ORBITAL EXPERIMENT CAPSULE

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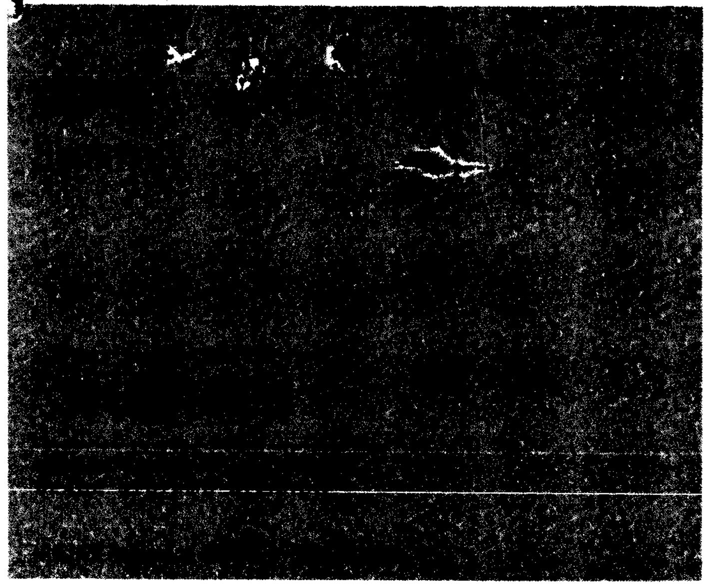
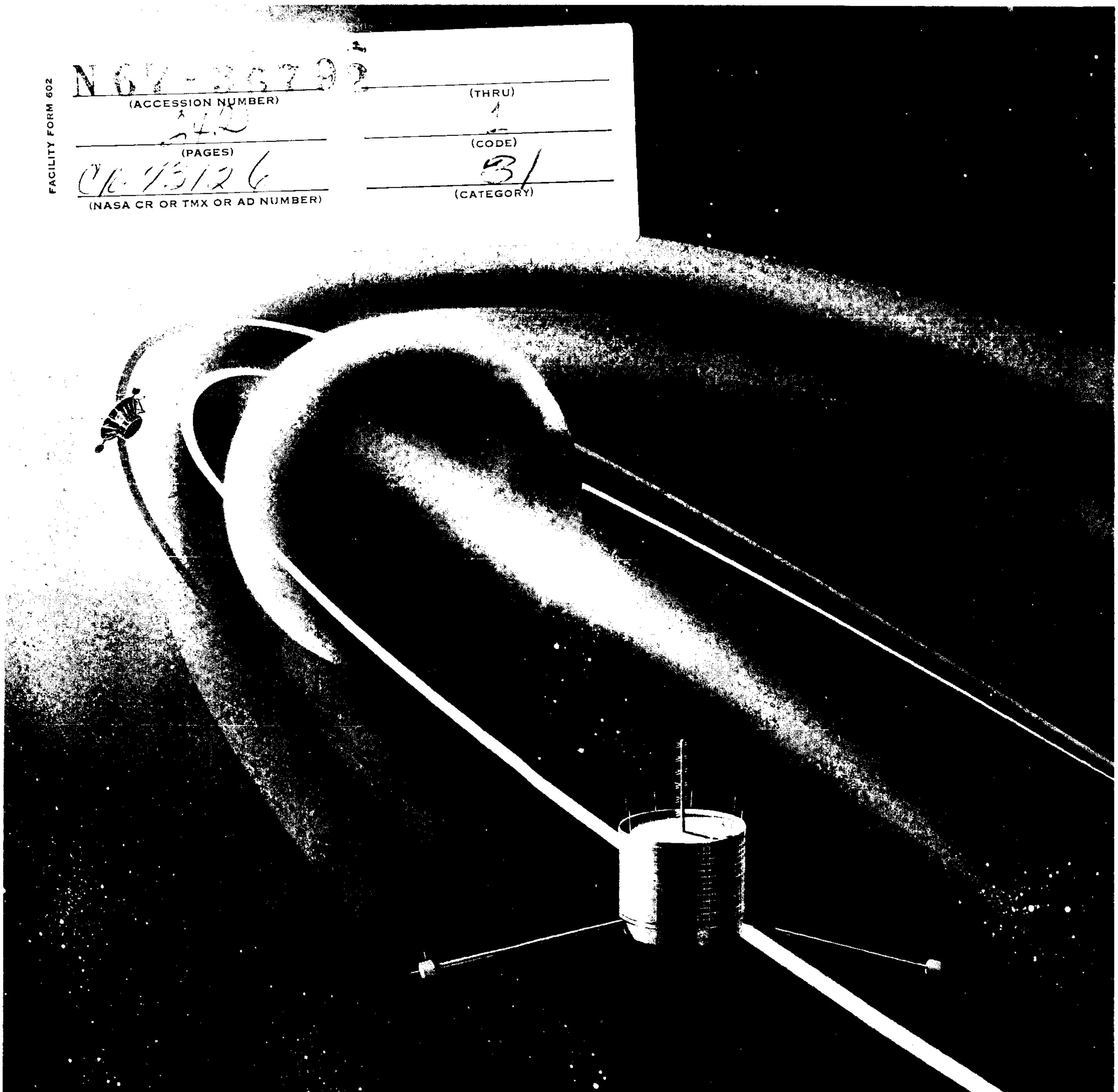
FEASIBILITY STUDY

