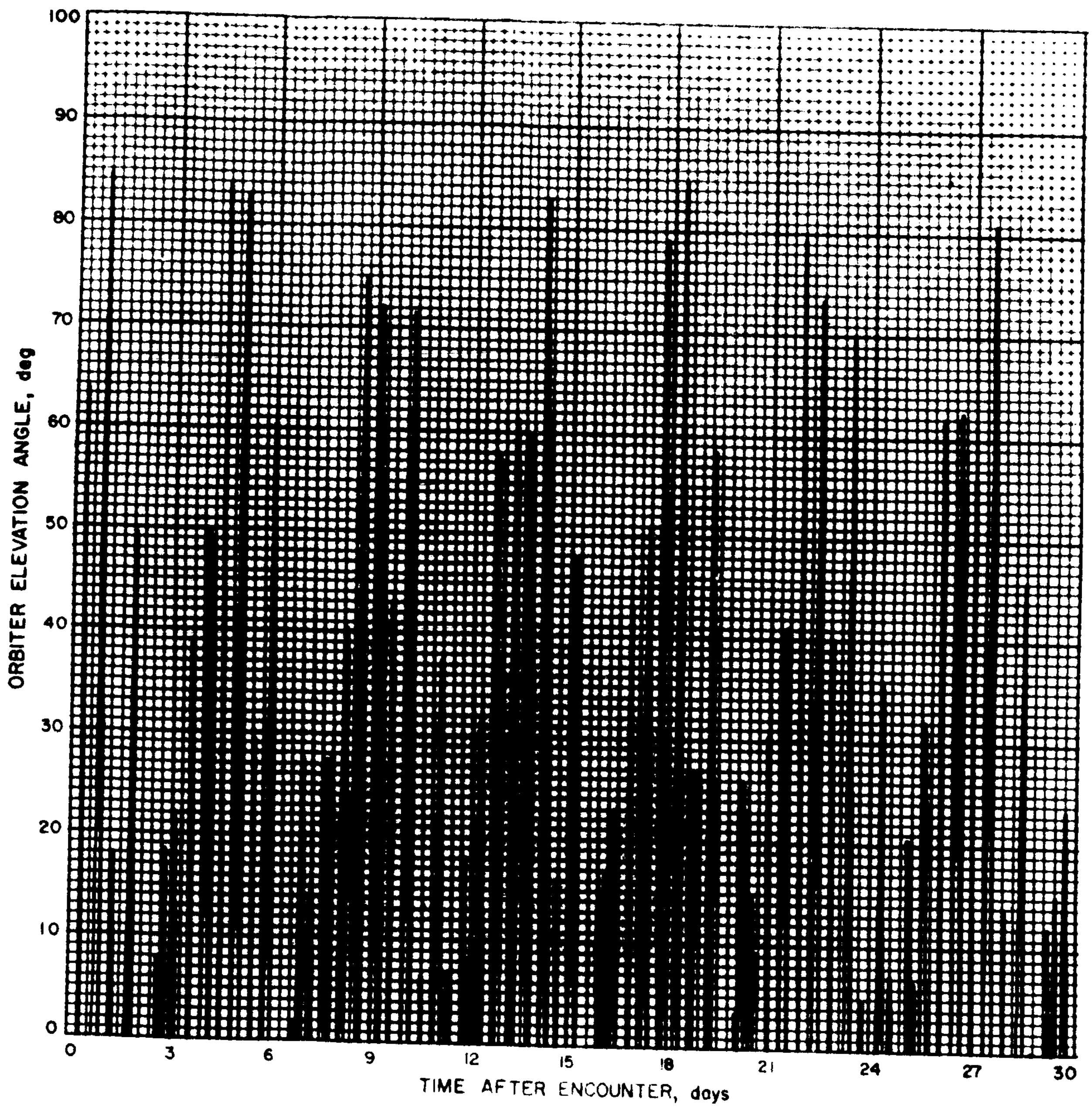


Figure 4-66. Elevation Angle of Orbiter During First 30 Days
(Elliptical Orbit)

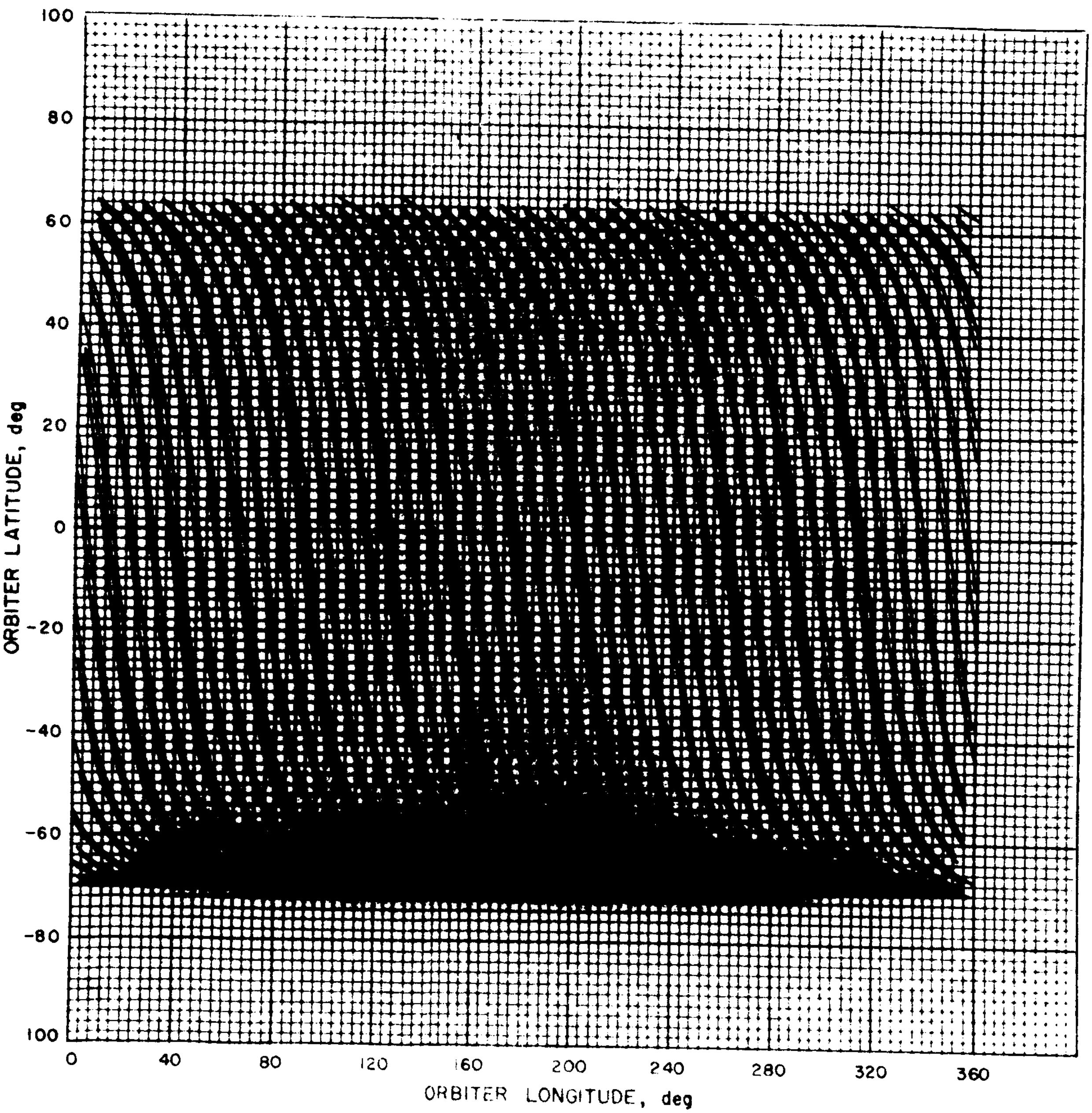


Figure 4-67. Orbiter Latitude vs Longitude During First [illegible] Days
([illegible])

inclination since this is about the minimum allowable inclination from a surface coverage point of view. However, the declination of the incoming asymptote (for the nominal Type II 1969 Mars trajectory) is such that 33 deg is the minimum attainable inclination without resorting to a plane-change maneuver. Hence 33 deg inclination orbits were studied instead.

In order to make a comparison between high and low inclination orbits, two types of orbits were investigated: a nominal altitude (2500 km) circular orbit, and a nominal altitude (1800 by 10070 km) elliptic orbit, both inclined 33 deg to the Martian equator.

The capsule landing site used for the circular case was at the equator, since the study was initiated before the exact location of the desired landing area was known. The capsule landing area for the elliptic case, which was studied later, was 10 deg North latitude, the center of the desired area. It should be noted that, although the capsule landing site for the 70 deg inclination elliptic orbit was on the equator, it is felt that the change in view periods for such a high inclination orbit which would result from choosing a landing site at 10 deg North latitude would not be materially different.

The most notable result of the study is the magnitude of the expected increase in the time the orbiter spends above 30 deg elevation angle (with respect to the capsule horizon). The 70 deg inclination nominal elliptic orbit is above 30 deg for an average of 150 min/day for the first five days, whereas the 33 deg inclination orbiter is above 30 deg for an average of 256 min/day for the first five days, an increase of almost two hr/day. Another desirable result is that Canopus is never occulted for the 33 deg inclination orbit throughout the desired lifetime of the orbit, whereas it is occulted for 30–35 minutes per orbit for the 70 deg inclination orbit. Undesirable results are that the Sun is occulted for about an hour per orbit (everyday), and the Earth is occulted after only 10 days. For the circular orbit, the 33 deg inclination orbiter spends an average of 120 min/day above 30 deg for the first five days compared to 55 min/day for a polar orbiter. Occultation-time studies were not made for circular orbits because it appears that elliptic rather than circular orbits will be used. The above effects and other results are shown in Tables 4-6 and 4-7 and Figures 4-68 through 4-72.

It should be noted that, although for the particular elliptical orbit studied the Sun was occulted for a few hours every day, it would still be possible to use a 33 deg inclination orbit and still have the Sun continuously visible for several days by increasing the perigee radius, but this latter expedient is generally undesirable. Hence, in performing the final tradeoff analysis, one must keep in mind the fact that not only inclination but periapsis and apoapsis heights could be varied. Also, studies on dispersed trajectories, similar to those which follow, must be done for low and medium inclination orbits so that the resulting variations can be noted.

8. Dispersed Trajectory Orbits

In a previous part of paragraph 6, results were presented for two nominal orbits about Mars, a 2300 km altitude circular polar orbit, and an elliptical, 1800 km by 10070 km altitude, 70 deg inclination orbit. Results of studies on orbits which assume that the incoming asymptote is displaced 500 km from the nominal position will not be presented.

The studies assume that the incoming trajectory is dispersed either:

a) 500 km high
b) 500 km low
c) 500 km to the left
d) 500 km to the right

It was further assumed that the capsule trajectories were dispersed in the same direction as the bus trajectories. This assumption is valid only when the dominant guidance errors are due to orbit determination uncertainties rather than maneuver execution errors. It was felt that at this time the current study did not warrant a Monte Carlo or equivalent generation of random pairs of capsule and orbiter trajectories.

If a simple retro-maneuver scheme is assumed which removes the nominal retro-velocity at periapsis, then the trajectory dispersions result in the following orbits for the elliptical case (1800 by 10070 km altitude, 70 deg inclination nominal conditions):

a) 2300 km by 12285 km elliptic, 70 deg inclination
b) 1300 km by 8060 km elliptic, 70 deg inclination
c) 1800 km by 10070 km elliptic, 75 deg inclination
d) 1800 km by 10070 km elliptic, 65 deg inclination

where a, b, c, and d refer to the above indicated displacements. When these same dispersions occur for the circular polar orbit, elliptic orbits result for the high and low displacements. However, as these orbits are still very nearly circular, for simplicity, it was assumed that circular orbits resulted for these cases. This assumption also provides additional information on two other possible "nominal" circular orbits which might be considered for an orbiter mission. The above dispersions then yield:

a) 2800 km circular polar orbit
b) 1800 km circular polar orbit
c) 2300 km circular orbit, 95 deg inclination
d) 2300 km circular orbit, 85 deg inclination

Table 4-6. Capsule Line-of-Sight Parameters for Dispersed Trajectories

Type[a] \ Day	[b] First Rise Time Above 30 deg (min)	Max Elevation During First Rise (deg)	Max Elevation During First Five Days (deg)					Time Above 30 deg During First Five Days (min)						Max Capsule Orbiter Range (km)	Min Capsule Orbiter Range (km)	[c] Surface Track Separation (km)
			1	2	3	4	5	1	2	3	4	5	Avg			
ENO	461	64	84	50	19	48	84	236	133	0	89	295	151	13000	1800	6630
ELO	1253	15	55	56	57	58	58	149	146	126	118	118	132	11000	1300	5340
EHI	562	52	59	66	51	51	81	37	192	153	103	207	140	15000	2300	8130
ERI	467	72	30	53	22	46	86	251	162	0	[illegible]	310	152	12000	1800	6630
ELE	461	55	80	46	21	52	88	272	112	0	87	281	142	12000	[illegible]	6630
ETH	1024	4	59	99	65	77	57	297	312	312	45	375	256	1300	1800	6630
CNO	662	8	73	46	56	79	56	37	37	74	[illegible]	[illegible]	[illegible]	4600	2200	3170
CLO	590	4	52	81	58	42	49	15	44	74	15	15	33	4000	1800	2760
CHI	749	21	84	50	60	75	70	74	162	30	74	74	[illegible]	5200	2900	3590
CRI	642	14	59	45	70	80	73	44	74	59	44	74	59	4600	2200	3170
CLE	662	13	80	36	58	45	70	59	44	44	44	59	50	4600	2200	3170
CTH	187	32	23	65	62	51	32	101	120	130	115	112	120	4600	2200	3170

[a] Key:
- E = elliptic
- C = circular
- NO = nominal
- LO = low dispersion
- HI = high dispersion
- RI = right dispersion
- LE = left dispersion
- TH = 33 deg inclination

[b] Time after orbit establishment

[c] Surface distance between two successive equatorial crossings

Table 4-6A. Orbit Parameters for Dispersed Trajectories

Type[a]	Periapsis Height (km)	Apoapsis Height (km)	Period (hr)	Inclination (deg)
ENO	1800	10070	7.61154	70
ELO	1300	8061	6.13226	70
EHI	2300	12285	9.32798	70
ERI	1800	10070	7.61154	65
ELE	1800	10070	7.61154	75
CNO	2300	2300	3.63730	90
CLO	1800	1800	3.17057	90
CHI	2800	2800	4.12493	90
CRI	2300	2300	3.63730	85
CLE	2300	2300	3.63730	95

(a) Key:

E elliptic
C circular
NO nominal
LO low dispersion
HI high dispersion
RI right dispersion
LE left dispersion

The parameters investigated and the results obtained were those which were significant from a communications and a photographic point of view, and the important points to be noted are the following: For the elliptic orbits the initial rise time above 30 deg elevation varies from 461 min to 1255 min after orbit establishment. The total time above 30 deg for the first day varies from 37 min to 251 min, and the average time per day above 30 deg for the first five days varies from 132 to 162 min. With a 500 km high dispersion the maximum communication distance can be as great as 15000 km. The surface distance between successive equatorial crossings is an important parameter for those concerned with surface photography and varies from 5540 to 8130 km. For the circular orbits the initial rise time above 30 deg elevation varies from 662 to 749 min, the total time above 30 deg for the first day varies from 15 to 74 min, the average time per day above 30 deg for the first five days varies from 33 to 71 min, and the surface track separation varies from 2760 to 3560 km. These and other results are given in Tables 4-6 through 4-7.

Table 4-7. Orbit Parameters for 70 Deg and 33 Deg Inclination Elliptic Orbits

Day / Type[a]	Time per Orbit that Sun is Occulted (min)								Time per Orbit Earth Occulted								Time per Orbit Canopus Occulted				Time per Orbit Vega Occulted			
	0	20	40	60	80	100	120	140	0	20	40	50	80	100	120	140	0	40	80	120	0	40	80	120
ENO	0	0	0	55	55	20	0	0	0	0	0	0	0	55	50	0	35	34	32	31	0	0	0	0
ETH	67	85	70	50	50	65	65	75	0	0	80	85	70	70	80	85	0	0	0	0	0	0	0	0

[a] ENO nominal 70 deg inclination elliptic orbit
ETH 33 deg inclination elliptic orbit

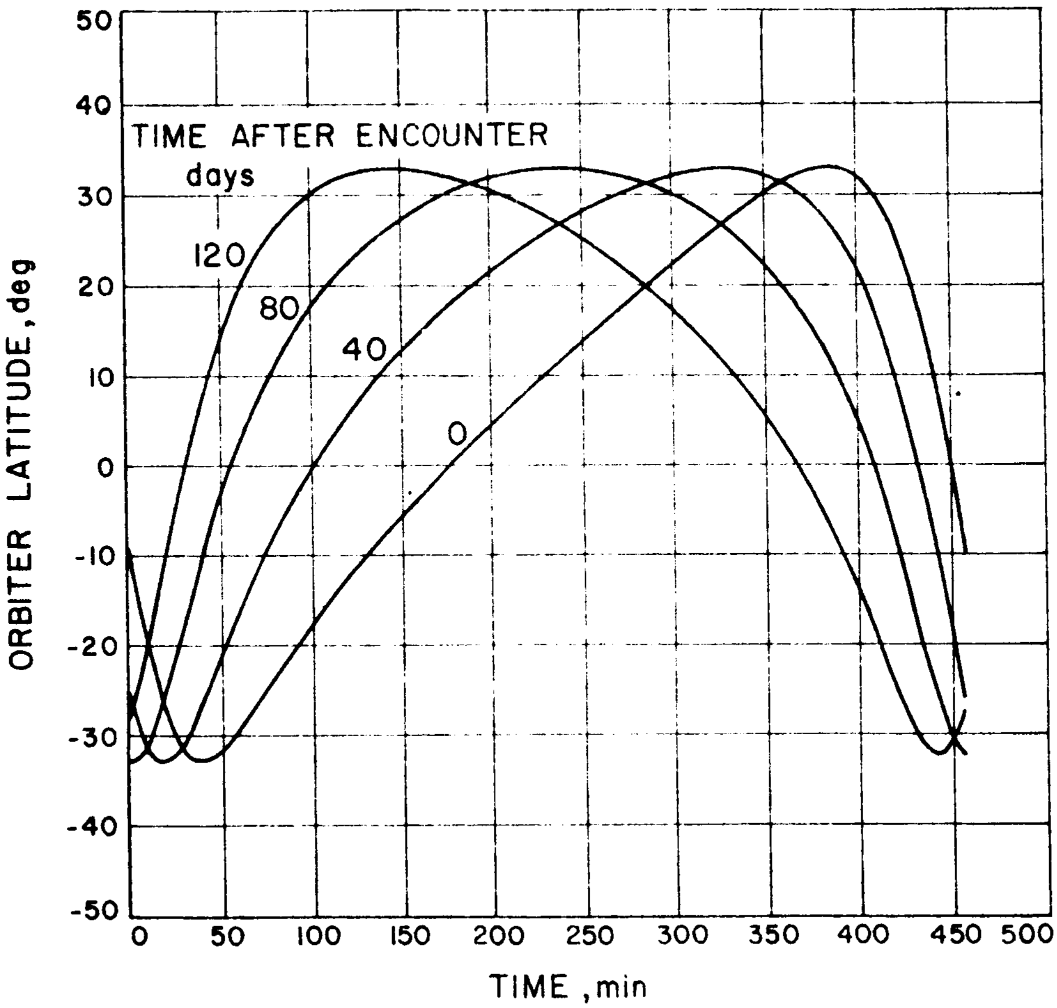


Figure 4-68. Orbiter Latitude vs Time During Single Orbit at 0, 40, 80, 120 Days (33-Deg Inclination Elliptical Orbit)

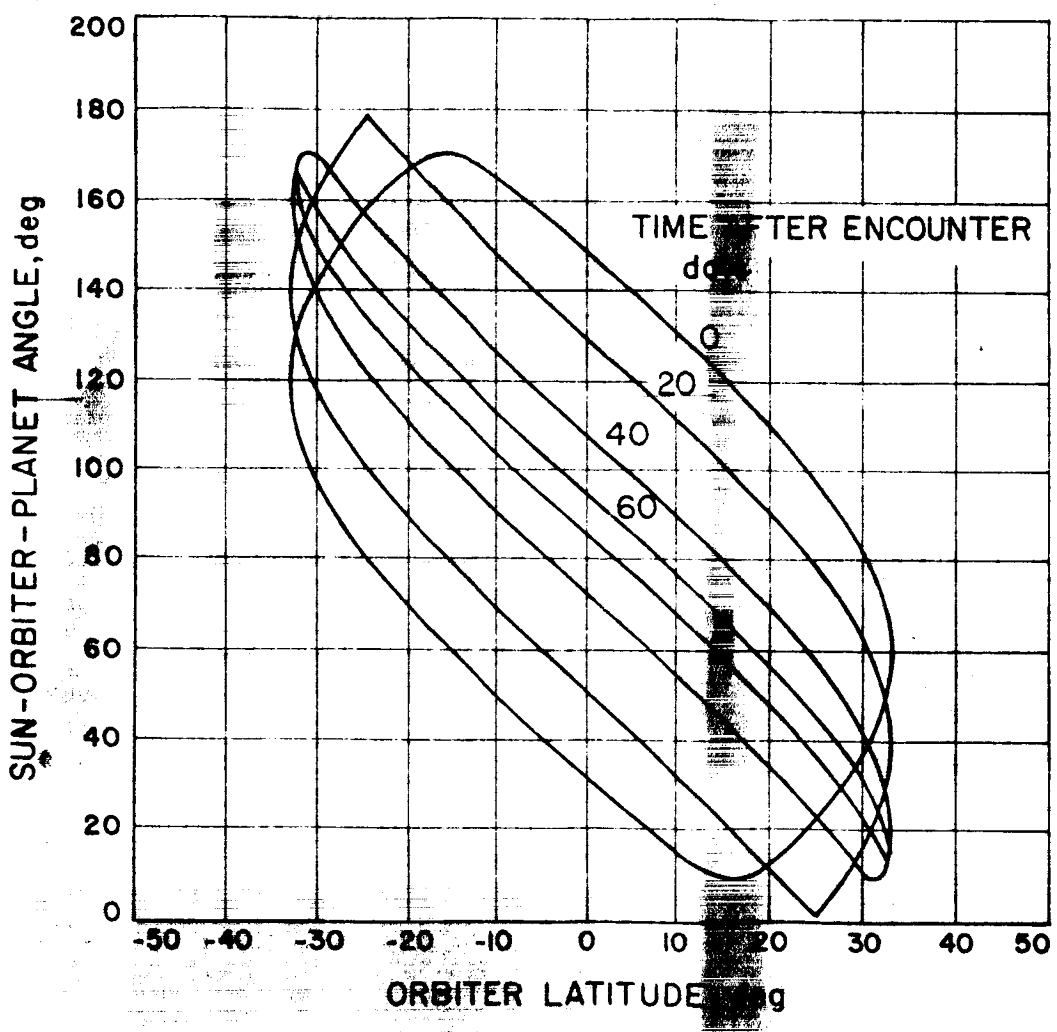


Figure 4-69. Sun-Orbiter-Planet Angle vs Latitude During Single Orbit at 0, 20, 40, 60 Days (33-Deg Inclination Elliptical Orbit)

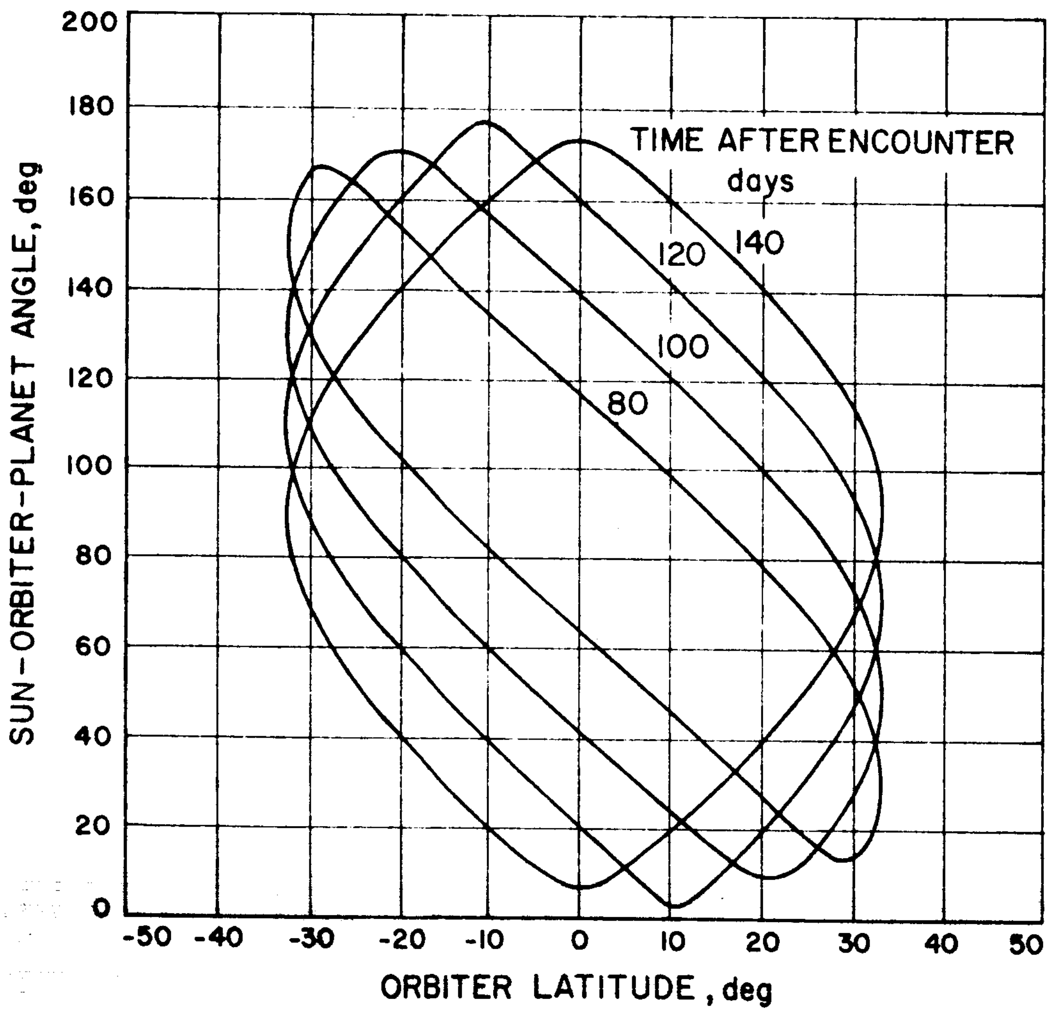


Figure 4-70. Sun-Orbiter-Planet Angle vs Latitude During Single Orbit at 80, 100, 120, 140 Days (33-Deg Inclination Elliptical Orbit)

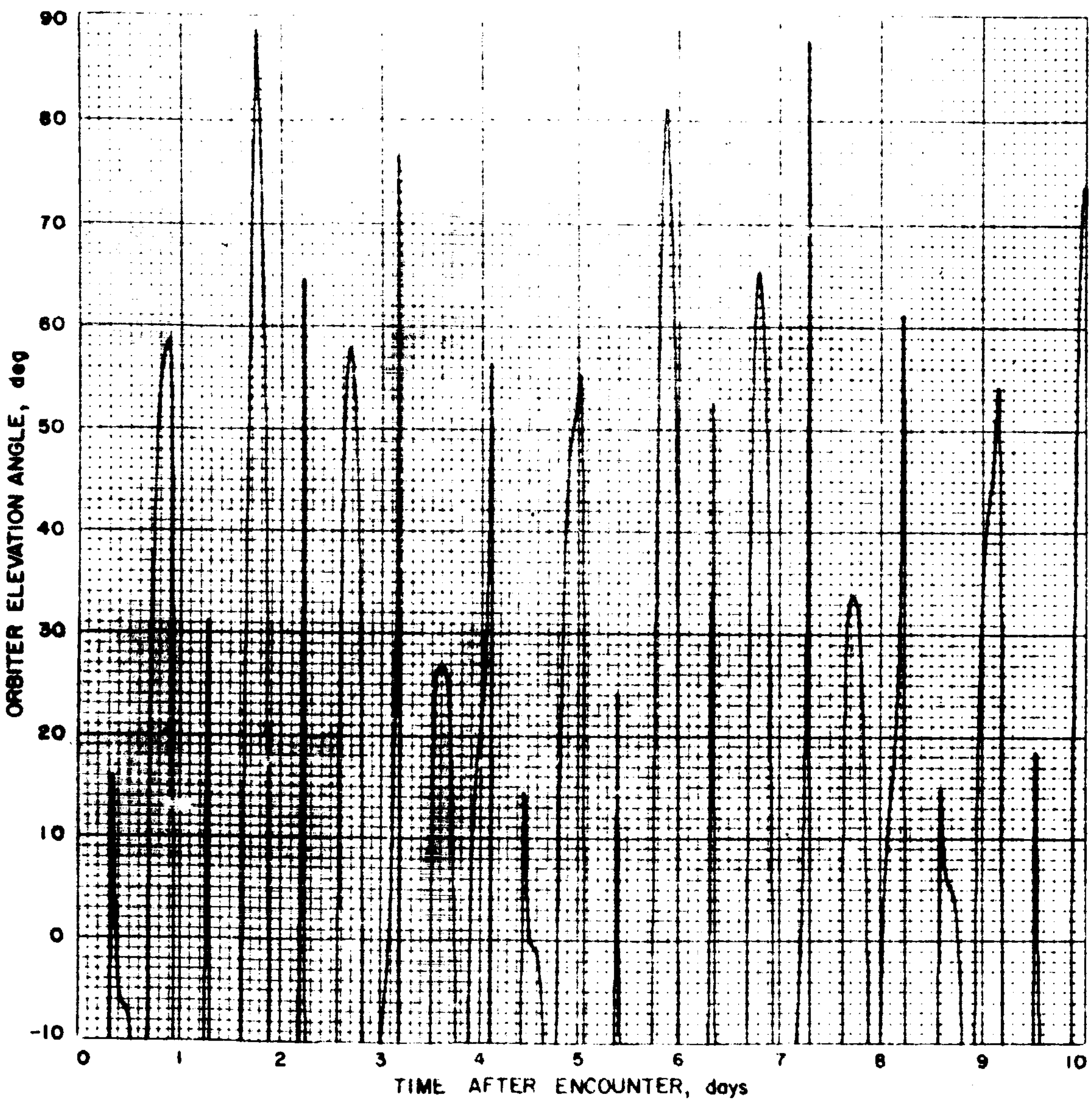


Figure 4-[illegible] Elevation Angle of Orbiter During First 10 Days
(55-Deg Inclination, [illegible] Orbit)

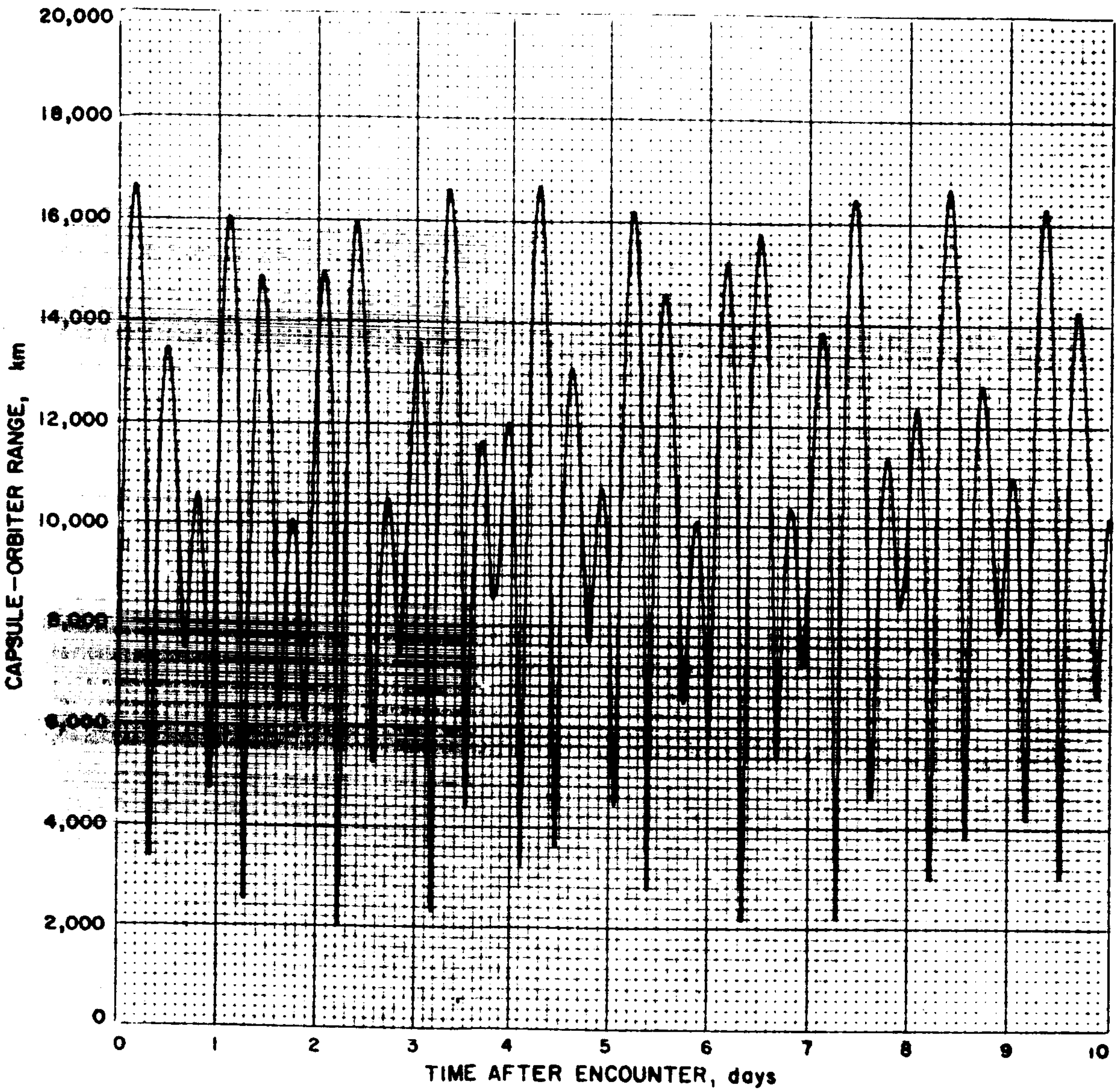


Figure 4-72. Capsule-Orbiter Range (km) During First 10 Days
(33-Deg Inclination Elliptical Orbit)

REFERENCES

1. J. Kork, "Satellite Lifetimes in Elliptic Orbits", I. A. S. paper No. 62-5, January 1962.
2. B. Billik, "Survey of Current Literature on Satellite Lifetimes", A. R. S. Journal, V. 32 No. 11, November 1962, pp. 1641-1650.
3. W. D. Parsons, "Orbit Decay Characteristics Due to Drag", A. R. S. Journal, V. 32, No. 12, December 1962, pp. 1876-1881.
4. A. J. Kiliore, "The Effect of Atmospheric Drag on the Lifetime of Mars Orbiters", JPL TM 312-307, May 6, 1963.
5. G. F. Schilling, "Limiting Model Atmospheres of Mars", Rand Corp. Report R-402-JPL, August 1962.
6. J. W. Chamberlain, "Upper Atmospheres of the Planets", The Astrophysical Journal, V. 136, No. 2, pp. 582-593, September 1962.
7. D. N. Vachon, "The Density of the Upper Atmosphere of Mars", General Electric Co., Space Sciences Lab., Report No. R62SD58, May 1962.

D. GUIDANCE AND CONTROL

1. Guidance System and Mission Profile Selection Factors

The choice of guidance system and mission profile for the combined orbiter-capsule mission is affected by the same factors that affect the orbiter-only mission. A brief indication of these factors is reproduced below from EPD-139, Volume II.

"During the current study, it has become increasingly clear that a careful tradeoff study between orbit accuracy, propulsion performance, mission success probabilities, and mission value must be conducted. Such a study has not yet begun; the remarks made here are speculations designed to indicate some of the questions that must be considered in such a study. Two mission sequences will be considered here:

1) A single near-Earth midcourse maneuver, a heliocentric-phase midcourse maneuver, injection into a satellite orbit, and orbital trim.
2) A single near-Earth midcourse maneuver, approach guidance, and injection into a satellite orbit.

These mission sequences are similar in terms of the total amount and complexity of the equipment carried aboard the spacecraft. The differences lie in the amount of equipment required to function before the spacecraft enters an initial orbit about the planet and in the propulsion performance.

For the first mission sequence, the midcourse equipment is required to function twice, and the injection equipment once before the spacecraft is injected into an orbit about the planet. This orbit may be less accurate than that produced by approach guidance, thus requiring the orbit-trim mode to function, and requiring extra propellant to trim the orbit to the desired accuracy.

In the second mission sequence, the midcourse equipment is still required to function twice, and possibly a third time, before an initial orbit is achieved, because the approach mode used the midcourse equipment for the execution of maneuvers. In addition, the added equipment for the approach mode, which is roughly equivalent to that used in orbital trim, is required to function before the initial orbit is achieved. However, once this orbit is achieved, trim may not be necessary. Thus, the propellant requirements may be somewhat less for achieving an orbit of the same accuracy.

It would appear that the first sequence is somewhat more desirable from the point of view of mission success probabilities, because even if the trim mode failed to function and produce an orbit of the desired quality for the full list of experiments, some or all of the experiments probably could be performed with at least partial effectiveness if the terminal equipment functioned to produce a crude orbit. This possibly-attractive capability would be bought at the price of added weight and a reduction in the experience of using approach guidance (which appears more necessary for lander missions than for orbiter missions)."

In addition to the above-mentioned factors, the time of capsule release and the way in which the times of arrival for bus and capsule are separated affect the mission profile. A discussion of these factors may be found in Chapter 4 of the present volume. A discussion of spacecraft and capsule maneuver errors, which are a major factor in the above choices, are discussed in the Guidance Section of the current volume, together with estimates of orbit-determination accuracy achievable with earth-based radio tracking.

The accuracy analysis for maneuver execution discussed here is an initial study based on a particular mechanization for which performance three estimates are currently available. The mechanization is similar to those used on the Ranger and Mariner spacecraft. The use of the particular mechanization discussed here as an example should not lead the reader to the conclusion that this particular mechanization is recommended as the desired choice for Voyager mechanization. Further study of other mechanizations, some of which may have improved accuracy characteristics, is necessary before such a recommendation can be formulated.

2. Spacecraft and Capsule Maneuver Execution Accuracies

This section describes the accuracies with which trajectory-correction and orbit-injection maneuvers may be executed from the spacecraft bus or the landing capsule.

a. Bus Maneuvers

The type of maneuver specifically analyzed consists of turning the spacecraft sequentially about two or more of the body-fixed control axes to point a rocket motor in a particular direction in space. Once the pointing of the rocket motor has been accomplished, the motor is fixed until the desired increment has been added to the spacecraft velocity.

Prior to initiating the maneuver, the spacecraft obtains an inertial reference frame for the initial set of turns by measuring the position of certain celestial bodies with respect to the body-fixed axes.

When the reference frame for the turns has been established, the attitude control system is commanded to turn the spacecraft through a preset angle about one of the spacecraft axes. After this turn has been completed, it is followed by one or more turns performed in a similar manner about one or more of the other spacecraft axes. The turns are precomputed from knowledge of the inertial direction desired for motor firing, knowledge of the spatial orientation of the spacecraft, and knowledge of the orientation of the motor in the spacecraft. Maneuver commands most likely will be computed on the Earth and transmitted to the spacecraft (although the entire process may be done on the spacecraft itself, if the necessary information and equipment are present).

After the motor is pointed in the proper direction by the turns, the motor is ignited on command and burns until an increment of the proper magnitude has been added to the spacecraft velocity. During the burning phase of the maneuver, the thrust vector is maintained in the proper direction by an autopilot in the spacecraft. The magnitude of the velocity increment can be controlled either by a timer or by an integrating accelerometer.

In the process of making a maneuver as described above there are errors which affect the ultimate pointing direction of the thrust vector and its magnitude. Errors in controlling the pointing direction and the magnitude of the thrust vector cause deviations in the velocity increment added to the spacecraft velocity. These deviations in turn cause errors in the final result desired of the maneuver, whether it be miss distances of a particular trajectory or periapsis altitude of a desired orbit. The maneuver errors arise from inaccurate knowledge of the spacecraft trajectory before computing the maneuvers (tracking errors) and from the inability of the spacecraft equipment to accurately execute the computed maneuver (execution errors). Only execution errors are discussed here. Tracking errors are discussed in Chapter 4.

The actual sources of error depend on the method of mechanizing functions involved in executing the maneuver. There are several ways in which the turns might be made and several ways in which the magnitude of the velocity increment might be controlled. Some of these have been analyzed and are discussed in the following paragraphs. Table 4-8 indicates the various sources of error for different mechanizations while Table 4-9 indicates the resultant error in maneuvers using various combinations of the possible mechanization schemes. Pointing error is defined to be the inaccuracy in the direction of the thrust vector from both the turn error and the autopilot error; magnitude proportional error is defined to be the percentage of the desired velocity magnitude by which the actual velocity magnitude may be incorrect; and magnitude resolution is defined to be the smallest amount by which one commanded velocity may differ from another.

In computing the error in pointing, magnitude, and resolution, it is assumed here that the errors are statistically distributed in a Gaussian fashion and the appropriate errors for each combination of the maneuver mechanization are added in a root-sum-square manner to arrive at a resultant error. The numbers which are given in the following paragraphs and in the tables are assumed to be equivalent to the three-sigma (3σ) values of the errors. Even though resolution errors cannot strictly be said to have a Gaussian distribution, this assumption does not significantly alter the results of the accuracy estimates.

Table 4-8. Error Sources and Their 3σ Values for Spacecraft Maneuvers

Error Source	3σ value
Spacecraft turns (including the acquisition of inertial references)	1.0 deg
Autopilot pointing	
With integrator (path guidance loop) – (side velocity error)	0.3 m/sec
Without integrator – (pointing error)	1.0 deg
Velocity Magnitude	
Accelerometer scale factor	0.05%
Null offset	10^{-4} g
Resolution	0.02 m/sec
Fixed impulse solid	1.0%
Timer shutoff accuracy	0.01%
Timer resolution	10^{-3} sec
Thrust level accuracy	2.0%
Rocket motor commanded shutoff uncertainty (equivalent time error at full thrust)	10^{-2} sec

Table 4-9. Resultant 3σ Root-Sum-Square Maneuver Error for Various Maneuver Mechanizations

Maneuver Mechanization	Pointing error (2 axes combined) (mrad)	Magnitude error	
		Proportional (%)	Resolution (m/sec)
Without path guidance loop, with accelerometer @ 5g full scale, 1 g thrust level	25	0.05	0.10
Without path guidance loop; with fixed impulse solid motor	25	1.0	....
Without path guidance loop, with timer shutoff, 1 g thrust level	25	2.0	0.11
With path guidance loop; with accelerometer @ 5g full scale; 1g thrust level	17.5	0.05	0.10
With path guidance loop, with fixed impulse solid motor	17.5	1.0	....
With path guidance loop; with timer shutoff; 1g thrust level	17.5	2.0	0.11

It is estimated that the turn error of the spacecraft will cause no more than a 17.5 mrad offset of the actual thrust direction relative to the desired thrust direction for the range of possible mechanizations considered for effecting the turns. The mechanization analyzed for the present estimates is based on the Ranger and Mariner series of spacecraft, in which rate sensing gyros are used in an integrating loop to control the turning of the spacecraft. The 17.5 mrad error includes any errors in locating the inertial reference for the turns.

The autopilot error analysis presumes an advanced version of the Ranger and Mariner types of autopilots. For reasons of stability, the autopilot deflects the thrust vector of the rocket motor to pass through the actual spacecraft center of mass. In one scheme, no compensation is made for such deviations of the thrust vector from its nominal direction in the spacecraft axis system (which was the axis system aligned by the initial turns). The pointing error resulting from this type of autopilot system is also on the order of 17.5 mrad. A second scheme utilizes an integrating "path guidance loop" to compensate for the movement of the thrust vector in space by commanding equal and opposite rotation of the spacecraft. The error associated with this type of autopilot is more difficult to ascertain, because it is a function of the length of the maneuver, and other parameters. For a motor burning time of more than 10 to 20 seconds, the error will not cause the transverse velocity error to exceed a value of 0.1 to 0.3 m/sec. For shorter burn times the error will never be greater than twice the error of the system without a path guidance loop.

Velocity magnitude errors depend upon the type of mechanism selected to shut off the motor after the proper velocity increment has been achieved. The most simple system (from a control point of view) is that in which the propellant loading determines the magnitude of the velocity increment. A fixed-impulse solid rocket motor can be made that has no more than 1 percent dispersion in the velocity magnitude. Such a motor would probably be used, if at all, only for orbit injection retro; liquid propellant motors remain the most likely choice for all bus propulsion functions. Time shutoff of the rocket motor is also straight-forward. Timer scale factor accuracies of better than 0.01 percent and timer resolutions of 0.001 second or better are relatively easy to achieve. The accuracy of the thrust level can be predicted to about 2.0 percent. An accelerometer can be made to measure the actual velocity added to the spacecraft with an accuracy of 0.05 percent if the temperature is known to ±10° F at the accelerometer (wider variations of the knowledge of temperature introduce greater inaccuracies); accelerometer null offsets of about 10^{-4} g's also affect the accuracy of an accelerometer. A reasonable resolution for an accelerometer with a full-scale range as high as 5.0g is 0.02 m/sec. Lower acceleration levels allow better resolutions but the change has an insignificant effect on overall accuracy. Also, with added shutoff computer complexity (more counter stages and higher rebalance pulse rate) this resolution could be reduced an order of magnitude for velocity increments as high as 3 km/sec.*

*To accommodate the retro maneuver as well

b. Capsule Maneuvers

A capsule maneuver begins with a change in orientation of the spacecraft, and is followed by the separation and stabilization of a capsule. This separation and stabilization is followed by the addition of an increment to the capsule velocity with the firing of a rocket motor. The capsule trajectory change is desired in order that the capsule may impact a target planet while the bus is injected into orbit around it, and to separate the arrival times of the capsule and bus (unless execution errors are such that this task must be performed by a bus trajectory change). Prior to separation, the capsule is mounted in a fixed orientation relative to the spacecraft coordinates and the rocket motor is fixed to the body of the capsule. The particular case analyzed is described by the following sequence. First, the spacecraft, with capsule attached, makes a series of turns similar to, and in the same way as, the turns described in the section on spacecraft maneuvers. These turns orient the capsule rocket motor to the proper spatial direction for the desired capsule velocity increment. The capsule is then freed from its mountings.

A set of small cold-gas jets located on the sides of the capsule are used both to spin the capsule for stabilization during motor firing and to give the capsule a small separation velocity relative to the bus. The definite separation velocity is provided to avoid the danger of damaging the bus when the capsule rocket is fired. An interval of time between the separation of the capsule from the bus and the actual firing of the rocket motor is necessary to accumulate adequate separation distance.

At the end of this interval, the rocket is ignited and burns until the proper velocity increment is added to the capsule. The motor thrust line is nominally colinear with the nominal capsule spin axis, the direction of which is determined by the spacecraft turns prior to separation of the capsule.

In the process of making a maneuver such as described above there are execution errors which affect the final pointing direction and magnitude of the actual capsule velocity increment. In the same manner as for the spacecraft maneuver, tracking and execution errors ultimately affect the location, time, and velocity of capsule impact. (If excessively large, these errors could cause the capsule to miss the planet altogether.) The errors discussed here are only those associated with the execution of the maneuver itself, and not those associated with inaccurate knowledge of the premaneuver trajectory. These latter errors are discussed in Chapter [illegible].

As with the spacecraft maneuvers, the sources of errors in making a capsule maneuver depend on the particular type of mechanization chosen to execute the maneuver. For preliminary analysis, a particular limiting case was chosen for investigation. Table 4-10 describes the capsule characteristics for this case. Deviations of the pointing

Table 4-10. Capsule Parameters

Parameter	Assumed Value
Capsule	Body of Revolution
Mass	1000 lb
Moments of inertia Spin axis Both perpendicular axes	 30 slug ft^2 90 slug ft^2
Spin rate	100 rev/min
Separation velocity	2 ft/sec
Time to spin	2 sec
Typical velocity increment	200 m/sec

magnitude, and resolution errors remain the same as defined for spacecraft maneuvers. Table 4-11 lists the contributors to capsule maneuver error and indicates the resultant error for such a maneuver.

The turns executed by the spacecraft prior to the separation of the capsule are assumed to be in error in the same manner as the turns for the spacecraft maneuver. The error expected from this phase of the maneuver, including the acquisition of inertial references, is on the order of 17.5 mrad.

The act of separating the capsule will cause an offset in the actual direction of the nominal spin axis of the capsule. This error arises from several sources, the most important ones being the initial misalignments of the capsule axes relative to the spacecraft axes and the tipoff errors associated with the separation. The latter errors arise from nonsimultaneous release of the fasteners holding the capsule, the capsule striking the bus during the process of moving away, and other such items. The overall effect of these errors is estimated to cause an offset of the actual spin axis from the desired direction by about 17.5 mrad.

In the process of spinning the capsule, giving the capsule a separation velocity, and firing the rocket while spinning, there are several opportunities for errors to arise. The rocket motor thrust line, although aligned to the nominal spin axis of the capsule as carefully as possible, will not be exactly colinear with the actual spin axis; this is due to effects such as mechanical misalignments of the motor itself, noncolinearity of the actual line thrust with the motor geometrical axis, and misalignment of the jets which spin the capsule. All these items interact in a complex manner to cause effective pointing errors of two kinds, each of which is assumed to be relatively independent of the other. The first is the effective error

Table 4-11. Capsule Maneuver Errors

Error Source	Pointing Error (mrad)	Magnitude Error (%)
Spacecraft turns prior to separation	17.5	...
Initial offset of nominal spin axis at separation (includes tipoff)	17.5	...
Resultant error from jet force unbalance and misalignments	24.0 (may be reduced by appropriate techniques)	...
Coast time effects	Unevaluated	...
Effect during burning of thrust axis misalignments from capsule spin axis	5.0 (typical number)	...
Solid rocket impulse	...	1.0
Resultant 3σ root-sum-square error	34.8 + that due to coast time effects	

caused by the thrust axis misalignments; this error is estimated to be on the order of 5.0 mrad. The second is the effective pointing error caused by the misalignment and unbalance of the spin jets; this error is estimated to be less than 24.0 mrad.

Immediately after separation and spin-up comes a period of coasting while the capsule is drawing away from the spacecraft to reach a safe distance for firing the rocket motor. Preliminary estimates of this coast period are one-half to three-quarters of an hour. If the capsule were a perfectly rigid body in the classical mathematical sense, it would remain in the same state throughout this coast period because no external forces act on the capsule during this time. In this ideal case, the capsule could be considered as a spinning gyroscope which maintains its spin axis in a constant inertial direction. However, an actual capsule is not rigid, and internal forces acting on the capsule cause a stressing of the body and a consequent conversion of kinetic energy to heat. The angular momentum remains unchanged, however, and thus the energy loss may cause the spin-axis direction to move considerably with respect to the body axes, although not with respect to an inertial frame. The result is an uncertainty in orientation of the capsule in space and an uncertain spin rate at the time that the rocket motor is fired. The actual magnitude of this effect has not yet been estimated. A detailed study of the rigidity of the capsule and its contents, the spin rates, moments of inertia, energy relationships, and other related items is necessary before such an evaluation can be made.

The thrust magnitude error associated with the rocket motor firing is assumed to be that of a fixed impulse solid motor, i.e., about 1.0 percent of the velocity increment desired.

In computing the total pointing and velocity-magnitude errors it is again assumed that the individual errors discussed above are statistically distributed in a Gaussian fashion and that the numbers quoted above are equivalent to the 3σ values of these errors. Table 4 summarizes the individual error sources and indicates the root-sum-square resultant error. It is again emphasized that the numbers quoted here are for a single capsule configuration and functional mechanization for effecting a capsule maneuver.

3. System Requirements and Capabilities

As mentioned in paragraph 1, the guidance system must ensure that (1) the orbiter portion of the spacecraft is placed in an acceptable satellite orbit for gathering scientific data, and (2) the capsule entry and landing conditions are properly controlled. The accuracy requirements for satisfying these two basic objectives will become tighter with each successive Voyager mission. Early Voyager missions to Mars may typically require control of the landing point to ±500 km (3σ), control of the periapsis altitude to ±500 km (3σ) and control of the orbit inclination to ±5 degrees (3σ). These numbers are based upon study results for a Mars mission, where the desire to control the orbit precession rate (to maintain acceptable lighting) and the capsule landing area (within Syrtis Major) requires about this degree of accuracy. Later Voyager mission requirements may reduce these numbers to 10 km, 100 km, and 1 degree, respectively. As Venus is nearly twice the size of Mars and as orbit precession and landing-point control do not appear to be as important as for Mars, the accuracy requirements for early Voyager missions to Venus could probably be relaxed to 1000 km. This degree of accuracy would permit the entry angle into the Venus atmosphere to be maintained below 30 degrees, if necessary.

Considering the question of partial mission success, the minimum achievements which could still permit a positive mission gain would probably be the establishment of any nondegenerative planetary satellite orbit and/or the safe landing of an exploratory entry capsule at any point on the planetary surface (with communication capability). These minimum objectives could be met with guidance accuracies as poor as ±5000 km (3σ, impact-parameter plane). This number is more reasonable for an orbiter mission than for an entry capsule, as the latter may require more accuracy in controlling the entry angle in order to effect a safe atmospheric transit than that degree of accuracy which permits any type of capture (shallow or steep).

Advanced launch vehicles such as Saturn should achieve 3σ target accuracies of 50,000-500,000 km (semi-major axis) with injection guidance during the Earth-escape phase of flight. This broad accuracy range covers the span of short Venus missions to long Mars missions. The exact target dispersions due to component errors in the Saturn guidance system are classified and have been included in a separate document (EPD-139, Volume III, Part 2) for typical Type I and Type II 1969 Mars trajectories. This document also contains the corresponding midcourse velocity requirements which are unexpectedly small.

The performance of one midcourse correction will improve the control of target miss to 2,000-15,000 km (3σ), depending upon the trajectory sensitivity (Type I Venus vs Type II Mars, for example) and the method of velocity control (accelerometer vs timer cutoff). The first midcourse maneuver will normally be performed within the first 5 days after injection. When only one maneuver is performed, the contribution to target miss from maneuver execution errors is competitive with or dominates the orbit uncertainty which existed immediately prior to the maneuver. The performance of a second midcourse correction could occur at almost any time between 2 days after the first maneuver and 10 days before encounter. The earlier time might be preferred in the event of a large nonstandard error which may have been committed by the first maneuver, whereas the delay of a second correction would probably be preferred when the first maneuver performed within predicted tolerance; such perturbations as gas leaks and solar pressure effects can be better determined with longer tracking times. The actual selection of the time for performing the second midcourse maneuver will probably not be made until during the actual flight, after the results of the first maneuver have been observed. The second maneuver should reduce the target miss to about 500-3,000 km (3σ), depending upon when the second maneuver is performed. Most of this error is due to orbit uncertainty rather than maneuver execution errors. Two midcourse maneuvers would make possible an excellent flyby mission, a crude orbiter mission (which could be improved with orbit trim), or a crude entry and landing mission; if the capsule is released during the approach phase of flight (or from orbit) when the orbit knowledge is much better than it was at the time of the second midcourse maneuver, a much more accurate entry and landing can be performed.

Although discussed in more detail under the separate classified paragraph (Chapter 4, IV. B. 4) dealing with injection and midcourse guidance, the problem of requring very small midcourse velocity increments should be mentioned here. With the currently planned 1500 lb single-thrust-level engine, the Voyager spacecraft will not be able to add a velocity increment smaller than about 0.283 m/sec (with a 3σ accuracy of 0.026 m/sec or about 10 percent). If one accepts the small midcourse velocity requirements listed in 2. B, then it would appear that some portion of all possible first midcourse maneuvers and a very large percentage of possible second midcourse maneuvers would be too small to be performed by the 1500-lb-thrust engine. Several ways to handle this situation are:

1) Bias the aiming points at injection and at each midcourse maneuver by a few m/sec; such a procedure may even be necessary to guarantee a probability as low as 10^{-4} of impacting Mars in the presence of a conservative confidence level in the operation of the spacecraft motor.

2) Wait until the corrective velocity requirement reached the minimum capability.
3) Correct the miss but alter the time of flight.
4) Employ a two-thrust-level main engine.
5) Install a small separate gas-expulsive system capable of adding only a fraction of a m/sec.
6) Use the attitude control system to provide very small velocity increments.

Although it currently appears that the most desirable of these alternatives would be (1), (2), and possibly (3), the significance of the other alternatives as related to the likelihood of small maneuvers should be more thoroughly investigated during the course of future studies.

Orbit knowledge will continue to improve as onboard measurements are combined with Earth-based tracking data during the approach phase of flight. The performance of an approach guidance correction would reduce the target miss to about 1500 km (3σ), which would satisfy the early Voyager requirements. It should be noted that this degree of accuracy could be achieved without actually performing a corrective maneuver during the approach phase by (1) in the case of the orbiter portion of the mission, "trimming" the orbit to this accuracy or better after the orbit had been established, and (2) for the lander portion of the mission, simply releasing the entry capsule when the orbit knowledge was acceptable without the main trajectory first requiring a correction. It would probably be wise, however, to perform at least one approach correction as (1) Voyager spacecraft will be designed with multiple-maneuver capability and such maneuvers should normally be performed in a routine manner, (2) the performance of an approach maneuver might bring to light some important nonstandard effect, the knowledge of which could be a very critical input for the performance of a successful retro maneuver, and (3) because the orbit-trim maneuvers would then be either unnecessary or small, a larger effective payload could be placed in orbit.

Studies to date[24] indicate that position in a planetary satellite orbit can be determined to an accuracy in the region of 1-10 km (3σ) from either Earth-based tracking (accurate two-way doppler) or onboard measurements of planetary angular diameter and Sun-orbiter-Planet angle. This degree of accuracy in orbit knowledge is quite satisfactory not only for possible orbit-trim maneuvers but also for reconstruction of photographic stereo pairs. As a matter of fact, as later Voyager missions tighten the accuracy requirements for controlling the entry corridor and landing point, it may be necessary to first enter a satellite orbit before detaching the exploratory capsule. This procedure should permit the highest accuracy both in determining the orbit and in controlling the landing point. At this point, it would be worthwhile to discuss the potential advantages and disadvantages of releasing the entry capsule before or after going into orbit. The advantages of the preorbit release are:

1) A larger total mission payload results.
2) There is an opportunity for a successful entry mission even if the bus fails to go into orbit and/or the capsule-to-Earth link fails.

The advantages of release from orbit are:

1) There is plenty of time to determine the satellite-orbit parameters very [illegible] and in an unhurried fashion, so that one might tend to feel more [illegible]
2) A slower, [illegible] entry [illegible]
3) Data [illegible]
4) [illegible] unless, in the preorbit-release case, a [illegible] correction was made to the capsule trajectory during the terminal phase of flight. It therefore appears that preorbit release will be satisfactory for early Voyager missions with the Saturn IB/SV (or any less capable launch vehicle), whereas postorbit release may be preferred for advanced Voyager missions as well as manned missions to the planets.

Finally, it is important to outline the capabilities of Earth-based orbit determination which might be anticipated for the 1969 Mars mission. Table 4-12 lists the assumed values for data noise and physical-constant uncertainties, and the resulting orbit uncertainties are shown in Figures 4-73 and 4-74. Figure 4-73 is based upon a 181-day Type I Mars trajectory launched April 2, 1969, and Figure 4-74 assumes a 279-day Type II trajectory launched February 14, 1969. C[illegible] doppler data (1 sample/min) was taken from three stations from injection (I) to I + [illegible] days; continuous doppler coverage from one station for about ten hours per day was taken from I + 15 to encounter. For both trajectories the first midcourse correction was performed at I + 5 days [illegible] halfway through the flight. The [illegible] statistic [illegible] Saturn [illegible] injection to Mars [illegible]

Referring to Figure 4-73, it can be seen that the 1σ [illegible] ellipse at Mars [illegible] about [illegible] days with [illegible] and station-location uncertainties. The orbit knowledge is then [illegible] by the first maneuver execution [illegible] later. The [illegible] knowledge [illegible] again by a second midcourse correction at the halfway point. [illegible] between a few [illegible] after the first maneuver and [illegible] days prior to encounter. Continued tracking produces only a small improvement in the knowledge of the orbit [illegible] approaches the [illegible]

Table 4-12. Assumed 1σ Data Noise and Physical-Constant Uncertainties

Error source	Assumed values for current study	Conservative estimates for 1969	Realistic estimates for 1969
Effective doppler noise (m/sec)	0.015	0.01	0.003
GM of the Earth (parts per 10^6)	10	1.5	0.6
Astronomical unit (km)	500	250	150
Ephemeris error (seconds of arc)	0.1 (100 km for Mars)	0.1 (100 km for Mars)	.05 (50 km for Mars)
Station location uncertainties u* (km) v w	100 100 120	10 10 30	5 5 15
Uncertainty in solar radiation force upon spacecraft (results from uncertainties in the reflective properties of the spacecraft)	5%	5%	2%

*the w axis is parallel to the Earth's spin axis and the u and v axes lie in the equatorial plane with u containing the Greenwich meridian.

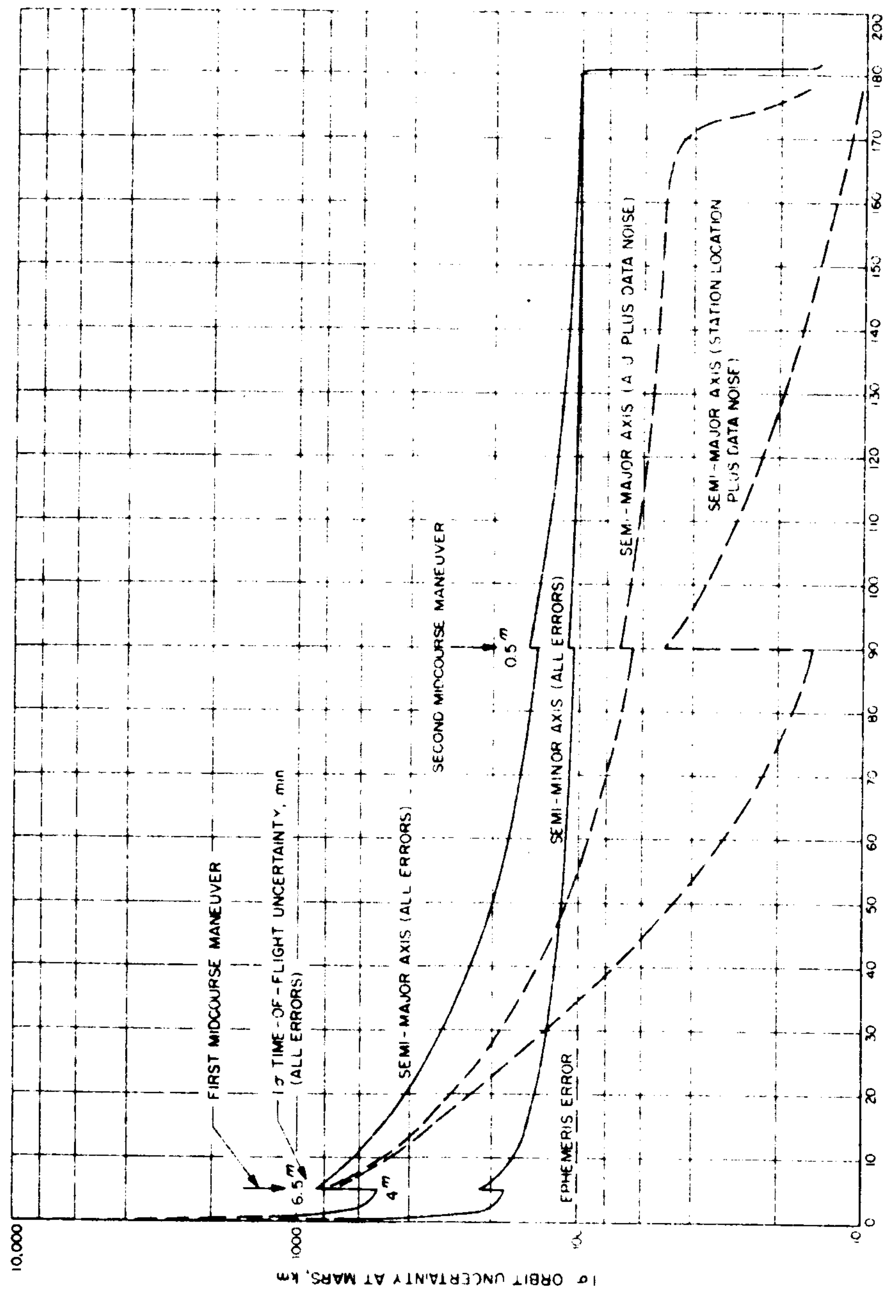


Figure 4-73. Orbit Determination Estimates for 181-Day Type I Mars Trajectory

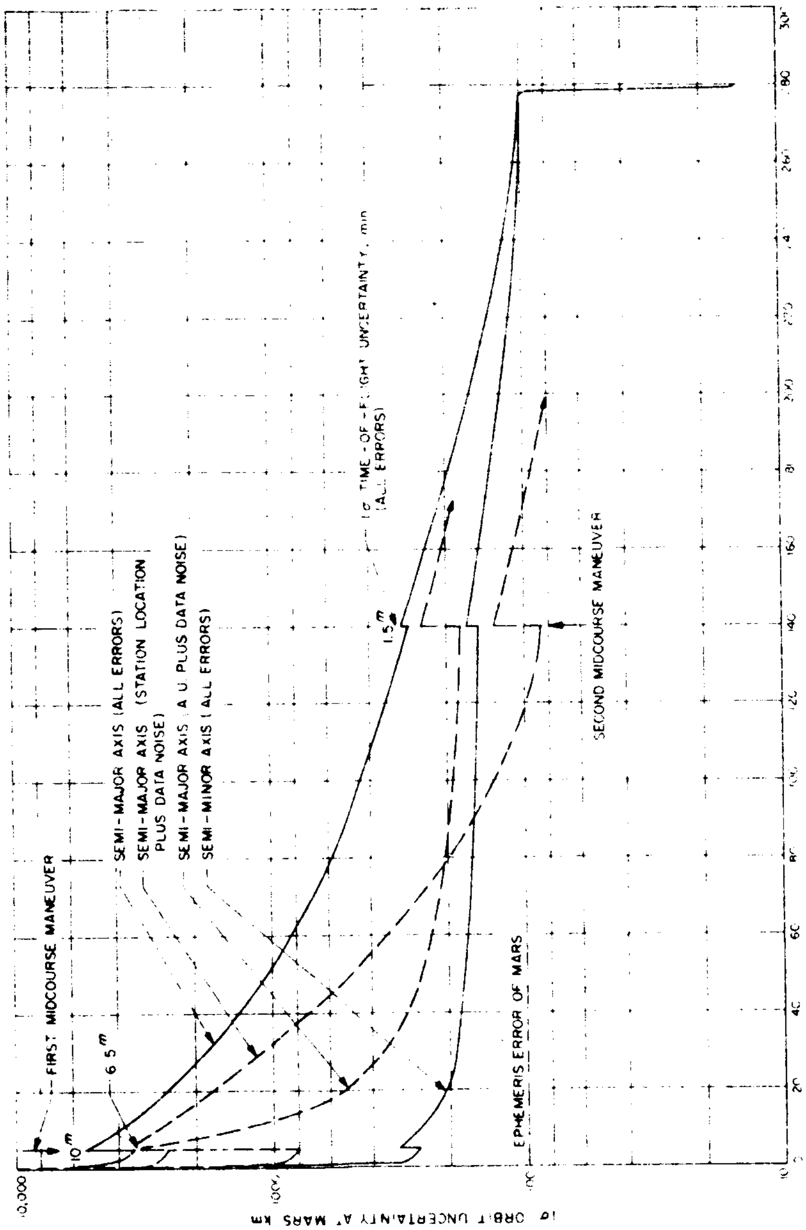


Figure 4-74. Orbit Determination Estimates for 279-Day Type II Mars Trajectory

ephemeris error of Mars. It is only during the last few hours before encounter that the gravitational influence of Mars allows an improvement in the knowledge of the near-Mars orbit down to a value of 10-20 km.

Figure 4-74 illustrates the orbit determination estimates for the longer Type II Mars trajectory, and it can be seen that the 1σ semi-major axis at I + 5 days is about 3500 km and is primarily due to station-location uncertainty. Although this 3500-km error is seven times larger than the corresponding uncertainty for the Type I trajectory, it has no significant effect upon the final mission accuracy which is essentially the same for both Type I and Type II trajectories.

The inclusion of an approach guidance system which would perform measurements of various angles between the Sun, Mars, and a star (probably Canopus) would provide an important independent determination of the near-Mars orbit and should also improve the orbit knowledge during the period when the Earth-based tracking is not contributing much new information a few days prior to encounter. By E-2 days, the approach guidance data should have permitted the orbit to be determined to the order of 50 km, 1σ. The entry capsule would be released at this time while the velocity requirements are not severe.

4. Injection and Midcourse Guidance.

This subject is classified and is the topic in EPD-139, Volume III, Part 2.

5. Satellite Orbit Determination

One of the techniques to be used for satellite orbit determination is Earth-based doppler radar. Other techniques using on-board optical measurements are currently under study. A study [1), 2)] using Earth-based doppler data indicates that an artificial satellite of Mars, equipped with the proper transponding equipment, can be located to within 1 km in position, and 1 m/sec in velocity after an observation period of two days.

The orientation and shape of the orbit are not predominant factors in determining the ultimate accuracy, with one notable exception. An orbit with a very low inclination with respect to the ecliptic plan causes a large uncertainty in the out-of-plane position and velocity component. Orbits of this type are not currently being considered for the Voyager series of spacecraft; therefore, this constraint is not considered to be restrictive. If a priori data are considered, the restriction can be eased.

It would appear that the attainable accuracy is sufficient for the reduction of stereo photographs.

REFERENCES

1. Barber, T. A., "Voyager Satellite Orbit Determination Accuracy Using Earth-Based Doppler Radar - Part I - Classical Orbital Element and Position Accuracies", T.M. 312-306, May 8, 1963

2. Barber, T. A., "Voyager Satellite Orbit Determination Accuracy Using Earth-Based Doppler Radar - Part II - Inertial Position and Velocity Accuracies", T.M. 312-312, May 17, 1963

E. SPACECRAFT CONTROL

1. General Discussion

The spacecraft control functions for this mission are quite similar to those discussed in EPD-139, Volume II. The primary difference is the fact that a separable capsule is included in this mission. The spacecraft control functions are again divided into three categories, attitude control, articulation control, and autopilot control. A new estimate of the fuel required for the attitude control system is given. Some comments will be made on the use of simultaneous lobing techniques for pointing the high-gain antenna toward the Earth. The only new articulation control function is that of pointing a receiving antenna on the bus in such a direction that communication between the bus and capsule can be maintained. The autopilot functions for this mission are the same as those discussed in Volume II. Since a landing capsule has not been previously considered, its control requirements will be discussed in detail.

2. Attitude Control

Further consideration of the hot and cold gas systems for attitude control mentioned in Volume II indicated that changes in the weight estimate are necessary. The mission considered for the orbiter with a separable capsule was assumed to consist of 250 days in transit to the planet and 150 days in orbit about the planet. Three to four days before reaching the planet, the capsule is to be released and, when the bus reaches the planet, a final retro maneuver is performed. Since an appreciable fraction of the mass of the bus before this maneuver is contributed by the propellant, the inertia of the bus after the final retro is approximately one-third of that before the maneuver. Depending on the magnitude of the solar torques or other disturbing torques, this decrease in inertia could increase fuel consumption during the orbital cruise phase by as much as a factor of three. This possible increase in fuel consumption can be eliminated by decreasing the minimum available impulse per nozzle actuation after the maneuver. The most convenient way to do this is to decrease the actuation torques which requires a dual torque level system. In estimating the gas consumption, it was assumed that the moment of inertia of the spacecraft (including the capsule) during transit was 4000 slug-ft^2 about all three axes. For orbital cruise, a value of 1300 slug-ft^2 was used. Except for the new fuel, tankage and plumbing weight estimates given below, the weight and power requirements for the remainder of the system are the same as given in Volume II.

For a hot gas actuation system, the estimated weight of fuel and tanks is 120-lb with an additional 28-lb necessary for hardware. These weights were obtained assuming that the minimum desirable thrust from a nozzle is 0.1 lb, that the torquing radius was 4 ft, and that no efforts were made to control unbalanced solar torques. Also, a single thrust level actuation system was used because the hardware needed to implement a dual level system would weigh more than the resulting fuel and tankage weight saved.

If a cold gas system using nitrogen were used instead of the [illegible] system and the configuration were kept the same, the weight of such a system would be approximately [illegible] lb. [illegible] gas system, other mechanizations [illegible]

[illegible]

[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
W[illegible]	[illegible]	[illegible]	[illegible]8	[illegible]

[illegible]

[illegible]

mounted at the ends of the solar panels, the area required for each vane would be approximately 50-ft^2. These vanes could further complicate the field of view problems mentioned above, but the system weight could be reduced to 150 lb including hardware.

Additional weight savings could be achieved for the cold gas system by mounting the nozzle actuators on booms extending from the ends of the solar panels. However, because of the additional hardware required and the added complexity of more folding booms with the resulting decrease in reliability, this technique does not seem to offer significant advantages.

All of the numbers given have been based on a solar panel powered spacecraft with the particular moments of inertia mentioned. For different moments of inertia or for an RTG powered spacecraft, both fuel requirements and hardware weights would differ from those quoted above.

3. Spacecraft Antenna Pointing

The following discussion of the antenna pointing problem is divided into three sections. The first section presents some of the reference systems available for pointing the antenna. Some of the problems of utilizing several different references are presented in the second section and, lastly, a summary of the probable ranges of the more important pointing errors are tabulated.

a. Reference Approaches

In order to fulfill the scientific requirements of the proposed missions, high data rates following planetary encounter are required. Communication can achieve the desired bit rates most efficiently with large diameter antennas provided they can be pointed with sufficient accuracy.

A number of alternate approaches might be considered:

1) A closed-loop RF link employing simultaneous lobing for fine pointing of the antenna with coarse pointing by means of a precalculated program;
2) A closed-loop optical system employing a long-range Earth sensor for fine control and also relying on programmed coarse control;
3) A programmed antenna carefully controlled and calibrated to permit reduction of fixed errors and compensation of variable errors within the requirements of a large diameter antenna;

4) A small diameter programmed antenna with sufficiently increased transmitter power to achieve the desired data rate;

5) A smaller diameter programmed antenna with the present power level and acceptance of a lower bit rate.

[illegible] the advantages and disadvantages of the above system are indicated below and are [illegible].

[illegible] antenna [illegible] independent [illegible] more or less [illegible] control. It has the advantage [illegible] that the Earth [illegible] correct RF signal. However, the possibility [illegible] concern [illegible]. The price paid for [illegible] the flight equipment and a substantial increase in the DSIF load. [illegible] of [illegible] with the 85 and 210-foot dishes to support at least part of each Voyager flight.

Optical line control of the antenna represents one compromise in antenna pointing. It suffers from some inaccuracy, [illegible] the antenna [illegible]; however, it has the advantage of direct information as to the Earth's position without any ground support. It does have several signal and disadvantages. It is a reasonably complex piece of flight equipment, it is not [illegible] when the Earth is [illegible] than 15 deg from the Sun and it is confused by other bright stars and planets in its field of view.

A programmed antenna with carefully controlled and compensated errors is probably the most questionable system from the point of view of feasibility. It is doubtful whether structural deformations due to thermal inputs could be [illegible], compensated or even predicted, particularly during orbital operation. This approach may be quite expensive in terms of preflight calibration and considerably complicates the problem of last minute replacements unless these difficulties can be designed out. [illegible] is reasonably simple and reliable [illegible] achieved.

A higher power transmitter operating through a smaller diameter dish represents a possible method of [illegible] the tight antenna pointing requirements. It has the advantage of [illegible] rate with relative simplicity [illegible] paid for the less optimum [illegible] weight [illegible] power systems. This weight penalty [illegible] by lowering the data rate; however, this would probably not meet the [illegible].

Table 4-13. A Comparison of Several Antenna Pointing - Reference Approaches

Antenna Pointing System	Scientific Acceptability of Data Rate	Complexity of Flight Equipment	Postlaunch Ground Equipment and Personnel Required	Relative Reliability	Unusable Periods	Relative Accuracy Probable Error Range	Adaptability to Nonstandard Flights	Possibility of Confusion	Comments
Closed loop RF link for fine pointing. Programmed coarse pointing	Satisfactory	Relatively complex	Fairly extensive	Least Reliable	1. When Earth is not transmitting 2. When Earth is occulted 3. When not in lock	Most accurate about 0.25 deg	Most adaptable	1. Might be confused by secondary lobe	1. This link could conceivably be closed by ground command
Closed loop optical link for fine pointing. Programmed coarse pointing	Satisfactory	Relatively complex	None	Least Reliable	When Earth Probe Sun angle is less than 5 deg during part of [illegible] & [illegible] 6 mos. Earth occulted	Almost as accurate as above system [illegible] deg	Most adaptable	1. Might be confused by other planet or bright star	The probability of confusion should be evaluated
Programmed fine and coarse pointing. Careful control and compensation of pointing errors	Satisfactory (if feasible)	Complexity depends on the compensation required	None unless compensation is inserted from the ground	Difficult to evaluate at this time	None	Must be very [illegible] between [illegible] 5 deg and [illegible] 5 deg	Least adaptable	None	1. This approach may not be feasible for the required accuracy 2. Requires much preflight calibration
Programmed coarse pointing. Smaller antenna dish with higher transmitter power, less critical to pointing errors	Satisfactory	Relatively simple	None	Most Reliable	None	Probable Accuracy between [illegible] deg and [illegible] deg	[illegible] adaptable than [illegible]	None	1. Several system tradeoffs are required to evaluate this technique
Programmed coarse pointing. Smaller antenna dish with same transmitter power. Less critical to pointing errors	Unsatisfactory	Relatively simple	None	Most Reliable	None	Probable Accuracy between [illegible] deg and [illegible] 5 deg		None	Low data rate may make this approach unworthy of further consideration

b. **Problems in Utilizing Several References**

The antenna servo might have one or more of the following signals available at any time to provide pointing information:

1) A precalculated program referenced to a Sun-star or other reference frame.
2) Direct measurements of the Earth's position from either RF or optical measurements.
3) Ground commands which might be based on the spacecraft's signal strength, on the Earth, or from telemetered monopulse signals from the spacecraft.

If more than one of these signals is available, there is some possibility of confusion or failure; for example:

1) A preflight program could be calculated or loaded incorrectly. It is dependent on the correctness (within specified limits) of the spacecraft Sun-star reference frame. It is dependent on the proper operation of the program sequencer.
2) Direct measurement of the Earth's position by RF systems suffer from confusion from noise and secondary antenna lobes. Optical trackers may be confused by other stars and planets or by reflectors entering the field of view. In addition, these systems rely on programmed coarse pointing of the antenna and, therefore, suffer from the afore mentioned difficulties.
3) The ground command system is dependent on the proper operation of all portions of the command link, including the human operators.

The problem in the design of a specific flight system is one of utilizing these references in a manner to obtain accurate pointing of the antenna at all times. To illustrate this problem, we might imagine a situation where the antenna pointing program indicates that the Earth should be at, say, 35 deg and the RF or optical link indicates a 40 deg Earth position. The antenna system might be designed to treat these conflicting instructions in a variety of ways.

1) Disregard the RF or optical link and remain at the programmed angle.
2) Disregard the programmed angle and rely on the RF or optical signal.
3) Move toward the angle commanded by the RF or optical link but do not deviate from the programmed angle by more than 1.5 or 2 degrees.
4) Move toward the angle commanded by the program, but do not deviate from the angle commanded by RF or optical servo by more than 1.5 or 2 degrees.

In addition to the question of how the antenna should be pointed at any time, there is also a question of whether any of these signals should be capable of altering any other signal or its priority. This raises questions such as:

1) **Should the RF or optical link be capable of affecting the program? If so, automatically or on command?**
2) **Should the command system be capable of instructing the spacecraft to disregard one of the references? If this reference turns out to be the stored program, what instructions should the antenna follow when the RF or optical link is unavailable?**

It is apparent that this list of questions can be quite long. The answers will ultimately depend on the relative reliability of the various references, the nature of their most probable failures, and the amount of back-up equipment that can be tolerated to overcome a failure.

c. Error Summary

A summary of the more important error sources is shown in Table 4-15. Their importance is dependent on such factors as the system mechanization, the celestial geometry, and the spacecraft environment.

Several examples will serve to illustrate this point:

1) Such errors as attitude control system deadband are of no importance if the antenna is aligned with the attitude control sensor signals rather than with respect to the spacecraft frame.
2) The importance of the sensor errors depends on the position of the Earth in the Sun-star reference system.
3) If simultaneous lobing is operating, only RF signal offset and antenna servo errors are present.

d. Antenna for Orbiter-Capsule Communications

A new requirement on spacecraft functions is the pointing of an antenna to receive telemetry data from the lander after separation. Since this antenna is to be pointed at the planet, the horizontal platform has been considered as a possible mounting location. The trajectory and orbit are such that the "look" direction of the antenna will be satisfactory for this pointing direction.

Table 4-14. Antenna Pointing Error Summary

Error Source	3σ Error Range (deg)
Attitude Control Sun Sensors	±0.3
Attitude Control Star Tracker	±0.3
Attitude Control Deadband	±0.12
Attitude Control System Electronics Offset	±0.07
Static Structural Errors	±0.5
Thermal Structural Errors	±0.5 to 1.5
Antenna Boresighting Errors	±0.25
Antenna Servo Errors	±0.2
RF Reference Null Offset	±0.1
Pointing Program Errors	±0.1 to 0.3

4. Capsule Requirements

This discussion presents various proposed separation and capsule maneuver techniques from an attitude control and autopilot standpoint. Since all the information necessary to make a choice of the most desirable system is not available, two possible systems will be discussed in some detail. A brief discussion of the other stabilization techniques will also be given.

In all cases, it was assumed that the capsule would have no capability of performing any commanded turns, but only of maintaining its initial position. Hence the commanded turn capability of the bus would be required to turn the bus and capsule to some arbitrary orientation before releasing the capsule. This position would be that required by the capsule during the interval from separation to entry, including the propulsion phase.

Since some of the stabilization techniques proposed invloved spinning the capsule or using a flywheel, a study was made to determine the time required for spin-up by electrical means. The constraints assumed were that the maximum allowable spin-up torque would have to be within the bus attitude control capability, there would be a constant power input, and, if a gear train were used, its efficiency would be 75 percent. A linear speed-torque characteristic of the spin-up motor was assumed with the no-load speed being 4000 rpm. Using a power-to-stall torque ratio of 5 w/oz-in, the curve of Figure 4-75 was obtained. The use of the curve is illustrated by the following example. Consider a capsule with a moment of inertia of 100 ft-lb-sec^2 about its spin axis. In order to reach a spin velocity of 7 rad/sec with a power input of 10 w and a gear ratio of 40:1, a spin-up time, t, of 5650 seconds is required.

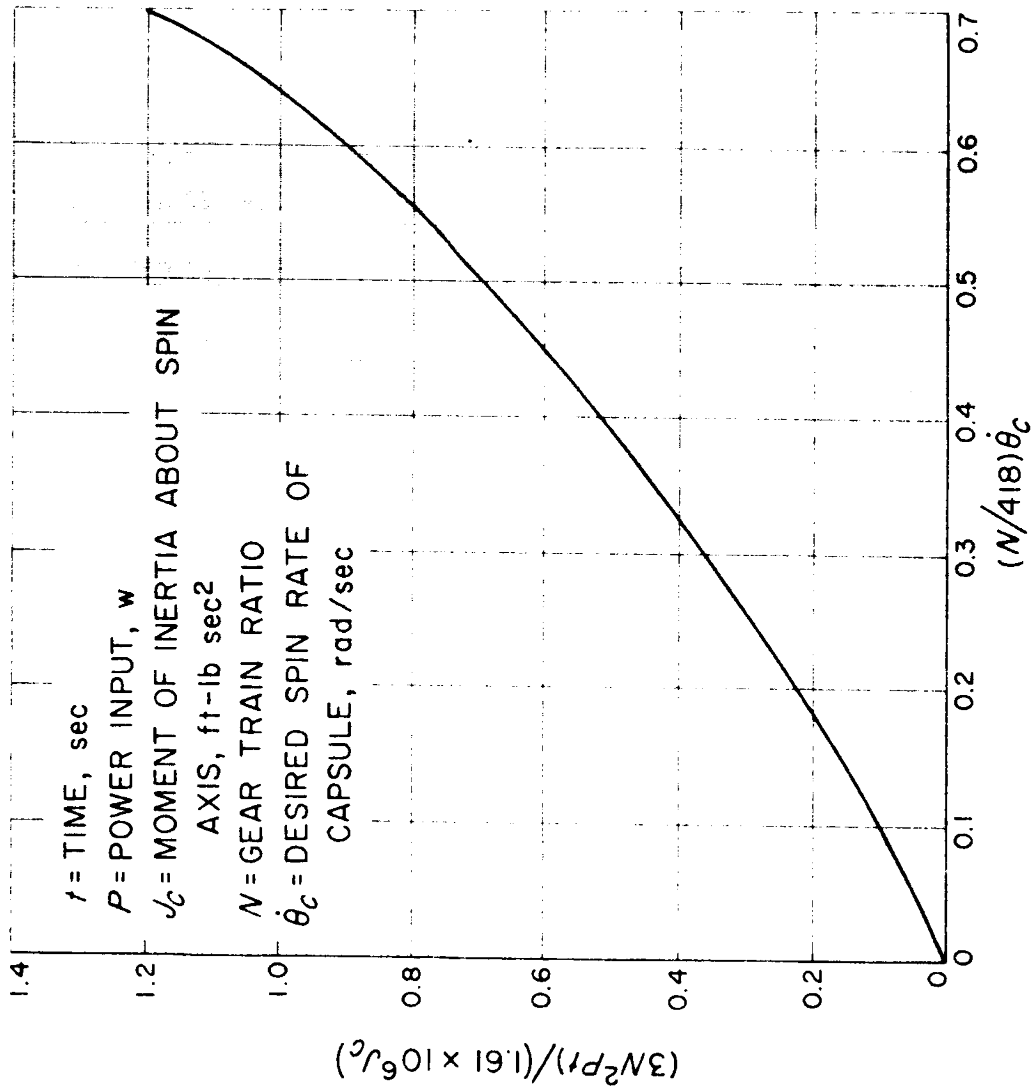


Figure 4-75. Spin-Up Characteristics for Capsule

Preliminary calculations indicate that the use of a flywheel attached to the capsule would give only marginally acceptable performance. Also the time required to electrically spin-up the flywheel while the capsule was attached to the bus would keep the bus away from its primary orientation longer than desirable. Because of this and the fact that more attractive mechanizations were available, no further consideration was given to the use of a flywheel for stabilization.

The use of optical sensors to determine capsule orientation after separation was investigated. In one mechanization, the spacecraft would be the target for a sensor. For the relatively short distances between the capsule and bus, up to and including the time when the capsule maneuver is completed, a two-axis capsule attitude control system using an optical sensor appears feasible. However, since bus-capsule distance increases rapidly after that to 36,000 km (in some cases), and since the capsule must be oriented until just prior to entry for communication purposes, the use of an optical sensor is no longer feasible. The possibility of using either the Sun or the planet as a reference for two-axis control and performing either a Sun-line or a planet-line maneuver is ruled out from trajectory considerations. In another mechanization, optical sensors would use both the Sun and destination planet as targets for control of the capsule about three axes. Placement of the sensors and field of view considerations present configuration problems for such a system. In addition to this, the desired capsule orientation and, hence, sensor location and pointing direction would not be known until after launch. Because of these difficulties, optical sensors have not been considered further.

Two techniques of capsule orientation remain: spinning the entire capsule for stabilization or using an active attitude control system with gyros providing a reference. For spin-stabilization the capsule could be spun up on the bus using either mass expulsion techniques or electrical means. The curve previously discussed can be used to evaluate the feasibility or desirability of using electrical power for spin-up. It has been found that a spinning capsule attached to the spacecraft is compatible with the contemplated bus attitude control system. Four important considerations governing the use of spin stabilization are whether:

1) Suitable timing and de-spin mechanizations can be found to reduce the spin rate of the capsule shortly before entry
2) The capsule antenna "look" angles will be such that the same orientation of the capsule can be used for both the maneuver and the cruise phase of the capsule trajectory
3) The capsule dynamics are such that the capsule will remain in the same orientation with respect to the spin axis until entry
4) Accuracy of pointing directions can be achieved to satisfy the guidance requirements

An active attitude control system using gyros for a reference would provide the most flexibility and the greatest growth potential of any of the systems discussed. Control could be maintained about either two or three axes of the capsule. Mass expulsion techniques would be used for actuation. Possible working fluids for such a system could be either cold gas or the products of combustion of a small solid-fuel charge. The gyros in the control system could also be used by the autopilot during the propulsive maneuver. Thrust vector control could be obtained with jet vanes, although other techniques such as gimballing or secondary injection present alternatives. With an attitude control system as described above, the orientation of the capsule during the maneuver does not necessarily have to coincide with that of the succeeding cruise phase of the capsule trajectory. However, if the orientation were not the same, mechanization complexity would increase. Other advantages of this system are that it does not give rise to any difficult configuration problems and that the zero reference does not have to be set until the gyros are turned on prior to separation.

Since the capsule is aerodynamically stable, no active attitude control will be provided during entry. The parachute to be used for the descent phase will also maintain a suitable capsule orientation during this period. In capsule configurations considered to date, no provision is being made to move the antenna with respect to its supporting structure. Thus, control systems for orienting the capsule antenna after landing have not yet been considered.

A further discussion of capsule separation techniques is presented in Section II-L of this chapter.

F. ELECTRICAL POWER

Design studies on power systems capable of satisfying the requirements of the Voyager spacecraft have been in progress. Nuclear and solar systems with several types of converters are being considered for the orbiter. Nuclear systems appear most attractive for the lander.

1. The Orbiter

a. Nuclear Systems

For this study, the orbiter power system was divided in three radioisotope modules each producing 285 w. This system is also compatible with the spacecraft design configuration. The design is to be taken as typical rather than final. It serves to illustrate the design flexibility provided by such a system.

A typical radioisotope module proposed for an orbiting spacecraft is shown in Figure 4-76. Figure 4-77 shows the location of the modules in the spacecraft. The characteristics of the modules are:

Fuel	Curium 244 as CmO_2
Thermoelectric elements	Germanium-silicon
Safety design	Intact reentry and helium containment
Operating temperatures	Hot junction 1500°F Cold junction 600°F
Number of couples	588 in 3 series strings of 186 couples each
Output voltage	28 v dc

It is estimated that this module will weight 91 lb, thus a total of 273 lb for the power source. If the hot junction temperature is incompatible with the materials used, a drop of 200 to 1300°F will result in an 11 percent weight increase for a total of about 304 lb.

b. Solar Photovoltaic System

A preliminary weight analysis has been performed for orbiter solar photovoltaic power systems. The assumptions, method of analysis, and pertinent results are summarized here. Further discussion of solar photovoltaic systems may be found in Section IV, B, 6, Electrical Power, of EPD-139, Volume II.

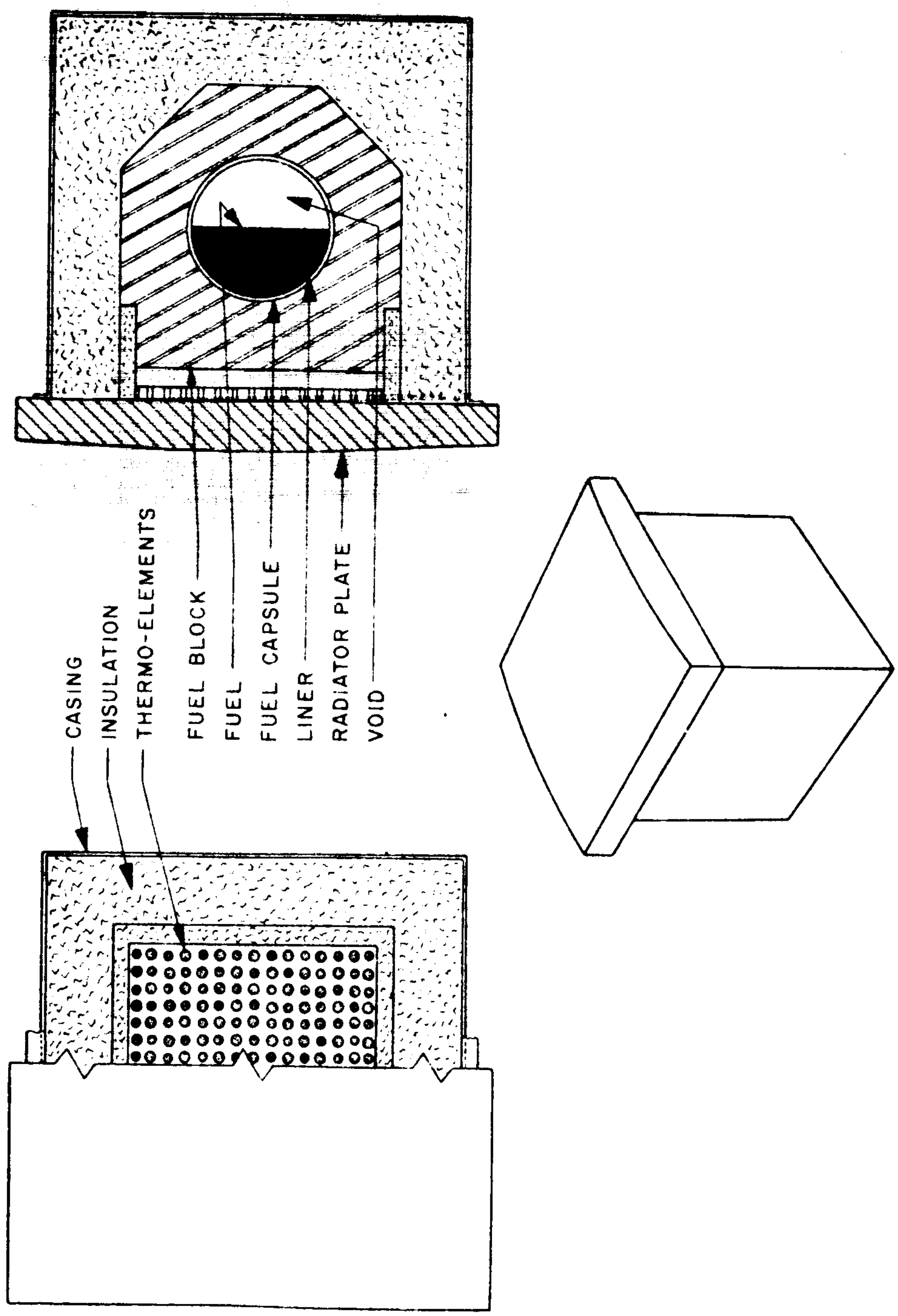


Figure 4-76. Generator Designed for Intact Reentry

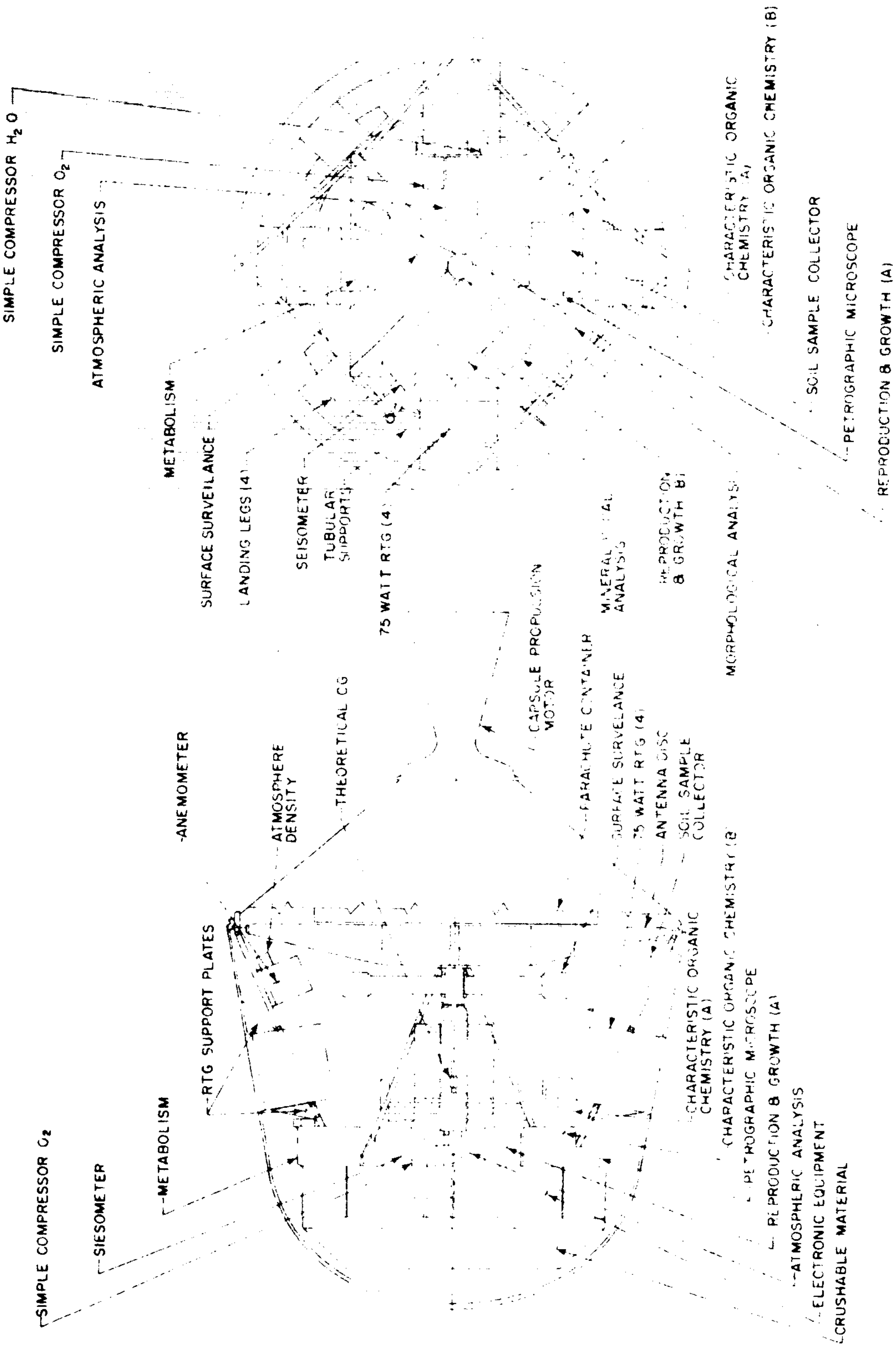


Figure 4-77. Generator--Space Capsule Design

(1) System Configuration and Operating Conditions. The elements of the power system are shown in the block diagram, Figure 4-78. At certain periods during the orbiting part of the missions, the orbiter passes through the planet's shadow during a fraction of each orbit. At such times, the power system is denied solar energy, and power must be drawn from another source. Because the orbiter completes many hundreds of orbits, a rechargeable storage battery is selected as the auxilliary power source. During the sunlit portion of each orbit, solar energy is converted to electrical energy and used for battery recharge. The time variations of power and energy at various points in the systems over the course of an orbit are shown in Figure 4-79.

The loads, P_U, demanded by the users from the system during the shadow period may be different from those demanded during the sunlight for at least two reasons:

1) The scientific experiments performed in the shadow period may differ from those performed when the spacecraft is in the sunlight.
2) Communications and other equipment may be put on a standby basis or turned off during the shadow periods to reduce the demands on the battery, thus lessening the sizes and weights of the solar panels and battery required to provide spacecraft power.