FINAL REPORT

SEPTEMBER 1967 • NASA Contract NAS 2-4236





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FINAL REPORT

NASA Contract NAS 2-4236

VOLUME I
Summary

SEPTEMBER 1967

HUGHES Reference B2810

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HUGHES AIRCRAFT COMPANY
SPACE SYSTEMS DIVISION

SSD 70384R

FOREWORD

In accordance with the requirements of the Ames Research Center NASA Contract NAS 2-4236 dated 13 March 1967, a feasibility study has been conducted for the Orbital Experiment Capsule (OEC) mission concept. This document represents its final report. The OEC would be a scientific satellite of Mars to be initially carried "piggy-back" on the Voyager spacecraft and ejected from it upon completion of the operational phase between the Voyager spacecraft and the Voyager Lander. This approach is so specified in order to eliminate any possible interference by the OEC with the Voyager prime operations. The OEC concept is such that it would serve as a scientific bus permitting the performance of a variety of experiments via a capability to carry various complements of scientific instruments within prescribed requirements and constraints. The necessity of an OEC as part of the Voyager mission is dictated by the mission scientific objectives and the impact of these objectives on the design and requirements of the Voyager spacecraft. The OEC concept permits the satisfaction of some of these scientific objectives with significant reduction in cost, complexity, and constraints on the spacecraft.

The objectives of the study were to perform a feasibility assessment leading to the definition of a mission philosophy, profile, and design. These objectives have been accomplished to the level of definition of a scientific satellite configuration and mode of operation which fully satisfies all of the prescribed mission requirements and, furthermore, introduces scientific data gathering flexibilities that would not be achieved with the Voyager spacecraft itself.

The Final Report is presented in three volumes. Volume I, entitled "Orbital Experiment Capsule Feasibility Study Final Report," provides all the study results in summary form. Volume II, "Supporting Technical Studies and Tradeoffs," provides all of the backup data to substantiate the findings and recommendations presented in Volume I. Volume III, "Budgetary Cost and Schedule Data," provides a preliminary look at the hardware implementation requirements of the OEC. Volume III presents plans and schedule data as well as budgetary cost estimate for the development phase of an OEC.

The OEC study has had many contributors. The following group of names summarizes only the most important of these contributions which have made possible the careful treatment of the OEC definition within the contractual cost and schedule constraints.

V. T. Norwood	Head, OEC Communication and Data Handling Studies

L. E. Schwaiger	Head, Configuration and OEC Subsystems Definition
W. Turk	Head, OEC Systems Analysis
E. E. Angle	OEC Experiment Requirements and Magnetic Control
R. J. Nowicki	OEC Cost and Schedule Data

A large portion of the material presented in the three volumes which comprise this Final Report is the result of the work of personnel from various laboratories. A partial list of these contributors is as follows: Systems Analysis, B. H. Billik, E. P. Harris, L. Schwartz; Propulsion, W. W. Butcher, L. M. Wolf; Power, H. F. Prochaska, P. S. DuPont; Thermal, R. J. Wensley, R. D. Welsh; Attitude Control, B. Porter, W. L. Townsend; Separation, L. P. Birindelli; Reliability, R. J. Schulhof, M. A. Anderson; Project Plan/Cost Data, H. Reich, A. Wenters, R. C. Summers.