The solar panel output power, P_A, is zero during the shadow periods, because no solar power is being received. During the sunlit periods, P_A exceeds the sum of the user's demands, P_U, and the power required to charge the batteries, P_C. The difference is the power dissipated in the charging, regulating, and power conditioning equipment.

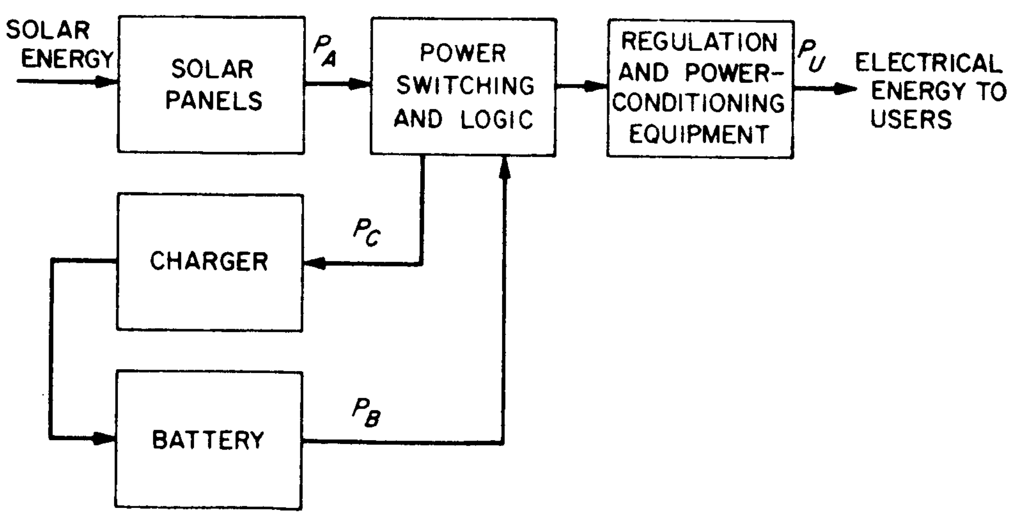


Figure 4-78. Orbiter Solar Power System, Block Diagram

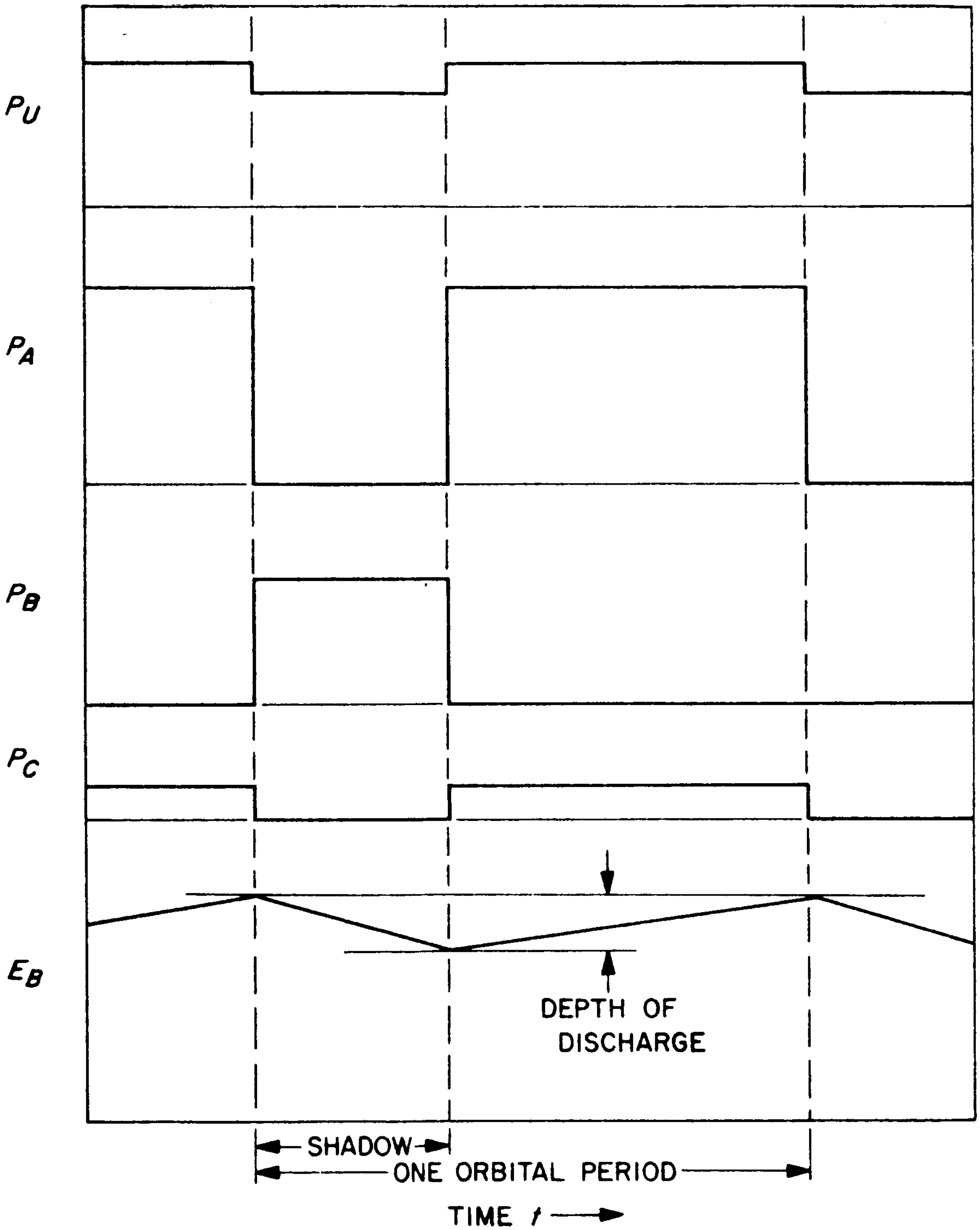


Figure 4-79. Power and Energy Variations with Time

The power drawn from the battery during the shadow periods, P_B, is sufficient to supply the shadow period demands of the users plus the losses in the conditioning and regulation equipment.

The energy available in the battery, E_B, varies from full-change at the beginning of the shadow period to a minimum at the end of the shadow period, and is built back up to maximum again by the charging power during the sunlit periods. It is, of course necessary that the charging power be sufficient to return the available battery energy level to its original value if a large number of cycles are to be covered without progressive depletion of the battery.

(2) Method of Analysis. A simplified analysis was used here to obtain an approximate idea of the weight characteristics of orbiter solar power systems. The computation scheme is illustrated in the block diagram, Figure 4-50, and is described in detail below.

Parameters that affect the solar power system size and weight include:

1) Power demands of users
2) Shadow and sunlit period durations
3) Solar panel performance characteristics
4) Losses in battery, charger, regulation, and conversion equipment
5) Safety factors and contingency allowances
6) Weight characteristics of components

The general scheme of computations is to determine:

1) Total user demand on power system from individual user estimates of power required
2) Power demands on solar panels and battery resulting from the users' demands
3) The energy required from the battery
4) The energy required to recharge the battery
5) The power demand on the solar panels to provide the recharging energy during the sunlit period of each orbit
6) The power available per square foot of solar panels at the orbiter location
7) The area of solar panels required
8) The weights of the power systems components

(a) Computation of user demands. For the present analysis, the power was assumed to be divided among five users, each of which may draw power at one level

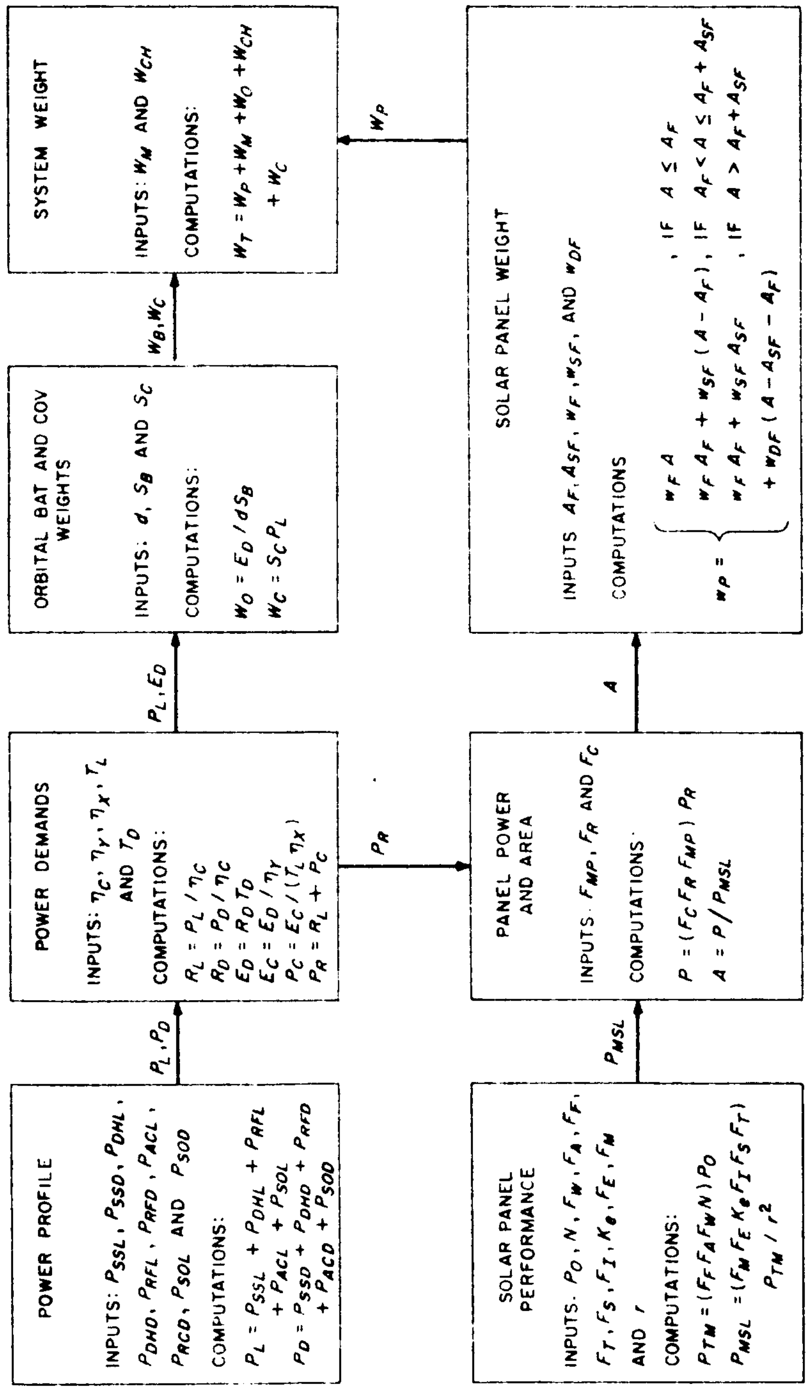


Figure 4-80. Analysis of Weight Characteristics

during the sunlit period and another level during the shadow period. The users and symbols for their demands are shown in Table 4-15, together with the values used for this study.

Three different levels of user demand during the shadow period were considered:

1) "Maximum Demand," where the loads in sunlight and shadow are identical
2) "Intermediate Demand," where the communications equipment and some science and associated data-handling equipment are in a stand-by condition
3) "Minimum Demand," where all science, communication, and data-handling equipment is on standby

The total user demands during sunlit and shadow portions are, respectively, P_L and P_D:

$$P_L = P_{SSL} + P_{DHL} + P_{RFL} + P_{ACL} + P_{SOL} \tag{1}$$

and

$$P_D = P_{SSD} + P_{DHD} + P_{RFD} + P_{ACD} + P_{SOD} \tag{2}$$

(b) Users' Demands on the Solar Panels and Battery. In this analysis, the losses in the regulation and conditioning equipment are described by an efficiency factor η_c, which is assumed to be independent of the load level. The demands during the sunlit and shadow periods, R_L and R_D, respectively, are drawn from the solar panels and battery, respectively, and are given:

Table 4-15. User Demands

User	Sunlight		Shadow			
	Symbol	Demand (·)	Symbol	Demand (·) Maximum	Intermediate	Minimum
Space Science	P_{SSL}	111.	P_{SSD}	111	59	0
Date Handling	P_{DHL}	103	P_{DHD}	103	53	37
Communications	P_{RFL}	157	P_{RFD}	157	102	102
Attitude Control	P_{ACL}	62	P_{ACD}	62	62	62
Sequencing, Operations Control, and Computing (SOCC)	P_{SOL}	20	P_{SOD}	20	20	20

$$R_L = \frac{P_L}{\eta_c} \tag{3}$$

and

$$R_D = \frac{P_D}{\eta_c} \tag{4}$$

For this study, $\eta_c = 0.7$.

(c) Energy Required from Battery. The energy required from the battery, E_D, is the product of R_D and the shadow period duration, T_D, so that

$$E_D = R_D\ T_D. \tag{5}$$

The durations of light and dark times are shown in Table 4-16.

(d) Energy Required to Recharge the Battery. The charging losses of the battery are described by an efficiency factor η_y, which is the product of two ratios,

$$E_C = \frac{E_D}{\eta_y}. \tag{6}$$

For this study, 1969 silver-cadmium battery capability was assumed, with $\eta_y = 0.594$.

Table 4-16. Sunlight and Shadow Durations

Orbit Type	Time	Distance (AU)	Time in Sunlight, T_L (hr)	Time in Shadow, T_D (hr)
Ellipse 1800 km - 10,000 km Altitude	Encounter E	1.388	7.61	0
	E + 40 days	1.414	7.61	0
	E + 60*	1.434	6.69	0.92
	E + 80*	1.457	6.69	0.92
	E + 100	1.482	7.28	0.33
	E + 120	1.506	7.61	0
	E + 140	1.534	7.61	0
Circle 2300 km Altitude	ENC	1.388	3.64	0
	E + 40	1.414	3.64	0
	E + 60	1.434	3.14	0.50
	E + 80	1.457	2.97	0.67
	E + 100	1.482	2.91	0.73
	E + 120	1.508	2.89	0.75
	E + 140	1.534	2.92	0.72

*Maximum time in shadow

(e) **Power Required to Recharge the Battery During the Sunlit Period.** Losses in the charger are described by an efficiency factor η_x. The minimum charger demand on the panels is that which, over the length of the sunlit period T_L, will exactly provide the energy E_C; thus

$$P_C = \frac{E_C}{T_L \eta_x} \tag{7}$$

For this study, T_L is shown in Table 4-16, and $\eta_x = 0.85$.

(f) Power Available Per Unit Area of Solar Panel at Orbiter Location. The estimation of solar-panel output per square foot of panel area involves the possible effects of many factors, identified in Table 4-17. In comparing solar-panel performance estimates, due account must be taken of all of these factors.

JPL maintains a solar-panel testing facility at Table Mountain, California. Standard conditions at Table Mountain are a solar intensity of 0.100 w/cm^2 and a cell temperature of 28°C. The power output per square foot of assembled covered-cell solar panel at Table Mountain, P_{TM}, is estimated from the factors in Table 4-17 by the formula

$$P_{TM} = (F_F F_A F_W N) P_O \quad \text{w/ft}^2 \tag{8}$$

The minimum power output per square foot of solar panel at a given location of the orbiter is estimated as

$$P_{MSL} = \frac{F_M \; F_E \; K_e \; F_I \; F_S \; F_T}{r^2} \quad P_{TM} \tag{9}$$

(g) Required Solar Panel Area. The total demand on the solar panels, P_R, is the sum of the users' demands and the power required for battery recharging, or

$$P_R = R_L + P_C \tag{10}$$

The estimated solar panel power output, P, that the panels should be sized for is somewhat larger than P_R to account for three factors:

1) The design of the regulation equipment may be such that stable operation cannot be guaranteed for demands greater than a certain fraction of the maximum panel output. This limiting fraction is described by the factor F_{mp}. A value of 1.05 is used for this study.

Table 4-17. Solar Panel Performance Parameters and Values

Factor	Symbol	Value[a]	Unit
Power output of individual uncovered cell under standard tungsten light source	P_o	0.0250	w/cell
Number of cells per square foot of panel area	N	400	cells/ft^2
Spectral conversion factor, tungsten light to Table Mountain[b]	F_W	1.00	
Performance loss due to handling, assembly, and cell-to-cell mismatch in assembled panel	F_A	0.97	
Power loss due to reflection and absorption in cover glass, filter, and adhesive	F_F	0.97	
Performance degradation due to flight acceptance test and launch environment	F_T	1.00	
Spectral Conversion factor, Table Mountain to Space	F_S	0.88	
Solar radiation intensity ratio, space at 1.0 AU to Table Mountain	F_I	1.40	
Distance from Sun	r	Table 4-16	AU
Correction to cell conversion efficiency as a function of intensity and cell temperature	K_e	0.955[c]	
Performance loss due to effects of space environment	F_E	1.0	
Possible difference between measured and actual performance due to measurement errors	F_M	0.95	

(a) Current estimate of 1969 performance.

(b) "Table Mountain" refers to JPL's solar cell test facility at Table Mountain, California. These values are included to permit comparisons of measured cell performance.

(c) Average value over orbiter mission lifetime at planet.

2) The solar cells should be arranged in electrically-isolated sections so that failure of one section by short-circuiting does not impose an additional drain on the other sections. For reliability through redundancy, it is desirable to be able to carry the spacecraft load on less than the full complement of panel sections. The degree of redundancy is described by a factor, F_R, which has a value of 1.07 for this study.

3) Past experience has shown that a contingency allowance in estimating solar panel size is prudent for early design studies. The contingency allowance is described by a factor, F_C, which has a value of 1.1 for this study.

The recommended minimum panel capability P is then obtained as

$$P = (F_C F_R F_{mp}) P_R \tag{11}$$

The recommended minimum panel area is then obtained as

$$A = \frac{P}{P_{MSL}} \tag{12}$$

(h) Weights of Power System and Its Components. The weight of the power system is the sum of the weights of four component groups: solar panels, batteries, battery charger, and the regulation and conversion equipment.

1) Solar panel weight. The weight of the solar panel assembly is a function of the method used to articulate the panels. A certain area, A_F, may be permanently fixed to the spacecraft frame without requiring any articulation. This requires the least weight, W_F, per square foot of panel area. An additional number of square feet, A_{SF}, may be articulated with a somewhat heavier single-folding arrangement with a weight, W_{SF} lb/ft^2. Panel area in excess of $(A_F + A_{SF})$ square feet must be articulated with an even heavier double-folding configuration with a weight, W_{DF} lb/ft^2. The weight of the solar cells and connections per square foot remains constant (at 0.25 lb/ft^2 for this study) and is included in the values for W_F = 0.75 lb/ft^2, W_{SF} = 1.35 lb/ft^2, and W_{DF} = 2.45 lb/ft^2. The values of A_F and A_{SF} are 50 ft^2 and 170 ft^2, respectively.

The panel weight W_P is computed according to the formula

$$W_P = \begin{cases} W_F\, A & ;\ A \le A_F \\ W_F\, A_F + W_{SF}\,(A - A_F) & ;\ A_F < A \le (A_F + A_{SF}) \\ W_F\, A_F + W_{SF}\, A_{SF} + W_{DF}\,(A - A_{SF} - A_F) & ;\ A > (A_F + A_{SF}) \end{cases} \tag{13}$$

2) Battery weight. It is expected that two batteries will be provided: (1) a higher-capacity "maneuver" battery capable of a small number of charge-discharge cycles, to carry the loads during maneuvers; and (2) a lower capacity "orbiter" battery capable of many charge discharge cycles to carry the loads during shadow periods.

The weight of the maneuver battery, W_M, is determined by factors external to the present study, and has been estimated at 45 lb for the present case. The weight of the orbiter battery is determined by the energy requirement, E_D. Complete discharge of the battery is not desirable, hence the battery should be large enough that only some fraction d of the total capacity is ever discharged. The battery weight is here assumed to be proportional to the battery capacity as described by the specific energy, S_B w-hr/lb. Hence the weight of the orbiter battery, W_O, is given by

$$W_O = \frac{E_D}{d\,S_B} \tag{14}$$

where

$d = 0.75$

and

$S_B = 30$ w-hr/lb

for this study.

3) Charger weight. The weight of the battery charger, W_{CH}, is estimated to be 2 lb.
4) Conversion and regulation equipment weight. The weight of the conversion equipment, W_C, is assumed to be proportional to the maximum user demands are described by the specific converter weight, S_C, which has a value of 0.07 lb/watt for this study. Hence

$$W_C = S_C P_L \tag{15}$$

because $P_L > P_D$ for this study.

5) Total system weight. The total system weight, W_T, is thus

$$W_T = W_P + W_M + W_O + W_{CH} + W_C \tag{16}$$

(3) Results. The power system weight requirements were investigated at encounter, and at 20-day intervals following encounter up to E + 140 days, inclusive, for two orbits: a circular polar orbit of 2300 km altitude above the surface, and an 1800 km by 10,000 km

elliptical orbit inclined 70 deg to the Mars equator. The results are listed in Table 4-18 and illustrated in Figures 4-81 and 4-82. (The distinctions between maximum, intermediate, and minimum demand affect the sizing only when the spacecraft passes through the shadow of the planet.)

2. The Lander

a. Introduction

The power system for a Mars landing capsule must satisfy the following general constraints:

1) Be capable of operation from the time it is separated from the bus until the end of mission, including such phases as cruise, entry, landing and post landing.
2) Fit within the space allocated within the capsule.
3) Have a minimum effect on other spacecraft components, subsystems and experiments.

Table 4-18. Orbiter Solar Photovoltaic Power System Weight Characteristics

	Time (days)	Maximum Demand				Intermediate Demand				Minimum Demand			
		W_T (lb)	W_P (lb)	W_O (lb)	A (ft^2)	W_T (lb)	W_P (lb)	W_O (lb)	A (ft^2)	W_T (lb)	W_P (lb)	W_O (lb)	A (ft^2)
Circular Polar	E	247	168	0	147								
	E + 20												
	E + 40	254	175	0	152								
	E + 60	341	248	14	206	313	225	9	189	299	214	7	180
	E + 80	398	300	19	234	343	252	12	209	324	235	9	197
	E + 100	440	340	21	250	362	270	14	221	339	250	10	207
	E + 120	470	369	22	262	387	294	14	231	351	262	11	216
	E + 140	480	381	21	266	399	306	14	236	360	271	10	222
Elliptical - 70 deg	E	247	166	0	147								
	E + 20												
	E + 40	254	175	0	152								
	E + 60	344	239	26	199	315	219	17	184	301	209	13	177
	E + 80	352	247	26	205	323	227	17	190	309	[illegible]	13	181
	E + 100	304	216	10	182	294	209	6	177	299	205	5	174
	E + 120	282	203	0	173								
	E + 140	293	214	0	181								

Definitions

W_T Total System Weight
W_P Solar Panel Weight
W_O Orbiter Battery Weight
A Solar Panel Area

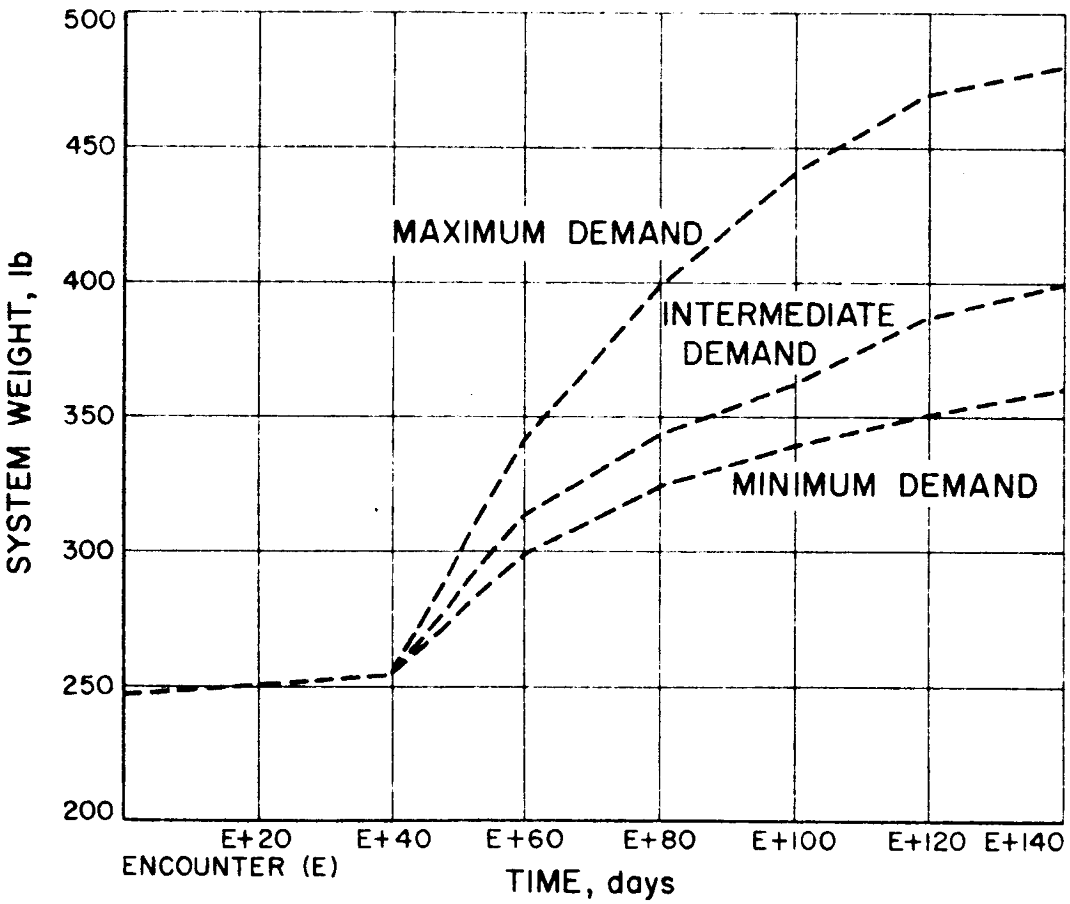


Figure 4-81. System Weight for 2300 km Altitude Polar Circular Orbit

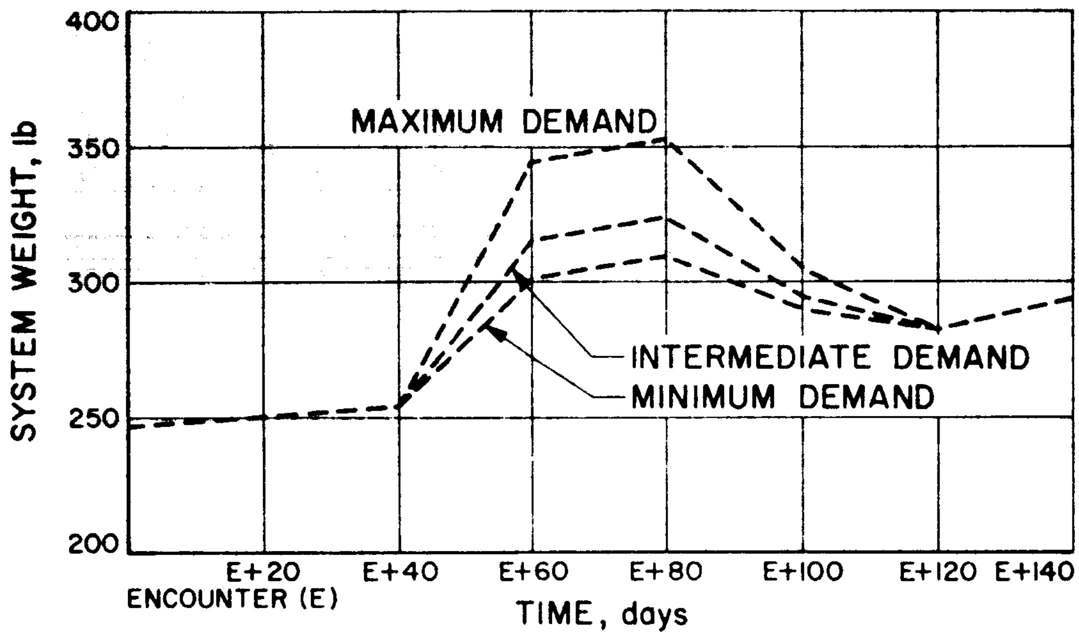


Figure 4-82. System Weight for 1800 km by 10, 000 km Altitude, 70 deg Inclination Elliptic Orbit

The first constraint means that the power system has to withstand the environment imposed on the capsule. This environment will consist of: high vacuum during cruise, acceleration due to capsule maneuver prior to entry phase, shock loadings due to ballistic decelerations and parachute opening during entry, and also due to landing, entry heating, i. e., high temperature for short periods of time, possible chemical attack from the constituents of the Martian atmosphere and chemicals on the Martian surface after landing.

The second constraint will be a function of the power system selected, since the space allocation will depend on the power system adaptability, i. e., whether it can be in the inside of the capsule or whether it must have radiators located on the outside. This requirement may also work in reverse, i. e., affect the design of the landing capsule itself.

The third constraint is also a function of the power system selected and will be discussed further in this section under Interface Problems.

b. Lander Power System Considerations

In the selection of a power system, it is necessary to evaluate many factors. The proper balance of these factors is obtained when the power system incorporated into the spacecraft results in a spacecraft of minimum weight and maximum reliable operational capability. Since the power system must be capable of being stored for long periods of time, chemical systems requiring storage tanks, such as fuel cells, are not desirable. With the best of available insulation and cryogenic storage, the weight of such systems is very large, even for modest power requirements.

A solar system deployable after landing is extremely cumbersome, and its expected reliability and performance unpredictable due to unknowns about the surface environment. The unknowns include such factors as landing terrain, solar constant, dust storms and other local surface phenomena. Solar mirrors or solar cells located on the skin or outside of a landing capsule would either burn up or be permanently damaged due to entry heating. It therefore appears necessary to rely on either a primary battery or a radioisotope-chemical battery power system. The final selection between these two types of power sources will depend mainly on:

1) Whether the communications system will transmit directly to Earth and/or use the orbiter as a relay link.
2) Whether the communication system will operate continuously or be cycled.
3) The expected life of the system after landing.

Communications and experiments are the largest power users and will, to a large extent, determine the power level of the system; experiment duration will determine the systems life requirement.

c. Power System Characteristics

(1) Primary Chemical Battery. Two types of primary batteries that may be considered are:

1) Those that are manually activated, i.e., the battery is active at the time of installation into the spacecraft.
2) Those that are automatically activated just prior to use.

Silver-zinc primary batteries have had the most extensive use in space power systems. Their size and weight will depend on such factors as length of operations, maximum sustained power level, duty cycle, storage time and storage, and operating temperatures. Capacity loss with storage time is a very strong function of storage temperature. Present test batteries will show losses for 210 days storage as shown in Figure 4-83. For manually activated batteries, the capacity loss at 50°F is expected to be less than 5 percent and at 120°F about 45 percent, providing the improvements predicted for 1968 are achieved. The automatically activated batteries have a negligible loss at 50°F and at 120°F the loss goes up to 10 percent.

Another difference in these two types of batteries, pertinent to our particular application, is their relative expected reliabilities. The manually activated battery, being active on installation into the spacecraft, can be checked-out prior to launch; its operating reliability would be a function of the reliability of the temperature control system. The automatically activated battery has to rely on a squib that will open a diaphragm or valve that allows the electrolyte to flow into the battery case. The assessed reliability of the temperature control system, in its capability of maintaining manually activated batteries between 0°F and 50°F during storage (transit time), must be compared to the assessed reliability of the activation mechanism of the automatically activated battery, for a proper evaluation of the system.

(2) Radioisotope-Chemical Battery. A radioisotope power generator would be either a thermoelectric or a thermionic converter operating with heat supplied by a long half life, alpha emitting, isotope. Such generators were described in EPD-139, Volume II. The generator would be sized to power the full lander and recharge a battery during its normal operation. The battery would be sized to supply peak loads when the power requirements exceeds the capacity of the radioisotope generator. In case of a failure of the radioisotope generator, the battery would be able to handle the full lander load for a period of a few hours.

The radioisotope generator must be located near the skin of the capsule in order to be able to radiate the waste heat as necessary to maintain the desired temperature gradient across the converter. If a single module is used, it would have to be located on the center line of the capsule. In such a location, unless it is completely at the rear of the capsule,

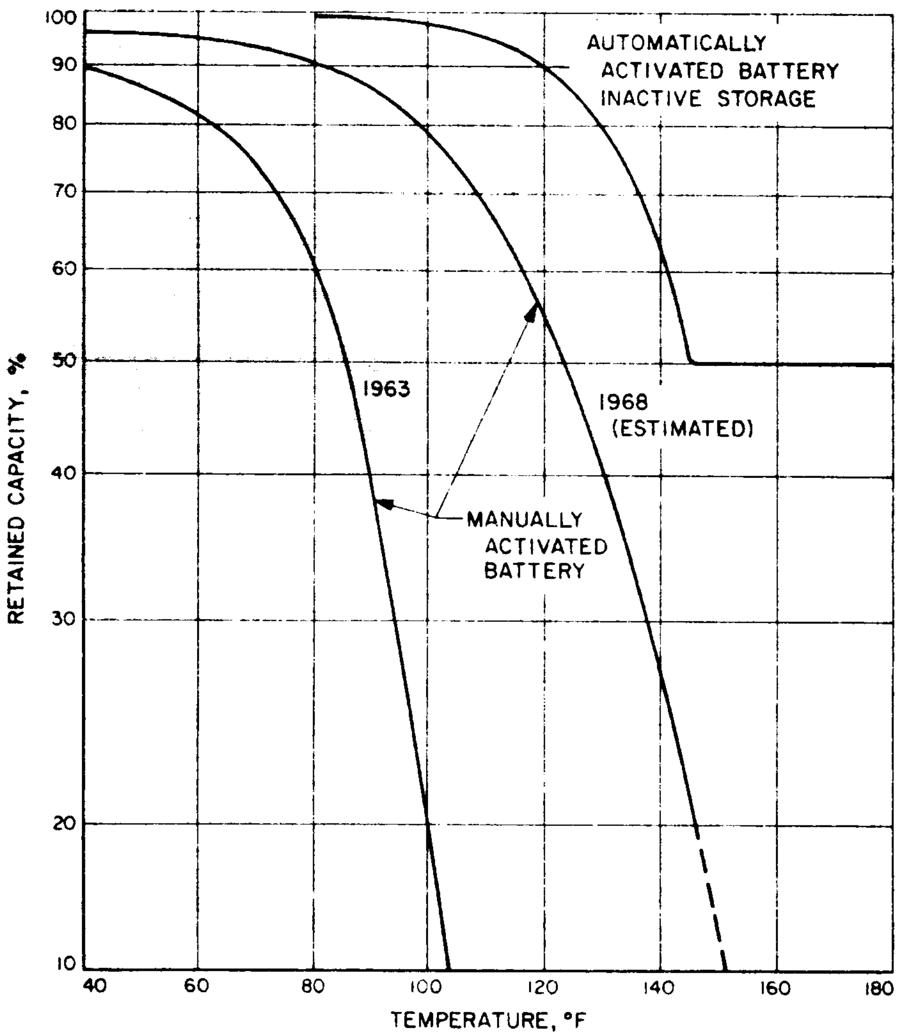


Figure 4-83. Retained Capacity of a Silver-Zinc Battery after 210 Days as a Function of Storage Temperature

where it would interfere with the propulsion system and antenna, it would have difficulty in disposing of its waste heat. It is therefore advisable to use a power generator consisting of multiple modules. Because of capsule entry-stability considerations it is desirable to have a minimum of three units. This breakdown in units of low power level is obtained at a penalty in specific power output. This penalty is illustrated in Figure 4-84, which shows that the specific power output (w/lb) of a module decreases as the power level of a module goes down.

The radioisotope generator considered is a plutonium 238 fueled, thermoelectric generator. The thermoelectric elements are germanium silicon operating at a hot junction temperature of over 1300°F and a cold junction temperature of over 500°F. The thermoelectric elements must be attached to the radiator in order to produce the desired temperature difference for their operation; they are therefore located only on one side of the fuel block with the other sides being covered with sufficient thermal insulation to minimize heat losses. This is done at a penalty in thermal efficiency and in specific power output as shown in Figure 4-84, which accounts for this penalty and represents designs compatible with the capsule configuration. A schematic of a typical module is shown in Figure 4-76. This module shows a fuel capsule surrounded by ablative material sufficient to protect it from burnup in case it enters any atmosphere separated from the mother vehicle.

(3) Safety Requirements for Radioisotopes. Five safety areas affecting the design of a radioisotope power generator have been outlined by the Atomic Energy Commission. These are:

1) Fuel containment against fire and explosion on the launch pad.
2) Fuel containment on high velocity impact due to flight abort prior to entering in orbit.
3) Fuel container to be corrosion resistant, mainly to sea water, if impact occurs in the sea. Release of the radioisotope should not occur before at least 10 half-lives. Certain chemically active forms of the radioisotope may require longer containment and slow release after that period of time.
4) Fuel burn-up in case of orbital reentry at altitudes above 100, 000 ft, and dispersed particle size not to exceed 1 micron. An alternative to this provision would be intact reentry of fuel container.
5) Maintain safe radiation levels during all operations close to personnel.

d. Selection Criteria

A preliminary screening so far has resulted in the consideration of two types of power systems:

1) A radioisotope thermoelectric generator with a rechargeable chemical battery/
2) A primary chemical battery.

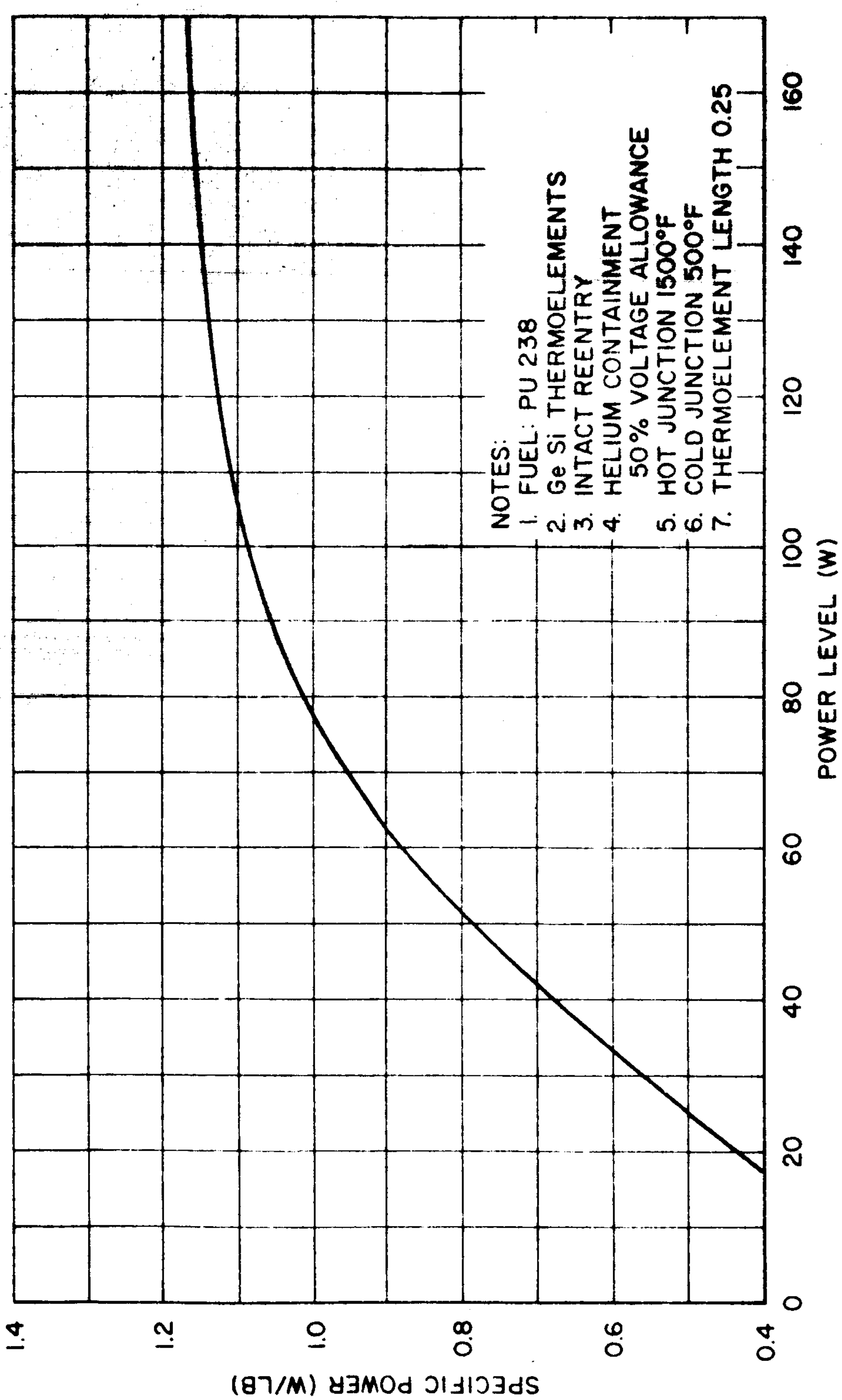


Figure 4-84. Specific Power (W/LB) vs Power Level for Radioisotope Power Generator Modules

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Figure 4-85.

A selection between these two types of systems, in all likelihood, will be made when the mission to be performed in firmly defined. Certain factors have a strong effect on the selection. The factors to be considered are:

1) Reliability
2) Availability
3) Interface problems

There is no doubt that if long system life is of primary importance, the system based on primary batteries will be eliminated, and the choice will fall on a radioisotope system. If other factors have equal weight (e.g., the effect of nuclear radiation on experiments, or undue complications in the temperature control system), the choice is not so obvious.

Primary consideration is given to the reliability factor because of the expense involved in launching the spacecraft considered here. A high degree of confidence must be placed on all the systems. The power system, however, must operate if anything is expected of the spacecraft. Batteries have had extensive use in spacecrafts of many types, and their reliability is well known. Although some advances in the state of the art on batteries are expected to achieve the weight of systems quoted above, the reliability of these batteries is not expected to be appreciably different. When radioisotopes are considered, not as much data on past performance is available; the systems so far under operation are much smaller in both size and power level. The radioisotope considered for the power system of the capsule, plutonium 238, has been used in two 2.7 w units flown on Transit IV-A and IV-B satellites and in several generators (SNAP 9A) designed for powering Transit V satellites. The thermoelectric material considered, GeSi, has been operated extensively at much lower temperatures in connection with the SNAP 10A program. Since only a limited amount of information is available at the expected operating temperatures, it is difficult to assess the expected system reliability.

A test module to operate at hot junction temperature above 1250°F will be placed under test at JPL during the second half of 1963. Additional data on germanium silicon thermo-elements is expected from RCA and other users within the next few years. This information will help in obtaining realistic performance and operating life data, so that reliability assessment can be more meaningful. From presently known properties of germanium silicon materials, it is expected that some shortcomings of the formerly used lead telluride thermoelements will be overcome.

One feature common to both manually activated batteries and radioisotope systems which is of great help in establishing system reliability, is their capability of being performance tested prior to flight at the launch pad.

System availability is another critical factor. This factor affects the battery system only in its expected performance. Battery systems can be made available today to perform the mission required. With radioisotope systems availability may be a very serious problem. Amounts of plutonium 238 presently under production and those planned for the next few years, could not be made available for a Voyager capsule, because they are committed to other programs. Unless arrangements are made early to affect production schedules, plutonium 238 may not be available for a 1969 Mars landing capsule.

Interface problems (the third factor listed) will require a more extensive discussion. This will be discussed in the following section. The power system affects the capsule design because of its thermal and structural characteristics. The battery requires a special temperature control to limit losses due to long term storage in transit. A radioisotope unit, because of the amount of waste heat generated which must be disposed of, also requires special consideration. In addition, there is the radiation that escapes from the fuel capsule, in a radioisotope system, consisting mainly of gamma photons and neutrons, which affects the operation of electronic systems and may also have an adverse effect on science experiments.

e. Interface Problems

Both batteries and radioisotope generators make their presence in the capsule felt, each producing problems due to its own characteristics. Batteries are fairly adaptable as to location, but demand for their operation a control of temperature, on the low side, to avoid capacity losses. Radioisotope units are heat generators and must be located close to the outer surface of the capsule so they can radiate their waste heat without affecting other capsule components. Each of these characteristics and others that effect the capsule design will be discussed below.

(1) Thermal Interface

(a) Batteries. A battery, in order to retain its power capacity for the long interplanetary cruise time, must be stored at a relatively low temperature. Although this control is not extremely critical, the storage temperature has a very strong influence on the capacity retention ability of the battery, as illustrated in Figure 4-83. A complete thermal analysis of the capsule is required to assess the magnitude of this problem.

(b) Radioisotope Generators. The thermal problem with these units is rather serious. They must be located near the outer surface of the capsule in order to radiate their waste heat. This, however, introduces the problem of protecting the radiator from burning up during planetary entry. The capsule has a heat shield around it and the generator's radiator must be integrated with it. The location is critical because it should not be located too far forward in the shield on account of the heating and yet the generators should not be located too far back in order to retain the capsule c.g. forward of its center of pressure at reentry. A conceptual design of a four-module generator, integrated into a practical capsule, is shown in Figure 4-86.

(2) Nuclear Radiation. This problem is peculiar to radioisotope generators and does not exist with batteries. The radioisotope planned for use with the capsule power system is plutonium 238, one of the least offensive; however, in addition to alpha particles which are completely stopped within the fuel container, it produces low energy gammas and neutrons generated by a small amount of spontaneous fission which occurs. For the design shown in Figure 4-77, using four 75w units, the radiation limits expected at the locations of the various experiments on-board is shown in Table 4-19. The total accumulated doses shown represent 10,000 hrs exposure.

(3) System Lifetime. A primary functional requirement is the desired life of the power system, since it determines the life expectancy of the lander system. In the case of a battery-based power system large weight penalties must be paid for each additional day of operation, while the radioisotope power system weight is relatively insensitive to operating life. Because of the choice of long half-life isotopes (e.g., plutonium 238 with a half-life of 89 years), there will be a negligible change in power output after 1 year of operation. The radioisotope power system is designed to produce maximum power immediately after landing; since the power requirements drop off with time, this radioisotope energy decay will be in harmony with the power profile.

(4) Orientation and Day-Night Cycling. By eliminating a solar powered system from further consideration, the requirement for orientation is removed. The day-night cycling will affect the temperature control system for the battery system, since it is advisable to avoid large temperature shifts on the battery even while it is operating. In the case of a radioisotope unit, the effect of the Sun is negligible because of the large amount of heat generated by the system itself.

i. System Performance Evaluation

(1) Desirable environment properties data. The more that is known about the environment in which a system will have to operate, the easier it is to design the system. For batteries, being located in the interior of the capsule, the outside environment may have a negligible effect on their performance. The environment will affect the operation and design of the environment control system, but the effect on the batteries will be indirect. A radioisotope power system, located with at least one surface exposed to the outside, is more dependent on the environment for its operation. The environment characteristics that have an effect on system performance are those that affect the heat transfer. Prevailing

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Figure 4-86.

Table 4-19. Unshielded Radiation Levels from Four - 75 Watt Radioisotope Thermoelectric Generator Units

Capsule Science Instruments	Total Neutron Dose[a]			Total Gamma Dose[a]	
	Rem/hr	Rem	Total Neut/cm^2 after 10^4 hrs	Rem/hr	Total Roentgens after 10,000 hrs
Metabolism	4.48×10^{-2}	4.48×10^{2}	1.15×10^{10}	2.23×10^{-2}	2.23×10^{2}
Reproduction & Growth (A)	3.21×10^{-2}	3.21×10^{2}	0.83×10^{10}	1.62×10^{-2}	1.62×10^{2}
Characteristic Organic Chemistry (A)	3.29×10^{-2}	3.29×10^{2}	0.84×10^{10}	1.65×10^{-2}	1.65×10^{2}
Atmospheric Analysis	4.10×10^{-2}	4.10×10^{2}	1.05×10^{10}	2.10×10^{-2}	2.10×10^{2}
Simple Composition – O_2	3.31×10^{-2}	3.31×10^{2}	0.85×10^{10}	1.66×10^{-2}	1.66×10^{2}
Mineralogical Analysis	4.55×10^{-2}	4.55×10^{2}	1.16×10^{10}	2.27×10^{-2}	2.27×10^{2}
Petrographic Microscope	4.13×10^{-2}	4.13×10^{2}	1.06×10^{10}	2.06×10^{-2}	2.06×10^{2}
Surface Surveilance	3.94×10^{-2}	3.94×10^{2}	1.01×10^{10}	1.97×10^{-2}	1.97×10^{2}
Seismometer	4.36×10^{-2}	4.36×10^{2}	1.12×10^{10}	2.19×10^{-2}	2.19×10^{2}
Reproduction & Growth (B)	3.83×10^{-2}	3.82×10^{2}	0.98×10^{10}	1.92×10^{-2}	1.92×10^{2}

Note:

[a] 10,000 hr exposure

temperature, atmosphere composition, density (pressure) and mobility (winds) are the main characteristics that affect the transfer of heat. A conservative design results by assuming that all waste heat will be disposed of by radiation. The presence of an atmosphere will tend to alleviate the heat rejection problems by adding convection and/or conduction to the radiation mechanism after landing.

Data on the planet's atmospheric and surface chemical composition would help in determining if the materials of which the generators are built are susceptible to chemical attack.

(2) System Performance Monitoring. To evaluate the performance of the system, and possibly improve its design for future spacecraft, it is indispensable to have a means of obtaining data on the following parameters:

1) Power output
   a) Voltage
   b) Current
2) Load variations
3) System temperatures at various locations

Continuous monitoring of these parameters would be ideal; frequent periodical checks are acceptable, since it is desired to observe performance as a function of time. Temperatures on batteries must be known for the outside of the battery case and the inside (electrodes and electrolyte). Temperatures on the radioisotope system should be obtained for the hot junctions, cold junctions, fuel and radiator.

Power output should be measured as it comes out of the generators (battery or radioisotope) and also after the regulator.

g. Power System Options

Presented herein are the power profiles and power system weights for several alternative approaches as described below. The variables considered are:

1) Radiated power for the direct and relay transmitters.

2) Type of power system.
3) Operating options for the transmitters.

There are also three phases of operation, namely:

1) Cruise, which covers the flight time between separation from the bus and landing.
2) First 24 hrs operation after landing.
3) Every subsequent 24 hr period.

The figures on power requirements are shown in raw watts and can be used directly for sizing a generator. For most cases an efficiency of 70 percent was used for the conversion of raw power into regulated power for the user. Two exceptions exist and they are as follows:

1) For the transmitters a conversion of 80 percent was used.
2) For data handling, the figure used for efficiency is 60 percent because of the lower efficiency, higher precision of regulation required for this equipment.

The data is presented in tabular and graphical form. The symbols used are explained below:

Option 1a-All equipment is ON continuously.

Option 1b-All equipment except the S-band direct transmitter is ON continuously. The cycle for the S-band direct transmitter is: ON 2 hrs after separation, OFF 56 hr during cruise, ON 2 hrs prior to landing, and ON 6 hrs of every 24 hr period after landing.

Option 2-Both the S-band direct transmitter and the VHF relay transmitter are cycled and the rest of the equipment is ON continuously. The S-band direct transmitter is cycled as in Option 1b and the VHF relay transmitter is ON while in view of the orbiter. The time varies on a day-to-day basis and it may change from zero to over 5 hrs. An average of 2.5 hr operation per day was assumed for sizing the power system.

Case A-In this case, the capsule is assumed to have a 50 w S-band direct transmitter and a 25 w VHF relay transmitter.

Case B - In this case, both the S-band and the VHF transmitters radiate 25 w.

Case C - In this case, both the S-band and the VHF transmitters radiate 50 w.

Case D - In this case, the S-band transmitter radiates 25 w and the VHF transmitter 50 w.

Phase I refers to cruise time (60 hr).

Ia refers to all equipment ON-time during cruise (4 hr)

Ib refers to the time when either one or both transmitters are OFF depending on whether we are in Option 1b or Option 2. The total time assumed is 56 hr.

Phase II refers to operation during the first 24 hr after landing

IIa refers to the time when all the equipment is ON (6 hr for Option 1b and 2.5 hr for Option 2).

IIb refers to the OFF-time for the S-band transmitter in Option 1b (18 hr) and the ON-time of the S-band transmitter only (OFF for the VHF transmitter) in Option 2 (3.5 hr)

IIc refers to the time in Option 2 when both transmitters are OFF (18 hr).

Phase III refers to operation during each 24 hr period starting 24 hr after landing.

IIIa refers to the time when all the equipment is ON (as in Phase IIa).

IIIb is identical to IIb for all days after landing except the first one.

IIIc refers to the time when both transmitters are OFF (as in IIc).

For computing the weights of radioisotope power units the curve presented in Figure 4-84 was used. The battery assumed for the radioisotope was a rechargeable silver-zinc, since very few cycles are required. The weights presented include the charger. The converter weight was held constant at 20 lb for all cases since the regulated power output is slightly below 300 w for most cases, options and phases; converter weight was computed on the basis of 7 lb/100 w regulated power output. To compute the installed energy required for a battery, the loss due to 210 day storage had to be accounted for. After computing the total energy required, it was multiplied by the following factors:

1) For a manually activated battery stored at 50°F; 1.101.
2) For a manually activated battery stored at 120°F; 1.913.
3) For an automatically activated battery stored at 50°F; 1.057.
4) For an automatically activated battery stored at 120°F; 1.162.

The storage capacity of batteries was projected to the expected state-of-the-art for silver-zinc primary batteries for 1969, and the values used were 102 w-hr/lb and 10.2 w-hr/in^3.

The power requirements for all options and all cases are presented in Tables 4-20, 4-21 and 4-22 and illustrated in the power profiles presented in Figures 4-87 through 4-98.

For radioisotope power systems, the battery is designed to take care of the power loads in excess of the radioisotope generator's capacity and in case of a failure of the radioisotope generator the battery must be able to handle the full load for a few hours The total power requirements are computed from the power profiles presented in Figures 4-87 through 4-98.

Table 4-23 below presents the weight of radioisotope generators in 3 or 4 modules, the weight of the batteries including charger and the total weight of the system including converter.

Table 4-24 presents the weights and volumes of primary batteries, divided in 3 phases as discussed above. Two storage temperatures were assumed for both manually activated and automatically activated batteries. This illustrates some of the trade-offs in reliability of the battery vs temperature control accuracy requirements. The power requirements and power systems shown above assume a spin-stabilized capsule. If an active attitude control system is required, the power drain during Phase I (cruise) will increase. This increase is not so severe as to affect substantially the weight of the radioisotope based power system. At worst, in some cases, a 5 lb increase in battery weight may be required. The power system based on a primary battery will, however, be affected to a much larger extent. A silver-zinc battery, manually activated, stored at 50°F and added to the battery already existing, will represent a weight increase of about 20 lbs for all options and cases The power required is estimated at 16,304 watt-hours distributed as shown in Table 4-25.

(1) Conclusion. If long time operation is desirable, a radioisotope power system is required for a Mars landing capsule. This can be illustrated by the operation under Option 2, Case B with two 25 w transmitters (both cycled); the radioisotope power system for that option can be realistically estimated at 356 lb, the primary manually activated battery system stored at 50°F operating for 3 days after landing will weigh 260 lb with 42 lb of battery required for each additional day In the case of Option 1a with two 50 w transmitters, the radioisotope-based power system weighs 600 lb and the primary manually activated battery system stored at 50°F will weigh 514 lb operating for two days after landing. The

radioisotope system presented herein are not optimized as to size and number of radioisotope units and batteries; the figures, however, are not expected to be much different unless some relaxation as to burn-up protection on the fuel capsule can be made. In such a case the weight advantage will shift much further in favor of the radioisotope power system. The weight of the batteries of 102 w-hr/lb, while optimistic, could be expected for 1968.

Table 4-20. Raw Power Requirements for Option 1a

Equipment	Case A	Case B	Case C	Case D
"S" Band Direct Transmitter	133	66 5	133	66 5
VHF Relay Transmitter	60	60	131	131
Data Handling Direct	28	25	28	25
Data Handling Relay	62	62	74 5	74 5
SOCC	10	10	10	10
Science Experiments I	19	19	19	19
Science Experiments II	129	129	129	129
Science Experiments III	31 5	31 5	31 5	31 5
Total for Phase: I	312	242 5	395 5	326
II	422	352 5	505 5	436
III	324 5	255	408	338 5

Table 4-21. Raw Power Requirements for Option 1b

Equipment		Case A	Case B	Case C	Case D
"S" Band Direct Transmitter	on	133	66.5	133	66.5
	off	6.5	1.5	6.5	1.5
VHF Relay Transmitter		60	60	131	131
Data Handling Direct		28	25	28	25
Data Handling Relay		62	62	74.5	74.5
SOCC		10	10	10	10
Science Experiments I		19	19	19	19
Science Experiments II		129	129	129	129
Science Experiments III		31.5	31.5	31.5	31.5
Total for Phase:	Ia	312	242.5	395.5	326
	Ib	185.5	177.5	269	261
	IIa	422	352.5	505.5	436
	IIb	295.5	287.5	379	371
	IIIa	324.5	255	408	338.5
	IIIb	198	190	281.5	273.5

Table 4-22. Raw Power Requirements for Option 2

Equipment		Case A		Case B		Case C		Case D	
"S" Band Direct Transmitter	on		133		66.5		133		66.5
	off	6.5		1.5		6.5		1.5	
VHF Relay Transmitter	on		68		68		131		131
	off	10		10		14.5		14.5	
Data Handling Direct			28		25		28		25
Data Handling Relay			62		62		74.5		74.5
SOCC			10		10		10		10
Science Experiments I			19		19		19		19
Science Experiments II			129		129		129		129
Science Experiments III			31.5		31.5		31.5		31.5
	Ia		320		250.5		395.5		326
	Ib		135.5		127.5		152.5		144.5
Total for	IIa		430		360.5		505.5		436
Phase:	IIb		372		302.5		389		319.5
	IIc		245.5		237.5		262.5		254.5
	IIIa		332.5		263		408		338.5
	IIIb		274.5		205		291.5		222
	IIIc		148		140		165		157

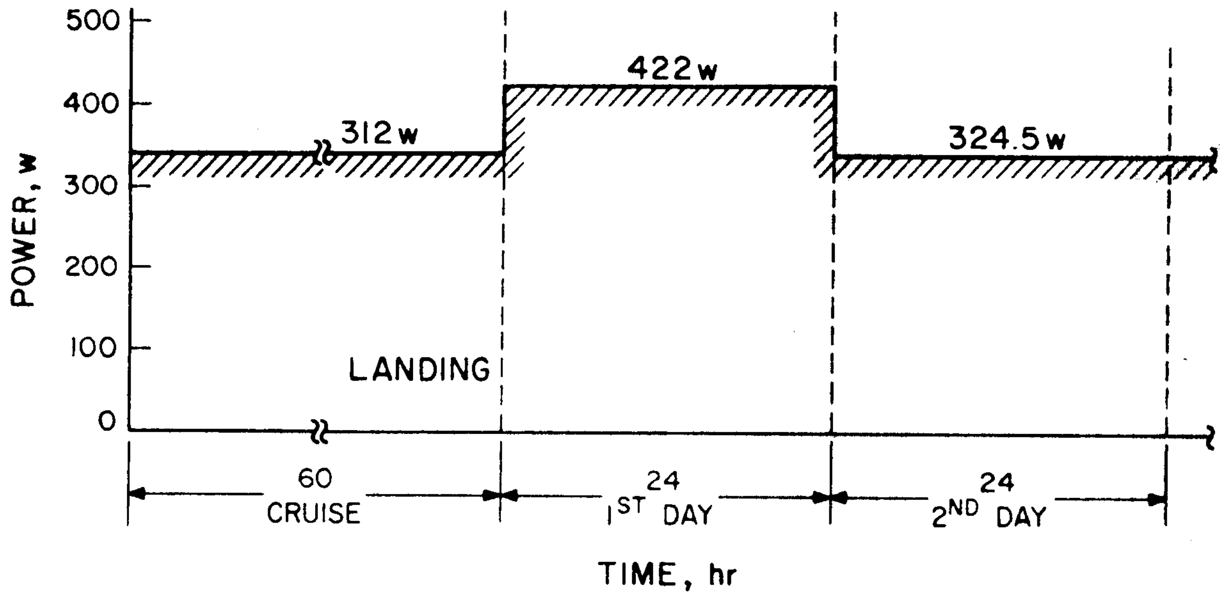

Figure 4-87. Power Profile for Option Ia, Case A

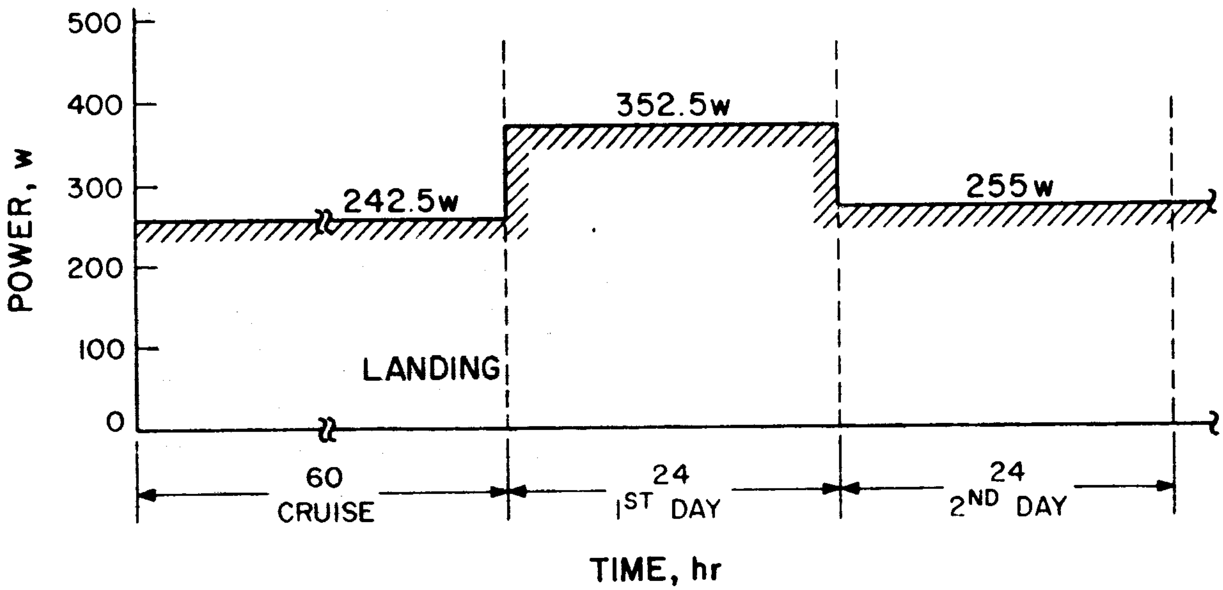

Figure 4-88. Power Profile for Option 1a, Case B

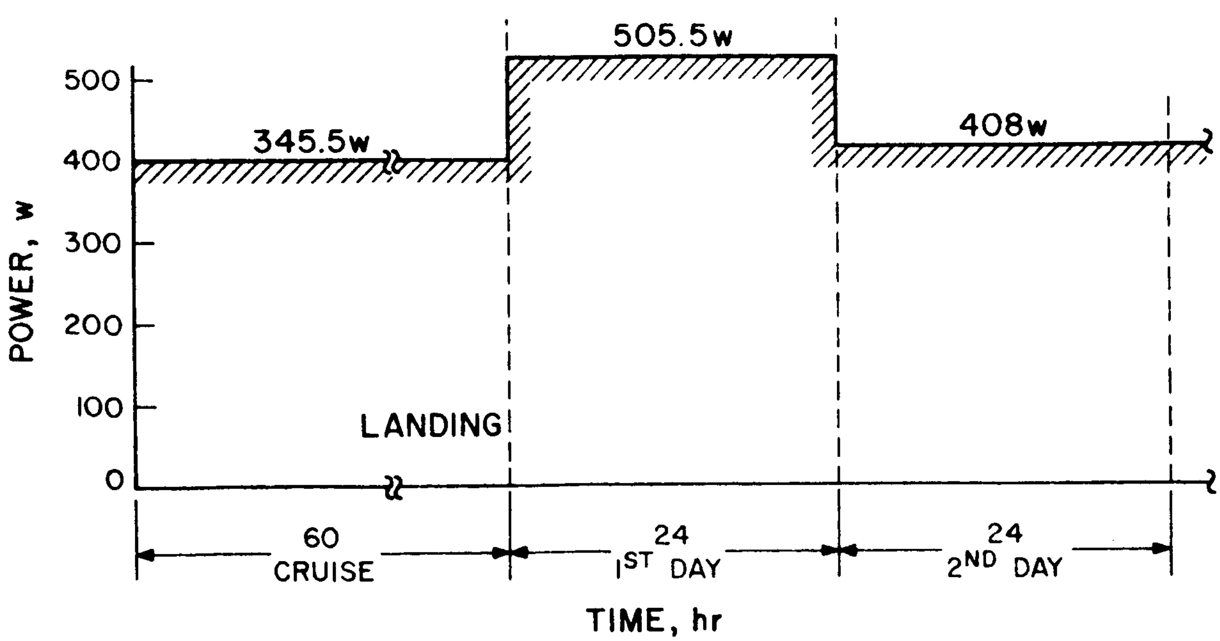

Figure 4-89. Power Profile for Option 1a, Case C

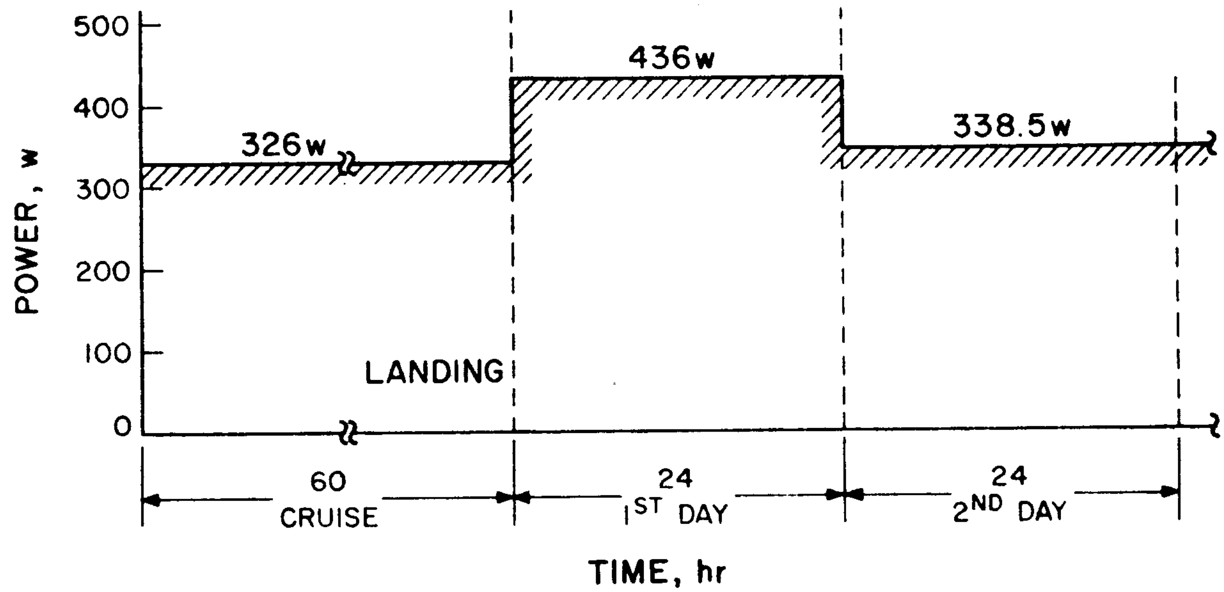

Figure 4-90. Power Profile for Option 1a, Case D

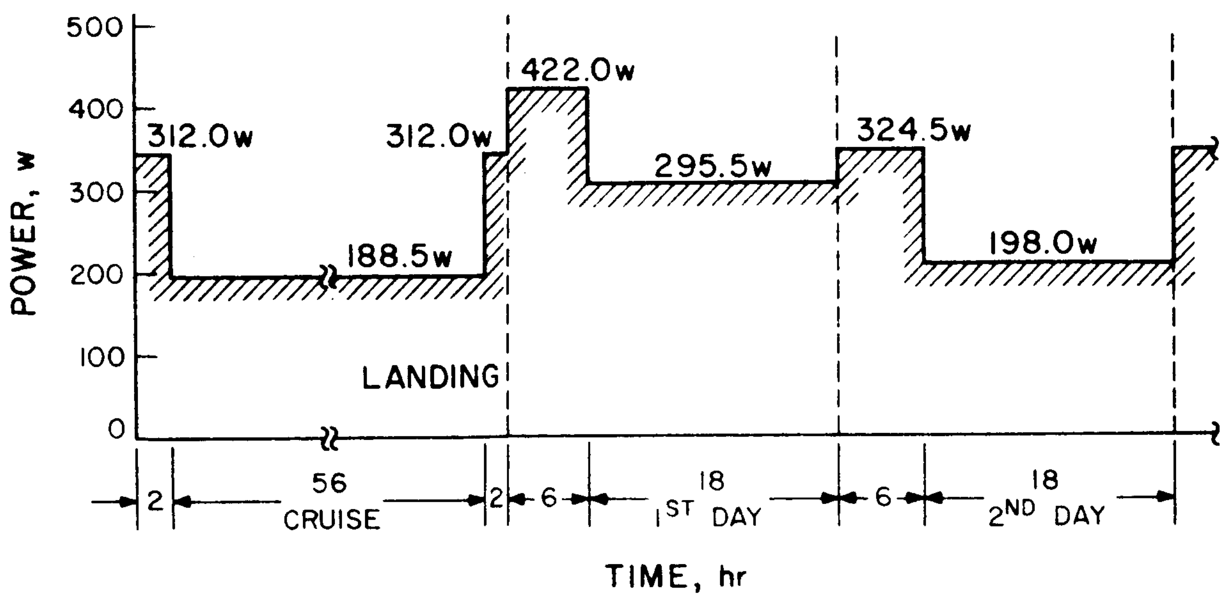

Figure 4-91. Power Profile for Option 1b, Case A

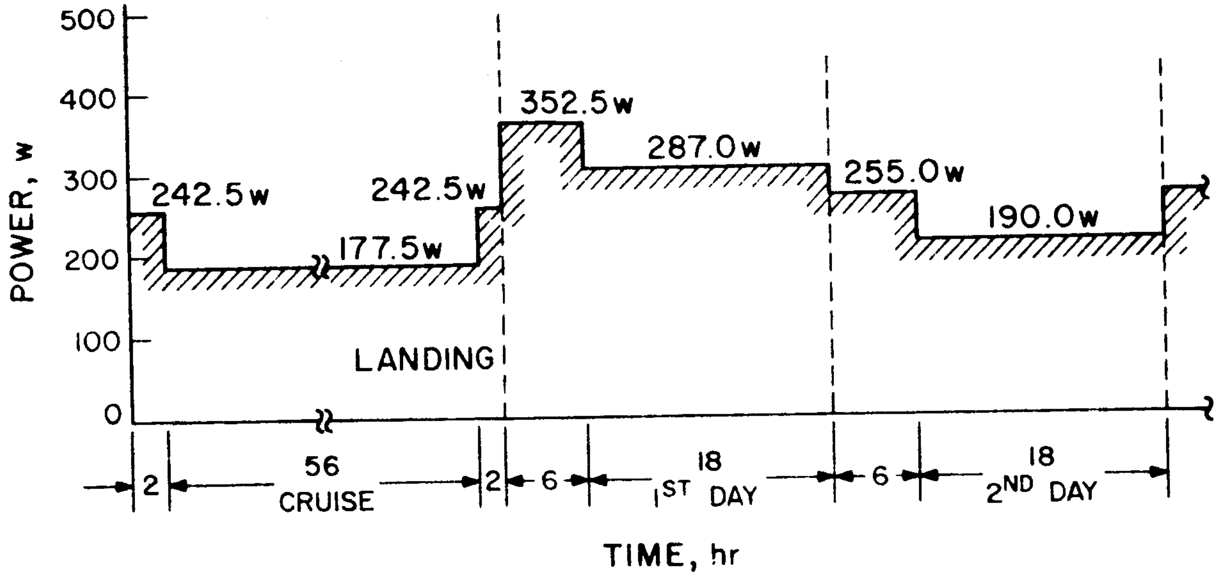

Figure 4-92. Power Profile for Option 1b, Case B

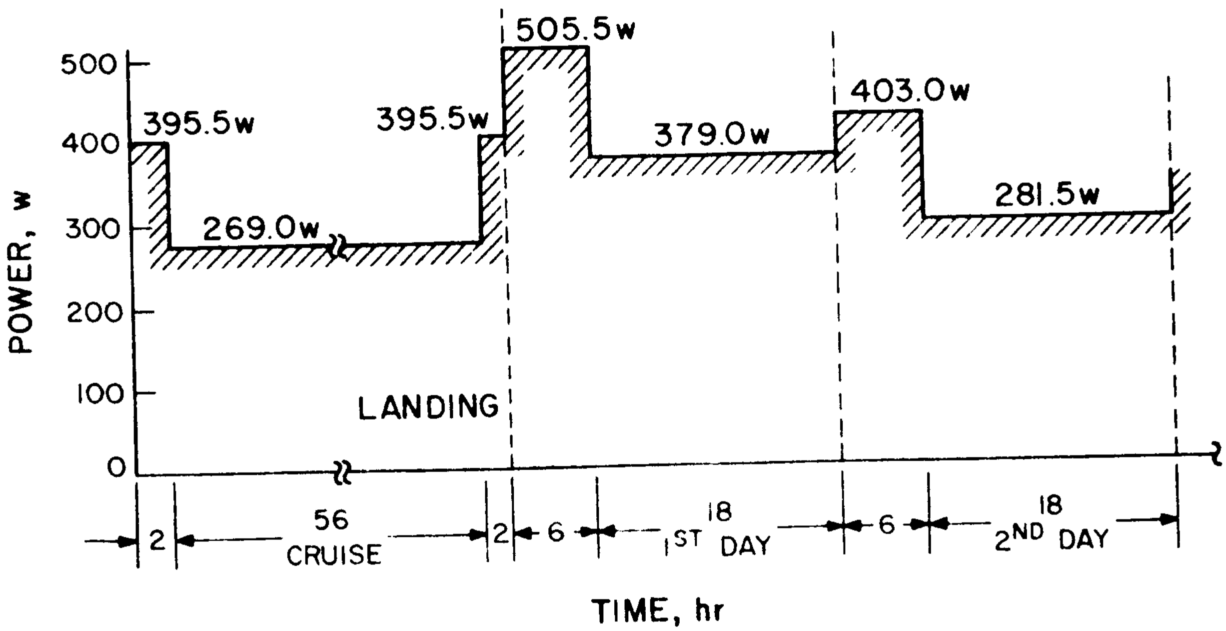

Figure 4-93. Power Profile for Option 1b, Case C

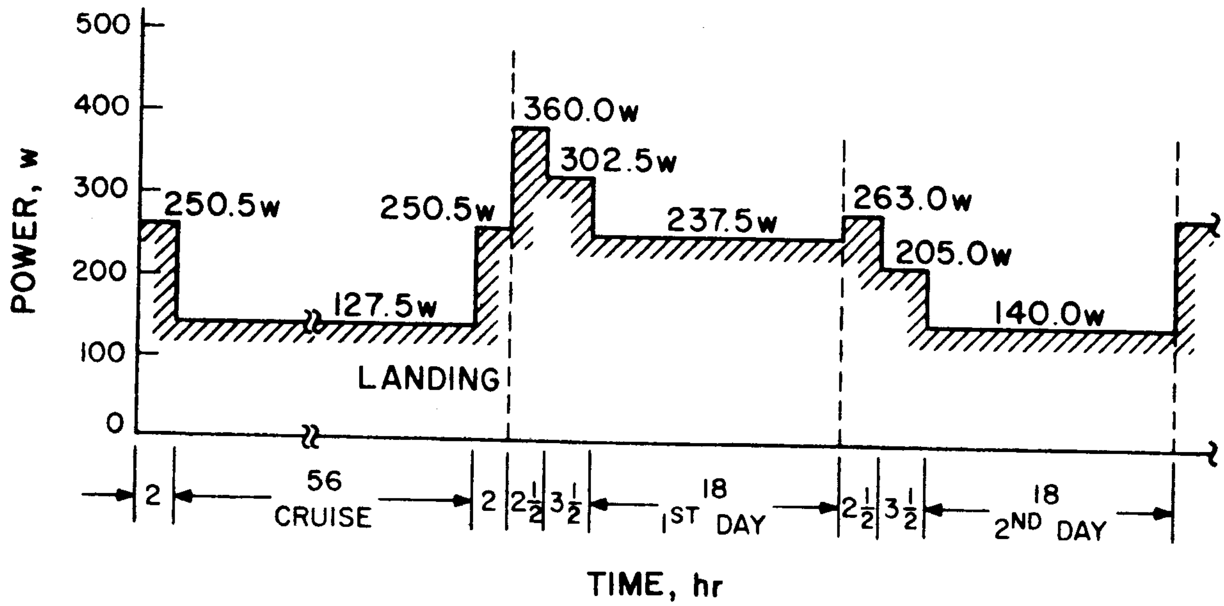

Figure 4-96. Power Profile for Option 2, Case B

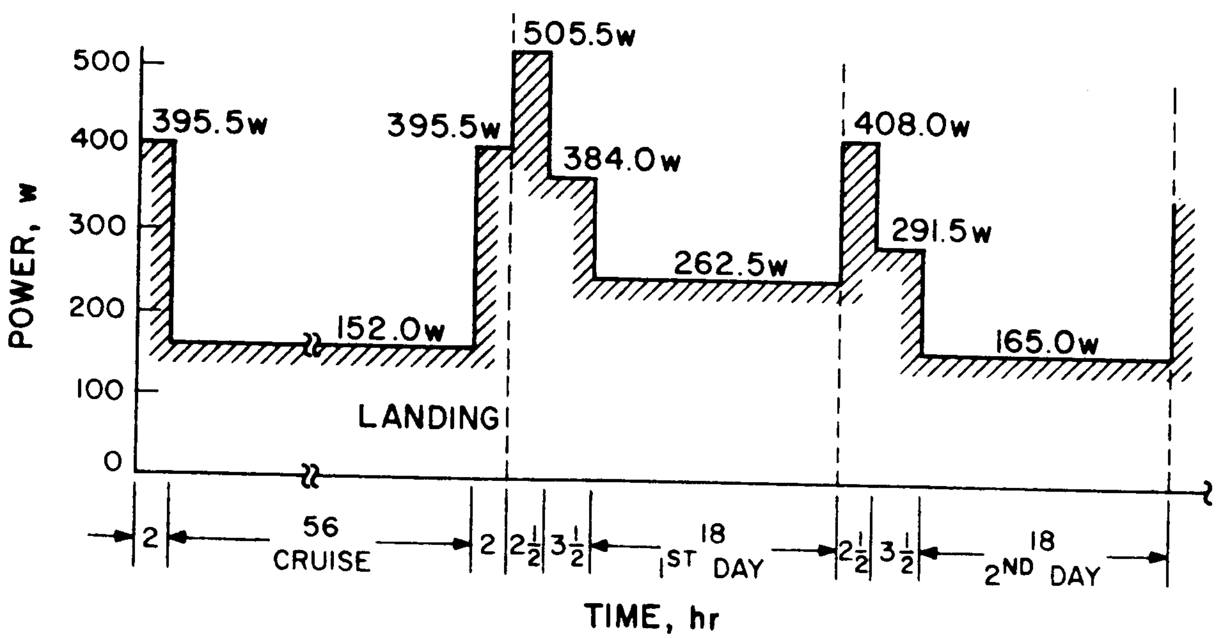

Figure 4-97. Power Profile for Option 2, Case C

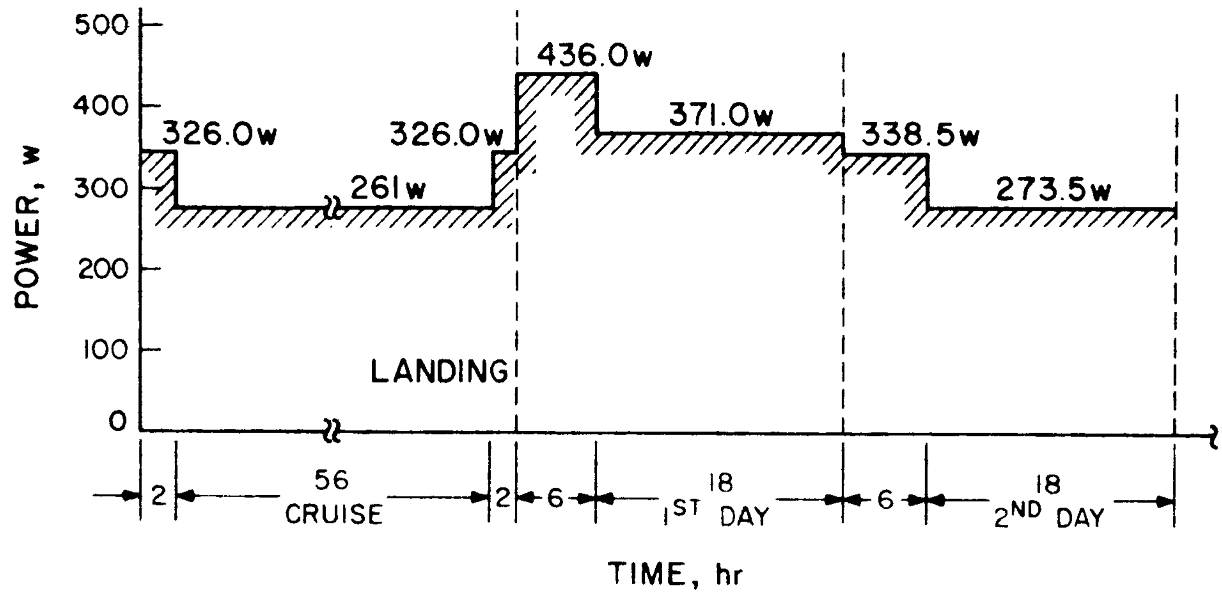

Figure 4-94. Power Profile for Option 1b, Case D

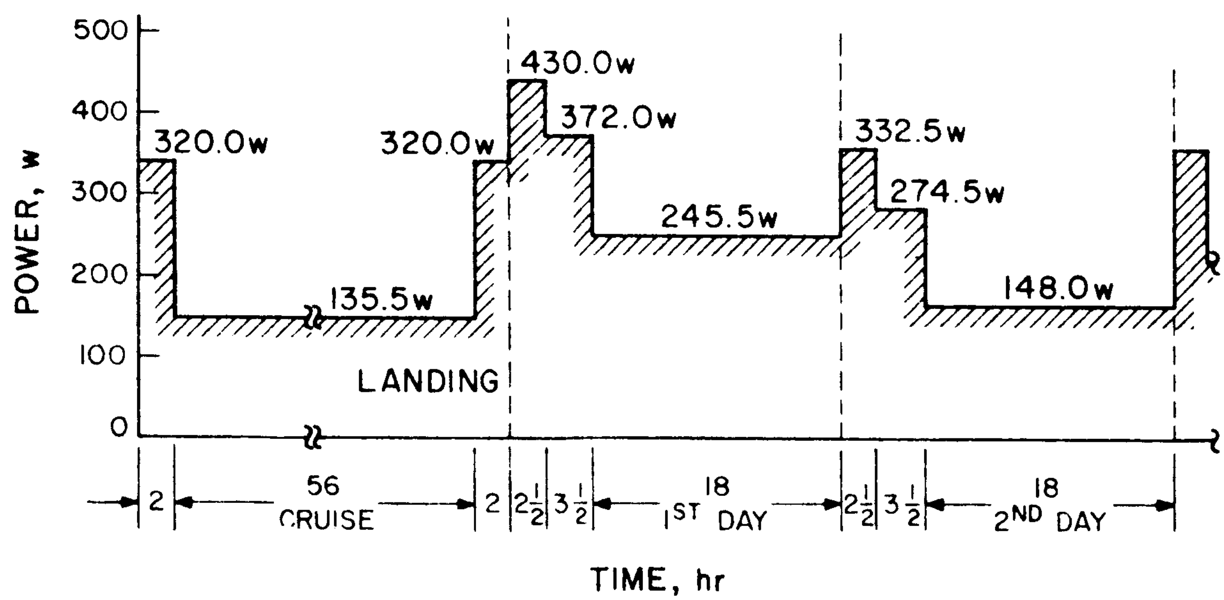

Figure 4-95. Power Profile for Option 2, Case A

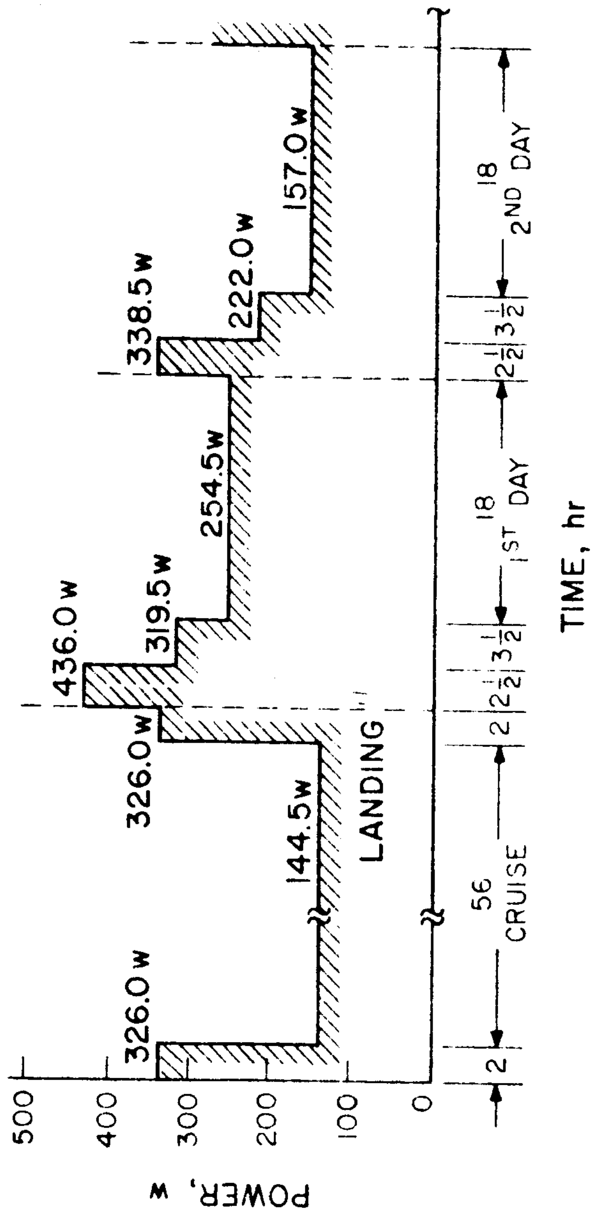

Figure 4-98. Power Profile for Option 2, Case D

Table 4-23. Radioisotope Power System Weights

Option	Case	Radioisotope Generators				Battery		Total System[b] Weight (lb)
		Number of Modules	Power Level per Module ([illegible])	Total Power ([illegible])	Total Weight of Generators (lb)	Energy ([illegible]-hr)	Weight[a] (lb)	
1a	A	4	90	360	446	3500	40	506
	B	4	75	300	395	3000	35	450
	C	4	115	460	540	3500	40	600
	D	4	90	360	446	3500	40	506
1b	A	3	90	270	335	3500	40	395
	B	3	80	240	308	3500	40	368
	C	4	75	300	395	3500	40	455
	D	4	75	300	395	3500	40	455
2	A	3	80	240	308	3000	35	363
	B	3	75	225	296	3500	40	356
	C	3	80	240	308	3500	40	368
	D	3	80	240	308	3000	35	363

Notes:

[a] Includes weight of charger

[b] Includes 20 lb for conversion equipment

Table 4-24. Primary Battery System Weights and Volumes

Option	Case	Phase	Raw Power Required (watt-hr)	Manually Activated Battery: Stored at 50 F: Installed (watt-hr)[a]	Weight (lb)[b]	Volume (ft³)[c]	Stored at 120 F: Installed (watt-hr)[d]	Weight (lb)	Volume (ft³)	Automatically Activated Battery: Stored at 50 F: Installed (watt-hr)[e]	Weight (lb)	Volume (ft³)	Stored at 120 F: Installed (watt-hr)[f]	Weight (lb)	Volume (ft³)
1c	A	I	18,120	20,611	202	[illegible]	[illegible]	352	[illegible]	[illegible]	174	[illegible]	[illegible]	211	[illegible]
		II	10,128	[illegible]	110	0.64	19,375	191	[illegible]	[illegible]	105	0.61	11,769	116	[illegible]
		III	[illegible]	[illegible]	[illegible]	0.49	14,598	146	0.85	[illegible]	[illegible]	0.47	[illegible]	[illegible]	[illegible]
	B	I	14,550	16,020	158	0.91	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	0.88	[illegible]	[illegible]	0.94
		II	8,460	9,315	92	0.53	[illegible]	[illegible]	[illegible]	[illegible]	88	[illegible]	[illegible]	[illegible]	[illegible]
		III	[illegible]	6,738	67	0.39	11,706	115	[illegible]	[illegible]	64	[illegible]	[illegible]	[illegible]	[illegible]
	C	I	23,730	[illegible]	257	[illegible]	45,395	446	[illegible]	[illegible]	[illegible]	[illegible]	27,574	[illegible]	[illegible]
		II	12,132	13,357	131	0.76	23,209	228	[illegible]	12,824	126	[illegible]	[illegible]	[illegible]	[illegible]
		III	9,792	10,781	106	0.62	[illegible]	184	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	0.65
	D	I	19,560	21,530	212	[illegible]	37,418	[illegible]	2.13	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
		II	10,464	[illegible]	113	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
		III	[illegible]	[illegible]	86	0.51	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	0.54
1b	A	I	[illegible]	12,996	128	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
		II	[illegible]	[illegible]	85	0.49	[illegible]	148	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
		III	[illegible]	6,068	60	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
	B	I	[illegible]	[illegible]	[illegible]	0.69	[illegible]	205	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
		II	[illegible]	[illegible]	[illegible]	0.46	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	0.48
		III	4,950	5,450	[illegible]	[illegible]	[illegible]	[illegible]	0.54	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
	C	I	16,662	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
		II	9,855	10,850	[illegible]	[illegible]	18,853	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	11,412	[illegible]	[illegible]
		III	7,515	8,274	[illegible]	[illegible]	14,376	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
	D	I	15,920	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
		II	[illegible]	[illegible]	101	0.59	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
		III	6,754	[illegible]	76	0.44	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
2	A	I	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
		II	6,769	7,482	74	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
		III	[illegible]	[illegible]	48	0.28	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
	B	I	[illegible]	8,964	88	0.51	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	0.54
		II	6,234	6,864	68	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
		III	[illegible]	4,286	42	[illegible]	[illegible]	74	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
	[illegible]	I	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
		II	[illegible]	[illegible]	90	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
		III	[illegible]	[illegible]	55	0.32	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
	E	I	[illegible]	10,345	102	0.59	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
		II	[illegible]	7,475	74	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
		III	4,449	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

Notes

[a] The installed to required power ratio is [illegible]

[b] The weight [illegible] was [illegible] at the rate [illegible]

[c] The volume was assumed at [illegible]

[d] The installed to required power ratio is [illegible]

[e] The installed to required power ratio is [illegible]

[f] The installed to required power ratio is [illegible]

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Table 4-25. Power Requirements for an Active Capsule Attitude-Control System

Equipment	Regulated Power Required (w)	Raw Power Required (w)	Time on	Energy (w-hr)
Gyros & Electronics	18.0 16.0 14.5	25.8 21.4 20.8	3 min at start up 5 min ≈ 60 hr	1.3 1.8 1248.0
Switching Amplifier, Logic & Power	4.0 8.0	5.7 11.5	54 hr 6 hr	307.8 69.0
Valves & Jets	4.0 8.0	5.7 11.5	0.1 sec/5 min ≈ 1.2 min 1 min	0.1 0.2
Autopilot Electronics	2.0 4.0	2.9 5.7	10 min 30 sec	0.5 0.1
Jet Valve Actuator	6.0 12.0	8.6 17.2	10 min 30 sec	1.5 0.1
Maximum Peak Average		71.7 27.8		1630.4

G. TELECOMMUNICATIONS

1. Introduction

a. Scope

In the areas of telecommunications, considerable effort has been directed toward an analysis of the radio tracking, telemetry and command functions for both an orbiter and a landing capsule. The four principal objectives of this study were to:

1) Determine by estimation and assumption the gross telecommunication requirements for an orbiter/lander spacecraft of the Voyager class.
2) Determine the capabilities, trade-offs, advantages and disadvantages for at least two alternate mechanizations that would meet the estimated telecommunication requirements.
3) Delineate the specific problems that must be solved before the feasibility of the proposed mechanizations can be realized. Feasibility, as used here, includes not only technical but also economic considerations.
4) Specify the effect of the infeasibility of a proposed subsystem mechanization on the communication capability in terms of the functional requirements.

The effort was directed toward the phases of the mission which follow the separation of the capsule from the bus approximately a million kilometers from Mars.

Between the orbiter subsystems and the capsule subsystems there are a large number of detail variations in the mechanizations that can be studied. These variations result not only from different possible modes of operation but also from the varying degrees of redundancy that may be specified. The two mechanizations that were analyzed vary only in the type of control that is used in the capsule communication system. The first, referred to as Option 1, utilizes no external radio control of the capsule communication modes. The second, Option 2, utilizes some form of radio control of the capsule.

As will be shown in Subsection J, the orbiter systems generally meet the desired communication requirements with a reasonably proportioned design, while the capsule systems do not quite meet these requirements. Thus, in successive iterations, the requirements, particularly telemetry, on the capsule system and the design must be reexamined so that a reasonable design can be achieved.

Two factors suggest that the demands on the communications system (and indeed the entire spacecraft) are quite severe and the mission may actually not be feasible. The first is the problem of reliability or the probability of performing a useful mission. For mission lifetimes of the duration being considered (about a year) and the complexity of the spacecraft, an acceptable probability of a successful mission, even considering the degree of component and system redundancy assumed herein, is seriously in doubt. Secondly, some of the concepts presented in this section (G) of the Voyager study require considerable development effort before being considered ready for flight use.

Even though the 1969 mission seems distant, the time required for preliminary design, hardware fabrication, and system checkout leaves only a short time for development work. Consequently, it may not be possible in the time available to develop all the techniques and hardware necessary to implement the designs presented herein.

b. Definitions and Block Diagram

A basic functional block diagram of the capsule-orbiter telecommunications systems is shown in Figure 4-99.

For this discussion relating to the telecommunication area, a number of terms are defined as follows:

(1) Orbiter Functions, Elements or Subsystems. Those functions, elements or subsystems such as telemetry or command which are related to the basic spacecraft which is eventually placed in orbit around Mars and which would be required even if there were no associated capsule.

(2) Capsule Function. Those functions which are strictly related to only the capsule system.

(3) Capsule-Orbiter Functions. Those functions, especially on the orbiter, which result from the interaction of the orbiter and capsule systems.

(4) Radio Subsystem. The radio subsystem for either the capsule or orbiter as shown in Figure 4-99 includes all of the antennas, receivers, transmitters, and associated RF circuits.

(5) Telemetry Modulation-Demodulation Subsystems. For the purpose of this study, the telemetry function has been split into two subsystems: data handling, which because of its complexity is covered in a separate Subsection I, and the modulation-demodulation subsystem. As shown in Figure 4-99, the latter subsystem includes the circuits that (1) accept the telemetry data in a serial binary waveform and combine it with synchronizing information into a composite signal for modulating the RF carrier, or (2) demodulate composite data and sync signals to produce data in serial binary waveforms.

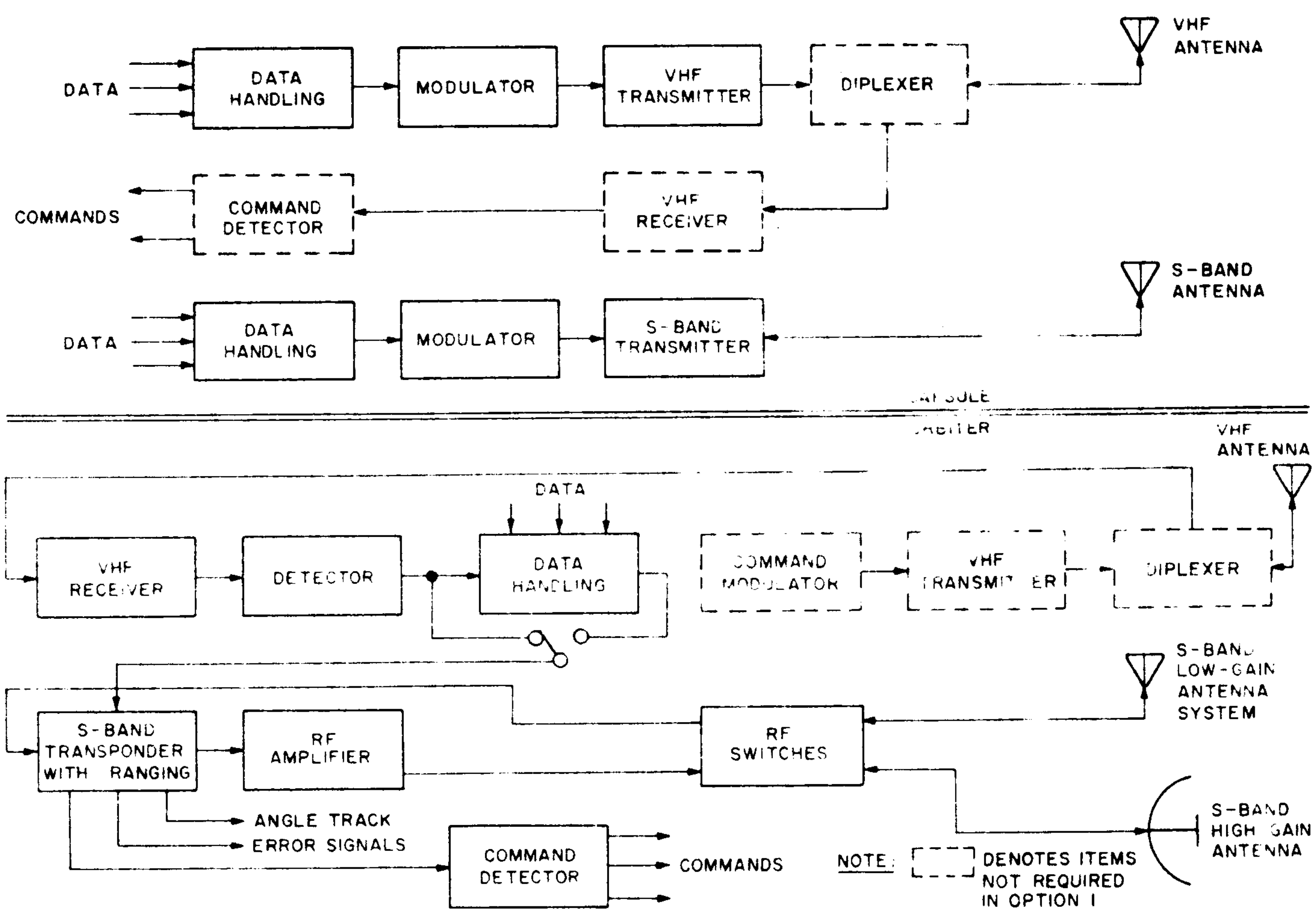


Figure 4-99. Telecommunications Functional Block Diagram

(6) Command Subsystem. The command subsystem includes the non-RF circuits which generate or detect and decode signals which convey control information from Earth.

(7) Link. A link is a radio frequency transmission circuit between pairs of transmitters and receivers from Earth-to-orbiter and vice versa.

(8) Direct Link. Direct link is used exclusively to mean an S-band link from the capsule to Earth.

(9) Relay Link. Relay link is used to mean a VHF link from capsule to orbiter or vice versa where the orbiter is used as a radio relay station between the capsule and Earth.

2. Basic Assumptions and Philosophy

a. DSIF Configuration and Parameters

(1) General Description. The DSIF Ground Radio Subsystem described herein consists of antenna reflectors, feeds, feed lines, diplexers, low noise amplifiers, receivers, exciters, data extractors, transmitters, and acquisition equipment. This S-band equipment will also include equipment necessary for subsystem test and calibration. The subsystem will provide angle-of-arrival error data to the tracking servo, wide-band and narrow-band telemetering outputs, one-way and two-way doppler outputs, ranging detection, ranging and command modulation capability, frequency synthesizers and exciters, and high-power RF generation. Equipments having these capabilities for support of Voyager will be located at the DSIF tracking and communications sites at Goldstone, California; Canberra, Australia; Johannesburg, South Africa; and Madrid, Spain. Launch Station equipment will be located at AMR.

(2) Antenna Reflectors. Antenna reflectors employed in the DSIF are of the paraboloidal type, and for S-band will use cassegrainian feed configurations. At present, the standard DSIF antennas are HA-Dec 85-ft reflectors, by the time Voyager is launched in early 1969, three AZ-El 210-ft reflectors will also exist.

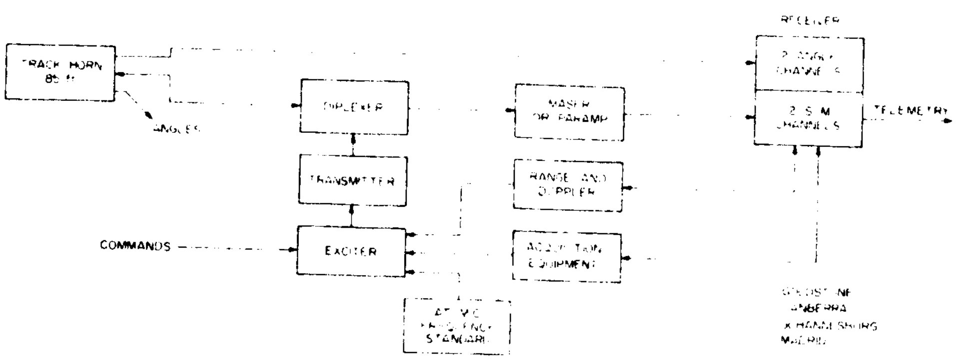
(3) Receivers. Receivers employed are double superheterodyne types which phase-lock with the incoming RF carrier. Angle-of-arrival tracking is accomplished by deriving angle error pointing information from the output of the simultaneous lobing feeds. The S-band receiver incorporates coherent AGC and ranging system reference loops. The basic S-band maximum bandwidth is 3.3 Mc. APC loop noise bandwidth is selectable from 12 cps to 150 cps at threshold. The receivers provide telemetry detection channels and

doppler extraction equipment. The receiver excess noise temperature, operating without a low-noise amplifier (as in the Launch Station and Range and Doppler Station), is approximately 3000°K. With a paramp the system T_{excess} is nominally 250°K and with a traveling wave maser, 60°K. Dynamic signal level range is from -65 dbm to threshold (typically -160 to -170 dbm). S-band receivers are tunable to 2295 ±5 Mc with crystals, each crystal oscillator being voltage tunable over a ±75 Kc range. Rapid change of frequencies will be available.

(4) Transmitters and Exciters. The S-band transmitters use 10 kw klystron amplifiers. Typical gain is 45 db, and the minimum bandwidth is 6 Mc. A backup 100 kw klystron amplifier will be available at Goldstone and possibly several other sites. Low power transmitters of approximately 100 w power will be employed for the Launch Station and the Range and Doppler Station. An exciter provides the stable frequency source, tuning capability, multipliers, and phase modulators for both the command and ranging system. Frequency stability better than 1 part in 10^{10} over a 15-minute period is provided by slaving to an atomic clock. Transmitter frequency tuning is provided in 100 cps steps throughout the transmitter channel (2115 Mc ±5 Mc), with a voltage-controlled crystal oscillator (VCO) locked to the synthesizer output. Separate VCO's will be provided to provide compatible transmitter frequencies with the receiver.

Standard Deep Space Station

(a) Block Diagram

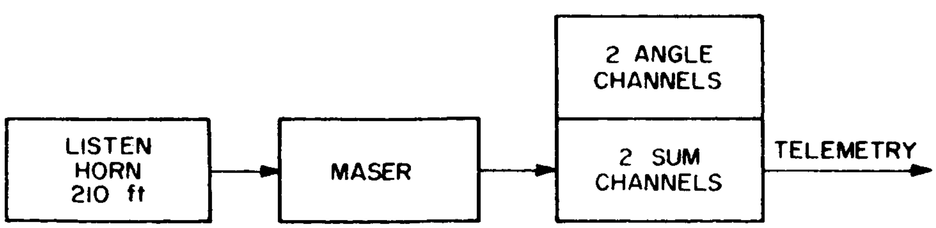


(b) Parameters

Parameter	Receive	Transmit
Antenna Gain	$53.0 ^{+1}_{-0.5}$ db	51.0 ±1 db
Antenna Beamwidth	~0.4 deg	~0.4 deg
Axial Ratio	0.75 ±0.25 db	0.75 ±0.25 db
Feed Line Loss to Low Noise Amplifier	0.18 ±0.05 db	--
Feed Line Loss to Xmtr.	--	0.5 ±0.1 db
Antenna Temperature (zenith)	10° ±3°K	--
Max Feed Power	--	10 kw
Polarization	LC & RC	LC & RC
Diplexer Loss (included in Item 4)	0.1 ±0.02 db	0.05 ±0.025 db
Paramp T_{excess} (including second stage)	$165° ^{+30°}_{-10°}$K	--
Maser T_{excess} (including second stage)	18° ±3°K	--
Transmitter Power	--	to 10 k
Transmitter Noise Contribution	10° ±3°K	--

Low Noise Listening Station

(a) Block Diagram



(Note two-site operations may be possible at Goldstone as was done on Mariner 2 for two-way doppler and command)

(b) Parameters

Parameter	Value
Antenna Gain	61.0 ±1 db
Antenna Beamwidth	0.15 deg
Axial Ratio	<0.5 db
Feed Line Loss to Low Noise Amp	0.02 ±0.01 db
Antenna Temperature	10° ±2°K
Polarization	RC & LC
Maser T_{excess} (including second stage)	18° ±3°K

(5) Dual Probe Coverage. In the event of two probes simultaneously occupying the same antenna beamwidth, it will be possible to receive telemetry from one probe while performing two-way doppler, command, and telemetry functions with the other. The period required to switch two-way doppler and command from one spacecraft to the other would take about one hour at planetary distances.

(6) Coverage. DSIF coverage will be maintained as follows:

1) Phase 1. Injection to First Midcourse +3 Days. Continuous coverage for 24 hours a day during the first five to ten days will be provided. The tracking data will provide angles for several days, precision two-way doppler, and ranging.
2) Phase 2. Additional Midcourse Maneuvers. Continuous coverage for 24 hours per day will be provided for two days prior to and three days after the maneuver.
3) Phase 3. Cruise (Tracking). Coverage will be one pass over one station every four days; the tolerance being not less than two, or more than six days between each pass. Tracking will provide precision two-way doppler and ranging.
4) Phase 4. Near Planet. From ten to five days prior to encounter, coverage will increase to one pass every other day. Tracking will provide precision two-way doppler and ranging.
5) Phase 5. Approach and Capsule Landing. From encounter minus five days to landing plus thirty days, continuous coverage 24 hours a day will be provided during capsule life. Precision two-way doppler and ranging will be used to determine the orbit of the orbiter vehicle after two days of tracking. (The method of establishing capsule position is unresolved at this time.)

6) Phase 6. Orbiter Life. From the end of capsule life, nominal-encounter plus four weeks, to the end of the mission, nominal-encounter plus 150 days, coverage will be 16 hours a day.

The DSIF requirements per spacecraft would be:

In any 24 hour period, there must be one period of at least 8 hours of continuous data reception. There cannot be a gap between 8 continuous hour reception periods of more than 16 hours.

This requirement when coupled with the spacecraft data transmission cycle will provide reception of all the spacecraft data.

Precision two-way doppler and ranging will be provided.

(7) Telemetry Coverage. Telemetry coverage during the cruise phase will be the same as that during phase 6, the orbital phase. The 85-ft net may be used for coverage when the spacecraft-to-Earth range is low enough to permit reception by this net.

b. DSIF Availability

In order to study the effect on the orbiter data system of the DSIF availability for periods other than 24 hours per day during the lifetime of the orbiter, one must make an assumption of the value of the data to be transmitted. The assumption made here is that all data transmitted by the orbiter is of great value, and random gaps in the receiving capability cannot be tolerated.

It is not practical to commit the DSIF net to 24 hours per day of coverage for extended periods of time for three important reasons.

1) Station scheduled and nonscheduled maintenance will be required and cannot be considered to occur only during station nonvisibility periods.
2) It is quite likely that more than one spacecraft will be in transit to the planet or in orbit about the planet at the same time.
3) The DSIF will be committed to other programs during this time period and, thus, must divide its time between programs.

One solution to the nonavailability problem would be to use an Earth-based radio command to initiate and cease spacecraft transmission. Data storage would be required during the nontransmission periods. This solution will not be considered further (except as a possible backup mode) since it would require transmission of three or four commands per day at station elevation angles close to the horizon.

Another solution is to use data storage and retransmission several time, coupled with a minimum DSIF coverage per spacecraft in order to ensure all data is received at least once. The method proposed is to transmit all data three times with an 8-hour cycle period. First transmit an 8-hour block of date in real time and simultaneously store it. Then during the next two successive 8-hour periods, retransmit this data twice in addition to continuous real-time data transmission. The effect of this solution would be to increase, by a factor of three, the required data transmission rate for a given input rate.

During the mission phase of capsule planetary entry and lander life on the surface, the DSIF requirements will be 24 hours per day coverage. This is expected to be a 4-week period. This coverage would be for lander direct-link reception or for orbiter transmission of lander data. It is recognized that this 24 hour per day coverage may conflict with reception from another spacecraft in transit to the planet.

The preceding has assumed that the orbiter will be continuously in view of the Earth during its orbit life. For the nominal orbits, this is expected to be true for about the first 90 days and then the orbiter will be occulted by the planet during a portion of each orbit after this period. Under conditions of occultation, an additional restriction would be placed on the DSIF in order to receive data during a retransmission period that otherwise would have been received had the orbiter not been occulted.

e. Lander-Orbiter Options

Two basic configurations of lander-to-orbiter data transmission will be considered. One will be a lower risk (from the standpoint of total hardware required), lower capability method, while the other will be a higher risk, higher capability method. These methods will be designated as Options I and II respectively. Options I and II are characterized by the following:

(1) Option I. - The lander data transmission to the orbiter is entirely controlled by lander on-board sequencing. No external control, such as Earth-based radio command or orbiter control, would be used to sequence the lander data. This option is lower risk since data transmission to the orbiter does not depend upon the proper functioning of an orbiter transmitter and lander receiver. Another advantage of Option I is that the lander and orbiter VHF antennas are not required to operate at two frequencies, thus simplifying the antenna design. However, it is a lower data capability method since the lander has no information to tell it that the orbiter is overhead and data may be transmitted. Thus, under this option, the lander must use a preprogrammed "shot-gun approach" to maximize the data transmission between the lander and orbiter. Data handling aspects of Option I are discussed further in Chapter 4, Section II, (I).

(2) Option II. The lander data transmission to the orbiter is controlled by signals from the orbiter and/or Earth. To accomplish this requires the use of a lander receiver, orbiter transmitter, and possibly an orbiter command modulator and lander command detector and decoder. This option requires the proper functioning of more equipment, but has a higher overall data transfer capability since the lander can sense that the orbiter is overhead and data may be transmitted.

Both options include the use of a direct lander-to-Earth communications link. The telecommunications Block Diagrams given in Figures 4-99 through 4-101 show items only associated with Option II as dashed lines. Data handling aspects of Option II are discussed further in Chapter 4, Section II, (I).

d. Orbiter System Philosophy

From a telecommunications viewpoint, the orbiter serves a very important function in the task of obtaining data from the surface of Mars. It serves as a relay station capable of pointing a high-gain, directional antenna towards Earth. Then, acting as a relay station, it can receive data from a lander on the Mars surface and retransmit this data to Earth at a much higher data rate than would be possible directly from the Mars surface to Earth.

In addition, the orbiter will have the capability for performing scientific experiments, and data from these can also be transmitted to Earth via the orbiter communication system. Earth-based radio commands may be used to control orbiter and possibly lander sequencing, selection of redundant elements, or to initiate other desired control functions.

Two-way doppler and ranging capability between the Earth and the orbiter will be provided. A block diagram of the orbiter telecommunications system is shown in Figure 4-100.

In general, the requirements on the communication system are more severe during the orbital phase than during the transit phase. Thus, by first considering the orbital requirements, the majority of transit phase requirements will be covered. Two exceptions to this are (1) the use of low-gain antennas for communication with the spacecraft when the attitude does not allow pointing the high-gain antenna toward Earth, and (2) the requirement for ranging data soon after injection from Earth into a planetary transfer ellipse, which implies small acquisition times for the ranging signal.

Most of the techniques proposed for the orbiter telecommunications system are those currently being implemented in the Mariner series of spacecraft. Much of this study is based upon past Mariner B work and, because some prototype hardware has been built and tested, the performance capabilities are able to be predicted with confidence.

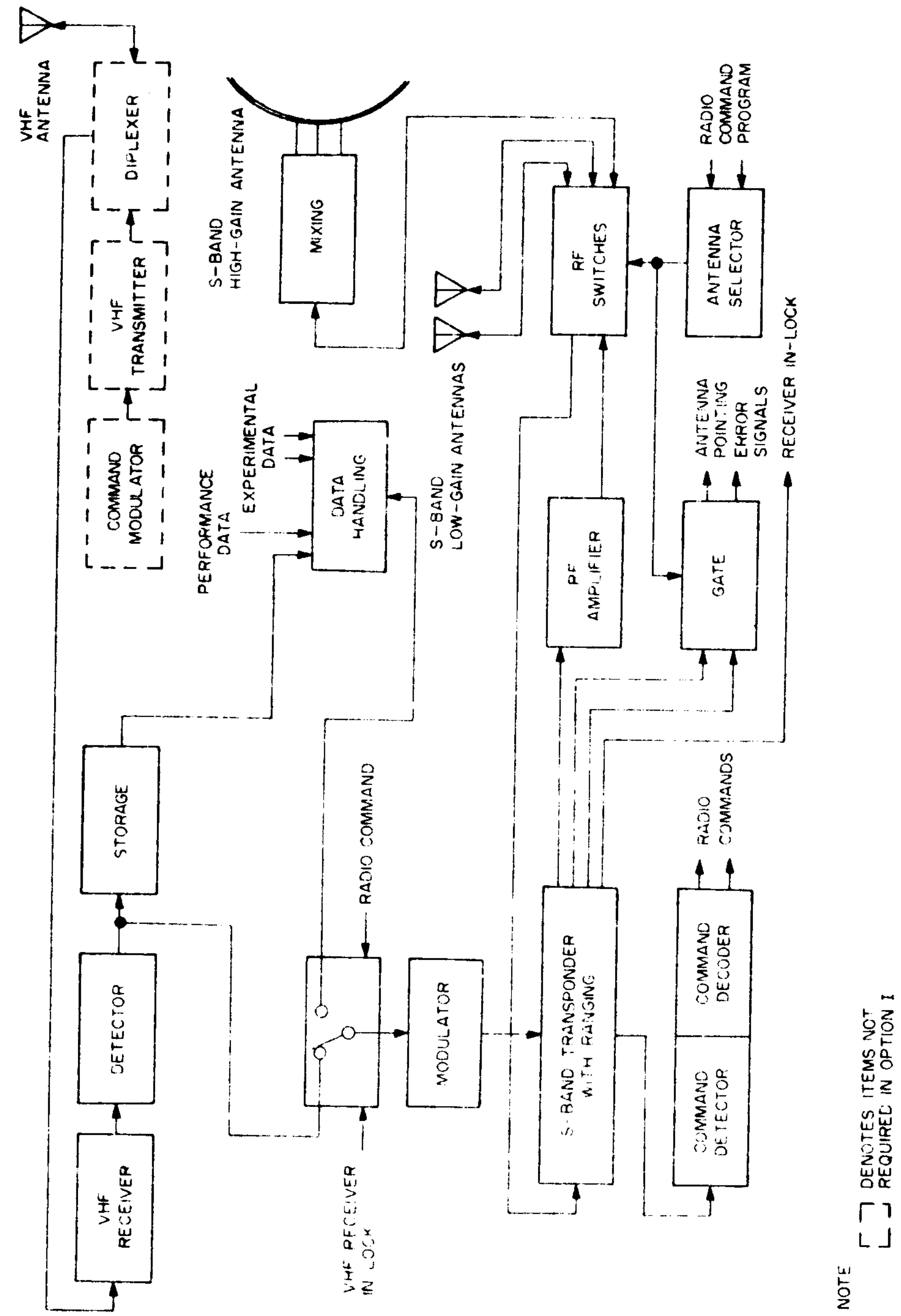


Figure 4-100. Telecommunications Bus Block Diagram

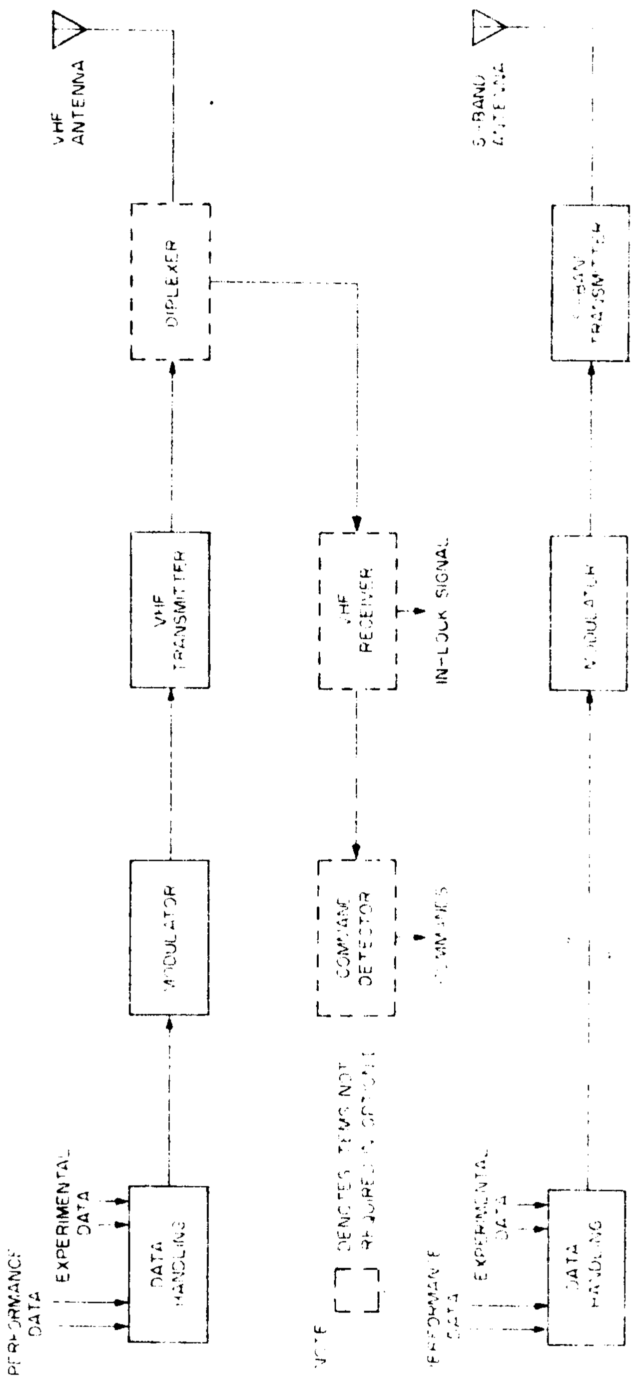


Figure 4-101. Telecommunications Lander Block Diagram

e. Capsule System Philosophy

The principle philosophy used in selecting capsule and capsule-orbiter subsystem configurations was to select those which resulted in the simplest, and therefore most reliable, mechanizations consistent with the communication requirements and the achievement of relatively high communication efficiency. In addition it was considered highly desirable to provide at least two links for transmission of data from the lander to Earth: the first, a relatively high capability relay-link utilizing the orbiter as a relay station, and the second, a minimal capability direct-link which did not rely upon proper operation of the orbiter once the capsule had landed on the planet successfully.

The modes of operation and transmitter power level for the relay link were selected within reasonable power and weight constraints to most nearly approach the real-time data transmission requirements for each capsule phase. Thus, capsule data collected during any one phase is transmitted as soon as possible so that, if a major capsule failure then occurs during the next phase, the data from the previous phase will not be lost.

For the direct link, it was considered desirable to use continuous transmission with but one mode of operation in order to keep the mechanization as simple as possible. As will be shown in Section II (I), even with minimal data capability, this concept will probably have to be abandoned for a cycled mode of operation in order to reduce the required power to a reasonable level.

3. Orbiter-Earth Functions

a. Antenna Configuration

(1) High-Gain Antenna. Based on an antenna size vs transmitter size optimization study (with weight as the primary parameter), an antenna size of 12.5-ft has been selected. An antenna of this size with an aperture efficiency of 55 percent would give a gain of 36.7 db. The polarization will be right-hand circular with less than 0.5 db axial ratio.

There are a number of problems associated with pointing this antenna at the Earth. If the pointing requirements were placed entirely on the spacecraft attitude control system, the following are assumptions of typical errors in pointing the antenna electrical axis toward the Earth:[1] ±1.0 deg for a stored program to point the antennas and for the spacecraft basic attitude stabilization, ±0.15 deg for the structure or arm between the spacecraft and the antenna, and ±0.25 deg for the antenna boresight error. This gives a

[1] See Section IV, B, 5 for additional discussion.

total error of ±1.4 deg. For a 12.5-ft antenna the pattern loss that results for errors of this magnitude is 5 db. If we could tolerate the 5 db loss, then the same performance could be had with a 10-ft antenna. The maximum gain would be lower, but the performance at ±1.4 deg would be the same as the 12.5-ft antenna. There are two advantages to the 10-ft size. It is a smaller and, therefore, lighter antenna. Also, the slope of the antenna pattern at 1.4 deg is less and, therefore, less sensitive to pointing errors. However, the 5 db loss in performance would have to be made up with 5 db more radiated power, and thus the overall system weight would increase, since we no longer have the optimum antenna and transmitter size for minimum weight. If the original gain of 36.7 db must be met, the best that could be done would be to go to an 18-ft antenna and hold the pointing error to ±0.8 deg. This approach, again, suffers from two problems: the larger and therefore heavier antenna upsets the weight optimization in the other direction, and the ±0.8 deg pointing is not consistent with the pointing capability of the spacecraft.

The best solution appears to be to remain with the 12.5-ft diameter antenna and use an angle tracking feed on the antenna to accurately point the antenna. This can be done for about 10 pounds, and pointing accuracies of the order of ±0.25 deg can be obtained. With this accuracy, pointing losses of only 0.2 db would result. For angle tracking, an Earth-based beacon must be in operation for the antenna to track. If the beacon is lost or the angle tracking becomes inoperable, then the accuracy of antenna pointing will become ±1.4 deg as mentioned before. To cover this situation, one solution might be to reduce the data rate to that compatible with 5 db loss in signal for those periods of time when high accuracy tracking is not available.

During normal operation, whenever a high data rate is required, a beacon would be used on Earth. Control of the data rate could be provided by the angle tracking gate that controls the time at which the angle tracking output is supplied to the servo system. Backup control could be supplied by command from the Earth. The antenna would have to have a large enough acquisition angle to be able to acquire anywhere over the ±1.4 deg region.

Investigations of antenna drift rates are necessary to determine if, when the Earth beacon is used intermittently, the antenna can be continuously pointed at the Earth within at least ±0.5 deg, which is the 0.5 db beamwidth of the antenna.

Figure 4-102 illustrates the antenna mechanization that would be required to provide angle tracking along with other components which are placed at the antenna to minimize the circuit losses of the RF portions of the communication system.

In Figure 4-102 those components which make up the angle channels include four feed elements, two error-summing hybrids, two bandpass filters, two tunnel diode preamps, two balanced modulators, and a summing junction. The modulators are used to suppress the

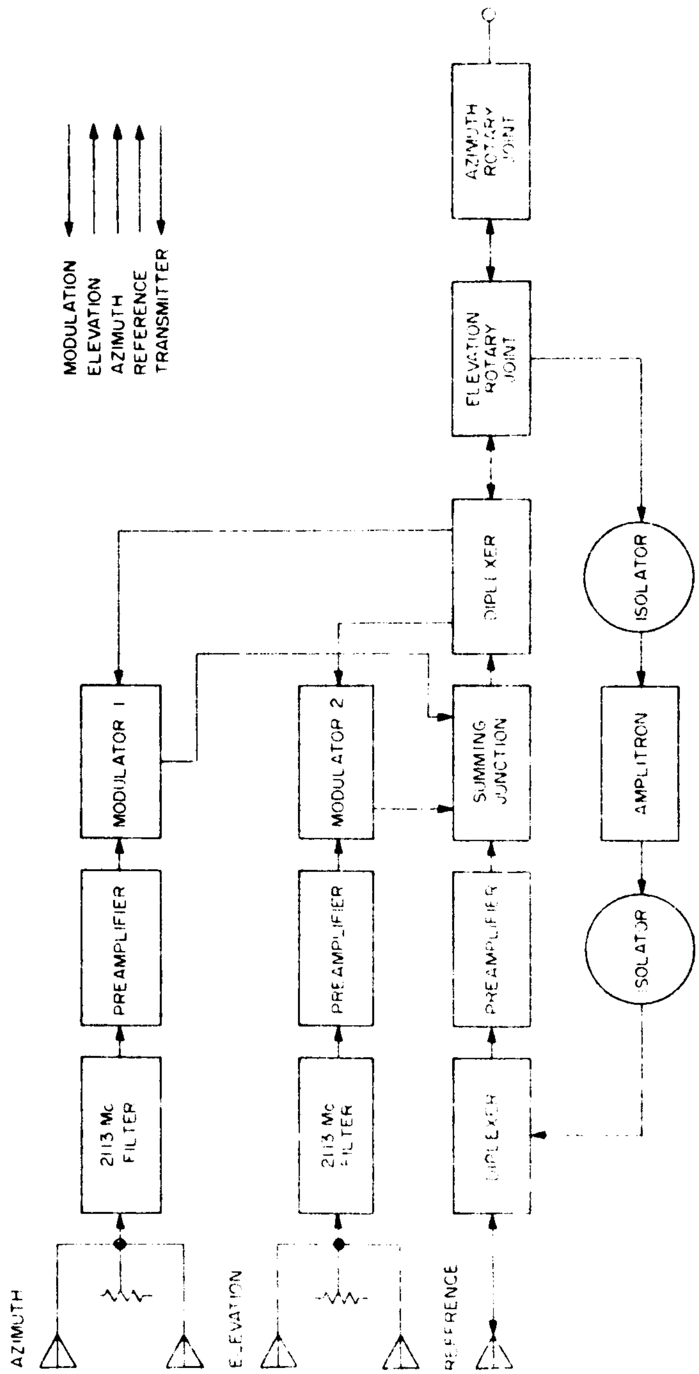


Figure 4-102. Antenna Located Hardware

carrier and to put the tracking information in side-bands so that the error signals could use the same transmission line components and receiver as the reference (command) signal.

It is not technically feasible to run three separate coaxial lines for these signals as this would require three-channel rotary joints which are items that do not exist; no techniques are currently available to mechanize such a device. Also, a one channel system reduces the weight required for the system. The preamps are used to make up for circuit loss in the modulators and transmission line components. The locating of the amplitron at the feed reduces the losses attributable to the transmitter system. The losses are 2 to 3 db in the transmission line components. Although the preferred location of these components is at the feed, they could be located behind the dish. The main problem with this system is that additional cables would have to be run between the feed and dish structure and would considerably complicate the feed erection scheme. To route the modulator signals and electrical power to the antenna, slip rings would be integrated into the coaxial rotary joint design.

Slip rings for this application present a design problem that must be solved.

Due to the antenna size, the antenna will have to be erectible to fit on the spacecraft within the 152 in. diameter upper stage. For a 12.5-ft antenna, it could be packaged in a 5.5-ft diameter by a 2.5-ft thick volume. The feeds and their associated circuitry, as illustrated in Figure 4-102, would be mounted on a ground plane located at the focus of the antenna. The weight at the feed would not exceed 21 lb. The reflector would be a paraboloid with an F/D of approximately 0.35. The weight of the antenna and structure exclusive of the feed weight would be 50 lb. Coaxial and power cables associated with the antenna would be within the state-of-the-art for Voyager. A program has been contracted by JPL to develop antennas of these types.

(2) Omni-Antenna. Two independent omni-antennas will be carried on the orbiter and each will provide a right-hand circularly polarized radiation. One, which will be considered the prime antenna, will be located in such a direction to give coverage during the transit mode and for the majority of the spacecraft maneuvers. The second or fill-in antenna will be used for the remainder of the maneuvers and for coverage in case the spacecraft assumes a nonstandard attitude.

The primary antenna would be similar to that used for Mariner C and would consist of a combination waveguide-structure which would support the antenna at a sufficient distance so that the pattern will not be obstructed by the spacecraft. The antenna itself will be a crossed-slot cap on the circular waveguide, with a coaxial to circular polarization transition at the base of the waveguide. The antenna pattern coverage will be symmetrical about its axis with +3 db on axis and -5 db at 110-deg from this axis. The axis of the antenna when pointed along the spacecraft roll axis gives the most optimum pointing when this axis is oriented toward the Sun. This antenna would weigh 4.5 lb. The second antenna would have its axis pointed in the opposite direction to the first and would weigh 0.3 lb. Its pattern coverage would be symmetrical about its axis and would be -5 db at 80 deg from this axis and +5 db on-axis. Cable weight would be 2 lb and the insertion loss for the primary antenna cable would be less than 0.5 db.

2. Radio Subsystem

a. System Elements

The basic RF system, shown in Figure 4-103, is an extension of the 1964 Mariner B design. Two low-gain antennas are used for maximum coverage. Redundancy is provided in the receivers, exciters, and power amplifiers. Switching of antennas, receiver inputs, and exciter outputs is accomplished by changing the electromagnet drive in the circulator switches. The power supply switching for the exciters is ganged with the drive to CS_3. Since the amplitron looks like a transmission line (0.5 db loss) when power is removed, no RF switching is needed in this area. Transfer between amplitrons is achieved by simply switching prime power between their power supplies. All switching will be controlled through the control unit which derives inputs from radio command, internal sensors, CC&S, and the power subsystem.

(1) Receiver Combinations. Reception with either receiver via any antenna is possible. Simultaneous operation of the receivers is possible using two different antennas. Either the pair A_1 - A_3 or A_2 - A_3 may be used. Thus, simultaneous reception with the two low-gain antennas is possible if desired during early parts of the flight.

(2) Ranging. Two types of ranging mechanizations are discussed. The first is called turnaround ranging and is characterized by direct modulation of the spacecraft transmitter by the ranging code as received by the spacecraft receiver. The second type is called planetary ranging and makes use of a ranging-code detector onboard the spacecraft. The output of this detector then modulates the spacecraft transmitter.

Each receiver includes turnaround ranging. It is recommended at this time that a full planetary ranging system be included on one receiver in a manner such that switching of that receiver between turnaround ranging mode and planetary ranging mode is possible. Switching the planetary system between receivers also may be desirable.

(3) Angle Tracking. Should angle tracking be desired, it is recommended that the AZ and EL modulators be mounted in the high-gain antenna feed (see antenna located hardware block in Figure 4-102). This will allow for a single coaxial line through the rotary

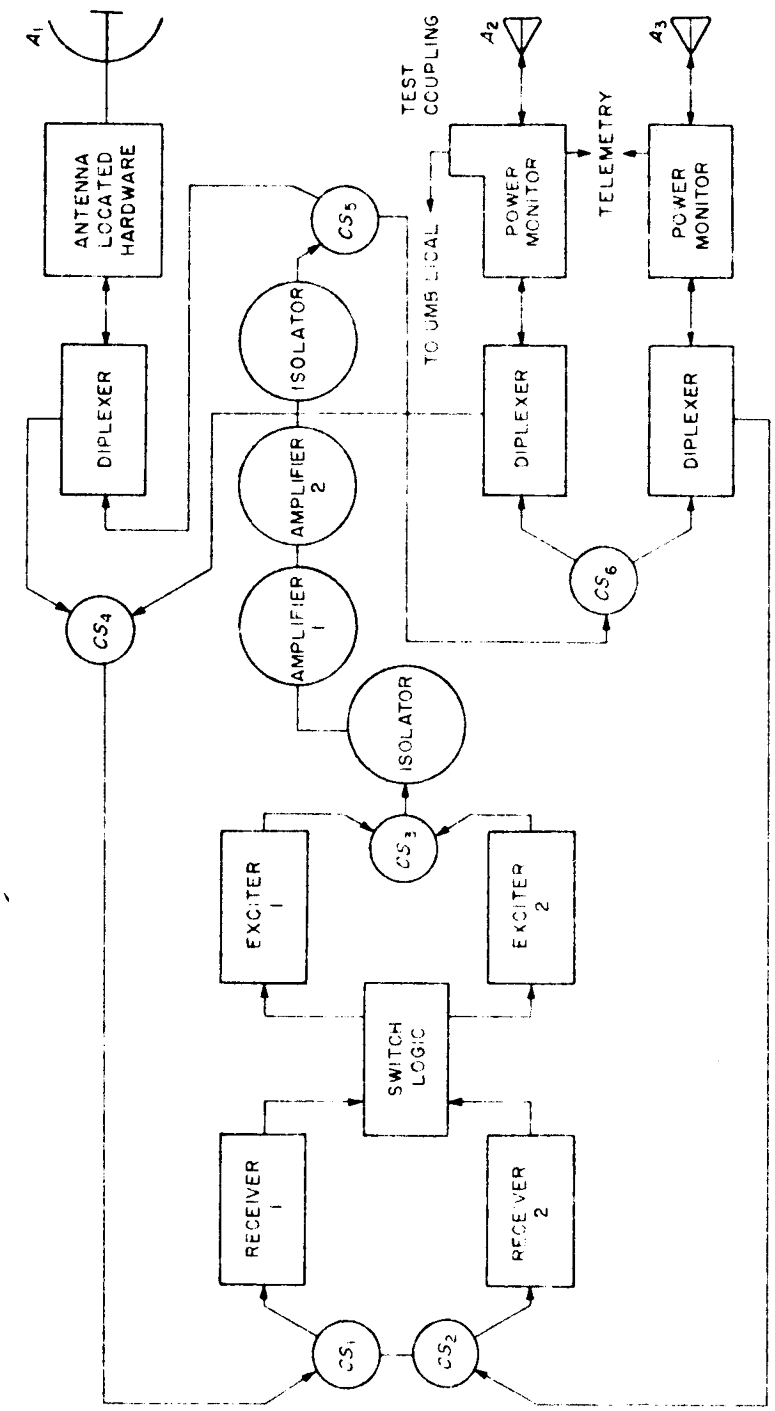


Figure 4-103. Radio Subsystem Mars 1969

joints between high-gain antenna and bus. In principle, it appears that angle tracking could be performed by either receiver. The feasibility of this and the location of the angle tracking receiver front end in the antenna feed should be studied in detail.

(4) Antenna Feed-Mounted Amplitron. It is certain that a large high-gain antenna will be used. Such an antenna could easily result in a large separation between the radio case and the antenna feed, thus producing a high RF loss along this path. In this case, it becomes attractive (from a power-loss consideration) to locate the final power amplifier in the antenna feed. Thus the RF loss would occur at a low power level (~1/2 watt). To provide for power amplifier redundancy with minimum loss via the high-gain antenna, two amplitrons are required in the feed. To allow their use via the low-gain antenna, another channel would be required in the rotary joint, and this is not practical. Another solution to the redundancy problem is to mount one or more power amplifiers in the antenna feed and another (perhaps more than one) on the bus. The amplifiers mounted in the antenna feed would be used only if the amplifiers mounted on the bus failed or if the RF power loss of the bus-to-antenna connection could not be tolerated. This concept is an area for more detailed study.

b. Transmitter Trade-Off Considerations

(1) Orbiter-to-Earth Transmitter. The orbiter transmitter under consideration will operate at S-band at the 10 to 25 w output level.

The first choice of configuration is a solid-state exciter followed by a power amplifier (see Figure 4-104). The exciter will contain the crystal-controlled oscillator, phase modulator, and amplifier multipliers required to produce the S-band output in the 0.4 to 3 w region. Present day efficiencies of this unit are 8 percent, but later developments could raise this from 10 to 12 percent. A vacuum-tube cavity amplifier is available at the 10 w output level, with a gain of 13 db and an efficiency of 54 percent. A 25 w tube is being developed with a 12 db gain and an efficiency of 34 percent. Thus, the overall efficiency of the vacuum tube transmitter would be 29.8 percent at 10 w and 28.8 percent at 25 w output.

At power levels between 20 and 100 w, the amplitron is an attractive device from many standpoints. At 25 w, an amplitron is available that has a gain of 20 db and an efficiency of 50 percent. Thus, this transmitter has an efficiency of 47 percent. Since the amplitron is a bilateral amplifier, an isolator (with 0.3 db loss) is required at input and output. The amplitron generates considerable noise over a very wide frequency range. If the diplexers do not provide sufficient rejection of noise at the receiver frequency, additional filtering (with 0.3 to 0.5 db loss) will be required. The effective amplitron transmitter efficiency could thus be as low as 39 percent.

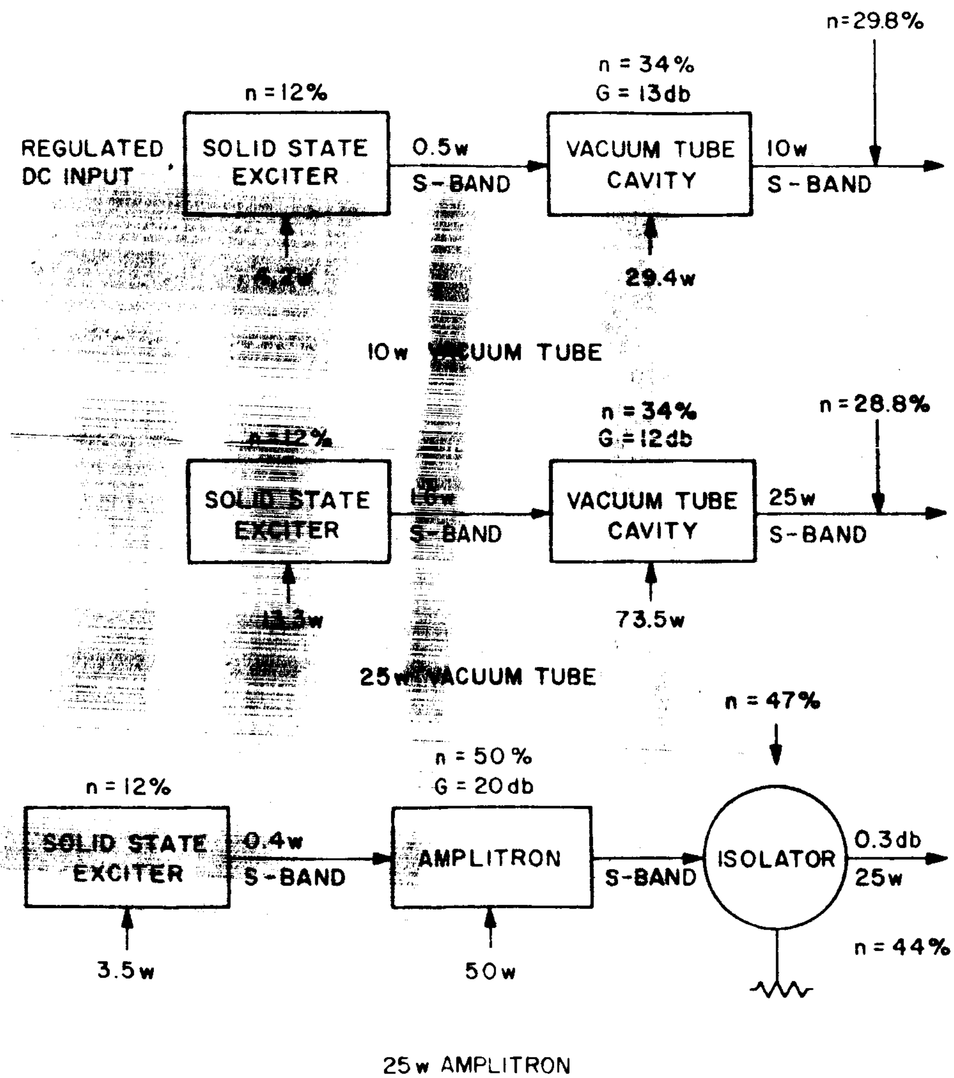


Figure 4-104. Bus S-Band Transmitters

Klystrons and traveling wave tubes have higher gains but lower efficiencies and, at present, look less attractive than the amplitron. There is little hope at this time of developing an all solid-state S-band transmitter at output levels above 5 w.

Ionic breakdown at critical pressure would occur during the launch phase and would be handled in the standard manner; i.e., lowering of vacuum tube voltage or turning-off amplitron voltages for the critical period.

As long as the 25 w power level is not exceeded, the high-vacuum breakdown (the multipacting problem) would not likely occur.

From a standpoint of redundancy, the amplitrons are favored, for a series connection may be used, and the deactivated unit presents only an 0.5 db loss to the circuit. With cavities, parallel circuitry and RF switching are required.

3. Telemetry Modulation Subsystem

It is anticipated that the highest telemetry data rate required for the mission will not exceed 10 K bps, and the lowest will not be below 5 bps. For this range of data rates, a word-synchronous single channel system is recommended for the following reasons:

1) A word-synchronous system provides capability for rapid data synchronization and, thus, more real-time telemetry data.
2) Since it is not necessary to insert word-sync information in the data format, the system efficiency is higher.

A pseudo noise (PN) code of length 63 (maximal-length linear shift-register), cycling at the word rate, will be used to obtain bit- and word-synchronization. Data rates may be conveniently changed by factors of two over the range of data rates given above.

Figure 4-105 shows the telemetry modulation system which is made up solely of digital type networks. The output signal consists of $PN \oplus 2f_s \oplus D$[1] where the D component may change state as many as seven times per PN code cycle. Seven word detectors connected to the PN generator serve to provide bit- and word-sync to the spacecraft data handling system.

A block diagram of the single channel telemetry detector appears in Figure 4-106. The detector is a fully automatic self-synchronizing system and may be broken down into four major areas:

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1) $\oplus$ = modulo two addition and is equivalent to multiplication in the analog domain.
D = data

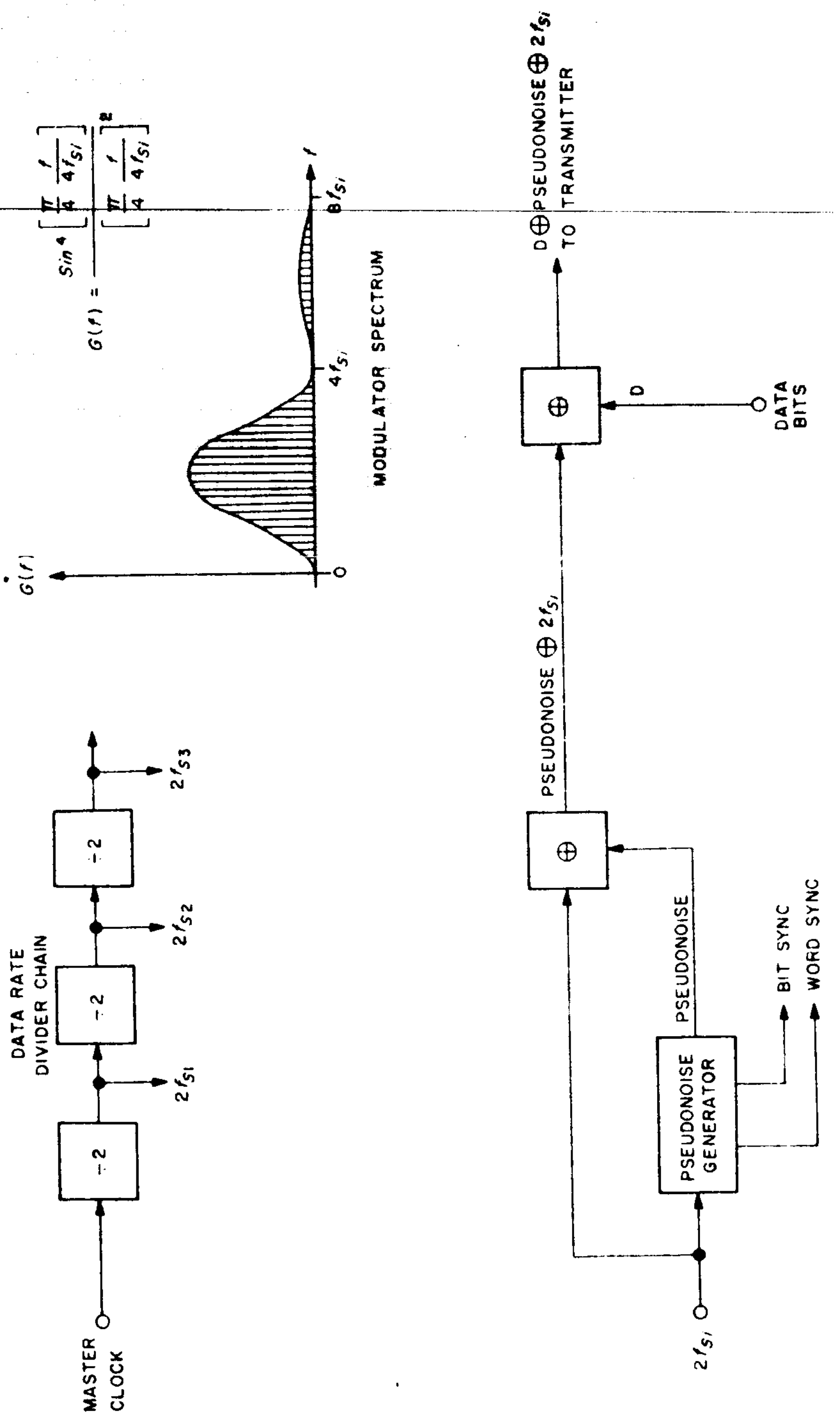

Figure 4-105. Single Channel Telemetry Modulator

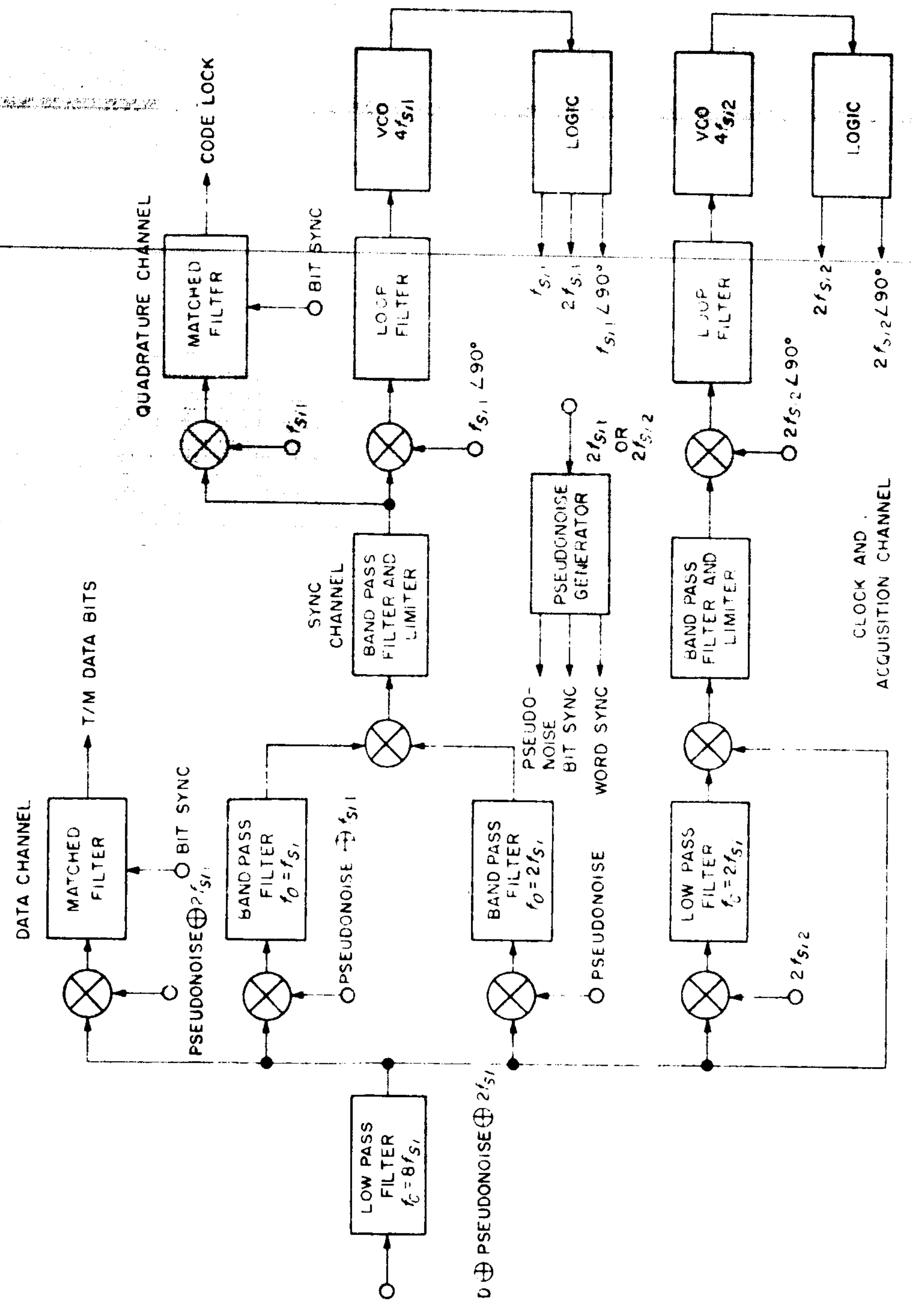

Figure 4-10c. Single Channel T/M Detector

1) the data detection channel,
2) automatic clock and code acquisition loop,
3) sync loop, and
4) the quadrature detector.

The automatic acquisition loop serves to search for the data rate being transmitted and to lock onto the PN clock frequency, $2f_s$. Once this is accomplished, the detector PN code is digitally stepped at the word rate until the quadrature-channel matched filter indicates code correlation. The maximum acquisition time (in seconds) for a telemetry data rate of R bps, ~~is 441/R. This acquisition time is for data threshold conditions.~~ Data threshold is defined as a bit error probability (P_e^b) of 5×10^{-3}. Since the integration period in the acquisition loop is seven times the integration period in the data channel, the probability of not acquiring the first time is 5×10^{-6}.

a. Performance

With a square-wave type signal being transmitted and a low-pass filter at the detector input with cut-off frequency 36 R cps, data channel losses of 0.5 db are normal. The required ST/N/B is therefore 5.7 db for a bit error probability of 5×10^{-3}.

b. Problem Areas

No major problem exists for the spacecraft-to-Earth telemetry modulation and detection subsystems, because the techniques involved represent the current state-of-the-art and have already been successfully implemented for the advanced Mariner-type spacecraft.

c. Options I and II

The telemetry modulation and demodulation system would be the same if either Option I or Option II were selected for the capsule-to-orbiter link.

4. Command Subsystem[1)]

a. Command Requirements

The following table lists the present estimate of the number and type of commands required for the orbiter subsystems.

[1)] For more information, see TR 32-314. Command Techniques for the Remote Control of Interplanetary Spacecraft by J. C. Springett, August 1, 1962.

Subsystem	Discrete Commands		Quantitative Commands
Radio	30		
Data Handling	40		10
Science	80		20
Attitude Control	30		20
SOCC	15		1
Power	20		
Pyrotechnics	3		
Capsule	19		
Totals	237	+	51 = 288

These requirements are considered to be an approximate upper bound with no contingencies, and are based upon studies conducted by the Telecommunications Division. Commands for control of redundant elements are included.

b. Command Link Design - Option I

Earth-to-Spacecraft Command Link

(1) Functional Description. Figure 4-107 shows a general block diagram of the Earth-to-spacecraft command modulation, detection, and decoding subsystems.

The Command Verification Equipment (CVE) functions to process and check the commands received at the DSIF station via the teletype line. In addition, the CVE initiates and monitors all command transmissions and will terminate any transmission that is determined to be in error.[1)]

The command modulator functions to generate all synchronization signals for the CVE, and to encode the command together with synchronization for subsequent modulation of the transmitter.

The spacecraft command detector locks to the received synchronization signals, and demodulates and detects the command. The detector also provides the command decoders with an in-lock signal so that the decoders will recognize only valid command signals.

[1)] For further information refer to JPL Spec 30715 B.

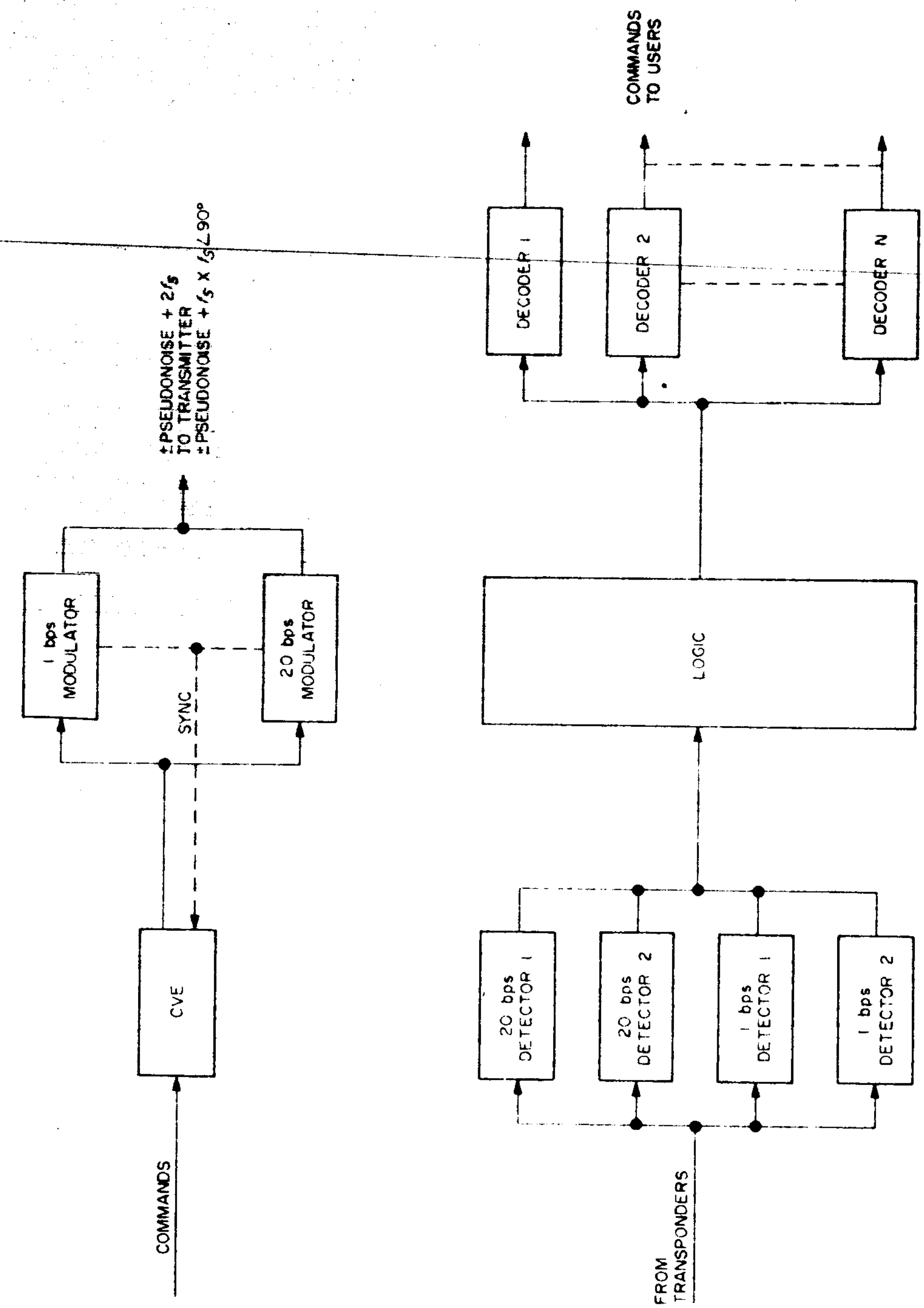

Figure 4-107. Bus Subsystems

The command decoders recognize their respective commands and initiate the desired action.

(2) Command Word Structure. Each command word may be broken down into its word sync, address, and magnitude. The word-sync code will consist of a 7 bit maximal-length shift-register sequence. The command address is 11 bits in length including parity. Command magnitude consists of 40 bits which will be transmitted as zeros for discrete (switching) commands.

(3) Command Data Rates. Two data rates, 1 bps and 20 bps, are recommended. The 1 bps will be used at planetary encounter and during periods when the 20 bps rate cannot be used because of low signal-to-noise conditions. The 1 bps rate is intended primarily for use with the low-gain antenna system.

The 20 bps system avoids some of the practical problems, such as building filters at low frequencies, that exist in a 1 bps system.

The 20 bps rate is intended for most nonencounter command functions such as mid-course maneuvers and cruise mode changes. In addition, this rate may be employed at, or near, planet encounter when the high-gain antenna system is available. The 20 bps system will also be valuable because of its fast acquisition time (1.6 sec or less) for commanding such events as Sun and Earth acquisition. During periods when the spacecraft is tumbling, or for emergency use, fast acquisition and rapid command reception is attractive.

(4) Command Modulation and Detection Systems. The modulation and detection systems for both data rates will make use of single channel techniques with automatic acquisition. Block diagrams of a typical modulator and detector appear in Figures 4-108 and 4-109. The modulator output consists of either $\pm PN \oplus f_s \times f_s \angle 90°$[1] or $\pm PN \times f_s \angle 90°$, where $\pm$ indicates the command modulation. The two different code formats, $PN \oplus f_s \times f_s \angle 90°$ and $PN \times f_s \angle 90°$, are used for address of redundant detectors operating at the same data rate.

The detector may be broken down into four major areas:

1) the command channel,
2) the quadrature or in-lock channel,
3) the sync loop, and
4) the automatic acquisition loop.

[1] x denotes multiplication

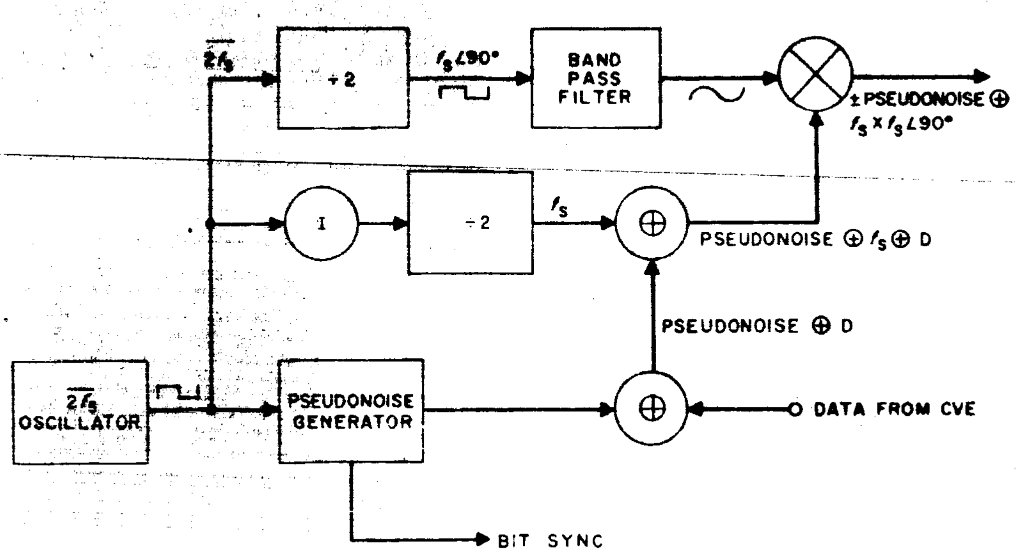

Figure 4-108. Typical Single Channel Modulator

It should be noted that automatic acquisition is not strictly required, but it does provide the fastest means of acquiring under threshold conditions. Maximum acquisition times for the 1 bps system are on the order of 8 to 9 minutes, and for the 20 bps system, 1.6 sec. Maximum acquisition time without automatic acquisition would be approximately 25 percent longer than the time stated for automatic acquisition.

Each subsystem may contain redundant decoders, and the selection of these redundant decoders would require only switching of the input signals. If, instead, two redundant central decoders were used, the output of each isolated command switch would need to be combined in some manner. The later is a much more complicated task. Each subsystem decoder will receive from the detectors the command format, bit sync, and in-lock signals by means of isolated solid-state switches.

(5) Spacecraft Redundancy. It is proposed that the detection system consist of four detectors, two 20 bps detectors, and two 1 bps detectors. The inputs of all detectors will essentially be tied together and connected to the RF receivers through solid-state

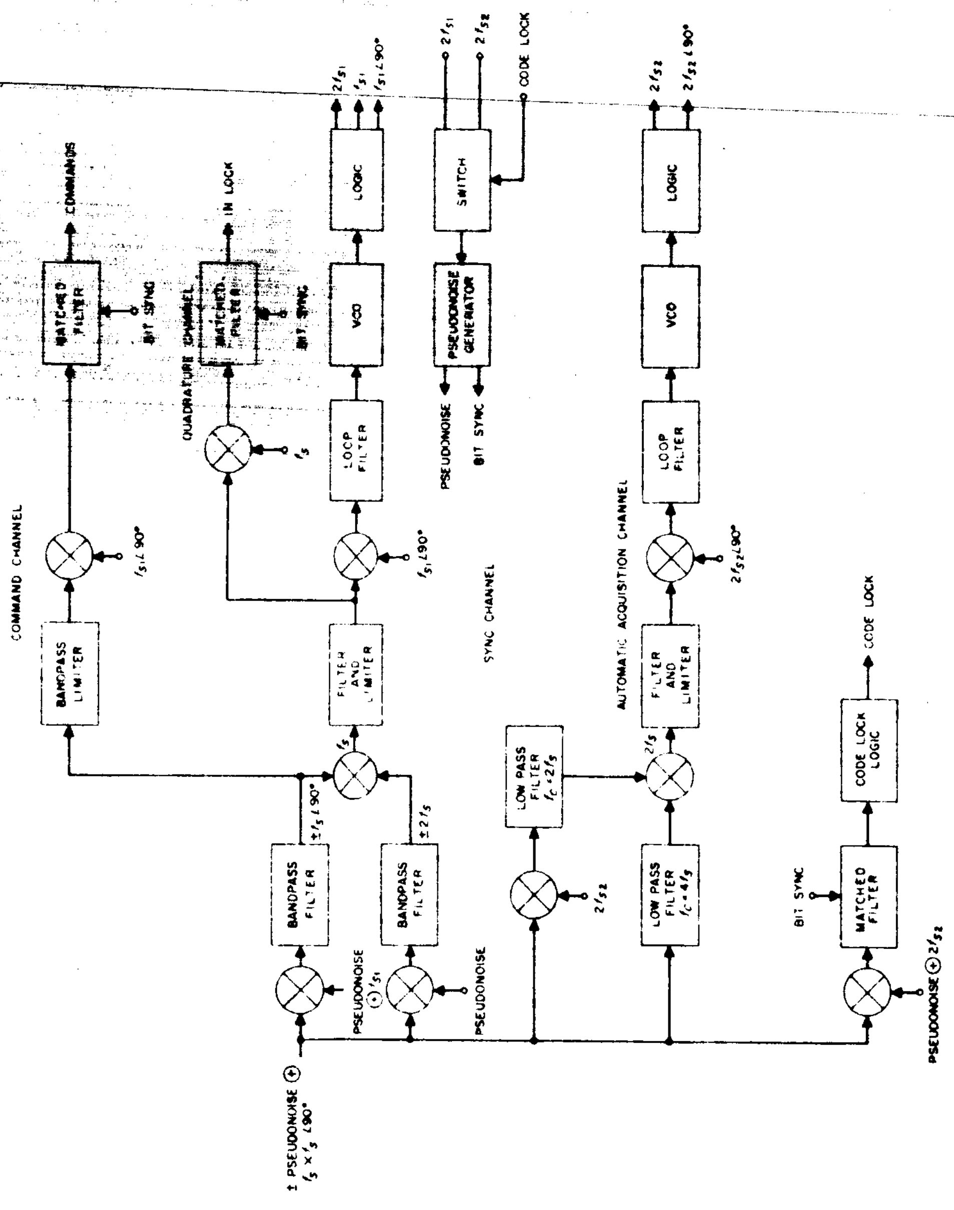

Figure 4-101. Single Channel Detector

switches. The outputs of the detectors will be connected to each subsystem decoder(s) through a logic and isolation network, a simplified version of which appears in Figure 4-110.

(6) **Modulation-Detection Parameters.** The following numbers represent the state-of-the-art in modulation and detection (exclusive of the RF link) and is considered reasonable for 1969 flights, with $P_e^b = 1 \times 10^{-5}$.

1) 1 bps automatic acquisition
   ST/N/B = 11.0 db
   Sync loop $2B_{LO}$ = 0.5 cps
   Acquisition loop $2B_{LO}$ = 0.5 cps
2) 1 bps no automatic acquisition
   ST/N/B = 13.0 db
   Sync loop $2B_{LO}$ = 2.0 cps
3) 20 bps automatic acquisition
   ST/N/B = 10.1 db
   Sync loop $2B_{LO}$ = 3 cps
   Acquisition loop $2B_{LO}$ = 3 cps
4) 20 bps no automatic acquisition
   ST/N/B = 10.1 db
   Sync loop $2B_{LO}$ = 6 cps

c. Command Link Design - Option II

Earth-to-Spacecraft Command Link

The Earth-to-space command system for Option II is the same as that for Option I, with the only possible addition being a subsystem decoder for the capsule commands.

4. Capsule-Orbiter Functions

a. Antenna Configuration

(1) Capsule VHF Antenna. The capsule antenna is required to radiate 25 w at 200 mc and be right-hand circularly polarized. These requirements can be met with several types of structures such as crossed slots or turnstile antennas. The slot antenna is favored because all the structure of the antenna is behind the antenna surface or aperture, thus protecting the sensitive antenna components from the unknown problems associated with landing a capsule on a planet. The turnstile antenna components are exposed and are, therefore, vulnerable if the capsule should roll on landing. A reinforced plastic dome could be used to protect the turnstile; however, the additional weight associated with the dome does

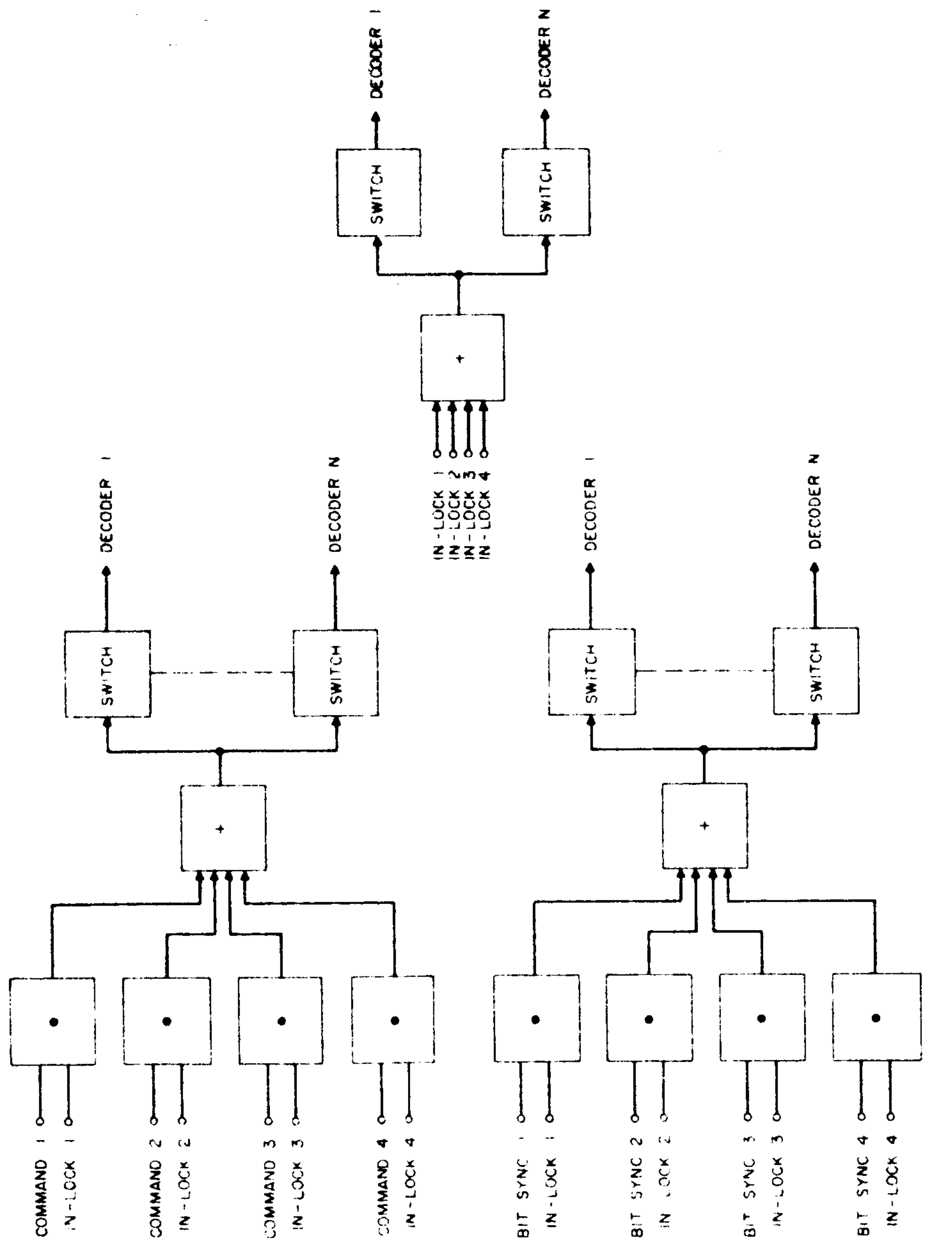

Figure 4-10. Isolation Network Logic

not make it competitive with the slot antenna. Also, a dome would interfere with other capsule items such as the parachute. The length of each of the crossed slots would be 2.5 ft, their depth would be 1.25 ft, and their width approximately 0.5 ft. Reduction on the depth of the slots detunes the antenna. The antenna could be retuned using reactive elements added to the antenna; however, this would:

1) raise the Q of the antenna and make it more susceptible to voltage breakdown during the planetary entry phase,
2) reduce the antenna bandwidth which can cause additional design problems if a two-way communication system is used, and
3) make the antenna more sensitive to damage as any damage that might be sustained by the antenna when the capsule is landed can upset the delicate antenna tuning; therefore, it is recommended that the antenna slot-depth not be reduced.

An antenna designed to withstand the landing loads to be encountered would weigh approximately 15 lb; if a soft, stabilized landing could be assured, this weight could be reduced. The weight of coaxial cables would be 0.3 lb.

It might appear advantageous to use one simple light antenna during the capsule landing phase and a second erectible antenna after landing. Reliability problems arise in using the latter system because

1) automatic switching is required between the two antennas at the proper time, which complicates a system which is, of necessity, already complicated,
2) the first antenna must be separated when it is no longer needed so that it does not interfere with the second antenna, and
3) an erectible antenna mechanism must be designed that could not be damaged in any way during landing so as to prevent deployment of the antenna.

If any of the logic that controls the antenna switching should fail to operate or if the antenna failed to deploy, there would be no way to receive a backup command to correct the problem. The antenna that is nonoperable is the one that is required to receive the command. It is not implied that a deployable system could or should not be built; it is only being pointed out that the simplicity of using one nonerectable antenna throughout the mission should materially increase the overall reliability.

The pattern coverage that can be provided is basically hemispherical; this will generally meet the communication needs during the transit phase of the capsule flight and after it has landed. Since the material which would enclose the antenna to protect it from damage during landing would absorb some of the radiated energy, only 2 db on antenna gain on-axis can be realized. This gain would drop off to -8 db at 70 deg off the antenna axis, where 70 deg is considered the maximum usable angle. This is because the communication capability is being designed around the possibility that the local horizon for the capsule on Mars could be as high as 30 deg due to the terrain. Antenna axial ratios will vary from 1 db on-axis to 10 db at 70 deg.

If Option II is used, the performance of the antenna for a received command or beacon signal depends on the frequency selected for that link. However, the performance for reception should not be lower than 2 db from the transmitter performance, and more likely within 1 db. Cable losses would be 0.3 db.

The low-gain performance is based on the fact that some sort of energy absorbing protective material will have to be placed over the antenna to protect it from being damaged when the capsule impacts the planet. Although the capsule is intended to land right-side-up with the antenna on top of the capsule, there is always a possibility the capsule will roll due to cross winds or landing on an irregular surface. Any protective material will absorb radio energy. A method to regain the performance inherent in the antenna is to devise a system to jettison the protective covering once the capsule comes to rest. The performance before landing would be as specified above; after landing the performance would be +7 db on-axis and -2 db at 70 deg, with the same axial ratios as above. The antenna should have an unobstructed location on top of the lander. Although the slots are only 2.5-ft long, the surface in which they are mounted should span the complete top surface of the lander, with no obstructions above this surface. This requirement is necessary if the antenna is to operate at its highest efficiency. Lowering the efficiency requires either higher transmitter power or lower data rate. However, the following exceptions could be tolerated:

1) A retro motor mounted over the antenna would be acceptable if, after the motor is used, it is jettisoned along with its support structure, leaving the antenna unobstructed as required above. The reason that is acceptable is that, before the motor is used, the distance between the capsule and orbiter is so short that the communication performance margin is more than enough to cover any antenna losses that would result.
2) A parachute package on top of the capsule would be acceptable if the package were toroidal so as to not cover the slots and if any packaging used to package the parachute were nonmetallic. Once the parachute is inflated, the antenna can radiate easily through it, if the shrouds are fabric. This is acceptable, if after the parachute is ejected, the top surface of the capsule remains free of obstructions.

(2) Orbiter VHF Antenna. A 200 mc right-hand circularly polarized antenna is to be flown on the orbiter. Since a palnetary horizontal platform (PHP) will probably be

used and will be pointed at the center of Mars after capsule separation, the antenna will be located on this platform. With the platform pointed at the center of Mars and the orbiter at its closest approach to Mars, the Mars horizon subtends an angle from the orbiter of approximately ±47 deg. This means that as the orbiter appears on the horizon for the capsule, the signal from the capsule will appear 47° off the axis of the orbiter-borne VHF antenna. A minimum antenna gain of +3 db to right-hand circular polarization with a maximum axial ratio of 5 db can be maintained over this angle. The approach recommended is to use a turnstile above a ground plane, as this would provide the lightest configuration and could be built to withstand the loads that it would be subjected to on the orbiter. The length of the elements would be 2.5-ft, located 1 ft above a ground plane of a minimum diameter of 3 ft. The antenna would weigh 3.0 lb. A coaxial rotary joint will be required and could be built for 2.0 lb and have an insertion loss of 0.1 db. Transmission lines associated with this system would weigh 0.4 lb and have an insertion loss of 0.5 db.

b. Radio Subsystem

(1) Relay-Link Transmitter.

(a) Choice of Frequency. Since communication over the relay link is between two nondirectional antennas, attenuation between the antennas increases as the square of the frequency. Thus, for a fixed capability (constant bit rate), the transmitted power should increase as f^2. On the other hand, antenna size and weight decrease as frequency increases. A parametric study (not included here) of communication system weight (transmitter, power and antenna) vs frequency has shown that a broad minimum occurs in the 175 to 200 Mc region. The assumptions used to obtain this conclusion were:

1) A base power of 25 w at 200 Mc.
2) Transmitter prime power supplied by a radioisotope power system characterized by a specific energy of 1.8 w/lb.
3) Power conversion (DC-DC) efficiency of 80 percent.

System weight remains fairly constant in the 175 to 200 Mc range. Antenna considerations of reliability and size point toward 200 Mc. For the transmitter, the case is not as clear. For vacuum tube transmitters, efficiency variations are negligible in the 175 to 200 Mc region. Solid-state transmitter efficiencies decrease slightly with increasing frequency. Thus, the recommended relay link frequency is 200 Mc.

The first choice of configuration is a solid-state exciter which contains the basic crystal-controlled oscillator and a phase modulator followed by a vacuum-tube cavity power-amplifier (see Figure 4-111).

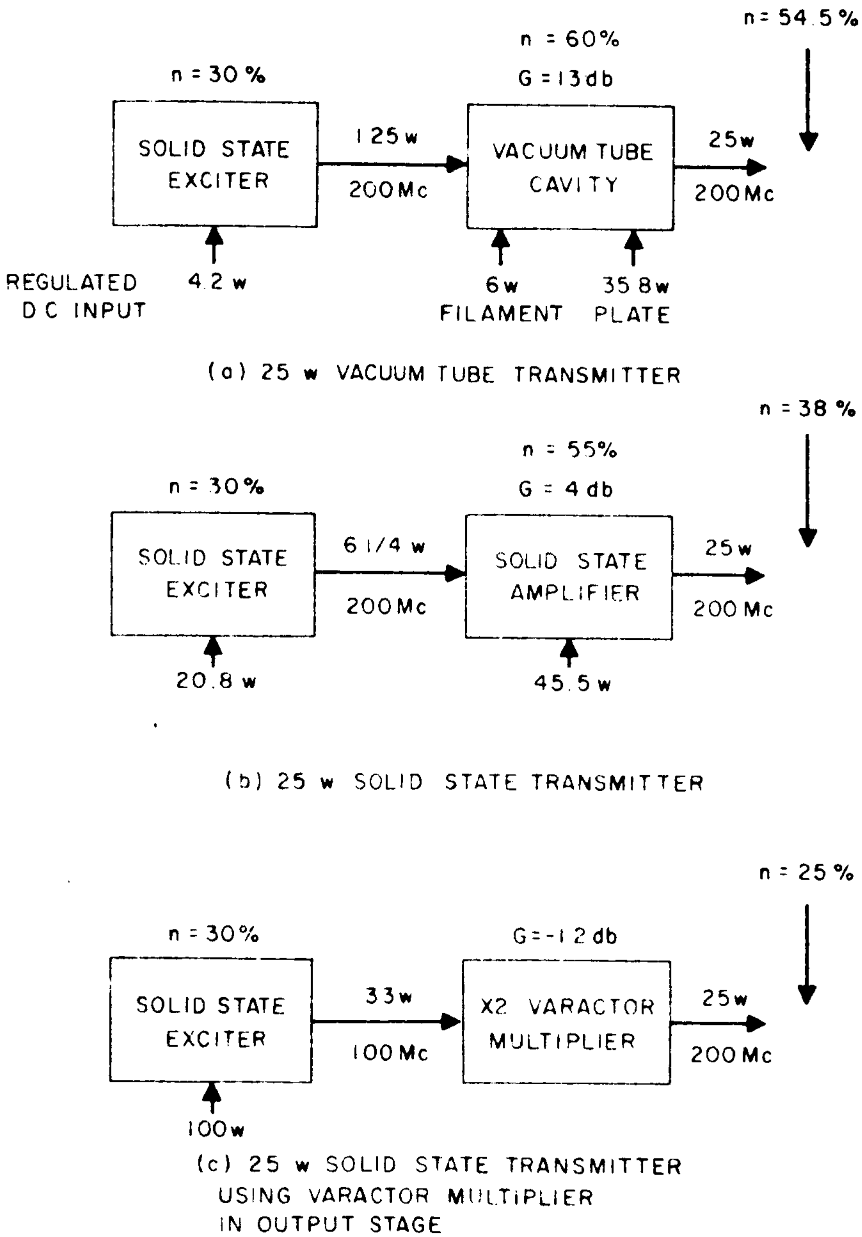

Figure 4-111. Ladder 200 Mc Transmitters

At the present time, flight exciter units are available that are capable of outputs up to 8 w at frequencies up to 160 Mc at an overall efficiency of 32 percent (ratio of RF output power to regulated DC input power). Future developments may result in a maximum of 10 w at 250 Mc with up to 30 percent efficiency.

Also available at present are vacuum-tube cavity power-amplifiers which have gains near 15 db with anode efficiencies of 70 percent, and require 6 w of filament power. Thus the overall efficiency of the first choice transmitter is 54.5 percent.

Two solid-state alternates could be considered. The first, shown in Figure 4-111F, contains an exciter to generate 6.25 w at 200 Mc. Following this is an amplifier that is 55 percent efficient, but has only a 6 db gain. Thus, the overall transmitter efficiency is 38 percent.

The second alternate, shown in Figure 4-111C, contains an exciter which generates 33 w at 100 Mc. An X2 varactor multiplier provides the 25 w at 200 Mc with a loss of 1.2 db. Overall efficiency is 25 percent.

These proposed alternate transmitters depend on advances in high-frequency, high-power transitors and diodes, but nonetheless seem reasonable for 1969 flights. The state-of-the-art in solid state transmitters is advancing rapidly. In sight are 10 w, 250 Mc devices with efficiencies and gains almost equal to those of vacuum tubes. It seems likely that for a 1969 Mission, solid state devices will be available in the 25 w, 200 Mc region with gains and efficiencies near those of vacuum tubes.

From a shock-vibration standpoint (less than 200 g) no one choice seems to have an edge. There is, however, the atmospheric breakdown problem to be considered in the Martian atmosphere. If an Earth-like atmosphere is considered, it is felt that the 25 w RF power level presents no great problem whereas a 50 w level would. It is the high DC voltages required for the vacuum tube (400-500 v) which would present a problem. On Earth, at the critical pressure, 180 v has been considered a 'critical' DC voltage level. During entry on Mars, either a decrease in DC voltage during the critical period or employment of a satisfactory insulation material on all high voltage surfaces would be required. Present state-of-the-art leaves much to be desired in the field of low loss dielectrics at RF frequencies. Because of this, solid state transmitters may be the only choice, unless we are willing to give up transmission during the critical entry phase.

(2) Relay-Link Receiver. At present a phase-lock-loop receiver with automatic acquisition is being considered. The maximum relative acceleration between orbiter and capsule, which occurs during the deceleration of the capsule entry phase is 1.3 km/sec^2. An RF loop noise-bandwidth ($2 B_{LO}$) of 77 cps is thus required to keep phase

error below 30 deg at data threshold (6 db above RF threshold). This bandwidth results in an acquisition time (the time required to sweep the VCO over the entire range of transmitter frequency uncertainty with a 90 percent probability of attaining lock on the first sweep) of 22 sec.

Since sideband frequencies should not occur within 20 Kc of the carrier frequency for the proposed acquisition method, an IF pre-detection bandwidth in the order of 50 to 60 Kc is required. If the IF filter bandwidth requirements become too great, the information can be extracted previous to the IF filter; a separate phase detector would be required, similar to the method used for ranging detection.

The performance of an FM-with-AFC receiver should be compared to that of a phase-lock-loop receiver before the orbiter receiver type is selected. The use of an FM type receiver may simplify the acquisition process.

(3) Beacon Transmitter. The beacon transmitter will operate at 200 Mc at the 5 w output level. An all solid-state unit similar to the low frequency portion of the S-band exciter will be used. A block diagram of a typical transmitter is shown in Figure 4-112.

(4) Beacon Receiver. It has not yet been decided what form the beacon receiver will take. A simple design would be able to detect the presence of a single carrier and could be a phase-lock-loop receiver or a wide-band fixed-tuned receiver, the latter requiring a higher signal-to-noise ratio at threshold.

If two or more commands are desired via the beacon link, an audio tone filter-detector can be used with either of the above mentioned receivers, and corresponding tone generators can be added to the beacon transmitter. Block diagrams of both the fixed-tuned and phase-locked receivers are shown in Figure 4-113.

c. Telemetry Modulation Subsystem

(1) Relay Link. The capsule relay link poses a considerably different design problem than does the spacecraft-to-Earth telemetry link because of the following limitations:

1) All sidebands within ±20 Kc of the RF carrier frequency must be down at least 60 db relative to the RF carrier so that the RF swept frequency acquisition system does not accidentally lock-on to a sideband.
2) Detector lock must be virtually instantaneous so that a minimal amount of data is lost during blackout or low signal periods.

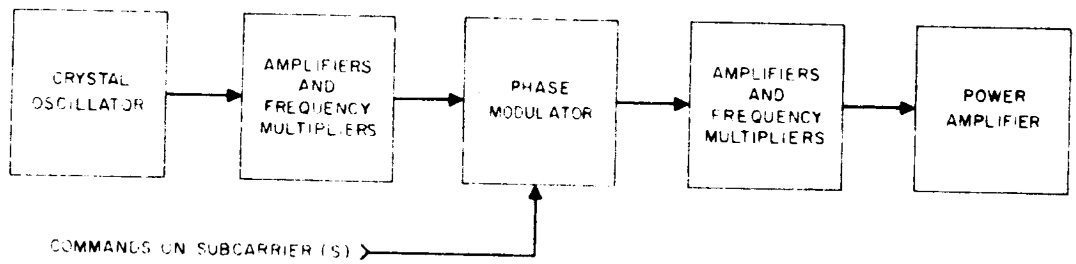

Figure 4-112. Orbiter Beacon Transmitter

3) No direct control over the detector can be exercised to aid acquisition.
4) The modulation and detection subsystems should be as simple as possible.

Two possible avenues of approach for the relay link would be (1) a feedthrough system where the output of the orbiter's relay receiver is used directly to modulate the orbiter transmitter, or (2) a system where bit-by-bit detection in the orbiter is employed to recover capsule data which is subsequently processed by the orbiter data handling system for transmission to Earth. Complete detection systems will be discussed first.

(a) Modulation-Detection System No. 1. An efficient bit-by-bit detection system is required so that a maximum relay data rate can be realized. This requirement suggests a coherent system with matched filter detection which in turn requires both a coherent reference and bit sync within the detector.

Because of the above limitations, the techniques employed for the spacecraft-to-Earth telemetry link cannot be employed.

Another approach would be to employ one of the widely known bit-synchronization systems. These systems essentially make use of two phase-lock loops -- one to provide a coherent subcarrier detection reference, and the second, in conjunction with a zero-crossing detector, to provide bit sync. A major drawback here is the rather complex bit-sync loop mechanization (primarily the zero-crossing detector) and the large number of components that have to be switched if multiple data rates are required.

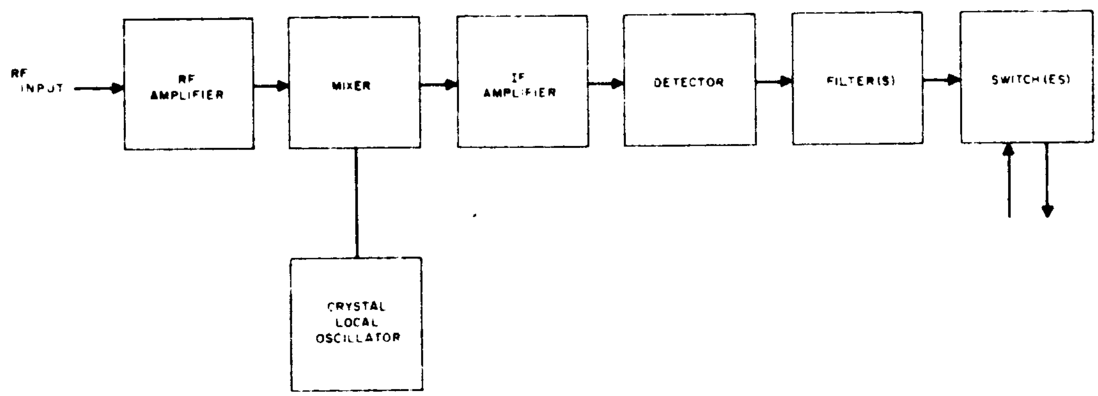

Figure 4-113A. Lander Beacon Receiver, Wide Band Fixed Tuned

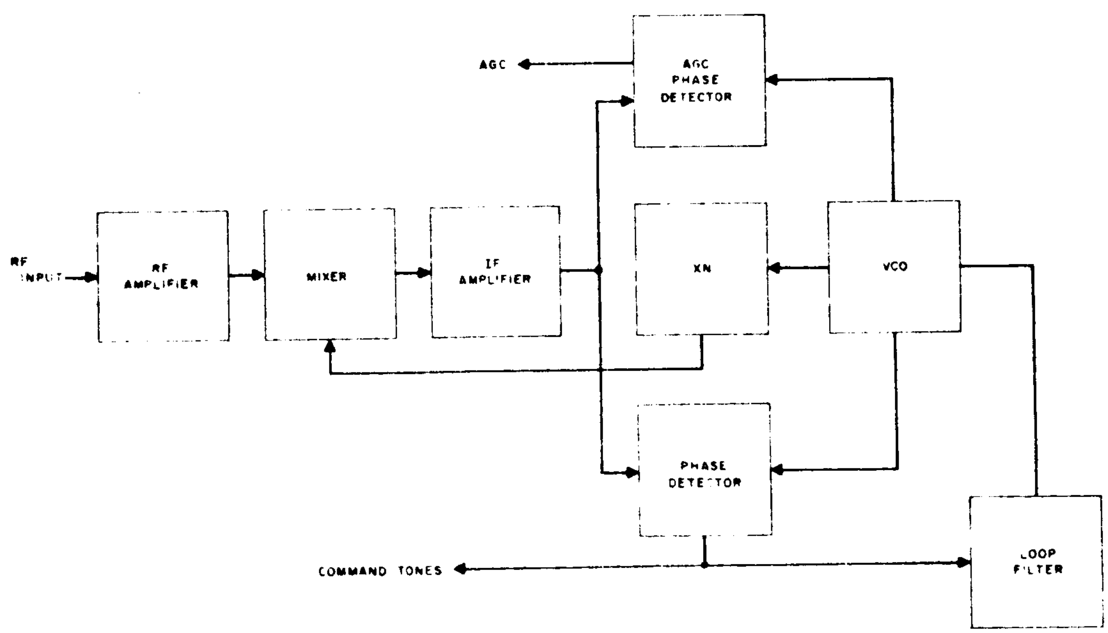

Figure 4-113B. Lander Beacon Receiver, Phase Locked

A somewhat more desirable solution, particularly when data rates over 50 bps are employed (the present situation), is to transmit the bit-sync signal over a separate subcarrier. Consider the modulation and detection system shown in Figures 4-114 and 4-115. An oscillator provides a frequency of $2f_o$ which becomes the subcarrier for the data bits. The frequency, $2f_o$, is divided by two to obtain a second subcarrier frequency, f_o, for the bit-sync information; f_o is further divided by some number, N, to obtain the bit sync clock. Bit-sync divided by two rather than bit-sync is transmitted in order to resolve the phase ambiguity within the detector. The two modulated subcarriers are linearily mixed in the proper ratio for subsequent modulation of the transmitter. Typical numbers for a 300 bps system are shown in brackets in Figure 4-114.

The detector aboard the orbiter consists of two loops (Figure 4-115) -- the first to obtain a coherent reference by locking to the subcarrier frequency of the squared f_o ($2f_o \angle 90°$), and the second to demodulate and lock to the bit sync. The data-modulated $2f_o$ subcarrier is coherently demodulated, and the noisy bits are detected in a matched filter.

The foregoing system has a number of distinct advantages:

1) The system is completely synchronous.
2) The modulator and detector are simple.
3) The amount of transmitter sideband power required for the bit sync subcarrier is 1/10 or less than that required for the data subcarrier.
4) The system contains no phase ambiguities.
5) Data rate may be easily altered, requiring only that the frequency, fo N, in the modulator and the bit-sync loop VCO frequency in the detector be changed.

The major disadvantage of the system is the rather wide IF bandwidth that must be employed in the RF receiver to pass the two subcarrier frequencies.

(b) Modulation-Detection System No. 2. To circumvent the spectral problems of the foregoing system, an alternate approach would be to use orthogonal subcarriers at the same frequency as shown in Figures 4-116 and 4-117. In order for this system to work, the bit-sync subcarrier power be considerably less than that allocated for the data subcarrier in order to derive a coherent reference through the squaring process within the detector. (If the powers are equal, no reference can be obtained through a single squaring process.) The operation of this system is nearly identical to the foregoing system, with the main difference being in the subcarrier frequencies.

A disadvantage of this system is that the data subcarrier coherent demodulation reference possesses a 180 deg phase ambiguity. This problem may be solved by searching

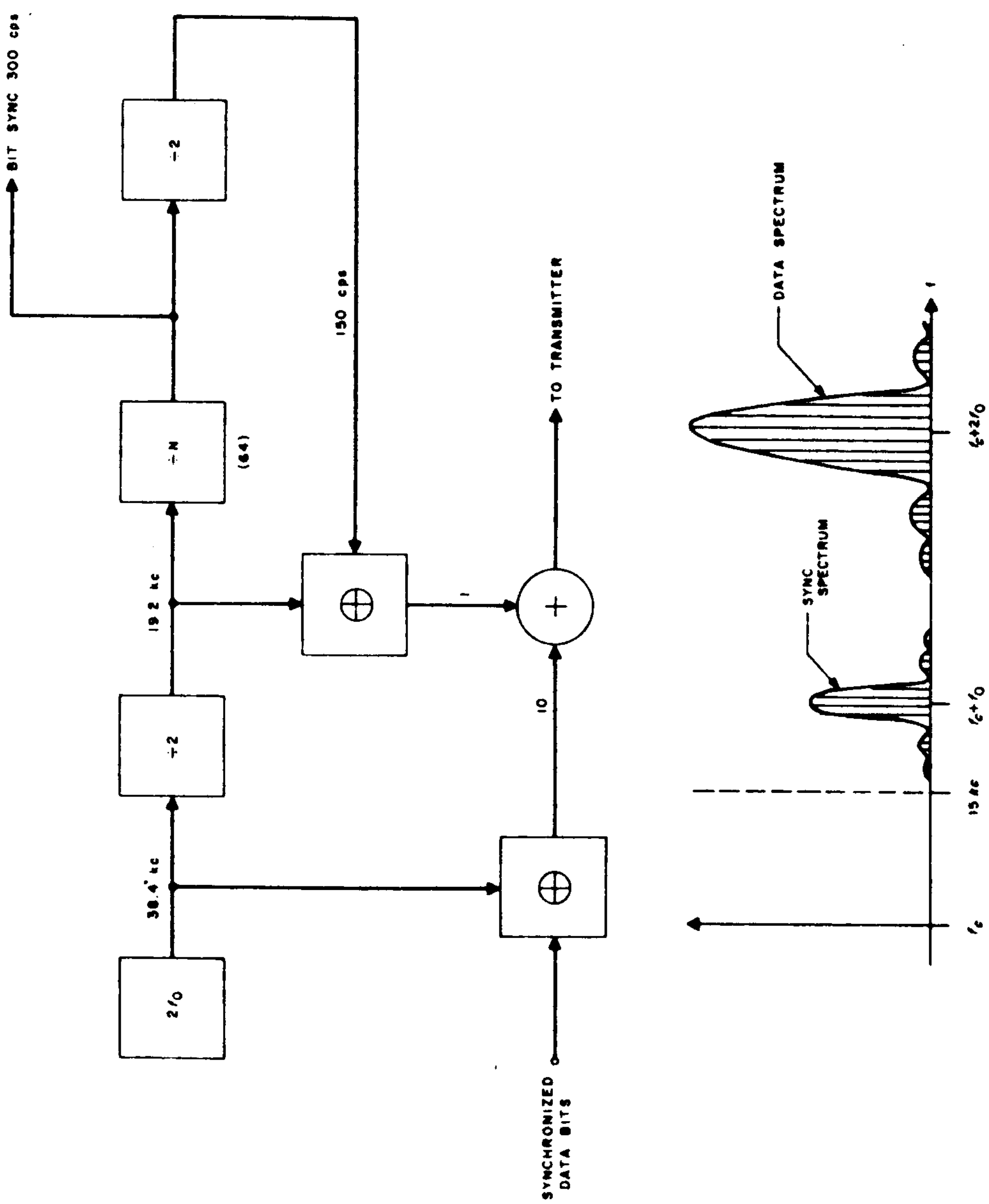

Figure 4-114. Lander Modulator

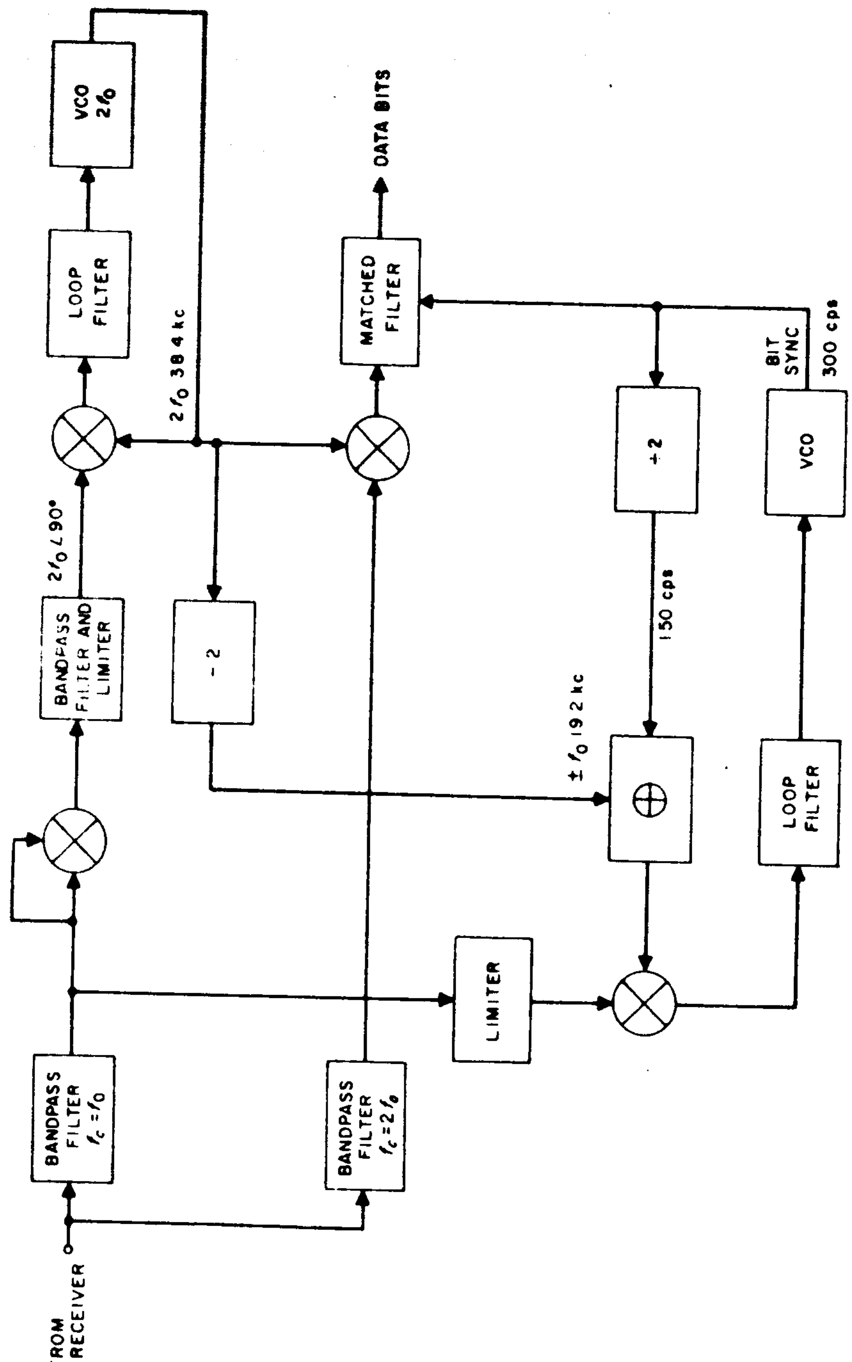

Figure 4-115. Lander Detector

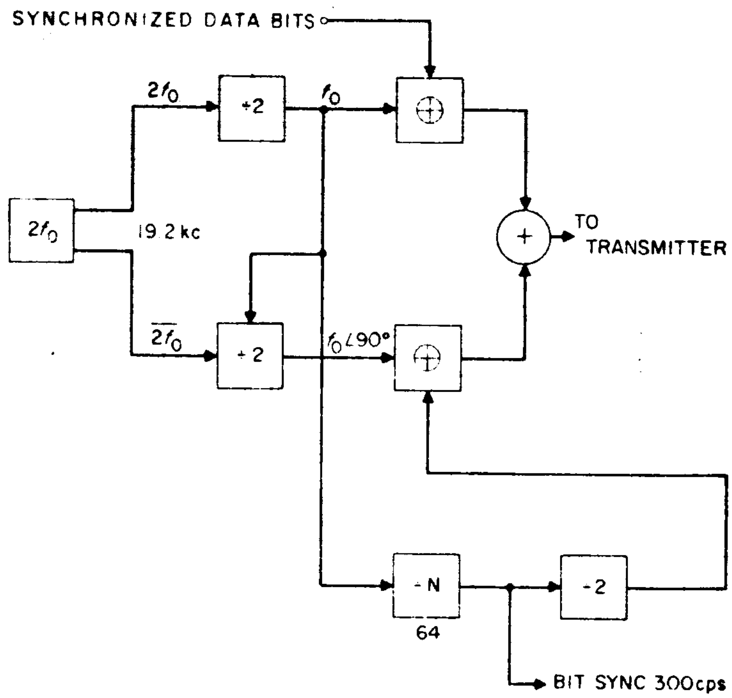

Figure 4-11c. Lander Modulator #2

for the proper phase of word and frame sync-words in the data format on the ground or by using change-of-state data format with a resultant doubling of the bit error probability for a fixed ST/N/B.