A list of the contributors to this effort would not be complete without the names of certain Ames Research Center Personnel. Mr. C. Privette of the Space Technology Branch, who managed this contract for the Ames Research Center, provided extensive direction and consultation during the course of the study. Valuable comments and contributions were also given by Messrs. J. Wolfe, C. Sonett, E. Iufer, and many others who participated in the review and discussions of the material presented during the course of the study.

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1.0 INTRODUCTION

The existence or nonexistence of a magnetic field of the planet Mars is still unknown. There have been no Earth-based observations of radio emissions characteristic of trapped particles nor auroral activity characteristics near magnetic poles to indicate any magnetic field. Direct measurements from the Mariner IV spacecraft indicated no magnetic field at a distance of 13,200 km from the center of the planet. Although the latter measurement did not indicate the presence of any magnetic field, it did establish an upper boundary. It is estimated from the Mariner IV measurements that the magnetic flux density at the equatorial surface is less than 100 gamma. Assuming a dipole field, this would give a polar value of 200 gamma. With such a possible weak field there is some question as to the source and nature of the field. The Mariner IV measurements indicated that any Martian dipole field strength could not be greater than 0.0003 that of Earth. Hence, it can be seen that the measurement of a Martian magnetic field, if it exists, requires a sensitive magnetometer, which in turn requires a magnetically clean spacecraft.

Measurements of this nature, as well as plasma environment and better definition of the Martian atmosphere, are within the scope of the overall space program and have been outlined as part of the scientific objectives of the Mars Voyager mission.

The limitation that has been placed on the inherent magnetic field of the spacecraft carrying the magnetic field sensors is 0.25 gamma at the sensor location. To provide a benign magnetic environment equal to, or lower than, this level in the vicinity of a spacecraft the size and complexity of the Voyager Orbiter spacecraft represents an extremely difficult if not impossible task. To impose magnetic control specifications and requirements on the spacecraft subsystems and components to ensure low permanent stray and induced magnetic fields has serious design, test, cost, and schedule implications. For example, the facilities that would be required to perform the system magnetic tests on a spacecraft the size of the Voyager spacecraft do not exist today; a 40 to 50 foot diameter Helmholtz coil facility would probably be needed. The addition of booms for providing a separation distance between the magnetometer and the spacecraft represents a partial solution to the spacecraft magnetic control problem. It is estimated that in order to meet the 0.25 gamma requirement, boom lengths in the range of 10 to 30 meters would be required for the Voyager spacecraft. The use of booms of this length presents problems with regard to stowage and/or deployment, weight, and alignment accuracy with regard to the three-axis position of the magnetometer. In addition, significant effects would be introduced on the spacecraft's control system design and performance.

To perform plasma probe measurements, a scanning (spinning) platform would be required on the Voyager Orbiter. Plasma wake effects associated with the presence of the large Voyager Orbiter in the vicinity of the instrument would jeopardize the plasma probe measurements. The need for various field and particle measurements also imposes a requirement for a flexible scientific interface between the Voyager spacecraft and the instruments. This flexibility has certain cost and schedule implications.

The problems briefly described above associated with performing the desired measurements on the Voyager spacecraft led to the definition of a feasibility study to explore the possibilities of conducting these types of measurements in a meaningful and economical manner via a separate spacecraft from Voyager. This separate spacecraft has been named the Orbital Experiment Capsule (OEC). This concept would not only minimize or eliminate the previously mentioned problem, but would provide additional advantages such as a fixed interface between the Voyager Orbiter and the complement of instruments that would be carried on the OEC. Changes in the type, number, or requirements of the instruments would be accommodated by the OEC/scientific payload physical interfaces with no effects on the Voyager spacecraft/OEC physical interface. In addition, the OEC concept provides an important opportunity to perform mother-daughter occultation experiments with the Voyager spacecraft.

This report presents the results of the OEC Feasibility Study performed by the Hughes Aircraft Company for the Ames Research Center under NASA Contract NAS 2-4236, dated 13 March 1967. The fundamental concept is one in which the OEC would be carried aboard the Voyager Orbiter. Subsequent to the establishment of an orbit about Mars and upon completion of Voyager Lander operations, the OEC would be ejected into an orbit of its own. The OEC would then conduct a variety of experiments while in this Martian orbit.