(c) Modulation-Detection Performance. Because the subcarriers are square-waves rather than sine-waves (to eliminate the need for bandpass filters in the capsule modulator), an approximate [illegible] db penalty is paid for bandpass filtering in the detector. It is estimated that all other detector losses (exclusive of RF system losses) will not amount to more than [illegible] db, requiring that ST/N/B = [illegible] db for a bit error probability of 5×10^{-3}.

(d) Problem Areas. No major problems are anticipated in the hardware mechanization of either of the two foregoing systems. A potential problem lies in using the same receiver narrow-band IF for data as well as carrier tracking. A wide band IF would be a solution, but this would complicate the receiver design and necessarily affect reliability.

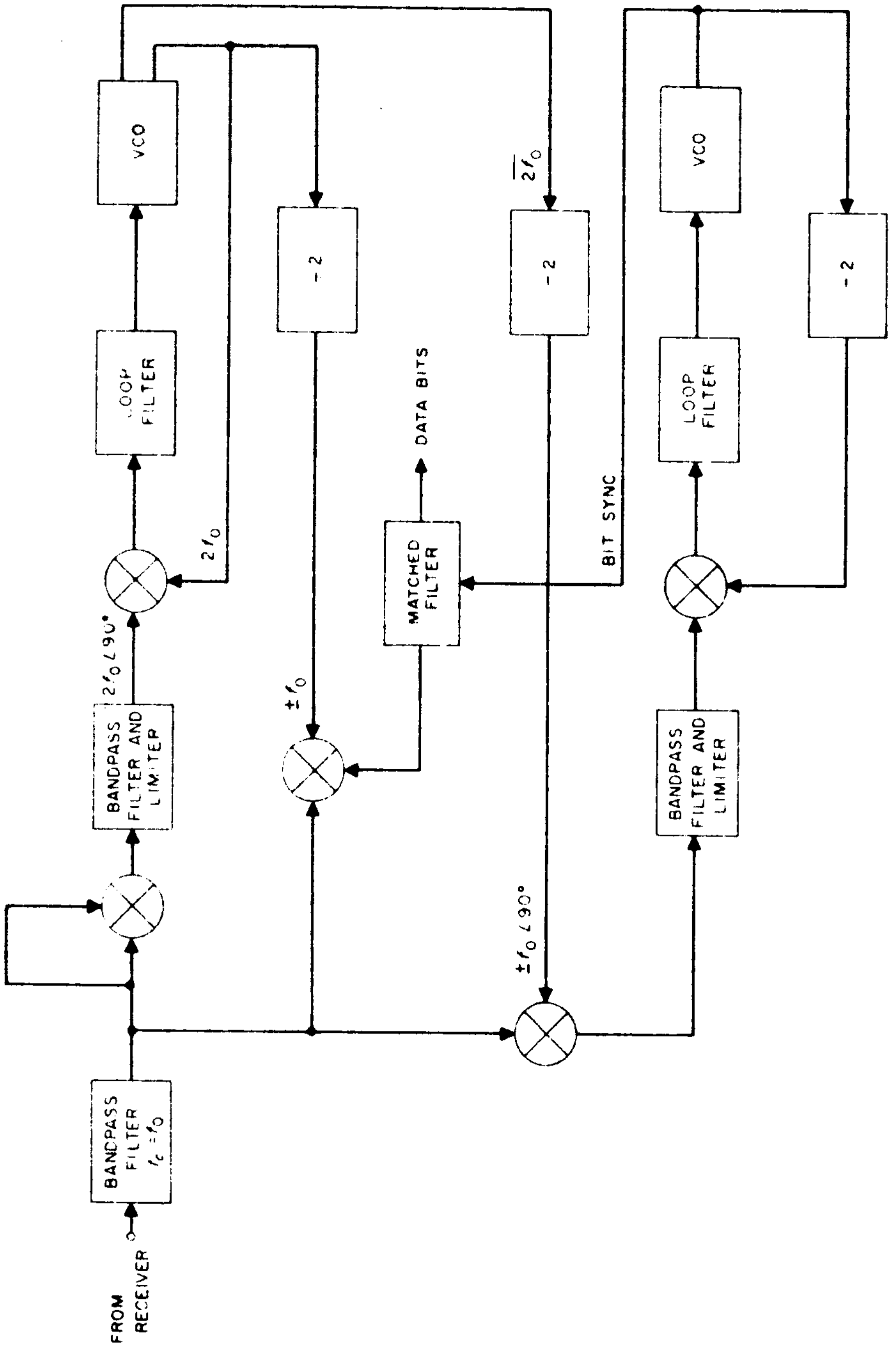

Figure 4-117. Lander Detector #2

(e) The Feed-Through Link System. As mentioned earlier, one of the simplest relay links would recover the subcarrier from the receiver, filter, and then use this signal to modulate the orbiter transmitter. When such a system is implemented, usually one subcarrier is employed so that a limiter may be placed prior to the transmitter modulator in order to control the modulation index. Detection is performed on the ground by means of a bit-synchronizer. One obvious disadvantage of the feed-through system is that the data rate is determined by the worst link; e.g., if the lander-to-orbiter link is normally able to support a higher data rate that the orbiter-to-Earth link, then the latter determines the maximum data rate. Some penalty is also paid for modulating the transmitter of the orbiter-to-Earth link by both signal plus noise, unless the signal-to-noise ratio is quite high. The exact degradation due to the noise-modulation is dependent upon the data rate and is proportional to exp $\left[\sigma n^2\right]$ where σn is the rms value of the noise.

(f) Combined Feed-Through and Bit-Detection Systems. When combined bit-detection and feed-through systems are employed aboard the orbiter for redundancy purposes, a problem arises in the lander modulation process. The bit detection system requires two subcarriers of different amplitudes, while the feed-through link requires a single subcarrier so that a limiter can be employed. The use of two different lander modulators is impractical, since it may be impossible to command the lander as to which one it should be using. As a result, the two subcarrier signals must be used with the feed-through link as well as the bit detection system. This poses a mechanization problem because a limiter cannot now be employed in the feedthrough link without changing the relative subcarrier power ratios. In addition, two orthogonal single frequency subcarrier systems must be employed to circumvent the problem of filtering two different subcarrier frequencies and then recombining them prior to modulating the transmitter.

One possible solution to eliminating the limiter is to provide the system with a fixed maximum gain so that the modulation index, when all noise modulation is present, can be fixed. With such a mechanization, the modulation index when signal is present would probably no longer be optimum. Before a choice can be made a more intensive study (beyond the scope of the present effort) will have to be performed.

(g) Recommandations. Because it appears that the problems of using a feedthrough modulation system are many, and because considerable effort must be expended to resolve the problems, the bit-by-bit detection system is proposed for Voyager use. However, the feedthrough system would be simpler and would not require an acquisition process or the proper functioning of a detector in the orbiter.

Since there would be much to gain by eliminating the detector, it is recommended that further effort be expended to solve the problems associated with the feedthrough modulation system. Additional problems in the mechanization of the data handling subsystem are discussed in Section IV, B (1).

d. Command Subsystem

(1) Command Requirements. Estimates of the capsule command requirements for the various subsystems have been made and are listed below. There is also an alternate method listed for each command in the event that Option I (no external control) is used:

Capsule Subsystem	Number of Commands	Alternate Method
Communications		
S-Band Xmitter Turn-On	1	Sensing landing
Data Rate Adjust	2	None
VHF Xmitter Power Control	2	Timer, sensing entry, and landing
Emergency Data Handling Mode	1	None
Science		
Instrument Control, Emergency Mode	5	Timer, sensing of events; none for emergency mode
Power		
Power Control	2	Timer, sensing of events
Attitude Control and Engineering Mechanics		
De-spin	1	Timer
Parachute Jettison	1	Sensing of landing
All		
Redundancy	4	On-board fault correction
TOTAL	19	

(2) Modulation and Detection System. At the present time there is no requirement for direct command transmission between Earth and the lander; therefore, all commands to the lander from Earth will be relayed via the orbiter.

(a) Functional Description. Figure 4-118 shows a block diagram of the modulation, detection, and decoding subsystem which makes up the lander command link. The modulator consists of a number of crystal-controlled tone generators operating in a frequency range between 15 Kc and 30 Kc which are gated into a summing network in accordance with the commands to be transmitted. Gating functions are derived either from Earth-command or spacecraft events, and are long enough in duration to provide a successful command actuation probability of 0.99999 or better.

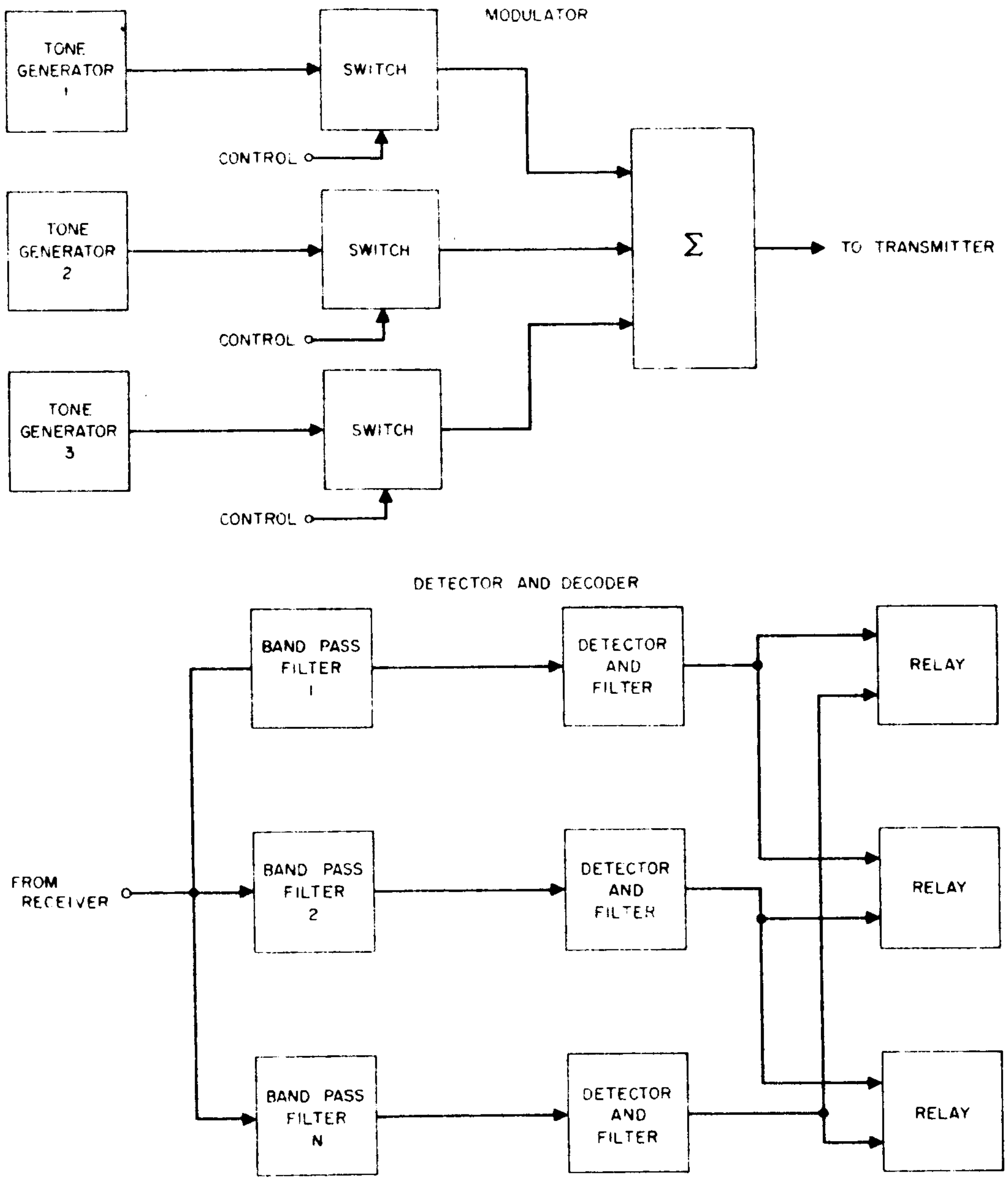

Figure 4-118. Lancer Command System

The lander command detector consists of a series of bandpass filters and detectors, the outputs of which are employed to switch relays or activate solid-state switches.

(b) Required Number of Tones. The following binomial coefficients, $\binom{n}{m}$, represent the number of commands that may be obtained from n tones taken m at a time:

$\binom{n}{1} = n$

$\binom{5}{2} = 10$

$\binom{6}{2} = 15$

$\binom{6}{3} = 20$

$\binom{7}{2} = 21$

It is recommended that 5 tones be used individually for up to 5 commands. For 6 to 10 commands, 5 tones used in pairs are recommended. Use of 6 and 7 tones in pairs is recommended for command requirements between 11 and 15 and between 16 and 21 respectively.

(c) Modulation and Detector Performance. It is anticipated that a signal-to-noise ratio of 19 db in the bandwidth of the detector filters, for a period of one second, will be required. If this figure cannot be met, then longer tone-on times will be required and integrating type detectors can be employed.

(d) Problem Areas. No major problems presently exist for either option.

5. Lander-to-Earth Functions

a. Antenna

For the direct S-Band link a cross-slot radiator would be the most simple. The requirements for this antenna would be similar to those for the VHF antenna. The pattern coverage would be the same since the local horizon on Mars is the controlling factor for the coverage. However, this antenna would be considerably smaller with slot lengths of 2.5 in. Its weight would be 0.5 lb and the associated cable weight would be 0.4 lb. This antenna would be mounted adjacent to the VHF antenna. Cable losses would be 0.5 db.

The direct S-band link provides a minimum capability for transmission of capsule data that does not depend on the proper operation of the orbiter or the primary capsule data transmission system. If the capsule is not orientated right-side-up after landing, then both the VHF relay and S-band antennas' radiation patterns could be drastically affected. Therefore, it might be advantageous to design the S-band atnenna system to operate regardless of the capsule orientation. Two antennas could be used simultaneously, with the second in the capsule base. This would result in a 3 db loss due to power spliting. However, the most serious drawback to this configuration is the interference band produced around the capsule equator and possible degradation of the main antenna pattern for a normally oriented capsule. To circumvent this problem, it might be possible to design a switch which senses the direction of the local vertical and switches the transmitter to the antenna closest to it. This scheme could be expanded to include four antennas, one located on the top of the capsule for operation when the capsule is oreinted properly and three located at 120 degree intervals around the rim of the base of the capsule; i.e., as if the four antennas were located on the verticies of a tetrahedron. The most logical orientation for the capsule is on its base or its side if no effort is made to control its orientation. Thus, the latter system using four antennas is the optimum. The weight of the redundant two-antenna system would be approximately 4-lb and that of the four antenna system would be 6 lb. Each individual antenna would have the coverage as specified for the single antenna system.

It would be desirable to design the VHF antenna system to function in a similar manner if the capsule were not upright. However, while the S-band system can be made redundant for a small weight penalty, the additional weight and volume required for a redundant VHF system is prohibitive and, therefore, is not considered.

b. Radio Subsystem

Since the capsule transmitter for the direct link will be an S-band unit of from 10 to 50 w, the basic unit will be similar to the orbiter-to-Earth transmitter. (See Part III, B, 1)

If the output level exceeds 30 w and transmission is desired before encounter, the multipacting problem should be examined. Methods of avoiding this problem might involve the use of metals with low secondary emission factors or the use of insulating coatings on metals near high RF field points. Since vacuum tubes require 400 to 600 volts and amplitrons require 1800 to 2200 volts, the ionic DC breakdown will be a problem during entry and possibly even at the surface. Many solutions to this problem are possible; for example, all high voltage points could be insulated. Since some of these points will be within tuned cavities, an insulating material that has low loss at S-band must be found.

If breakdown and impact problems become too severe, a solid-state transmitter would be indicated, but present indications are that a solid-state transmitter above 10 w would not be feasible.

If breakdown during entry is the only breakdown to consider, and loss of signal would occur due to blackout anyway, current limiting can be incorporated in power supplies to prevent permanent damage. A second alternative is to simply reduce voltages below the critical voltage and suffer the corresponding loss in communication capability. The performance capability of the capsule-to-Earth link during the entry and landing phases have not been investigated. It may not be desirable to energize this link before landing due to power and trajectory constraints.

c. Telemetry Modulation Subsystem

The modulation and detection system employed for the direct link is dependent upon the available transmitter power and on the data rate desired. As a result, a different system is recommended for each of the following data-rate ranges, in order that an efficient modulation and detection system can be employed:

1) $R \geq 3.0$ bps
2) $0.5 \leq R \leq 3.0$ bps
3) $R < 0.5$ bps

$R \geq 3.0$ bps

For this range of data rates, it is recommended that a single channel system similar to that used for the orbiter-to-Earth telemetry link be employed. Typical single channel mechanization was described previously for the spacecraft-to-Earth telemetry link.

The 3 bps minimum is chosen because below this rate, ST/N/B must be increased in order to meet the required signal-to-noise ratio in the sync-loop noise-bandwidth, $2B_{LO}$, due to the fact that a lower practical limit on $2B_{LO}$ is approximately 0.25 cps.

System performance figures are

$$ST/N/B = 6.2 \text{ db}$$

and

$$2B_{LO} \text{ (max)} = 0.23 R.$$

These numbers do not take into account RF system losses.

$0.5 \leq R < 3.0$ bps

From 0.5 bps to approximately 3.0 bps, a different detection method may be employed to efficiently demodulate the data, assuming again that a phase-lock RF receiver is employed. Figure 4-119 shows the capsule modulator and Figure 4-120 the ground detector.

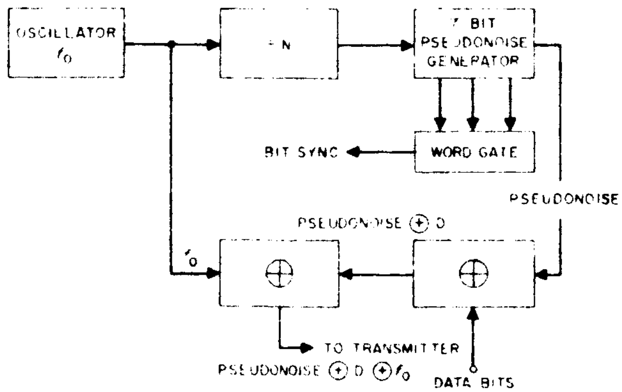

Figure 4-119. Lander Modulator $0.5 \leq R < 3.0$ bps

The modulator operates from a stable clock source, f_o, which provides the data subcarrier, and which is divided by N to obtain the clock frequency for a 7-bit PN code generator which cycles at the data bit-rate. Bit sync is derived by a word gate placed on the PN generator. Data is sent in the form of either + PN or - PN, which modulates the data subcarrier, f_o.

The detector obtains a coherent demodulation reference by filtering and squaring the received data subcarrier, phase locking to the resulting $2f_o \angle 90°$ signal, and dividing by two to obtain $\pm f_o$. Although the reference waveform contains a 180 deg phase ambiguity, this can be resolved through examination of sync words in the data format.

Subsequent to subcarrier demodulation, the transmitted PN code is correlated with 14 possible shifts of the local PN code and 14 matched filters are employed to recover the data. Of all 14 matched filters, one output will be quite close to giving maximum-data channel performance. In the worst case, two of the matched filters will indicate identical performance, which will be 2.5 db from optimum. A method by which the nearest-to-optimum filter could be chosen would be to follow each filter by a full-wave rectifier and low-pass filter combination, and then determine which low pass filter has the greatest average output. Once the best matched filter has been selected, phasing adjustments can be made to obtain optimum performance. All of these operations can be performed automatically if desired.

Generation of the 14 PN code shifts requires that two code generators operating from inverse clock phases be employed, each generator providing 7 shifts through modulo-two adders connected to the generator shift-register stages as follows:

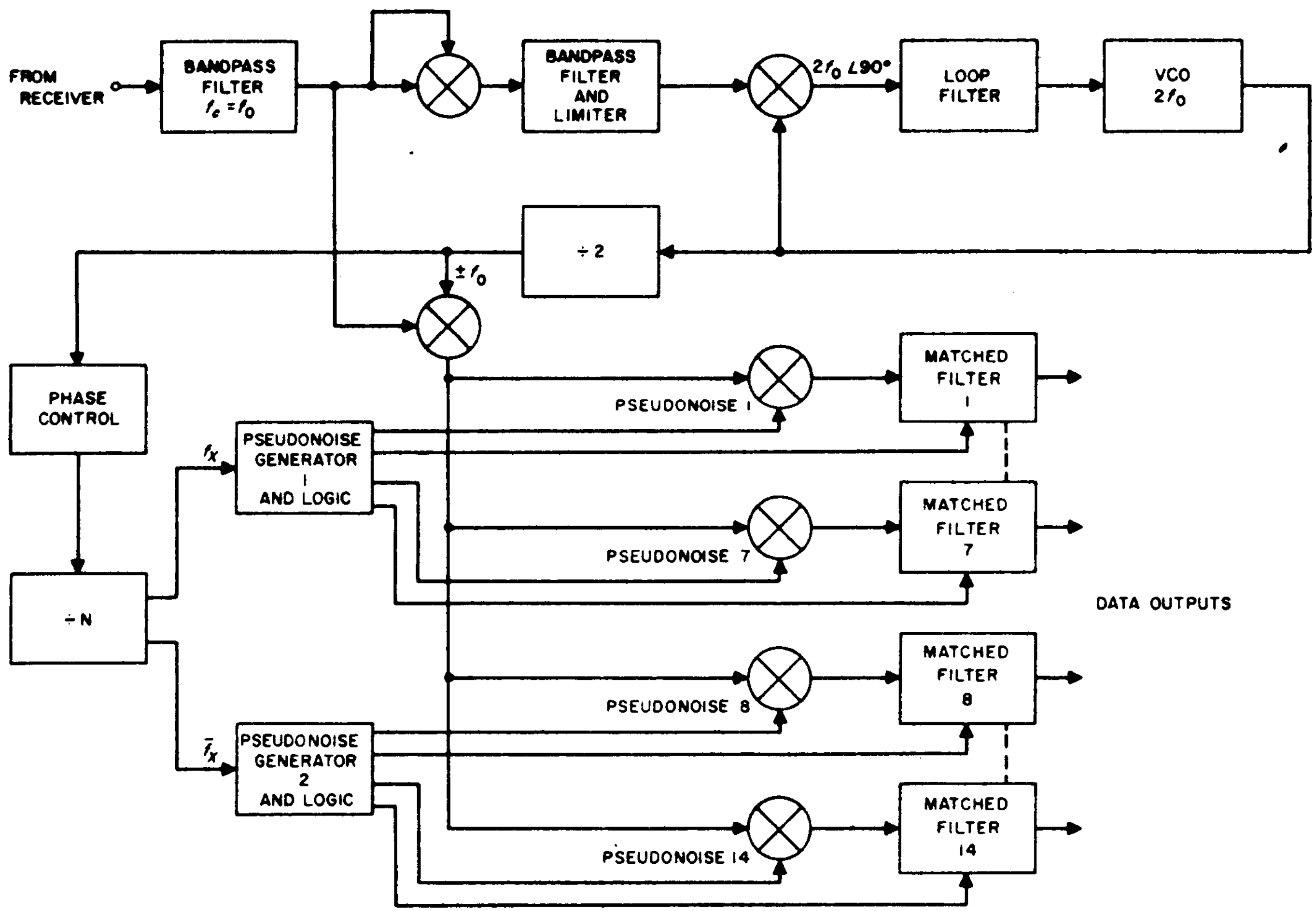

Figure 4-120. Lander Detector $0.5 \leq R < 3.0$ bps

Code Shift	Stage Logic
1110010	Stage 1 only
0111001	Stage 2 only
1011100	Stage 3 only
0101110	Stage 1 ⊕ Stage 3
0010111	Stage 1 ⊕ Stage 2 ⊕ Stage 3
1001011	Stage 1 ⊕ Stage 2
1100101	Stage 2 ⊕ Stage 3

System performance figures, when the local and received PN codes are properly aligned, are:

ST/N/B 6.2 db

and Clock loop noise bandwidth, $2B_{LO}$ (max) 0.25 R. It should be noted that these figures are the same as specified for a single channel ($R \geq 5.0$ bps) system. The reason for this is that lower clock-loop noise bandwidth, $2B_{LO}$, may be used due to not having to acquire a PN code with its associated discontinuous loop-error function. Also, less signal-to-noise loss is obtained in the subcarrier squaring process due to the smaller filter noise-bandwidths that can be employed, resulting in no ST/N/B increase.

$R < 0.5$ bps

The third system considers a method by which extremely low data rates may be realized. The method is based upon the successful techniques that have been employed by JPL for the radar exploration of Venus and other planets.[1)] The basic detection system is shown in Figure 4-121. The binary signal consists of an RF carrier shifted between two or more different frequencies. The use of more than two frequencies is equivalent to coding the channel. The detection system provides a filter for each of the frequencies, followed by an autocorrelator and an integrator. The net effect is to perform a spectral analysis at each of the possible received frequencies to determine which frequency has been received. The accuracy of the method depends upon the integration time and the filter bandwidths employed for a given transmitter power, and the method exhibits a rather sharp threshold.

This system, as used for the planetary experiments, made use of a long integration time. Because experience is lacking with short integration times performance of the system at typical Voyager data rates cannot be accurately predicted at this time.

1) See JPL TR 32-350; Richard M. Goldstein; "Radar Exploration of Venus."

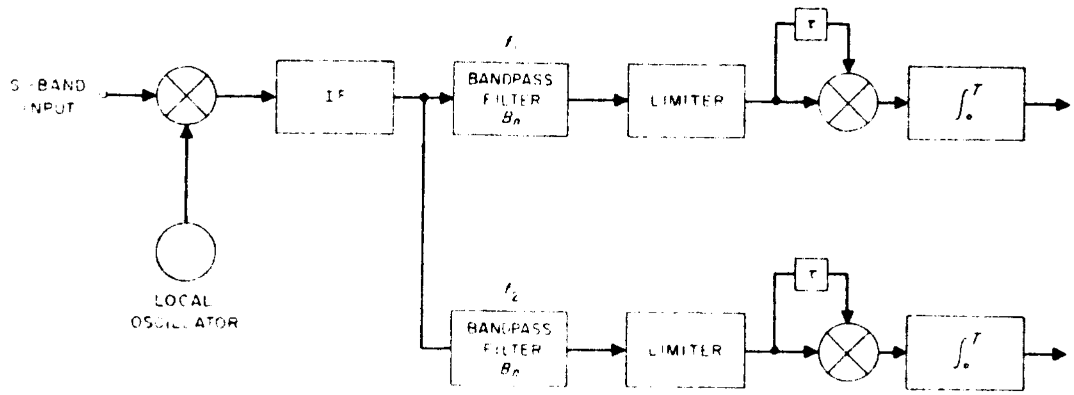

Figure 4-121. Lander Frequency Shift Keyed Detector

Several other variables must also be considered in the system operation. First, all doppler shifts due to planet rotation, etc., must be taken into account in order that the received signal can be in the center of the bandpass filter, B_n. Second, the transmitter frequency drifts in time with temperature, and there must also be compensation to keep the signal in the filter bandpass.

A possible solution would be to use many filters displaced in frequency, which would expand the detector system complexity by at least two times the number of extra filters (plus autocorrelators and integrators) employed. Finally there is the problem of determining when to begin and terminate the integration period (the bit sync problem). This could be solved by using a number of integrators with staggered integration periods during initial acquisition.

It is anticipated at this time that the entire detection system, including the filters, can be constructed utilizing a general purpose digital computer.

6. Performance Capabilities

a. Capsule-to-Bus

The relay data link must function during two entirely different conditions or phases. These are:

Phase I Capsule separation to capsule landing on surface.
Phase II Lander surface operation.

In estimating the performance capabilities of this data link, several assumptions are made. They are:

1) A maximum capsule-to-bus range of 36,000 km based on an encounter separation time, Δt, of two hours as described in Chapter 4, Section II, (C).
2) After separation from the bus, the capsule has some form of attitude stabilization, such that the capsule-to-bus direction is within ±45 deg of the antenna axis up to the time of start of atmospheric entry.
3) After atmospheric entry, to compensate for parachute sway, capsule-to-bus direction is within ±60 deg of the capsule antenna axis.
4) The orbiter receiving antenna is oriented toward the planet center during period from capsule separation through capsule lifetime, so that the capsule-to-orbiter direction is within ±47 deg of the orbiter antenna axis.
5) After landing, lander orientation is maintained, so that its antenna axis is within 10 deg of the local vertical.
6) An elliptic 1800 km by 10,000 km orbit with 3σ dispersions of 500 km is assumed.

(1) Phase I - Capsule Separation to Capsule Landing. Table 4-26 lists the assumed parameters and estimates of tolerances for the communications link design for Phase I. Figure 4-122 shows the available data rate vs capsule-to-bus range for various transmitter powers. It may be noted from Figure 4-122 that a change in transmitter power by a factor of two does not result in a like change in information rate. This is because the transmitter modulation index, or the division of total power into carrier and subcarrier, has a different value for each transmitter power level.

This phase of the capsule operation can be divided into subphases. These are:

1) The high-speed entry into the atmosphere and
2) The parachute phase prior to capsule landing.

These two subphases have not been investigated in detail. During the high-speed entry phase it may not be possible to maintain communications due to the plasma sheath that will very likely surround the capsule. By the time of the mission, technology may exist to solve this problem, but at present, communications during this phase should not be planned. During the parachute phase, oscillations of the capsule may present some unfavorable antenna look-angles. However, the capsule-orbiter range is decreasing and may compensate for unfavorable angles. The net effect of these variables has not been investigated. This is a task for further study since it is quite desireable to have communications prior to impact in order to:

Table 4-26. Communication Parameters, Capsule to Orbiter - Phase I

Parameter	Value	Tolerance	Comment
Transmitting Circuit Loss (Total losses between transmitter and antenna terminals. Includes losses mentioned in antenna discussion.)	0.5 db	± 0.2 db	. . .
Transmitting Antenna Gain	2.0 db	± 2.0 db	. . .
Transmitting Antenna Pointing Loss	4.0 db	± 1.0 db	± 45 deg
Transmitter Frequency	200 Mc	. .	. . .
Polarization Loss Axial ratio Transmitter 5 db Axial ratio Receiver 5 db	0.5 db	± 0.3 db	. . .
Receiving Antenna Gain	7.0 db	± 1.0 db	. . .
Receiving Antenna Pointing Loss	4.0 db	± 1.0 db	± 47 deg
Receiving Circuit Loss (Total losses between receiver and antenna terminals. Includes losses mentioned in antenna discussion.)	0.8 db	± 0.2 db	. . .
Receiver Noise Spectral Density	-169 dbm/cps	± 1.0 db	5 db NF
Carrier APC Noise BW	75 cps	± 7.5 cps	. . .
Required Carrier SNR in $2B_{LO}$ for data transmission	6.0 db	. . .	. . .
Required Data ST/N/B $P_e^b = 5 \times 10^{-3}$ 1.5 db RF losses, 0.5 db misc. losses	7.2 db	$^{+0.5}_{-0.0}$ db	. . .
Total Transmitter Power	variable	± 1.0 db	. . .

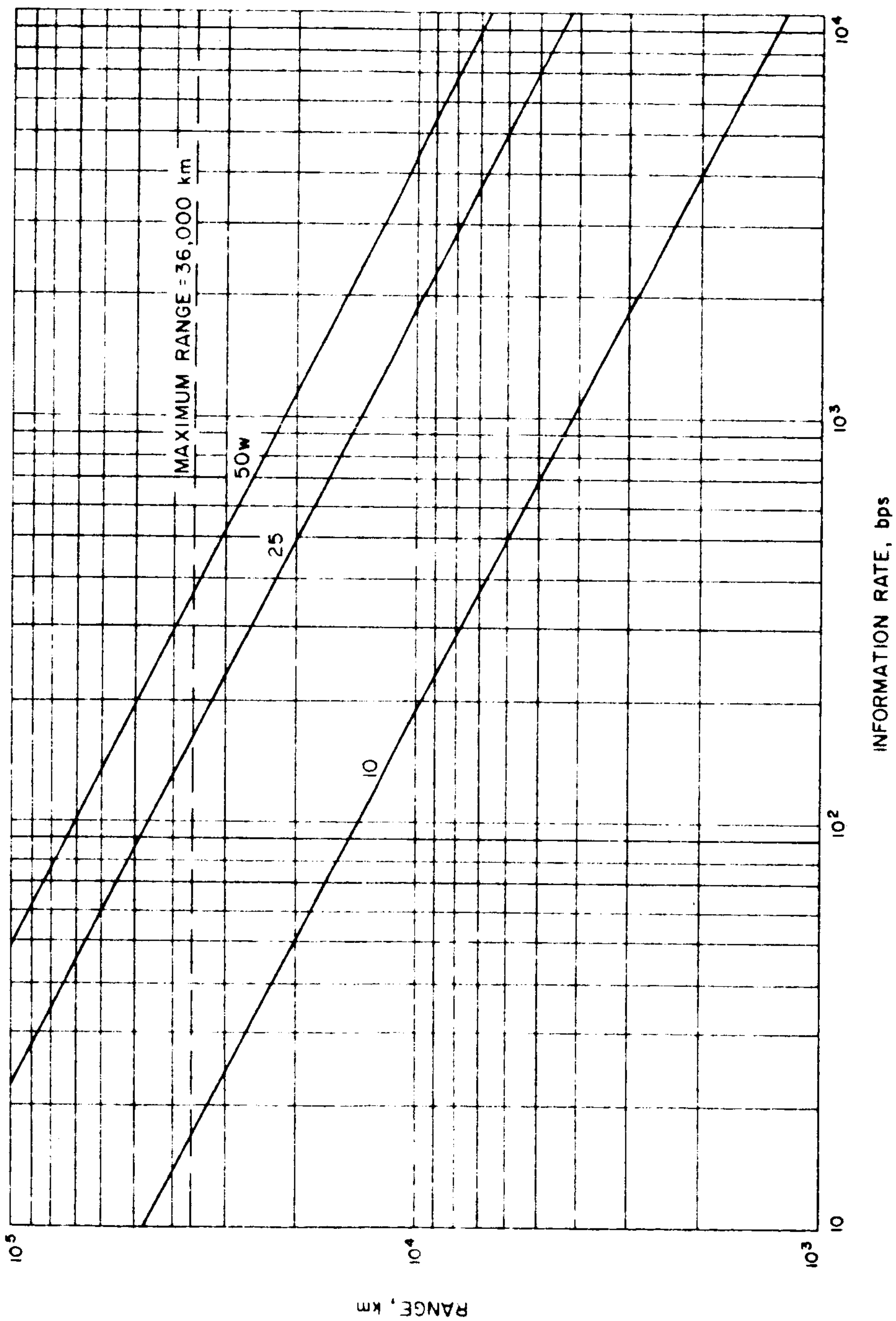

Figure 4-122. Information Rate vs Capsule-to-Bus Range for Various Capsule Transmitter Powers Phase I

1) determine if the capsule survived the entry phase and
2) to read-out data from entry experiments.

(2) Phase II - Lander Surface Operation. Table 4-27 lists the assumed parameters and estimates of tolerances for the communications link design for Phase II. Figure 4-123 shows the available data rate vs lander-to-orbiter range for various transmitter powers. Again, the transmitter modulation index has been varied, so the information rate does not change by the same factor as the transmitter power.

(3) Choice of Power Level. Two possible capsule transmitter power levels are presently under consideration, 25 and 50 w. Concurrent weight and power estimates have therefore been compiled for these two levels.

In making any final decision the following factors will have to be considered:

1) At the 25 w level it may be possible to use all solid-state techniques with relatively low supply voltages, thus reducing the potential problem of high voltage breakdown during entry and on the planet surface.
2) The 50 w power level would be approximately that needed to satisfy the desired data-handling requirements, but may be more than the capsule power system can support.

b. Orbiter-to-Earth

(1) Orbiter-to-Earth Data Link. Tables 4-28 and 4-29 list the assumed parameters and estimates of tolerances for the orbiter-to-Earth communications link for both the 85 ft and 210 ft diameter receiving antennas.

Estimates of orbiter data rate capability vs range are presented in Figures 4-124 and 4-125. The parameter used is power-gain product which is equal to the sum of the transmitter power, in dbm, and the transmitting antenna gain, in db.

(2) System Optimization. A study has been made to find the size of the orbiter high-gain antenna that yields the minimum system weight. While the assumptions used for this study are not in complete agreement with all the values of parameters presently recommended, the results are instructive, because they show the (typical) discontinuities of such an optimization. The following assumptions were made:

1) The antenna system was assumed to be constrained to fit within a 152 in. diameter upper stage.

Table 4-27. Communication Parameters, Lander-to-Orbiter - Phase II

Parameter	Value	Tolerance	Comment
Transmitting Circuit Loss (Total losses between transmitter and antenna terminals. Includes losses mentioned in antenna discussion.)	0.5 db	± 0.2 db	–
Transmitting Antenna Gain	2.0 db	± 2.0 db	–
Transmitting Antenna Pointing Loss	10.0 db	± 1.0 db	± 70 deg
Transmitter Frequency	200 Mc	–	–
Polarization Loss Axial Ratio Transmitter 10 db Axial Ratio Receiver 5 db	1.5 db	± 1.2 db	–
Receiving Antenna Gain	7.0 db	± 1.0 db	–
Receiving Antenna Pointing Loss	4.0 db	± 1.0 db	± 47 deg
Receiving Circuit Loss (Total losses between receiver and antenna terminals. Includes losses mentioned in antenna discussion.)	0.8 db	± 0.2 db	–
Receiver Noise Spectral Density	169 dbm cps	± 1.0 db	5 db NF
Carrier APC Noise BW	75 cps	± 7.5 cps	–
Required Carrier SNR in $2B_{L0}$ for data transmission	6.0 db	–	–
Required Data ST N B P_e^b 5 · 10^{-3} 1.5 db RF losses, 0.5 db misc losses	7.2 db	$^{+0.5}_{-0.0}$ db	–
Total Transmitter Power	variable	± 1.0 db	–

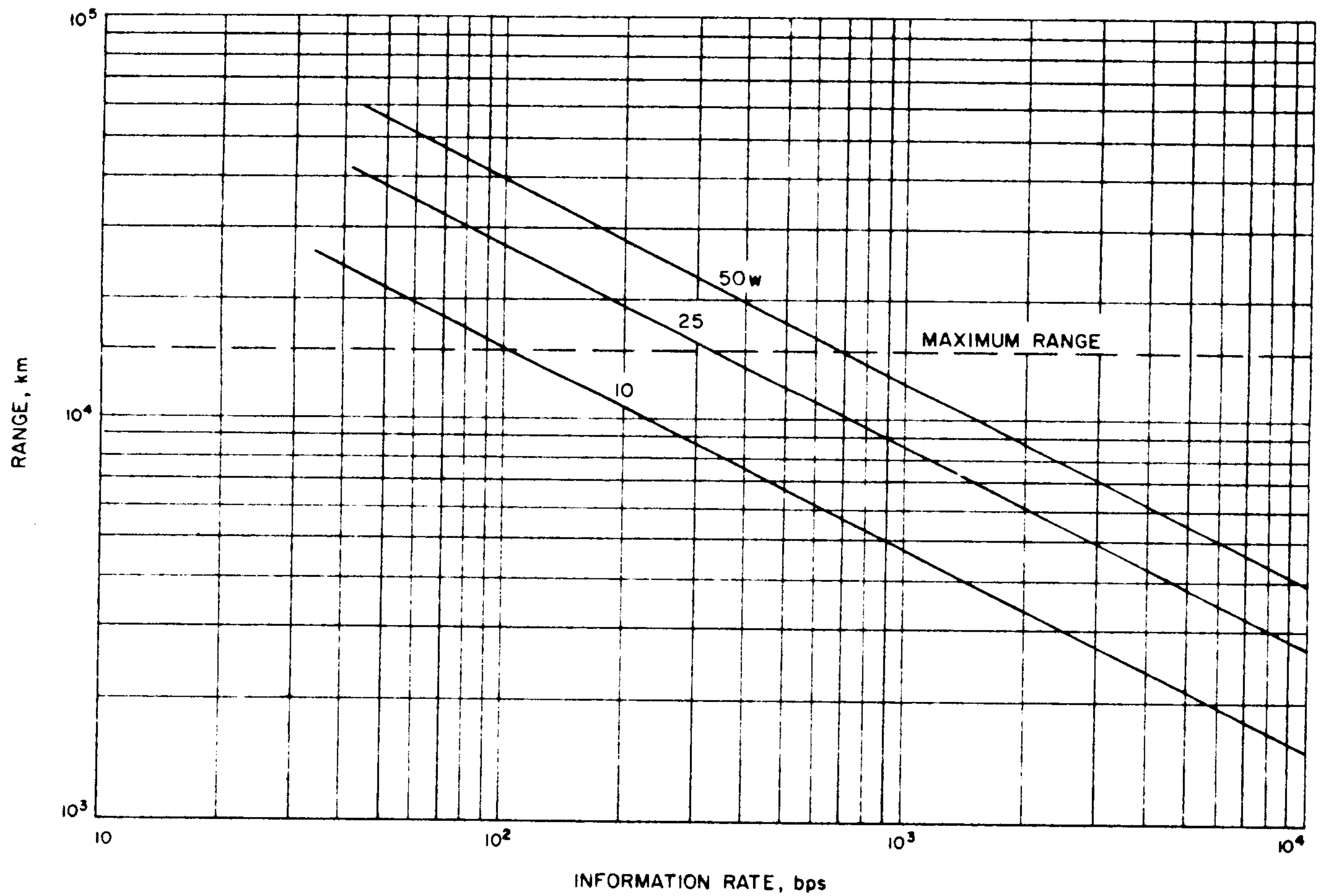

Figure 4-123. Information Rate vs Lander-to-Orbiter Range for Various Capsule Transmitter Powers Phase II

Table 4-28. DSIF Parameters, Orbiter to Earth

	Standard Deep Space Station 85 Foot Antenna		Low Noise Listen Station 210 Foot Antenna	
Item	**Value**	**Tolerance**	**Value**	**Tolerance**
Antenna Gain	53.0 db	$\pm^{1.0}_{0.5}$ db	61.0 db	± 1.0 db
Axial Ratio	0.75 db	± 0.25 db	0.5 db	–
Feed Line Loss to Low Noise Amplifier (Includes diplexer loss)	0.18 db	± 0.05 db	0.02 db	± 0.01 db
Antenna Temperature (zenith)	16 K	± 3 K	10°K	± 2°K
Maser T excess 2nd Stage	18°K	± 3°K	18°K	± 3°K
Transmitter Noise Contribution	10 K	± 3°K	–	–
Carrier APC Noise BW	12 cps	$\pm^{0.79}_{0.97}$ cps	12 cps	$\pm^{0.79}_{0.97}$ cps
Required Carrier SNR in $2B_{L0}$ (for data reception)	6.0 db	–	6.0 db	–
Required Data $ST/N/B$ ($P_e^b = 5 \times 10^{-3}$, includes 1.5 db R.F. losses)	+7.2 db	$\pm^{0.5}_{0.0}$ db	+7.2 db	$\pm^{0.5}_{0.0}$ db

Table 4-29. Orbiter Communication Parameters, High Gain Antenna

Item	Value	Tolerance
Total Transmitter Power	Variable	± 1.0 db
Transmitting Circuit Loss	2.5 db	± 0.5 db
Transmitting Antenna Gain	36.7 db	± 1.0 db
Transmitting Antenna Pointing Loss	0.5 db	$\pm^{0.5}_{0.0}$ db
Antenna Axial Ratio	1.0 db	–
Frequency	2295 Mc	–

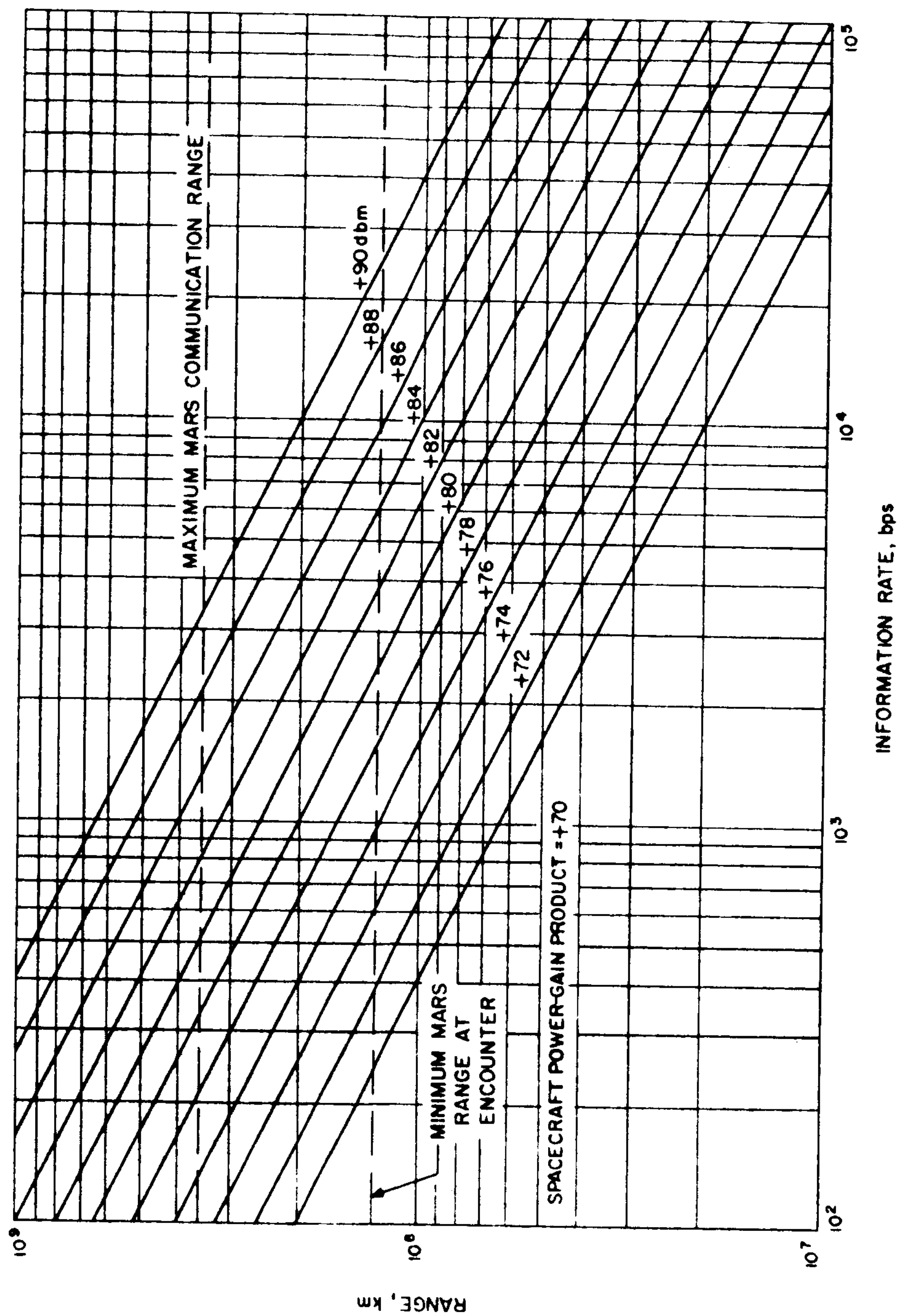

Figure 4-124. Information Rate vs Range with an 85 ft Diameter Ground Antenna

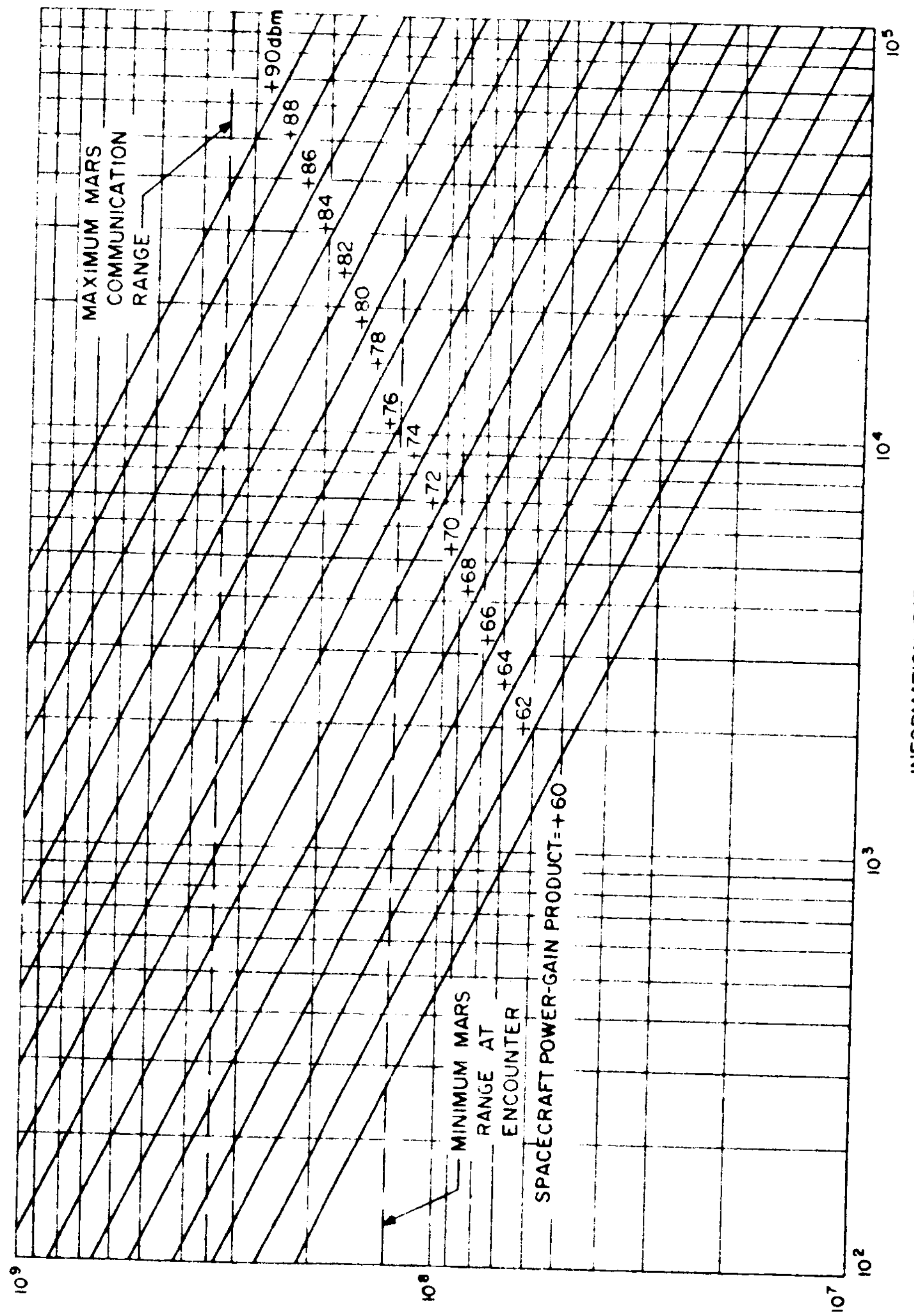

Figure 4-125. Information Rate vs Range with a 210 ft Diameter Ground Antenna

2) The weight of batteries and battery chargers was assumed to be independent of the communications system, since the communications system was assumed to be operating only during the phase of the orbit when the solar panels are illuminated.

3) The transmitter system was assumed to be that described, previously, in part III B of this section; a completely redundant system is assumed.

4) The antenna was assumed to be a swirl-rib, foldable configuration that is currently under development by industry.

5) The solar panels were assumed to weigh 2 lb/ft^2 and deliver 2.5 w/ft^2 at Mars distance. This yields 1.25 watts of raw power per pound of solar panel weight.

6) Active pointing angle control was assumed to be necessary for antenna diameters in excess of 8 ft. The additional weight necessary for this was assumed to be 10 lbs. However, without it the accuracy with which the electrical axis of the antenna could be pointed was assumed to be ±1.4 deg, leading to an assumed gain degradation of 2 db for an 8 ft antenna and 1 db for a 6 ft antenna.

The system weight was computed as a function of antenna diameter for various gain-power products of the spacecraft communications system. The resulting curves are shown in Figure 4-126a. The large discontinuities at 8 ft are the result of change to simultaneous lobing for pointing-angle control. Other discontinuities are caused by the necessity of switching from an amplitron transmitter to a vacuum-tube cavity type and also to the change in the number of ribs in the antenna at 12 and 16 ft, respectively. The minimum points are marked by dots.

The antenna aperture diameters corresponding to the minimum system weight vs information rate are plotted in Figure4-126b. Communications distance is assumed to be 350 by 10^6 km. The transmitter radiated power corresponding to each antenna size is also shown in Figure 4-126b. The discontinuity at approximately 1.5×10^3 bps is caused by switching from the vacuum tube cavity to the amplitron transmitter. The plateaus at 12 ft and 16 ft are caused by the assumed increase in the number of ribs of the foldable antenna. The increases were arbitrarily assumed to occur at these diameters, and no generality would be lost in assuming that they occur elsewhere.

It is interesting to note that, if a transmitter system of the type previously described is assumed, the vacuum-tube cavity transmitter is used only for levels of radiated power below 41 dbm. At this point it becomes advantageous to use the amplitron transmitter and an antenna of a smaller diameter.

The results of Figure 4-126b apply for a 210-ft Earth-based antenna. For the 85-ft antenna the information rate scale would be changed by a factor of 10^{-1}.

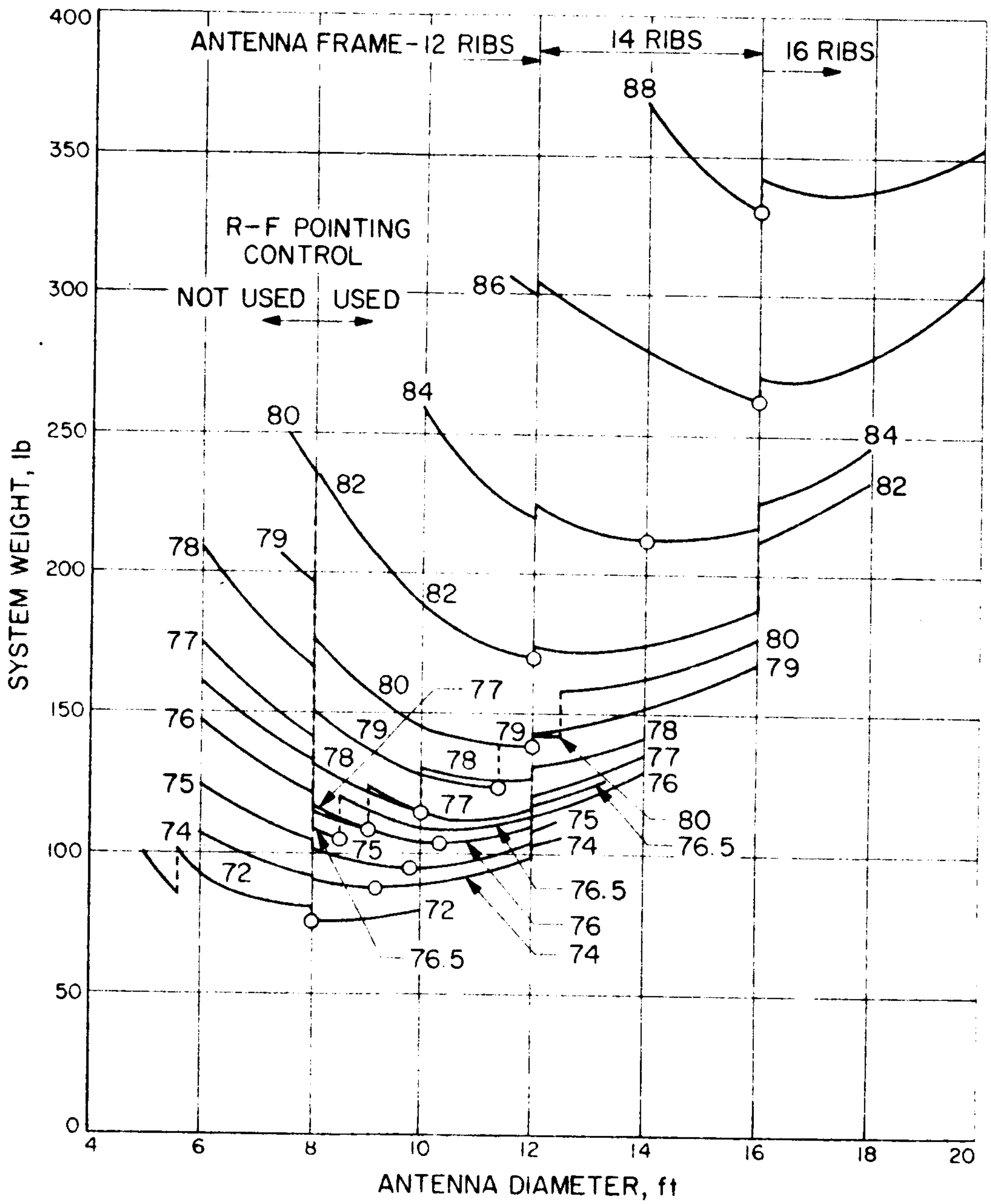

Figure 4-126A. System Weight vs Antenna Diameter

Table 4-30. Earth-to-Orbiter Link

Parameter	Value	Tolerance
Transmitter Power Output	10 kw	± 0.2 db
Feed Line Loss to Xmitter	0.5 db	± 0.1 db
Antenna Axial Ratio	0.75 db	± 0.25 db
Diplexer Loss	0.05 db	± 0.025 db
Antenna Gain	51.0 db	± 1.0 db
Frequency	2113 Mc	

DSIF Transmitting Parameters, Standard Deep Space Station

Parameter	S C Omni Antenna		S C High Gain Antenna		Comment
	Value	Tolerance	Value	Tolerance	
Antenna Gain	3.0 db	± 1.0 db	36.7 db	± 1.0 db	
Antenna Axial Ratio	6.0 db	$^{4.0}_{6.0}$ db	1.0 db	–	
Antenna Pointing Loss (110)	8.0 db	± 1.0 db	0.5 db	$^{0.5}_{0.0}$ db	
Receiving Circuit Loss	2.5 db	± 0.5 db	2.5 db	± 0.5 db	
Noise Spectral Density	169.5 dbm cps	1.0 db	169.5 dbm cps	1.0 db	4.5 db NF
Carrier APC Noise BW	20 cps	$^{0.79}_{0.97}$ cps	20 cps	$^{0.79}_{0.97}$ cps	
Required Carrier SNR in $2B_{L0}$ (for command reception)	8.0 db	–	8.0 db	–	
Required Command ST N B for 1bps (P_e^b 1 · 10^{-5}, includes 1 db RF losses)	14.0 db	$^{1.5}_{0.0}$ db	14.0 db	$^{1.5}_{0.0}$ db	
Required Command ST N B for 20bps (P_e^b 1 · 10^{-5}, includes 1 db RF losses)	11.1 db	$^{1.5}_{0.0}$ db	11.1 db	$^{1.5}_{0.0}$ db	

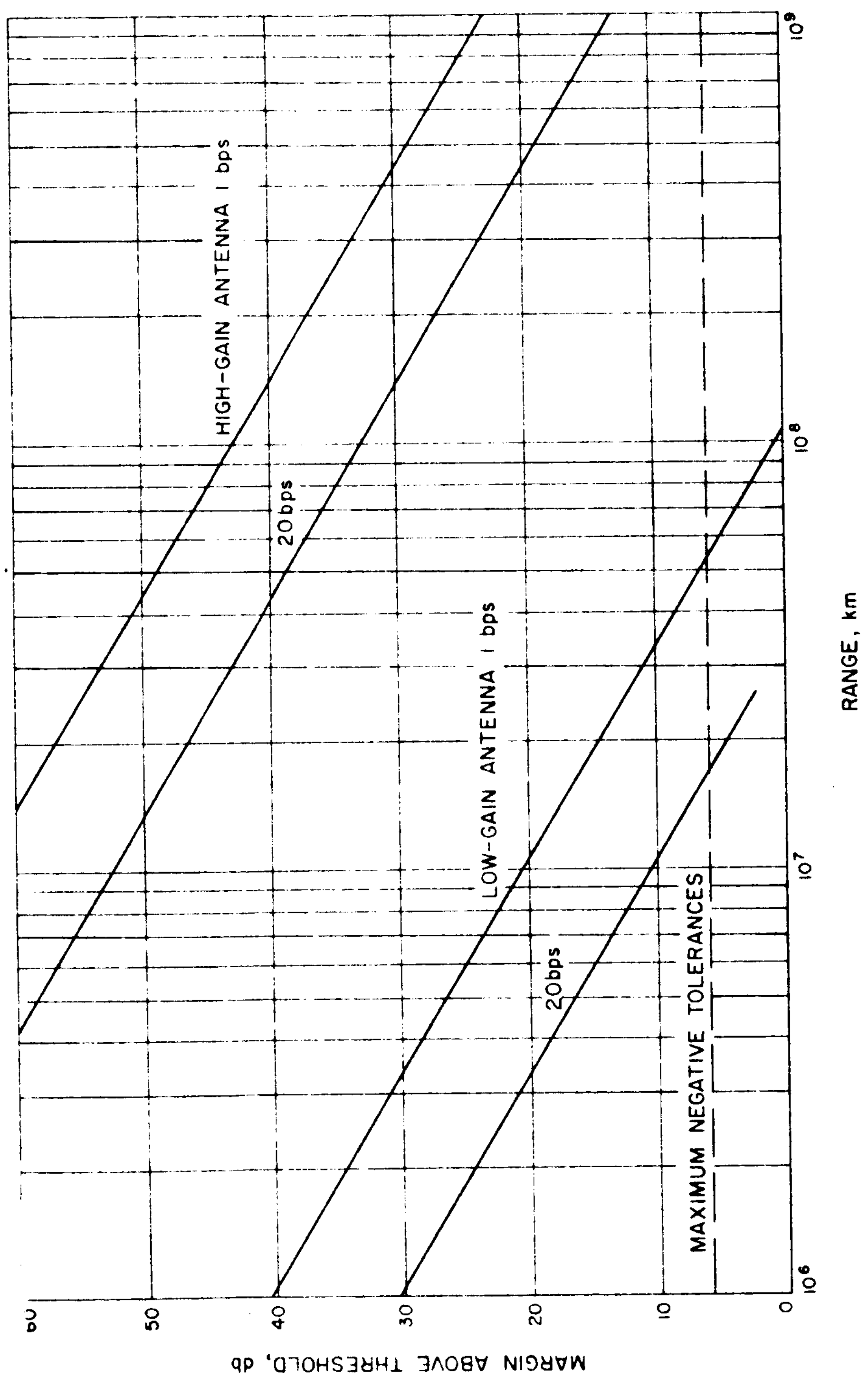

Figure 4-127. Earth-to-Spacecraft Command Performance with a 10 kw Transmitter

c. Capsule-to-Earth Direct Link

The capsule-to-Earth link is an S-band link operating directly from the lander on the Mars surface. The data rate capability of this link is considerably less than that of the relay link.

An estimate has been made for a system that results in a data rate greater than 0.5 bps. For this estimate several assumptions have been made. They are:

1) The design range is 235 by 10^6 km, which corresponds to a 30 day lander life.
2) The lander is located within ±5 deg of the Mars equator, and the lander antenna axis is pointed within 10 deg of the local vertical. For this condition 7.1 to 8.3 hr of communications time per day could be realized and still be within 70° of the lander antenna axis.
3) After landing the antenna is not covered by protective material or masked by structure.

Both the 25 w and 50 w levels of direct link transmitter power have been assumed in preparation of the weight and power estimates as given in Table 4-31. The information rate that this link can provide will depend on the mechanization that is used and could range from 0.1 to 1.0 bps at the 25 w level, and to 2 bps at the 50 w level. In order to define the performance more accurately, further study will be needed.

Even though we are not now able to accurately predict the performance of a direct link with rates less than 0.5 bps, we strongly feel that a direct link should be retained. If a worthwhile data rate with a reasonable transmitter power can be provided, then a capsule communications capability that does not depend upon the presence of the orbiter can be achieved.

7. Weight, Power and Volume Summary

a. Lander

Weight, power and volume estimates for Option I and Option II are given in Tables 4-31 and 4-32, respectively. All power requirements are at the input of the Transformer - Retifier system except for the transmitter power requirements, which are at the power source (solar cells, battery, radioisotope thermoelectric generator, etc.).

Table 4-31. Weight, Power and Volume Estimates for Communication Option I

	25 w level			50 w level		
	Weight (lb)	Power (regulated watts)	Volume (in.^{3})	Weight (lb)	Power (regulated watts)	Volume (in.^{3})
Relay Link						
VHF Antenna	15	–	5900	15	–	5900
VHF Transmitter	3.5	57.5[a, c]	100	4.0	116 [a, c]	130
Modulator	1.0	1.88	50	1.0	375	50
RF Cables	0.3	–	–	0.3	–	–
Power Supply (Modulator & Transmitter)	3.5	–	60	4.0	–	75
Direct Link						
S-Band Antenna	0.5^3	–	10	0.5[b]	–	10
S-Band Transmitter	4.5	65 [c]	350	5.5	126	400
Modulator	0.5	1.25	nil	0.5	6.25	nil
Antenna Cables	0.4	–	–	0.4	–	–
Isolator, Filters, RF Cables	3.2	–	16	3.2	–	16
Power Supply (Modulator & Transmitter)	6.0	–	125	8.0	–	175
Totals	38.40	125.63	6611	42.40	252.25	6756

Note:
[a]Continuous transmission assumed
[b]No redundancy assumed
[c]Assumes input voltage between 25 and 50 volts

Table 4-32. Weight, Power and Volume Estimates for Communication Option II

	25 w level			50 w level		
	Weight (lb)	Power (w)	Volume (in.^{3})	Weight (lb)	Power (w)	Volume (in.^{3})
Same Systems as for Option I	38.40	125.63	6611	42.40	252.25	6756
Relay (Command)						
Diplexer	1	–	30	1	–	30
VHF Receiver	3.5	4.38	100	3.5	4.38	100
Command Detector	5.0[a]	3.75	50	5.0[a]	3.75	50
Power Supply (Receiver & Command)	3.0	–	60	3.0	–	60
Total	50.9	133.76	6851	54.9	260.33	6996

Note:
[a] Commands other than just presence of orbiter are assumed

b. Orbiter

Weight, power and volume estimates for both Option I and Option II are given in Table 4-33. All power requirements are dc, regulated and are taken at the transformer-rectifier output except the transmitter which is taken at the output of the dc-to-dc convertor. Where redundancy is indicated, the figures shown are total requirements.

Table 4-33. Orbiter Weights, Power and Volume Estimates for Communication Option I and II

Orbiter – Capsule Link						
VHF Antenna	3.0	–	–	3.0	–	–
RF Rotary Joint	2.0	–	10	2.0	–	10
Diplexer	–	–	–	1.0	–	30
VHF Transmitter (beacon - 5 w)	–	–	–	2.5	13.3	80
Command Modulator	–	–	–	3.0	5.0	60
Power Supply (transmitter & Command)	–	–	–	2.0	–	75
VHF Receiver (2)	9.0	9.0	300	9.0	9.0	300
Data Detector (2)	6.0	4.0	100	6.0	4.0	100
Power Supply (receiver & detector) (2)	2.0	–	100	2.0	–	100
Circulators & Miscellaneous Hardware	4.0	–	100	4.0	–	100
RF Cables	0.4	–	–	0.7	–	–
Totals	186.2	95.4	3021	195.0	113.7	3266

Notes:
(a) Includes feed and dish
(b) Only one is energized at any time
(c) Only two units within communications case, 100 in.^{3} each
(d) Power to be supplied by user subsystem

H. SCIENTIFIC MEASUREMENTS

1. Introduction

In Chapter 3, Section II of this report, the scientific objectives of the mission were outlined in detail. The purpose of this section is to further define the requirements of the scientific system and enumerate typical constraints imposed upon both the orbiter and the capsule. It must be realized that these requirements are only approximations. The final scientific requirements will be established during preliminary design when the selection of the mission scientific payload is made by the NASA Space Science Steering Committee.

2. Scope

The material presented is divided into two sections. The first section will describe measurements during the cruise portion of the flight and from an orbiter. The second section will describe the measurements to be conducted from the descending and landed capsule. Each section will further describe several typical instruments that could be flown in each measurement category, general constraints peculiar to each measurement category, and the general physical characteristics of each category such as weight, power, and data.

3. Interplanetary and Orbiting Spacecraft

a. Interplanetary and Orbiter Measurements

Tables 4-34, 4-35 and Figure 4-128 summarize typical weight, power, data requirements, and general constraints placed on the interplanetary and orbiting spacecraft by the scientific experiments. Since these are typical constraints and not necessarily those of the final scientific payload, the spacecraft design should be flexible enough to accommodate different instruments. This requirement is also a prerequisite for intra-mission flexibility.

The instruments and constraints in the following sections will be used to illustrate how a given set of experimental objectives can be translated into a design concept. It should be noted that the data requirements delineated are those of the experiments and do not reflect the necessary coding and addresses required for transmission. No consideration has been given to reducing telemetry data requirements through data compression; however, it is felt that by appropriate mechanization a considerable reduction in data transfer can be achieved.

b. Interplanetary and Orbiter Power Profile

The interplanetary power profile for the scientific experiments represents continuous operation of all instruments during the total period required to traverse the distance from

Table 4-34. Summary of Scientific Measurement Requirements from the Bus During Cruise

Measurements	Weight (lb)	Power (w-regulated ± 5%)	Data Requirements	Spacecraft Requirements
Fields and Particles	55	25	100 bps minimum for total period of cruise (approximately 1.6×10^9 minimum bits for 200 day cruise flight). Instruments can utilize additional bandwidth above this requirement if available	Orientation of instruments with respect to solar coordinates required
Micrometeoroid	8	1	1. Approximately 20 bits/impact 2. Estimated 50 impacts in 200 day flight to Mars. 3. Can be sampled on fixed format data system or upon recording of event.	1. Measurements will be omnidirectional and the bus should obstruct as small a subtended angle as possible with respect to the instrument. 2. Orientation with respect to solar coordinates
Totals	63	26	Approximately 1.8×10^9 bits for 200 day cruise to planet.	

Table 4-35. Summary of Scientific Measurement Requirements from the Orbiter

Measurements	Weight (lb)	Power (w-regulated ± 5%)	Data Requirements	Spacecraft Requirements
Atmosphere	64	25	7.0×10^8 bits for 150 day orbit	1. Oriented to planet vertical. 2. Orientation with respect to planet vertical and solar coordinates.
Surface Structure	130	60	1. 5.0×10^9 bits for selected stereo map of 1/4 of planet surface (1/2 km resolution) and total thermal profile map. 2. Stereo area coverage should be increased (if possible) to utilize maximum data storage and transmission capabilities.	1. Controlled area coverage. 2. 53 deg angle separation for forward and back cameras or other optical systems. 3. Oriented to planet vertical. 4. Orientation with respect to planet vertical and solar coordinates.
Fields and Particles	55	25	20 to 100 bps for a considerable fraction of the total mission (approx. 1.3×10^9 bits for 150 day orbit).	Instrument orientation with respect to planet vertical and solar coordinates.
Micrometeoroid	8	1	1. Approximately 20 bits/impact. 2. Estimated 1500 impacts in 150 day orbit. 3. Can be sampled in fixed format data system or upon recording of event.	1. Measurements will be omnidirectional and the orbiter should obstruct as small a subtended angle as possible with respect to the instrument. 2. Orientation with respect to planet vertical and solar coordinates.
Totals	257	111	Approximately 7.0×10^9 bits for 150 day orbit.	

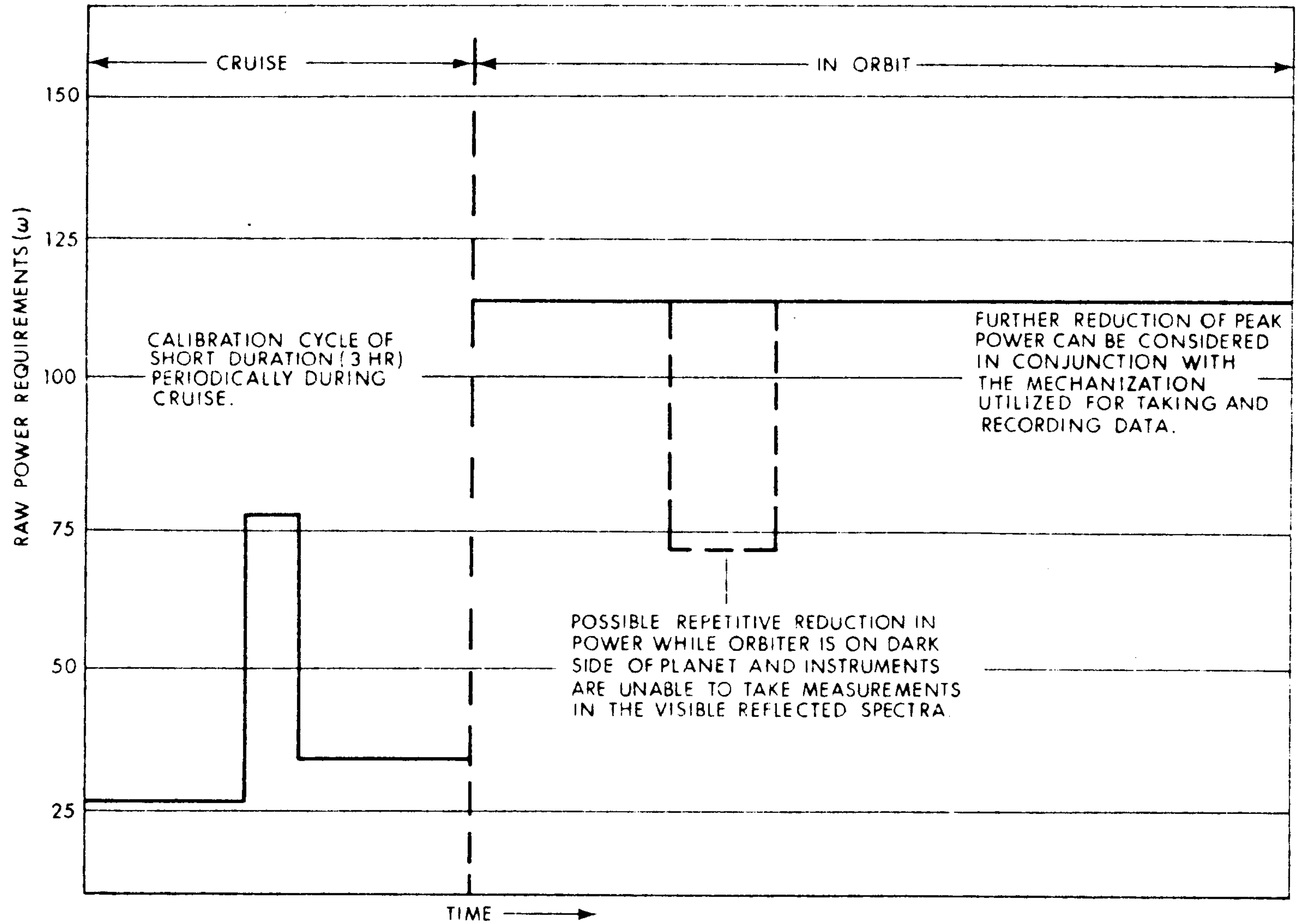

Figure 4-128. Tentative Raw Power Profile During Cruise and Orbit

Earth to Mars. The profile is interrupted periodically during cruise for short calibrations of the planetary scientific instruments. The power profile for the orbiter is more difficult to establish. It depends not only on demands of the instruments, but on the method in which they are mechanized to handle the data, their operational characteristics, and the desire to smooth the peak power requirements by the proper use of sequencing. Since the scientific payload being considered for this study is only typical of the requirements of the final payload, a thorough study of the means of reducing power to the minimum peak load has not been completely investigated. Figure 4-128 reflects the power required to operate the instruments on a 100 percent duty-cycle, with the exception of portions of the surface structure measurements, and does not take into consideration any method of sequencing to ease the storage requirements on the data system or the peak power load on the power system.

c. Typical Bus/Orbiter Instrumentation and Constraints

The instruments listed under each of the following measurement categories are representative of the types available to meet the scientific mission objectives. From these instruments (and others not listed) the selection of the final scientific instrument payload will be made.

(1) Atmospheric Instruments

1) UV Spectrometer	Determination of the abundance and height distributions of the principle atmospheric constituents obtained from UV spectra of the day, night, and twilight portions of Mars with emphasis on lighted side.
2) IR Spectrometer	Determination of those minor constituents of the upper cloud structure whose absorption spectra are in the IR region. Measurement of the atmospheric profile of temperature and pressure.
3) Microwave Spectrometer	Determine the molecular composition of the atmosphere (clouds, below clouds, at surface) characteristic lines.
4) UV Photometer	Determination of the presence of particular constituents of the upper atmosphere by means of filter photometry.

(a) Temperature Control Requirements

	Instrument	Operating (°C)	Nonoperating (°C)
1)	UV Spectrometer	-20 to +60	-40 to +85
2)	IR Spectrometer		
	sensor	-20 to +0	-150 to +50
	electronics	0 to +50	-70 to +100
3)	Microwave Spectrometer	-30 to +85	-50 to +85
4)	UV Photometer	-20 to +60	-40 to +85

(b) View Angle and Orientation Requirements. The instruments in this category will be aligned to the planet vertical and will require a pointing accuracy of 1/10 of the beam width. The view angles will be solid cones ranging from 1 to 30 deg and will require unobstructed view of the planet surface.

(c) Data Requirements and Sequencing. The instruments in this measurement category are divided into two distinct groups. The first requires a constant data rate during the entire life of the orbiter at relatively low bit rates (10 to 30 bps). The second group will be operated by command, will have high data rates (400 to 2000 bps), and will not be in continuous operation.

In the second category the experiments will take complete spectra for a specific section of an orbit, transmit this information to Earth, and analyze the content of the data. Upon completion of the analysis, the instrument will be commanded to look to specific portions or lines of the spectra. The duty cycle of the instrument is governed by the time it takes to transmit the data, analyze it, and send a command for the next segment of data from the instrument. The whole process will be repeated for the next portion of data.

(2) Surface Structure Instruments

1)	IR Spectrometer	Simultaneous mapping of composition of surface and correct radiation temperature from orbiting vehicle.
2)	Extended Microwave Spectrometer	Will measure quantities stated for the Microwave Spectrometer and also will give a geographic temperature study of Martian surface.
3)	Microwave Radiometer	Thermal mapping of the surface (33mm) and structure of atmosphere (8mm).
4)	Television	Low and high resolution monoscopic and/or stereoscopic coverage of designated planetary surface area.

5) IR Radiometer — Thermal mapping of the surface to give a geographic temperature profile.

(a) Temperature Control

	Instrument	Operating (°C)	Nonoperating (°C)
1)	IR Spectrometer		
	sensor	-20 to 0	-150 to +50
	electronics	-10 to +50	-70 to +100
2)	Extended Microwave Spectrometer	-30 to +85	-50 to +85
3)	Microwave Radiometer	-30 to +80	-50 to +85
4)	Television		
	camera & optics	-20 to +40	-50 to +100
	electronics	-20 to +70	-40 to +100
5)	IR Radiometer	-30 to +80	-50 to +85

(b) View Angle and Orientation Requirements. All instruments currently under consideration for this measurement category, with the exception of stereoscopic television, will have their view angles bisected by the planet vertical. The stereoscopic television experiment will utilize two cameras or an optical system with the center of field of view of each lens system separated by 53 deg and aligned to the orbital plane of the spacecraft with the angle of lens separation bisected by the planet vertical. It is not anticipated that the total field of view of any of the instruments under consideration will exceed 5 deg.

The orbiter coordinate system should be oriented within 5 deg of the satellite orbit coordinate system and known to 1 deg. The maximum excursion of the orbiter should not be more than 0.1 deg about its nominal orientation. Maximum slewing rate should not exceed 1.0 deg/hr.

(c) Data and Sequencing Requirements. With the exception of the vidicon experiments, the remaining instruments under consideration operate at constant data rates during the entire life of the orbiter. The bit rates of these experiments will be between 8 and 30 bps. Use of the vidicon system will be limited to the portion of time that adequate lighting is available and the distance above the surface is at a minimum. The additional constraints of specific area coverage will further reduce the time of instrument operation.

The final decision of operational modes for the vidicon experiment cannot be made until preliminary design; hence, the exact data rates per unit time cannot be specified at this time. However, the total data requirement specified for this experiment is considered to be adequate for the required area coverage. This requirement does not take into consideration data compression or the necessary coding or addresses necessary for the transmittal of this information.

The vidicon experiment is the only instrument in this category that will require extensive sequencing. It is likely that this sequencing will be almost entirely by ground command.

(3) Fields and Particle Instruments

1)	Particle Flux	(Interplanetary) Used in conjunction with the Ion Chamber to monitor the energetic particle and photon radiation in interplanetary space. Provides semi-quantitative information about the energy and particle types composing the radiation.
2)	Ion Chamber	(Interplanetary) To measure the total ionizing radiation (cosmic rays) and the variation with time and position in the solar system.
3)	Trapped Radiation Detectors	To search for magnetically-trapped charged particles in the vicinity of Mars and study their intensity, directional and spatial distributions.
4)	Medium-Energy Proton Directional Monitor	(Interplanetary) Observation on the temporal and spatial variations of cosmic-ray intensity. (Planetary) To search for magnetically-trapped charged particles in the vicinity of Mars and to measure the spatial extent and anisotropy of the particle fluxes in the low intensity regions of the trapping regions.
5)	High-Energy Proton Directional Monitor	(Interplanetary) To provide information about the temporal, spatial and directional variations of galactic cosmic ray flux.
6)	Cosmic Ray Spectrum	(Interplanetary) To investigate the intensity and the gradient of flux of charged particles in space as functions of nuclear species and time.
7)	Plasma Instrument (Electrostatic Analyzer)	(Interplanetary) To measure the flux of positive and negative components of solar plasma traveling outward from the Sun with good energy and directional resolution. To correlate these data with the magnetic field and the low energy plasma.

8)	Bi-Static Radar	(Interplanetary) To obtain information on the absolute electron density in interplanetary space and the fluctuation of this density as a function of time. (Planetary) Obtain information on the electron density profile of the ionosphere about Mars by progressive and selective absorptions of the Earth-transmitted signals during the occulting pass.
9)	Plasma Instrument (Faraday Cup)	(Interplanetary) To determine the particle number, density, the distribution of velocity vector, and the temporal and spatial distributions of the interplanetary plasma. To correlate these data with the magnetic field measurements.
10)	Magnetometer	(Interplanetary) Variation in magnetic and direction of the interplanetary field with heliocentric distance and longitude. (Planetary) To establish the existence of a planetary field and to determine its characteristics, e.g., magnitude, direction, multipolarity, and orientation. To investigate the nature of the interface between planetary and interplanetary magnetic fields.

(a) Radiation Constraints. The effects of radiation on the particle and fields experiments can be separated for the two main phases of the mission: cruise to the planet, and the orbit phase. When considering only the measurement characteristics of the instrument and not the physical problems due to radiation damage of components, etc., it is possible to ascertain the effect that radiation will have on the scientific experiments for either mission phase.

In the cruise phase the experiments are designed to measure the natural interplanetary radiation background. This radiation consists of a flux of heavily charged particles (protons and heavier nuclei) with energies from 10 Mev/nucleon up to more than 10^6 Mev/nucleon. The average omni-directional flux of these particles is 2 to $3/cm^2$ sec, and they ionize at the rate of 1 to 2 mr/hr. In addition, there is a small flux of electrons and possibly of gamma rays and solar neutrons amounting to no more than a few percent of the heavy particle flux. In measuring this radiation, the spacecraft radiation produced should not exceed more than 1 percent of the natural radiation for an optimum experiment. For a lesser experiment, it is possible to study the natural radiation using simple detectors of the type already developed, as long as the spacecraft radiation produced is equal to or less than the natural radiation and is strictly constant in time. For example, one could study the heavily charged particles to a background of 1 mr/hr of gamma-rays and a fet neutrons/cm^2 sec, but could not measure natural gammas or neutrons in such a flux. The presence of such a source would require very careful calibrations of all instruments on the spacecraft.

When artificial radiation exceeds the natural radiation, simple detectors can no longer be used. It would be necessary to develop solid state detectors employing fast coincidences. Gamma-rays with flux of greater than 1 mr/hr will create problems and neutrons will cause difficulty by activating material in the spacecraft and detectors, thus producing additional gammas of unknown energy.

In the orbiter phase, the problem can be greatly alleviated if the source of large spacecraft radiation is carried only on the capsule. After the capsule is released, the only radiation problem remaining is due to the activated material generated by neutron capture during the cruise phase of the mission. However, trapped radiation belts similar to the Earth's can be studied in the presence of spacecraft-produced radiation on the order of 10^3 higher than can be allowed in interplanetary space.

In order to measure the complete interplanetary and planetary radiation environment, it is necessary that the radiation flux produced by radioisotopes and other radiation generators aboard the spacecraft and measured at the radiation detectors be small enough that:

1) No detector is saturated.
2) The artificial flux can be subtracted from the measured flux in such a way that the subtraction introduces an uncertainty no greater than $\pm\delta$ percent in the smallest natural flux observed. The uncertainty, δ, should be of the order of the instrument uncertainty, but no greater than 5 percent in any case.

Generally, item 2) will impose a much lower limit on the spacecraft-produced radiation than item 1).

Items 1) and 2) are satisfied if the spacecraft-produced radiation of each type is no greater than 1 percent of the anticipated minimum natural radiation of the same type. These upper limits are shown in Table 4-36. The allowable flux is defined as the flux from 2π steradians which enters any surface of the sensitive volume of a detector.

In general, it will not be necessary to measure the artificial flux or to guarantee its consistency so long as it remains below the values specified. However, particular instruments may require detailed data even at these low levels. All instruments will require calibration on the spacecraft with all artificial sources in place.

Table 4-36. Upper Limits of Spacecraft-Produced Radiation

Radiation	Maximum allowable flux Particles/cm^2 sec at the detectors
Protons E > 10 Mev 10 Mev. E > 0.02 Mev[a] 20 Kev > E > 1 ev	 0.003 0.01 10^6 per decade of energy
Alphas and heavier nuclei E > 10 Mev/nucleon 10 Mev. E > 0.02 Mev/nucleon[a] 20 Kev > E > 1 ev	 0.0003 0.001 10^6 per decade of energy
Electrons E > 0.5 Bev 0.5 Bev > E > 1 Mev 1 Mev > E > 0.02 Mev[a] 20 Kev > E > 1 ev	 0.0001 0.001 0.001 10^6 per decade of energy
Neutrons E > 1 Mev 1 Mev. E > 0.001 Mev 1 Kev > E	 0.01 0.01 0.01
Protons[b] E > 50 Mev 50 Mev. E > 3 Mev 3 > E > 0.3 Mev	 10^{-6} $5 \cdot 10^{-3}$ 10^{-3} with no resolvable peak containing more than 5% of this energy level
300 Kev > E > 100 Kev 100 Kev > E > 10 Kev 10 Kev > E > 100 ev	0.1 0.1 100

Notes

[a] These are estimates

[b] This includes electron bremsstrahlung produced in all parts of the spacecraft and in any detector

Some instruments may be usable if the spacecraft-produced radiation is no higher than 5 times the values listed in Table 4-36. In these cases, the level must be known to within ±20 percent and must be reproducible to this accuracy. The possibility of raising the artificial radiation to this level must be negotiated for each radiation detector aboard.

If the design of the spacecraft requires the use of a radioisotope thermal generator for the generation of power, then the ability to measure the spectrum of interplanetary and planetary particles will be severely limited. The following cases using two types of radioactive material under consideration illustrate this problem.

1) 500 w generator using Pu 238; generator $\geq$ 1 m from detectors.
   a) Heavily charged particles can be measured by presently developed instruments such as the ion chamber or geiger counters; however, elaborate spacecraft calibration will be required. In addition, the performance of these instruments will be degraded.
   b) The large neutron flux associated with the radioisotope generator may cause undesirable activation.
   c) Heavy charged particles can easily be measured by solid state telescopes.
   d) Natural neutrons, electrons, and gammas cannot be studied.
   e) The effects of neutron activation must be carefully studied.
2) 500 w generator using Cm 241; generator $\geq$ 1 m from detectors.
   a) Heavily charged particles can be measured only by advanced systems using solid-state detectors with coincidence outputs.
   b) Other natural radiation cannot be studied.
   c) Neutron activation must be carefully considered.
   d) Elaborate calibration of spacecraft system will be required.

(b) Spacecraft Magnetic Flux Constraints. Satisfactory measurement of the magnetic flux requires that the total field of the spacecraft at the magnetometer sensor should be no greater than 1 gamma. This requirement can be satisfied if each assembly or subassembly meets the following requirement:

1) Static Fields
   a) The magnetic field of an assembly or subassembly (due to perm fields and current-loop fields) in the frequency range of DC to 0.1 cps shall be less than 1.0 gamma at 1 m.
   b) The stability shall be such that the field shall not change by more than 0.1 gamma (at 1 m) or 10 percent of the original field (whichever is less) throughout the environmental testing of the assemblies, and the field shall not change by more than 0.1 gamma (or 10 percent, as above) between the various modes of operation.

2) Dynamic Fields

a) The spectral magnetic field density of an assembly or subassembly in the frequency range of 0.1 cps to 10^3 cps shall be less than 0.01 gamma rms per cycle of bandwidth at 1m.

b) This dynamic field shall be stable in that it never exceeds 10 times its nominal value at any time throughout its life.

c) The dynamic magnetic field of frequencies greater than 10^3 cps is not of importance.

The reduction in the total magnetic field of the spacecraft will be of significant value in maintaining attitude control of the spacecraft when passing through any magnetic field associated with the planet.

If the magnetometer sensor were deployed from the spacecraft by a boom or possibly by a subsatellite, the spacecraft magnetic constraints could be relaxed. This will greatly facilitate the spacecraft magnetic control problem.

(c) Temperature Constraints

		Operating (°C)	Nonoperating (°C)
1)	Particle Flux	-30 to +70	-50 to +90
2)	Ionization Chamber	-30 to +125	-50 to +125
3)	Trapped Radiation Detectors	-30 to +65	-50 to +65
4)	Medium-Energy Proton Directional Monitor	-30 to +50	-50 to +65
5)	High-Energy Proton Directional Monitor	-30 to +50	-50 to +65
6)	Cosmic Ray Spectrum Analyzer	-40 to +50	-50 to +65
7)	Plasma Measurement (Electrostatic Analyzer)	-20 to +80	-55 to +100
8)	Plasma Measurement (Faraday Cup)	-20 to ±100	-55 to +100
9)	Bi-Static Radar	-30 to +80	-50 to +85
10)	Magnetometer		
	sensor	-30 to +100	-50 to +100
	electronics	-30 to +100	-40 to +100

(d) View Angle Orientation Requirements. The orientation of these instruments will be negotiated when the final payload is selected. In general, it is possible to satisfactorily orient these instruments without affecting the major structure. This is possible because the instruments are usually light and manipulatable, making attachment to the outer perimeter quite feasible.

The view angles, however, cause more difficulty because each view must be unobstructed. Thus, with angles varying from a few degrees to almost 2π steradians, early consideration must be given to providing for this requirement during the design of the spacecraft.

(e) Data and Sequencing Requirements. The particles and fields experiments will collectively generate approximately 100 bps minimum during the interplanetary phase of the mission. During the orbit phase, the rate is expected to vary from 20 to 100 bps. The total amount of data from these experiments will remain relatively constant, but the rates from individual experiments will vary as a function of the data content. In essence, this means that the operation of the experiments will be controlled by the measured quantities within the data and variations may require different sequencing and data modes depending upon the measured values or subsequent ground commands. If greater communication can be allocated its wider transmission bandwidth can be effectively utilized by this set of experiments during any period of the mission.

(4) Micrometeroid Instruments

1) Cosmic Dust	(Interplanetary) To measure the flux of cosmic dust particles as a function of direction, distance from the Sun, and momentum with respect to the spacecraft.
2) Micrometeoroid Detector	(Interplanetary) Direct measurement of the velocity and cumulative mass distribution of the cosmic dust of the Zodiacal cloud in the vicinity of Mars.

(a) Temperature Constraints

	Operating (°C)	Nonoperating (°C)
1) Cosmic Dust		
sensor	-50 to +100	-75 to +100
electronics	-40 to +100	-50 to +100
2) Micrometeoroid		
sensor	-50 to +100	-75 to +100
electronics	-40 to +100	-50 to +100

(b) Orientation and View Angle Requirements. Since the measurements will be omni-directional, the spacecraft should obstruct as small a subtended angle as possible with respect to the instrument. The sensors will be oriented with respect to planet vertical and solar coordinates, looking in the plane of ecliptic and in the direction of and opposite to the velocity vector.

(c) Data and Sequencing Requirments. The instruments are designed to operate continuously during cruise and in orbit. During transit, the data can be sampled upon the recording of an event and transmitted in real time; however, while orbiting the planet, the data will be sampled in a fixed format, stored with other data, and transmitted at appropriate intervals. At a resolution of 20 bits/impact, about 1000 bits are expected during cruise and about 3 by 10^4 bits are expected during orbit.

4. Capsule

a. Scientific Measurement Requirements of the Capsule

Table 4-37 specifies weight, power, data requirements, and general constraints for a typical capsule scientific payload. These requirements will be covered in more detail in the following sections. Table 4-38 summarizes the radiation constraints.

b. Tentative Power Profile for Capsule Scientific Payload

Figure 4-129, a typical power profile, illustrates the use of a particular power-switching method which assumes that all instruments will be activated immediately after landing. Although this results in a high peak power requirement during the first day, it is based on the fact that the less time an instrument operates, the more reliable it is. Only those instruments which require more time will operate after the first day. If a radioisotope thermal generator system is used, it may be desirable to operate all instruments continuously. This may simplify temperature control and mechanization.

c. Typical Capsule Instrument and Constraints

The instruments listed under each of the following measurement categories are representative of the types available to meet the scientific mission objectives. From these instruments and others not listed, the selection of the final scientific payload will be made.

(1) Biology

1) Radioisotope Growth Detector

Uses radiosotopic techniques for detection of metabolic and growth process. Carbon 14 included in a liquid media is converted to $C^{14}0_2$ which is detected with a Geiger-Mueller tube or solid state detector. Control experiments are performed as a contrast to the actively metabolizing experiment.