The purpose of this study, which was conducted over a period of 6 months, was to perform the necessary systems analysis, communications and data handling, configuration, and associated studies to evaluate the technical, reliability, cost and schedule, and Voyager program impact implications of such a concept. Numerous technical studies and tradeoffs were conducted during the first 3 months of the study in order to define an OEC mission design spectrum. During the final phase of the study specific conceptual approaches were selected and studied in detail which led to the definition of a recommended feasible and highly flexible OEC configuration that could be used to conduct a variety of experiments in the vicinity of Mars such as:

- Map the magnetic and electric fields
- Measure solar winds
- Map the "captured" or trapped particle radiation
- Measure the dust particle and distribution near Mars
- Monitor solar flare intensities

- Measure aerographic distribution of the atmosphere
- Measure composition of the atmosphere

The results of the feasibility study are presented in three volumes. Volume I, entitled "Orbital Experiment Capsule Feasibility Study Final Report," presents a detailed summary of the work accomplished and significance of results. In the first portion of Volume I, data is given on the Voyager mission, the OEC mission and its requirements and constraints, the spectrum of possible OEC approaches and designs, and the specific configurations which have evolved as a result of the various technical studies and tradeoffs. Data is also presented on the OEC impact and constraints on the Voyager program and the cost and schedule implications of the OEC concept for the Voyager program. Summary data is then presented in Volume I on the systems analyses which were conducted during the course of the study. These include orbit analysis in which the relationship of the OEC orbit to the magnetosphere is evaluated, orbit lifetime and solar eclipse studies, and problems of the Voyager Orbiter-OEC occultation and their antenna field of view requirements. Other studies include the relative Voyager Orbiter-OEC geometry, the separation problem, and perturbation effects. The problems of the OEC attitude and orbit determination, attitude control, and orbit change possibilities were explored in depth and tradeoffs conducted to define the most appropriate approach within the given constraints. The problem of acquisition and relay of scientific and engineering data was studied and the results are presented in the communication and data handling section of this volume. A summary of the OEC configuration and subsystem studies conducted is also presented in this volume in addition to the results of the reliability and magnetic assessments.

Volume II, entitled "Supporting Technical Studies and Tradeoffs," presents in detail the technical data generated during the course of the study. In addition to the studies mentioned as contents of Volume I, detailed information is given on the Mars environment and the range of experiments that could be conducted on the OEC mission as well as details on possible instruments.

Volume III, "Budgetary Cost and Schedule Data," presents a preliminary assessment of the hardware implementation phase for the development and flight of the OEC. A master phasing schedule has been prepared, based on the recommended OEC configuration and schedule goals for the Voyager mission. A preliminary budgetary cost estimate was developed on the basis of the recommended configuration and certain program ground rules and preliminary project plans.

Although a specific OEC configuration has been selected and is recommended for development, the numerous studies conducted led to the definition of a range of possible OEC configurations within the prescribed constraints which vary incrementally in complexity and flexibility. The recommended configuration exceeds the fundamental requirements in terms of scientific objectives, and its flexibility is limited only by the weight allocation to the OEC. The baseline has a gross weight of less than 125 pounds and provides the following features:

- It is capable of carrying a complement of experiments weighing 15 pounds and requiring 10 watts of power continuously.

- It will be capable of operating for 6 months after ejection from the Voyager spacecraft.

- It will be capable of collecting and transmitting data to the Voyager spacecraft on a continuous basis or will be able to store the data and effect transmission at optimum times.

- It will be capable of receiving commands directly from Earth; data can be transmitted direct to the Deep Space Network in degraded modes.

- It will be capable of performing orbital changes, i.e., periapsis drop, in order to explore other regions in the vicinity of Mars.

- While being able to operate in the above modes, the recommended configuration also meets all of the scientific objectives related to magnetic cleanliness, attitude orientation, knowledge of the attitude orientation, and orbital position and data rates.

- This mission design imposes minimal requirements on the Voyager spacecraft. These requirements are primarily a location for stowage and release corridor, power for thermal control during the transit phase, a near isotropic receiving capability, and data storage and transmission capability. Of the above, only one represents a new hardware requirement on the spacecraft (isotropic antenna), while the others are fully compatible with the proposed spacecraft designs by General Electric, TRW, and Boeing.

2.0 SUMMARY

2.1 MISSION REQUIREMENTS

The general requirements of the OEC mission derive from two fundamental sources: scientific requirements and Voyager compatibility, and Voyager related constraints and interactions.

2.1.1 Scientific Requirements

It is of fundamental scientific importance to determine if Mars has a magnetic field. Equally important is a determination of the interaction of solar wind with the Martian environment. Does Mars have a magnetopause or do the solar wind particles interact directly with the Martian atmosphere so as to produce some sort of a bow-shock? Perhaps a combination of both mechanisms is at work. Considerations such as these dictate the instrument complement to be flown, capsule orbits, scientific data rates and accuracy, and accuracy of determination of capsule attitude and position. Other desirable mission requirements, such as the ability to perform occultation experiments with the Voyager spacecraft, impose additional requirements on the capsule design.

Scientific Payload. The scientific payload will consist of a complement of instruments which may vary from one OEC to another. Certain payload parameters have been defined in order to establish boundaries to the OEC requirements. A representative payload consisting of one magnetometer, one plasma probe, and an electric field meter has been selected in order to establish these boundaries. This payload establishes a power requirement (10 watts continuous), a weight budget (15 pounds), and a data rate capability (550 bits per second). In addition, volume and viewing requirements are defined by these instruments, as well as location requirements.

The specific instruments which were used in the course of the study to further define the spacecraft requirements were the Ames Research Center (ARC) three component flux gate magnetometer, the TRW electric field meter such as was used in the Pioneer program, and the ARC spherical plate analyzer for positive ion and low energy electron measurements. Other instruments were considered including the requirements for conducting occultation experiments. These instruments are discussed in detail in Volume II, Section 2.1.

Spacecraft Magnetic Field. The requirement which has evolved for the magnetic level at the location of the magnetometer is 0.25 gamma. This value represents the net effect from the spacecraft due to its inherent properties and represents an OEC design objective.

Stabilization. A requirement for spin stabilization has been dictated for the OEC mission (Reference 1). This rate has been stated as 60 rpm $\pm$ 10 rpm.

Lifetime and Operational Life. The lifetime (orbit life) requirement (Reference 1) which has been imposed on the OEC is that the probability of an accidental impact or orbit decay shall be less than or equal to 3×10^{-5} over a 10-year period. The operational life requirement is not less than 6 months after separation from the Voyager spacecraft after being stowed on the spacecraft during transit for a period of 12 months.

Attitude Requirements. The attitude requirement which has been assigned to the OEC is that the spin axis be normal to the Mars ecliptic within 5 degrees. This requirement is dictated by the scientific payload. In addition, from an optimal communications and power point of view, the spin axis should be normal to the sunline.

Attitude and Orbit Determination Requirements. Certain accuracy requirements as to knowledge of the attitude and orbit positions have been specified in Reference 1 and evolved during the course of the study. The spin axis must be known to 1 degree; orientation with respect to the sunline has been specified at 1/4 degree. The OEC position in orbit is to known to 20 km at periapsis and 100 km at apoapsis.

Reliability. The basic reliability requirements for the OEC mission are those outlined in Reference 1. A fundamental requirement is a "fail-safe" design that would not jeopardize the success of the overall Voyager mission. The reliability design goal for the OEC has been specified as 0.75 for a minimum operating period of 6 months, after a year's transit stowage.

Communications Frequencies. S-band RF frequencies will be utilized for the Earth-Voyager spacecraft and Earth-OEC radio communications up and down links. The OEC-Voyager spacecraft radio relay link will be consistent with the Lander spacecraft communication system.

Simplicity. An important requirement imposed on the OEC mission design is simplicity. This approach was to be carried to the extent of making use of existing concepts and hardware whenever possible.

2.1.2 Voyager-Related Requirements

The nature of these requirements derive from the need for compatibility in design and operation, minimization of constraints to be imposed on the Voyager mission, OEC weight allocation, and the need for "fail-safe" design throughout where OEC failures would not compromise the primary Voyager mission. These basic requirements are briefly discussed below.