Table 4-37. Scientific Measurement Requirements of the Capsule

Measurements	Weight (lb)	Power (w-regulated ±5%)	Energy (w-hr)	Data Requirements	Capsule Requirements
Biology	28	21	2980	1×10^8 bits total for two weeks survival	1. Radiation constraints as stated in included section 2. Operating lifetime of two weeks required 3. Soil plus atmospheric sample required 4. Top of capsule should have a 360 deg unobstructed view of the Sun after landing 5. Any mechanism used for slowing descent should be detached upon landing 6. Additional heater power required during Martian night
Geology	32	23	1100	1×10^8 bits total for one month survival	1. Must obtain soil sample. Can be from the top surface layer 2. One month minimum lifetime required 3. Capsule orientation required
Atmospheric	17	12	2950	3.1×10^5 total	1. Must start atmospheric sampling at highest possible altitude during descent 2. Atmospheric sampling required after landing 3. One month lifetime on surface desired 4. Radiation constraints as stated in included section
Atmospheric (cont)					5. Temperature control of operating instruments required during descent 6. Erection of sensor will be required
Surface Surveillance	20	20	75	5×10^8 bits for 360 deg coverage with 2 cameras. (Partial stereo included)	1. Cameras or mirrors must be erected to height which will allow their view angles to be unobstructed, and also allow 360 deg coverage 2. Orientation of capsule required
Soil and Atmosphere Sample Acquisition and Processing	20	10	20		1. Two to three ports are required on opposite sides of the capsule 2. Will satisfy biological, atmospheric and geological sample requirements 3. Orientation required
Sounds	3	3	1080	5×10^7 bits total for two weeks survival	1. Should be outside capsule wall if capsule cover is left on 2. Two weeks lifetime desired
Totals	121	90	8205	7.5×10^8 bits	

Table 4-38. Capsule Instruments Radiation Constraints

Experiment	Beta	Gamma	Neutron Flux	Soft X-rays	Alpha	Protons	Remarks
1 Radioisotope Growth							
a Detectors	x						1.5 Kev max background Beta
b Media			x	x			10^6 Rad maximum allowable with Cm 244
2 Turbidity and pH Growth			x				Cm 244 flux should be reduced to 1 Rem/hr
3 Multiple Chamber Biochemical			x				40 Rems/hr for 10,000 hours maximum allowable with Cm 244
4 X-ray Diffractometer							Information not available
5 Petrographic Microscope							Radiation clouding of plastic sample holder
6 Simple Composition							
a H_2O / b O_2	x	x			x		Alpha background should be less than 1% of expected levels. Background Beta should be less than 1% of expected. Tolerable gamma radiation depends on detection efficiency
c N_2		x				x	Background proton flux should be less than 1% of expected. Gamma background depends on detection efficiency
d A_r		x					Gamma background should not exceed 1 gamma/cm^2/sec in the instrument
7 Mass Spectrometer		x					Ion current produced by Cm 244 inducing 0.1 Mev gamma by secondary photo-electrons and 2 Mev gamma by secondary compton electrons. Pu 238 produces ion current by photo electrons
8 Gas Chromatograph		x					Same as Mass Spectrometer
9 Density		x					Same as A_r constraints
10 Vidicon		x	x	x			Radiation damage to photosensitive surfaces. Specific levels not available
11 Sounds							No information available
12 Soil & Atmospheric							No information available

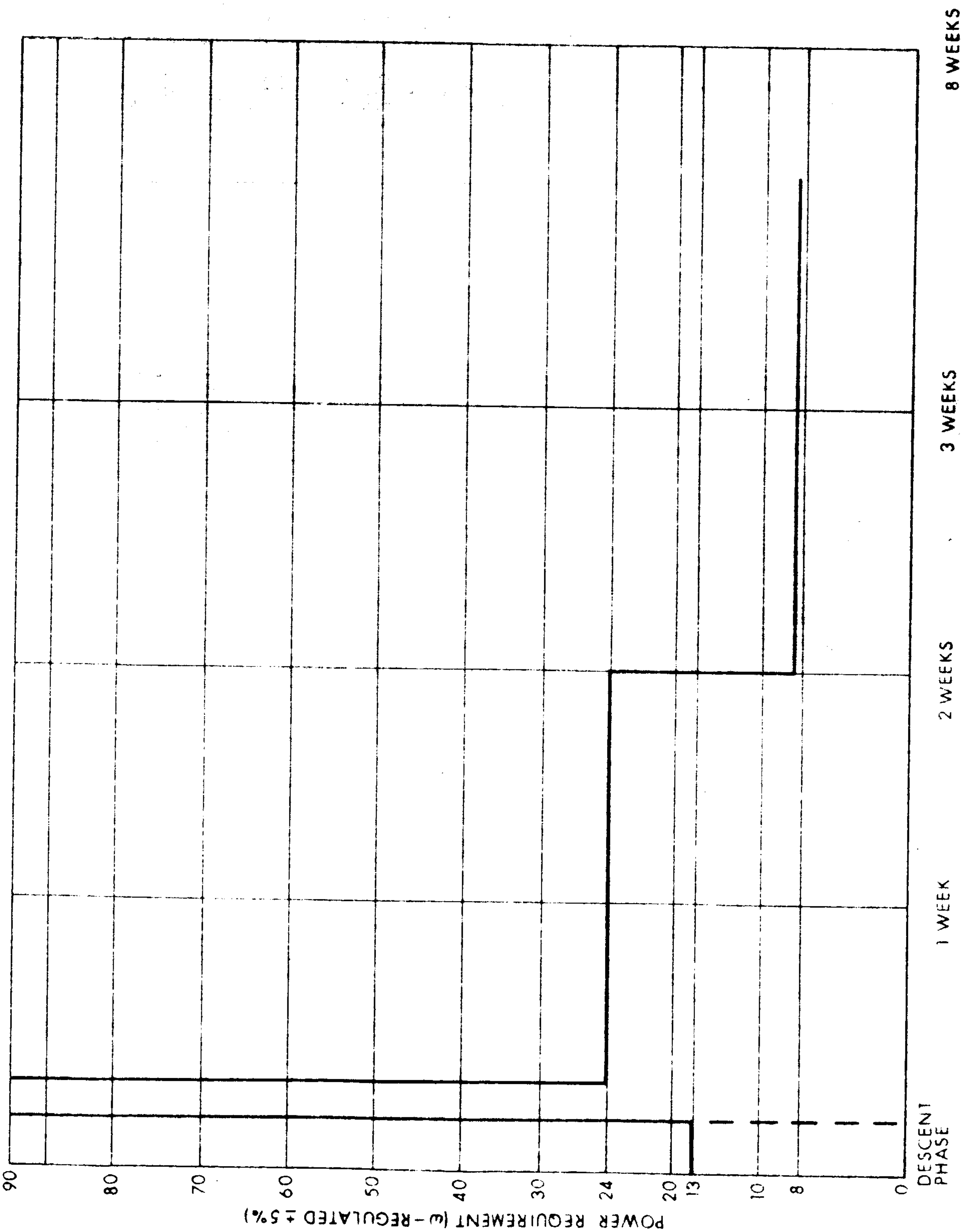

Figure 4-129. Tentative Power Profile for Capsule Scientific Payload

2)	**Reduction-Oxidation Potential Experiment**	Multiple experiment designed to measure changes in Eh (redox potential) in which organisms are actively metabolizing and, therefore, changing the redox potential of that media. Large numbers of samples can be run simultaneously.
3)	Microscope	Visual observations of the morphology and optical characteristics of small organisms. This could be used as a specialized instrument for microcolorimetry, spectrophotometry and fluorescence.
4)	Organic Gas Chromatograph	Detection of organic compounds and possibly organisms by measurement of the volatile organic substances and degradation substances from pyrolyzed samples. Characteristic "fingerprints" of micro-organisms may be found.
5)	Turbidity and pH Growth Detector	Changes in optical density and other light scattering phenomenon plus changes in pH are used to detect growing organisms in a transparent liquid culture media.
6)	Multiple Chamber Biochemical Experiment	Multiple, versatile, miniaturized biochemical laboratory capable of detecting reactions characteristic of biological life. At present, the multiple chamber is used to measure enzymatic hydrolysis using a sensitive fluorescent technique. Capable of carrying out growth and control experiments.
7)	Stain Experiment	Selective staining of soil particles to provide a positive identification of organic materials present. The capability may be extended to determine whether or not the organic material is living.
8)	Ultraviolet Flux	Determination of the integrated ultraviolet flux at the surface as a function of time.

(a) Radiation. In this instrument category, there are two possible problem areas. One is in the detectors used to measure radiation and the other is in the organic media used to grow or operate on bacteria. Some of the typical instruments in this category are listed below with their specific radiation tolerances.

1) **Radioisotope Growth Experiment.** In this instrument, both the problems appear because a detector and an organic growth medium are used. The tolerances are as follows:

   a) The radiation detector can tolerate a maximum background beta energy level of 1.5 Kev. The sensitivity to other radiation is not known at this time.

   b) Many of the proposed media will contain organic compounds in the liquid or solid state. These are subject to decomposition by radiation, i.e., gammas, neutrons, and soft X-rays. Sensitivity varies depending upon the molecular bonds and the bond energy. In general, significant changes do not begin to occur until approximately 10^6 rads are reached. In this case, these substances will be exposed decomposition radiation for the duration of the mission and will be subject to the cumulative changes. Only in the case of the Cm 244 is 10^6 rads approached. It does not appear that Pu 238 would cause any problems.

2) Turbidity and pH, Growth Detector. The growth media which may be used in this instrument have the following radiation tolerances:

   a) SR 90 is not acceptable.

   b) The unshielded neutron flux from Cm 244 power generator would have detectable, and possibly deleterious, effects on the culture of micro-organisms. The flux should be reduced to 1 REM/hr to assure no interference with the experiment.

   c) The neutrons from the Pu 238 generator would most likely have no detectable effect.

3) Multiple Chamber Biochemical Experiment. The sensitivity of this instrument's media to the neutron flux can be stated as follows:

   a) Total flux acceptable is 10^{13} neutrons/cm^2.

   b) In a 10,000 hr mission, the total acceptable flux/hr is 10^9 or approximately 40 REMS/hr; therefore, the Cm 244 seems to be marginal and the Pu 238 presents no problem.

(b) Temperature Requirements. Tentative temperature limitations are:

• Nonoperating maximum	+100°C
• Nonoperating minimum	0°C to +2°C
• Operating maximum	+30°C
• Operating minimum	+2°C

The range of minimum nonoperating temperatures cannot be specified until the "state" in which the various media will be transported to the planet can be defined. The operating temperatures indicated are for the media only and are not required for the remainder of the instrument.

(c) View Angle, Access Port, and Orientation Requirements. A minimum of two ports (on opposite sides of the capsule) is required to provide samples of loose surface soil for the biological instrumentation.

Some of the biology instruments contain liquid media through which measurements must be made. These instruments may require orientation to within ±30 deg of the planet vertical.

The ultraviolet flux instrument will most likely use a hemispherical lens to collect energy from the Sun. The lens should have an unobstructed view angle of 180 deg. The plane surface of the lens should be parallel to and within ±30 deg of the planet horizontal.

(d) Data and Sequencing Requirements. The total data requirement for this package will be approximately 1 by 10^8 bits during a two-week period. The largest bulk of this data will be taken during the first day with the microscopic instrument having the highest data rate. The other instruments will have a much lower average bit rate and will operate for lengths of time varying from 4 hrs to 2 weeks.

The majority of the experiments will require a soil sample. Some of the instruments will require only one, while others will require as many different samples as possible.

All instruments should have power applied and squibs fired immediately before and/or after landing. It may be desirable to apply power to the instruments before landing, so that a check can be made on instrument voltages.

(2) Geology

1) X-ray Diffractometer	Identification of minerals and determination of relative quantities and precise composition of mineral assemblages.
2) Seismograph	Study of the internal activity and heterogeneity of the planet.
3) Petrographic Microscope	Textural analysis of rock units. Vidicon pictures will be taken through the microscope.

(a) Radiation

1) X-ray Diffractometer. One system being considered will consist of a radiation detector. Radiation tolerances for this instrument cannot be specified at this time.

2) Petrographic Microscope. One possible instrument problem may be radiation-clouding of the plastic in which the sample is embedded.

(b) Temperature Requirements. Tentative temperature limits are:

- Nonoperating maximum +100°C
- Nonoperating minimum -30°C
- Operating maximum 1) +40°C to +70°C
- Operating minimum 1) -40°C to 0°C

(c) View Angle, Access Port, and Orientation Requirements. Instrumentation within experiment package will require several top-surface soil samples collected from different locations within a small radius of the capsule.

The measurement of seismic activity requires that the orientation of the instrument, with respect to the planet vertical, be known. In the past, a most effective method of providing orientation has been to use a floatation mechanism which automatically aligns itself with the planet vertical.

(d) Data and Sequencing Requirements. The geology package will require two modes of operation. The first mode will be initiated after landing and will continue for approximately 24 hr. During this period, most of the required 1 by 10^8 bits will be taken. The power and data requirements after this time (second mode) will be considerably reduced for the remainder of the mission. It is desirable, but not mandatory, that the major part of the experiment be completed in 24 hr.

Several soil samples will be needed for this experiment, preferably from different locations within a small radius of the capsule.

(3) Atmospheric Analysis

1) Static Temperature	Determination of the static atmospheric temperature as a function of altitude during descent and as a function of time on the surface.
2) Static Pressure	Determination of the static atmospheric pressure as a function of altitude during descent and as a function of time on the surface.
3) Density	Determination of the atmospheric density profile.

1) A range of operating temperatures indicates the possibility of instruments with different limits.

4)	Mass Spectrometer	Determination of the distribution and quantity of the major constituents of the atmosphere between certain atomic mass limits.
5)	Atmospheric Gas Chromatograph	Determination of the distribution and quantity of the atmospheric constituents over a wide range of concentrations.
6)	Anemometer	Determination of the short term and average wind velocities on the surface.
7)	Simple Composition	Quantitative determination of specific atmospheric constituents.

(a) Radiation. In this instrument category, the majority of the problems will be in the radiation detectors and special components. Problems of typical instruments in this category follow:

1) Simple Composition. The atmospheric composition experiment measures several specific elements by radiation detecting techniques. The following paragraphs describe the energy levels of the particles being detected and specify the radiation levels that can be tolerated from the capsule.

a) H_2O: One method involves measurement of a change in intensity of an alpha particle source as a function of frost layer thickness. Solid state (semiconductor) detectors are used. The counting rates currently used are approximately 10^3 alpha particles/cm^2 sec. These detectors are relatively insensitive to gamma radiation. All that can be specified at this time is that the background radiation be no greater than 1 percent of the level of detected events. The actual allowable flux would depend upon the efficiency of detection for a given gamma ray energy.

b) O_2: One measurement of oxygen involves detection of 0.7 Mev beta particles at a flux of about 10^2 beta particles/cm^2 sec. One percent of this level (1 detected event/cm^2 sec) would be allowable. The actual allowable gamma ray flux again depends upon the efficiency of detection with this crystal-phototube combination.

c) N_2: This measurement involves the detection of protons in the 1 to 3 Mev range. The allowable detectable flux would be about 1 percent or approximately 0.01/cm^2 sec. (The actual allowable flux of gamma rays would be larger, again, because of efficiency considerations.)

d) Ar: One method of detecting argon involves the attenuation of low energy gamma rays (X-rays) in the range of 2.5 to 6 Kev. This measurement probably imposes the most severe restrictions on allowable background gamma ray levels. In this energy range, the intensity level used will be approximately 100 gamma/cm^2 sec with an allowable background level about 1 gamma/cm^2 sec. This is approximately 5 orders of magnitude lower than the lowest energy level of Pu 238.

2) Mars Spectrometer. One hazard to the mass spectrometer is gamma radiation caused by the production of free electrons in the mass spectrometer envelope material (assumed to be stainless steel #304). If these electrons strike the collector plate, they produce a background current of opposite sign to the normal ion current. The lower limit of ion current to be expected is on the order of 10^{-13} amps. Hence, if an electron current of this order of magnitude or higher is generated by the radioisotope power supply, the mass spectrometer would be greatly handicapped.

Calculations have been made on the number of electrons generated in the envelope material (assumed to be 0.16 cm thick) that could penetrate the envelope and strike the collector. The sources of these electrons are the Compton effect, the photo electric effect and pair production. When the energies, mean free paths, and reaction cross sections are taken into account, there are only two significant contributors to the electron current from Cm 244:

a) Secondary photo electrons from the 0.1 Mev gamma

b) Secondary Compton electrons from the 2.0 Mev gamma

The total electron current generated is 6.4 by 10^{-15} amps of which 95 percent is contributed by the 0.1 Mev gamma.

In the case of Pu 238, only photo electrons contribute to the electron current, the value of which is 3.2 by 10^{-16} amps.

Because the values of these currents are minimum, no secondary interactions have been considered. However, in practice, it is likely that electron current could be one and possibly two orders of magnitude higher. Consequently, operation with Cm 244 will be very close to the borderline, while about 10 percent of the minimum ion current will be induced by Pu 238.

If an electron multiplier is to be used with the mass spectrometer (as in the case of an extremely high altitude composition analysis), the radiation from either the Cm 244 or Pu 238 power supplies would overwhelm the mass spectrometer output. For an electron multiplier with 10^{6} gain, one could assume 10 percent of the induced electrons would receive this full gain, generating output electron currents of 10^{-10} amps (Cm 244) and 10^{-11} amps (Pu 238) of the same sign as the amplified ion currents.

A second hazard is gamma activity induced in the mass spectrometer envelope from neutron activation. The main isotopes contributing would be:

- Mn^{56} ($T^{1/2}$ = 2.5 hr, E = 1.15 Mev)
- Cr^{51} ($T^{1/2}$ = 25 days, E = 0.32 Mev)
- Fe^{59} ($T^{1/2}$ = 45 days, E = 1.2 Mev)
- Co^{60} ($T^{1/2}$ = 5.3 yr, E = 1.25 Mev)

If all the induced gamma activity from Cm 244 was converted to secondary electrons and all these electrons struck the collector, a current of 7.9 by 10^{-15} amps would result. This is a maximum and is probably one or two orders of magnitude higher than would actually occur. Under the same assumptions; Pu 238 would generate only 1.2 by 10^{-17} amps. Consequently, the neutrons from either radiosotope power supply would not be harmful.

3) Gas Chromatograph. The gas chromatograph will be faced with problems similar to those of the mass spectrometer.

4) Density. The radiation sensitivity of the density instrument will very similar to the sensitivity of the argon detection instrument.

(b) Temperature Requirements. Tentative temperature limitations are:

- Nonoperating maximum +100°C
- Nonoperating minimum -30°C
- Operating maximum +70°C
- Operating minimum -20°C

(c) View Angle, Access Port, and Orientation Requirements. The instruments used to measure the characteristics (temperature, pressure, and composition) of the atmosphere must have atmospheric samples available during both the descent and post-landing phases. The sample must be representative of that at the capsule when the measurement is being made. To obtain accurate temperature and pressure measurements, oscillations of the capsule after parachute deployment should be held within ±10 deg.

(d) Data and Sequencing Requirements. The total information required during a capsule descent of twenty minutes will be approximately 5 by 10^4 bits. The information received during descent should be transmitted in "real time" and also stored because of the possibility of losing communications during this period. It will be necessary to activate the descent instrumentation during the preentry phase to allow time for measurement preparation and instrument calibration. These instruments will operate after landing for periods varying from 1/2 hr up to the duration of the mission.

(4) Surface Surveillance

- Vidicon System. Observe the structural and temporal relations of the surface upon which the capsule has landed and is performing petrological experiments.

(a) Radiation. Photosensitive surfaces may not be able to tolerate low level penetrating radiation, i.e., soft X-rays, gammas, neutrons, for the extended period of the mission. The optical system which will be used is not known at this time; therefore, no specified tolerances can be stated.

(b) Temperature Requirements. Tentative temperature limits are:

- Nonoperating maximum +100°C
- Nonoperating minimum -40°C
- Operating maximum +40°C
- Operating minimum -20°C

(c) View Angle, Access Port, and Orientation Requirements. The instrumentation must have an unobstructed view between 10 deg above and 50 deg below the local horizontal (local horizontal is defined as being perpendicular to the planet vertical), and throughout a 360 deg scan about the planet vertical.

Two cameras will be used to achieve stereo coverage. In order to obtain optimum stereo, the distance between cameras should be the same as the distance from a point mid-way between the cameras to the point of desired stereo coverage.

The instrumentation will require either a vertical orientation of the capsule or knowledge of the capsule attitude so that correct measurement interpretation can be made.

(d) Data and Sequencing Requirements. The data requirement for this experiment is 5 by 10^8 bits. This information will be generated by covering 360 deg in azimuth and also taking pictures above and below the local horizontal plane. There will be two cameras, and sequencing must be such that both cameras simultaneously take pictures of the same point as the cameras or mirrors rotate through 360 deg. All of the vidicon pictures can be taken during the first day. This would be a desirable mechanization for vidicon reliability purposes.

It would be desirable that data transmission from capsule-to-orbiter be mechanized so that a picture would never be interrupted from day to day, i.e., complete pictures should be sent each day.

(5) Sounds

Audio Receiver	Measure the energy distribution and magnitude within the audio frequency spectrum.

(a) Radiation. The general electronic components will be the most critical portions of this instrument; the circuitry must be designed so that parameter drift and degradation due to radiation will not significantly affect the performance.

(b) Temperature Requirements. Tentative temperature limits are:

• Nonoperating maximum	+100°C
• Nonoperating minimum	-30°C
• Operating maximum	+80°C
• Operating minimum	-20°C

(c) View Angle, Access Port, and Orientation Requirements. If the capsule is landed with the cover or shroud attached, the sensor must be placed outside the capsule wall. The sensor should also be placed in such a position that sounds can be detected omni-directionally.

(d) Data and Sequencing Requirements. The total data requirement for this experiment will be 5 by 10^7 bits. The instrument should be turned on prior to landing so that the sounds occurring during this period can be recorded. The outuput of this instrument must be sampled continuously during the post-landing phase since it is not known when a sound will occur.

(6) Soil and Atmosphere Sample Acquisition and Processing

1) Soil Acquisition	To obtain a soil sample at different locations on the surface and deliver it to the various instruments.
2) Atmospheric Acquisition	Enable atmospheric sampling to proceed during the descent and post-landed phases.

(a) Radiation. The instrumentation development has not progressed sufficiently to permit a quantitative statement to be made at this time.

(b) Temperature Requirements. Tentative temperature limits are:

- Nonoperating maximum +100°C
- Nonoperating minimum -30°C
- Operating maximum +70°C
- Operating minimum -30°C

(c) View Angle, Access Port, and Orientation Requirements. The access requirements of this package have been included in the description of the individual experiments.

The orientation requirements, during both the descent and post-landing phases, are extremely important. It is imperative that uncontaminated atmospheric samples be available during descent and that the soil acquisition device be placed on the surface after landing. Because of these requirements, redundancy (i.e., sampling devices on opposite sides of the capsule) will be used.

(d) Data and Sequencing Requirements. The data of the sampling system will consist of engineering measurements such as voltages, temperatures, indication of squib firing, etc. Squibs will be fired before and/or during descent, and after landing. The various functions of these squibs will be to puncture seals, open and close valves, remove access port covers, eject the soil samplers, etc.

The soil acquisition system may consist of several different devices operating in parallel. The two methods presently being considered are differential pressure and sticky string systems. Other techniques are being explored. The reliability of obtaining a soil sample is increased by using a movable device, which itself must be highly reliable and have the ability to cover a large surface area.

I. SPACECRAFT DATA HANDLING

1. Introduction

a. Purpose of Data Handling Study

This study was undertaken to investigate the requirements placed upon Voyager-class data handling systems and to reveal problem areas in sufficient detail to meaningfully guide future study and design efforts. In many cases, the requirements imposed had to be hypothesized, based upon engineering judgment and extrapolation of present requirements to 1969. System mechanizations are proposed for the primary purpose of revealing problem areas and arriving at realistic weight and power estimates. A secondary reason for proposing these mechanizations is to advance thinking into new methods for solving the data handling problem for this and other classes of future missions. The main areas of concentration in this study are the capsule and orbiter missions after separation of the capsule. Time limitations have not permitted sufficient focusing on pre-separation data handling aspects; however, it is felt that pre-separation data handling will be relatively straight forward if post-separation problems can be solved.

b. Scope of Data Handling Study

This study includes the following:

1) The prediction of specific data, mode and storage requirements based upon estimates of orbiter-capsule configurations and mission requirements.
2) Possible trade-offs between requirements and capabilities.
3) The outlining of a set of general functional requirements which the data handling system must satisfy.
4) A discussion of data-handling design philosophies and intra-system trade-offs.
5) A presentation and discussion of data-handling-system functional block diagrams for the following cases:
   a) Orbiter
   b) Capsule relay link
      - Option I no capsule external control
      - Option II capsule external command control
   c) Capsule direct Earth link
6) Weight and power estimates
7) Problem areas, which requires solution prior to realization of system mechanization, and areas for future investigation. Discussions of these subjects are presented separately in Chapter 6 of this volume.

c. Definition of Terms

In order to clarify the discussions in this report, the following definitions will be be assumed:

1) Signal. A voltage, current or impedance time function.
2) Measurement. A signal at a test point or instrument output. May also be used in the context of what quantity is being measured.
3) Instrument. At a minimum, a transducer, but may also include signal conditioning elements.
4) Data. Signals in any form which represent measured phenomena.
5) Information. That part of data which contains what is to be learned about the measurement or measurements.
6) Data Conditioning. An information-preservation of data retrieval.
7) Sync. Signals which remove ambiguity in the data such that the data will correctly represent the measured phenomena.
8) Channel. A medium of transmitting signals from one point in space to another (e.g., wire, RF link, etc.).
9) Multiplex. The process of combining a large number of channels into a smaller number of channels. In this report, multiplexing will refer primarily to the process in the time domain.
10) Commutation. A multiplexing process which time-shares input signals.
11) Mode or Program. A scheme of processing data.
12) Storage. A medium for preserving signals or data which may also be stated as data storage, program storage, etc.
13) Data Compression. A processing scheme to remove redundancy from data so that it will approach its information content, thereby reducing required data rates. Information is used here as defined above.
14) Conversion. A term used to describe the functions of multiplexing, conditioning and quantizing exclusive of power equipment, program storage and processing, and data storage.
15) Performance Measurement. A term used to describe a measurement which is in one of the following categories:
    a) Operational. Measurements which are vital to vehicle operation. Foreseeable use with high probability in a closed-loop vehicle-Earth-vehicle decision process which will affect the state of the vehicle. Does not imply high frequency of usage.
    b) Evaluation. Measurements which monitor performance to verify design in the space environment, accompany scientific experiments to aid in their interpretation, provide feedback to future designs, and assist in fault evaluation.

2. General Assumptions Affecting the Data Handling Study

The problem of iteration required between communication capabilities and data requirements poses a problem somewhat analagous to the chicken and egg riddle. Therefore, the method of approach will be to first estimate data, mode, and storage requirements as independently as possible from the capability of the communications channel. It should be mentioned that performance requirements are best specified in terms of a rate (bits per second) and most experimental data requirements by total information (bits). Assumptions for estimating data, mode, and storage requirements are stated in the body of the report as needed.

An investigation of the ability of the communications channels to support the requirements will then be made, but a reiteration of capabilities and requirements will not be attempted. Instead, possible trade-offs will be investigated in those areas where capabilities and requirements are not compatible. Investigation of channel capabilities is based on the following assumptions:

a. Capsule-Orbiter Communications Relay Channel

1) 50 and 25 w transmitter power levels and other communication parameters as outlined in Section II (G). During the period from capsule separation to bus-retro, the two transmitter power levels provide capabilities of 377 and 160 bps, respectively, at an assumed maximum range of 36,000 km. There is a high probability of channel interruption during the descent phase. During the orbital phases, 690 and 330 bps rates, respectively, will be available during periods of mutual visibility, based on a maximum range of 15,000 km over all reasonable dispersions of elliptic orbits. Bit rate is considered adjustable under Option II operation.
2) Bit-synchronous telemetry channel.
3) Remodulation of the orbiter telemetry subcarrier after bit-by-bit detection of the capsule data, provided the data rate of the capsule does not exceed the orbiter data-rate capability.
4) Possibility of receiving true or complemented capsule data upon Earth receipt because of the detector mechanization outlined in Section II (G).
5) 150 day design lifetime.

b. Orbiter-Capsule Command Channel

1) Only available under Option II.
2) Discrete commands during mutual visibility.

c. Capsule-to-Earth Communications Channel

1) Data rates of 2 bps based upon a 50 w transmitter power level and other parameters as outlined in Section II (G). A rate of 0.2 bps is also considered for comparison purposes.
2) Mutual visibility between Earth and capsule at least 25 percent of the total time.
3) Bit-synchronous telemetry channel.
4) Possible receipt of true or complemented data upon Earth receipt.
5) 30-day design lifetime.

d. Earth-to-Capsule Command Channel

- Not available.

e. Orbiter-to-Earth Communications Channel

1) 79 dbm spacecraft power-gain product with corresponding bit rates as outlined in Section II (G).
2) 24 hr/day DSIF coverage from Encounter -5 days to Landing +30 days; thereafter, a minimum 8 hr/day uninterrupted coverage until end of mission at 150 days.
3) Earth occultation possible at anytime after encounter. Times of possible occultation are not available except for nominal orbits.
4) A bit- and word-synchronous telemetry channel.

f. Earth-to-Orbiter Command Channel

- Quantitative and discreet command capability at 20 bps to Encounter +150 days, during periods of mutual visibility.

g. Mission Geometry, Ranges, and Times

- Information as outlined in Section II (C). The time between capsule parachute deployment and bus periapsis passage (defined as Δt in Section II (C) is assumed to be 2 hr. Other assumptions will be stated as required in the body of this report.

3. Data Handling Requirements and Channel Capabilities

a. Capsule-to-Orbiter Link

(1) Performance Data. To estimate capsule performance data requirements, a study was conducted utilizing existing Mariner R, A, and C telemetry lists. The purpose of

this study was to obtain realistic estimates of future data requirements. The results of the study are used frequently in this report, but should not be considered final nor dictating concerning firm system mechanization.

A basic capsule system was hypothesized and the above telemetry lists examined for possible measurements which could be included. Additional measurements which would be pertinent to the capsule system were also considered. It was felt that predicting the number of measurements on a measurement-per-pound basis would be unrealistic, since attitude control and sequencing systems will be simple and major allocations of weight will be given to the propulsion, heat shield, and parachute systems. Assumptions for the capsule systems are as follows:

1) Capability of receipt of discrete commands. However, the data rate requirements are relatively insensitive to the inclusion of command capabilty.
2) No shut-off of units such as transmitters, receivers, and science monitoring.
3) A capsule separation maneuver which includes spin orientation and a capsule motor-burn duration of no more than 600 sec.
4) An elementary sequencing system for initiation of timed events.
5) Measurement of the local planet vertical after landing.
6) Seven capsule mission phases as follows:

   I. Preseparation.
   II. Separation, which includes motor burn and spin orientation.
   III. Preentry, commencing upon completion of separation phase.
   IV. Entry, commencing upon detection of the Martian atmosphere. Detection of Martian atmosphere is considered to be that point where the fine accelerometer commences measuring a finite deceleration.
   V. Descent, from parachute deployment to capsule landing.
   VI. Capsule landing to bus preretro maneuver.
   VII. Orbital.

   a) Beacon out-of-lock (Option II), not present with Option I.
   b) Beacon in-lock (Option II) or internal indication with Option I.
7) No additional redundancy of capsule hardware except as explicitly outlined in this volume.
8) Occurrence time for certain event measurements.
9) No data compression.

Table 4-39 summarizes the number of capsule measurements by subsystem. Measurements may vary from 1 to 40 bits. The total number of measurements may seem somewhat large but a great many of the measurements are only 1 or 2 bits. Subsystems and types of measurements are grouped in Table 4-39 as follows:

1) Communications. Measurements of VHF and S-Band transmitters and beacon receiver with associated power converters.
2) Power. Radioisotope thermoelectric generator system and primary converter measurements, but not subsystem power converter measurements.
3) Propulsion and Orientation. Motor measurements, velocity increments, spin and despin measurements, and associated events.
4) Engineering Mechanics. Heat shield measurements, temperature control measurements, structural and parachute monitoring, planet orientation, and events throughout the capsule mission.
5) Sequencing subsystem (SOCC). Events and occurrence times where necessary.
6) Data Handling and Scientific Instrument Subsystems. Scientific instrument performance data (less calibration sequences), voltages, temperatures, instrument temperature control, and reference data. Mode and sync information, which is dependent upon the organization of the data handling system is not included.

Table 4-39. Capsule Performance Measurement Summary

Type of Measurement	Number of Measurements	Bits for complete read out
Communications	36	196
Power	26	152
Propulsion and Orientation	20	143
Engineering Mechanics	26	110
Sequencing	4	48
Data Handling and Scientific instruments	60	284
Totals	172 measurements	1033 bits

Table 4-40 lists data-rate allocation to subsystems as a function of the capsule mission phases. The table shows the bandwidth which a data user may be allocated though it may not be indicative of maximum requirements for two reasons. First, the requirements may change during preliminary design, and second, requirements may increase during the mission itself if the data indicates failures or abnormalities. In the absence of data compression, it is conceivable that the user's requirement could not be satisfied, especially in the latter case. The highest sample rate was chosen to be 1 bps, and the number of measurements requiring this rate in any phase was considered to be a maximum of two.

Table 4-40. Capsule Performance Measurement Data Rate Allocation vs Mission Phases

Phases	I Pre-separation	II Separation	III Pre entry	IV Entry	V Descent	VI Landing	VII Orbital Out of-Lock	VII Orbital In-Lock
Communications	1.96	1.9	[illegible]	11.9	11.9	11.9	0.12	1.46
Power	1.52	14.7	1.52	14.7	14.7	14.7	1.47	1.47
Propulsion	0.06	63	0.06	0	0	0	0	0
Engineering Mechanics	0.22	0.22	0.22	11.0	11.0	11.0	0.1	0.1
Sequencing	0.24	0.24	0.24	0.24	0.24	0.24	0	0.24
Data Handling and Scientific Instruments	3.50	3.50	3.50	35.0	35.0	35.0	3.50	3.50
Totals	7.5	83.7	6.9	73	72	73	5.2	6.8

(2) Experimental Data

(a) Accelerometer. A group of accelerometer measurements may be flown on the capsule. This group is considered to be of an experimental nature although they will be of value in assessing the performance of capsule engineering subsystems. It is assumed that the experiment will be complex; that is, it will include two measurements (coarse and fine), each with three accelerometers situated on orthogonal axes, with provision to detect peaks in the absolute value of resultant vectors.

(b) Scientific Experiments. Scientific data requirements will commence prior to entry where calibration data will be taken. During entry, scientific experiments will be oriented towards taking meaningful atmospheric measurements. Sample rates are scaled for a minimum entry time of 15 min. Upon landing, the arrival of scientific data into storage can be high enough for the data rate required on the channel to be considered as an impulse function of weight 7.5 by 10^8 bits. (See Section II (H) for details of capsule scientific experiments.) The data rate allocations for capsule experiments are shown in Table 4-41.

Table 4-41. Capsule Experimental Measurement Data Rate Allocation

Phases	I Pre-separation	II Separation	III Pre-entry	IV Entry	V Descent	VI Landing	VII Orbital
Accelerometer	0	0	60 (a)	60	30	0	0
Calibration	0	0	2 (b)	10	0	0	0
Science Experiment	0	0	0	40	40	40	Impulse Function of Weight 7.5×10^8 bits
Totals	0	0	[illegible]	[illegible]	70	40	Impulse Function of Weight 7.5×10^8 bits

Notes: (a) [illegible] Mars atmosphere
(b) [illegible] Mars atmosphere

(3) Comparison of Total Data Requirements and Channel Capabilities. Table 4-42 lists total data requirements for the capsule-orbiter relay link along with channel capabilities at the 25 and 50 w transmitter power levels. Sync requirements are considered to be about 15 percent. Thus, the combined data requirements during the orbital phase consists of the impulse function of weight 8.6 by 10^8 bits commencing at landing plus a constant requirement of 6 (or 5) bps during the period of time it would take to read out 8.6 by 10^8 bits after capsule landing.

Table 4-43 shows the assumptions and necessary parameters for computation of capsule relay channel capability after landing. The table is based on the nominal elliptic orbit and on dispersions about this nominal covering the first five days after landing. These data were used in making up the table and were assumed to be a valid approximation to the parameters for the duration of the landing period. Some data in Table 4-43 will be used later to compute storage requirements for the capsule relay system. Option II was used for computing channel capability. Data capability for Option I is dependent upon the type of sequence generation aboard the capsule and possible orbit dispersions. This capability could range from the maximum obtainable under Option II to possibly less than half that amount. Some considerations for Option I are discussed later in this subsection. A complete assessment of channel capabilities under Option I requires further investigation.

Figure 4-103 shows a graphic comparison of data requirements and channel capabilities after capsule landing and includes the results of Tables 4-42 and 4-43.

Table 4-42. Total Capsule Relay Link Data Requirements and Channel Capabilities (BPS)

Phase	I Pre separation	II Separation	III Pre entry	IV Entry	V Descent	VI Landing	VII Orbital Out of Lock	VII Orbital In-Lock
Performance Data	7.5	83.7	6.9	[illegible]	73	[illegible]	[illegible]	[illegible]
Experimental	0	0	62	130	70	40	Impulse Function of Weight 7.5 × 10^8 bits	
Sync (15%)	1.1	13.5	11.3	[illegible]	22	1?	1 Impulse Function of Weight 8.? × 10^8 bits	
Total	[illegible]	97	80	[illegible]	165	130	8 Impulse Function of Weight 8.? × 10^8 bits	
Channel Capability at 25 w	Orbiter Rate	160	160	160	160	160	330	
Channel Capability at 50 w	Orbiter Rate	377	377	377	377	377	690	

Table 4-43. Parameters and Assumptions for Computation of Capsule Relay Link Channel Capability After Landing Under Option II for 25 w and 50 w Transmitter Power Levels

1. Capsule on equator oriented to planet vertical. 30 deg horizon in all directions, limited by antenna cone-angle only.
2. Elliptic orbit. (See Section IV, B.3. for a complete discussion of elliptic orbits and their expected dispersions.)
3. Design range of 15,000 km (high dispersion)
4. Total orbiter availability time above 30 deg and within design range:

Maximum	813 min (right dispersion)
Minimum	658 min (low dispersion)

5. Percent of total time orbiter is available:

Maximum	11.3%
Minimum	9.0%

6. Time between orbiter availabilities:

Maximum	48 hr
Minimum	4 hr

7. Availability times:

Maximum	300 min
Minimum	20 min

8. Time from landing to first availability:

Maximum	561 min (high dispersion)
Minimum	369 min (low dispersion)

9. Bit rate capability during orbiter availability:

25 w	330 bps
50 w	690 bps

10. Channel capability equation:

    (Channel rate (bps) during orbiter availability) × (8.65 × 10^4 sec/day) × (% time available) × (days after landing)

(4) Sequence of Operation, Mode and Storage Requirements

(a) Phase I - Preseparation. Provision must be made for telemetering through the bus umbilical. There should be at least two capsule status checks sent prior to the separation maneuver. One should be prior to, and the other after quantitative commands are loaded into the capsule.

(b) Phase II - Separation. During this phase, there will be a problem of communicating at close ranges because of high input power to the orbiter receiver. The problem has two possible solutions: the first is to transmit in real time from the capsule and attenuate the received signal on the orbiter until some preset time; the second is to delay transmission from the capsule until some preset time. In either case, there is a good possibility of interruption of the channel at a time when vital data are needed. During this phase, the mode required should be transmission in real time and simultaneous storing. Some high-rate data which the channel cannot support in real time may have to be stored for later

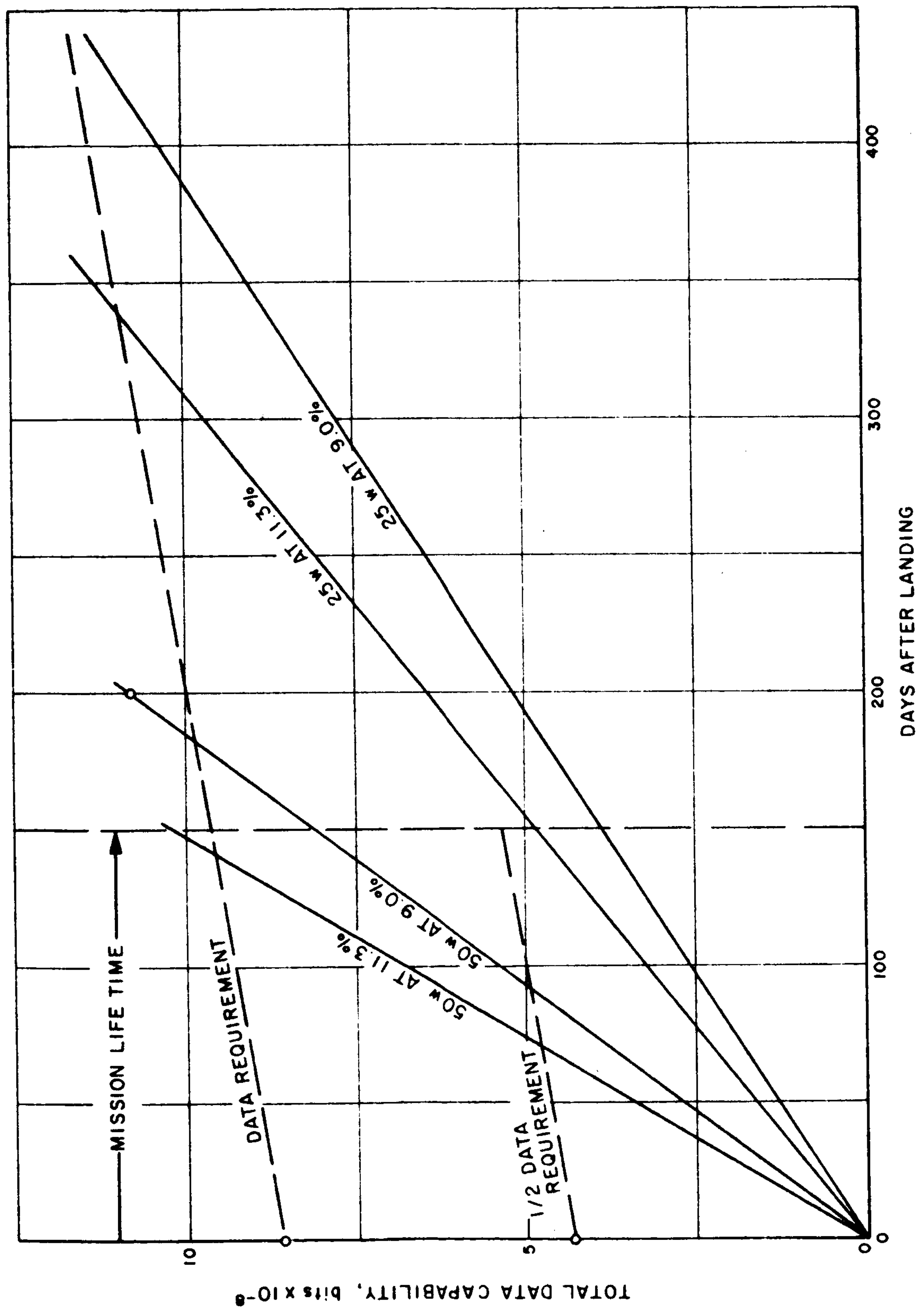

Figure 4-130. Relay Link Channel Capability and Data Requirements for 25 w and 50 w Transmitter Power and for Maximum and Minimum Available Communication Time

transmission. Redundant transmission from storage of data previously transmitted in real time can be made after the maneuver. To aid in reconstruction, these data should preferably be transmitted from storage in the same format as it was sent in real time. Any high-rate data not sent in real time can also be transmitted redundantly from storage after the maneuver. Assuming a 600 sec maximum maneuver time, maximum storage requirements could be 60 K bits. There should be at least one status check of the capsule sent after the maneuver.

(c) Phase III - Preentry. Data requirements are low during this phase until a few hours prior to predicted entry. During most of this phase, there is a possibility of the bus not listening to the capsule, or of capsule transmitter shut-down (particularly if a battery is used as a primary capsule power source). However, these considerations will not significantly affect the data handling system. About 2 hr prior to detection of the Martian atmosphere, actuation of entry science instruments and transmission and storage of calibration data is required. Storage required is 5 K bits. Accelerometer instruments will be activated prior to entry in sufficient time to transmit some deceleration profile prior to blackout. Storage of experimental data should commence when the fine accelerometer detects deceleration due to the Martian atmosphere.

(d) Phase IV and V - Entry and Descent. There is a high probability of channel interruption during these phases because of ionization, rapid decelerations, and capsule sway on the parachute. Channel interruption because of ionization is almost certain. Data transmitted during these phases should be stored for later transmission to the orbiter or to Earth via the direct link. Because of the possibility of a capsule failure after landing or its failure to assume a proper orientation on the planet surface, the capsule data handling system should be programmed to transmit as much of the stored entry and descent data as possible <u>prior to landing</u>. This type of mode will place higher rate requirements on the communication link and, at most, could double the existing required rates shown in Table 4-42. Because of rate limitations for both power levels considered, there must be some selectivity exercised in choosing the data to be retransmitted from storage. Specific solutions to this problem are not considered within the scope of this study. However, the mechanization proposed later in this report does account for the possibility of the mode discussed above. To mechanize this mode of operation, two storage elements are provided. One of these is temporary storage which can be used to store selected data for retransmission during descent; the other is of a permanent nature and is used for transmission of entry and descent data to Earth via the direct link. Parallel filling of storage media will considerably reduce the possibility of a complicated addressing system to withdraw selected data from storage during descent. Assuming a maximum entry and descent time of 50 min, storage requirements are 600 K bits for the permanent storage media. The storage capacity for the temporary storage medium is dictated by later requirements imposed during the orbital phase.

(e) Phase VI - Capsule landing to bus pre-retro Maneuver. Because of uncertainty in orbiter operation after the retro maneuver, the information transferred on the capsule-to-orbiter link should be as meaningful as possible. A time duration of 2 hr between parachute deployment and bus periapsis passage will provide a minimum of about 30 minutes before the capsule-to-bus channel is no longer available. If this minimum time is assumed, the capsule data-handling system can remain in the descent and entry mode, continuing to take performance and atmospheric measurements as before, and with the same storage and transmission plan in effect. After about 5 min, the system may revert to a number of possible modes:

1) Completely redundant transmission of stored data collected during entry and descent, interleaved with low-rate capsule status checks.
2) Transmission of post-landing scientific data. Post-landing scientific instruments will be activated upon stabilization of the capsule on the planet surface, and data will be collected in scientific storage media. The time between activation of post-landing scientific instruments and the time at which meaningful data can be transmitted is not specifically known at present. However, a minimum time of 30 min surface time before the bus is no longer available for data receipt demands that consideration be given to this mode.
3) A combination of 1) and 2) above. More specific allocation of the relay link channel to the various types of data considered cannot be made at present. Regardless of the mode chosen, however, scientific storage media must be available to store post-landing experimental data. If entry and descent experimental data were stored in scientific media, the requirement for simultaneous collection of post-landing data and transmission of descent data would result in undue access problems for scientific storage media. To alleviate this problem, experimental entry and descent data can be collected, along with performance data, in the two performance storages previously mentioned. Once the capsule has landed, post-landing experimental data can be collected in scientific storage independent of the mode requirements for the performance storages. Collection of experimental and performance data in the same performance storage during entry and descent also simplifies this problem of retransmission of data prior to the capsule landing.

(f) Phase VIIa - Orbital (beacon out-of-lock). During this phase, limited performance measurements are taken to verify scientific measurements. Both performance and experimental scientific measurements are stored separately. Appropriate time-tagging is especially required to provide correlation between experimental and performance measurements.

(g) Phase VIIb - Orbital (beacon in-lock). Data is multiplexed from storage according to a predetermined plan. For reliability reasons, and to minimize dependence on external commands, data should be multiplexed to include a segment of each scientific measurement rather than devoting long periods to reading out a single measurement. Performance data should be a combination of real time and stored data to provide automatic means for data retrieval in case of a failure of performance data storage. On the first orbiter pass over the capsule, redundant read-out of entry and descent data should be made. Performance data storage required for the landing phase is 1.4 by 10^6 bits (storage of data for 48 hr at an 8 bps rate). These storage requirements do not include experimental post-landing data storage which is on the order of 7 by 10^8 bits. Most scientific storage is sized for high arrival rates over a short period of time, providing the advantage of short instrument operating-time. Possible trade-offs between science instrument operating-time and storage requirements have not yet been made.

(5) Discussion of Possible Trade-Offs. Table 4-42 and Figure 4-130 show the following:

1) A 25 w capsule transmitter power level will satisfy prelanding data requirements except during two phase. The 25 w level limits the available channel data rate during the most critical entry and descent phases. This channel limiting also prevents any retransmission of data prior to, and after, landing unless the data requirements are considerably reduced. The reduction in data-rate required would probably decrease capsule mission value because of less performance data. Scientific mission value would probably not decrease significantly.
2) Only about half the suggested post-landing data requirements are provided by the 25 w level within the 150 day lifetime.

Assuming a 25 w transmitter level on the capsule, there are basically two alternatives to matching data requirements with the channel capability. The first is a reduction in both the prelanding and post-landing data requirements. It was mentioned previously that capsule mission value would probably suffer with respect to performance measurements during the prelanding phases, but this could conceivably be tolerated if the scientific mission value of the capsule could be maintained. However, as shown in Figure 4-130, the post-landing scientific data requirements would have to be reduced by an approximate factor of 3 to conform to the 150 day lifetime capability. It is felt that this may be a significant lowering of capsule scientific mission value.

The second alternative to matching data requirements would be to employ data compression. The most rewarding area for this technique, from the viewpoint of both data requirements and storage reduction, is its application to the capsule vidicon system. The vidicon experiments comprise about 8.2 by 10^8 bits of the total capsule post-landing data requirements. This study assumes a conservative 2 to 1 compression ratio for vidicon data (see Section VI). With this ratio, the post-landing requirements would not be quite satisfied, still necessitating some reduction in basic data requirements. However, it is felt that the further reductions needed would be nominal compared to the sizeable reduction by vidicon data compression. The reduction in required storage capability by vidicon data compression will probably not be significant enough to employ anything other than tape recording units. Thermoplastic storage techniques might be considered an alternate solution but their feasibility is yet to be demonstrated. Tape recording units could be employed redundantly or made much simpler with decreases in required storage capacity due to data compression. System integration problems of tape units should not be excessive since vidicon systems can be functionally independent of the data handling system except during read-out, when buffer units can be employed.

During the descent and entry phases, the most promising application for data compression seems to be in the area of the accelerometer measurements. Accelerometer measurements occupy a sizeable portion of the available channel rate, as shown in Table 4-41. Slow sampling could be utilized for transmission in real time with higher sampling employed in parallel only for resolution of peaks. When peaks occur, they may be stored, along with times of occurrence, for later transmission to the bus. The reduction in performance data-storage during entry and descent would be about 120 K bits. In addition, the required data rate would be lowered by about 50 bps during Phase III and IV and by 25 bps during phase V.

There is a possibility of employing data compression for the remainder of the performance measurements during entry and descent. The required ratio would have to average about 3 to 1 for the 25 w level, because of the necessary reduction in basic rate, and the reduction needed to retransmit entry and descent data prior to landing. With present methods, the complexities may become excessive. It is possible that other performance data compression methods may become available in the near future. If they show moderate increases in complexity when incorporated into a system where reliability, sound organization, and flexibility are major requirements, then their adoption is certainly within the realm of consideration. At present, reduction of basic performance data requirements seems to be the most attractive solution to matching requirements with channel capability during entry and descent.

3) A 50 w transmitter power level satisfies all capsule prelanding data requirements and in addition, provides enough margin during the descent and landing phases to allow for retransmission of entry and descent data. Accelerometer data compression during entry and descent remains a promising technique, primarily for the 120 K bit reduction in storage capacity obtainable.

4) Although a 50 w transmitter power level marginally satisfies capsule post-landing data requirements within the 150 day lifetime, the application of data compression to the vidicon system remains a promising approach. In the vidicon system operation, it is the storage media which must operate over long periods of time. The vidicon instruments and associated data compression equipment will be operating over a relatively short period of time. Data compression will not only reduce the required capacity of the storage media, but will make it possible to read out the accumulated data in less time. Decreased operating time results in reliability increase. If a conservative data compression ratio of 2 to 1 is assumed, it would require about 100 days after landing to fully satisfy capsule data requirements. The prospect of decreasing capsule design lifetime is an attractive one. Further investigation is needed into the gain of capsule scientific mission value with nonvidicon experiments after the approximate 100 day point. If little scientific value is gained with the nonvidicon experiments, then further consideration should be given to shortening the required capsule lifetime with vidicon data compression.

Data compression for the remainder of the performance measurements could also result in considerable decrease in storage capacity, especially during the landing phases, where requirements approach 1.4 by 10^6 bits, for the temporary storage media discussed previously. This requirement is much larger than all other performance storage requirements. Tape recorders with this capacity are available now, but the problems associated with integrating them into a data handling system which requires interrelated operations with its performance storage media makes them appear unattractive. The preferable approach to the performance data storage problem is to reduce sampling rates or the number of measurements, until storage media with better access properties and higher reliabilities are available. Halving the post-landing performance data sampling rate would not be a serious detriment to mission success and performance storage required would be reduced to a practical 700 K bits. For simplicity and reliability, it would also seem preferable to forego data compression during descent and entry phases, especially since data requirements are already satisfied at the 50 w level.

The importance of maximizing the time between capsule landing and bus retro maneuver cannot be overstressed. If the bus fails to gain orbit, this time will be the only time that a relatively high data-rate for capsule data is available. After the bus is out of capsule view, the only link available for sending capsule data to Earth would be the low-rate direct link.

Capsule Phase VII was previously discussed assuming Option II. The essential difference between Option II and Option I is that in Option II the beacon signal specified the orbiter availability, whereas in Option I there must be at least an estimate of the following quantities aboard the capsule:

a) Time between availabilities

b) Time duration of availability

Best range (for data-rate setting)

The situation is depicted in Figure 4-131. For purposes of illustration, assume that the availability profile of the orbiter is perfectly periodic, as shown in the figure. ETA_j and $ETBA_j$ represent estimated time duration of availability and estimated time between availabilities aboard the capsule under Option I. The ETA_j would determine the size of the data blocks to be transmitted from the capsule; the $ETBA_j$ would determine the number of redundant transmissions of a given data block and subsequent updating to a new block for transmission.

The TA_j and TBA_j represent actual times of availability and times between availabilities, θ is some phase offset between simulated and actual orbiter availabilities. Assuming the periodic sequence of orbiter availability as mentioned before, then,

$$TA_j = TA_k$$

and

$$TBA_j = TBA_k$$

then there would be no data lost by reverting to Option I if

$$ETA = TA$$

and

$$ETBA = TA$$

regardless of the phase angle θ. However, if

$$ETA \neq \ldots$$

$$ETBA \neq ETBA_{}$$

and

$k \neq 1$, then if $k > 1$

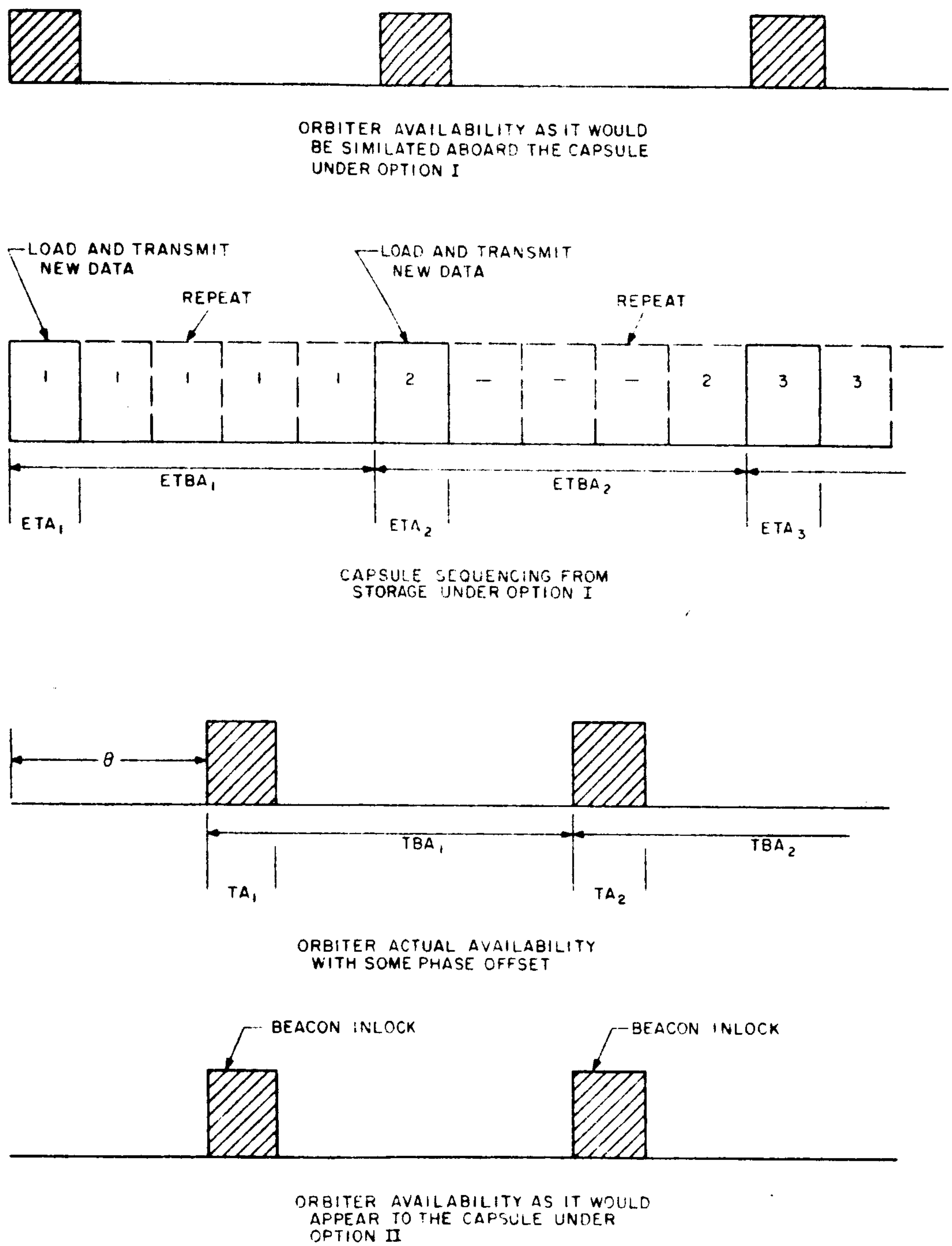

Figure 4-131. Capsule-to-Orbiter Communication Availability

there would be data lost and if $k < 1$, there would be redundant transmission of information along with some possibility of loss when a complete cycle of data blocks fell between two actual availabilities. The data loss would become somewhat dependent upon the phase angle θ on a short term basis, but independent of θ as time approached large values. The problem of computing best estimates of range, TBA, and TA over possible dispersions in landing site, orbiter range and periodocity parameters would be a severe one under the assumption of perfectly periodic availability profiles. However, the problem is compounded by the fact that the orbiter availability profiles will not be periodic; that is, the TA_j and TBA_j are not equal although they are computable for any given orbit. Under realistic assumptions, the problem of forming the best estimates of range, availability times, times between availabilities, and possibly the phase angle, depends upon the complexity of the capsule equipment. It can be safely assumed that the maximum complexity on-board the capsule will be limited to generation of an aperiodic sequence to simulate the orbiter availability profile, with updating of the sequence parameters by quantitative command via the bus prior to capsule separation. For this case the problem is one of finding that aperiodic sequence and bit rate which will minimize the information loss over the ensemble of possible landing sites, coast times, entry times, orbit dispersions and ranges known prior to separation, and mechanizing the sequence generator and selecting bit rates with updating provisions by quantitative command via the bus. If, for simplicity reasons, the sequence generator on the capsule were periodic, then the problem is one of minimizing the information loss by:

a) Forming the best estimate of bit rate over the possibilities above and mechanizing a number of bit rates in the capsule, one of which would be selected prior to separation.

b) Forming the best estimate of TA and TBA over possible TA_j and TBA_j in each orbit and over the possibilities mentioned above, with provision for updating them prior to separation.

c) Forming the best estimate of the phase angle with provision to set the time of activation of the periodic sequence by bus quantitative command prior to capsule separation.

It is not known at this time whether the parameters needed can be computed independently; it seems unlikely that they can. If this is the case, the solution to these estimation problems are possible only with Monte Carlo type analyses. Some considerations in approaching the problem are outlined later in this section, but particular solutions to the problems are considered an area for further study.

The advantages of Option I would be:

a) No VHF beacon receiver and command equipment on the capsule or orbiter would be required.

b) No diplexing equipment or VHF antennas which must be designed for two carrier frequencies would be needed on the capsule and orbiter.

c) The data storage mechanization would be simpler because less data can be transferred.

d) Though not an advantage over Option II, the possibility of using Option I as a back-up for Option II warrants consideration.

e) Sequence generators on the capsule would probably be simple compared to the complex beacon equipment of Option II.

The disadvantages of Option I would be:

a) Estimates of necessary parameters to generate the capsule simulated availability profile would be a difficult problem.

b) Loss of information would be greater than for Option II, especially for the elliptic orbits. The degree of the minimum loss is not known, as also mentioned previously. It should be noted that the problem of availability time is present with both Options. Under Option II the data handling system will know when the orbiter is available, but will have no a priori knowledge of the length of the availability time.

c) Option I would probably require greater bandwidth allocation to sync information.

d) Power could not be duty cycled without great possibility of information loss.

e) Although sequence generators on the capsule may be less complicated than beacon equipment under Option II, the peripheral circuitry to update the generators and to provide a selection of a bit rate at capsule separation offset this advantage.

f) There would be no possibility for command, and therefore all redundancy would have to utilize on-board fault correction.

b. Capsule - Earth Direct Link

(1) Performance Data. A typical set of performance data requirements on the direct link are assumed to be the following:

1) Measurements which can be used to best evaluate the performance of those scientific instruments which utilize the direct link.

2) Direction of the planet vertical with respect to the capsule.

3) Some group of measurements which will give data users a quick look at the status of the capsule. Included in this category would be the following:
   a) Events issued
   b) Events occurred
   c) Capsule identification
   d) Capsule, where needed
   e) A small group (perhaps 5 maximum) of critical measurements which best indicate capsule performance
4) Data stored from the entry and descent phases amounting to 600 K bits without compression.

Table 4-44 summarizes the performance measurement requirements for the direct link.

A data rate summary is not a realistic measure of the direct link requirements since a major part of the data will be entered into storage media with the experimental data and, therefore, depends on the rate of arrival of scientific data into storage. The remainder will be sampled and transmitted in real time and will amount to 100 bits for a complete readout. Assuming 1 readout every 1000 sec the data rate required for this remaining data is 0.1 bps.

(2) Experimental Data. Scientific instrument data requirements are presently considered to be a minimum of 1 by 10^6. This will satisfy the requirements for most of the low-rate scientific experiments envisioned but will not provide capability for the vidicon or high data-rate experiments.

(3) Comparison of Total Data Requirements and Channel Capabilities. Table 4-45 lists total data requirements for the direct link. It is assumed that all scientific data is entered into storage within a minimum of 30 days and that performance data for instrument evaluation will be stored concurrently at a rate of 1 sample every 1000 sec with times of occurrence to provide correlation with experimental data. Sync requirements for this channel are low because of the decreased need for fast acquisition of data.

Table 4-44. Capsule Direct Link-Typical Performance Measurement Summary

Type of Measurement	Number of Measurements	Bits for Complete readout
Scientific Instrument Evaluation	20	140
Planet Vertical	3	21
Capsule Evaluation	25	79
Stored Entry and Descent Data	--	600 K
Totals	48 measurements	240 bits per readout + 600 K bits

Table 4-45. Capsule Direct Link Data - Typical Requirements Channel Capabilities

Measurement	Data rate (bps)	Total data for 30 days (bits)
Scientific Instrument Evaluation	0.14	0.36×10^6
Scientific Instrument	--	1.0×10^6
Descent and Entry	--	0.6×10^6
Real time Performance	0.1	0.26×10^6
Total		2.2×10^6

Based on the parameters listed in Section II (G), Earth will be within the capsule antenna cone about 1/4 of each Martian day. This results in an effective direct-link data-rate which is 1/4 of the actual channel bit rate.

Figure 4-132 gives a graphic comparison of the data requirements and capabilities for a 30 day lifetime, during which the DSIF is continuously available. The channel capability lines in the figure take into account the reduced effectiveness of the channels.

(4) Sequence of Operation, Mode and Storage Requirements.

(a) Preseparation Phase. Provision must be made to obtain a status check of the system through the bus umbilical.

(b) Entry and Descent Phase. Performance and experimental data is gathered and stored by the relay link system. The direct link system need not be activated during this time unless the storage media is designed to be a functional part of this system.

(c) Capsule Landing to End of Mission. The direct link system is activated and stores data from post-landing scientific instruments and samples the required performance measurements. Some of these performance measurements are stored and the remainder are transmitted to Earth in real-time. The descent data storage is read-out nondestructively on the relay link. For simplicity, a fixed time-sharing plan among performance and experimental data should be started during this phase and should be maintained during the remainder of the mission. For the same reasons as for the relay link, scientific data is programmed to contain a segment of all scientific instrument data rather than to devote long periods of time to single instruments.

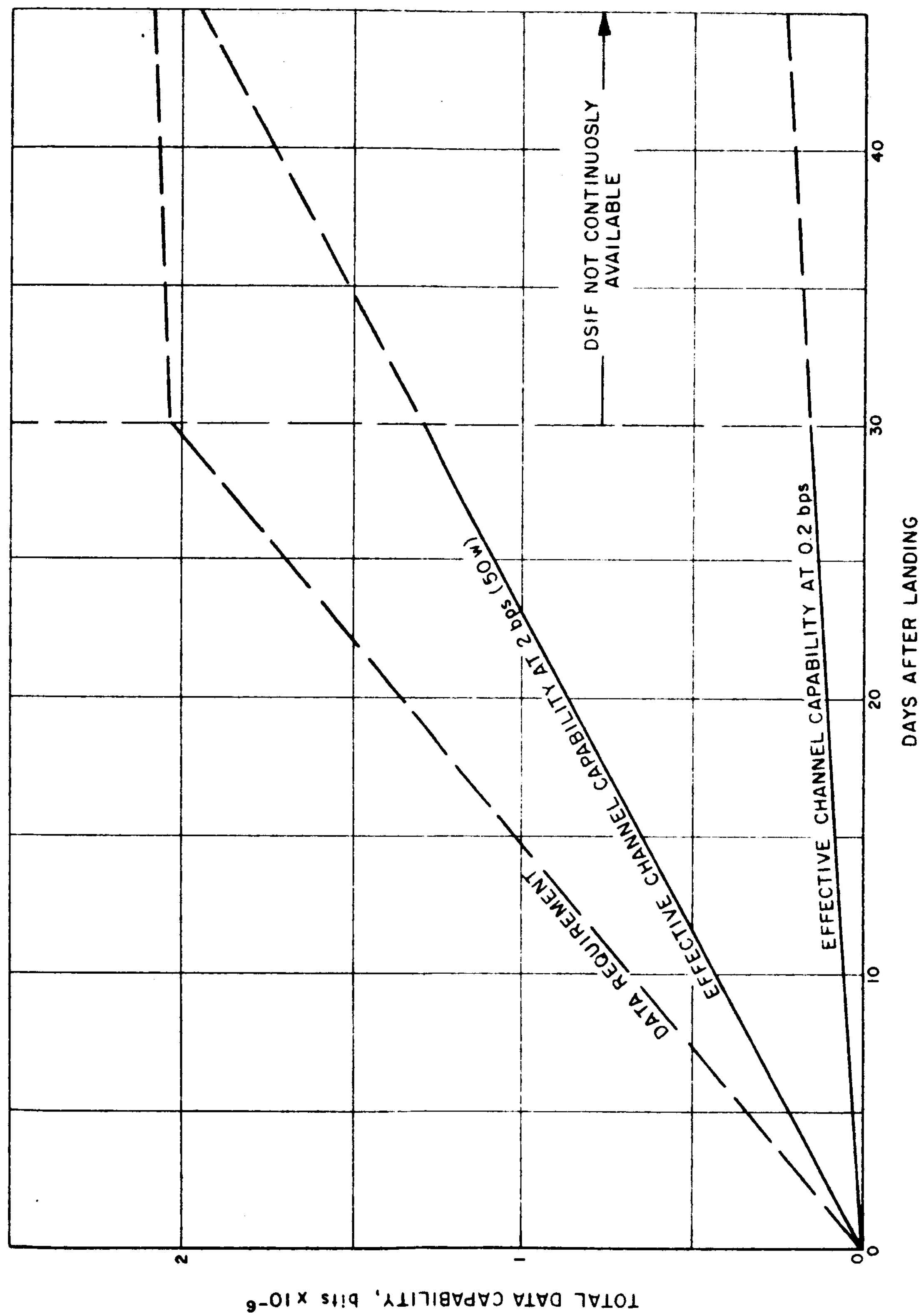

Figure 4-132. Channel Capability and Data Requirements for Capsule Direct Link

The mutual visibility between Earth and the landed capsule is interrupted daily. This situation is analogous to the landed capsule-orbiter mutual visibility problem for the relay link. However, the Earth-capsule availability profile is much less random and is primarily dependent upon the Mars rotation rate, the capsule antenna cone angle, and capsule position and orientation on the planet surface. As with the relay link, two options for communications control have been considered. They are shown in Section II (F) for determining capsule power requirements and are classified as Option Ia and Option Ib. The definitions of these two options are listed here for convenience:

Option Ia - The direct link communication system is operated continuously.

Option Ib - The direct link communication system is cycled and would operate at full power for 6 hr/day.

Under Option Ia, data should be transmitted redundantly in blocks. The time required to transmit one data block should be the time in one Martian day multiplied by the fraction of total time during which there would be mutual visibility. The block should be transmitted redundantly over one Martian day, after which time a new block of data should be loaded and transmitted redundantly. This will insure reception of a complete profile of information from the capsule regardless of phase differences in the start of mutual visibility and the starting of data blocks. Since the mutual visibility time is about 1/4 of the total time, blocks would be transmitted 4 times and updated after the fourth transmission. Block length would be 50 K bits for a 2 bps channel bit rate and 5 K bits for a 0.2 bps rate.

Under Option Ib, redundant transmission of data blocks is not necessary. However, provision must be made for correctly phasing communication system operation with the Earth-capsule availability profile. One possible system for achieving the correct phasing would be to activate the direct link communication system by a Sun sensor. If the capsule has landed in the area specified in Section II (C), then this system will provide a 6 hr mutual visibility period above the capsule antenna horizon, as specified in Section II (G).

Storage requirements for the direct link system will vary widely, depending upon the degree of independence built into the system. There must be at least 50 K and 5 K bit storages for the two rates considered. Entry and descent data storage will require 600 K bits regardless of the relay link storage requirements, since it was determined that the maximum storage for the relay link was dictated by the nonavailability of the orbiter, not the descent data requirements. Scientific instrument data storage could range from 1.3 by 10^6 bits (if data were taken at the instrument outputs), to 50 K and 5 K bits (if data were taken directly from relay link scientific storage). However, this latter system would require complicated program logic and addressing systems to withdraw this data from relay link storage, since it is a requirement that most scientific data on the direct link be sequential in time. Thus, the direct link system must be able to sequentially withdraw data at a low rate from storage media which have been filled at much higher rates. Such a system would be extremely complicated unless scientific data handling utilized random access storage media. A great deal of independence between the direct and relay link systems is sacrificed if the systems must share common storage media.

(5) Discussion of Trade-offs. Figure 4-132 shows that the direct link data requirements are not satisfied by a 50 w transmitter power level within 30 days after capsule landing. Trade-offs between data requirements and channel capability will be necessary to resolve this discrepancy. As was mentioned in discussion of the relay link, accelerometer data compression could decrease the total data requirements by about 120 K bits and, in addition, would decrease the required direct-link storage capacity. With a 120 K bit decrease in data requirements, the total direct link requirements could be satisfied within about 44 days after capsule landing, assuming that the DSIF were continuously available after the 30 day point. It is not known at this time what other specific trade-offs can be made in the proposed data requirements. One possibility is to reduce the requirement for entry and descent data on the direct link. Figure 4-132 shows that the direct link requirements have to be reduced by an additional 600 K bits to satisfy a 30 day lifetime. In the telecommunications area, the possibility of extending DSIF continuous single station coverage for 1 to 2 weeks beyond the 30 day point should be considered. It should be mentioned however, that even though DSIF coverage could not be extended and total data requirements subsequently reduced, significant worthwhile scientific value from the direct link would still be gained

A requirement for television pictures on the direct link would be a severe problem. High channel capacities would be required. Also, the sending of a continuous block of television data is in conflict with the requirement for the direct link not to be devoted to single sources for long periods of time. Long blocks of television data would also be in conflict with an assumed fixed format requirement for the direct link, since formats would have to be changed for efficiency of multiplexing. If picture blocks are broken up into smaller segments, such as lines, then some difficulty in reconstruction may be experienced. Television requirements for the direct link must be investigated in the future.

There is a problem with availability of the permanent descent data storage if the descent data are to be transmitted via the relay link to the orbiter on its first pass over the capsule. Weight and power considerations indicate that there only be one such permanent storage device; consequently, the relay and direct link may want to read from the storage at the same time. This problem is not too severe, however, compared to the difficulties of sampling scientific instruments from possibly two different clock sources or providing access to relay-link scientific storage as discussed previously.

c. Orbiter-Earth Link

(1) Performance Data. To estimate the bus/orbiter performance data requirements, a study was conducted utilizing Mariner B telemetry lists, EPD-139, Volume II, and engineering judgment on a likely configuration for a Voyager bus/orbiter. Measurements and sampling rates during the course of the mission were estimated in detail, and total measurements compared to a measurement-per-pound estimate. The results of this study are used during the remainder of this report. However, the results should not be considered as dictating firm system mechanizations. Assumptions for the orbiter configuration are as follows:

1) Primary Sun orientation reference for all flight modes.
2) Secondary star orientation reference.
3) Orbiter weight of approximately 2000 lb.
4) Two midcourse trajectory corrections.
5) One capsule separation maneuver (no motor firing on bus).
6) One retro-maneuver.
7) Power subsystem
   a) Fixed solar-panel power source.
8) Guidance and attitude control subsystem
   a) Gas attitude control system.
   b) Solar sail system (in cruise mode).
   c) Inertial guidance system (in orbital mode).
   d) PHP oriented to planet vertical.
   e) Guidance Sun sensor, planet tracker, and horizon scanner.
9) SOCC subsystem
   a) On-board special purpose computer.
10) Communications subsystem
   a) Articulated antenna oriented to Earth by program and simultaneous lobing.
   b) Planetary ranging on-board.
   c) Beacon transmitter for capsule.
   d) VHF capsule receiver.
11) Propulsion subsystem
   a) Single, constant-thrust, gimbaled nozzle engine with roll jet generator.
12) Data handling and scientific instrument subsystem
   a) No data compression.
   b) Orbiter time and identifier transmitted.
   c) Scientific instrument performance monitoring.
13) No additional equipment redundancy other than that explicitly stated in this volume.

Table 4-46 lists measurements by subsystem. Some of the factors affecting the number of measurements for a Voyager class mission are (1) greater subsystem complexity, (2) greater accuracy required on many measurements due to better resolution of sensors and finer degree of system control, and (3) requirement for better time resolution on certain measurements. Based on EPD-139, Volume II, page V-96, a value of 0.35 measurements/lb seems a realistic figure for prediction of the number of orbiter measurements. For a 2000 lb orbiter, the number of predicted measurements for comparison with Table 4-46 is 700.

Table 4-47 lists data-rate allocation as a function of the orbiter mission phases. Two methods considered for estimating required data rates are (1) fixed sampling rates for fast and slow data, and (2) variable sampling rates with changing flight mode.

(a) Estimates by Fixed Sampling Rates. Measurements are classified as requiring slow or fast sampling in each flight mode. The fixed sampling rate is determined by the fastest required measurements in the fastest mode. A fast sampling rate of 1 sample/sec should be satisfactory for most measurements. Those requiring greater time resolution may be redundantly sampled during a sample period. Special time-tagging or counting techniques may give sufficient information. On-board storage of data sampled at higher rates (e.g., 100 samples/sec) during midcourse or other critical periods may subsequently be interlaced with real-time transmission. Slow measurements can satisfactorily be sampled at rates 1/10 or less of the fast rate. The resulting communication-rate profile for fixed-rate sampling is actually a rather unrealistic worst case due to the fast sampling rate. In general, required data rates by this calculation will be about twice the variable sampling rate results calculated below. This method of estimation forms a worst case upper bound to the possible required rates and will be treated as such during the remainder of this report.

Table 4-46. Orbiter Performance Measurement Summary

Subsystem	Number of Measurements	Number of Bits for complete readout
Engineering Mechanics	58	406
Power	73	511
SOCC	41	364
Guidance	119	1092
Attitude Control	122	959
Propulsion	59	413
Communications	89	623
Command	22	154
Data Handling and Scientific Instruments	72	504
Total	655 Measurements	5026 bits

Table 4-47. Orbiter Performance Measurement Data Rate Allocation vs Mission Phases

Table 4-47. Orbiter Performance Measurement Data Rate Allocation vs Mission Phases

Variable Sampling Rate

Phases	Launch and Acquisition		Transit	Trajectory Corrections			Transit	Approach and Capsule Separation			Orbital Injection			Orbit
	I		II	III			II	IV			V			VI
	a	b	a	a	b	c	a	a	b	c	a	b	c	a
Engineering Mechanics							19.8	2.7	2.7	2.7	2.7	1.7	2.7	2.7
Power							24.5	6.9	6.9	8.0	6.9	6.9	8.8	8.8
SOCC							97.4	23.5	23.5	12.7	12.7	12.7	12.7	12.7
Guidance							130.8	117.0	98.0	132.8	132.8	65.7	173.7	172.4
Attitude Control			Estimated from IV, V, and VI				101.2	236.4	161.8	135.6	290.3	261.8	172.9	172.9
Propulsion							3.1	0.17	0.17	0.17	3.9	290.0	72.3	- -
Communications							99.5	105.9	174.3	175.2	175.2	64.9	178.0	175.2
Command							38.8	79.1	79.1	79.1	79.1	7.7	79.1	79.1
Data Handling and Scientific Instruments							26.7	64.2	64.2	64.2	64.2	64.2	51.1	64.2
TOTALS	432	573	547	602	569	501	547	636	611	601	768	776	752	688
Fixed Sampling Rate (Worst Case)														
TOTAL	432	1177	1125	1237	1051	1158	1125	1307	801	1517	1558	1434	1737	1552

(b) Estimates by Variable Sampling Rate. For convenience in computation, four sampling rates were chosen to cover the required range: 1 sample/sec, 1 sample/10 sec, 1 sample/100 sec, 1 sample/1000 sec. Each measurement was assigned a specified sampling rate for each flight mode according to estimated user requirements. Because of the greater range of sampling rates, this is a more realistic estimation of actual requirements than the fixed sampling rate case. Communication rates were calculated for the late phases of the orbiter mission and only estimated for the earlier phases, based on the same ratio of reduction from the fixed sampling rates. The results of this type of estimation will be considered as the orbiter data requirement for purposes of comparing system capabilities.

The mission phases assumed for the orbiter are as follows:

1) Phase I. Launch and Acquisition
   a) Launch and Injection. Nominal amount of engineering data for subsystem performance analysis.
   b) Acquisition and Extension. Transition from launch configuration to cruise configuration. Systems monitored as they go into operation, e.g., solar panels unfold, Sun acquired, etc.
2) Phase II. Transit
   Generally slowly varying information.
3) Phase III. Trajectory Correction Mode
   a) Premaneuver. Commands transmitted to the spacecraft, verified by retransmission to Earth. A complete status check of all subsystems.
   b) Initiation and Duration. High sampling rates desired. Lower priority data stored and transmitted later. Higher priority measurements transmitted in real time.
   c) Post-Maneuver Sequence. Transition to cruise. Reacquisition of attitude control references. Status check of all subsystems for comparison to premaneuver conditions.
4) Phase IV. Approach and Capsule Separation
   a) Preseparation. Increased transmission of guidance information; location of reference objects in spacecraft coordination system. On-board guidance computation. Otherwise similar to Phase IIIa.
   b) Separation Sequence. Similar to Phase IIIb but with no orbiter motor firing, proportionally decreasing the high rate sampling required.
   c) Post-Separation. Similar to Phase IIIc with continued guidance calculations. Capsule relay link activated.
5) Phase V. Orbital Injection Mode
   a) Preretro. Establishment of horizon scanner, planet vertical, and planet range. High guidance and attitude control sample rate. Phase IIIa comments also apply.

b) Retro-Initiation. Similar to Phase IIIb. Long duration motor firing increases data storage required.

c) Post retro. Transition to orbital mode. Propulsion system jettisoned. Fewer guidance calculations. Inertial guidance system activated. Capsule relay link active.

6) Phase VI. Orbital

Similar to cruise, but additional sensing in relation to planet for guidance, attitude control, and power sequencing during Sun occultation. Capsule communication link active.

(2) Experimental Data. For purposes of this report, experimental data requirements can be divided into 4 phases: the interplanetary or transit phase, approach (between capsule separation and bus retro), and the planetary (or orbital) phases before and after Encounter +30 days. Table 4-48 below lists experimental data requirements by 3 appropriate groupings for these phases.

(3) Comparison of Total Data Requirements and Channel Capability. Comparisons here are devoted to the orbital phase as a function of days after encounter. Performance data requirements during this phase are about 700 bits/sec and experimental data requirements are at their maximum value. If data requirements can be satisfied during this phase, they will be satisfied for the preceding phases of the mission. Actually, performance data requirements will reach their maximum during retro maneuver, but the total data rate requirement will decrease because high data-rate experimental measurements cannot be accomplished during maneuver. Table 4-49 shows the total average requirements imposed on the orbiter channel, including capsule data plus sync. The total data shown is cumulative data transmitted after bus retro. The effective channel capability for the first 30 days after bus retro exceeds the total data requirements by about 3,000 bps. Methods of utilizing the

Table 4-48. Orbiter Experimental Data Requirements

Data type	Transit Phase	Approach Phase	Orbital Phase	
			Before E - 30 days	After E + 30 days
Particles, Fields, and Micrometeoroid	125	125	125	125
Surface structure (non-vidicon) and Atmospheric	0	75	75	75
Surface structure (vidicon) and Atmospheric	0	2200	2200	20
Totals	125	2400	2400	220

excess capability are discussed on the preceding page. The channel capability after 30 days is decreased by the effect of noncontinuous DSIF availability.

(4) Sequence of Operation, Mode and Storage Requirements. The sequence of operation for the bus/orbiter data handling system will be from just prior to capsule separation to the end of mission.

(a) Phase IVa - Preseparation. Provision should be made to telemeter capsule status checks interleaved with preseparation bus data.

(b) Phase IVb - Separation. The bus should revert to real time transmission with concurrent storage of data. Capsule data received during this phase must also be stored for later read-out to Earth. The capsule data storage required during this phase is small compared to later requirements.

Table 4-49. Orbiter Total Data Requirements and Channel Capability After Bus Retro

Data Type	Encounter + 30 Days		Encounter + 150 Days	
	Average Data Rate (Bps)	Total Data (Bits)	Average Data Rate (Bps)	Total Data (Bits)
Performance	688	1.79×10^9	688	8.95×10^9
Experiments	2,400	6.24×10^9	220	8.53×10^9
Sync (15%)	464	1.21×10^9	134	2.61×10^9
Capsule 50^w at maximum availability	79	0.2×10^9	79	0.96×10^9
Total Requirements	3,631 bps	9.44×10^9 bits	1121 bps	2.3×10^{10} bits
Channel capability for 79 DBM Spacecraft power-gain product at worst case distance during mutual visibility	6,600 bps		2900 bps	
Worst case distance	235×10^6 KM		355×10^6 KM	
DSIF availability	Continuous		1/3 of total time	
Effective channel capability	6600 bps		970 bps	

(c) Phase IVc - Post Separation. Upon restoration of cruise orientation, the bus should read out stored bus and capsule data. After this period, the real time capsule relay link can be activated for purposes of aligning ground receiver parameters and receiving redundant read-out of capsule storage and real time capsule system checks. During the time that the bus data modulator is slaved to the capsule clock, the orbiter data handling can remain on its own clock in the sampling mode desired. Capsule data storage on the bus can be filled in parallel for possible read-out at a later time. After a short period of time, the orbiter will place an inhibit on the ability of the capsule clock to control the modulator and devote the channel to approach planetary science. There will be from 28 to 58 hr until capsule entry for this mode. Approach planetary science is considered a requirement at present because of the uncertainty in bus operation after retro maneuver. The bus channel should remove capsule relay inhibit about 2 hr prior to capsule entry and, because of uncertainty in capsule operation, should be devoted exclusively to capsule data relay until the bus receiver falls out of lock. Capsule data-storage should be filled in parallel during this time for possible later retransmission. Approach planetary science and performance data-handling should continue sampling and storing for later transmission. Orbiter performance storage requirements during this phase would be about 8 by 10^6 bits if total storage of performance data were assumed. Assuming the 50 w capsule transmitter power and an available time of 4 hr (corresponding to a Δ T of 2 hr plus 2 hr prior to entry), capsule data-storage requirements are about 3.5 by 10^6 bits. With the same time and a 25 w capsule power level, capsule data storage is about 1.5 by 10^6 bits. Time to read-out the maximum storage at a 3,600 bit bps bus data-rate is only about 18 min. The capsule data should not be destructively transmitted from bus storage until some time after retro maneuver, to allow for orbiter failure to gain an orbit but successful reacquisition of Earth.

(d) Phase Va - Pre-retro Mode. Relay of capsule data to Earth should be inhibited at this time and the channel should be devoted primarily to data performance concerned with preparation for the maneuver. There is still a bandwidth margin for sending of other information. Two possibilities exist for usage of this bandwidth:

1) Readout of capsule data storage.
2) Readout of stored approach planetary science measurements taken during capsule relay activation.

(e) Phase Vb and c - Retro Initiation and Post Retro. Bus data should be stored and transmitted in real time. Assuming a 776 bps input rate and 15 min of motor burn, storage requirements for performance data are 650 K bits. After Earth is reacquired, storage would be read out.

(f) Phase VI - First 30 Days of Orbiter Operation. The orbiter should activate the beacon transmitter (under Option II) and also revert to transmission of performance and planetary experimental data. When the VHF receiver comes into lock, capsule data-relay to Earth is activated with parallel storage of capsule data. Parallel storage can be read-out, if desired, at a later time. This constitutes a back up for DSIF failures, Earth occultation, or other high priority orbiter information requiring the channel. Maximum storage requirements for capsule data on the orbiter at the 50 w capsule transmitter level are 1.25 by 10^7 bits, for a 5 hr maximum period of mutual visibility (Table 4-43) and a 690 bps capsule data-rate. At the 25 w capsule transmitter level, the storage is 6 by 10^6 bits.

[illegible] requirements are substantially greater than the requirements for the capsule entry phase. During capsule data-relay, the orbiter data handling system may store all or any part of the performance data. The performance storage will be about the same as capsule storage required, or 1.3 by 10^7 bits. Experimenter data storage required is about 6 by 10^8 bits.

(g) Phase VI - Orbiter operation after first 30 days. The DSIF availability significantly affects the data handling storage requirements in this phase. Performance data should be stored in 8 hr blocks for redundant transmission, and there must be provision for 3 of these blocks. Assuming the 688 bps performance [illegible] requirement, a performance storage of about 6 by 10^7 bits is needed. This requirement is excessive. The capsule data relay link may still be activated [illegible] capsule data must positively be stored to prevent its loss. The capsule data storage must be sized for the maximum availability which could occur between capsule and orbiter during a 24 hr period, or about 6 hr. This increases capsule data storage to 1.5 by 10^7 bits for the 50 w capsule transmitter level and to 7 by 10^6 for the 25 w level. Experimental storage will be much less than that required in the first 30 day period.

(5) Discussion of Trade-Offs. A 77 dbm orbiter power-gain product provides an effective channel capability only 151 bps lower than the data requirements at 150 days after Encounter. This is not a serious problem, however, since it is possible that performance data requirements could be reduced during the latter part of the mission. However, if Earth occultation occurs, or performance measurement requirements increase toward the worst case estimated in Table 4-47, then the channel may become limited in capability during the latter part of the mission. The 79 dbm power-gain product easily satisfies the estimated data requirements at the Encounter +30 day point. There are a number of ways to utilize this channel capability margin, which is of the order of 5,000 bps. First, [illegible] storage capacities could be [illegible] because of the high channel-capability to transmit data in real time. Second, scientific mission value could be increased by transmitting more planetary scientific information during the first 30 days after Encounter. Third, the channel bit-rate could be decreased for the purpose of minimizing problem areas in the communication system. At present, a major [illegible] area in this system is antenna lobing. However, any decrease in the channel capability to match data requirements at Encounter +30 days requires reduction either of the data requirements at the Encounter +150 day point or of the orbiter mission lifetime.

It should be mentioned that the channel capability and requirements were only considered at two times after retro-maneuver. This discussion then, while only considering two bit-rates, does not preclude the possibility of stepping down through a greater number of bit-rates during the orbital period. For example, if experimental data were required at a rate of 1000 bps after 30 days, then the total data requirements could still be supported by the channel for an additional 10 days, even without full DSIF coverage.

If it is desired to preserve all performance data, as assumed in the preceding discussion on storage capacities, then orbiter performance data storage requirements become excessive in view of the very probable requirement that performance storage must be an integral part of the data handling system, working sometimes in an interleaved fashion with real-time transmission. A feasible solution to this problem is to reduce the performance data requirements during periods of channel interruption, consequently reducing the storage. However, if the channel is interrupted and the data handling system has no knowledge of it, then data could be lost. If the channel is known to be interrupted, then performance storage data could be filled with vital data to be read-out at a later time. At present it should be a requirement that the performance data storage be limited to the region of 600 K to 800 K bits to allow for more reliable types of memories and to minimize system integration problems.

Capsule data storage problems are as critical as performance data storage, but here the primary problem is reliability. Although the capsule data storage is used as a back-up during the first 30 days, it becomes a series element after this period. There are a number of solutions to this problem; first, in the event of a failure of the storage media after the first 30 days, DSIF could be placed on continuous coverage; however, maximum time of coverage could be excessive and Earth occultation problems would still remain. A second solution would be to utilize existing scientific storage media, which may be partially available after the first 30 days. A third solution would be outright redundancy. Integration problems for capsule storage are not too severe and, since the maximum storage requirement is about 1.5 by 10^7 bits, tape recording media are feasible for use in redundant configurations. For nonredundant usage their reliability should be questioned for this high priority application. Thermoplastic recording techniques may prove to be a solution to this problem in the future but, for the present, their feasibility is still in question.

The worst-case performance data requirements shown at the bottom of Table 4-47 are in some cases twice the requirements used for this discussion. The primary reason for this difference is that, in estimating the worst-case requirements, low system flexibility was assumed. More explicitly, only a limited ability of the system to tailor a wide range of sampling rates to the individual measurements was assumed. A substantial amount of flexibility over the mission phases is needed to achieve the reduction in data requirements from the worst case. However, 655 measurements predicted for the orbiter is about twice the number incorporated into a Mariner series spacecraft. Building flexibility into such a large data handling system will pose severe design and checkout problems in the future. Data compression techniques for bandwidth reduction are within the realm of consideration for application within the orbiter data handling system, but these techniques should not be used at the expense of sound system organization and good reliability practices.

The most promising area for data compression within the orbiter data handling system [illegible] system, which produces a substantial amount of the total orbiter data requirements, but which can operate somewhat independently from the data handling system. In addition to achieving a sizeable reduction [illegible] required data rate, data compression in the vidicon system would cause a reliability gain from reduction in the required [illegible] storage capacity. Another benefit may be realized with vidicon data compression, coupled with reduction of performance [illegible] requirements at the Encounter [illegible] day point. For example, if performance data requirements at the [illegible] days point are reduced to [illegible] bps, total data requirements become 439 bps. [illegible] rate of [illegible] DSIF duty cycle, or 1[illegible]17 bps, which corresponds to a power-gain [illegible] db. At the distance corresponding to [illegible] days after Encounter, 75.[illegible] dbm will provide a rate of [illegible]00 bps. Vidicon data uses about 2000 bps of the total [illegible] bps data requirement at Encounter + 30 days. Therefore, with a data compression ratio of 2000/(2000-[illegible]), or 1.5 to 1, the data requirements at Encounter + 30 days can also be satisfied. The reduction in power gain product is 2.4 db. Further investigation into this area is warranted for possible minimization of problems in the communications system.

4. Data Handling System Design Philosophy

a. General Requirements

A primary requirement for deep space missions in the future will be that of reliability. The requirement can be divided into two parts. First, there is the reliability of the elements comprising the data handling system, which includes such considerations as the various levels of redundancy, component reliability, detailed design and fabrication of the system; the second is a requirement which dictates that system design philosophy should allow for failures regardless of their probabilities. Design philosophy for missions of the durations considered for a Voyager class spacecraft must recognize the possibility of failures; system design should localize failure effects such that some mission value is realized when failures occur. Probabilities of failures which could cause mission failure must be minimized in susceptible areas, first by minimizing the number of such areas, and then by reducing the resulting failure probabilities [illegible] those areas. The same philosophy should prevail in those areas where failure [illegible] the higher priority portions of the total mission value. The application of such a philosophy necessitates assignment of multiple risk levels to the various functions within the data handling system and subsequent synthesis to minimize overall risk. System design concepts must also tend to elements which are inherently more reliable or can be made more reliable by application of redundancy techniques.

Flexibility is the second most important requirement and is basically of two types. Inter-mission flexibility is the ability of the system to accommodate measurement changes without affecting the entire system organization. This results in greater case of design. Inherent in this type of flexibility also is the ease with which varied measurement types are incorporated, a desirable characteristic when data compression and varied performance and scientific measurements are considered. Scientific payloads, especially, will probably vary considerably between missions of the same class spacecraft. Intra-mission flexibility is the system's ability to provide a number of data handling modes for an ensemble of measurements with some efficiency of data transfer to the user. For a Voyager class mission, this flexibility will be mandatory. Efficient utilization of bandwidth requires a close fit of the sampling plan to the expected information of the data sources, either by a fixed program in each mode or by data compression. There are many additional requirements such as interleaving stored data with real-time data, destructive and nondestructive storage read-out, and conditional data modes based upon probabilistic spacecraft events.

Another important requirement is the reduction of the uncertainty in making a measurement which is introduced by the measuring device. This implies independence of the measuring device from the measured quantity and the influences which act upon it. The influences may be environmental or those induced by abnormalities in the measured system. If independence cannot be achieved, then the influences should be at least as separable as possible.

The last major requirement is efficiency with respect to weight and power invested and with respect to utilization of available bandwidth from a given information transfer.

b. System Design Philosophy

Reliability requirements for the design of Voyager class data handling systems can be partially satisfied by decentralization. This includes the following concepts:

1) Make the data paths from the sources to the final output as independent as possible with respect to functional elements required. Thus, elements which are time-shared in more centralized systems will be replicated to provide this independence. The major disadvantage of this concept is that weight and power requirements increase; however, in view of greater weight and power allowances for a Voyager-class spacecraft, this may be a reasonable price to pay for increased reliability. More independence tends to insure that the functional elements required to be in series with any given measurement will be a minimum, and hence the success probability for the given measurement will be a maximum. However, with more independence, the probability of success for the set of all measurements becomes less than for the centralized system. A

decentralized system more readily satisfies intermission flexibility requirements in that measurement changes during the design phase do not influence the entire system just as measurement failures during the mission would have less influence on the system. This type of system is more complex, but the complexity can be made separable, resulting in relatively easy design and check-out.

2) Any central control elements such as master programmers or multiplexing elements should be as simple as possible, to provide for inherent high reliability and to render the addition of redundancy an easier task. Conceivably, these central control elements could be mechanized for specific control of every data source and, consequently, would have to remember all possible formats, speeds, and modes for the data sources they must control. When commands or other stimuli are received by the control elements, they must in turn transform these stimuli into detailed instructions for the data sources. To simplify the central control elements, as many of these detailed control functions as possible should be delegated to lower control levels, with the central elements retaining only the essential parts of the control functions. The design problem is one of how to delegate these control functions to lower levels subject to a set of constraints retained in the central elements for proper system coordination, while at the same time retaining the intramission flexibility to satisfy the mission requirements.

The decentralizing process for more independence can quite rapidly approach very generous limits of weight and power, therefore, the design process must be a discriminating one. Obviously, in any endeavor with the potential for scientific gains typified by a Voyager-class mission, it is the scientific payload and data handling requirements which must receive the first priority for survival. Fortunately, mutual independence among the scientific instruments and attendant data handling functions does not result in as rapid an increase in weight and power as for the performance measurements, because of the hybrid nature of the individual data handling requirements for the scientific instruments.

Because of the weight, power, and survival constraints above, performance measurements must be limited to groupings of measurements. The major classification of the groupings should satisfy both intramission flexibility requirements and some priority of survival. These two goals are somewhat compatible in that the data-rate allocation profile generally reflects a measure of survival priority as it changes during the mission. One difficulty is that survival priority may vary with time and thus the possibilities for fixed independence by this type of grouping are somewhat limited. There are other possible major classifications for performance measurement groups. These classifications are presented in summary form below:

a) Sampling Rate. This classification is used in present multiplexing systems when measurements are grouped into sampling rate categories for the purpose of matching the sampling process to the expected information of the measurements. This type of grouping is efficient from a bandwidth utilization viewpoint. This efficiency increases as the groups become smaller. If the groups are mechanized independently, system weight, power, and complexity may increase rapidly. The many flight modes of a Voyager class spacecraft tend to complicate the problem of matching the sampling process to the groups of measurements.
b) Mission Priority. Two groupings in this category include the operational and evaluation measurements. Operational grouping may be inefficient in bandwidth utilization, since measurements may have low frequency of usage. Generally, this type of organization becomes more compatible with sampling-rate groupings as flexibility requirements decrease.
c) Subsystem. The outstanding advantage of this primary group is the reduction in interface complexity. Measurements may be conditioned remotely from central control elements. Ease of checkout and design are also advantages. The primary disadvantages are the weight and power increases, along with the fact that if a subsystem data-handling mechanization fails catastrophically, all measurements from one subsystem would be lost. However, organization within a subsystem could be on the basis of multiple risk levels to alleviate this disadvantage.
d) Common Data Handling Functions Required. This type of grouping results in low weight and power but does not satisfy priority of survival requirements. Flexibility is also hard to introduce.

The system functional block diagrams which follow in the next subsection reflect the philosophies discussed above but have been adapted to the particular application.

5. Data Handling System Functional Block Diagrams

a. Introduction

The functional block diagrams in this subsection are based upon a representative group of scientific experiments and the performance data requirements presented in paragraph 3. It is difficult at the present time to postulate specific experiments, and no further progress in firming up the list of experiments will be made until the preliminary design has started. The primary purpose of these diagrams is to arrive at realistic weight and power estimates and, secondarily, to postulate data handling mechanizations as an initial solution to data handling requirements. It must be stressed that the solutions are initial and, in many cases, very general. Future iterations and quantitative measures for comparison will be necessary to determine the mechanization which is best qualified to satisfy the final mission requirements.

b. Capsule Relay Link Data Handling System (Figure 4-133)

The system shown reflects some of the trade-offs discussed in paragraph 3. No vidicon data compression is assumed, and a more pessimistic weight and power estimate will result from this assumption. Accelerometer data compression is assumed, which will result in a higher estimate for the relay-link system, but a lower estimate for the direct-link system. Descent and entry data storage is shown on the direct-link diagram with weight and power charged to that system. The performance data storage is sized for 700 K bits under Option II; under Option I this storage would decrease to the range of 300 K to 400 K bits capacity. The performance data storage is used for temporary storage of selected entry and descent data. To simplify the diagrams, lines of control, quantitative updating of programs, telemetry via the bus umbilical, and some outputs to the direct-link system are not shown. Telemetering through the orbiter umbilical could be accomplished in parallel with the outputs leading to the master multiplexer and by injecting the bus clock into the system for phase slaving. A description of the various functional blocks in the system follows:

1) Control Elements. There are 3 main control elements. The master controller primarily influences multiplexing from the storage media of the system and from the final output of the performance data commutator. Periodically, the master controller interrupts clocking to the system and injects a master sync sequence into the data stream. This sequence is required for initial acquisition of capsule data. Since the capsule relay link is a synchronous bit system with possibility for true or complemented data upon receipt, this master sync sequence will be long. The minimum duration of time which the capsule relay link is predicted to be in lock (with positive data transfer required) will dictate the approximate frequency of the master sync sequence. All other sync for stored performance data, scientific data, and commutator groups is generated within the area of interest when the data is multiplexed on the channel. The remaining control functions are vested in the experimental and performance data programmers, which contain nonvolatile storage and program logic to carry out detailed instructions. Hopefully, these functions can be more decentralized in future studies. Another control element is required under Option I and may be used as a back-up under Option II. It is shown in dotted lines in the diagram. This element simulates the orbiter availability profile

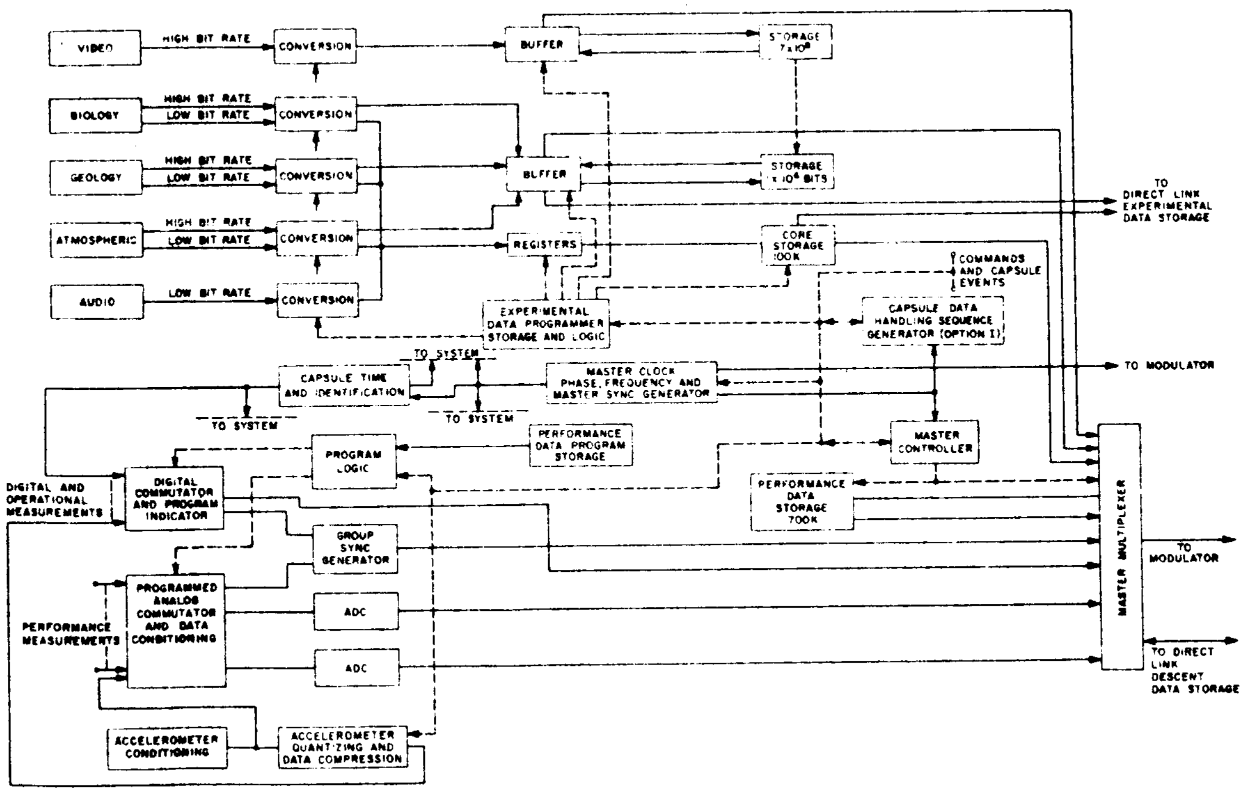

Figure 4-133. Capsule Relay Link Data Handling System Functional Block Diagram

in the absence of external command control. It can utilize its own independent clock since synchronous mode-changing is not required.

2) Master Clocking Elements. The master sync sequence and all the phases and frequencies required by the system are generated within this element. Capsule time and identification is also generated in a separate element for use of any group requiring it. Capsule time will be required mainly for the accelerometer experiment and for stored data which must be correlated with previously transmitted real-time data.

3) Scientific Instrument Data-Handling Elements. A great deal of independence of conversion equipment and storage media is used for the scientific experiments. Most of the experimental data is multiplexed directly from storage according to a sampling plan stored in the master controller. Some portion of this sampling plan may be carried out by the experimental data programmer.

4) Performance Data Handling Elements. Primary grouping of performance measurements is accomplished by priority of survival. The digital commutator generally processes the high priority or operational measurements. These measurements are: events issued and received along with times of occurrence, redundancy states, capsule time and identification, command verify, data program in effect, accelerometer peak readings and times, and other digital measurements such as threshold failure indicators, motor burn durations, and velocity increments. The digital commutator is internally programmed for more efficient bandwidth utilization and generates its own sync indication when addressed. Conceivably, readout from this commutator should immediately follow the master sync sequence.

The analog commutator processes the measurements which will generally fall into the evaluation category. This commutator is programmable for more efficient bandwidth utilization during the many capsule modes. The basic programmed unit is a deck of measurements called a group. There are 3 or more relative rates to which any number of groups may be assigned within a program. Once the particular assignment is made, the order of sampling of the groups is also specified. The program has capability of randomly specifying groups, the groups and the measurements on the decks are then sequentially sampled. All decks except the deck being sampled are turned-off (or isolated from the measurement bus). The deck to be sampled is turned-on and then sequentially sampled. Each deck generates its own group sync. When the master sync is being sent, every deck in the commutator is turned-off. Each group has an equal number of measurements; consequently, the period of the master sync sequence can be made to be a multiple of the group length and, when this is done, the reset to the commutator will have no effect unless the commutator is out of synchronism with the master sync period. An approximate allocation of measurements to the groups under Option II has been made to arrive at some measure of the group length. Too large a group results in less efficient bandwidth utilization; too small a group results in undue complexity. Table 4-50 summarizes this allocation. The group size chosen was 10 measurements. The maximum differences in the chosen group size and the number of measurement allocated to the group, was +4 and -3. Groups of 5 were thought to result in too complex a programming system. The number of groups shown in the listing is 12 and the group length in bits would be 77 bits, which includes group sync and 7 bits per measurement. The remainder of the measurements are in the digital commutator, which will comprise about 2 groups. Performance program storage is about 500 bits maximum over all possible modes. It should be noted that the group size places a constraint on

Table 4-50. Capsule Analog Commutator Group Allocation

Group Descriptions	Nearest Number (Divisible by 10) to the Number of Measurements in the Group	Groups
Scientific instrument evaluation: temperatures and temperature control	20	2
Engineering Mechanics: Structural and heatshield temperatures Temperature and control	10	1
Other systems: Temperatures	10	1
Communications and Data Handling: various Planet orientation	30	3
Power subsystem: voltages and currents	10	1
Propulsion and in-flight orientation	10	1
Scientific instruments: various (less calibration), Accelerometer performance	30	3
Totals	120 Measurements	12 groups

the freedom of choice in telemetry channel assignments, but the group size is small enough so that this constraint is minor when compared to the constraints placed upon past systems.

5) Accelerometer. This measurement is fed to the analog commutator for relatively slow sampling. Most sampling, conditioning, quantizing and peak detection is conducted by an independent unit and the results fed to the digital commutator. To conserve weight and power, one performance-measurement analog-to-digital converter could be diverted to the accelerometer to accomplish quantizing during capsule entry and descent. However, this would leave just one converter in series with performance measurements at a critical time. These performance converters should provide redundancy at all times. Therefore, the accelerometer experiment contains its own analog-to-digital converter.

c. Capsule Direct-Link Data-Handling System (Figure 4-134)

The direct-link data handling system operates independently from the relay-link system except in the four following major areas. First, the direct-link clock would normally be phase-slaved to the master clock in the relay-link system. If the master clock should fail, the direct-link clock would continue to time the direct-link system, to withdraw experimental data from relay-link storage media, and to takeover capsule time indicator clocking. Operation of the direct-link commutator would not be impared if the master clock failed. Second, some control information from the relay system must be utilized in the direct-link system. Third, the descent data storage may have to be time-shared between the two links. The relay performance storage would be filled, in parallel, with the direct-link storage during entry and descent of the capsule. If there were requirements for continual read-out of descent data on the relay-link after the first orbiter pass over the capsule, the direct-link storage would have to be read-out on the relay link, because the relay-link storage would have to be used for storing of post-landing performance data. Fourth, it is assumed that the direct link would withdraw experimental data from the relay-link storage media whenever this storage was not being read-out on the relay link. An alternative to this system would be to store all experimental data in direct-link storage as it is generated by the instruments. This alternative would bypass the problem of time-sharing relay-link storage between the two systems, but direct-link data storage would become excessively large. A memory with good access properties and capability for close system integration would be required for this application. With the system assumed here there would still have to be some nominal amount of experimental data storage for the direct-link. In addition, a programmer would have to be provided for gaining access to relay-link storage as it is available for read-out to the direct link. The frequency of access would be inversely proportional to the direct-link data storage capacity. In the diagram, direct-link data storage is sized for 100 K bit capacity.

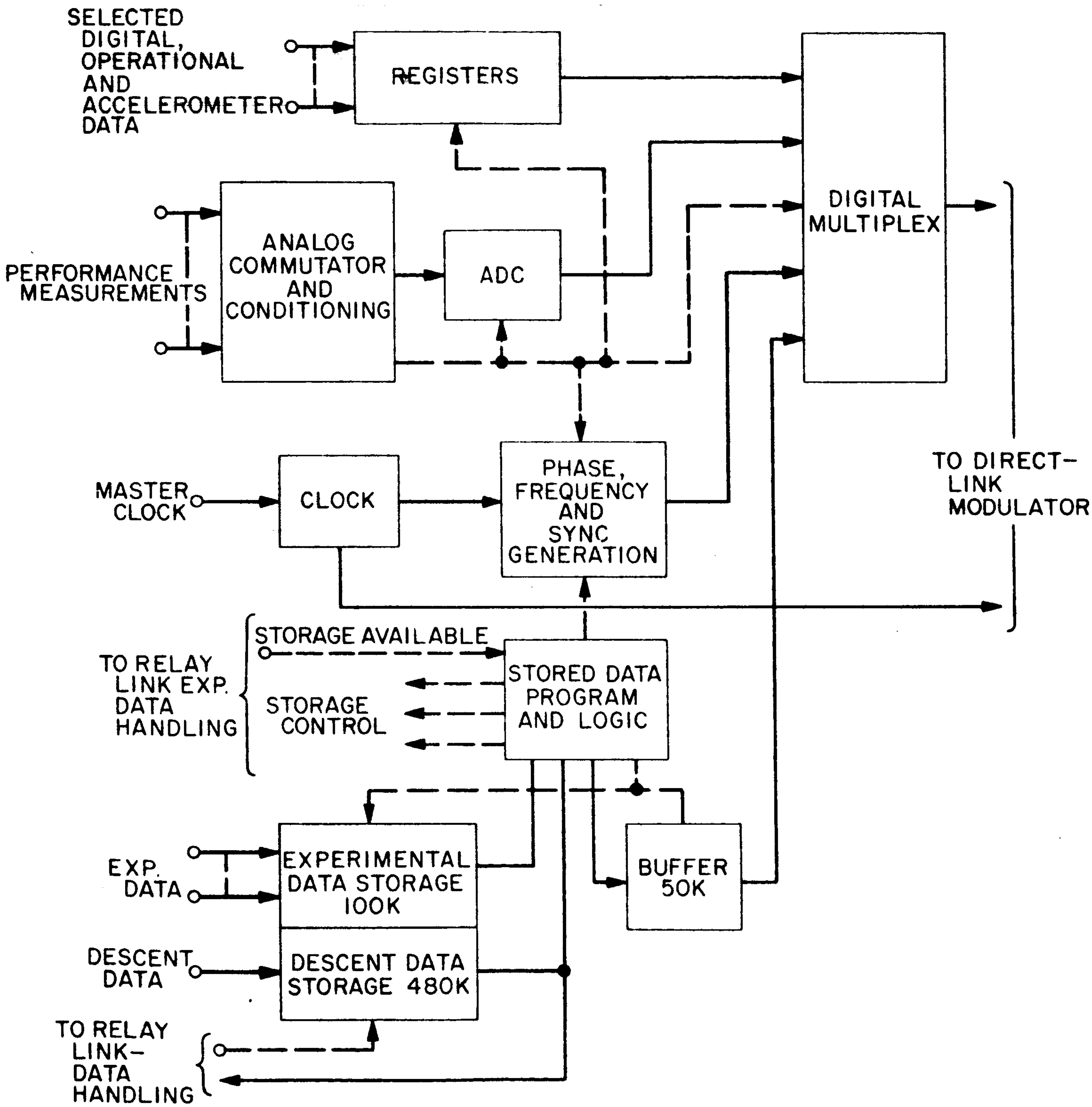

Figure 4-134. Capsule Direct Link Data Handling System Functional Block Diagram

The storage capacity for the entry and descent data storage is sized for 480 K bits. This is the total descent and entry data requirements less the data saved by assuming accelerometer data compression.

The buffer storage shown in Figure 4-134 reflects the assumption of Option Ia, where the communication system for the direct link operates continuously. The buffer is updated with a new block of information every fourth readout. Redundant read-out is needed to ensure reception of a complete data profile with a 1/4 duty cycle of mutual visibility between the capsule and Earth. Buffer storage is sized for the 50 w transmitter level corresponding to a 2 bps channel bit rate.

A direct-link time indication is needed on the channel to provide correlation between the performance engineering measurements and the stored experimental data. Capsule time is utilized for this purpose. Capsule time is also needed for the scientific measurements when they are taken and stored.

d. Orbiter Data Handling System (Figure 4-135)

The orbiter data-handling system diagram reflects some considerations discussed in subsection 3. Potential vidicon data compression is indicated, but storage sizes shown are sufficient for an approximate 2 day accumulation of uncompressed data, resulting in a slightly greater weight and power estimate than could be expected with compression. Data compression, or at least adaptivity to system activity, is provided for in the performance data-handling area. Performance engineering storage is sized for 700 K bits. Capsule data storage is sized for the maximum estimated in subsection 3. In weight and power estimates, provision is made for two redundant capsule data-storage units of 1.5 by 10^7 bit capacity. More specific comments on the various elements of the system are listed below:

1) Control Elements. There are three major control elements in the data handling system. The master controller transforms commands and other spacecraft events into usable form for the system, controls multiplexing of the storage outputs with real-time data from the performance area, specifies what data is to be stored in the performance storage, and controls generation of master sync. In many cases commands and other spacecraft events will be directly routed to the control elements on a lower level than the master controller, depending upon the decentralization. Only a few of these lines of control (dotted) are included on the diagram for simplicity reasons.

   The other two main control elements are the experimental and performance program storage and logic elements. The experimental programmer is shown as a centralized unit, but it is likely that its functions will be more dispersed

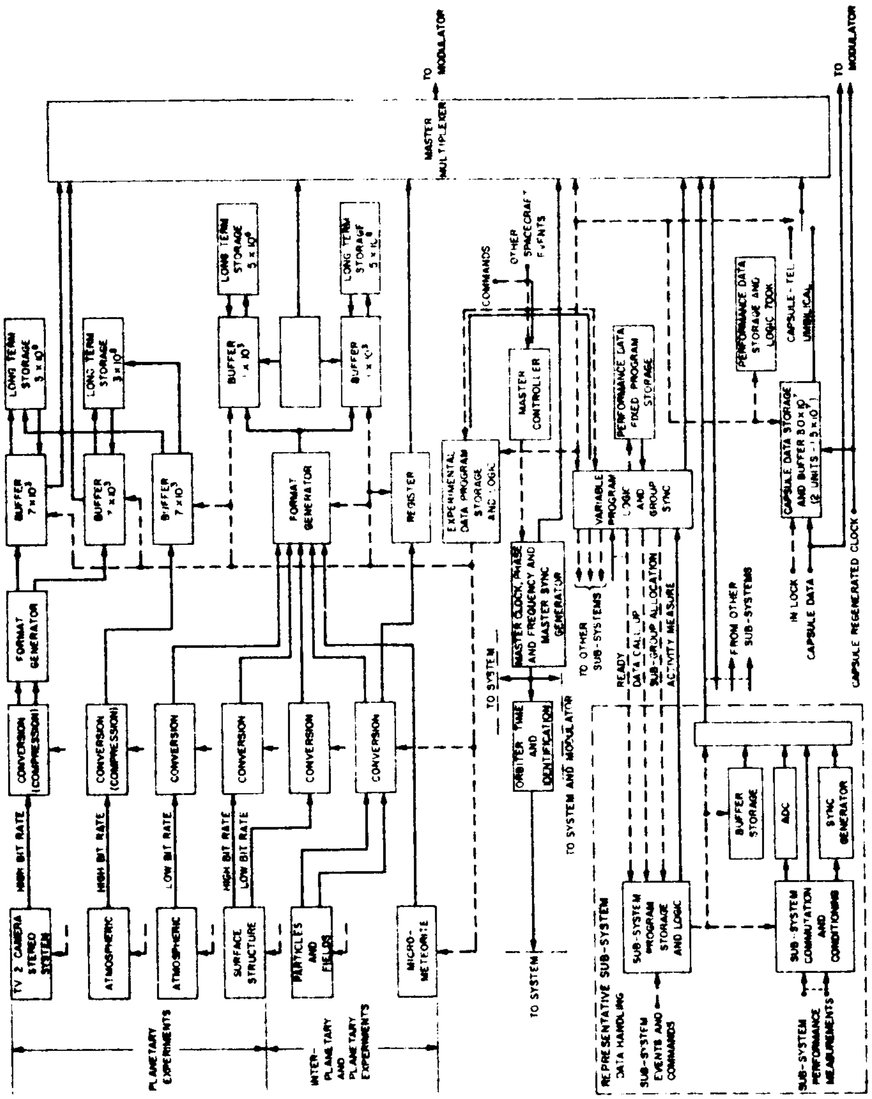

Figure 4-135. Orbiter Data Handling System Functional Block Diagram

after greater detailing of scientific instruments. It is possible to predict several of the major types of control that will be characteristic of the types of experiments being considered for this mission. There are four major types of control for the orbiter experiments, which follow:

a) Data Value Control. The particle and fields experiments are characteristic of this type of control, where the value of the data being obtained by the instruments controls its area of the data system, causing data rates and instrument sequencing to be modified. Commands will also be necessary for operation of this type of experiment and only minor operational changes are anticipated from the cruise to the orbiter phase of the mission.

b) Constant Sampling. Several of the planetary experiments will run at relatively low data-rates for the duration of the orbital phase. Data rates from 8 to 30 bps are currently being considered for these experiments and very little data system or external control is anticipated.

c) Command Control. Certain types of planetary instruments will be operated almost entirely by command. The sequence of operation for this type of experiment will provide for the taking of a complete spectrographic map of a planet segment during one orbit and then, by command, changing the spectral coverage of the experiment so that only specific lines or portions of the total spectra are covered on later orbits. These decisions will be based on the analysis of data received from the experiment.

d) Area Coverage. This type of experiment will be selected to provide coverage of a specified segment of the planet surface. The control of this experiment will be based on lighting constraints, orbit parameters, area coverage, etc.

The performance data-handling programmer imposes a fixed set of programs on the performance measurements. The programs are very general in that they only specify what number of subgroups the groups are allocated in a given period of time, e.g., between master sync sequences. Once this allocation is made, a sampling plan is carried out. When a given group is being addressed, a sync sequence is generated by the performance data-programmer. The group addressed is given a "ready" signal at this time, and then the data "call up" from the group is made by the programmer. It is then the responsibility of the group to specify what subgroups it is transferring-out by means of its own sync. The group may transfer-out any data in real time or from storage, depending upon its own internal program. A group is, of course, limited to the subgroup allocation it has been given in the time the channel is made available to it. When the group is not being addressed, it is free to carry out its own internal sampling or to search for measurement activity.

Superimposed upon the fixed set of programs in the performance programmer is a provision for adaptation of a set which is based upon measures of activity generated within each group, subject to the constraints of a fixed set of priority rules which can be included in the main program. The activity measures could be generated in the form of requests for a greater subgroup allocation, whereupon the performance programmer would take action to grant these requests. If a group cannot be given the channel immediately, then the data may be buffered in group storage or diverted to performance storage depending upon the mode of operation specified by the master controller. There are many variations allowable with this basic system. Provisions need not be made for adaptation in every group; in such cases the fixed set of programs and decision rules would be simpler. This last statement infers that the fixed set of programs should attempt to match the expected information from the various groups. Measures of activity can also be sent to the master controller, e.g., for extreme measures of activity where performance measures should take precedence over experimental or capsule data. Finally, the basic internal organization of the groups can remain flexible and amenable to different data-handling or compression systems.

2) Master Clocking Elements. All phases, frequencies and time indications required by the system are generated within these elements. The master sync is also generated here. Included in the master sync must be provisions for specifying the programs in effect. Since the telemetry detection system will be a word-synchronous system, master sync requirements will not be extreme. The determination of telemetry code length (or basis for word organization) and frequencies of master sync to achieve the minimum acquisition time is an area for study in preliminary design phases.

3) Scientific Instrument Data-Handling Elements. The fact that flexibility is a prime prerequisite for the scientific portion of the data-handling system implies that each of the experiments should be implemented so as to maximize its independence. It can easily be seen that it is not practicable to completely implement a system in this manner. However, if instruments with like characteristics and modes of operation are grouped together to form a separate portion of the data system, the approach to the problem becomes realistic. During the preliminary design, the choice of the elements that comprise the independent portions of the data system becomes the all important aspect of the data-system design. It is essential that weight and power constraints be met while giving consideration to reliability and simplification of operation. These competing characteristics may be incompatible and the system will

require extensive analysis before the final mechanization is chosen. Primary grouping of the scientific instruments at present is by mission phases; planetary and interplanetary. Within these groups, independence has been chosen with reference to the various data-rate classifications.

4) Performance Data-Handling Elements. The basic organization of the performance data-handling measurements into the groups previously mentioned is by subsystem. A performance group is composed of measurements from one or more subsystems. Based upon the list of measurements for the orbiter, a representative grouping is shown below:

Subsystem	Number of Measurements
Guidance	119
Attitude Control	122
Communications, Data Handling and Scientific Instruments	183
Propulsion, SOCC	100
Power, Engineering Mechanics	131
Total	655

The system weight and power estimate is based upon 5 groups of 130 measurements each. However, groups need not be constrained to equal size, only to equal basic subgroup length. Therefore, the possibility of a high priority, low risk group of measurements is not precluded. The subsystem grouping of the orbiter performance measurements has been chosen because of potential problem areas associated with interfaces, system design, and checkout in any other type of organization. With 655 measurements to be considered, these problems could be extreme. In some cases, subsystem organization would permit group data-handling to be accomplished remotely from the master control elements, providing much simpler interfaces. However, this advantage must be weighed against the prospect of losing independence between the measuring device and the measured quantity. Such a case would occur if the data handling and measured quantities were supplied from the same internal power source.

5) Reliability Prospects. In both the orbiter and capsule relay systems the prospects for inclusion of redundancy are good. Because the orbiter system data-handling is adaptive by nature, it would seem that this type of system should also function in the presence of failures. Because there is a great deal of functional independence among the elements of the orbiter data-handling system, a promising reliability approach would be mutual checking of functional elements, especially in the area of logical operations. By providing for common-check sequences or other stimuli for which the response is known, failed portions of the system could be isolated or data could be rerouted. The

prospects for passive techniques such as component and logical redundancy may be somewhat difficult except in areas where the data handling functions are relatively simple; e.g., in the master clocking elements. System design, however, must endeavor to simplify any series elements so that passive redundancy techniques can be implemented.

In the capsule relay data-handling system, the application of passive redundancy techniques is a more attractive approach to reliability. The capsule data-handling systems should be designed to minimize dependence upon external command; this is a natural attribute if passive redundancy is used. Most elements of the capsule data systems will be relatively simple, making it an easier task to implement passive redundancy.

Weight and power estimates for the capsule and orbiter data handling systems reflect the potential application of redundancy techniques.

6. Weight and Power Estimates

a. Assumptions

Tables 4-51, 4-52 and 4-53 show weight and power estimates for the data handling systems under the assumptions outlined in the preceeding subsection 5. The following additional comments apply:

1) Experimental data handling estimates utilize interim component state-of-the-art, i.e., dot and microminiature components. wherever possible. This technology is available now, though in some cases, not flight proven.
2) Performance data-handling estimates reflect state-of-the-art advanced to the integrated circuit stage about 3 years from the present.
3) In some areas where miniaturization has advanced slowly, standard components are assumed. Two such areas are power and data conditioning.
4) Storage elements available now or within the next two years are assumed. storage sizes are as listed in Figures 4-133, 4-134 and 4-135.
5) Scientific instrument and performance transducers weights and power requirements are not included.
6) Total power requirements shown are average power inputs to subsystem converters.
7) Because of possible inclusion of component redundancy in power equipment, 60 percent efficiencies were assumed. Efficiency without redundancy would be approximately 70 percent.
8) Component and logical redundancy measures were assumed for master control elements and multiplexers, as well as for some group programming and conditioning elements.

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Table 4-51. Capsule Relay Data-Handling Weight, Power and Volume

Function	Option I		Option II	
	Weight (lb)	Power (w)	Weight (lb)	Power (w)
Conversion	7.8	13.3	7.8	13.3
Programming	6.3	4.8	5.4	3.8
Storage	36.0	8.9	44.0	9.7
Power	4.6	--	5.0	--
Sub-totals	54.7	27.0	62.2	26.8
Power Converter Loss (60% efficiency)		18.0		17.8
Total Weight and Power	54.7	45.0	62.2	44.6
Volume	2.0 ft^3		2.1 ft^3	

Notes: (a) System sized for 50 w transmitter power level

(b) Estimates for the 25 w transmitter power level have not been computed in detail, since it was previously determined that this level would not satisfy capsule data requirements. Weight and power for the 25 w level would be approximately 43 lb, 37 w for Option II and 38 lb, 36 w for Option I.

Table 4-52. Capsule Direct-Link Data-Handling Weight, Power and Volume

Function	2.0 bps Rate		0.2 bps Rate	
	Weight (lb)	Power (w)	Weight (lb)	Power (w)
Conversion	1.9	4.2	1.9	4.2
Programming	0.6	1.3	0.3	0.7
Storage	19.5	4.6	18.5	4.0
Power	2.3	--	2.0	--
Sub-totals	24.3	10.1	22.7	8.9
Power Converter Loss (60%) efficiency)		6.7		6.0
Total Weight and Power	24.3	16.8	22.7	14.9
Volume	0.6 ft^3		0.6 ft^3	

Table 4-53. Orbiter Data-Handling Weight, Power and Volume

Function	Weight (lb)	Power (w)
Conversion	14.1	29.5
Programming	10.5	11.2
Storage	76.7	21.5
Power	10.5	--
Sub-totals	111.8	62.2
Power Converter Loss (60% efficiency)		41.4
Total Weight and Power	111.8	103.6
Volume	4.6 ft^3	

J. PROPULSION SYSTEMS

1. Orbiter Propulsion Systems

The propulsion system on-board the orbiter will provide impulses for midcourse and approach corrections (possibly several of each), retro into orbit and, perhaps, arrival-time-adjustment and orbit-trim maneuvers. A detailed discussion of systems suitable for these applications for a wide variety of mission constraints has been presented in Volumes I and II of EPD-139.

The arrival-time-adjustment maneuver is an exception to the preceding statement since such a maneuver is meaningful only for a capsule-carrying mission. An arrival-time-adjustment maneuver is required to ensure that the capsule will be able to maintain communication with the bus during the descent through the atmosphere to the planetary surface. In order to accomplish this, either the capsule may be accelerated or the bus decelerated. The difference between the payload masses attainable by the two methods is negligible. The velocity increment requirements for the arrival-time-adjustment maneuver are given in Figure 4-39 of this report.

Since publication of the previous volume, certain features of the proposed bus propulsion system (reference EPD-139, Volume II, Section V. C. 6. d) have been considered in greater depth; a discussion of these considerations follows.

It is now proposed to alter the thrust level previously selected for the proposed system. A thrust-to-mass ratio at initiation of the retro maneuver of 0.15 lbf/lbm had been chosen previously. More recent analysis has shown that the expected minimum total impulse and associated 3σ error for the midcourse maneuvers are acceptable for a motor which provides a thrust-to-mass ratio of 0.3 lbf/lbm at the time of the retro maneuver. The conditions of acceptability are that the value for the minimum impulse "bit" should be substantially less than that required by the smallest correction maneuver that would ever be attempted with the main propulsion system. Also, a 3σ error in final approach maneuver must not result in an unacceptably large error at the planet, or the desired orbit will not be obtained. The increased thrust level has a beneficial effect in that the gravity-burning-time loss during the retro maneuver is reduced. The reduction in the present case amounts to 115 m/sec of effective velocity increment. This results in a retro propellant mass saving of about 180 lb. It may be possible to increase these savings by going to an even higher initial thrust-to-mass ratio, say 0.5 lbf/lbm. More study is required before the higher thrust level can be chosen, but it is probable that the finally selected thrust-to-mass ratio will be between 0.3 and 0.5 lbf/lbm.

The optimized propulsion system parameters, which are given in Table C of EPD-139, Volume II, Part 2, are influenced by the choice of thrust level. For an orbiter total mass of around 5,000 lb, the values for thrust, chamber pressure and expansion ratio become 1,500 lb, 150 psia, and 90:1, respectively. These values have been used to size the propulsion system shown in the configurations presented in this report, Figures 4-139a through 4-146a.

Some consideration has been given to the possibility of experiencing an explosion in the bus during the lengthy period of orbital coast subsequent to the conclusion of scientific measurements. It is believed that the probability of suffering an explosion and contaminating the planet prior to completion of the measurements can be reduced to the required level through proper design. However, in order to meet the sterilization requirement, the orbit will be chosen such that the orbiter will not enter the atmosphere and contaminate the planet for a period of approximately 50 years. As noted in Section II (C) 1. and 2., it is a virtual certainty that some pieces of the unsterilized spacecraft will enter the atmosphere if an explosion involving either the partially filled propulsion system or the attitude control system should occur.

The most likely sources of explosions would involve either leakage, mixing and reaction of the residual propellants, or tank ruptures. Leakage could be caused by a meteorite puncturing the tank walls and the separating diaphragms. The pressure differential resulting from puncture and blowdown of either the fuel or oxidizer chamber would collapse the diaphragms and allow the propellants to mix. Tank ruptures could result from loss of attitude stabilization because the concomitant loss of spacecraft thermal control and subsequent overheating of trapped vapors could overpressurize the tanks.

A modification to the previously proposed bus propulsion system, which would circumvent the explosion problem, consists of adding a system for venting the propellant chambers and the pressurization system. The vent valves would be operated automatically by a long duration timer which could be overridden by command from the Earth. To avoid interaction with the attitude control system, operation of the valves would be delayed until the spacecraft had completed its useful life. If the venting system were used, its reliability would have to be of a very high order, since a premature failure would eliminate the possibility of mission success.

2. Capsule Propulsion Systems

a. Mission Considerations

Propulsion systems on-board the capsule may be required to provide velocity increments for:

1) Deflection of the capsule into an impact trajectory
2) Acceleration of the capsule for arrival-time adjustment
3) Cushioning of the capsule touchdown

The capsule deflection maneuver is required since the avoidance of planetary biological contamination dictates that the bus be placed on a nonimpacting trajectory. The rejected alternative is to place the bus on an impacting trajectory and to deflect it after capsule separation. As stated previously, the arrival-time adjustment may also be provided by the capsule propulsion system. The option exists of performing portions of this maneuver with both the capsule and the bus propulsion systems. The landing cushion application for the capsule arises because parachutes become very ineffective at low speeds. In this regime, the performance of the rocket must be compared to the energy absorbing efficiency of other landing devices such as hydraulic shock absorbers and crushable structure.

The capsule deflection maneuver requires a maximum velocity increment of 50 m/sec (see Figure 4-39). Velocity increment requirements for the arrival-time-adjustment maneuver are given in Figure 4-39 as a function of the required difference in the times of capsule entry and arrival of the orbiter at periapsis. The required velocity increments vary between 120 and 190 m/sec for a time separation of 2 hr and for the range of approach speeds of interest (4.0 to 5.0 km/sec). It is not certain at this juncture that a landing rocket will be used in the touch-down operation. Preliminary analysis has indicated that the maximum velocity increment which might be required for touch-down would be on the order of 20 m/sec. The mass of the rockets will be presented parametrically as a function of the velocity increment; therefore a more precise determination is not required prior to initiation of the preliminary design.

The other parameter needed in propulsion system sizing is the capsule gross mass. The range of allowable capsule gross mass at capsule-bus separation is given in Figures 4-6 through 4-11 for Mars missions in 1969 and 1971, which are the launch opportunities of current primary interest. One may observe that for an orbiter mass close to 1950 lb inserted into a 1000 km by 10,000 km elliptic orbit, the allowable capsule mass is about 1000 lb in 1969 and greater than 2000 lb in 1971. As indicated in Chapter V of this report, primary consideration has been given to capsule masses on the order of 1000 lb. Since many items are jettisoned prior to touch-down, the ratio of landed mass to capsule mass must also be known in order to size the landing rocket. Preliminary estimates for this ratio range between 0.3 and 0.55.

A primary requirement of the capsule motor is that it be sterile. As presently envisaged, such a requirement is tantamount to a requirement that the entire capsule be designed to endure a post-assembly heat sterilization. A minimum amount of developmental experience exists for propulsion systems capable of being heat sterilized.

Other requirements occur in the areas of reliability, performance, geometric restrictions, impulse magnitude, thrust vector control, and compatibility with a spin environment, if the capsule is to be spin-stabilized. Payload performance, measured inversely in terms of the total propulsion system mass, should in general be maximized. However, for the relatively small units which are being considered here, the advantage of the minimum-mass (maximum performance) propulsion design may amount to only a few pounds. Therefore, the choice of the propulsion system may not be based on performance considerations per se. The design of smaller propulsion systems will be strongly controlled by the space available. For example, the volume and length available in the capsule is governed by the shape of the heat shield and the capsule center-of-gravity location required for aerodynamic stability. Impulse magnitude must be controlled to the extent that the expected error will not cause the capsule to land outside of the desired impact area. Thrust vector control is necessary if the capsule is not spin-stabilized, except perhaps in system designs incorporating a high thrust-to-mass ratio (short burning-time). For this exceptional case, the motor burning phase would be completed before an excessive deflection of the added velocity vector could develop. Systems of this type are suitable only when relatively small velocity increments are to be attained.

b. Propulsion System Design and Performance

(1) Solid Propellant Rocket Motors. Figure 4-136 illustrates propulsion mass requirements for separation and acceleration of a 1000-lb entry capsule using solid propellant motors. The two curves represent two design approaches to the capsule sterilization requirement:

1) Sterile casting and assembly
2) Terminal heat sterilization

The lower curve of Figure 4-136 represents a high performance, sterilely cast and assembled motor. The design concept is an internal burning, case-bonded motor with a submerged nozzle, similar to motors in existence and being developed. Motors of this type should be available for use on a Voyager capsule with a delivered vacuum specific impulse of 300 lbf-sec/lbm and a propellant mass fraction of 0.90. These performance numbers were used for both Figures 4-136 and 4-137, except that, for the smaller velocity increments, the mass fraction was considered to decrease gradually from 0.90 at ΔV = 200 m/sec to 0.80 at ΔV = 50 m/sec.

Figures 4-137a and b depict the configurations of both the sterilely-assembled and the terminally heat-sterilized designs. Two sizes of each motor are represented. The larger yields a velocity increment of 200 m/sec, which corresponds to a combined deflection and arrival-time-adjustment maneuver. The smaller design, which yields 60 m/sec, velocity increment, provides a full deflection maneuver, but only a partial arrival-time adjustment. The remainder of the arrival time correction would be provided by deceleration of the orbiter. The smaller capsule maneuver would be used only if aiming errors precluded provision for

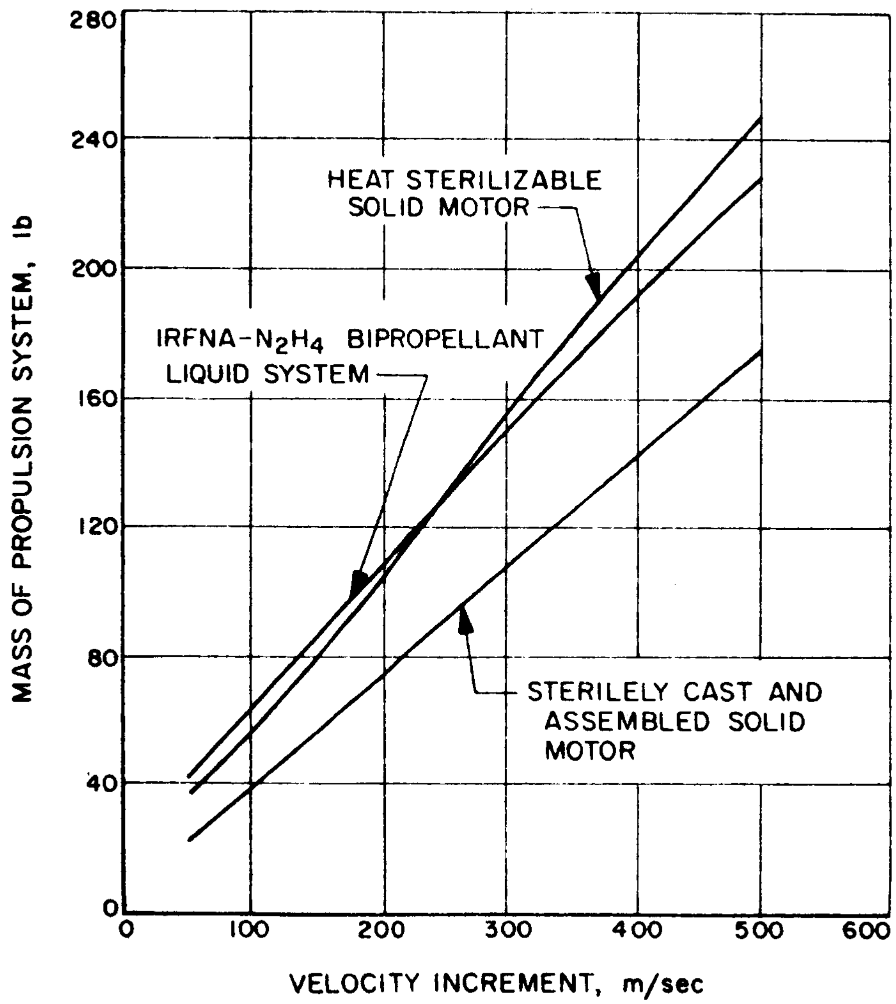
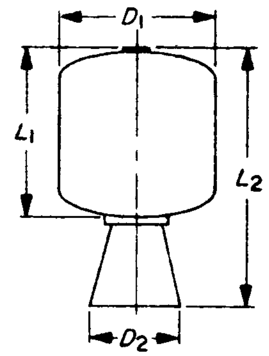


Figure 4-136. Mass of Propulsion Systems for Separation and Acceleration of 1000 LBM Entry Capsules

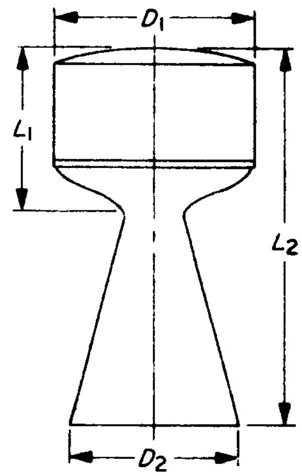
(a) STERILELY-CAST AND ASSEMBLED MOTOR

VELOCITY INCREMENT, m/sec	60	200
SPECIFIC IMPULSE, lbf-sec/lbm	300	300
PROPELLANT MASS FRACTION	0.80	0.90
EXPANSION RATIO	35:1	35:1
MAXIMUM THRUST, lbf	700	1540
BURNING TIME, sec	10.6	15.8
D_1, in.	8.3	12.3
D_2, in.	4.7	7.0
L_1, in.	8.9	13.1
L_2, in.	13.6	20.2



(b) TERMINALLY HEAT-STERILIZED MOTOR

VELOCITY INCREMENT, m/sec	60	200
SPECIFIC IMPULSE, lbf-sec/bm	230	230
PROPELLANT MASS FRACTION	0.60	0.80
EXPANSION RATIO	35:1	35:1
MAXIMUM THRUST, lbf	1350	2950
BURNING TIME, sec	4.46	6.59
D_1, in.	10.8	15.9
D_2, in.	9.2	13.6
L_1, in.	8.6	12.8
L_2, in.	20.3	30.0



(c) BIPROPELLANT INHIBITED RED FUMING NITRIC ACID-HYDRAZINE

VELOCITY INCREMENT, m/sec	60	200
SPECIFIC IMPULSE, lbf-sec/lbm	301	301
PROPELLANT MASS FRACTIONS	0.44	0.62
EXPANSION RATIO	35:1	35:1
THRUST, lbf	10	33
BURNING TIME, sec	600	600
CHAMBER PRESSURE, lbf/in^2	150	150
MIXTURE RATIO	1.4	1.4
D, in.	11.0	16.3
L_1, in.	12.4	18.3
L_2, in.	17.3	23.4

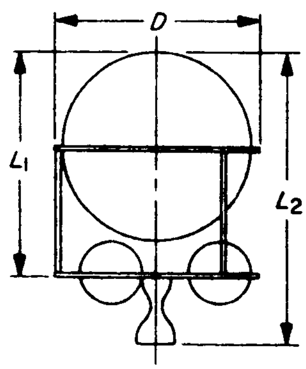


Figure 4-137. Configurations for Capsule Propulsion Systems Gross Mass = 1000 LBM

the full arrival-time adjustment by capsule propulsion. Provision for the partial arrival-time adjustment in the smaller capsule maneuver results in an attitude of the capsule which is compatible with communication requirements (see Figure 4-38).

Production of the high-performance, sterilely assembled motor is accomplished by a method now in the final stages of development. Briefly, sterilization is accomplished by the following steps:

1) The inert parts, igniter assembly, and propellant oxidizer are heat sterilized.
2) Liquid ethylene oxide is added to the propellant during the mix cycle, thus sterilizing the propellant. The ethylene oxide evaporates during propellant curing.
3) The propellant is cast and the motor is assembled in a sterile environment.

The sterilely cast and assembled motor design would be definite first choice from a mass and size point of view; however, it does not meet the current ground rule that all Voyager capsule systems must be terminally heat sterilizable. For this reason, further discussion in this section will deal with a conceptual design of the heat-sterilizable, solid-propellant capsule separation motor.

As implied by the proposed sterile assembly method, solid motor inert components and igniter assemblies have been developed which will withstand terminal heat sterilization with no apparent degradation of properties. Attempts to develop a high-performance, case-bonded, heat-sterilizable motor, however, have been beset by failures in one or both of the following modes:

1) The propellant grain has cracked or melted.
2) The propellant-liner and/or liner-case bonds have failed.

The weaknesses which lead to these failures have generally been traced to a combination of propellant properties, liner properties, and differences in coefficients of thermal expansion.

The mass of a terminally heat sterilizable motor, 50 percent heavier than the corresponding sterilely cast and assembled motor, is given by the middle curve of Figure 4-136. This curve is based on an estimated specific impulse of 230 lbf-sec/lbm at a 35:1 expansion ratio and a propellant mass fraction of 0.80. However, the lower end of the curve has been adjusted to account for decreased mass fraction on motors of less than 100 lbm. The corresponding configuration for the terminally heat sterilizable motor is shown in Figure 4-137b.

The low performance, highly heat resistant propellant proposed for use in the heat sterilizable design is formed into a single tubular grain. Lack of stress concentrations in such a grain and the propellant's heat resistant properties should provide sufficient grain integrity during sterilization heating. The thermal expansion and bonding problems will be solved by using a free standing, cartridge-loaded grain with the ends inhibited. This allows burning on both the inside and outside surfaces of the cylinder. The grain supports will be designed to allow a maximum degree of propellant expansion consistent with a sound mechanical design. A spider-web trap, placed across the entrance to the nozzle and supported by a rod passing axially through the center of the grain to the motor head end, will hold the grain in place during burning.

The propellant performance quoted for the heat sterilizable motor is based on the results of extensive tests of a propellant which has been successfully fired at 500°F.[1)] JATO motor inert masses were used to arrive at the quoted mass fraction by first scaling to the propellant mass for a 200 m/sec velocity increment and then adjusting the case mass to accommodate a chamber pressure at 400 psi. Prior to initiating a motor development program, further propellant testing should be done to guarantee the successful performance of this propellant and to reevaluate other propellants with a somewhat higher performance level.

A more extensive analysis should also go into the optimization of the configuration. If considerations of spacecraft configuration (space available) or motor-capsule dynamic stability (location of c.g.) place restraints on the length of the motor or if the high-thrust, short-burning-time separation scheme is desired, the motor would probably appear quite different from Figure 4-137b. For example, it might be feasible to use several tubular grains, lying side-by-side with axes parallel to the thrust axis, rather than the single grain proposed. Also the designed chamber pressure, propellant burning rate, and nozzle expansion ratio can be varied to fit the design requirements. Figure 4-138 shows how the multiple grain configuration could lead to a wide choice of motor geometries. Generally, the length tends to increase as burning-time decreases for a given number of cylindrical grains. Also, the motor performance tends to decrease as the number of grains is increased at a given burning time. The propulsion system mass could be as much as 20 percent greater than that shown in Figure 1 for a motor with a short (1 to 2 sec) burning time. This mass increase is due to decreased volumetric loading with a multiple grain design and decreased propellant performance due to a shortened nozzle. If, in the future, the short-burning-time, high-thrust separation scheme becomes more attractive than it is now, further analysis should be made of the multiple grain design. Even without any changes in capsule constraints, the geometry and ballistic parameters should be optimized to give near maximum performance.

An intertial guidance autopilot system coupled with altitude control jets has been proposed as an alternate to spinning for capsule stabilization. The solid capsule separation motor, if a solid motor were used, would have to provide a separate thrust vector control (TVC) system because the side thrusts encountered would be too high for reasonably-sized attitude control jets to handle. A mechanical (jet tabs) or fluid injection TVC system for the motor could be controlled by the capsule guidance system. Either system would add about 10 percent to the total propulsion system mass for a velocity increment range of 100 to 300 m/sec.

Capsule motor variances are of interest for guidance accuracy calculations. Generally, the two cases that must be considered are motor burnout, and thrust termination:

1) Motor Burnout. The propulsion system mass curve of Figure 4-136 is based on a motor which burns to fuel depletion. No data is available for the total impulse variances of heat sterilizable motors, but analyses of high temperature tests of the propellant under consideration show no trend toward significant increases in these variances. The following 3σ limits can be assumed for total impulse and burning time variances at a known temperature:

Total impulse	±0.5%
Burning time	±2.5%

The total impulse increases and burning time decreases as temperature increases. If the bulk grain temperature at the time of firing is known, the temperature effects can be calculated from the following relationships:

Total impulse	$\% \Delta I_t = 0.006 \ \Delta t\ (°F)$
Burning time	$\% \Delta t_b = 0.176 \ \Delta t\ (°F)$

If the temperature at the time of firing is unknown, these relationships must be combined with estimates of temperature variation to evaluate the total uncertainties in total impulse and burning time.

2) Thrust Termination. A thrust termination system can be built into the capsule separation motor if desired. The system would consist of an accelerometer-operated blow-off nozzle and separation ports at the motor head end. A 5 percent increase in propulsion system mass (in addition to shielding of the capsule aft end from hot gases) would be required to incorporate the system. Total impulse variances of this system would be set primarily by the accuracy of the integrating accelerometer which controls cutoff, since the variances in tailoff velocity-increment are extremely small compared to the overall velocity increment. The variances would be at least as small as those quoted for a motor designed to run to burn out. While the use of a thrust termination system provides some mission flexibility, the increased complexity and mass of such a system cause one to recommend a fixed impulse system.

The possible use of propulsion for a small landing impulse was mentioned in Section 2.A. p. 4-344, Mission Considerations. The heat-sterilizable solid motor would be well suited for this application. Assuming a required velocity change of 20 m/sec, a propellant specific impulse of 230 lbf-sec/lbm, and a propellant mass fraction of 0.5; a solid-motor-mass-to-capsule-landed-mass ratio of 0.0175 will be required. The low propellant mass fraction assumed is due to the small size of the motor.

(2) Storable Liquid Propellant Systems. Three terminally heat sterilizable liquid systems were considered for capsule applications: hydrazine monopropellant, Cavea B monopropellant, and nitric acid-hydrazine bipropellant. The requirement for terminal heat sterilization may be satisfied by providing adequate ullage volume and propellant tank strength to accommodate the tank pressure increase which occurs during the heat cycle. Material incompatibility and/or thermal decomposition of the propellants at the elevated sterilization temperatures are not anticipated, but tests verifying this assumption will be required.

As a result of the initial comparison, only the nitric acid (IRFNA)-hydrazine bipropellant combination was retained for further consideration. The propulsion system mass characteristic of this system is compared with the solid systems in Figure 4-136. The performance of either of the monopropellant systems is inferior even to the relatively low impulse, heat-sterilizable, solid system. Also, the monopropellant systems occupied larger volumes and are functionally more complicated than the solid motor. One might question the retention of the bipropellant system since, while it is similar in size and performance to the solid system, it is even more complicated than the monopropellant system. In comparison with the solid motor, the bipropellant system's low thrust characteristics might permit a smaller capsule-orbiter separation distance prior to capsule motor start-up. Further, if it were decided to use a gyro-stabilized capsule with cold-gas jets for initial separation and for coast-phase attitude control, the liquid system thrust level could be designed low enough to permit use of these same jets for attitude control during burning.

Figure 4-137c shows the configuration and design parameters for the bipropellant system. The propellant feed system of the proposed spin-stabilized system utilizes separate nitrogen pressurization containers. The spherical propellant tank is divided fore and aft into oxidizer and fuel chambers by a rigid bulkhead. Ullage separation is provided by the spin environment. The propellants are expelled from the tank above and below the bulkhead at the bulkhead-tank intersection. The feasibility of a somewhat similar ullage separation scheme has been demonstrated in a monopropellant system using a toroidal tank.[2)3)] A propellant

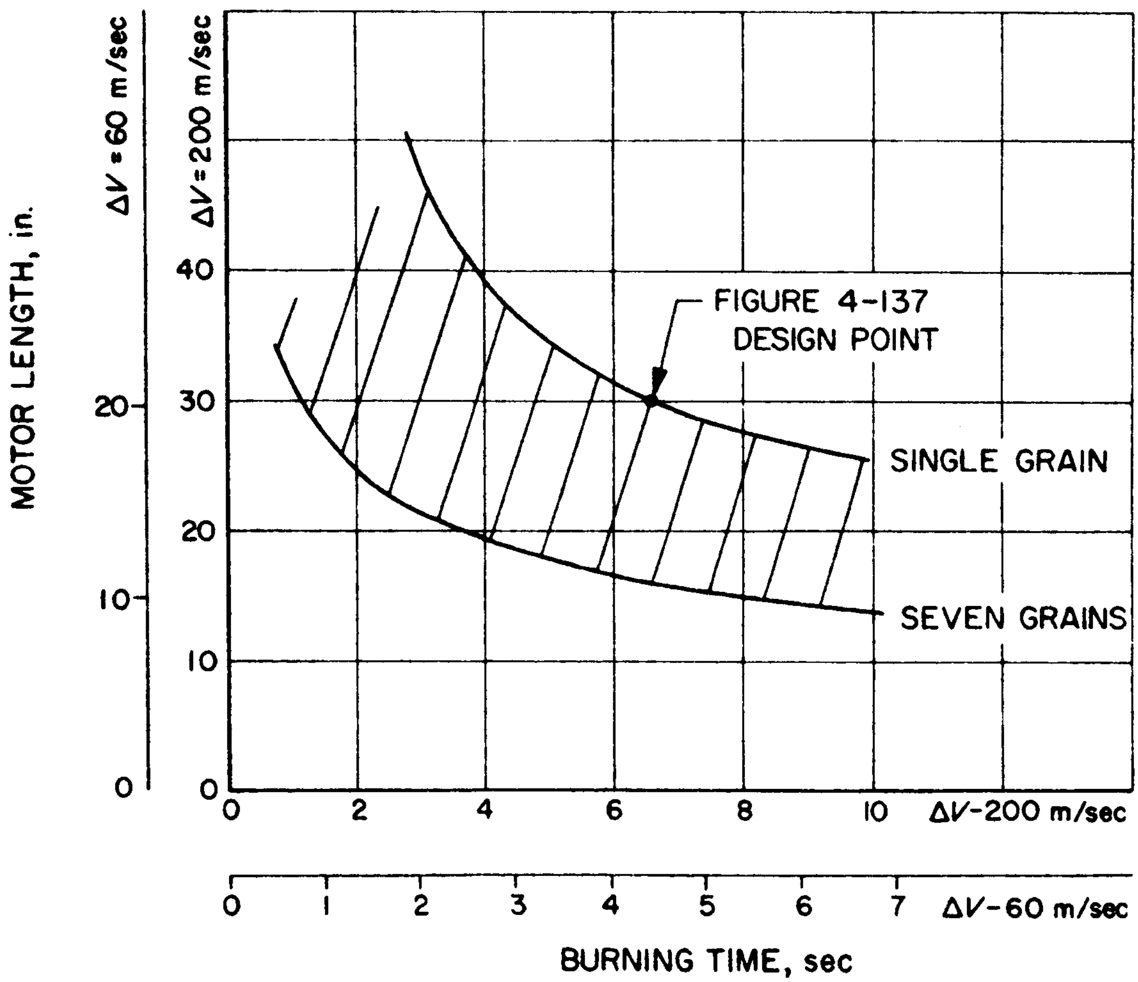


Figure 4-138. Typical Lengths of Heat-sterilizable Tubular-Grain, Solid-Propellant, Capsule-Separation Motors as a Function of Burning Time for Two Velocity Increments.

expulsion efficiency of 96 percent was obtained with this system in tests run to fuel depletion. If a gyro-stabilized capsule were used, positive expulsion schemes, similar to those discussed for the orbiter propulsion system, would be incorporated for ullage separation.

Thrust levels shown in Figure 4-137c correspond to an arbitrarily long burning-time (600 sec). Longer burning times could be used. However, the advantages in terms of lower system mass are negligible, since system operating pressures are fixed by heat sterilization and connot be lowered with lower thrust. The empty propulsion system masses were read from curves presented in another study.[4)] Slight modifications were made which reflect work done since those curves were prepared. Other configurations which would result in liquid systems shorter than that shown in Figure 4-137 are possible. For example, elliptical or toroidal propellant tanks could be used. Also, the pressurizing gas could be stored in the propellant tank. A blowdown system such as this would eliminate the separate nitrogen pressurization system. The system described in References 2) and 3) uses both a toroidal tank and a blowdown pressurization system. These alternatives would increase the propulsion system mass, but this may be justified by the configurational requirements of a specific capsule design.

For a landing motor application, the liquid systems would not be competitive with a solid unit because of the very small amount of total impulse required.

c. Proposed Capsule Propulsion System

Three propulsion systems have been presented for consideration for the Voyager mission. These are:

1) A sterilely-cast and assembled solid motor.
2) A heat-sterilizable solid motor.
3) A heat-sterilizable bipropellant liquid system.

Figures 4-136 and 4-137 show masses and sizes of these motors for velocity increments of 60 m/sec and 200 m/sec. Assuming that these velocity increments span the range of interest and that the capsule will be spin-stabilized, the sterilely cast and assembled solid motor 1) would be an obvious first choice because of its high performance and low volume. The current terminal heat sterilization requirements, however, require that a choice be made between 2) and 3).

The heat-sterilizable solid motor shows performance and possible configurational advantages over the bipropellant liquid system for the size range of interest. In addition, the inherent simplicity of a spin-stabilized solid motor tends to support its choice. Assuming that the solid motor design proposed, or a similar design, can be developed to satisfy the heat sterilization requirements, system b. is recommended for use on the Voyager capsule. The choice between systems b. and c. must be reconsidered should any of the following contingencies arise:

1) Severe sterilization problems occur in the development of system b.,
2) The interaction of the capsule propulsion system exhaust with the orbiter spacecraft requires excessive post-separation coast duration, or
3) The angular accuracy requirements preclude the use of the proposed spin-stabilization scheme.

REFERENCES

1. Development of Solid Propellant for Operation over the Temperature Range of -65 to 500°F, G. Gustavson, et al., AGC, Sacramento, Rept 0296-01F, August, 1959 - August 1960, September 20, 1960.

2. Research Summary No. 1, Jet Propulsion Laboratory, February 15, 1959, pp. 53-55. CONFIDENTIAL.

3. Research Summary No. 2, Jet Propulsion Laboratory, April 15, 1959, pp. 30-31, CONFIDENTIAL.

4. Lee, D.H., et al., Applications for Monopropellants in Space Vehicles, Technical Report No. 32-174, Jet Propulsion Laboratory, October 5, 1961, CONFIDENTIAL.

K. SPACECRAFT CONFIGURATIONS

1. Bus Configuration

The bus configuration studies conducted during this phase of the APSS work are described in the following section. In order to coherently follow this discussion the reader should have read the corresponding sections in EPD-139, Volume II.

The major attention in these configurational studies was devoted to investigating the perturbations on the basic bus by the addition of an atmospheric entry and landing capsule with a mass of 1000 to 2000-lb. The major features of the main bus which were kept constant were the power system with either a large solar panel area or some equivalent combination of radioisotope thermoelectric generator units, the planet-oriented science instrument pointing capability, the available packaging volume for electronic equipment, and the available propellant volume (although some changes in the propulsion system geometry were affected). An 8-ft pointable high-gain antenna is shown on all configurations with the 154-in. upper-stage diameter. A folded 12.5-ft pointable high-gain antenna can be used interchangeably with these configurations with the proper folding technique. The most attractive folding technique being considered is one in which only the outer portion of the antenna is folded. It is folded in such a manner that, if it should fail to erect, there is still a major portion of the reflector available allowing the antenna to operate at a reduced gain. If the 260-in diameter upper stage (S-VI) were used, then the 12.5-ft antenna would not have to be folded. This and other configurational considerations of using this larger stage will be considered in further detail in subsequent studies.

Several aspects of the capsule-bus interface were established and maintained constant in all the configurations considered. A short adapter section exists between the capsule and bus. This adapter section accommodates some fundamental aspects of the interface. On one end of the adapter is the capsule in-flight separation joint (both electrical and mechanical); at the other end of the adapter is the capsule-bus field joint. This joint is the critical interface between the capsule and bus in that all the basic capsule loads will be routed through this joint. The bus can then be designed to accept any capsule whose adapter section can be accommodated by this joint. This allows major changes in the size, shape, and weight of the capsule without effecting the basic structural design of the bus. This kind of growth potential and flexibility is mandatory in considering a spacecraft design which is to be used over an extended period of opportunities and is to take advantage of the design evolution

The configurations to be discussed show alternate approaches in the power and propulsion subsystems. The power subsystem employs either a solar photovoltaic cell system, with an available panel area up to 360-ft^2, or a radioisotope thermoelectric generator system. The radioisotope thermoelectric generator system is composed of several separate generators.

Each generator is a self-contained power unit with its own radiator, fuel, thermo-elements and insulation. Configurations using both three and six units have been investigated with a slight preference for using three units. This yields a higher power-to-weight ratio and a more harmonious integration with the multi-engine propulsion system.

The variations in the propulsion system which have been studied are the tankage configuration and the number of engines used for primary thrust. Three basic tankage configurations have been investigated, these are:

1) A single spherical tank with diaphragms separating the fuel and oxidizer compartments.
2) Six small spherical tanks, three fuel tanks and three oxidizer tanks.
3) Propellant tanks integral with the basic structure.

The third method offers a potential structural weight advantage by using six cylindrical propellant tanks as the main load-carrying members of the basic structure. These six tanks are at the points of a basic hexagonal structure, the electronic equipment is then packaged between these tanks. With this arrangement, the proximity of the fuel and electronics enhance the thermal control system by easing the problem of keeping the fuel warm and by increasing the thermal inertia of the spacecraft electronic compartments; thus, making their temperature control less sensitive to transients from changes in orientation during midcourse and approach maneuvers. This advantage is somewhat reduced after the retro maneuver because of the reduction in fuel content. The multi-propellant tank system may also lead to instability during engine start. Slight tank-to-tank variations in the stiffness of the expulsion bladders can produce unequal distribution of the propellants during the transit phase; the resultant shift in the center of gravity may be difficult to accommodate during subsequent firing of the propulsion system. This situation can be controlled by the addition of six valves, one at each tank outlet; this solution, however, will reduce the system reliability and increase its complexity.

Both single-thrust-chamber and multiple-thrust-chamber configurations have been considered for the propulsion system. In the single-thrust chamber system the engine is gimballed for spacecraft pitch and yaw control with the spacecraft attitude control system providing roll control during the propulsion maneuver. The multiple-engine system uses three throttleable engines, one of which is hinged for roll control. The three-engine system offers configurational advantages in that it allows significant shortening of the spacecraft and adapter. These advantages must be weighted against the increase in system complexity of the three-engine system and its associated decrease in reliability.

In addition to the above-mentioned equipment, a capsule receiving antenna was added to the bus for the lander-to-orbiter VHF relay link. This antenna is mounted with the planet oriented science equipment and points to the center of the planet. The antenna has a 47 degree half-cone angle which allows for a total included planet angle of 94 degrees corresponding to a lowest orbiter periapsis altitude of 1800 km.

2. Capsule Configuration

In studying the bus configuration it was necessary to assume a capsule aerodynamic shape around which the bus design could evolve. For the purpose of this configuration study the Discoverer shape was used as a typical capsule shape. Section II (L) will consider the capsule design in detail.

a. Configuration Number 1 (Figures 4-139a, 4-139b and 4-139c)

This configuration has a single-engine, single-tank propulsion system. Separate fuel and oxidizer compartments are contained in a spherical tank from which the engine is gimballed for pitch and yaw control. The attitude control system supplies roll control. Spacecraft electrical power is provided by a solar conversion system using solar cells as the collectors. The solar panel array, with a total available area of 354-ft^2, is composed of a fixed panel of 40-ft^2, mounted on the Sun side of the electronic equipment, and six folding panels. Each of the six folding panels has two smaller folds which erect to complete the array as shown in Figure 4-139.

The communication system on the spacecraft operates at both S-band and VHF frequencies. The S-band is used for the spacecraft-to-Earth link and is maintained with the high-gain antenna. This antenna has a two-degree-of-freedom pointing capability and is pointed toward the Earth for the cruise and in-orbit portions of the mission. The VHF antenna, used for the lander-to-orbiter communication link, is mounted with the planet-oriented science. It operates in its stowed position during the interval between capsule-bus separation and the retro maneuver, and is pointed toward the center of the planet after the retro maneuver is completed.

Attitude control is provided by a redundant, three-axis cold-gas system which, although considerably heavier than a comparable hot-gas system, appears inherently more reliable. The Sun-Canopus system is used for spacecraft orientation. The Sun is used as a pitch and yaw reference, so that the vehicle is oriented with the propulsion system pointed toward the Sun with the plane of the solar panels normal to the Sun line. Roll control is provided by the secondary reference, Canopus. This attitude control system is discussed in greater detail in Section II (E) of this document.

The capsule is shown with the propulsion and attitude control systems described in the capsule section (Section II (F).

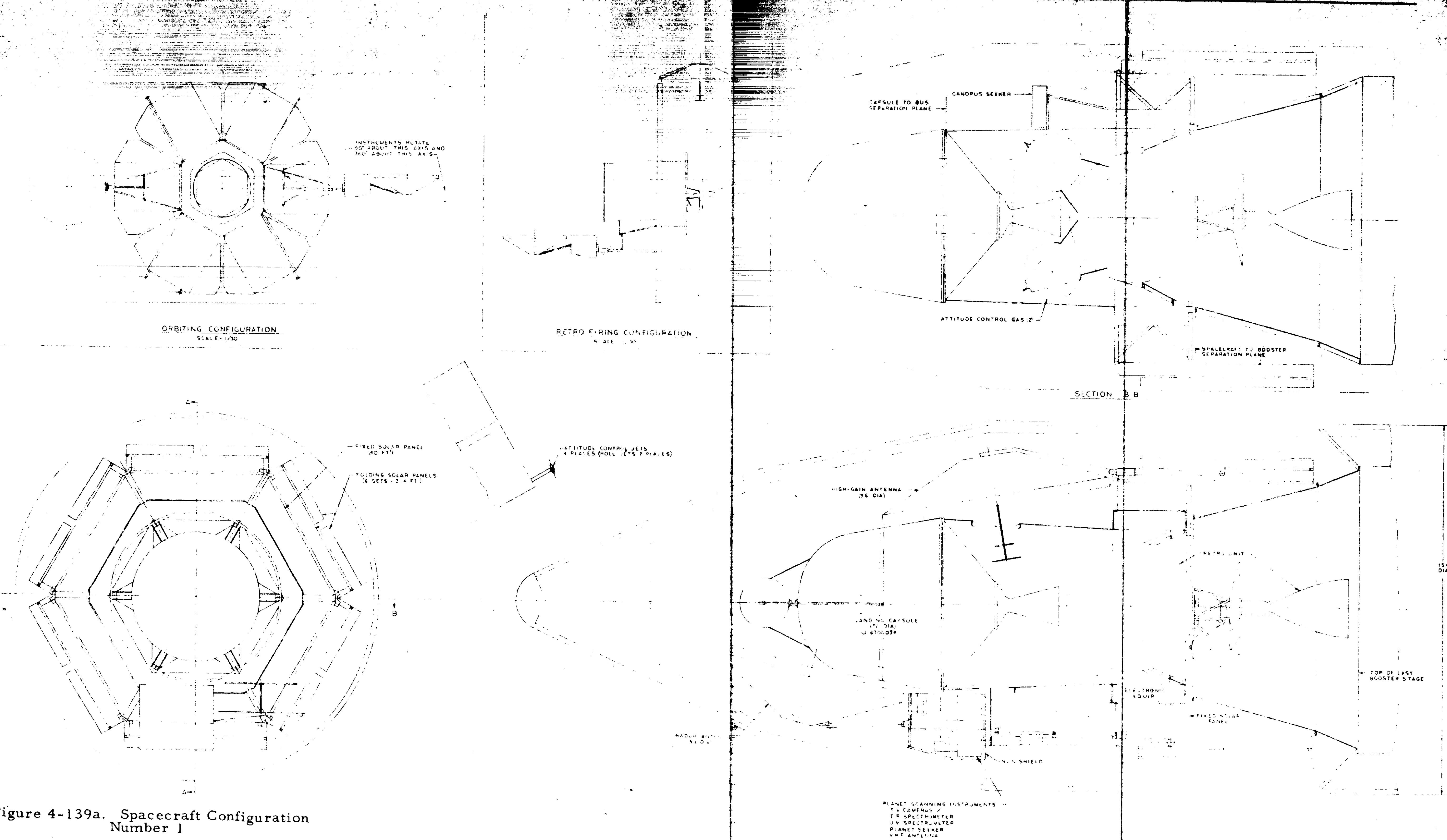


Figure 4-139a. Spacecraft Configuration Number 1

Configuration	Weight	Center of Gravity(1)			Moments of Inertia(2)			Ballistic Coefficients(3)		Mass-to-Area Ratio(4)
	W (lb)	X (in.)	Y (in.)	Z (in.)	I_X (slug ft^2)	I_Y (slug ft^2)	I_Z (slug ft^2)	$\left(\frac{M}{C_DA}\right)_N$ (slug/ft^2)	$\left(\frac{M}{C_DA}\right)_Z$ (slug/ft^2)	$\left(\frac{M}{A}\right)_Z$ (slug/ft^2)
Launch, 100% Fuel	6550	-0.48	0.43	463.8	2980	2840	1230			
Cruise With Capsule; 100% Fuel	6550	-0.48	-0.04	459.1	3940	3040	2970			0.0031
Cruise Without Capsule; 100% Fuel	5550	-0.57	-0.05	473.4	2200	1300	2840			
Cruise Without Capsule; 0% Fuel	2370	-1.34	-0.11	472.5	2200	1300	2840			
Orbit; 0% Fuel	2370	-1.34	2.79	476.1	2440	1130	3240	0.41	0.080	

Notes: (1) X-pitch reference axis, Y-yaw reference axis, Z-roll reference axis, Z coordinate is referred to spacecraft separation plane, defined as station 500.

(2) All I's about axes through center of gravity parallel to reference axes.

(3) $\left(\frac{M}{C_DA}\right)_N$ is ballistic coefficient normal to z-axis, $\left(\frac{M}{C_DA}\right)_Z$ colinear with z-axis, where A is the maximum appropriate area and $C_D = 2$ was used for this analysis.

(4) $\left(\frac{M}{A}\right)_Z$ is mass-to-area ratio colinear with z-axis.

Figure 4-139b. Configuration Number 1 Design Summary

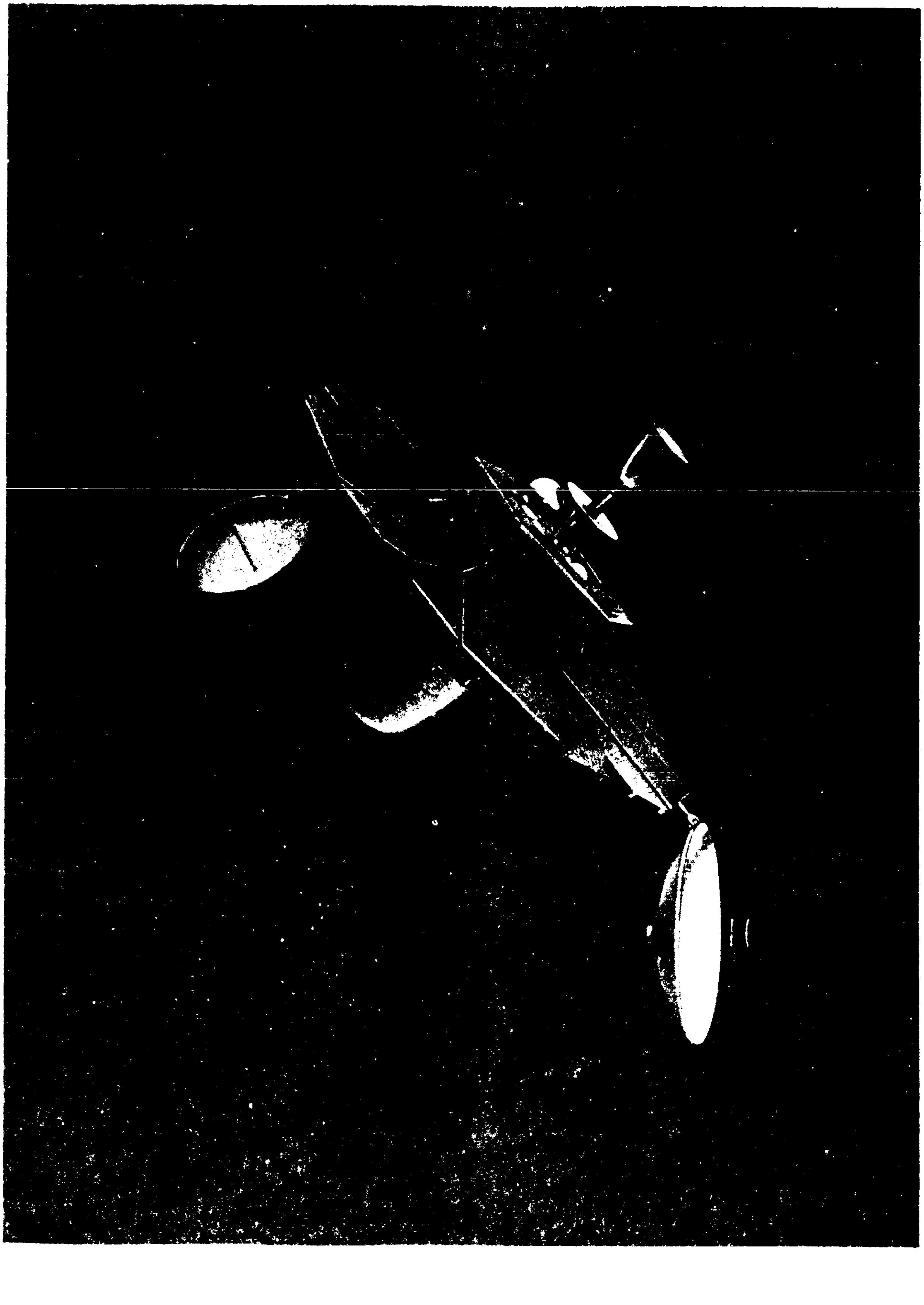


Figure [illegible] Artist's Rendering of Spacecraft Configuration Number 1

The capsule, with its propulsion system, and the single-tank single-engine bus propulsion system lead to an extremely long configuration, and past experience has highlighted some basic inherent disadvantages in the structural requirements and dynamics of such vehicles. Also, the look angles, or field-of-view requirements, are most difficult to satisfy with this type of configuration without resorting to excessively long booms for the instruments and antennas. The resulting low-resonant-frequency structures require complicated support mechanisms during boost and also cause undesirable coupling with the control system. Finally, the use of a long, heavy adapter section is a payload performance penalty which the spacecraft must absorb.

b. Configuration Number 2 (Figure 4-140a, 4-140b and 4-140c)

The communication system, attitude-control system and capsule are similar to those discussed in Configuration Number 1.

In order to alleviate some of the attendant problems of Configuration Number 1, the single propellant tank was replaced with six cylindrical tanks. This allows use of the tanks as load-carrying columns and enables the capsule to be lowered into the center of the spacecraft. The length was further reduced by using a multiple-engine propulsion system. It is readily apparent that the shortening of the spacecraft adapter section and capsule adapter results in a weight advantage. The potential control problem associated with the multi-tank configuration, which was discussed previously in this section, is a drawback of this configuration.

With the propulsion tanks integral with the structure, the basic size of the hexagon was increased to achieve the same electronic volume as in Configuration Number 1. This allows a larger fixed-panel area and support of the fold-out panels closer to the main structural columns. With this type of panel support, external bracing is no longer required, thus improving the panel support structure. This also provides a continuous, uncluttered exterior surface which simplifies the temperature control system by reducing secondary reflections.

c. Configuration Number 3 (Figures 4-141a and 4-141b)

In this configuration the electrical power system has been changed from a solar cell system to a radioisotope generator system. This particular design is composed of six individual generators, any five of which can supply the maximum expected power requirements; thus, in the event of a failure of any one unit, the spacecraft can still function properly.

Each of the six generators is mounted adjacent to one of the six spherical propellant tanks and is separated from the spacecraft electronic equipment by a nuclear radiation shield. With this arrangement, the thermal radiation from the generators can be used to keep the fuel warm without affecting the electronic equipment. Each generator has almost a 360 degree

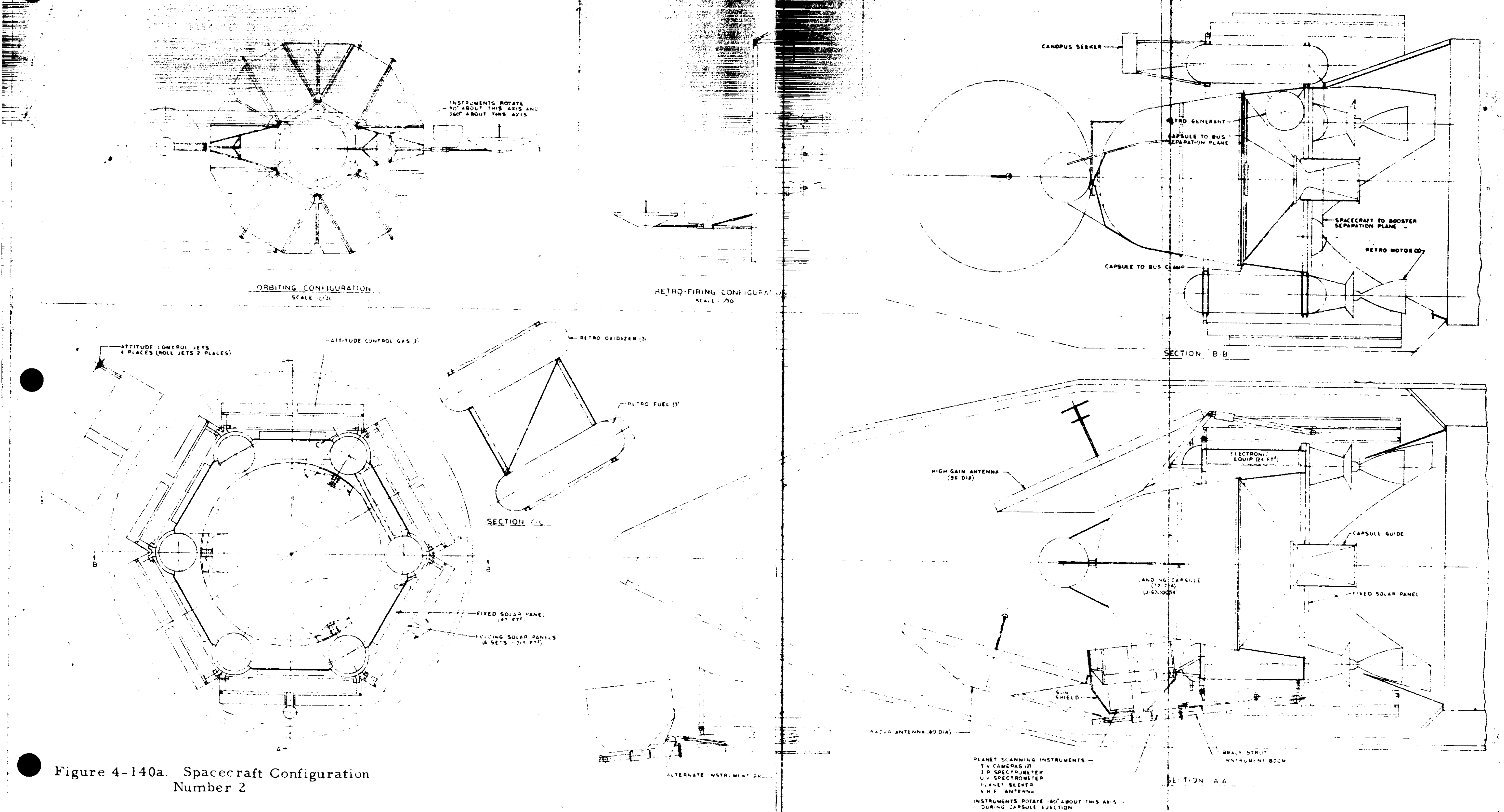


Figure 4-140a. Spacecraft Configuration Number 2

Configuration	Weight	Center of Gravity[1]			Moments of Inertia[2]			Ballistic Coefficients[3]		Mass-to-Area Ratio[4]
	W (lb)	X (in.)	Y (in.)	Z (in.)	I_X (slug ft^2)	I_Y (slug ft^2)	I_Z (slug ft^2)	$\left(\frac{M}{C_DA}\right)_N$ (slug/ft^2)	$\left(\frac{M}{C_DA}\right)_Z$ (slug/ft^2)	$\left(\frac{M}{A}\right)_Z$ (slug/ft^2)
Launch; 100% Fuel	6620	-0.27	0.44	467.0	2640	2520	3020			
Cruise With Capsule, 100% Fuel	6620	0.27	-0.56	461.9	3680	2950	4570			0.0031
Cruise Without Capsule, 100% Fuel	5620	-0.32	-0.66	468.2	3220	2500	4450			
Cruise Without Capsule, 0% Fuel	2440	-0.73	-1.51	463.2	2200	1480	2840			
Orbit, 0% Fuel	2440	0.73	2.66	466.0	2550	1300	3380	0.36	0.082	

Notes: (1) X-pitch reference axis. Y-yaw reference axis. Z-roll reference axis. Z coordinate is referred to spacecraft separation plane, defined as station 500

(2) All I's about axes through center of gravity parallel to reference axes.

(3) $\left(\frac{M}{C_DA}\right)_N$ is ballistic coefficient normal to z-axis. $\left(\frac{M}{C_DA}\right)_Z$ colinear with z-axis, where A is the maximum appropriate area and $C_D = 2$ was used for this analysis

(4) $\left(\frac{M}{A}\right)_Z$ is mass-to-area ratio colinear with z axis

Figure 4-140b. Configuration Number 2 Design Summary

4-1

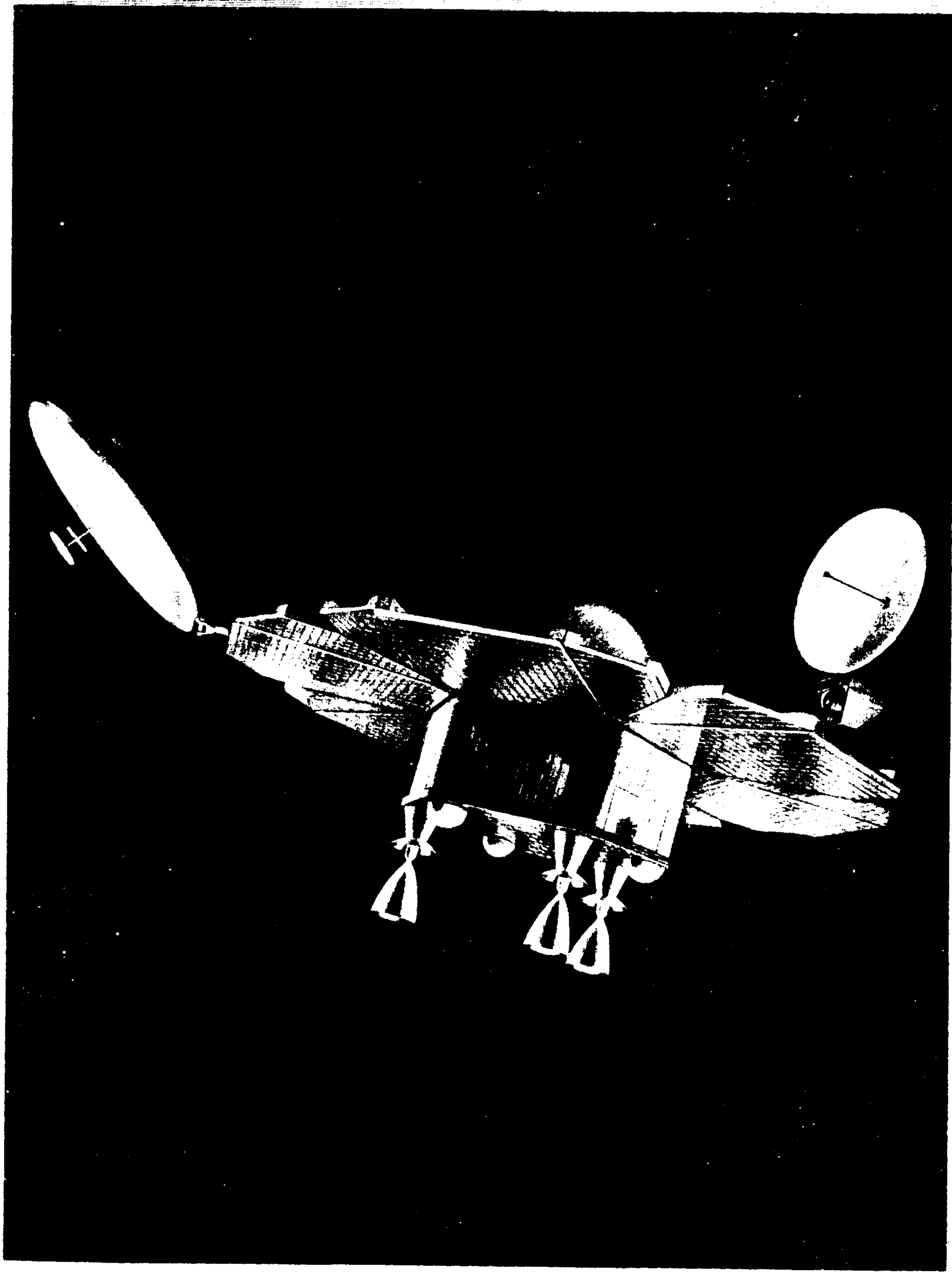


Figure 4-140c. Artist's Rendering of Spacecraft Configuration Number 2

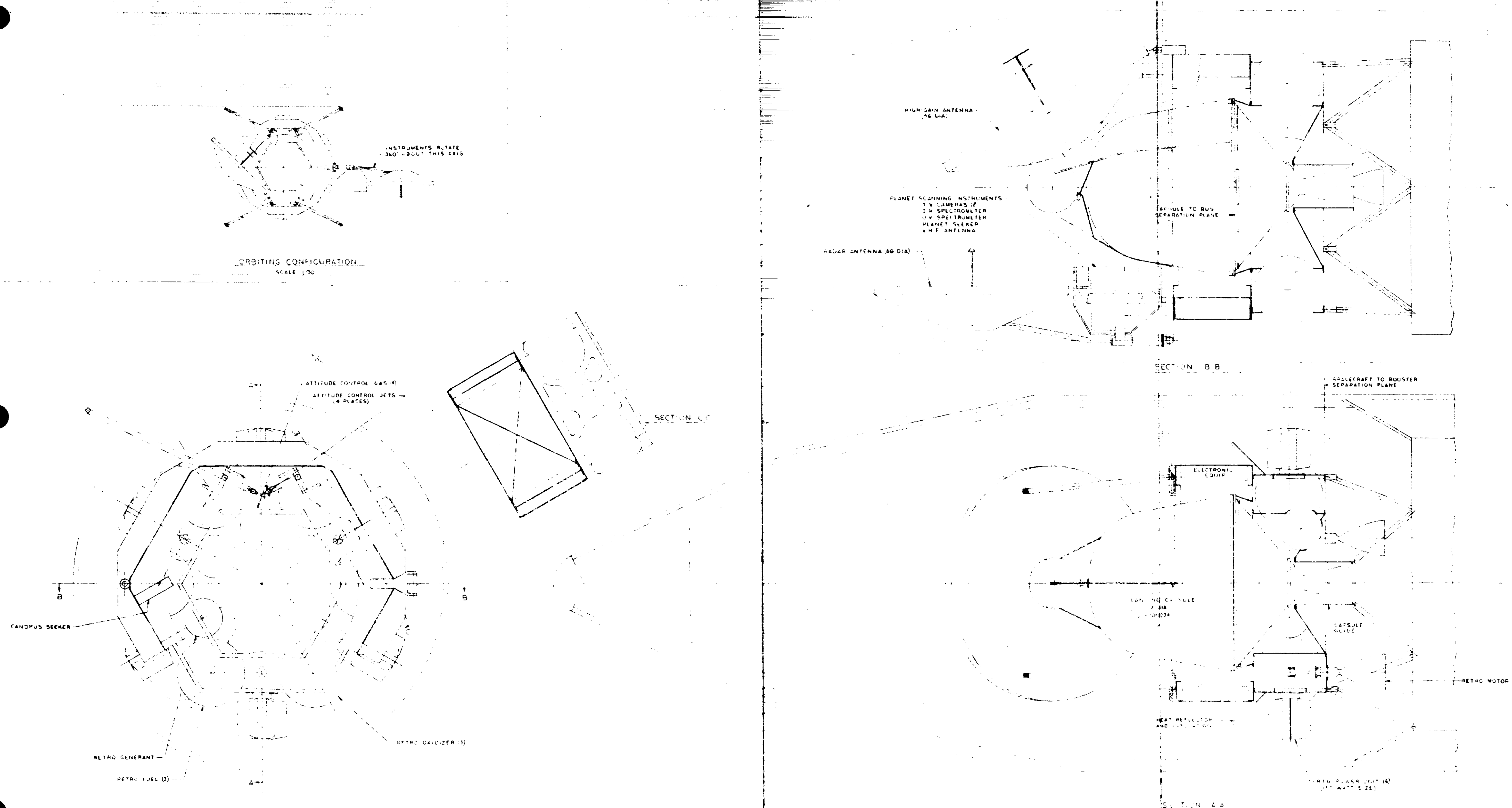


Figure 4-141a. Spacecraft Configuration Number 3

Configuration	Weight	Center of Gravity[1]			Moments of Inertia[2]			Ballistic Coefficients[3]		Mass-to-Area Ratio[4]
	W (lb)	X (in.)	Y (in.)	Z (in.)	I_X (slug ft^2)	I_Y (slug ft^2)	I_Z (slug ft^2)	$\left(\frac{M}{C_D A}\right)_N$ (slug/ft^2)	$\left(\frac{M}{C_D A}\right)_Z$ (slug/ft^2)	$\left(\frac{M}{A}\right)_Z$ (slug/ft^2)
Launch; 100% Fuel	6260	0.31	0.12	468.2	1990	2080	2080			
Cruise With Capsule; 100% Fuel	6260	−0.37	0.12	468.5	1940	2140	2200			0.0105
Cruise Without Capsule; 100% Fuel	5260	−0.43	0.14	477.0	1290	1500	2100			
Cruise Without Capsule; 0% Fuel	2080	−1.10	0.36	464.7	630	835	995			
Orbit; 0% Fuel	2080	0.98	0.36	465.8	575	915	1120	0.29	0.23	

Notes: (1) X-pitch reference axis, Y-yaw reference axis, Z-roll reference axis, Z coordinate is referred to spacecraft separation plane, defined as station 500.

(2) All I's about axes through center of gravity parallel to reference axes.

(3) $\left(\frac{M}{C_D A}\right)_N$ is ballistic coefficient normal to z-axis, $\left(\frac{M}{C_D A}\right)_Z$ colinear with z-axis, where A is the maximum appropriate area and $C_D = 2$ was used for this analysis.

(4) $\left(\frac{M}{A}\right)_Z$ is mass-to-area ratio colinear with z-axis.

Figure 4-141b. Configuration Number 3 Design Summary

field of view about an axis through its center and normal to the spacecraft roll axis; in addition, no generator will be affected by radiation from any other generator.

The multi-engine propulsion unit is also used on this configuration to shorten the spacecraft adapter section.

d. Configuration Number 4 (Figures 4-142a, 4-142b and 4-142c)

This is essentially the same as Configuration Number 3, except that six cylindrical tanks are used, as discussed under Configuration Number 2. In general, the relative merits of this design are similar to those of the preceding configuration.

e. Configuration Number 5 (Figures 4-143a and 4-143b)

A configuration with single-tank multi-engine propulsion system was investigated to alleviate the potential cg shift problem due to the tank-to-tank bladder difference while maintaining a minimal overall length.

Because of the propulsion system configuration, which makes it impractical to incorporate a fixed solar panel, the entire 360 ft^2 of panel area is contained in the erectable panels. It should be noted that a radioisotope generator system could be employed in place of the solar conversion system shown.

The effects of the single propellant tank on the entire system can readily be appreciated by comparing this configuration with the preceding two configurations.

f. Configuration Number 6 (Figure 4-144a and 4-144b)

This configuration employs a novel technique developed to meet the sterilization requirements. The capsule, as mounted on the spacecraft, is enclosed in a sealed canister, or "cocoon", which constitutes the sterile interface between the capsule and the bus. The entire assembly can be subjected to the terminal sterilization procedure and then mated in its sterile condition to the bus at the upper assembly-joint. The entire vehicle can then be mounted on the booster at the lower joint. With this procedure, the capsule is maintained within its sterile envelope without any additional sterilization procedures. The spacecraft-to-booster separation joint is located between the upper and lower joints on the canister. When this joint is released, the spacecraft is separated from the booster and the capsule sterilization envelope is opened. This technique avoids performing an extra function in separating the capsule from the bus.

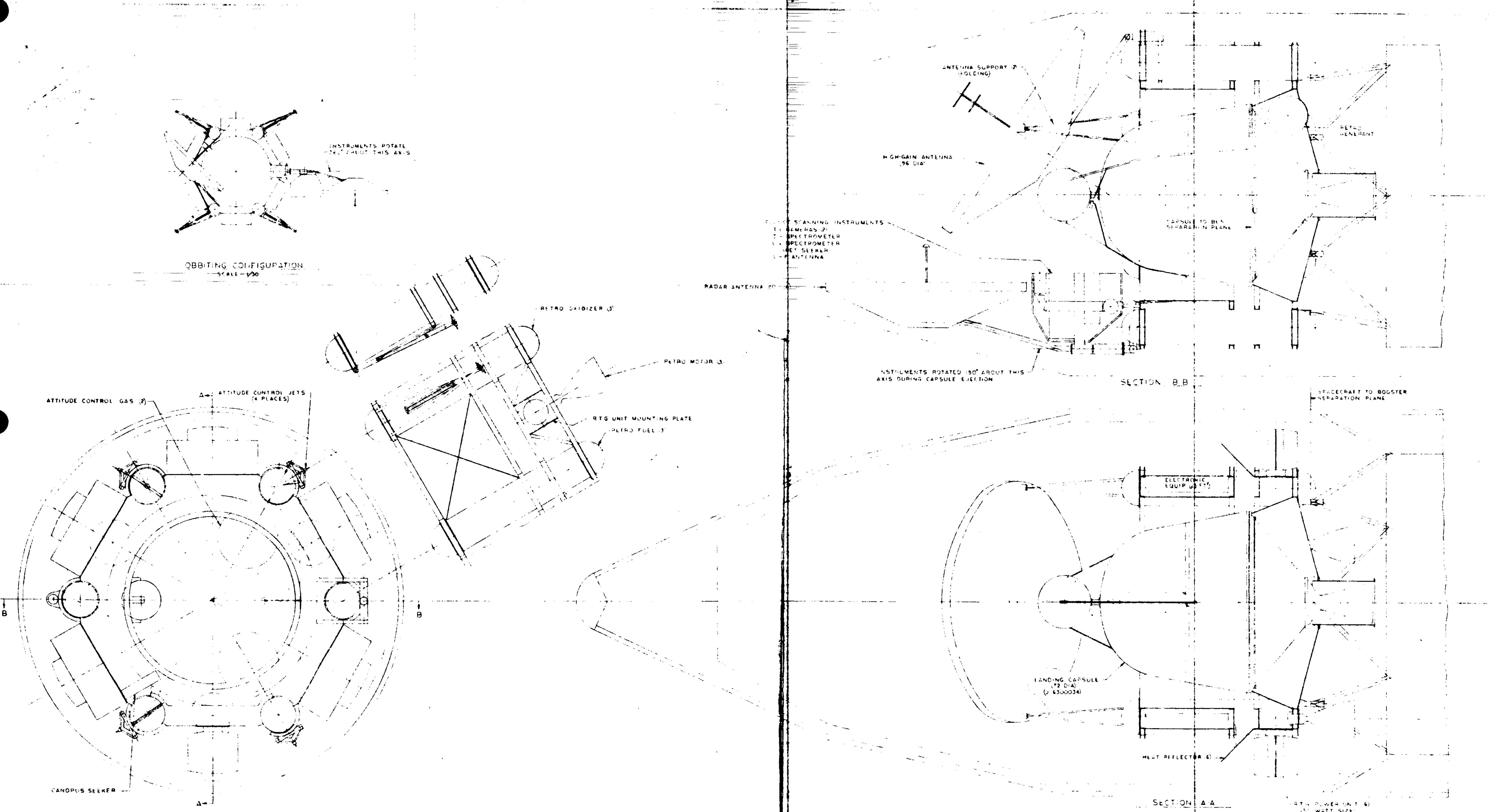


Figure 4-142a. Spacecraft Configuration Number 4

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Configuration	Weight	Center of Gravity(1)			Moments of Inertia(2)			Ballistic Coefficients(3)		Mass-to Area Ratio(4)
	W (lb)	X (in.)	Y (in.)	Z (in.)	I_X (slug ft^2)	I_Y (slug ft^2)	I_Z (slug ft^2)	$\left(\frac{M}{C_D A}\right)_N$ (slug/ft^2)	$\left(\frac{M}{C_D A}\right)_Z$ (slug/ft^2)	$\left(\frac{M}{A}\right)_Z$ (slug/ft^2)
Launch, 100% Fuel	6340	0.32	0.16	458.2	2440	2520	3140			
Cruise With Capsule, 100% Fuel	6340	-0.46	0.16	458.5	2400	2640	3280			0.0100
Cruise Without Capsule, 100% Fuel	5340	-0.54	0.20	462.5	2150	2380	3180			
Cruise Without Capsule, 0% Fuel	2160	-1.34	0.38	461.7	860	1090	1290			
Orbit, 0% Fuel	2160	0.72	0.48	462.0	835	1200	1430	0.29	0.23	

Notes: (1) X-pitch reference axis, Y yaw reference axis, Z roll reference axis, Z coordinate is referred to spacecraft separation plane, defined as station 500

(2) All I's about axes through center of gravity parallel to reference axes.

(3) $\left(\frac{M}{C_D A}\right)_N$ is ballistic coefficient normal to z-axis, $\left(\frac{M}{C_D A}\right)_Z$ colinear with z-axis, where A is the maximum appropriate area and C_D 2 was used for this analysis.

(4) $\left(\frac{M}{A}\right)_Z$ is mass-to area ratio colinear with z-axis

Figure 4-142b. Configuration Number 4 Design Summary

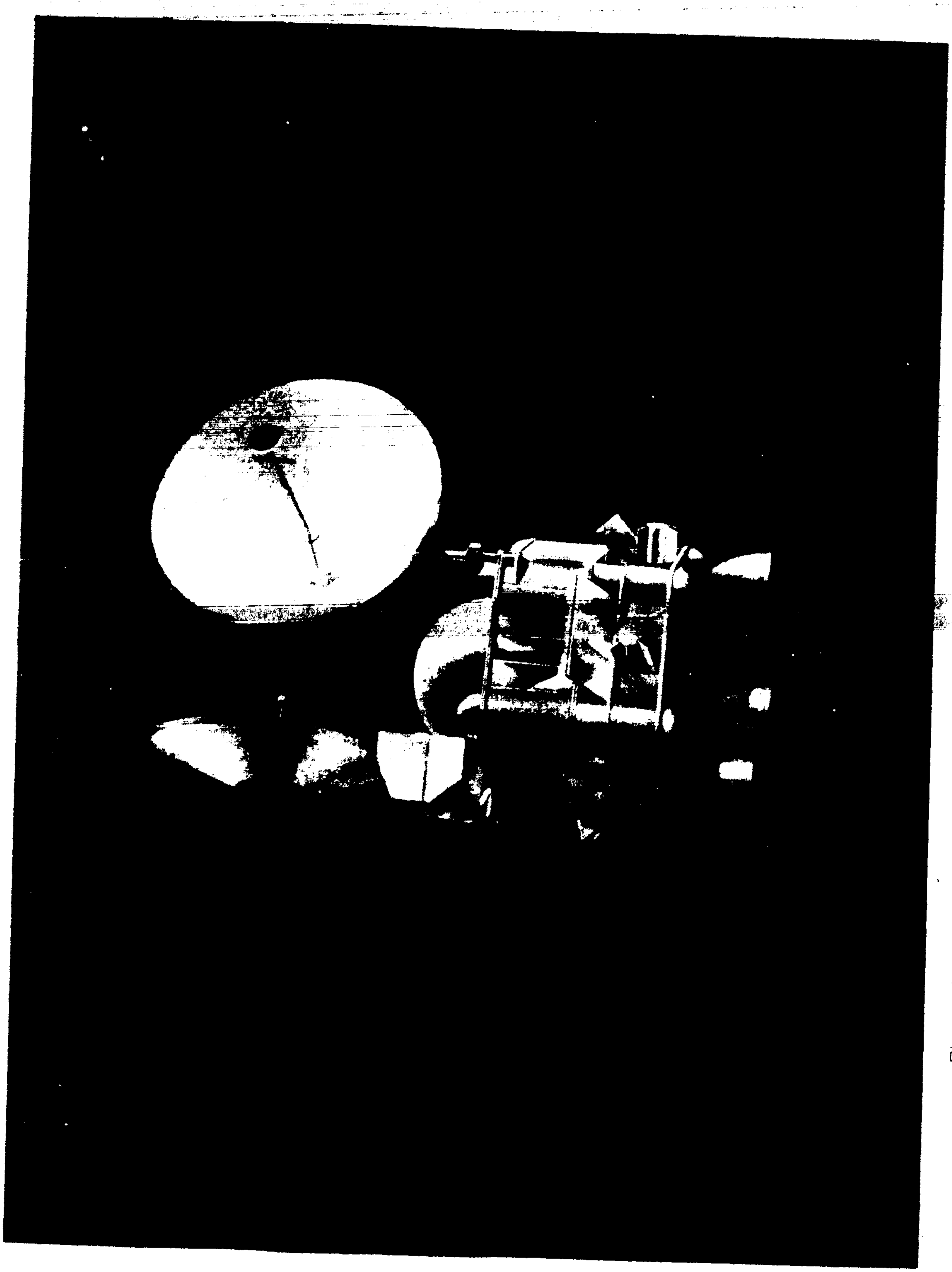

Figure 4-142c. Artist's Rendering of Spacecraft Configuration Number 4

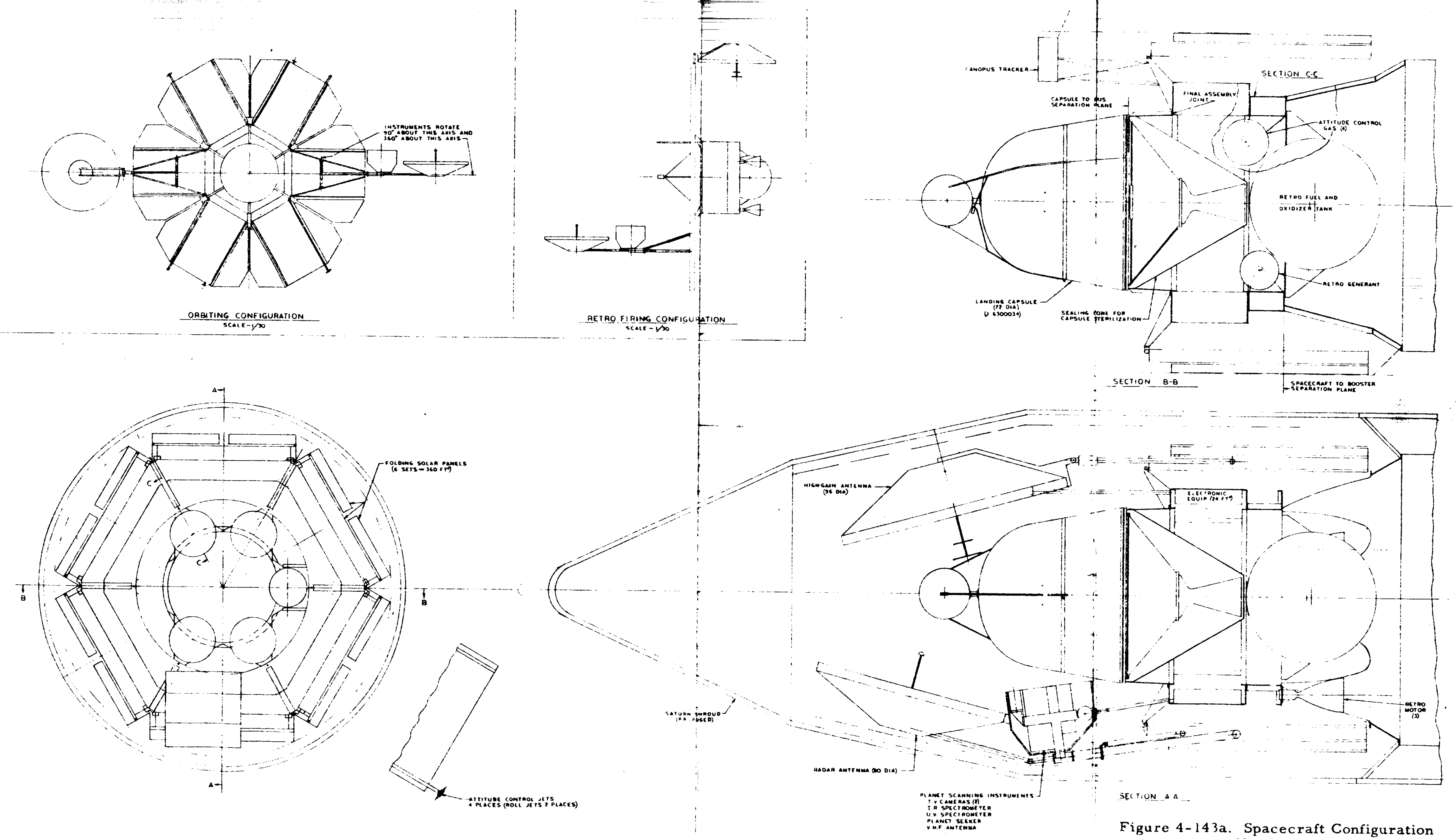

Figure 4-143a. Spacecraft Configuration Number 5

Configuration	Weight	Center of Gravity[1]			Moments of Inertia[2]			Ballistic Coefficients[3]		Mass-to-Area Ratio[4]
	W (lb)	X (in.)	Y (in.)	Z (in.)	I_X (sluf ft^2)	I_Y (slug ft^2)	I_Z (slug ft^2)	$\left(\frac{M}{C_DA}\right)_N$ (slug/ft^2)	$\left(\frac{M}{C_DA}\right)_Z$ (slug/ft^2)	$\left(\frac{M}{A}\right)_Z$ (slug/ft^2)
Launch; 100% Fuel	6560	−0.32	0.43	480.0	3740	3620	1490			
Cruise With Capsule; 100% Fuel	6560	−0.32	−0.75	474.7	5110	4330	3410			0.0028
Cruise Without Capsule; 100% Fuel	5560	−0.38	−0.88	488.9	3400	2620	3310			
Cruise Without Capsule; 0% Fuel	2380	−0.89	−2.06	458.0	2540	1750	3310			
Orbit; 0% Fuel	2380	−0.89	2.64	461.3	2920	1540	3920	0.33	0.072	

Notes: (1) X-pitch reference axis, Y-yaw reference axis, Z-roll reference axis, Z coordinate is referred to spacecraft separation plane, defined as station 500.