Voyager Compatibility. The OEC must be compatible with the 1973 Voyager mission spacecraft. This compatibility requirement dictates specific mission design constraints not only in terms of hardware and interfaces but also in the nature of the orbit the OEC would be carried into at the beginning of its operation, as well as the nature of perturbations that might result from the spacecraft operation. Specifically, in terms of orbits and perturbations, the range which has been specified in Reference 2 is shown in Table 2-1.

TABLE 2-1. OEC MISSION CONSTRAINTS DICTATED BY 1973 VOYAGER MISSION

1)	Periapsis altitude, h_p	500 km $\leq h_p \leq$ 1500 km
2)	Apoapsis altitude, h_a	10,000 km $\leq h_a \leq$ 20,000 km
3)	Orbital inclination to Martian equator, i	$i \geq 30$ degrees
4)	Orbital inclination to ecliptic, i_e	$i_e \leq 45$ degrees
5)	Latitude of periapsis, ω_p	$-60 \leq \omega_p \leq 40$ degrees over 6 months
6)	Central angle between sub-periapsis point and nearest terminator, λ_{ω_p}	$0 \leq \omega_p \leq 45$ degrees for first 3 months, $-30 \leq \lambda_{\omega_p} \leq 90$ degrees thereafter
7)	Angle between orbit plane and terminator plane, i_p	$i_p \geq 30$ degrees for first 3 months, $i_p < 30$ degrees for total of 1 month over next 3 months
8)	Solar eclipse duration, T_e	$T_e = 0$ for first 30 days, T_e = minimum (8 percent of orbit period or 60 minutes/orbit) for next 5 months
9)	Capability required to rotate periapsis by at least ±20 degrees from initial (hyperbolic) location	
10)	Capsule de-orbit maneuver will be performed between 3 and 12 days after orbit insertion. Capability for delaying this operation for 30 days required.	
11)	Unpredictable translational accelerations originating in Voyager will not exceed total average value of 0.6×10^{-7} cm/sec^2 (3σ), time average over 1 hour.	
12)	Orbit trim maneuvers may be required for post-landed orbital operations.	

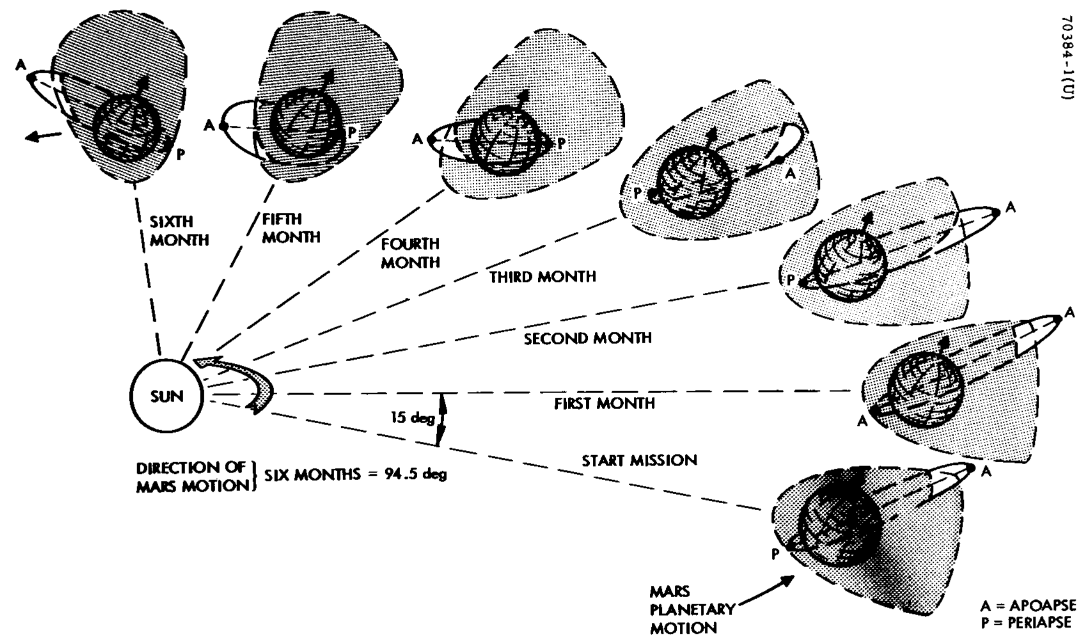

Figure 2-1. Orbit—Magnetosphere Motion

The general compatibility requirement also provides the fundamental criterion for environmental design of the OEC since it must survive all of the environments outlined for the Voyager spacecraft. In addition, other environmental requirements have evolved from the nature of the OEC mission; i.e., thermal design must account for the survival of the OEC under eclipse conditions which are more severe than those outlined for the Voyager spacecraft.