(2) All I's about axes through center of gravity parallel to reference axes.

(3) $\left(\frac{M}{C_DA}\right)_N$ is ballistic coefficient normal to z-axis, $\left(\frac{M}{C_DA}\right)_Z$ colinear with z-axis, where A is the maximum appropriate area and $C_D = 2$ was used for this analysis.

(4) $\left(\frac{M}{A}\right)_Z$ is mass-to-area ratio colinear with z-axis.

Figure 4-143b. Configuration Number 5 Design Summary

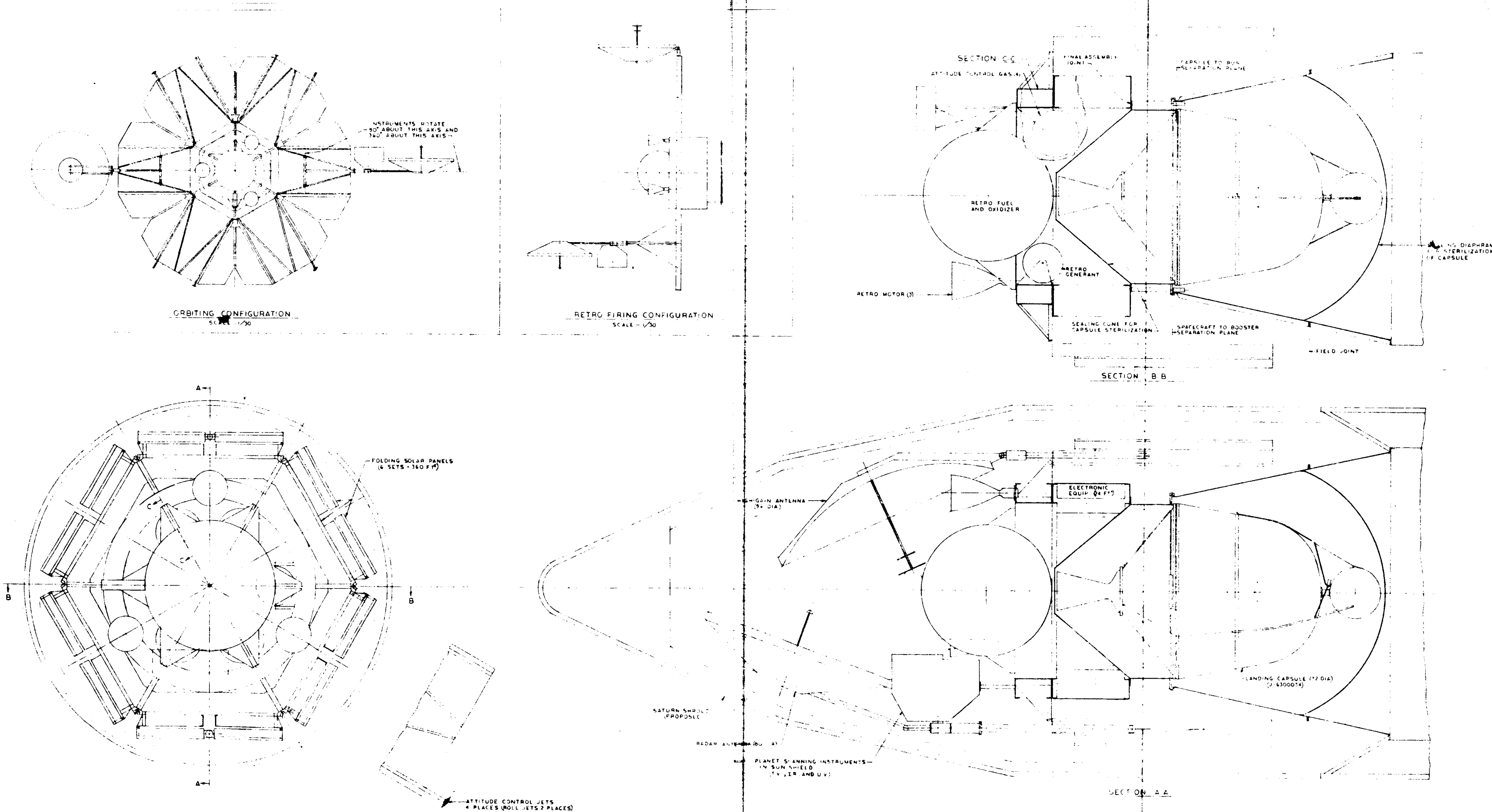

Figure 4-144a. Spacecraft Configuration Number 6

Configuration	Weight	Center of Gravity[1]			Moments of Inertia[2]			Ballistic Coefficients[3]		Mass-to-Area Ratio[4]
	W	X	Y	Z	I_X	I_Y	I_Z	$\left(\frac{M}{C_D A}\right)_N$	$\left(\frac{M}{C_D A}\right)_Z$	$\left(\frac{M}{A}\right)_Z$
	(lb)	(in.)	(in.)	(in.)	(slug ft^2)	(slug ft^2)	(slug ft^2)	(slug/ft^2)	(slug/ft^2)	(slug/ft^2)
Launch; 100% Fuel	6560	−0.12	0.32	456.7	3850	3760	1520			
Cruise With Capsule; 100% Fuel	6560	−0.12	−0.75	450.8	4940	4060	3460			0.0028
Cruise Without Capsule; 100% Fuel	5560	−0.14	−0.89	434.8	2660	1890	3350			
Cruise Without Capsule; 0% Fuel	2380	−0.33	−2.07	450.5	2440	1670	3350			
Orbit; 0% Fuel	2380	−0.33	2.92	454.2	2860	1460	3980	0.31	0.072	

Notes: (1) X-pitch reference axis, Y-yaw reference axis, Z-roll reference axis, Z coordinate is referred to spacecraft separation plane, defined as station 500.

(2) All I's about axes through center of gravity parallel to reference axes.

(3) $\left(\frac{M}{C_D A}\right)_N$ is ballistic coefficient normal to z-axis, $\left(\frac{M}{C_D A}\right)_Z$ colinear with z-axis, where A is the maximum appropriate area and $C_D = 2$ was used for this analysis.

(4) $\left(\frac{M}{A}\right)_Z$ is mass-to-area ratio colinear with z-axis.

Figure 4-144b. Configuration Number 6 Design Summary

Although this sterilization canister has several operational advantages, there are also several drawbacks of such a system. In using the technique it is necessary to have the capsule under the spacecraft, and have the propellant rather far forward. This is a poor structural arrangement since the propellant weight is about 60 percent of the total injected weight. Also the spacecraft is over-constrained by the requirements of such a capsule configuration.

g. Configuration Number 7 (Figures 4-145a and 4-145b)

As discussed in Section IV-13-(F), power system weight for a given total power capability decreases with a decreasing number of radioisotope power generators. This configuration takes advantage of this fact by using three instead of six power units. The three generators are mounted on a structure at the aft end of the spacecraft. The multi-engine propulsion unit also mounts on this same structure, and a thermal and nuclear radiation shield is again provided between the electronics and the generators. The generators have their major axes tilted aft so that they may have complete 360 degree radiation.

The remaining aspects of this configuration are similar to those of Configuration Number 4.

3. Required Spacecraft Propulsion Throttling Capability

The geometrical properties of the previous seven configurations require varying propulsion control moments during maneuvers. These requirements are summarized in Figure 4-146.

Configuration	Weight	Center of Gravity[1]			Moments of Inertia[2]			Ballistic Coefficients[3]		Mass-to-Area Ratio[4]
	W (lb)	X (in.)	Y (in.)	Z (in.)	I_X (slug ft^2)	I_Y (slug ft^2)	I_Z (slug ft^2)	$\left(\frac{M}{C_DA}\right)_N$ (slug/ft^2)	$\left(\frac{M}{C_DA}\right)_Z$ (slug/ft^2)	$\left(\frac{M}{A}\right)_Z$ (slug/ft^2)
Launch; 100% Fuel	6130	0.57	0.26	459.7	2400	2480	3020			
Cruise With Capsule; 100% Fuel	6130	−0.29	0.26	460.1	2340	2580	3180			0.0118
Cruise Without Capsule; 100% Fuel	5130	−0.34	0.31	464.8	2050	2300	3080			
Cruise Without Capsule; 0% Fuel	1950	−0.90	0.82	462.7	815	1055	1180			
Orbit; 0% Fuel	1950	0.91	0.82	463.1	786	1140	1290	0.22	0.25	

Notes: (1) X-pitch reference axis, Y-yaw reference axis, Z-roll reference axis, Z coordinate is referred to spacecraft separation plane, defined as station 500.

(2) All I's about axes through center of gravity parallel to reference axes.

(3) $\left(\frac{M}{C_DA}\right)_N$ is ballistic coefficient normal to z-axis, $\left(\frac{M}{C_DA}\right)_Z$ colinear with z-axis, where A is the maximum appropriate area and $C_D = 2$ was used for this analysis.

(4) $\left(\frac{M}{A}\right)_Z$ is mass-to-area ratio colinear with z-axis.

Figure 4-145b. Configuration Number 7 Design Summary

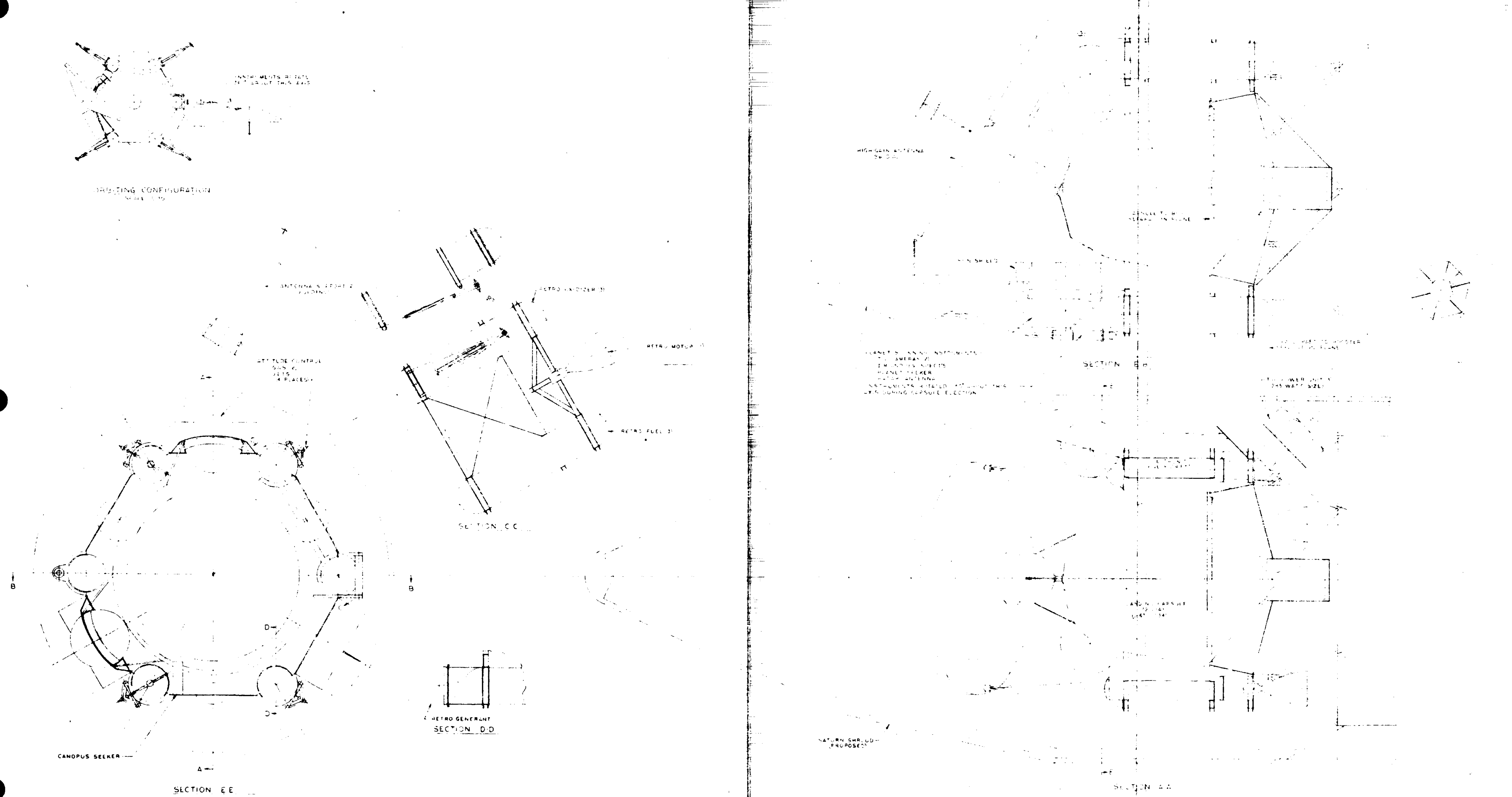

Figure 4-145a. Spacecraft Configuration Number 7

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① PERMISSIBLE RADIUS FROM S/C CG TO GEOMETRICAL CENTER OF THREE-ENGINE SYSTEM IN x-y PLANE.

② $\bar{r}_{max}/R_{retro}$. THE DIMENSIONLESS RADIUS FOR THROTTLING CAPABILITIES OF O TO 20% ("NEAR-HEXAGONAL" CURVES).

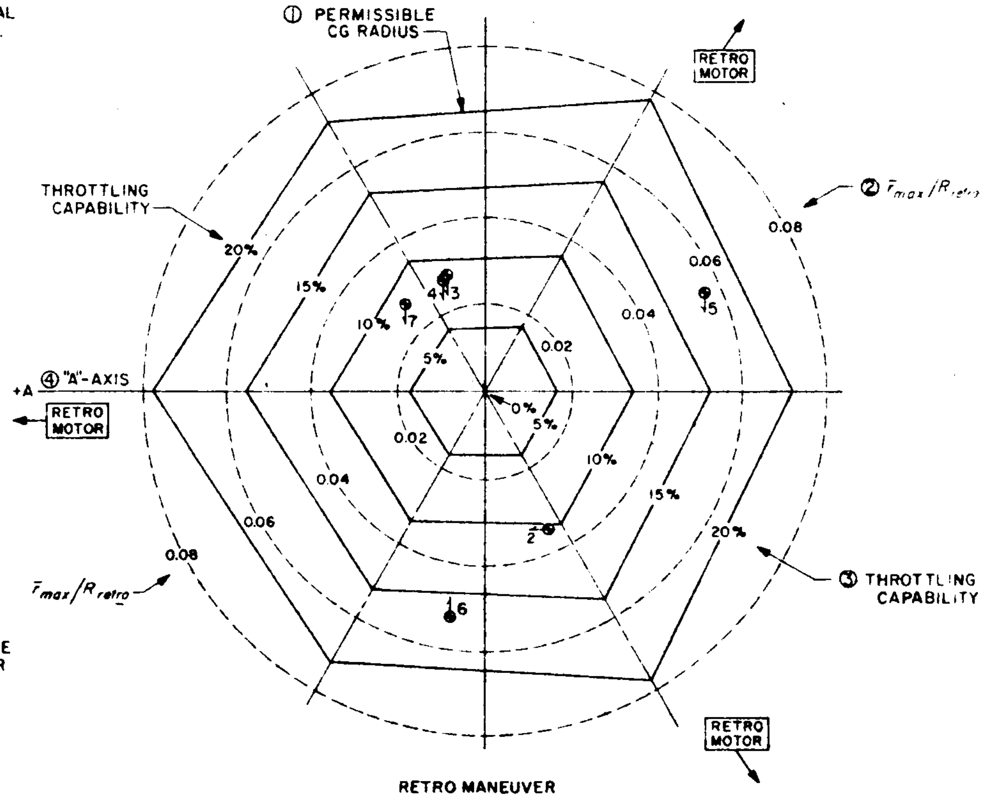
③ RESULTANT MINIMUM REQUIRED THROTTLE CAPABILITY FOR EACH DESIGN, i.e., THE THROTTLING REQUIRED FOR THE THRUST-VECTOR RESULTANT TO PASS THROUGH THE CG (WITH NO MARGIN ALLOWED FOR CONTROL), IS TABULATED BELOW.

④ A-AXIS IS AXIS, EITHER PITCH OR YAW, PASSING THROUGH A RETRO MOTOR.

CONFIGURATION	1	2	3	4	5	6	7
MAX CG RADIUS, $\bar{r}_{max}$ (in.)	*	1.68	1.11	1.42	2.24	2.10	1.22
RADIUS TO RETRO ℄, R_{retro} (in.)	*	48	40	52	40	40	45
MINIMUM THROTTLING CAPABILITY	*	11%	9%	9%	18%	17%	9%
+A EQUALS	*	+X	+Y	+Y	+Y	-Y	+Y

* CORRESPONDING MINIMUM GIMBAL HALF-CONE ANGLE FOR THE SINGLE-ENGINE SYSTEM OF DESIGN 1 IS 4.6 deg.

NOTES:
1. CALCULATED MAXIMUM CG RADII DURING MIDCOURSE AND RETRO MANEUVERS ARE BASED ON COMPUTER RUNS COMPLETED 15 JULY 1963.
2. ARROW ON CG SYMBOLS POINTS IN POSITIVE X-DIRECTION.



RETRO MANEUVER

① PERMISSIBLE RADIUS FROM S/C CG TO GEOMETRICAL CENTER OF THREE-ENGINE SYSTEM IN x-y PLANE.

② $\bar{r}_{max}/R_{retro}$. THE DIMENSIONLESS RADIUS FOR THROTTLING CAPABILITIES OF O TO 20% ("NEAR-HEXAGONAL" CURVES).

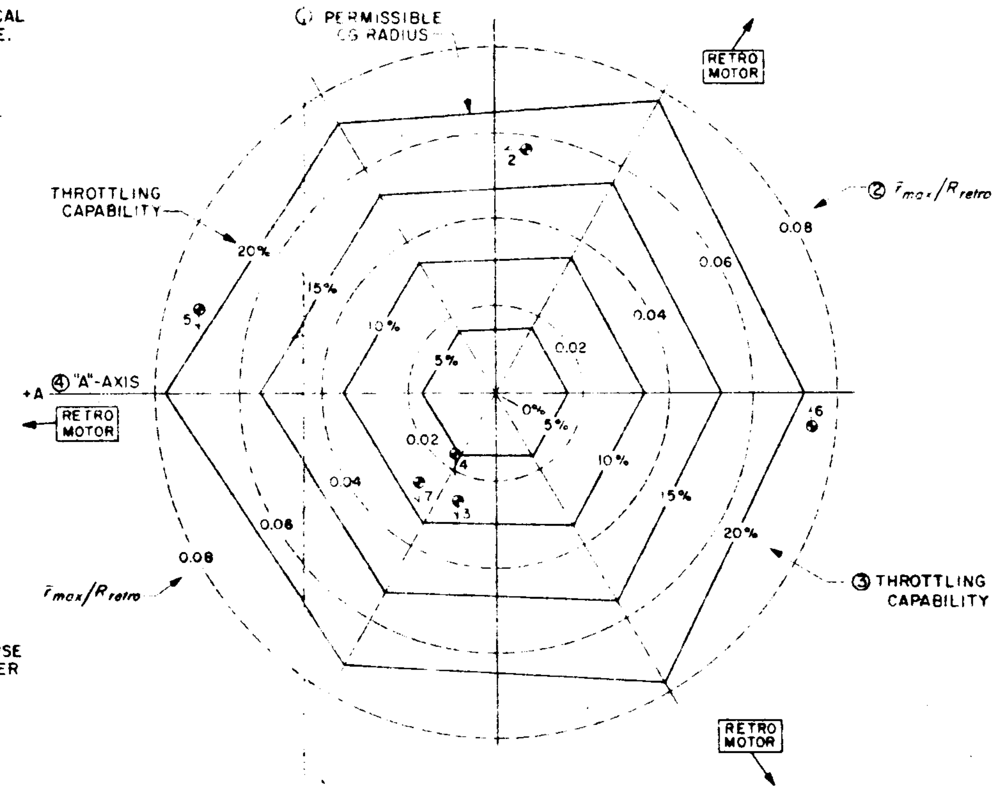
③ RESULTANT MINIMUM REQUIRED THROTTLE CAPABILITY FOR EACH DESIGN, i.e., THE THROTTLING REQUIRED FOR THE THRUST-VECTOR RESULTANT TO PASS THROUGH THE CG (WITH NO MARGIN ALLOWED FOR CONTROL), IS TABULATED BELOW.

④ A-AXIS IS AXIS, EITHER PITCH OR YAW, PASSING THROUGH A RETRO MOTOR.

CONFIGURATION	1	2	3	4	5	6	7
MAX CG RADIUS, $\bar{r}_{max}$ (in.)	*	2.76	1.04	0.86	2.78	2.94	1.22
RADIUS TO RETRO ℄, R_{retro} (in.)	*	48	40	52	40	40	45
MINIMUM THROTTLING CAPABILITY	*	18%	9%	5%	21%	21%	9%
+A EQUALS	*	+X	+Y	+Y	+Y	-Y	+Y

* CORRESPONDING MINIMUM GIMBAL HALF-CONE ANGLE FOR THE SINGLE-ENGINE SYSTEM OF DESIGN 1 IS 3.7 deg

NOTES:
1. CALCULATED MAXIMUM CG RADII DURING MIDCOURSE AND RETRO MANEUVERS ARE BASED ON COMPUTER RUNS COMPLETED 15 JULY 1963
2. ARROW ON CG SYMBOLS POINTS IN POSITIVE X-DIRECTION



MIDCOURSE MANEUVER

Figure 4-146. Voyager Spacecraft Minimum Required Throttling Capability (Midcourse) and Retro Maneuvers)

L. CAPSULE

In studying the capsule configuration it was necessary to assume a capsule aerodynamic shape around which the capsule internal design could evolve. For the purpose of this configuration study the Discoverer shape was used as a typical capsule shape.

Several typical configurations have been developed to indicate some of the major subsystem interfaces. One of the most significant constraints is the communication antenna system. The capsule communication system operates at both S-band and VHF, the S-band is for the direct capsule-to-Earth link and the VHF is for the capsule-to-orbiter relay link. In order to avoid complicated and unreliable antenna switching, it is desirable to use one antenna at each frequency for all phases of the capsule mission. In the VHF system this is accomplished by mounting an antenna on the aft end of the capsule with a 60 deg half-cone angle. With this antenna pattern it is necessary to maintain a capsule-bus relationship so that the bus is always within 60 deg of the capsule roll axis after separation and through landing. Also, because of these capsule antenna limitations, it is necessary to have a preferred orientation on the surface after landing. This subject is discussed more thoroughly in the post-landing section.

It is desirable that the S-band communication link be established in the event of a capsule erection system failure. This can be accomplished by using four S-band antennas located at the apexes of an imaginary tetrahedron.

The capsule power systems investigated were a battery power system and a radioisotope thermoelectric generator power system. The reasons for considering only these two types of power systems are discussed in more detail in Section II (e). The capsule power system interface is relatively simple for the case where a battery is used as the primary power source, whereas, when a radioisotope thermoelectric generator system is used, the interface is a much more complex [illegible]. [illegible] significant [illegible] of the radioisotope thermoelectric generator system is the [illegible] thermal radiation problem. The radioisotope power systems [illegible] large amounts of thermal energy. This energy must be removed [illegible] capsule [illegible] designed for a ballistic entry into the planet atmosphere and, therefore, must withstand [illegible] heating levels. Several solutions to this problem have been investigated and are discussed in more detail in the sections on power [illegible]. [illegible] of the more promising solutions has the radiator surface [illegible] external to the [illegible] heat shield. The [illegible] are located on the capsule in such a position that the integrated heat load during entry is low enough to allow the use of a [illegible] heat sink to protect the thermoelectric elements of the generator.

1. Separation

Several capsule-bus separation systems have been studied and one has been selected for further study. The reasoning behind this particular choice and some of the major features of this system are presented here. The advantages and disadvantages of some of the other options available are also discussed but in much less detail.

In order to properly make a choice of a primary system, it is first necessary to consider all reasonable existing possibilities. The final choice will be heavily influenced by trade-off considerations which include sterilization, accuracy requirements, reliability degradation to the capsule or spacecraft, and state-of-the-art development. The intent here is to outline the reasonable choices and the requirements associated with choosing each particular system so that the information required to make the proper and timely choice is available.

To date, three types of capsule-bus separation maneuvers have been considered. These are:

1) Early capsule-bus separation
2) Bus miss maneuver
3) Propulsion on the capsule

"Early capsule-bus separation" is accomplished by simply ejecting the capsule from the bus with a few m/sec separation velocity at a time in the flight when the trajectory is most velocity sensitive. Such a separation might be performed around 60 to 100 days after injection for typical Mars trajectories. The relatively small velocity increment which is required may be provided by a separation spring.

The bus miss-maneuver is a technique in which the capsule and bus are initially on an impact trajectory. Some few hundred thousand kilometers before impact, the capsule is separated and a subsequent bus propulsion maneuver is made to cause the bus to miss the planet.

In the third type of maneuver, which is propulsion on the capsule, the capsule and bus are on a flyby trajectory. A few hundred thousand to a million kilometers before perifocal passage the capsule is separated from the bus and a capsule propulsion maneuver is made, putting the capsule on an impact trajectory.

The first type of maneuver is the simplest, but because of its inherent inaccuracy is not considered in this study. In order to use the second technique, the probability of properly performing the miss-maneuver with an unsterilized bus must satisfy the contamination requirements. This requirement to date, cannot be done. Thus, the third type of maneuver is the only one which will be considered for the remainder of this discussion.

Within the technique of the propulsion on the capsule, there are three basic forms of separation mechanization to choose from:

1) Spin, on the bus (Figure 4-147)
2) Spin, off the bus (Figure 4-148)
3) Non-spinning (Figure 4-149)

In all of these cases specific contraints are assumed. Some form of capsule attitude control is required for the propulsion phase. The separation must be such that after separation the capsule and bus will function properly; this implies that the capsule propulsion maneuver must have no deterrent effect on the bus. The separation system is independent of the flight time at which the separation is performed. Capability must be provided for capsule-bus communication during capsule motor firing and prior to entry. For envisioned capsule antenna patterns, this requires that the capsule-bus line be within 60 deg of the capsule roll axis. Another requirement is that the capsule itself should not require any active attitude control immediately prior to or during the aerodynamic entry phase of the flight, and that the capsule should be capable of entry at any flight path angle.

Figures 4-147, 4-148 and 4-149 include all the reasonable methods to be considered within each of the above systems. By moving down and laterally along the lines of a particular chart, one can follow any one of the many functional block diagrams available for the separation-to-entry capsule sequence. The intent here is to identify the major problems associated with each block diagram. Then the large number of conceptual methods can be reduced to several realistic methods.

a. Spin, on the Bus

In order to spin-stabilize the capsule on the bus before the capsule-bus separation is accomplished, it is necessary to understand the effects of several basic phenomena. These phenomena occur whether the capsule itself is spinning or a flywheel or momentum wheel on the capsule is used.

If the stabilized bus is used as a reference body to torque against, the torque transmitted to the bus must not exceed the torque capability of the spacecraft attitude control system. The torque imparted depends primarily on the spinning moment-of-inertia, the spin-up mechanism and the desired angular acceleration. The low torque-resisting capability of the spacecraft leads to low spin-accelerations; thus, long time periods are required to achieve the desired spin rates. If electrical means are used for the spin-up, the available-power limitation may

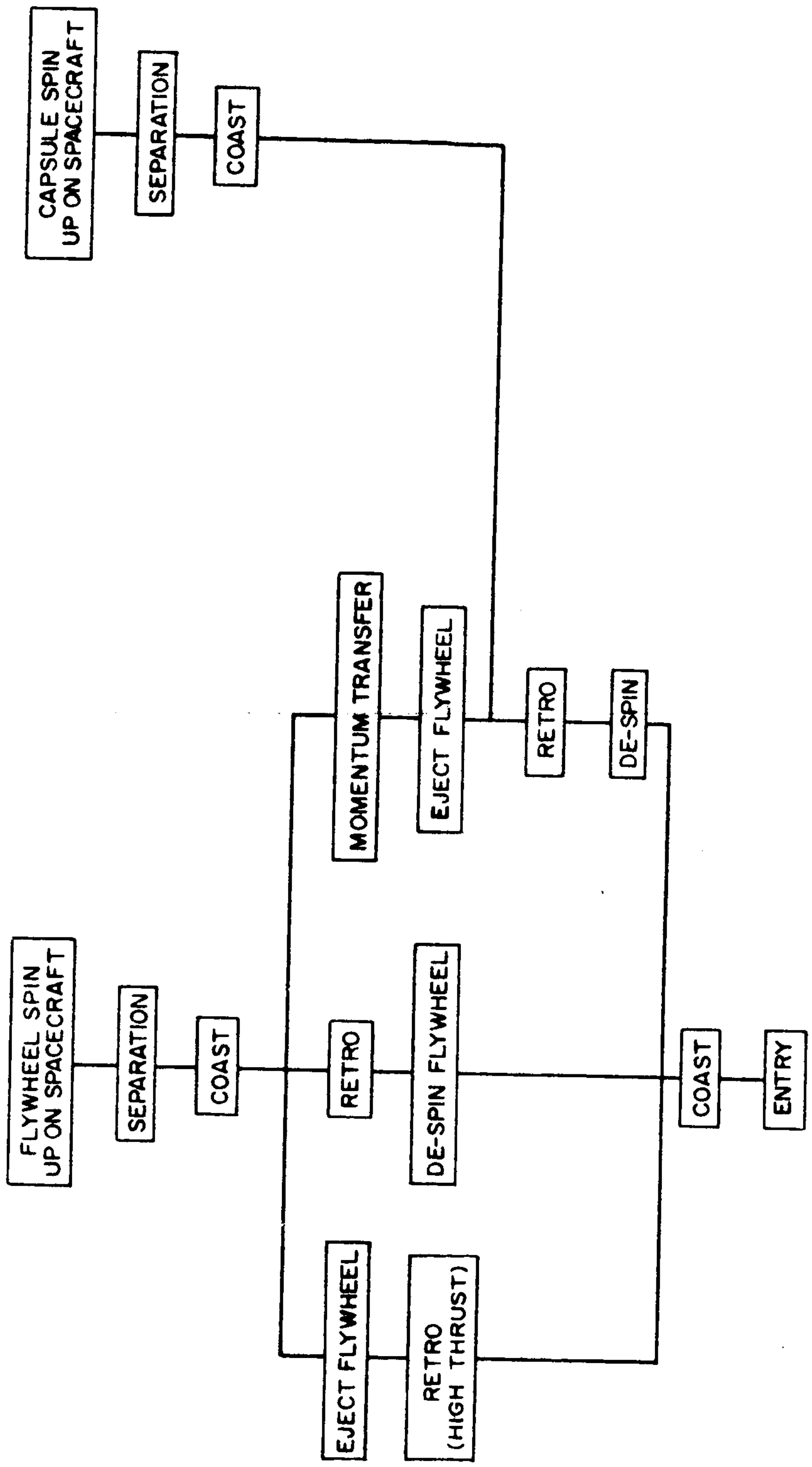


Figure 4-147. Spin Stabilized, on Bus

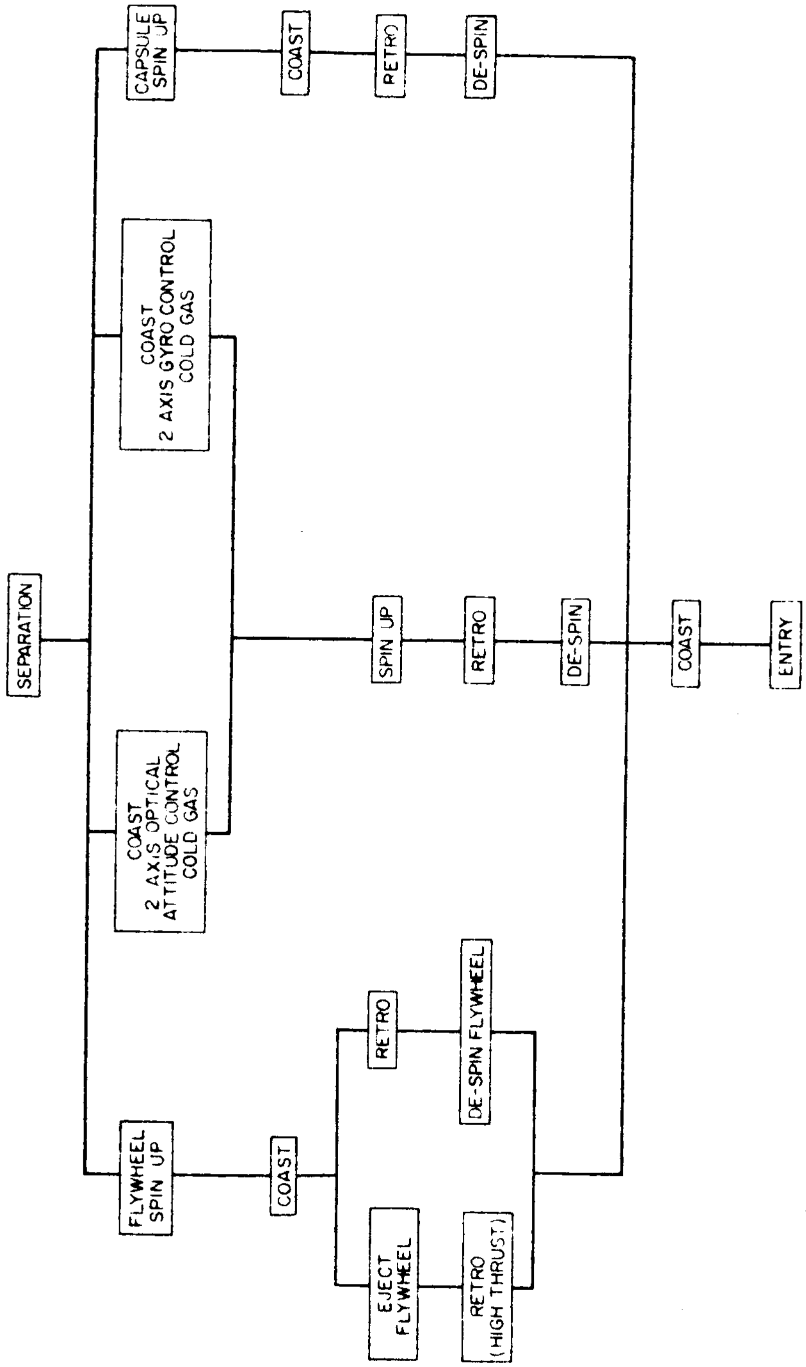


Figure 4-148 Spin Stabilized, off Bus

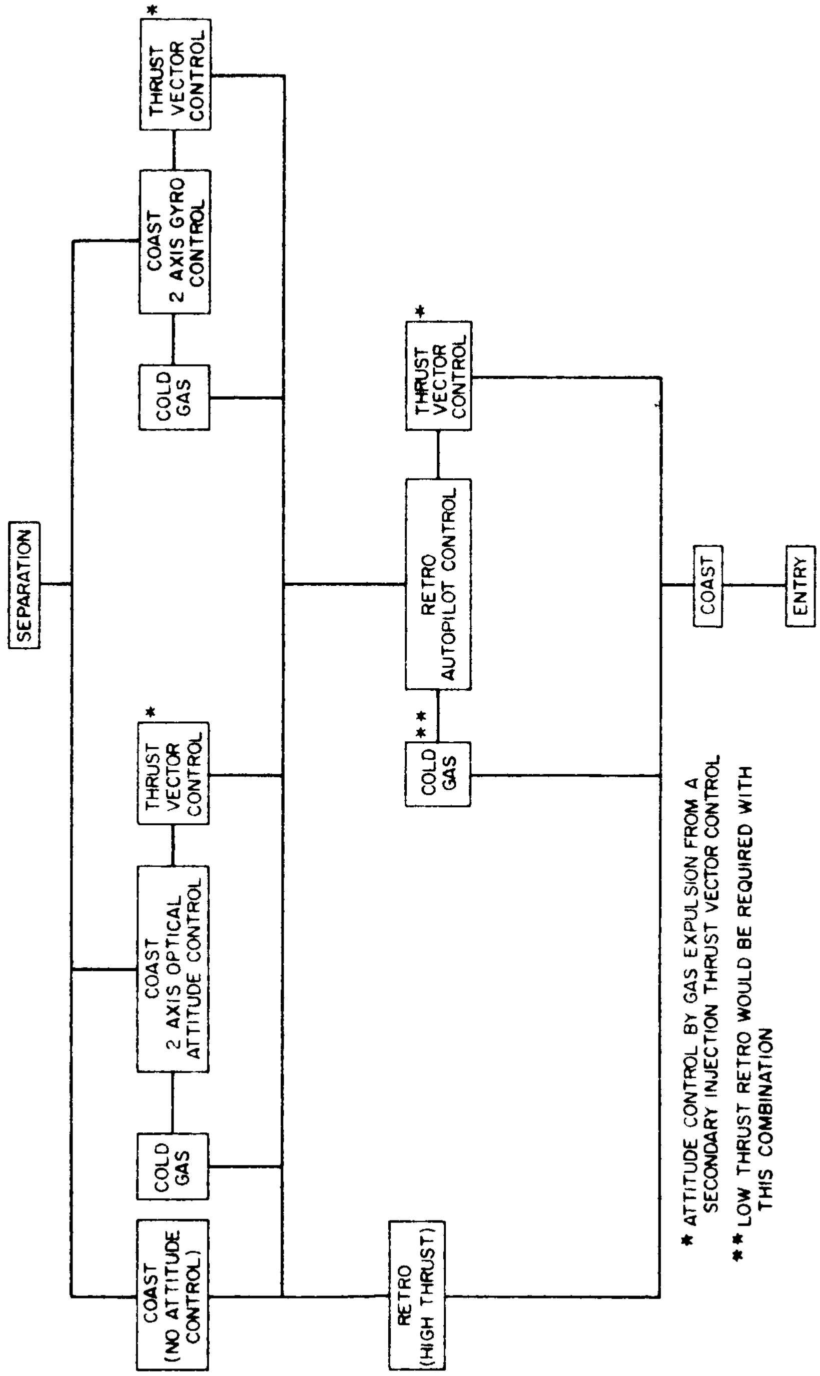


Figure 4-149. Non-Spin Stabilized Techniques

force use of spin-up times even longer than those associated with the torque limitation. Since the spacecraft may have to remain off its primary reference axis for these lengthy periods, problems in the power system (for the solar-powered systems) and the thermal control system may result. Also, increased pointing errors would have to be accommodated. Conversely, a slow spin-up with the spacecraft stabilized to its primary references, followed by an attitude maneuver to capsule-release orientation, would be unattractive because of gyroscopic interactions during the maneuver.

A relatively rapid spin-up could be accomplished by using gas jets or spin rockets on the capsule, provided exhaust particle impingement on the bus and subsequent optical surface contamination can be avoided. In this case, only the bearing friction would have to be overcome by the attitude control jets. The gas-jet system is of interest, but a solution to the bearing problem would be required. It may be possible to eliminate the bearing, thus achieving a "spin-off bus" system. Such a system will be discussed in the following paragraph.

b. Spin, Off the Bus

Spin stabilizing the capsule after it is separated from the spacecraft has a singular advantage in that the effects and requirements on the spacecraft are kept to a minimum. The capsule may be either actively stabilized or non-stabilized prior to spin-up. Within the former class there are two basic methods. In the first method the capsule spin axis is actively controlled by using the spacecraft as a unique (IR or optical) source. The rate and position measurements necessary to keep the capsule roll axis colinear with the capsule-bus line are derived from sensors on the capsule. The second method inertially stabilizes the capsule using axis-fixed gyros on the capsule. Each of these methods requires a two-axis attitude control system. In either case, the capsule would remain stabilized in this manner until spin-up. It would then be gyroscopically stabilized until and during the retro-maneuver. Prior to atmospheric entry it would probably be de-spun.

In the case where the capsule is spin-stabilized off the bus and no active attitude control system is used on the capsule, the final pointing error is very sensitive to the capsule-bus separation disturbance and the time between separation and spin-up. Although the capability of using a flywheel does exist, only limited mechanization advantages over spinning the entire capsule are to be realized.

The post-spin-up sequence is essentially the same with or without the active attitude control system. The drawback of these off-bus spin-up methods is the active attitude control system on the capsule. The complexity of using inertial or optical references must be evaluated against the reduced accuracy of a system which has no active attitude control. This type of reliability-performance trade-off must be handled with extreme prudence and judgment.

c. Non-Spinning

The propulsion maneuver can be made without spinning by using either an autopilot on the capsule or a high-thrust technique. The high-thrust technique is discussed in more detail in the propulsion part of this section. Prior to the propulsion maneuver, the capsule is stabilized either optically or inertially as in the previous spinning case. Available TVC (thrust-vector control) techniques during the propulsion maneuver include secondary gas-injection, jet-vane exhaust deflection, use of the cold-gas attitude control system, or gimballing the propulsion unit. Using the cold-gas attitude control system requires a low-thrust long-burn motor. If the TVC is provided by the use of secondary gas-injection, then this secondary injection system can also be used for the capsule attitude control during the coast phase. This option is shown in Figure 4-149.

The choice of separation schemes is strongly influenced by the decision of whether the capsule propulsion system is to provide only the lateral impulse for establishing an impact trajectory for the capsule or is also to provide the required capsule-bus arrival time separation. In the latter case, the capsule impact error is very sensitive to angular errors in the capsule velocity-increment vector. For example, if a separation velocity of 200 m/sec is required at 60 hr prior to encounter, and if it is desired to impact the planet in a target area of 200 km radius, the required angular accuracy in the velocity increment vector is approximately 0. 3 deg. The accuracy of any of the spinning systems is likely to be well outside such a limit; analyses are underway which should confirm this estimate. The non-spinning, high-thrust system will also probably be eliminated by such restrictive angular accuracy requirements. When the capsule-bus separation velocity is provided by the capsule propulsion system, meeting the pointing accuracy requirements is further complicated by the need for a long capsule coast after separation to prevent interaction of the propulsion exhaust plume with the bus. The non-spinning, attitude-stabilized, low-thrust system is the only one which shows promise of providing the desired accuracy. The ability of either the optically or inertially stabilized system to maintain the attitude of the capsule for the duration of the post-separation coast is under investigation.

When the capsule propulsion provides only the lateral maneuver, with possibly some forward component for communications antenna-coverage purposes, the sensitivity of impact error to angular errors in the capsule velocity vector is reduced to a point where any of the suggested separation systems are possible. These systems have been discussed in detail in the Spacecraft Control Section, II (E).

Both solid propellant and liquid propellant propulsion systems have been considered for the capsule. Each system is capable of being heat sterilized and used with a spin-stabilized capsule or with an autopilot using some form of TVC. The simplest and most reliable system is the spin-stabilized solid-propellant system, which avoids use of a complicated TVC systems.

Use of a high-thrust solid-propulsion system has been considered. The high-thrust propulsion maneuver obtains the total required impulse by using an extremely short burn-time and high thrust-level. The velocity increment can be achieved without actively stabilizing the capsule because the large pitch and yaw moments-of-inertia of the capsule and the relatively small thrust-misalignment do not permit the velocity vector to be rotated significantly during the short burning time. Thus, it is theoretically possible to perform the propulsion maneuver using such a technique. But, in practice, there are several other parameters which must be satisfied before this method can be adopted.

In order to use such a method, the capsule roll axis (thrust axis) must be inertially oriented in the proper direction at the time of motor ignition. This orientation is dependent both on the accuracy to which the spacecraft can be pointed and on the mechanics of the separation system and the subsequent attitude of the capsule. If the capsule attitude is not actively controlled, the small angular rates due to separation are integrated over the time between separation and retro. This time is set by the minimum safe distance requirement between the bus and capsule at the time of motor firing, along with the bus-capsule separation velocity. The capsule axis may be actively stabilized to give a better-controlled attitude at the time of firing. Attitude control after the propulsive maneuver may be required if the capsule-bus communication must be performed during the period between separation and entry. Also, for a non-attitude-stabilized capsule, the capsule aerodynamic configuration must be such that it will achieve proper orientation prior to the maximum heating period during entry.

Another point to be considered is that, although the capsule rotates only through a small angle during the high thrust burning, the burn-out angular rate may be rather high. The burn-out rate for a representative case (40 m/sec velocity increment, 1000 lb capsule, 5 sec burning time, 1 millirad thrust misalignment) is 0.8 deg/sec. A total turning angle of approximately 45 deg would result before attitude control jets of 0.3 lb thrust could overcome this rate. Since the resultant turning angle is approximately an order of magnitude higher than that which a body-fixed gyroscope could tolerate, it appears that an actively-stabilized capsule could not employ this simple form of mechanization.

Only a few capsule separation systems appear (from the previous discussion) to be of primary interest for Voyager, based upon following logic.

For the case where the capsule is spin-stabilized on the spacecraft (Figure 4-147), the flywheel technique was eliminated for reasons described in the earlier text of this report, leaving the remaining sequence: capsule spin-up on the bus, separation, coast, retro, de-spin, coast and entry. In the case where the capsule is spin-stabilized off the spacecraft (Figure 4-148), the flywheel again was eliminated. The two-axis optically-stabilized option was eliminated for reasons stated in Section II (E). This leaves the on-board capsule spin-up with gas jets and the off-board spin-up as the two remaining spin-stabilized systems of interest.

The two non-spin stabilized systems of interest use a two-axis, gyro-controlled gas-jet system with either a very high-thrust capsule propulsion system, requiring no TVC, or a very low-thrust propulsion system for which the coast-phase gas-jet system provides the thrust vector control.

For the non-spin-stabilized techniques (Figure 4-149), the two-axis optically stabilized [illegible] is [illegible] by the requirement to communicate up to entry.

The [illegible] of primary interest need to be considered [illegible] to the required attitude stabilization method and the [illegible] of the capsule [illegible] velocity is to be provided entirely by the capsule propulsion system. The methods of interest are summarized in Table 4-4[illegible].

In Table 4-[illegible] there are two capsule propulsion techniques [illegible] "capsule lateral-deflection plus capsule-bus arrival-time separation", and [illegible] "capsule lateral deflection only". In the "capsule lateral deflection plus capsule-bus arrival time-separation" the entire propulsion maneuver is performed by the capsule propulsion system [illegible] the capsule is fired ahead of the bus and toward the [illegible] with a velocity increment of perhaps 100 to 200 m/sec.

In the "capsule lateral-deflection only" method the capsule is fired normal to the flight path at 30 to 40 m/sec, and the bus is subsequently retarded to achieve the proper capsule-bus arrival time difference. With a lateral capsule deflection, the landing site dispersion is less sensitive to pointing errors, since some of the pointing error shows up as a flight-time error, which is not as serious as a lateral dispersion at the target area. In the full capsule-maneuver method, where the velocity increment is made in a direction more nearly normal to the critical plane (miss-[illegible] maneuver plane), not only do the pointing errors show up almost entirely as lateral dispersions at the target area, but also the actual velocity errors are larger due to the larger velocity increment required.

Thus, methods 5, 6 and 7 could be eliminated by accuracy constraints as discussed. This leaves method 8 as the only choice for achieving the desired accuracy with the capsule propulsion system supplying the total separation velocity. Method 8 is the most logical choice, if the accuracy requirements could be met.

The accuracy requirements for the capsule lateral-deflection-only maneuver are loose enough to permit use of off-board spin-up, method 2, instead of the on-board spin-up, method 1. [illegible] with method 2, the problems of a low-friction bearing on the spacecraft can be avoided. Method 1 would probably be favored over method 3 because of the difficulties in the latter method of stabilizing the capsule after the propulsion phase. In general, the spin-stabilized combinations are preferred over the gyro-stabilized systems because of the

Table 4-54. Separation Methods

Capsule Propulsion Provides:	Spin-Stabilized		Stabilization by two-axis, gyro-controlled, gas-jet system.	
	On-board spin-up with gas jets	Off-board spin-up with gas jets	Very-high-thrust maneuver	Very-low-thrust maneuver
Capsule Lateral-Deflection Only	1	2	3	4
Capsule Lateral-Deflection Plus Capsule-Bus Arrival-Time Separation	5	6	7	8

greater operational simplicity, the reduction in system complexity, and the associated increase in reliability

The following comments can be made concerning the propulsion system types to be used for the various separation systems shown in Table 4-54. A medium thrust, heat-sterilized solid unit is suggested for use with methods 1 and 2. For method 3, a high-thrust, truncated-nozzle, heat-sterilizable solid unit is the only contender. For methods 4 and 8, use of a low-thrust nonopropellant liquid system is proposed; an alternate for these situations is the use of a medium-thrust solid unit. The TVC for this medium-thrust solid could be provided by the coast-phase attitude control system.

An attractive separation system is one which is compatible with both methods 2 and 6. Method 2 would be used if the high accuracy requirements are to be adhered to. The main disadvantage of method 2 is that a subsequent propulsion maneuver on the bus has to be made in order to achieve any substantial capsule-bus arrival-time separation. Also, with method 2 the capsule-bus communication line-of-sight is marginal. Method 6 is a more desirable system since the entire capsule-bus separation velocity is provided by the capsule propulsion system The accuracy limitation of method 6 makes it difficult to realize landing sight dispersions of 200 to 300 km. Relaxing these dispersion requirements makes method 6 more attractive.

The proposed separation system and sequence of events is as follows.

1) The capsule-bus release mechanism is actuated. This occurs at a distance of approximately 10^6 km from the planet.

2) The capsule is axially separated from the bus at approximately 0.3 m/sec and simultaneously spun up to approximately [illegible] radians/sec for attitude stability. This separation and spin-up system is composed of three gas-jets located in a plane containing the capsule c.g. and normal to the capsule spin axis. The thrust vector of the jets is oriented 4 deg aft of this plane and normal to the capsule roll-axis radius. Cold gas (nitrogen) is used in the spin-up and separation system for simplicity, reliability, sterilization compatibility and for avoiding spacecraft optical surface contamination. Thus, with the cold gas there is no minimum capsule-spacecraft separation distance requirement as there is with a hot-gas spin system.

3) The capsule coasts for approximately [illegible] to [illegible] min and the propulsion maneuver is performed.

4) At the end of the propulsion maneuver the propulsion unit and the spin system are jettisoned. The spin system, bottle, jets, tubing and associated valving and other hardware are all mounted externally on the capsule. This allows placing the jets in the plane of the capsule c.g., minimizing the pitch and yaw torques of the system due to thrust-build-up delays in the nozzles and thrust unbalance, without bringing hardware through or putting fittings into the heat shield or ablative material. The spin system and propulsion unit are released by a single-release mechanism which requires only one command from the capsule.

5) Prior to entry the capsule is de-spun using a "yo-yo" de-spin technique. The capsule is not completely de-spun so as to maintain proper attitude for communications prior to entry, but its angular momentum is reduced sufficiently to allow it to have passive aerodynamic stability. The exact value of the residual spin rate which is compatible with these two requirements has not yet been established. It is possible that a de-spin maneuver will not be necessary.

2. Ballistic Entry

The capsule ballistic entry phase covers that portion of the capsule mission from atmospheric entry to parachute deployment. The atmosphere entry is defined as that point at which the capsule first encounters the uppermost layer of the planet's atmosphere. At this point the capusle will have ejected its propulsion system, attitude control system and any other equipment not required to survive entry. The external shape presented at this time will be the atmospheric entry shape. This aerodynamic shape is totally passive; that is, there is no requirement on entry angle or angle-of-attack-control. Recognizably, this type of design is non-optimum and accrues penalties in both structure and heat shield weights, but at this time it is felt prudent to absorb these performance degradations to enhance mission reliability.

A few seconds after the planet's sensible atmosphere is encountered, the capsule begins to orient its roll axis along the flight path. Figure 4-150 shows a typical decay in the amplitude of the angle of attack for two entry angles as a function of altitude. The frequency history of the angle-of-attack oscillations is shown in Figure 4-151, again as a function of altitude. In order to time-correlate these two phenomena, capsule altitude as a function of time is presented in Figure 4-152.

After the capsule roll axis is nominally oriented along the flight path maximum heating occurs, immediately followed by maximum deceleration (see Figure 4-153). In developing Figures 4-150 through 4-153, the Discoverer shape was used; however, the phenomena exhibited on these curves is typical of the behavior to be expected of a passive entry vehicle. Changing the aerodynamic characteristics of the capsule causes phase shifts and amplitude changes in the curves. After the capsule velocity decays from hypersonic to supersonic, the aft cover is removed, the supersonic drogue chute is deployed, and the capsule heat shield drops off. The drogue chute decelerates the capsule through transonic velocities. The reefed main-chute is then deployed at high subsonic velocity and is subsequently de-reefed for the terminal parachute descent phase. A typical mechanization for these events is indicated in Section II-B with the sequence of events.

3. Terminal Parachute Descent

The parachute descent phase covers that portion of the capsule mission from the time the main chute is fully deployed until landing. At the start of this phase both the heat shield and aft cover have been removed and only the landing portion (referred to as the lander) remains suspended by the parachute. It is anticipated that all the larger amplitude oscillations will have decayed and the lander will be nearly aligned along the local vertical. The actual attitude of the capsule roll axis at this phase of the flight will be determined by several considerations. Some of these considerations are the magnitude of the ground wind speed, the parachute design including its attach method to the landing package, and the weight and aerodynamic characteristics of the landing package. With the capsule roll axis aligned near the vertical and the bus line-of-sight within 60 deg of the capsule roll axis, capsule-to-bus communications can be re-established.

The data which will have been collected and recorded during the ballistic entry can now be transmitted to the bus. Ideally, this data should all be transmitted before landing. This is done so the important atmospheric data can be collected in the event of a landing system failure.

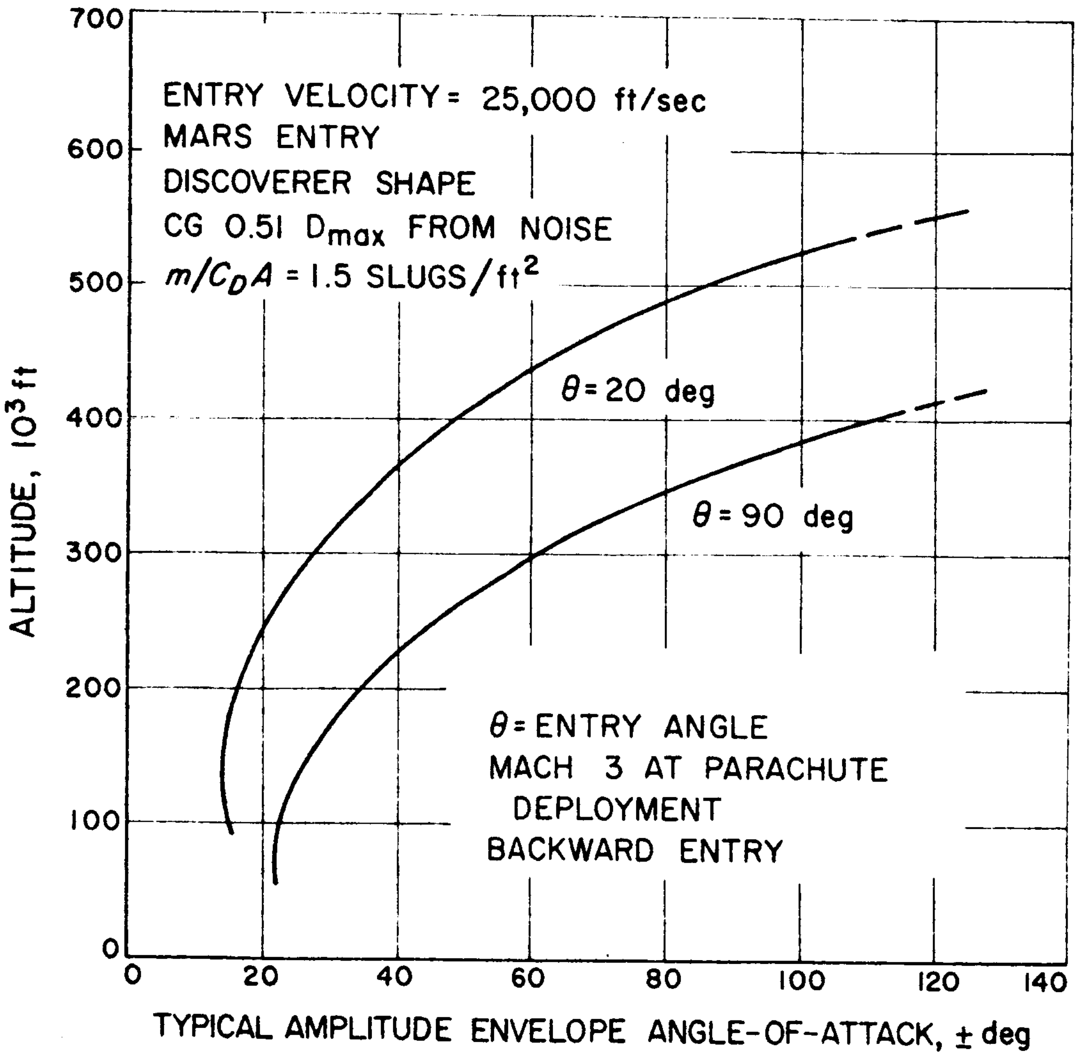


Figure 4-150. Capsule Angle-of-Attack Vs Altitude

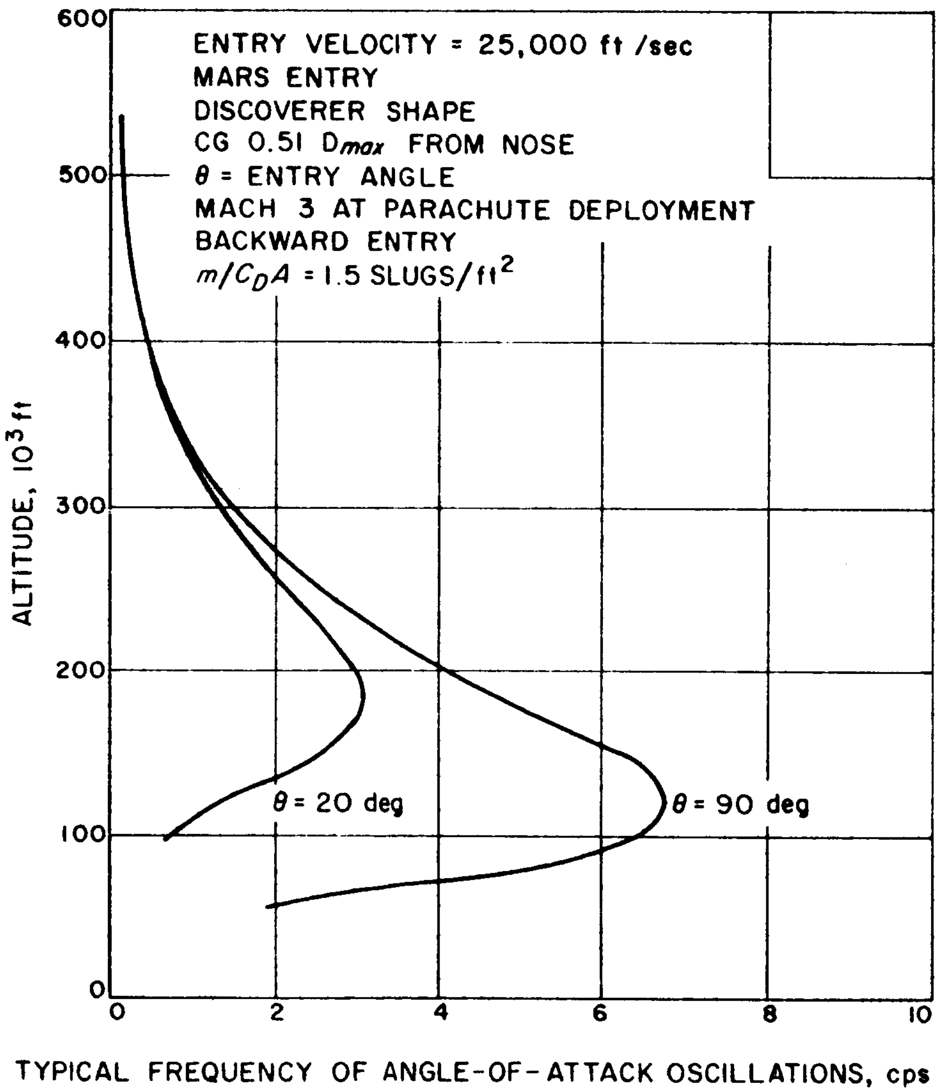


Figure 4-151. Capsule Angle-of-Attack Frequency vs Altitude

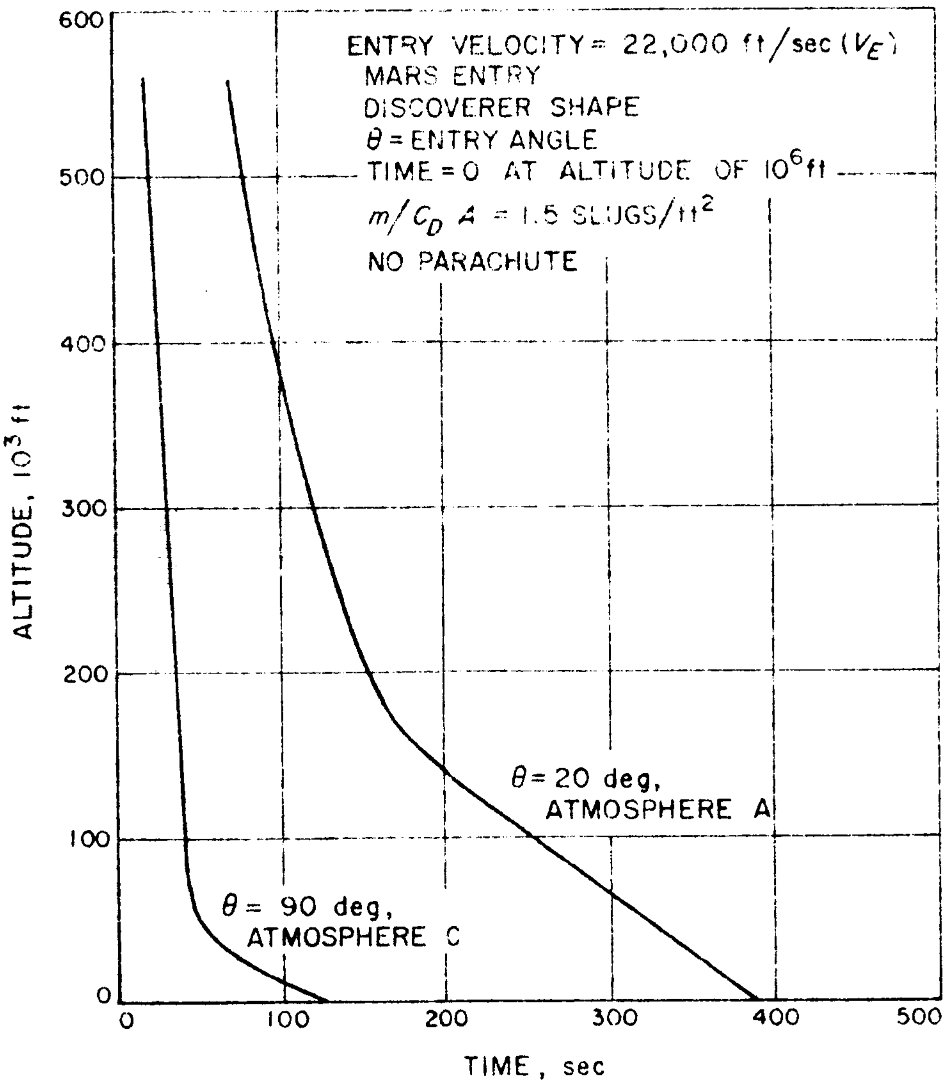


Figure 4-152. Capsule Altitude Vs Time

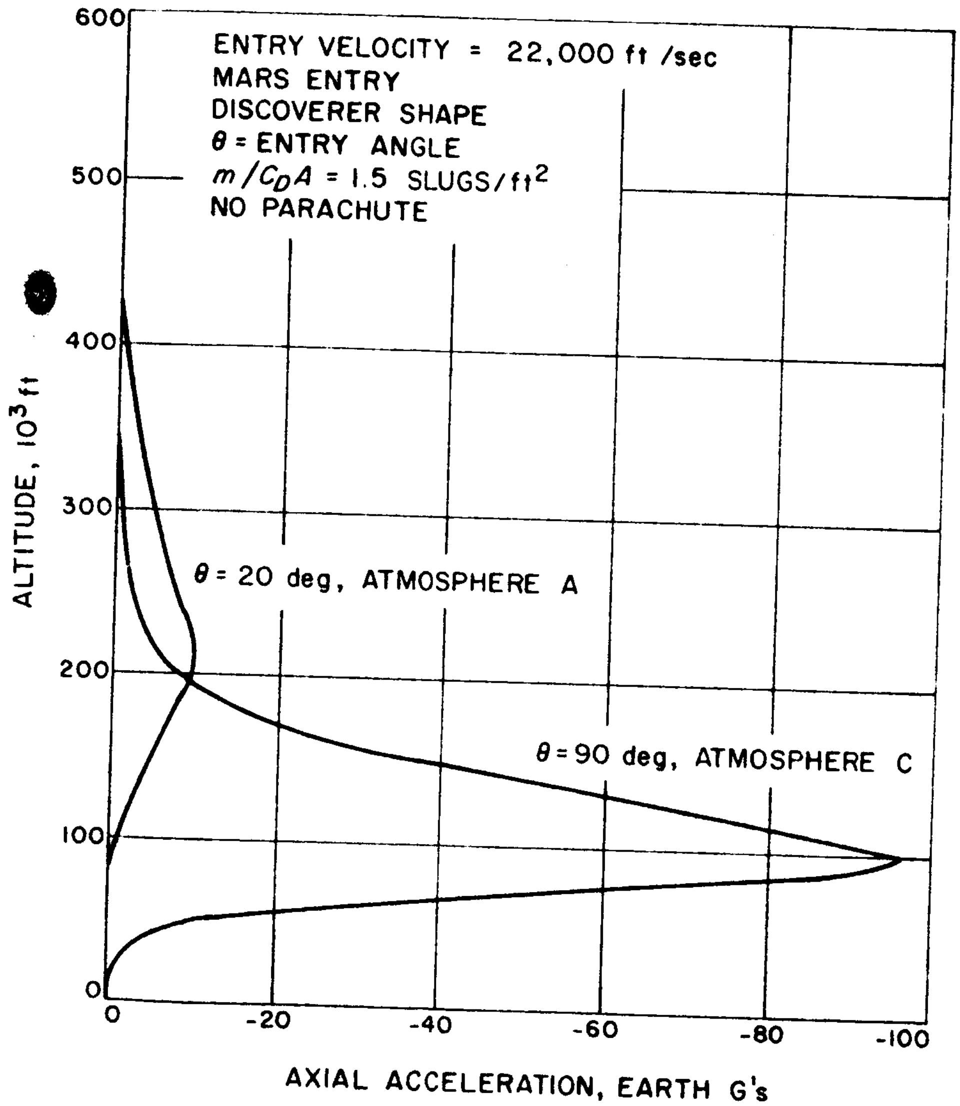


Figure 4-153. Capsule Axial Acceleration vs Altitude

Prior to the end of the descent phase all the prelanding operations are accomplished. This may include erection of landing mechanisms, orientation of impact absorbers and any other necessary landing operations.

Figure 4-154 shows certain parameters for a typical parachute system. The sensitivity of terminal descent time to retardation system weight is readily apparant.

4. Landing

The landing phase of the mission includes the initial touch-down, parachute release and impact absorption.

The primary function of the landing system is to remove the residual parachute descent velocity and maintain the equipment within its design environment. The normal component of the final descent velocity is determined by the parachute system design, and can be maintained as low as 20 to 30 ft/sec. if necessary. There is a weight trade-off between the parachute and landing systems because the weight of the parachute increases with decreasing landing velocities. The landing system design, in addition to complementing the parachute system, must also consider the expected ground winds. Present estimates of these winds are rather uncertain but also rather high (for some estimates, as great as 200 ft/sec). The resulting velocity is the vector sum of the ground winds and parachute descent velocity. It is this velocity or range of velocities in conjunction with the local ground conditions, that will determine the landing system design.

The landing system can be one of two types, the first is a directional or quasi-directional system, the second is an omni-directional system. The first system is dependent on sensing the direction of the landing velocity vector with respect to a fixed reference axis on the lander and the expected ground conditions. Although this type of design can yield a lighter weight system, the confidence level which can be established for the design conditions is rather low at present.

The second type of landing system is one in which the velocity can be absorbed from any arbitrary direction with respect to the lander. This type of system is by far the most reliable, but also the heaviest and, hence, least efficient.

The design of a reliable system to perform this task will require a large amount of engineering effort. Several preliminary conceptual designs using crushable material as a terminal energy absorbant have been investigated, but these still do not constitute even a preliminary design.

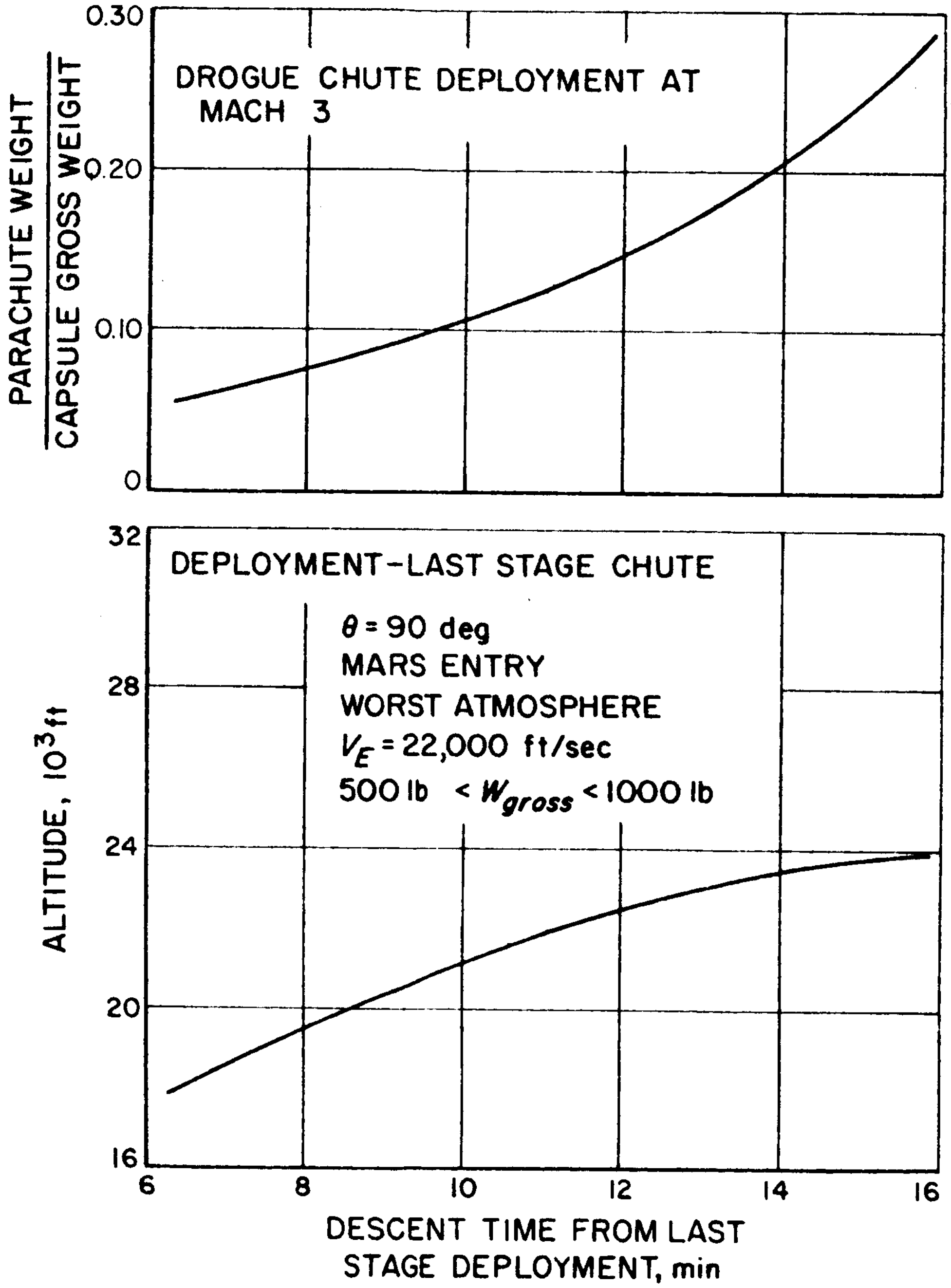


Figure 4-154. Typical Parachute Parameters

5. Post-Landing

The post-landing phase of [illegible] mission [illegible] and subsequent [illegible] of the lander on the planet surface.

After the capsule is successfully landed it is necessary to erect the communication antenna, certain of the science instruments and other equipments. This can be done either by having the landing system land the capsule in a known attitude or by using an omni-directional landing system, then subsequently erecting the capsule or parts of it.

A landing system which would land the capsule in a known orientation would possibly yield a lighter overall landing system. Because of the uncertainties in the ground conditions and the magnitude of the ground winds, it would be a very risky if not impractical design. The omni-directional landing system, on the other hand, will place an extreme penalty on the payload capability but may well be the only means to insure a reasonable probability of successful operation of the landing system.

Using such a system requires that the capsule erection system perform from any arbitrary orientation after initial impact. The mechanization of such a system is rather complicated and heavy. Several such systems have been investigated in order to highlight some of the more critical design and development problems to be confronted. Until some resolution of the surface environment can be established, an omni-directional system will have to be considered.

3. Effects on Entry Vehicle Design Due to Atmosphere Change

Recent observations of Mars and subsequent changes in the atmospheric model cause serious perturbations on the design philosophy for a Mars entry vehicle. The work done during the major portion of this study was based on the Martian atmosphere developed by the RAND Corporation for JPL.[1] Since the aforementioned observations were made late in the study period, it was felt prudent to continue the effort with the original atmosphere. This approach allowed timely completion of the study and the identification of many problem areas in the capsule-bus interface which are essentially independent of the atmosphere.

The following discussion will concern itself, in a general manner, with the effects of such a modification to the Martian atmosphere on the entry vehicle design. Throughout this discussion the terms "new" and "old" atmosphere will be used; the "old" atmosphere is as in the RAND Report, while the "new" atmosphere is the RAND model modified to encompass the results of the recent spectrographic observations. The new model does not contradict the old model, but rather widens the uncertainty in the atmosphere.

In considering the effects of the new atmosphere on the entry vehicle design, the most significant change occurs in the ballistic coefficient ($\frac{m}{C_D A}$). It can be shown that:

$$\frac{m}{C_D A} = \frac{P(y)}{2g \sin \theta E \ln\left(\frac{V_E}{V(y)}\right)}$$

where P(y) and V(y) are the pressure and velocity at any altitude y. From the above expression it can be seen that, if g, θ_E and $\ln\left(\frac{V_E}{V(y)}\right)$ are kept constant, the ballistic coefficient varies linearly with pressure. With the new atmosphere the surface pressure is reduced by about a factor of five; therefore, it becomes necessary to reduce the ballistic coefficient accordingly. If the drag coefficient, C_D, remained the same as used for old atmosphere, the capsule cross-sectional area would have to increase by a factor of five. This means (for the same capsule performance) a diameter increase of 2.24. However, by increasing diameter only, there is a point at which the useable payload vanishes due to increase of structure weight. Thus it becomes necessary to investigate new entry-vehicle shapes, where the ballistic coefficient ($\frac{m}{C_D A}$) can be reduced by increasing both the drag coefficient (C_D) and the cross-sectional area (A).

The shapes which have been considered for the new atmosphere are flat-nosed with high drag coefficients and large cross-sectional area. The structural design approach of such a vehicle is considerably different from that of the low-drag, Discoverer-type vehicles investigated previously. The dynamic behavior of these new types of vehicles may be considerably different, for both the c.g. and the center of pressure are further forward and the ratio of pitch and/or yaw radius of gyration to capsule diameter changes.

The implications of these new environmental estimates on the design of an entry vehicle will be further explored in succeeding paragraphs. It must be emphasized that the differences between the basic data supporting the earlier atmosphere model and that forming the basis for this newer model have not yet been fully explained. Therefore, this new model not only serves to broaden the uncertainty as to the environment that an entry vehicle must withstand but also points out the necessity for emphasis on in-flight measurement of the actual atmosphere.

With the basic assumption that the descent velocities near the surface should be the same for both atmospheres, the following comparisons can be drawn:

1) The ballistic coefficient ($\frac{m}{C_D A}$) will be roughly proportional to atmosphere surface pressure, so that five times as much cross-sectional area will be required in the new atmosphere. If a flat-nosed shape is employed to increase the drag coefficient, the cross-sectional area might only have to increase 2.5 times, which corresponds to a diameter increase of 1.58.
2) The maximum drag load will be proportional to the atmospheric density gradient, β. The maximum β ratio indicates that total drag loads will increase by 1.3 although nose pressures will now decrease by a factor of $1.3/2.5 \simeq 0.5$.
3) The altitude of peak acceleration is an inverse function of β and a log function of $\frac{P_{surface}}{m/C_D A}$. Since the ratio of $\frac{P_{surface}}{m/C_D A}$ is kept approximately constant to satisfy the criteria of equal descent velocities, the increased β for the new atmosphere model tends to lower the altitude at which the atmosphere decelerates the capsule. The ratio of new peak deceleration altitude to the old altitude and therefore roughly the ratio of new descent time to the old time is approximately $1/1.3 = 0.77$ (or 0.72 if pressure uncertainties are included). This altitude was already marginal for acceptable data transmission time during parachute descent in the old atmosphere.
4) The change of atmosphere will affect certain major-weight components of the capsule. These components are the heat-shield convective- and radiative-ablation materials, the insulation, the external pressure-load bearing shell, and the parachute (or any other retardation system). These components are most immediately affected by the environment, and as such, they are also most amenable to scaling versus atmospheric and flight performance parameters.

The fundamental relationships between heat-shield weight fractions and atmospheric-performance parameters were outlined in IAS Paper 62-96[2], and a revised version follows:

Convection Ablation

$$\frac{W_{AC}}{W_G} \alpha \frac{1}{C_D^{3/4} W_G^{1/4} \left(\frac{m}{C_D A}\right)^{1/4} (\beta \sin \theta_E)^{1/2}}$$

Radiative Ablation

$$\frac{W_{AR}}{W_G} \alpha \frac{W_G^{1/2} (R_N/R)^{2.62} (\beta \sin \theta_E)^{1/2}}{C_D^{2/3} \left(\frac{m}{C_D A}\right)^{0.1}}$$

Insulation

$$\frac{W_I}{W_G} \alpha \frac{(S/A)}{C_D \left(\frac{m}{C_D A}\right) (\beta \sin \theta_E)^{1/2}}$$

Heat Shield Structure

$$\frac{W_S}{W_G} \alpha \frac{W_G^{1/2} (S/A)(R_N/R)(\beta \sin \theta_E)^{1/2}}{C_D^{3/2} \left(\frac{m}{C_D A}\right)}$$

A similar approximate relation for a generalized parachute retardation system may be expressed as follows:

$$\frac{W_{RET}}{W_G} \alpha \frac{W_G^{1/2}}{C_D^{1/2} \rho_O^{1/2} V_T}$$

Symbols	Definition
A	Projected Frontal Area = R^2
C_D	Drag coefficient based on A
P	Atmospheric pressure
m	Mass of capsule
g	Acceleration of gravity
R	Maximum radius of capsule
R_N	Nose Radius
S	Surface Area
V_T	Parachute terminal velocity
W_G	Capsule gross weight
β	Atmosphere inverse scale height
ρ_o	Atmosphere surface density
θ_E	Entry path angle

Payload capabilities of the Discoverer-type configuration in the old atmosphere are shown on the left side of Figures 4-146a and 4-146b. The decrease of payload at the higher gross weights is due to the general inefficiency of parachutes at high weight and the particular inefficiency of the type of load-bearing structure assumed. It must be noted here that the structural dependencies are based on a probably over-conservative pressure-buckling shell design and on load transmission paths between payload and shell similar to those used on Discoverer. Due to expected lower unit surface pressures on the capsule in the new atmosphere, it seems likely that more appropriate and efficient structural and packaging methods would be used. However, because of current lack of such advanced method, the derivation of component weights for the new atmosphere uses the former method.

In order to determine an appropriate $\frac{m}{C_D A}$ for a capsule in the new atmosphere the following expression was used:

$$\frac{m}{C_D A} = \frac{P(y)}{2g \sin \theta_E \ln\left(V_E / V(y)\right)}$$

Assuming minimal parachute deployment conditions to be Mach 3 at 15,000 ft of altitude, the maximum allowable capsule ballistic coefficient is approximately 0.32. As discussed earlier, the Discoverer-type capsule in the new atmosphere produces no payload at this $\frac{m}{C_D A}$. The right side of Figures 4-14a and 4-14b show the limited payload capabilities of the Apollo-type configuration under the same scaling laws. These figures assume no entry-angle control, and therefore that $_E = 90°$. It will be noted that, if 1300 ft/sec impact velocities are tolerable, a hard lander can develop about twice the payload of the soft lander at about twice the gross weight. Close entry-angle control (to near the skip-out limit) woul be necessary to attain the payload capability of the A.32-1350 curve with a soft landing.

Apollo-type configurations could have been designed for payloads in the old atmosphere. As seen from the left-hand curves, the soft-landed payload capability of such a configuration would be intermediate between those of the old-atmosphere Discoverer and the new-atmosphere Apollo designs. This is a rough indication of the payload loss inherent in designing for large dispersions in the atmosphere model.

5) The larger size for the same weight creates two related problems with the Environmental Control subsystem. Due to the new shape, the electronic payload may be distributed over a relatively large area and thus larger conduction paths are required to obtain equivalent conductive linking between parts. Since the conductance is proportional to the cross-sectional area divided by the length, either the weight must be increased or the conductive linking must become worse. Similar comments apply to the path from the electronics to the radiator. Some sort of fluid transport device may find application here.

The other problem associated with the new shape is the larger surface area. This gives additional places for undesirable heat leaks to occur, thereby increasing the weight of insulation required.

Another aspect of the reduced atmospheric density is the reduction in the free convection coefficient (h_c) at the surface of the radiator, according to the relationship $h_{c\ (NEW)} = h_{c\ (OLD)} \left(\frac{\rho\,(new)}{\rho\,(old)} \right)^2$; this effect is not deemed serious since free convection is only a small contributor to the total heat-rejection capability of the radiator, even in the old atmosphere. However, the existence of high-velocity winds, possibly dust laden, may be more serious. Should they tend to sandblast the radiator surface, and thus alter the surface properties, a serious degradation of heat-rejection capability might result.

a. Retardation System

A parachute-type retardation is still preferred for the following reasons. Although much less efficient in the new atmosphere, it is still more efficient than a retrorocket on a velocity-increment-per-pound basis. In addition, it can more easily provide a longer atmosphere sampling and transmitting period if such is desired.

b. Outlook

A new version of the atmosphere has only recently been defined enough for the conduct of an engineering investigation into its effects on capsule design. There is at present no rational basis for choosing between the old and new atmosphere models. As evidenced by the weight comparisons, it appears that a larger capsule may be feasible in the old atmosphere rather than in the new one.

There are four alternatives available for obtaining the obviously required direct atmosphere measurements.

1) Small, simply instrumented capsules would have larger payload fractions available to obtain limited direct measurements.
2) Close observations of the electromagnetic properties of the Mars atmosphere from a flyby spacecraft, from which thermodynamic properties could be inferred.
3) Observations similar to (2) from Earth satellite or surface. Further explorations should be made of the possibility of significantly improving knowledge of the planet atmosphere by more extensive use of existing astronomical facilities.
4) Earth-based observations of the doppler frequency shift of a radio signal passing through the Mars atmosphere as a spacecraft passes "behind" Mars, as seen from Earth.

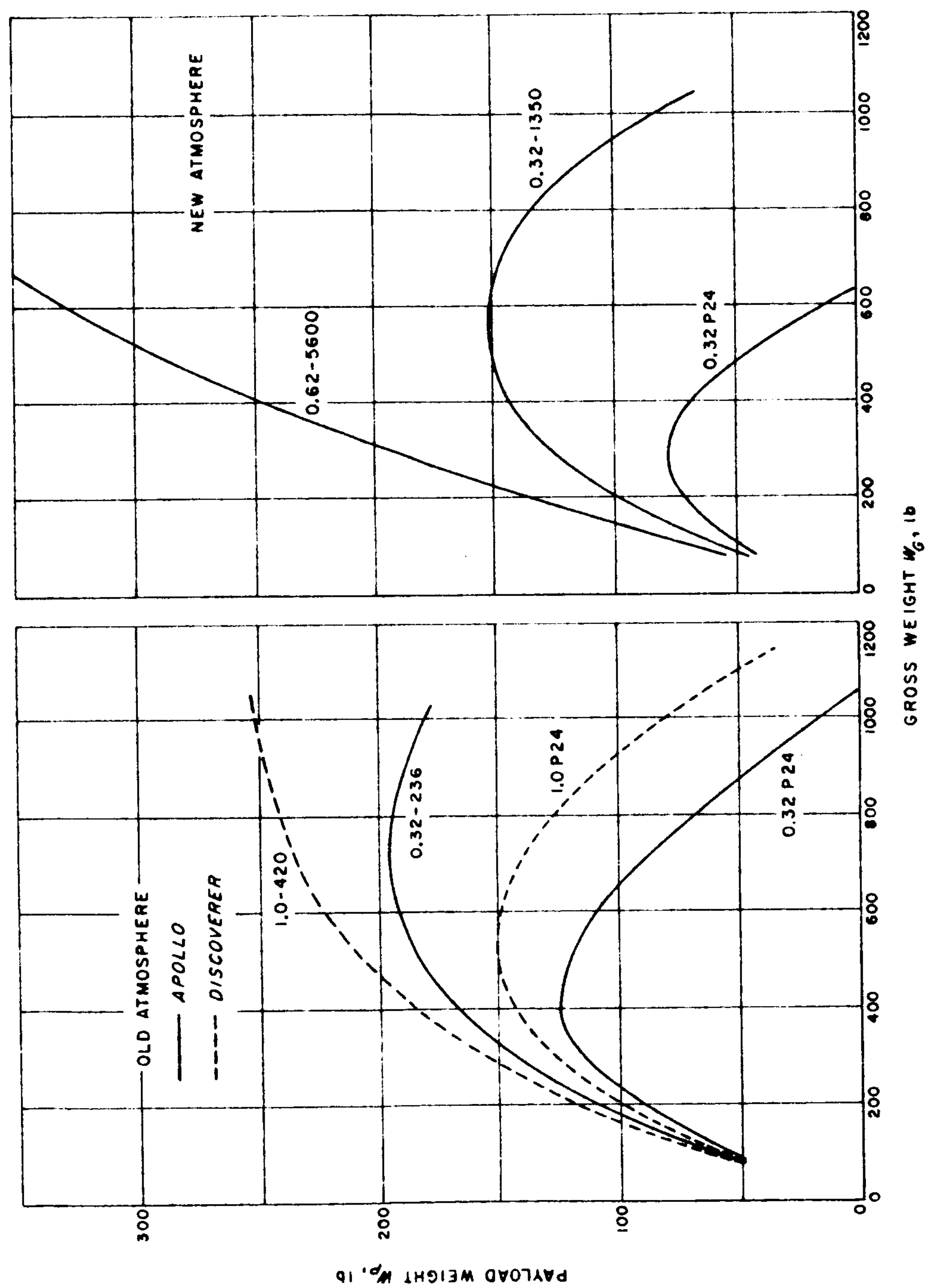


Figure 4-154a. Effect of Atmosphere on Mars Capsule Weights

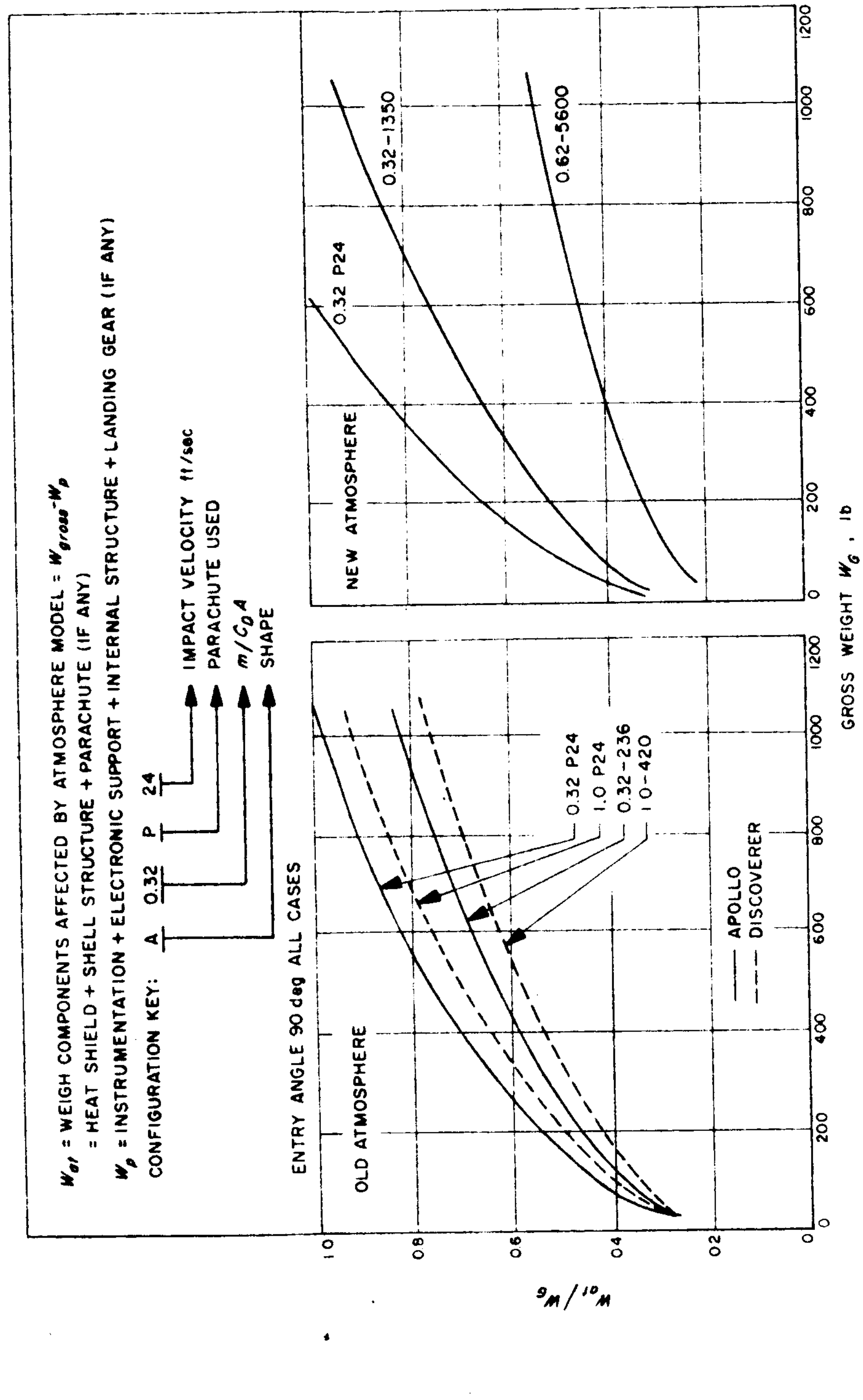


Figure 4-154b. Effects on Capsule Weights

REFERENCES

1. G. F. Schilling, "Limiting Model Atmospheres of Mars," Rand Corp. Report R-402-JPL, August 1962.

2. J. M. Spiegel, "Effects of Mars Atmosphere Uncertainties on Entry Vechicle Design," Aerospace Engineering, December 1962.

M. ENVIRONMENTAL CONTROL

The following discusses the capsule thermal control system. Since the orbiter thermal control system will be maintained independent of the capsule, no discussion of it will be included. The reader is referred to Volume II, EPD-139 for the discussion of the orbiter thermal control problem.

1. Thermal Control System

The thermal control system for a planet-entry capsule must function over a broad range of environmental conditions. These conditions can be categorized as follows:

1) On bus cruise. Quite possibly in the shade of the spacecraft; no power dissipation; long time duration.
2) Preseparation Check Out. Still attached to bus; possibly high internal power dissipation; relatively short time duration.
3) Separation and Retro Sequence. Free of bus, relatively fixed orientation relative to Sun; may be spinning; relatively short time duration.
4) Free Flight. Spinning, three to four days duration; probably low power dissipation.
5) Entry. Short duration; very high heat fluxes; high "G" loads.
6) On Surface. Unknown environment relative to gas composition, density, temperature, and velocity; temperature tolerances relatively tight due to biological experiments; high power dissipation.

Due to the influence of the entry phase of flight upon the overall configuration, the general capsule layout seems to contain three or four relatively isolated subsystems from the thermal standpoint. These are the:

1) Heat Shield-Outer cover, consisting of an ablating surface to dissipate the high aerodynamic heat fluxes produced at entry, plus the necessary structrual support.
2) Net Electronic Payload. The operating part of the capsule once on the surface; biological experiments, radio gear, etc., contained within the heat shield.
3) Retro-Attitude Control System. A strap-on subsystem, external to the heat shields, used during the separation and retro sequence, and subsequently dropped.
4) Radioisotope Power Supply (if used). This unit should be mounted on the external skin and insulated from the interior of the capsule. While on the bus, provision must be provided to radiate the heat away without interfering with the spacecraft proper. Other than these considerations, the radioisotope

ower system presents no conceptual problems and will not be further considered in this discussion.

The following list outlines briefly the major features of a possible temperature control system for the Mars entry capsule:

1) The three major subsystems, i.e., the net electronic payload, the heat shield, and the attitude control-retro motor package must be thermally isolated from each other and from all mounting rings or attachments. This isolation will probably consist of fiberglass structural connections as well as aluminized mylar superinsulation. In addition, the net electronic payload must be isolated from the atmosphere when the heat shield is removed.
2) Due to the difficulty of calculating, measuring, or controlling the heat transfer rates from the electronics through the heat shield, no attempt will be made to control the temperature of the heat shield by allowing heat to leak from inside the capsule.
3) A radiator large enough to provide adequate heat rejection from the electronic payload under the most adverse conditions on the surface of Mars will form a part of the afterbody of the capsule. This radiator will be attached to the electronic payload at all times through thermal switches and insulating structure. These switches provide the variable heat path necessary to maintain the electronics package temperatures during all phases of separated flight, yet are rugged enough to withstand landing loads.
4) Thermostatically controlled electrical heaters may be required during the cruise portion of the flight, either in the sterilization can or in the electronic and retro/attitude-control packages.
5) A metal heat-shield liner will also probably be required to reduce the thermal gradient through the heat shield.
6) The diaphragm separating the bus and the capsule will be insulated so that the thermal effect on the bus from capsule separation will be negligible.

During the analysis involved in arriving at the above temperature control system, the capsule was considered as a group of isothermal parts, with thermal connections as shown in Figure 4-155. These connections are as mentioned in item 1) above.

2. Detail Description

Following is a more detailed description of each of the three major components investigated:

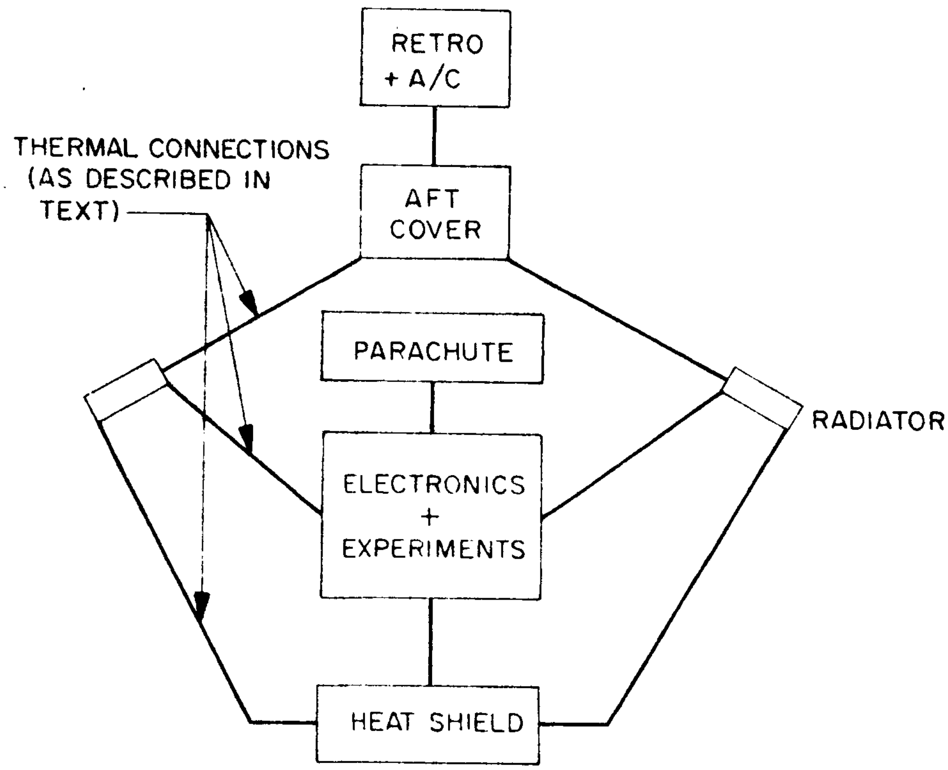


Figure 4-155. Capsule Thermal Schematic

a. Heat shield

The general problems here are as follows (based on an existing Discoverer study):

1) The material of the shield begins to deteriorate below approximately -150°F, making it unable to perform its ablating function during entry.

2) Temperature gradients over 300°F around the shell cause excessive thermal stresses in the shield materials.

Consequently, estimates of steady state temperatures for the heat shield both on and off the bus have been made. On the bus, if the capsule is in the shade of the spacecraft during the whole transit, the only heat inputs are the IR from the back of solar panels. This is a very small input, giving a heat shield termperature of approximately -275°F. Thus, some means must be provided for keeping the shield warm. The most promising method seems to be to insulate the sterilization can, keep it over the capsule until capsule separation, and provide thermostatically controlled heaters inside the can. An alternate method would be to provide some sort of sunlight reflectors to deflect sunlight onto the heat shield. For the RTG powered spacecraft, the heat shield can be maintained in sunlight, thereby maintaining adequate temperatures during the entire transit.

After the capsule is separated, Figure 4-156 shows the average heat shield temperature as a function of the surface α/ϵ ratio and the distance from the Sun. These average temperatures are representative if a pitch or yaw tumbling rate of at least 2 rev/hr is achieved about an axis perpendicular to the Sun-capsule line. In case no tumbling rate is imparted, Figure 4-157 illustrates the temperature extremes which will be experienced by a cylinder constructed of various materials. This figure indicates that for a cylinder made of 1/2 inch Phenolic Nylon with an emissivity of 0.04 at 90 deg to the Sun line at Mars aphelion, the temperature at the subsolar point will be 777°R and on the opposite side the temperature will be 283°R, a gradient around the cylinder of 494°F. On the other hand, if an aluminum liner 0.020 in. thick is installed, the corresponding temperatures are 706°R and 440°R, a gradient of 266°F. Consequently, if no tumbling rate can be imparted, the metal liner is mandatory to avoid large temperature gradients.

b. Retro Motor and Attitude Control System

The general problems are as follows:

1) The system must be maintained at some temperature while on the bus.
2) The system must not overheat during the separation maneuver.

If the retro motor and the attitude control system are thermally well integrated as a system, and the system thermally insulated from the rest of the capsule and from space except for the end of the nozzle, the system time constant is approximately 200 hr. Thus, it appears that the thermal inerial will be enough to provide for less than 100°F temperature rise during the entire separation sequence.

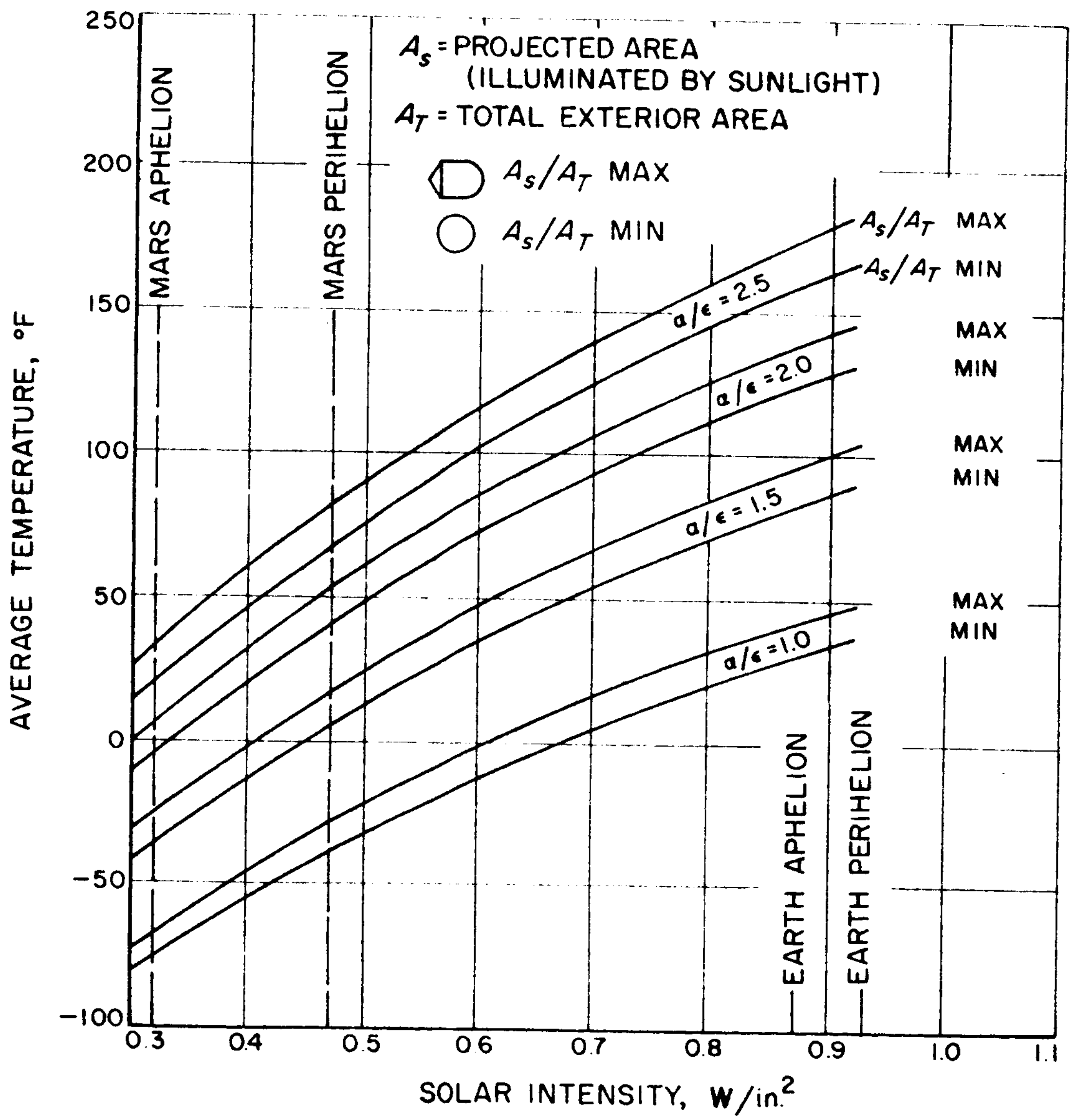


Figure 4-156. Average External Temperature for a Mars Entry Capsule

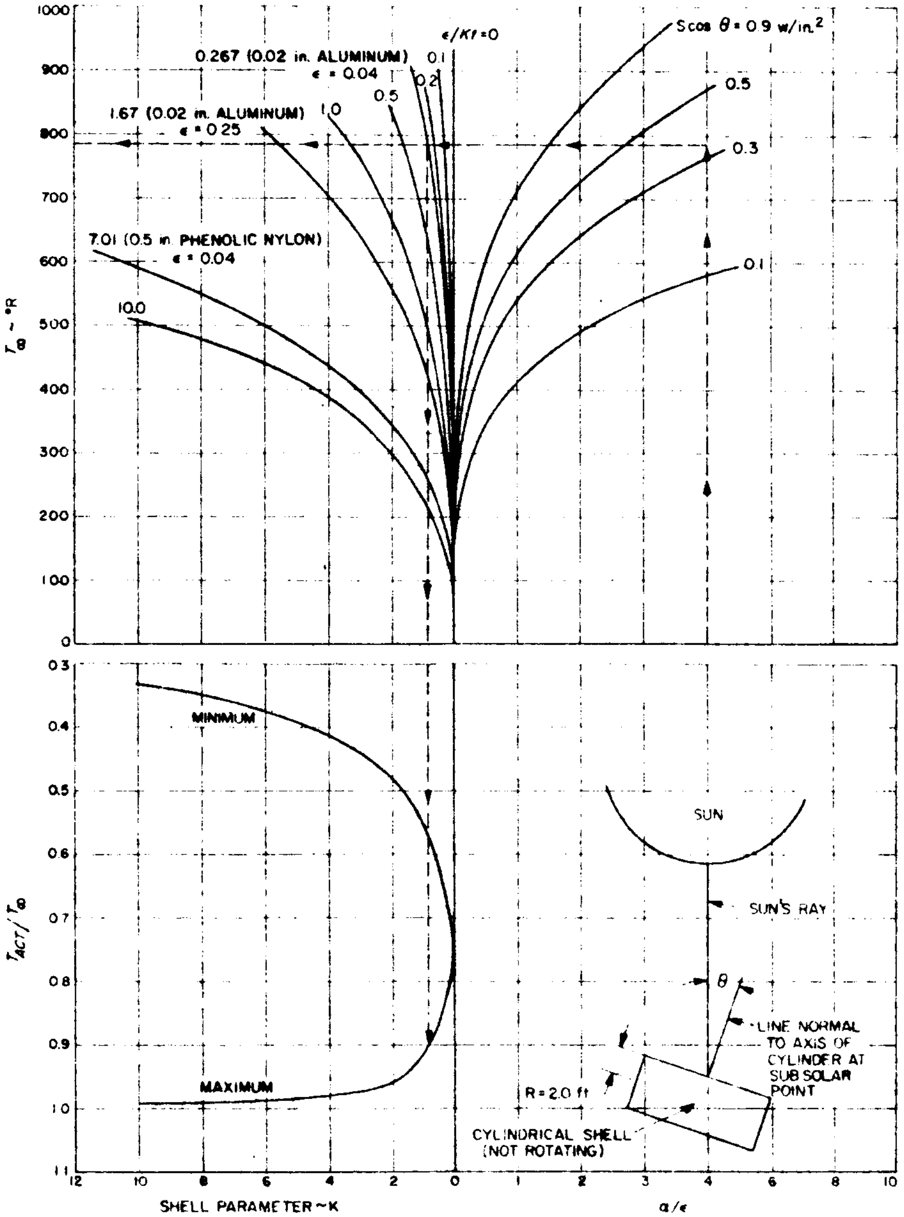


Figure 4-157. Temperature Extremes in a 4 Foot Diameter Cylinder

Prior to release of the capsule there are two solutions to the thermal control problem. If the sterilization can is left on and thermostatically heated, the separation system will be maintained at an adequate temperature. If the can is removed, a small independent thermostatically controlled heater must be provided for this system.

c. Net Electronic Payload.

The primary problem in this section of the capsule is maintaining the temperatures within reasonable bounds:

1) During the cruise portion of the transit while the capsule is on the bus.
2) During entry.
3) On the surface (to allow biological experiments to be made).

During the on-bus period of the transit, if the sterilization can is left on, no special controls are required for this portion of the capsule. If the sterilization can is removed, thermostatically controlled electrical heaters will be required in the electronic payload.

Once the capsule is separated from the spacecraft, the amount of power dissipated varies over a large range. Since the external environmental conditions change over a wide range, some manner of controlling the path through which the dissipated power can flow is necessary. Probably the most rugged devices for doing this are thermally-actuated variable-conductance switches, similar to the type developed for Surveyor. Figure 4-158 is a sketch of the Surveyor switch. These devices have a control ratio of approximately 100 to 1 in space, somewhat less in an atmosphere. The minimum time constant for the electronic payload with the switches closed is on the order of 4 hr. The radiator for the electronic packages will be an annular ring or a series of segments flush with the aft cover. Figure 4-159 shows the location of this radiator.

During the hard entry, if the capsule comes in backwards at a grazing angle, the aft cover and the radiator experience the maximum heat pulse. Figure 4-160 shows the magnitude of this heat pulse, compared with a forward entry. This maximum heating pulse generates (at most) a 30°F temperature rise within the electronic payload.

Once hard entry has been accomplished, the heat shield should probably be dropped since heat soak from the heat shield may prove undesirable.

Convective heat fluxes during the descent and on the surface are largely indeterminate due to the uncertainty in the atmospheric properties. It appears, however, that a large percentage of the heat transfer will be radiative, even on the surface. Figure 4-161 illustrates a first approximation of the heat transfer characteristic of a radiator in still air on the Martian surface.

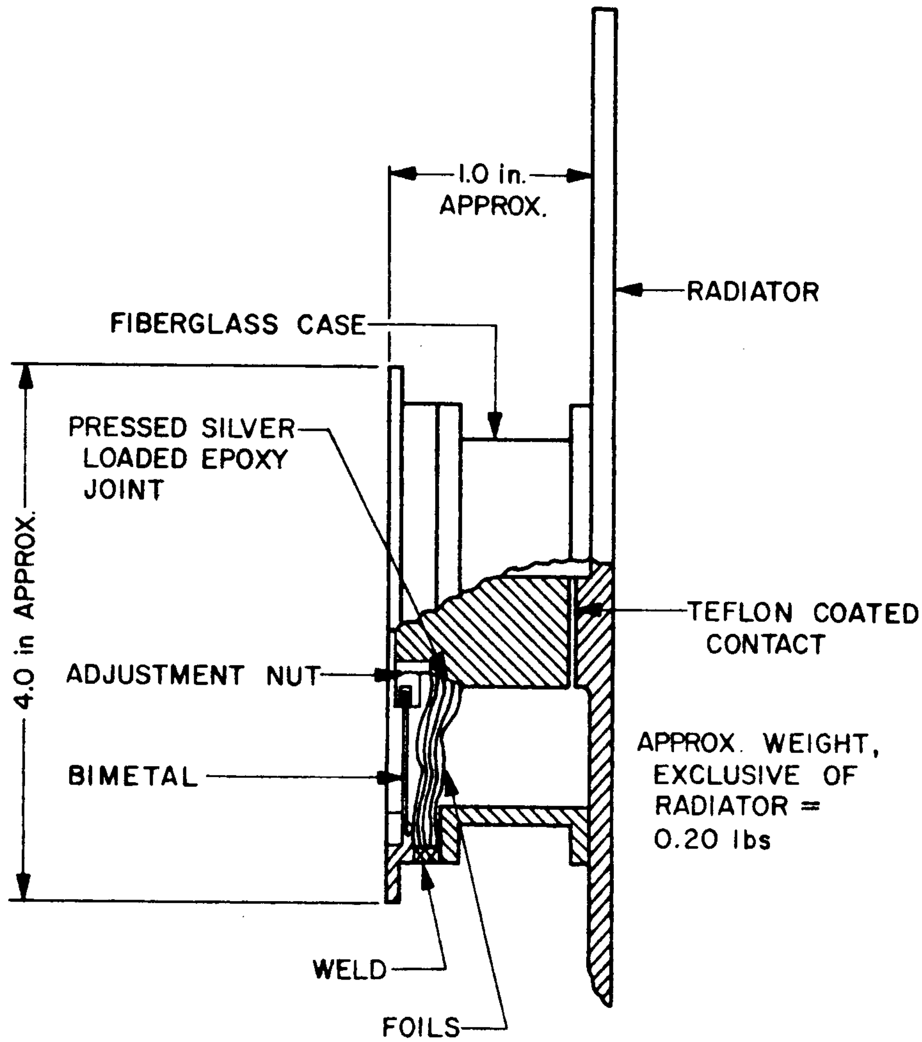


Figure 4-158. Surveyor Thermal Switch

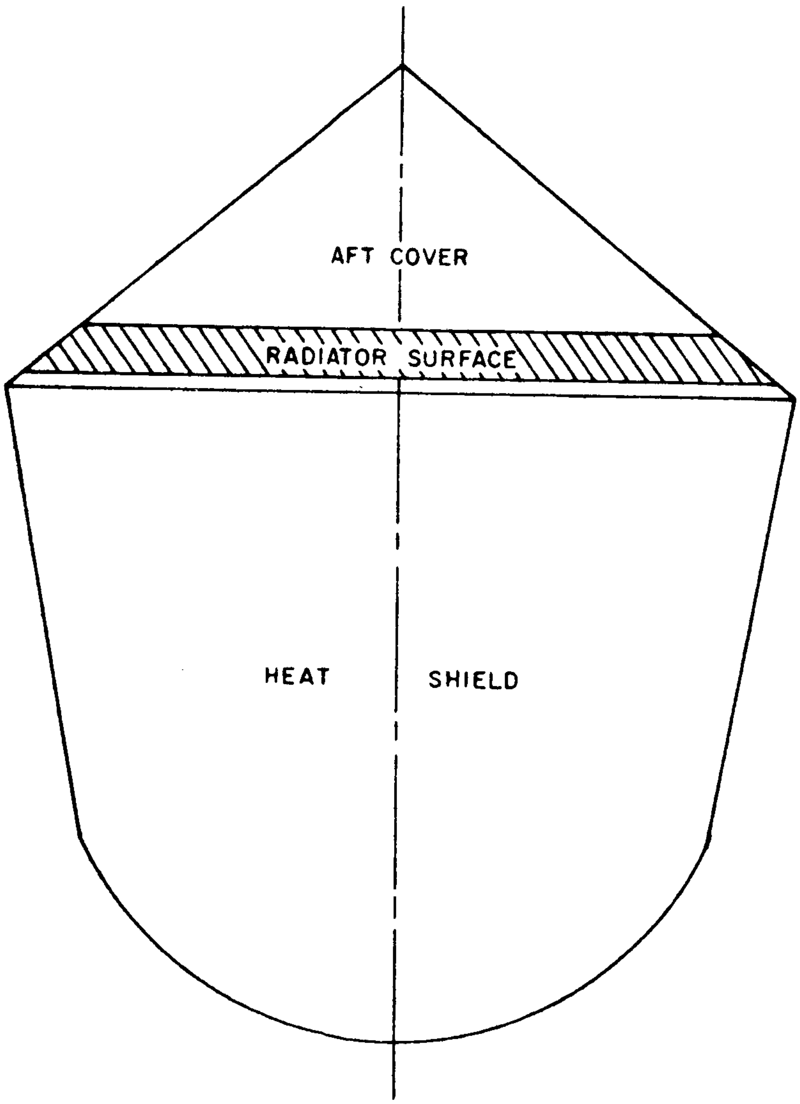


Figure 4-159. Radiator Location for a Mars Entry Capsule

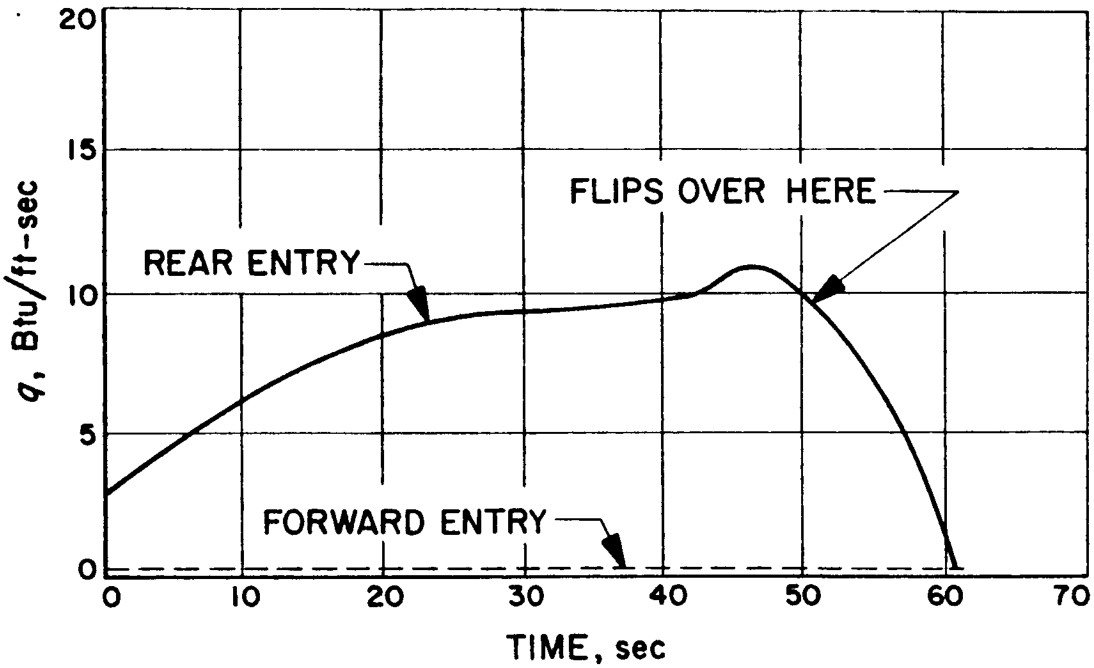


Figure 4-160. Aft Cover Heat Fluxes for a Mars Capsule

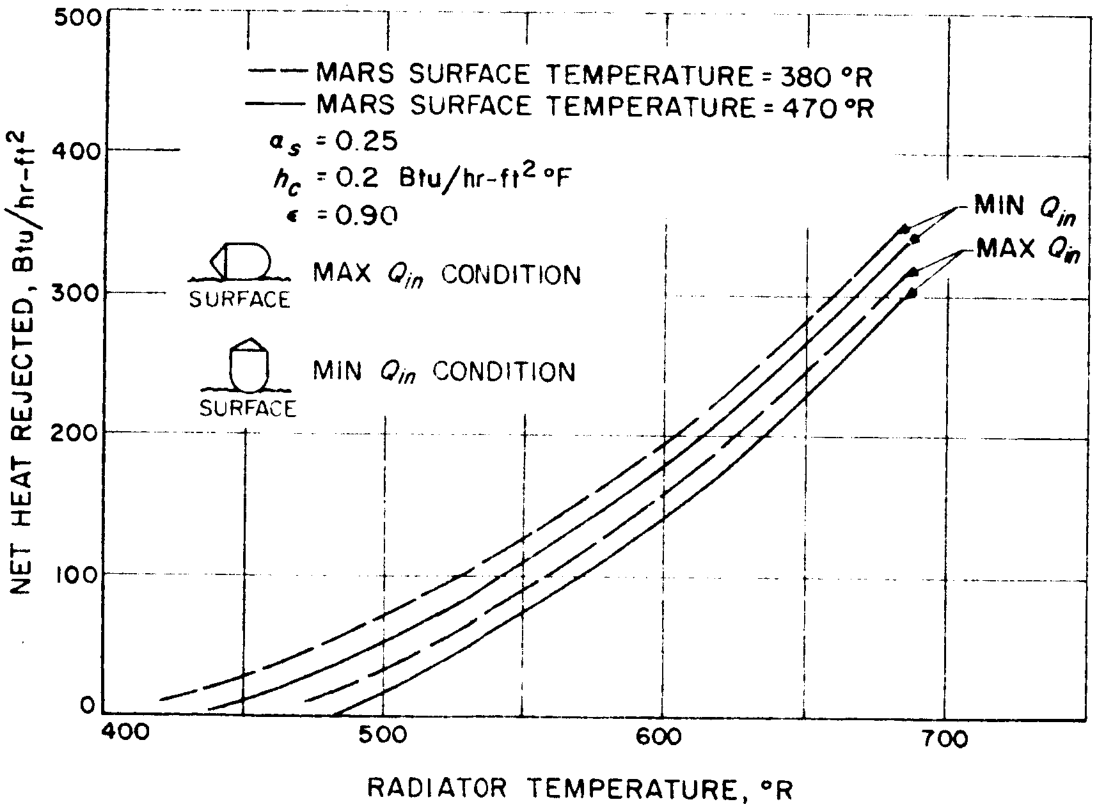


Figure 4-161 Radiator Characteristics on Martian Surface

3. Sterilization

A requirement exists to heat sterilize the entire capsule once it has been placed inside the sterilization can. As a result, an estimate of the time involved in the sterilization process is desired. Figure 4-162 illustrates the time required for each of the three major parts of the sterilized package to come to sterilization temperature vs the oven temperature. It should be noted that these curves indicate order of magnitude only.

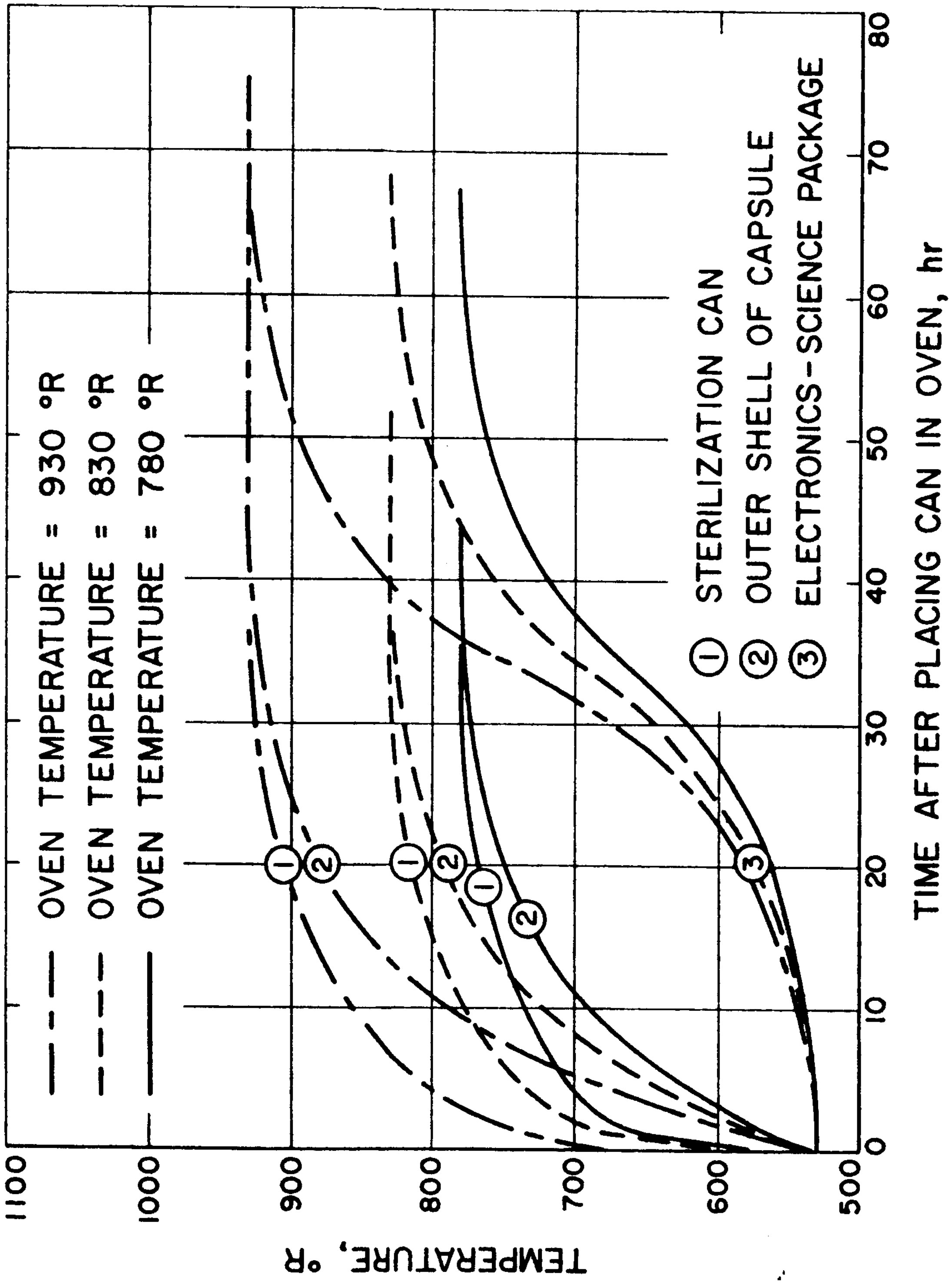


Figure 4-162. Effect of Oven Temperature on Sterilization Time for a Mars Capsule

CHAPTER 5

SPACECRAFT SYSTEM CONCEPTS

I. INTRODUCTION

Alternate technical approaches to meeting the functional requirements of a planetary orbiter/split-capsule mission have been outlined in detail in Section IV of this report. Various techniques have been examined from the point of view of resulting subsystem capabilities and interactions. In this section a typical spacecraft system concept is synthesized to indicate possible weight requirements and performance capabilities.

Variations in spacecraft weight-in-orbit and capsule weight for various trajectory parameters were given in Chapter 4 B-2 (see especially Figures 4-10 and 4-11). For a nominal 1800 km x 10,000 km orbit, orbiter weights in the range of 1500 lb to 2000 lb with corresponding capsule weights of 2250 lb to 900 lb are possible.* If the orbit apoapsis altitude is increased from 10,000 km to 20,000 km, the orbiter weights could be increased about 150 lb for the corresponding capsule weight and a further increase in apoapsis altitude to 40,000 km would add about another 150 lb. Thus a fairly wide range of orbiter weights may be achieved by allowing both capsule weight and orbit apoapsis to vary.

For the particular system described in this section an orbiter weight of 1820 lb and two capsule weights of 1380 lb and 1725 lb are indicated. However, as the description of the system will show, the weights of the subsystems which comprise these configurations are subject to considerable variation (either upward or downward) and the system totals should not be taken too literally. They are, however, indicative of the general weight allocations to various subsystems which are to be expected.

It should be noted that cruise operations and system modifications necessary to perform interplanetary investigations have been specifically excluded from the spacecraft system discussion. (A nominal interplanetary science package might, for example, weight 40 lbs and require 15 watts of regulated power. These requirements and any accompanying configuration problems should be included in later studies if the mission objectives selected for an orbiter/split capsule system include such interplanetary investigations.

II. DETAILED SEQUENCE OF EVENTS

The operation and functioning of the spacecraft system can best be introduced by reference to a flight operational sequence. Table 5-1 outlines a general sequence of events

*The orbiter weight cited does not include the dry propulsion weight carried into orbit.

Table 5-1. Overall Flight Operational Sequence

Event Number	Event
1	Lift-off AMR
2	Spacecraft injection
3	Spacecraft separation
4	Unfold solar panels
5	Initiate attitude acquisition a) Extend antenna b) Start automatic Sun acquisition
6	Sun acquisition complete, start ROLL search
7	Canopus acquisition complete
8	High-gain antenna pointed to Earth
9	Initiate cruise measurements
10	Complete midcourse tracking and correction computation; transmit trajectory correction commands. a) ROLL turn polarity and duration b) PITCH turn polarity and duration c) Velocity increment
11	Initiate midcourse maneuver sequence
12	Stop ROLL turn
13	Stop PITCH turn
14	Command motor shut-off
15	Commence automatic reacquisition
16	Sun acquisition complete
17	Canopus acquisition complete (If second midcourse is desired, the same procedure, steps 11-17, would be repeated)
18	Complete capsule separation maneuver and approach maneuver tracking and computation, based on RF and/or optical data.
19	Transmit separation maneuver commands a) ROLL turn polarity and duration b) PITCH turn polarity and duration c) Velocity increment
20	Initiate capsule separation sequence
21	Stop ROLL turn
22	Stop PITCH turn
23	Capsule separation
24	Monitor capsule separation events via spacecraft bus
25	Commence automatic reacquisition
26	Sun acquisition complete
27	Canopus acquisition complete

Table 5-1. (cont)

Event Number	Event
28	Transmit approach maneuver commands a) ROLL turn polarity and duration b) PITCH turn polarity and duration c) Velocity increment
29	Initiate approach maneuver sequence
30	Stop ROLL turn
31	Stop PITCH turn
32	Command motor shut-off
33	Commence automatic reacquisition
34	Sun acquisition complete
35	Canopus acquisition complete (If second approach maneuver is desired, the same procedure, steps 28-35, would be repeated)
36	Complete terminal maneuver tracking and computation; transmit terminal maneuver commands. a) ROLL turn polarity and duration b) PITCH turn polarity and duration c) Velocity increment
37	Monitor and relay capsule entry via spacecraft bus
38	Initiate spacecraft retro-maneuver sequence
39	Retract solar panels prior to retro-maneuver
40	Retract antennas
41	Stop ROLL turn
42	Stop PITCH turn
43	Initiate retro-maneuver using main retro
44	Terminal retro-maneuver
45	Main retro jettisoned
46	Solar panels extended
47	Commence automatic reacquisition
48	Sun acquisition complete
49	Canopus acquisition complete
50	Antennas extended
51	High-gain antenna pointed toward Earth
52	Planetary platform extended
53	Horizon scanners activated
54	Commence orbiter experiments
55	Commence relay of capsule landing experiments via orbiter

for the bus/orbiter part of the mission. Except for the addition of a capsule separation maneuver (Events 18-27) and relay of capsule data via the orbiter (Event 55), the operation is similar to that described in detail in Volume II, Section V.

The acquisition and cruise phases are almost familiar ones by now, following closely the pattern of Mariner 2, Ranger, and Mariner C spacecraft. Celestial objects are used for attitude reference sources. Trajectory correction maneuvers are commanded from Earth and make use of inertial sensors for relatively short-term operation. Upon completion of maneuvers, the cruise phase is resumed.

Planetary approach and subsequent phases are the really new functions peculiar to this particular mission. A detailed sequence of capsule-related events (Table 5-2) and a functional block diagram (Figure 5-1) provide a compact picture of the functional execution of the Mars 1969 landing capsule mission, as it is presently envisioned.

The sequence of events seeks to define the logical step-by-step evolution of the mission, and to determine the necessary control functions. The timing of the events is approximate and will depend on the choice of trajectory and separation philosophy. Timing on the surface is based on a Δt (time between capsule planetary entry and bus closest approach) of 2 hours.

The block diagram is simply a graphical statement of the functional sub-division of the capsule system necessary to implement the sequence of events and should not be confused with one showing the actual subsystems.

The material is based on a ballistic entry and parachute descent through the presently assumed Martian atmosphere, and would require revision to conform to other entry and descent schemes that might be necessary in the presence of significantly lower density atmospheres.

III. SYSTEM CONFIGURATION AND WEIGHT SUMMARIES

The spacecraft system is made up of two basic modules, the orbiter/bus and the landing capsule. The landing capsule can be further partitioned by defining a "net electronic payload" which is the total active payload weight landed on the Martian surface. (The net electronic payload does not include the internal structure and energy absorption material which is also landed.) Three weight summaries are shown here in Tables 5-3 through 5-5:

1) The orbiter, which is the weight in orbit, less the propulsion system weight.
2) The net electronic payload for two alternate communication capabilities.
3) An overall capsule weight summary corresponding to the two alternate net electronic payloads.

Table 5-2. Detailed Sequence of Events - Mars 1969 Capsule

Event Number	Mission Time	Event	Control Source	Data* Mode	Remarks
1	E-70hr	Activate and set pre-entry sequencer	Ground Command	S/C	Time-delay setting will depend on trajectory parameters and capsule status
2	E-65hr	Start separation sequence a) turn on capsule Separation Maneuver Sequencer b) turn on capsule internal timer power, transmitter, receiver (Option II) c) turn on bus relay receiver d) perform communications lock-up e) telemeter capsule status f) turn on bus Separation Maneuver Sequencer	Ground Command	S/C	Capsule operating on external (bus) power
3	E-63hr	Complete capsule check-out	Bus Separation Maneuver Sequencer	S/C	Capsule operating on internal power
4	E-62hr	a) establish go/no-go for capsule status b) start bus separation maneuver	Ground Command Bus Separation Maneuver Sequencer	S/C	
5	E-60.1hr	Pre-separation events a) arm capsule release b) activate capsule Separation Maneuver Sequencer c) arm capsule spin-up system	Bus Separation Maneuver Sequencer	S/C	
6	E-60hr	a) initiate bus-capsule separation ans spin-up b) arm capsule propulsion	Capsule Separation Maneuver Sequencer	I	Capsule status and separation events transmitted
7	E-59.5hr	Propulsion maneuver	Capsule Separation Maneuver Sequencer	I	
8	E-59.4hr	a) jettison propulsion and spin-up systems b) initiate de-spin system	Capsule Separation Maneuver Sequencer	I	

Table 5-2. (cont)

Event Number	Mission Time	Event	Control Source	Date* Mode	Remarks
9	E-59hr	Turn off capsule subsystems	Capsule Separation Maneuver Sequencer		Only pre-entry sequencer remains on
10	E-2hr	Pre-entry events			
		a) turn on capsule transmitter, receiver (Option II) b) perform communications lock-up c) telemeter capsule status d) activate capsule recorders e) activate entry sensors f) arm G-switch	Pre-entry sequencer	I	Capsule status and pre-entry events are transmitted
11	E	Entry	G-switch		Record all entry parameters during "blackout"
12	D	Jettison back cover Deploy hypervelocity chute Jettison heat-shield	Entry Sensors		Deployment occurs at Mach 3. Dynamic pressure of 500 lb/ft^2 and temperature of 700°F sensed by accelerometer and thermocouple probe
13	D+1~2 sec	Deploy reefed main chute Start main chute timer	Entry sensors		Occurs at mach 0.9; dynamic pressure 50 lb/ft^2
14	D+9~10 sec	Unreef main chute Start descent sequencer Perform communications lock-up	Main chute timer	II	
15	D+15~25 sec	Deploy atmospheric and engineering instruments Activate impact sensor	Descent Sequencer	II	Start transmission of stored plus current data
16	D to L (7 to 40 min)	Atmospheric and engineering data collection	Descent Sequencer	II	
17	L	Capsule Impact a) parachute release b) start lander sequencer	Impact Sensor		
18	L+5 min	Capsule erection	Lander sequencer		

Table 5-2. (cont)

Event Number	Mission Time	Event	Control Source	Data* Mode	Remarks
19	L+10~50 min	Deploy surface instruments	Lander sequencer	III	Relay link transmitter on higher bit-rate (surface instrument deployment may require sequencing)
20	L+45~78 min	a) turn off relay link transmitter b) continue collection and storage of data c) turn on capsule receiver d) deploy Sun-activated Direct-link sequencer	Lander sequencer		Bus begins retro-maneuver, direct-link transmitter continues
21		Surface Operation a) direct-link transmitter turned on as direct-link sequencer detects Sun above horizon. Turned off 6hr later; sequence repeats for the duration of the mission b) relay link	Direct-link sequencer	III	Transmits 3 bits/sec directly to Earth for 6hr of every rotation of Mars
		Option I operates continuously for the duration of the mission at constant information rate		III	Transmits one block of data repeatedly until reception by orbiter is assured, then begins transmitting new block, etc.
		Option II turned on by VHF beacon on bus, turned off by loss of VHF beacon; data is stored for transmission during next pass; sequence repeated for duration of the mission	Orbiter beacon through capsule receiver		

*Explanation of Data Modes

S/C - data transmitted directly through bus umbilical
I - relay link transmits real time capsule status telemetry at low bit rate (100 bps)
II - relay link transmits real time data time-shared with recorded data (low bit rate)
III - relay-link transmits at higher rate (700 bps)
direct-link transmits at 3 bps.

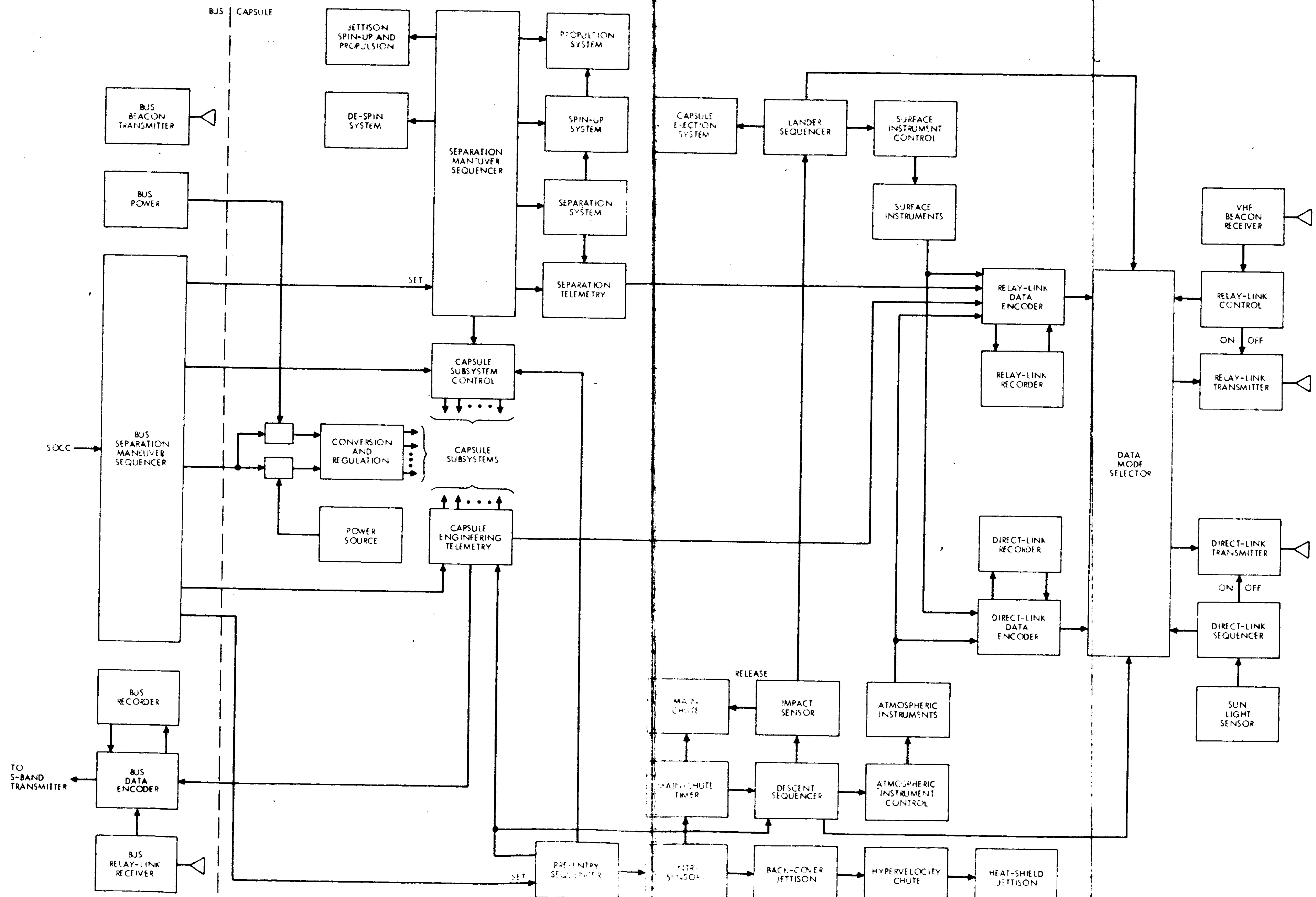


Figure 5-1. Conceptual Block Diagram
1969 -- Mars Capsule

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Table 5-3. Typical Orbiter Weight Summary

Function	Weight in Pounds	
Power Supply (solar photovoltaic)		388
Panels	283	
Converters	34	
Midcourse Maneuver Battery	45	
Shadow Operation Battery	26	
RF System (25 watt S-band with 12.5 ft. diameter antenna)		195
Data Handling		112
CC&S		25
Guidance and Attitude Control (nitrogen system)		348
Space Science		257
Structure		275
Bus	200	
Planetary Horizontal Platform	75	
Cabling		150
Thermal Control		60
Pyrotechnics		10
TOTAL ORBITER WEIGHT less propulsion system		1820

Note: 1. Nominal orbit altitude 1800 km x 10,000 km.
2. 150-day orbit lifetime.
3. No weight contingency included.

These weights do not constitute a recommendation for a final configuration; however, they are typical and indicate the general weight allocation to various subsystem.

For detailed descriptions of the subsystems listed in these weight summaries the reader is referred to the specific technical sections of Chapter 4 of this volume. However, certain brief descriptive comments about the configuration corresponding to the weight summaries are made in the following paragraphs.

A. The Orbiter

Table 5-3 shows a weight summary for a solar-powered orbiter/bus, as shown earlier in Figures 4-139 and 4-140. The power subsystem is sized for a worst-case situation; i.e., the orbiter nominally enters an 1800 km to 10,000 km altitude orbit which remains in the sunlight for at least the first sixty days of operation. The power system here is capable of operation in a close 1500 km circular equatorial orbit, provided that the 25w final amplifier of the RF system is turned off whenever Sun shadow is experienced.

Table 5-4. Typical Lander Net Electronic Payload Weight Summary

System	Communication Option I*	Communication Option II
RF System		
Direct	15	15
Relay	24	36
Data Handling		
Direct	24	24
Relay	55	43
CC&S	7	7
Thermal Control	20	20
Cabling	20	20
RTG Power System		
RTG	342	220
Battery	40	40
Conversion	20	20
Space Science	121	121
Net Electronic Payload	688 lb	566 lb

*Both communication options are capable of communication relay from the lander through the orbiter to Earth. However, Option I operates with the relay link on 24 hours per day, whereas Option II utilizes a command link from the orbiter to turn on the lander relay link when line-of-sight is established.

The RF subsystem consists of two parallel S-band transmitters and receivers, either of which may be turned on and off by Earth-command. The power output of the final RF stages is 25w. The antenna system consists of two omni-directional antennas plus a 12.5 ft parabolic antenna with the requisite feeds and equipment for simultaneous lobing operation.

The data handling subsystem provides both engineering and scientific data accumulation, storage and distribution in proper format.

The central Sequencer and Computer (CC&S) subsystem is described in detail in Section V.C.4 of EPD-139, Volume II, under the more general function, Sequencing, Operations Control, and Computing (SOCC).

The guidance and attitude control subsystem includes the following sensors:

1) Sun sensors
2) Canopus tracker
3) Horizon scanner

H 1?

Table 5-5. Typical Capsule Weight Summary

	Communication Option I	Communication Option II
Net Electronic Payload	688	566
Structure	230	188
Aft End	22	17
Lander Weight*	940	771
Parachute	128	105
Shell/Aft Cover Release	6	6
Weight at Parachute Deployment	1074	882
Avco Heat Shield**	292	240
Weight at Entry	1366	1122
Despin Mechanism	18	15
Weight at Propulsion Burnout	1384	1137
Propulsion (300 m/sec)	266	220
Spinup System	45	38
Release Clamp	5	5
Capsule Gross Weight	1700 lb	1400 lb

*The lander is designed to have a terminal velocity of approximately 40 ft/sec. The landed capsule uses omni-directional antennas for both direct communication to Earth (S-band) and for relay communication to the orbiter (VHF).

**"Project Voyager Status Report" RAD Division, AVCO Corporation, Technical Memorandum RAD-TM-63-17, March 25, 1963.

4) Gyros and accelerometer modules
5) Gyro compass and electronics

In addition, the actuators for the autopilot (4), antenna (2), and planetary horizontal platform (3), are included in this subsystem. The bulk of the 348 lb shown is for the attitude control system which uses nitrogen gas. The weight assumed here is an "average" one; earlier calculations in Chapter 4 B-1-4 indicated that this weight might be substantially reduced by clever design, balancing of solar torques, etc.; it was also pointed out that the weight required for a cold gas system might increase substantially (by over several hundred lb) if conservative design were employed to assure full 150-day operation in a Martian orbit.

The one remaining figure which may require explanation is the 150 lb allowance for cabling. While this may seem excessive, experience with other spacecraft systems (e.g., Ranger, Mariner and Surveyor) has indicated that total cabling weight, including connectors, can run as high as 7 to 10 percent of the total system weight.

B. The Net Electronic Payload

Two capsule net electronic payload weights are itemized in Table 5-4, corresponding to alternate communication options. Both communication options are capable of communication relay from the lander to the orbiter to Earth. However, Option I operates with the relay link (25w VHF) on 24 hr/day, whereas Option II utilized a command link from the orbiter to turn on the lander relay link only when line-of-sight is established between orbiter and lander. Both Option I and Option II operate their direct Earth links (50w S-band) continuously.

Option I is the simplest and, presumably, most reliable mode of operation: both communication links are turned on and left on. This mode is also most wasteful of power as indicated by the weight summary.

Option II, in which the relay link transmits on command (from the orbiter) for an average of 2.5 hr/day, provides a considerable power saving; the relay link data-rate is also more efficient, since no data need be transmitted more than once; however, the reliability of the relay link is somewhat reduced.

A third communication Option (not illustrated in Figure 5-1) would cycle the direct link as well as the relay link; this would result in even greater power savings, but again reliability would probably be degraded. One mechanization proposed would use a Sun sensor to turn on the direct link whenever the Sun were visible. For the first 60 days after landing on the Martian equator, this would occur daily about 2 to 3 hr after the Earth rose above the capsule's horizon (30° to 45° elevation angle). A timer would turn off the direct link 6 hr later. It is estimated that this communication option could reduce the net electronic payload weight to about 500 lb.

C. The Capsule

The capsule weight summary shown in Table 5-5 represents the total weight which is separated from orbiter/bus at the time of the capsule separation maneuver. Two net electronic payload weights corresponding to the two communication options of Table 5-4 are considered.

The structure weight shown includes the energy absorbtion mechanism and/or material necessary to accommodate landings up to 40 ft/sec horizontal and vertical velocity, as well as the basic capsule and payload structure itself. The landed capsule uses omni-directional antennas both for direct communication to Earth and for relay communication to the orbiter. In the interests of reliability, no attempt has been made to deploy or orient a high-gain antenna.

The heat shield weight assumed here is based upon recent studies by AVCO[1] and includes both ablation and thermal insulation requirements.

The spinup, propulsion, and despin mechanization is based on techniques discussed previously in Chapter 4 B-12-a of this volume. Cold gas is used for the spinup, and a yo-yo despin mechanism is employed prior to atmospheric entry. Allowing for separation velocity capability of 300 m/sec is quite conservative and provides considerable flexibility in the choice of separation philosophy and a possible weight contingency in the estimates given.

If the communication Option III described in the preceeding section were employed with a separation velocity capability of only 150 m/sec, the capsule gross weight might be reduced to as little as 1150 lb. However, the gross weight range of 1400 lb to 1700 lb shown in the table would appear to be a more reasonable indication of what is required for reliable capsule operation.

IV. MISSION CAPABILITY

The system configurations described here for a 1969 Mars mission represent a significant step beyond the capability of any foreseeable Mariner system. With a gross capsule weight range of 1400 lb to 1700 lb it appears possible to land a scientific instrument payload of approximately 120 lb on the Martian surface with a few hundred kilometer dispersion. This payload can be operated for many months sending data directly back to Earth at about 3 bit/sec rate (about 8 to 10 hr/day); by using an orbiter as a communication relay when it is in view (about 2.5 hr/day average), this data rate can be increased to about 3000 bit/sec. The lifetime of the lander is limited only by the half-life of the radio-isotope used for power generation.

The system described can place a spacecraft in excess of 1800 lb into a Martian orbit of 1800 km x 10,000 km altitude. Initial periapsis dispersions will be on the order of a few hundred kilometers, and subsequent orbital trim is possible. A useful scientific instrument weight of approximately 250 lb appears possible with an operating life in orbit of up to

150 days. Data from the orbiter (whether it originates in the lander or in the orbiter) can be sent to Earth at a rate of 6000 bit/sec at encounter and 4000 bit/sec 150 days later.

While it is possible to "take-apart" many of the subsystems described here and to show how performance can be increased for the same weight (or weight reduced while maintaining the same capability), it is felt that reasonable conservative design would dictate otherwise. Undoubtedly, many of the subsystems described herein will be modified as the design process evolves; however, the composite picture of the system and its performance appears reasonable and capable of achievement.

REFERENCES

1. "Project Voyager Status Report," RAD Division, AVCO Corp., Technical Memorandum RAD-TM-63-17, March 25, 1963.

CHAPTER 6
MAJOR PROBLEM AREAS

CHAPTER 6

MAJOR PROBLEM AREAS

I. INTRODUCTION

Many technical problems have been discussed in considerable detail in previous sections of this report. Some of these problems are related to specific system configurations, mission geometrics, and more detailed requirements. Other problems are of a more general nature, being common to all missions of the class considered. The purpose of this section is to single out, and in some cases describe in more detail, those major technical problems which require long lead-time for their analysis and solution; these are problems for which early effort must be expended if the development of a Voyager-class system is to proceed at an orderly pace with a high degree of confidence in its ultimate reliability.

II. STERILIZATION

The sterilization problems, from an engineering viewpoint, have been adequately considered and are described in previous JPL studies.(1) It may be advisable, however, to point out the major problem areas related to sterilization requirements and procedures.

Since absolute sterility of a material cannot be demonstrated with present test procedures, the condition of sterility actually becomes a probability statement in that, given sufficient experience and testing, it can be said that a specific sterilization procedure has a certain probability to produce sterility when properly applied. For Mars landers and orbiters, a probability level of 10^{-4} has been established which means that the chances of a single spacecraft being non-sterile are no greater than 1 in 10,000. Thus, one of the greatest problem areas from the sterility viewpoint is the establishment of proper conditions so that the sterilization procedure can attain the desired probability level for sterility. This requires proper design of sterilization procedure to function adequately. For example, if heat is the sterilization method, design engineering must consider the adequate distribution of heat to all parts of the spacecraft.

A second large problem area concerns the ability to demonstrate that sterility or a specific probability for sterility has been attained. Based upon the sterility test procedures being used today, the degree of sterility desired could only be determined after a great number of tests and the acquisition of much experience. It has been calculated that approximately 30,000 tests would be required to establish the 10^{-4} level of probability. More reliable standardized test procedures must be developed in order to establish sterility probability more simply and reliably.

From the hardware viewpoint, the greatest problem area concerns the compatibility of spacecraft materials with the sterilization procedure and the functional reliability of the materials or components after being exposed to the sterilization process. Many materials used in spacecraft construction are degraded by the various sterilization procedures, and heat is no exception, since the thermal procedure constitutes an artificial aging condition. Of course, compatibility reflects directly upon the ability of the material or component that has been processed to function properly at a satisfactory reliability level. Reliability testing of sterilized components must include life testing and not just functional testing after processing, since components on planetary spacecraft are expected to function either during or after long periods of space flight. At the present time, there are many questionable areas of compatibility and there has been only a modest beginning in a reliability test program based upon heat sterilization treatment.

Another major problem area concerns the maintenance of sterility after the final assembled spacecraft has been sterilized. This area involves the many various handling procedures for the spacecraft from the time it enters the terminal sterilization procedure until it is beyond the Earth's atmosphere after launch. Thus, very precise, monitored handling procedures must be developed and rigorously executed if sterility is to be maintained.

The above four areas represent the major problem areas, and it will be readily recognized that there are and will be many so-called minor problems which are actually a part of one or more of the major areas.

III. RELIABILITY

Because Voyager is a much more complex spacecraft and has greater mission capability than the Mariner series, its reliability problems will tend to be correspondingly greater. Recognition of the increased mission complexity and of the higher launch costs dictates early attention to all the trade-offs which include reliability as one of the parameters. These trade-offs should be examined increasingly as the conceptual studies progress and as preliminary design is begun.

The purpose of this section is to suggest certain guidelines and approaches to the Voyager reliability problem. The solutions will come only through the dedicated efforts of all concerned with the project planning, design, test, and operation of the Voyager mission.

A. Project Planning

1. Mission Objectives

The magnitude of the Voyager reliability problems can be reduced through mission planning as well as through sound design techniques. By holding spacecraft complexity to levels justified by growth in spacecraft subsystem reliability, early missions will tend to have simpler goals than later missions. Thus mission planning can be used, for example, to ensure that science experiments are scheduled so that only tested and proven subsystems are flown at each successive opportunity.

For each experiment, at each opportunity, acceptable reliability of the subsystems should be either:

1) Alreadv demonstrated
2) Capable of attainment through such means as redundancy and derating
3) Insured through substitution of an alternate experiment.

By planning missions on the basis of our confidence in the science subsystems available for each planetary opportunity, we attain Voyager goals in an economical fashion. Thus, the sequence of planetary data acquisition over the life of Voyager is biased by reliability considerations, but otherwise is ordered in accordance with the most logical pattern for scientific exploration.

2. Scheduling

Reliability planning is a necessary and vital part of project planning during the conceptual and preliminary design phases of a mission. In these phases major decisions and the primary system trade-offs are made. Here, also, is where the basis for all reliability measures during the detailed design phase originates.

Reliability practices, bad as well as good, will perpetuate themselves from these initial phases. Consequently, it is mandatory that sufficient time be alloted to the conceptual and preliminary design phases to insure that reliability goals and decisions are sound. Equally important is the fact that later phases of design should not suffer from lack of sufficient time allocations to implement the established reliability goals. Time must be allowed for completing the interative feedback of test results into the detail design process.

B. Design

One of the greatest contributions to reliability that can be made by designers is to avoid "exotic" designs. Exotic design is one which emphasizes elegance over excellence. This requires a self-discipline which puts project goals ahead of individual goals. The project viewpoint favors getting some data with high probability rather than complete data with low probability.

Exotic designs can be avoided by acceptance of the design constraint that subsystem functions be performed using components drawn from a set restricted as to the number of different types, all of which are acceptable as to their inherent reliability. Again, this constraint on design requires self-discipline in setting mission objectives and in designing to meet the objectives.

C. Manufacturing

The creation of a necessarily limited set of reliable parts requires that the manufacturing process be under control that will give a high degree of statistical significance to the observed quality of the parts. The manufacturing process, however, must be a part of a bigger loop so that test data at all levels, up through the system level, are fed back in as direct a fashion as possible. Thus, the manufacturer of a resistor, for example, must feel that he is participating in the construction of a spacecraft and not just producing resistors.

D. Operational Concepts

The traditional approaches to reliability enhancement include derating, redundancy, and marginal checking coupled with simplicity in design and quality workmanship. In addition, however, operational concepts such as hibernation (wherein the spacecraft systems are kept in as unstressed a condition as possible during the cruise phase) should be considered. Not enough experience with this technique has accrued so that it is fully accepted and can be applied with confidence.

Hibernation as an operational technique, therefore, requires further study and experimental evaluation. The quantitative increase in reliability attributable to this technique when applied over interplanetary flight times should be determined as part of a program to insure economical solutions to Voyager reliability problems.

E. Test Programs

Testing programs imply the existence of test criteria, usually related to reliability and functional operation, or both. Test results, without direct feed-back paths to the spacecraft design process, will only indicate parts of the system to "worry about," thus limiting their usefulness. It follows that project planning schedules must make adequate provisions for changes in the spacecraft design during the testing program.

The launch costs associated with two Saturn launches per Voyager opportunity are considerable. Thus economic considerations alone justify expenditures for early Voyager subsystem tests under space environments. The need for the scientific data which justifies Voyager missions, together with the infrequency of the planetary opportunities, heavily reinforce the logic of advance testing of prototype subsystems.

F. Sterilization

Sterilization of the lander is a firm requirement at this time. Experience to date has shown that heat sterilization is the preferred method from the standpoint of biological effectiveness by providing highest sterilization reliability. Unfortunately, however, minimum sterilization temperatures of 135° C can degrade electronic and electro-optical equipment. Thus, a very difficult problem arises in determining the best trade-off between increased sterilization reliability and decreased capsule reliability.

This problem is widely recognized and will be under continuous study. For a battery powered capsule, the isolation of the capsule as the only sterile assembly allows favorable mechanical packaging and handling so that complications in spacecraft and launch vehicle are a minimum. The problem area, therefore, tends to become one that can be resolved through research and development aimed at:

1) Increasing the temperature tolerance of capsule components
2) Developing alternate sterilization techniques
3) Devising capsule detail design approaches that are most favorable to meeting the reliability compromise between capsule life expectancy and sterility.

IV. LAUNCH VEHICLE OPERATIONS AND INTERFACES

A. Launch Operations

1. Launch Pad Allocations and Availability

Assuming a requirement for two launches per opportunity and a firing period of about 30 days, it is apparent that two launch pads are necessary. Alternative solutions to this problem using AMR pads 34 and 37 separately and jointly are being investigated by MSFC.

Data necessary to complete this study include the following:

1) Injection energy variation over the firing period for orbiter/lander mission. (not the same as for flyby missions)
2) Launch vehicle assembly and checkout times.
3) Pad refurbishment times, as applicable
4) Effect of launch azimuth constraints on 1) above
5) Constraints on launch time of second spacecraft arising from DSIF and clumping of the two spacecraft at the target planet
6) Countdown time, and hold time limits on the pad

2. AMR Tracking and Telemetry Coverage

Over the anticipated life span of the Voyager mission, a wide variation occurs in the declination of the outgoing geocentric asymptote and the corresponding launch azimuths. Although range safety considerations may largely govern the permissible extremes in launch azimuth, the configuration of AMR tracking and telemetry installations may add additional launch azimuth constraints. The problem here is to determine the probable additions to the tracking and telemetry net by the date of the first operational Voyager mission in 1969.

3. AMR Range Safety

The price for constraining launch azimuth is higher geocentric injection energy, C_3. Because the launch vehicle capability is limited, so is the extent to which launch azimuth constraints can be compensated by higher injection energy. The optimum declinations for some of the launch opportunities correspond to launch azimuth less than 50 or greater than 130 degrees. Such azimuths far exceed the limits reasonably expected to be granted by waiver. The problem is to clearly portray the trade-offs involved and begin early discussions with AMR in order to minimize the degradation in payload capability resulting from launch azimuth constraints.

An equally important objective is to obtain the firmest possible ground rules from AMR such that spacecraft redesign can be held to a minimum. Range safety considerations will affect the design of a radioisotope-powered spacecraft. If intact radioisotope recovery is required, then the spacecraft destruct system must not rupture the radioisotope case. If a parachute system is used for the radioisotope system recovery, this problem becomes more acute.

The two spacecraft configurations under study, solar powered and isotope powered, may present different problems of range safety. It is possible that more optimal azimuths would be granted for solar powered spacecraft. A determination of the AMR waiver limits is desirable so that this factor can be introduced into the process of selecting the Voyager spacecraft configuration.

B. Launch Vehicle

1. Plane Change to Gain Greater Declinations

Among the possible means for reaching declinations of the outgoing geocentric asymptote which lie beyond the launch azimuth limits is a dog-leg maneuver by the launch vehicle. Past studies of this problem, particularly in reference to obtaining synchronous equitorial orbits from AMR, have shown a rapid degradation in booster payload performance as the plane change increases beyond a few degrees.

MSFC has been asked to estimate plane change losses for the Saturn S-I/SV vehicle as part of a comprehensive evaluation of various methods for overcoming the adverse effects of restrictions on launch azimuth.

2. Vehicle Stability at Launch from Parking Orbit

For third stages having very low fineness ratios (length to diameter), it is important that the spacecraft center of gravity be located as far forward as possible. This provides greater stability and reduces the severity of pitch or yaw motions which occur at ignition.

The transient motion is a result of differences in the thrust-time history of the two engines, which are located off the longitudinal axis of the stage. Further study is required to determine:

1) How high the spacecraft center of gravity can be raised
2) What the third stage gains for the corresponding compromise to the spacecraft

Coincident with this study is the resolution of mechanical interface problems between the spacecraft and the third stage of the launch vehicle. This item is discussed below.

3. Mechanical Interface

The presence of the Saturn instrument unit between the spacecraft and the third stage of the launch vehicle imposes the current requirement that if spacecraft loads are transmitted through the IU, they must be uniformly distributed as seen by the IU. The adapter design, in addition, may be strongly affected by the size and shape of the lander package, particularly if the lander lies inside the adapter and below the spacecraft separation plane.

Among alternate solutions to be examined is an adapter configuration in which struts extend through the interior of the instrument unit and attach to a circumferential ring near the top of the third stage.

The dynamics of shroud separation are an intimate part of the mechanical interface. The shroud must separate and jettison without damage to the spacecraft; heat leakage from the shroud into the cryogenic heat sink of the upperstage must not induce malfunction from freezing of moisture; also, thermal stresses in the adapter from this source must be accommodated or suppressed.

4. Third Stage Design Point

The currently used design point for the third stage is a geocentric injection energy, C_3, of 22 km^2/sec^2. This value can be confirmed only when trade-off studies, such as discussed earlier, are complete. Many of the launch constraints can be compensated by increasing C_3. Thus, the stage design point of 22 km^2/sec^2 should be considered tentative, pending confirmation.

The primary factors involved in the trade-off with C_3 are firing period, launch azimuth, and flight time. Uncertainties in launch azimuth constraints and in spacecraft reliability should be reduced before the third stage design point C_3 of 22 km^2/sec^2 can be considered as the final JPL recommendation to MSFC.

C. Spacecraft Interface with the Launch Environment

1. Vibration, Shock, and Acoustic Excitation

These environmental factors will require increasing attention due to the more severe environment posed by Saturn compared to Atlas. Also, the spacecraft weight and antenna diameter are considerably greater than for previous spacecraft.

The vibration, acoustic, and shock specification for the Saturn I has been obtained from MSFC. It includes environmental data applicable to the payload area.

2. Air Conditioning in the Launch Phase

The heat load to be removed from the spacecraft is higher for a radioisotope-powered configuration than for a solar-powered configuration. If a radioisotope system is used, the heat output must be removed continuously from the time the unit is first assembled.

No problem is anticipated for air conditioning to remove radioisotope generated heat while the spacecraft is on the pad. Further study is required to determine the method for heat rejection during that part of the launch phase prior to ejection of the nose fairing. The solutions to be investigated should include the use of the coolant system of the Saturn instrument unit.

3. Destruct System

The retro propulsion of the spacecraft will be designed for de-activation during the ascent phase, and possibly during coast in parking orbit depending on AMR detailed requirements applicable to Voyager. Among the problem areas are:

1) Eliminating or minimizing payload loss attributable to provisions for destructing the retro propulsion system.
2) For radioisotope system configurations, insuring against damage to the radioisotope unit if intact recovery is required rather than dispersion in the upper atmosphere.

4. Spacecraft Telemetry

No firm design criterion has yet been established for spacecraft telemetry during ascent or parking orbit periods. However, further study may indicate the need for temperature measurements for radioisotope system configurations. The critical thermal phase of flight would occur prior to shroud ejection because of internal generation of heat from a radioisotope source coupled with external heat of aerodynamic origin.

V. MARS ENVIRONMENT

Uncertainties in knowledge of the Martian atmosphere, meteorology, and surface environment place a severe hardship on the design of any survivable landing capsule. The effects of these uncertainties will be discussed in turn:

A. Atmospheric Model

At the outset of this study it was assumed that the limiting model atmospheres described by G. F. Schilling(2) were a satisfactory representation upon which capsule entry design could be based. More recent observations (3) have indicated that the surface pressure is much lower than that originally assumed, perhaps as low as 10 mb. If the surface pressure is this low, then the design of a passive ballistic entry system becomes less efficient. This is true because the weight of an optimum heat shield structure and the weight of an optimum retardation system (parachute) increase as the atmosphere becomes less dense. (The term 'optimum' used here means "minimum weight" for a known atmospheric model.)

If the exact atmosphere is not known, so that the entry system must be capable of satisfying a wide range of possible atmospheric models, system performance is further degraded. This decrease in efficiency is due to the fact that the entry heat shield, structure, and insulation which is optimized for the lowest-density atmosphere is not capable of handling an entire range of possible models (e.g., 10 mb to 135 mb surface pressure). Thus, when gross uncertainties exist in knowledge of the atmosphere (as they do at the time of this writing) even more conservatism must be built into the entry system to accommodate all possible entry situations.

B. Meteorology

Uncertainties in the surface wind conditions on Mars make an efficient design of a landing retardation system difficult, unless one is willing to accept a high risk of failure at landing. While the previous sections of this report assumed surface winds of less than 40 ft/sec, recent estimates of extremely low surface pressures have been accompanied (in some cases) by estimates of surface winds up to several hundred ft/sec. If such high wind conditions do exist, there is little point in attempting to use parachute deceleration devices to reduce the vertical descent speed below a similar value of velocity. If these estimates are to be accepted and accounted for by conservative design, then very severe penalties on the weight of useful electronic payload placed on the surface of Mars in a survivable capsule must also be accepted. The design of the survivable capsule would have to be based on high speed impact similar to the Ranger rough landing capsule. To appreciate the severity of this problem, it is necessary to realize that a payload landing at 200 ft/sec with the same deceleration stroke as one landing at 40 ft/sec experiences a G-load twenty-five times as great. Thus, a payload landing at 40 ft/sec with a uniform deceleration stroke of 6 inches would have a loading of 50 G's while the same payload landing at 200 ft/sec would experience 1250 G's. If such severe surface winds are shown to exist or, at least, are considered likely, it may well be necessary to turn to an active controlled vernier propulsion descent as is presently being developed for the Surveyor lunar landing system.

C. Surface Environment

Lack of any knowledge of surface terrain conditions creates a major problem for lander design. In all likelihood, certain parts of any landed payload will require erection of some sort. It may only be a simple omni-antenna that requires orientation to (or near) the local vertical; it may be that certain experiments will require specific orientation. In any case, it is clear that a great deal of conservatism coupled with design ingenuity will be required to permit satisfactory surface operation over the widest possible range of uncertainties in surface terrain.

VI. SPACECRAFT SUBSYSTEMS

A. Electrical Power

1. Lightweight Photovoltaic Structures

In photovoltaic systems it will be required to develop large lightweight support structures, capable of withstanding accelerations, due to maneuvers for trajectory correction and orbit injection maneuvers.

2. Thermionic Diode Lifetime

For solar thermionic system, which may be expected to be in contention, some further advances in diode technology can be expected; however, before trusting a one year mission to such a system much more will have to be known about the behavior of specular surfaces of solar concentrators in a space environment.

3. Longlife Batteries

Batteries will require some additional work to be able to withstand long periods of inactive storage without loss of activity.

4. Radioisotope Production

The development of facilities for the production of radioisotopes for radioisotope power systems will have to be expanded and completed several years in advance of their expected use.

5. Radiation Effects

Radioisotope power systems are to be considered and further studies should be made to investigate the radiation tolerance of space science experiment instrumentation. Perhaps even a grouping of experiments sensitive to radiation could go on a solar powered spacecraft and the bulk of long lasting experiments, more tolerant to radiation, would go on a radioisotope powered spacecraft.

6. High-power Radioisotope Generators

Since only low power level radioisotope power generators have reached the hardware stage, units of 100 w to perhaps 500 w should be built and tested to verify the results of parametric studies in this area.

7. Voltage Converters

High efficiency voltage converters to raise the low-voltage power-outputs of thermionic systems to useable levels should also receive attention.

B. Guidance and Control

1. Attitude Control

As pointed out in Chapter 4 B. 8. b of this Volume and 6. 8(7) and 6. 9. b of Volume II, the current research programs on hot-gas actuators for attitude control do not indicate the certainty of having available hardware for the mission under consideration. As a result, a thorough investigation should be made of the gas requirements for the cold-gas system.

Unbalanced torques due to solar pressure represent a major contributor to fuel usage; therefore, the accuracy of the unbalanced torques becomes quite important. At the present time, no techniques are known to precisely estimate the forces on a spacecraft due to solar pressure. Present methods of controlling solar pressure torques consist of making conservative estimates of the torques expected and sizing control vanes accordingly. On the Voyager spacecraft, however, the control vanes become quite large and can easily obstruct the fields of view of sensors and scientific instruments. It is therefore necessary to search for better techniques of estimating solar pressure forces and for new methods of controlling them.

2. Antenna Pointing Control Systems

The investigation of possible schemes for pointing the high gain antenna has thus far served only to uncover some of the problems and questions that must be considered in making the final selection. It has indicated that the problem is not merely to select a suitable control system but is one in which philosophy of antenna design and of operation of the DSIF is also involved. A complete investigation is necessary in which the antenna design and the pointing control system are studied to a depth which permits tradeoff considerations to be made. This should include accuracy requirements, possible control techniques, failure mode analyses, mechanization complexity, and degree to which DSIF would be involved.

3. Study of Mission Objectives

A study of mission objectives, derived quantitative requirements, the work of achieving the individual objectives, and the relationship of these factors to the guidance requirements is required to avoid including unnecessary guidance equipment and system complexity. This study is discussed briefly in Chapter 4 B. 5. 1.

4. Guidance Accuracy Capabilities

Reliable, consistent estimates of guidance accuracy capabilities for possible system configurations must be prepared to serve as inputs to the study discussed above. An area requiring a particular amount of attention is the estimation of maneuver-execution accuracy by a spinning capsule that coasts for an appreciable period of time before rocket firing.

5. Capsule Release

The method and location on the trajectory of capsule release must be analyzed in considerably more detail.

C. Telecommunications

1. DSIF

The DSIF transmitter performs the following functions during the Voyager mission:

1) Provides transmitter signals for two-way doppler measurements.
2) Provides modulated carrier for ranging to the bus.
3) Provides a command link to the bus.
4) Provides a beacon for bus antenna automatic tracking.

The doppler and ranging functions can be provided to meet the requirements of trajectory and orbit determination. The requirements for such coverage are limited throughout the flight and can be scheduled with the DSIF existing facilities.

The command and beacon functions, however, require more continuous coverage throughout the mission and represent an operational and configuration problem to the DSIF. Should command and beacon functions be required on a daily basis throughout the entire mission, several possible configurations must be considered. Each has its own advantages and disadvantages.

a. Case A

The DSIF can offer a combination capability involving the use of the 85 ft reflector with transmitter capability and the 210 ft reflector with listen-only capability. (A similar mode of operation was typical of the Mariner R support provided by Goldstone Pioneer and Echo Sites combined.) Essentially, all transmissions are made through the 85 ft system for command, ranging, doppler and spacecraft antenna tracking reference. All telemetry, ranging and doppler reception will be accomplished by the 210 ft antenna at each of the operational sites.

The advantage of this mode of operation is that the listening capability is optimum, therefore providing the maximum capability of data rate. The disadvantage is that two antennas are involved at each site for each tracking mission. Therefore, two networks comprised of 85 ft antennas and 210 ft antennas will be involved during the life of this program. Because of the lengthy nature of the Voyager mission, the DSIF will be required to commit two networks for the operation for many months. From a reliability standpoint, there is no backup for such a complex configuration.

The only manner in which such a configuration of antennas would be reasonable is if the command and beacon functions were limited to only occasional coverage, thereby freeing the 85 ft antennas for other duty. The main DSIF functions therefore would be that of only listening by use of the large 210 ft antennas.

The current policy is to restrict the 210 ft to listen-only and not to provide a transmitter capability with such an antenna. This assumption therefore forces the need for an 85 ft transmitting combination to provide the command and beacon capability. The only remaining question is: What proportion of the time will be required to cover the Voyager mission?

b. Case B

By the inclusion of a waveguide switch to the listening capability of the 210 ft antenna, a modified transmit capability can be provided. The approach would be one of either receiving or transmitting, but not both simultaneously. (This mode would be equivalent to the planetary radar capability established at the Goldstone Venus Site.) This approach is meaningful when a long time exists between the transmission and reception of a signal to and from the spacecraft.

This approach implies that when commands are required and the antennas are switched to the transmit position, the commands are sent in a blind fashion. The receive mode is then restored and telemetry coverage continued.

The advantage of this mechanization is that it provides a simple mechanism for providing a command capability, and it has been demonstrated in the field. The disadvantages are that reception is terminated during periods of transmission. It is impractical to provide a beacon reference for spacecraft antenna tracking except on a periodic basis. It therefore requires the automatic tracking of spacecraft to have a reference for periodic correction rather than continuous correction.

The inclusion of a switch increases the system temperature some 5° Kelvin. For a nominal 30° K system temperature this would mean an 0.7 db degradation of system performance.

c. Case C

A 210 ft antenna with diplexer and transmitter can be provided. This configuration would be like that of the GSDS 85 ft station configuration. The advantage of this mode of operation would be continuous command, ranging, doppler and beacon capability. Procedurally, one antenna would be involved in a simple configuration. No switching or change of modes would be required. The increased antenna gain will give the equivalent of a 50 KW ground transmitter and 85 ft antenna.

The disadvantages of the diplexed approach are as follows:

1) The diplexer would introduce a 6° K degradation of system temperature because of insertion loss.
2) Experience has shown that wideband noise from diplexed transmitters can introduce an additional 10° K in system temperature.

This assumes that the problem of breakdown and noise burst is understood and controlled. Therefore, the above two losses total about 16° K, with a resulting loss of system performance of some 2 db.

Considering the extensive expenditure of funds to achieve the gain improvement of the 210 ft advanced antenna system, it seems wasteful to reduce performance by 2 db, when the increased performance costs nearly 2 million dollars per db to attain.

d. Conclusions

Case A provides a full capability at the cost of committing two DSIF networks to support the mission. It provides the optimum listening capability at the price of system complexity. This mode would be preferable for only a limited amount of time. The four weeks of capsule life will probably be a reasonable limit for tying up this much capability. The above statement assumes the commitment of the DSIF to a large number of other programs.

Case B offers a compromise capability in which a slight degradation of performance is accepted; this case greatly reduces complexity of the equipment required. It has a reduced capability in that command, ranging, doppler or beacon tracking are provided on a programmed basis with no listening capability during transmission.

Case C provides the simplest equipment mechanization and still preserves simultaneous transmit-receive capability at the price of 2 db degradation in system performance. For a particular mission, it still might be desirable to offer such a diplexed capability at the price of providing continuous transmit and receive capability simultaneously with the least amount of DSIF equipment being involved.

The Laboratory will continue to study this problem as it affects the Voyager communication system performance and operations development program.

2. Orbiter High Gain Antenna Pointing

The use of a relatively large diameter antenna on the orbiter requires that it be pointed toward Earth with a fairly high accuracy in order to effectively use the high gain of the antenna. The 12.5 ft diameter being considered would have a half-power beamwidth of ±1.1 deg, but to keep the pointing loss less than 0.5 db, the maximum pointing error should be no more than ±0.5 deg from the antenna electrical axis.

Pointing errors can result from the following sources:

1) Antenna boresight
2) Spacecraft orientation
3) Antenna pointing program
4) Structural

In order to effectively use the high-gain antenna, the combined pointing errors due to these sources must be less than ±0.5 deg.

A technique of reducing the effect of these errors is to point the antenna toward a continuous RF signal beamed from Earth. One method of accomplishing this pointing is by angle tracking this RF signal by a simultaneous-lobing system. If continuous angle-tracking were used, then the only antenna pointing error would be that associated with the angle-tracking system and this can usually be made small compared to the pointing requirements.

It does not seem practical from a DSIF standpoint, or desirable from a reliability standpoint, to require the use of angle tracking whenever data is to be received from the bus. A more practical approach would be to use intermittent angle-tracking to remove bias errors associated with the errors listed above and to up-date the antenna pointing program. If this is done, then only the random errors associated with those above must be kept less than ±0.5 deg. As the pointing error approached ±0.5 deg (possible due to the pointing program not matching the trajectory), the pointing error could be removed by use of angle tracking.

Thus, the time period of no angle-tracking is determined by the magnitude of the random errors and the accuracy of the pointing program to the required pointing profile.

Another requirement on the bus antenna pointing is initial acquisition of the angle tracking system. For the 12.5 ft diameter antenna, the acquisition angle would be ±1.4 deg. This means that in order to start angle tracking the antenna electrical axis must be pointed toward Earth with less than ±1.4 deg error.

At this time, it is not clear that the random errors associated with the antenna pointing can be kept less than ±0.5 deg. If they cannot, the result would be additional pointing losses and decrease in the data rate capability of the bus. It then might make more sense to reduce the diameter of the antenna to that resulting in a required pointing accuracy equal to the random errors. Again this would have the effect of reducing the bus data rate capability over that computed elsewhere in this report. Preliminary study of this general problem area was begun in FY '63 as part of the SR and AD program (tasks 5406-3346, Angle-tracking Spacecraft Antenna Techniques, and 5407-3346, Erectible Large Aperture Spacecraft Antennas). This effort will continue in FY '64.

3. Orbiter Detection of Capsule Data

The capsule-to-orbiter data link will make use of bit detection in the orbiter before modulation of the orbiter's transmitter for transmission to Earth. It would be desirable to eliminate dependence on the proper functioning of the detector in the orbiter in order to receive data on Earth.

The simplest relay link would consist of recovering the capsule data subcarrier from the receiver, filtering, and then using this signal to modulate the orbiter transmitter. The problems to be solved before adopting the feed-thru relay system are:

1) Determine the overall link degradation due to modulating the orbiter's transmitter with both signal and noise.
2) Determine ways of avoiding the changes in modulation index of the orbiter's transmitter due to the presence of a limiter in the orbiter, or determine ways to avoid use of a limiter in the orbiter.

Preliminary study of the general capsule-to-orbiter relay problem was begun in FY '63 under SR & AD Task 5421-3341, Telemetry Capsule Relay Techniques. It is presently planned to continue this effort, and include study of feed-thru techniques, in FY '64.

4. Orbiter Relay Link Receiver

The VHF relay link will require a receiver in the orbiter. Since the capsule-orbiter mutual visibility periods are limited, rapid acquisition of the capsule's signal by the orbiter is important.

Use of a phase-locked-loop receiver in this application will require automatic acquisition sweep circuits to search for the capsule's signal and will require that no data sidebands are within the range of acquisition.

An FM receiver with AFC used in this application may both simplify the acquisition process and allow use of reduced IF bandwidths. The performance of an FM receiver for this application needs investigation.

At present there has been no effort placed on this problem area; furthermore, there are no resources presently available to investigate this in FY '64.

5. Capsule Mars Atmospheric Entry Communications

During the high speed entry of the capsule into the Mars atmosphere, a plasma will surround the capsule making communication to the orbiter difficult, if not impossible. Investigation is needed into ways of quenching the plasma.

A partial study of this problem will be undertaken in the 1964 SR & AD program under task 5428-3346, Investigation of Capsule Antennas in Planetary Atmospheres. However, it is not felt that the level of resources presently available will allow a thorough definition of, or solution to, the problem in FY '64.

6. Low Bit-Rate Telemetry Techniques

A critical problem area brought to light by this study is that of communicating directly from a capsule to Earth with limited transmitter power and nearly omni-antenna performance. It appears that bit rates in the range of 1.0 to 0.1 bps or lower should be investigated since even this rate might satisfy a minimal scientific mission if the capsule lifetime were sufficient. The effort should logically include analysis and experimental verification of candidate techniques.

To date there has been no real study of this problem, and there are presently no resources available in the FY '64 SR & AD program with which to support this area. At most it is now possible to devote only part-time effort to investigation of this problem.

7. Circuit Losses

Circuit losses between the communications case and the spacecraft high-gain antenna are likely to be high. One method of reducing these losses is to mount the final RF amplifier stage on the antenna support structure. This, however, requires that all power and RF drive signals be conducted through some flexible device, or perhaps a rotating conducting device such as slip rings. At present only a cursory study of this problem has been made and more effort is required. However, there are no resources available in FY '64 to direct toward the solution of this problem.

8. Propagation Anomalies

The calculations of the capsule-to-orbiter communication links have not considered the propagation anomalies that may exist in the Martian atmosphere. Dust storms may cause RF attenuation. Current data on the ionosphere indicates that there should be no problem, but more investigations are needed. However, this problem area is not covered by any existing or proposed SR & AD tasks.

9. Antenna Alignment

Investigations are needed into ways of keeping the capsule antennas pointed upward even though the capsule may not come to rest with the antenna axis at the local vertical alignment.

The problem will be investigated as part of the FY '64 task entitled High Impact Capsule Antennas. The effort will include both the mechanical survival and subsequent orientation problems associated with planetary entry capsules.

10. Effects of a Low Pressure Martian Atmosphere Model

If the recently postulated low pressure Martian atmosphere (10-25 mb range) proves valid, solid-state components at reduced power output are indicated due to the possibility of ionic breakdown and high G levels. At reduced power output the data capabilities of both capsule links are greatly reduced. Shock requirements may prevent vidicon experiments because of the lack of storage media that can meet these requirements.

1) At the lower pressure levels and different composition postulated in recent models for the Martian atmosphere, there would be severe problems because of ionic breakdown from both RF field strength and DC field strength throughout the capsule mission in the atmosphere. With this atmospheric model the critical breakdown voltages are far below the voltage range required

for vacuum tubes. If vacuum tubes were required, satisfactory insulation would have to be provided for all high voltage surfaces. Although the electrical characteristics of dielectrics at DC present no problem, developments in the area of low-loss dielectrics for use at radio frequencies are not adequate to meet the insulation requirements imposed by the low pressure atmosphere. A partial solution to the breakdown problem would be the use of solid state components since the operating voltages required are far below the critical levels.

2) If the density of the Martian atmosphere is as low as 10-25 mb and the surface winds are correspondingly high, then the possibility exists that the capsule will experience accelerations at landing greater than 200 G's. These accelerations would be probable if the design of a Surveyor type soft landing capsule is not considered for this mission. In the communications area, solid-state transmitters will be needed to satisfy the high shock requirements. The use of solid-state transmitters in the capsule will probably limit the power levels on both the direct and relay communications links. Very optimistic predictions for the S-band direct link indicate that a 25 w solid-state transmitter would be available for a 1969 mission, but that a 10 w transmitter would be a more feasible choice for a power level. Power levels for the 200 Mc relay link transmitter are predicted to be somewhat higher than for the direct link. A 25 w relay-link solid-state transmitter is already considered as an alternate configuration to a vacuum tube transmitter. Consideration of a 50 w solid-state transmitter is thought to be optimistic at present. Should vacuum tubes be utilized on either link, an additional problem would be to develop insulation materials which have desired electrical properties and which could withstand high shock levels without cracking.

The possible need to maintain lower transmitter power levels on the capsule-direct and relay-links may materially decrease the scientific value of the capsule mission. If a 25 w transmitter power level can be assumed for the direct link, only 1.6×10^5 bits could be transmitted to Earth by this link over the 30 day design lifetime assumed at present. This is a 93 percent decrease from the assumed direct-link data requirements at the 30 day point. From the viewpoint of the scientific value obtainable, a 10 w direct-link power level is considered undesirable. At a 25 w relay-link transmitter level, there would have to be a 57 percent decrease in the data requirements at capsule landing plus 150 days. However, as mentioned previously in the body of this report, vidicon data compression would greatly ease the problem of meeting relay-link data requirements with a 25 w transmitter power. Compression ratios would have to be on the order of 3 to 1 to completely satisfy the data requirements. This ratio is greater than the 2 to 1 ratio assumed for this study.

D. Data Handling

Many of the problem areas discussed in the data handling section of this report are tied to specific system realizations, mission geometries, and more detailed requirements. Problems of a more general nature, common to all missions of the class considered, are summarized in the following discussion:

1. Reliability

The most significant data handling problem in Voyager class missions is reliability. This problem has dominated thinking in the entire aerospace industry for a number of years. Improvements in basic parts quality, fabrication, and developments in design and redundancy techniques have been significant. A Voyager class mission must utilize the best reliability improvements developed to date and must stimulate further investigation on all inroads to the problem. Accentuation of one particular area at the expense of others is not sufficient to insure adequate reliability margins. Each area not only supports but reinforces the effect of other reliability applications. As an example, a quick calculation of mission success, assuming a lifetime of 13 months (nominal Type II trajectory flight time plus 150 days after encounter); an electronic part total of 50,000 critical or "series" parts (reasonable for an 1800 lb spacecraft and 1500 lb capsule); adequate design margins; and a chance part failure rate of 0.0001 per cent per thousand hours (not currently possible, but perhaps achievable in 2 to 3 years) yields a reliability figure for the total mission success of 0.9 percent. The results of such a calculation show that the use of high quality parts and fabrication techniques alone will not achieve the required spacecraft reliability. Another facet of the solution to the reliability problem is the application of redundancy techniques. It is generally an accepted rule that as the basic non-redundant cell (part, logic block, subsystem, etc.) is made more reliable, the reliability increases when the cell is included in redundant configurations. The foregoing statements imply two things:

1) High part quality, sound design and fabrication techniques are a vital necessity for employing redundancy in this type of mission.
2) Redundancy measures on lower levels are needed.

Recently, the Laboratory has initiated developmental work in the area of worst case design techniques in-house and on outside contract. The efforts are basically of two types: first, there has been work done in the area of applying existing worst-case computer and hand design techniques to the Mariner series spacecraft. The techniques have been growing in sophistication and many useful lessons and design changes have resulted. In the same area, a developmental contract for an advanced command flight system with Philco Western Development Laboratories(4) has provision for application and evaluation of worst-case computer design techniques by a recent series of programs developed by Autonetics for the Recomp computer. Second, the Laboratory has initiated contractural negotiations for lease of a small general purpose engineering computer and joint development of a new series of worst-case design programs with the Control Data Corporation.(5) Major effort is also planned to apply lessons learned in the first area to this development work. Continuation and augmentation of these efforts in the future is warranted for partial solution to Voyager class reliability requirements.

Redundancy techniques have received considerable attention in recent years. A considerable amount of theoretical work has been done in active switched redundancy and passive techniques along the approaches outlined by Shannon, Moore, and Von Newmann. Unfortunately, only a small part of this theoretical work has had practical application in systems, mainly because of the prices in weight and power which have to be paid and the lack of sufficient data to verify theoretical calculations. The Jet Propulsion Laboratory has been following these redundancy efforts with interest in the past and has contracted for redundancy studies with Stanford Research Institute(6) for the primary purpose of investigating techniques to reduce redundancy ratios below limits existing now. A completely redundant analog to digital converter has also been developed under JPL contract by Texas Instruments Incorporated.(7, 8)

In the future, a more concerted effort to develop and apply practical redundancy techniques is considered a vital necessity for a Voyager class mission. In support of this effort, new system design philosophies and procedures for building reliability into the system during the conceptional stages of design are needed.

2. Data Compression Techniques

Interest in this area, as with redundancy techniques, has increased throughout the aerospace industry, although only comparatively recently. The type of data compression which will first be discussed deals with an ensemble of different sources, rather than the single source problem which arises in vidicon compression. Theoretical methods for compression may be the same in some cases, but system integration problems and final mechanizations may differ considerably. For convenience we will call the area of present discussion "multiple source" compression. Industry efforts in this area have been productive to the point of actual realization of flight systems; for example, Lockheed Missiles and Space Division is at present fabricating a flight system based upon the "floating aperture" principle and has demonstrated average compression ratios of greater than 30 to 1 with actual reruns of flight data.

The Jet Propulsion Laboratory has been following developmental work in multiple source compression with considerable interest. However, some problems in this area are found to be somewhat unique to deep space applications and, understandably, have not been given much attention by industry in general. Most notable is the problem of multiple flight modes which tend to render information more nonstationary than for a short or single purpose mission. Nonstationary information prevents evaluation of compression techniques by the classical methods of information theory. Data from deep space missions is also somewhat lacking, making evaluation of any given compression scheme a difficult one. There are extremely long periods of time in which information from the spacecraft is relatively invarient and data rates are more than adequate, even for uncompressed data. Application of compression techniques during periods where data is relatively invarient may be analogous to pounding the proverbial tack with a sledge hammer, to the detriment of reliability. However, during these periods, there is always the possibility of abnormalities in the measured systems, and the ability of the data system to adapt to, and to capture, this type of information is an extremely desirable characteristic. During short periods of time in which the information is varying rapidly, available data rates are usually insufficient to support the required rate in real time and, in fact, may drop to zero. Here, compression techniques may find useful application, especially from the viewpoint of reduction in storage, which is already required because of zero data-rate possibilities. In summary, the data compression techniques to be investigated fall into two categories:

1) Compression during those periods of known high acitvity where the transmission channel may not be available to all. Classical compression techniques based upon information theory concepts may be applicable in this area.
2) Adaptation to data of a transient nature and transmission of this data in some usable form to the users. The criteria for information is more subjective here.

Spacecraft in the future will contain a variety of measurements, such as digital or analog measurements, time measurements, time durations or frequencies; and measurement of the order of discrete events. System integration problems for such measurements, even with no compression, will become more difficult in the future. It is desirable to incorporate multiple source compression techniques into systems without undue revision in sound system design concepts or increases in complexity.

The investigation of multiple-source data compression techniques, or in its simplest sense more efficient multiplexing and transfer of data to the user, is a relatively new field within the Laboratory. Future effort in this area must involve a great deal of in-house activity, with close coordination with industry, to obtain effective support toward the solution of the particular problems associated with deep space missions.

The second area of discussion in data compression is vidicon data compression, which has been mentioned in this report as a promising area for future investigation. A study in the application of vidicon data compression techniques to planetary spacecraft leads to the conclusion that they offer tradeoffs in two primary areas: the amount of data which has to be obtained, and subsequently stored or transmitted, and the time required to transmit this data to Earth. They are therefore related to net mission reliability.

Vidicon data compression allows tradeoffs in the following areas: image resolution/frame, total number of frames or resolution elements, size of bulk and buffer storage devices required, time required to transmit data to Earth, and probability of success. As channel capacities increase, direct alternatives in the area of transmitter power or transmission bit-rate may be realized.

It is possible to order vidicon data compression techniques in several different ways. The first is a function of the manner in which the information is read out of the imaging device:

1) With linear, or constant, scan velocity
2) With variable scan velocities

The former method is, of course, the simplest and requires rather conventional techniques, but the latter offers the most potential as far as compression techniques are concerned. A variable-velocity scan system is sometimes called an "elastic" system, in that the length of time it takes to read out an image is proportional to the amount of detail, or information, contained in the scan. It is for this reason that higher efficiencies are potentially possible with the variable-scan approach.

The second possible technique is related to the kinds of redundancies which are eliminated by data compression. The simplest possible methods operate on the element-to-element correlation. The next simplest method operates on groups of elements or line-to-line correlation. The third (and most complex approach) is to remove redundancy between successive areas (or frames). It also tends to be true that the efficiency, or compression ratio, improves in the same order as given.

These two possible techniques are obviously related. The constant-scan approach lends itself exceptionally well to compression at the element level, and with the addition of more complexity, including nominal buffer storage, to element-group or line techniques. When the constant-scan approach is applied to area or frame techniques, the amount of buffer, or bulk, storage and peripheral electronics appear prohibitive. The variable scan approach therefore is most logically mated with area (or frame) and element-group or line compression techniques.

JPL has followed developments in industry and at other NASA centers with considerable interest, and has let a contract to attack the problem independently and to complement its in-house activity.(9) FY'62 funds have been spent to procure an extremely versatile video imaging and display system for in-house studies.

The Laboratory is presently constraining its major in-house and contractual efforts to the study of vidicon data compression techniques bounded by linear-scan readout of the imaging device. This appears to be a logical first step in the evolution and application of these techniques to the needs of planetary spacecraft missions. A minimum design goal has been set at a compression ratio of 2/1, with respect to a "standard" 6-bit PCM system. It is felt that this is a reasonable estimate of the efficiency possible with a constant-scan technique, assuming that the ultimate use of the data is objective and not subjective. The time scale for this effort has been established to support a Mars 1966-67 mission, and it is felt there is a reasonable probability of employing such a technique at that date.

The next logical step is to investigate the so-called "elastic" or variable-scan compression techniques. It is believed that these should make possible savings of 3 or 4 to one, but to date this is still an opinion, unsupported by analysis or experiment. For this reason, it is recommended that for the present study afford a savings of only 3/1 be conservatively assumed, measured with respect to conventional 6-bit PCM.

3. Storage Capability

Capability required for Voyager class missions will fall into three general classes:

1) Capacity in the range of 1×10^7 to 1×10^9 bits, non-volatile, non-destructive readout, with capability for close system integration desirable, but not a major requirement at present.
2) Capacity in the 6×10^5 to 1×10^6 bit range, non-volatile, destructive and non-destructive readout, and capability for close system integration with minimum complexity.
3) Capacity in the range of 500 to 5×10^4 bits, non-volatile, generally with non-destructive readout, and capability for close system integration with minimum complexity.

The first two classes or memory constitute a major problem area. EPD-139, Volume II discusses several approaches to meeting memory requirements in the first class listed above and, in general, the more specific requirements outlined there are applicable. However, application of these approaches to solving capsule storage problems will involve attendant higher risks because of sterilization requirements and a more severe environment. Of the approaches outlined in EPD-139, Volume II, thermoplastic recording techniques seem to hold a slight reliability edge in that moving parts are minimized and large buffering systems are not necessary to integrate the technique into a system. Potential problem areas associated with the present development of a thermoplastic memory system are as follows:

a. Miniaturized Electron Gun

Present record/playback electron guns are conservatively designed for a length of 15 in. Potential problem areas associated with shortening the gun for miniaturization purposes include:

1) Increased magnetic fields affecting secondary paths.
2) Increased variation of readout signals as a function of data location.
3) Diametric focusing circuitry may be required.
4) Miniature magnetic lenses capable of deflection angles greater than ±12 deg will have to be developed.

b. Electron Gun Power Requirements

Present electron guns require approximately 10 w to heat the cathode. Low power gun techniques are presently under development.

c. Operating Temperature Limitations

Present commercially available thermoplastic films will not maintain permanent storage at 80°C. A miniaturized heat control unit is available for controlling the film temperature to not exceed 50°C for a fixed plate configuration. However, a thermoplastic film possessing a higher operating capability should be selected for spacecraft applications.

d. Vacuum Requirements

The electron gun must operate in a vacuum. For capsule applications in which exposure to an atmosphere may result, a sealed vacuum is required. The use of a miniaturized sustainer vacuum pump which is commercially available could be used to maintain an adequate vacuum for years.

e. Electronic Circuitry

Redesign of present electronic circuitry must be accomplished to convert from vacuum tube to solid-state form.

In the second class of required storage capacities, study contracts are under way with Univac (10) and IBM (11) for flight storages in the range of 1×10^5 to 1×10^6 bits. Film memory with electronics has been demonstrated at Univac in the 1×10^5 bit size. Memories are under construction using 20 mil core. Plated wire and ferrite "cross-tube" techniques are 1 to 3 years away. There is a reasonable expectation that a 1×10^6 bit memory can be packaged into 0.25 ft^3, weigh about 30 lb, and consume less than 10 w of power at 1×10^5 bps rates by the 1965-1967 period. Future development efforts in this area should primarily be directed towards achieving greater weight reduction, because the relatively low input-output rates required of this memory class tend to minimize the power problem.

The third memory class listed should not present any major problems.

4. Option I: Data Handling Sequence Generation

A comparison between communication Option I and Option II was made in Chapter 3 B-9 of this report. Option I considered the capsule without external command control. Option II considered the capsule with external command control. This control included an orbiter-to-capsule beacon link which, in effect, would notify the capsule of the orbiter availability for receipt of data. Without this control under Option I, the availability profile of the orbiter must be generated on-board the capsule. The problem of how best to obtain a profile to be generated on the capsule was alluded to in Chapter 3 B-9 and is considered to be an area requiring considerable further study.

5. Effect of Low-Density Martian Atmosphere

A high G level resulting from recently considered low-density atmospheres will also significantly affect the choice of hardware for large bulk memories in the data-handling system. At present, tape recorders are about the only devices capable of meeting the capsule vidicon data-storage requirements. High anticipated shock requirements may prevent their usage, even in redundant configurations. Thermoplastic recording techniques are under development, but many problems remain to be solved before these techniques can be utilized in the capsule. One of these problems is the conversion of the hardware to solid-state. Thermoplastic recording equipment must also operate in a vacuum, necessitating the use of a seal for capsule applications. In summary, an increase in G level requirements for the capsule data-handling system may prevent the carrying out of vidicon experiments because of the lack of storage media which can meet the shock requirements.

REFERENCES

1. L. D. Jaffe, "Sterilization of Unmanned Planetary & Lunar Space Vehicles -- An Engineering Examination," JPL Technical Report No. 32-325 (Rev.)

2. G. F. Schilling, "Li[illegible] Model [illegible] of Mars," Rand Report No. R-402-[illegible], August 1962.

3. L. D. Kaplan, "An Analysis of the Spectrum of Mars," Astrophysical Journal, January 1, [illegible].

4. JPL Design Specification, 31022 "Mar[illegible] [illegible] Flight Equipment Command Subsystem Single Channel Command Detector," [illegible] 1962.

5. JPL EPD 157, "A Worst-Case Circ[illegible] [illegible] Technique Utilizing a Small Digital Computer," August 14, 1963.

6. SRI report M-4450 under NASA contract NAS-w-8, "Automatic Fault Detection and Conversional Switching Networks," Technical Report No. 1, April, 1961. And SRI report M-4450 under NASA contract NAS-w-6, "Development and Techniques for Improving Reliability for Digital thr. Logical Redundancy," Reports 1 thru 7, January 1960 thru December 19[illegible].

7. JPL Specification 30302, "Design Specification - Mariner Flight Equipment Telemetering Development Analog-to Digital Converter," April 27, 1961.

8. JPL SPS 37-2[illegible], Volume [illegible], page 23[illegible], "[illegible] Analog to Digital Converter Development," August [illegible].

9. JPL Job No. 325-0230[illegible]-[illegible]341.

10. JPL contract No. 950[illegible], May 20, [illegible]

11. JPL contract No. 950411, "Advanced Solid-State Magnetic Memory Study,"