The telemetry data acquisition and command must also be compatible with the present plans for the Voyager spacecraft; i.e., the Deep Space Network (DSN) will provide tracking, telemetry data acquisition, and command coverage for the Voyager spacecraft from injections to the end of the mission.

Weight Allocation. The weight allocated to the OEC system including the scientific payload is in the range of 75 to 125 pounds. It is within this constraint that the various possible OEC configurations were studied.

"Fail-Safe" Philosophy. In the mission design approach, of primary importance is the fail-safe philosophy to be used in order to eliminate any possibility of interference with the primary Voyager mission. For example, the OEC separation technique must ensure that no detrimental perturbations are induced on the Voyager spacecraft, while eliminating the possibility of collision during and after separation.

The fundamental requirements briefly discussed above served as guidelines for the definition and study of the OEC mission. In the following pages the mission philosophy, possibilities, and various solutions and modes of operations are discussed.

2.2 STUDY AND MISSION PHILOSOPHY AND POSSIBILITIES

The philosophy followed in performing the OEC feasibility study was one in which within the given mission requirements and constraints a wide spectrum of possibilities were considered in defining the mission design. This mission design range which covered the possible conceptual modes of operation of the OEC was bound by the various constraints associated with the Voyager mission. In the simplest concept, the concept of a co-orbital OEC mission was evolved. In the most flexible cases, the concept of an orbit change mode was defined and studied. These terms are used with respect to a mode of operation.

2.2.1 The Co-Orbital Concept

In the co-orbital concept, the OEC remains close to and essentially in the same orbit as the Voyager spacecraft during its entire lifetime. The orbit difference would be strictly that caused by the low separation velocity from the Voyager and its related effects. In this case the OEC relies on the natural motions of the orbital plane (regression, rotation of line of apsides) to perform an extension mapping of the environment in the vicinity of Mars. The motions of the orbital plane with respect to the planet and an assumed magnetosphere are illustrated in Figure 2-1.

In the simplest version of a co-orbital OEC and mode of operation, a mission can be feasibly designed with a gross weight in the vicinity of 75 pounds. In this simple co-orbital case, the OEC would be attached to the Voyager spacecraft at such location(s) that, upon separation from the spacecraft, the OEC would be in the required attitude with respect to the sunline and the Martian ecliptic. This approach necessitates a careful design of the separation system since the separation velocity must be minimized in order to minimize the separation distance between the spacecraft and the OEC during the 6 months of operation so as to prevent occultation by Mars between the OEC and the spacecraft. This low velocity also allows the required power levels for communications to be maintained at reasonably low levels.

Another important factor associated with the separation system design is the need to limit the transverse loads at separation. This requirement evolves from the fact that this simple design would not incorporate an active attitude control system. The OEC would be stabilized by means of spin, and the spinup would be performed after ejection from the spacecraft. The lower the transverse loads, the lower the attitude errors the OEC would have.

A lower limit exists, however, for the magnitude of the separation velocity. This value is related to the collision criteria between the OEC and the spacecraft. Once separated, the OEC would be spun up by a simple blowdown cold gas system to the desired level of 60 rpm ±10 rpm. At this point, scientific data gathering operations would begin. The mode of operation would be constant during the lifetime of the OEC. Data would be obtained continuously and relayed real time to the spacecraft. No receiving nor data storage capability would be incorporated on the OEC. During the required 6 months of operation, the OEC would not be occulted from the spacecraft, assuming no Voyager orbital maneuvers are made, and the power supply design would be based on the criteria of being able to transmit continuously up to the end of the 6 months.

The type of sensors for attitude and orbit determination which would be incorporated in this simple co-orbital OEC would be a sun sensor for defining the spin axis location in one direction and a Mars sensor for defining the second attitude angle. The use of a pair of Mars sensors in conjunction with knowledge of the size of Mars permits determination of OEC attitude with respect to the planet, and hence definition of the OEC orbit about Mars.

The simplest OEC version, however, fails to precisely meet two of the previously stated requirements: the normality of the spin axis to the Martian ecliptic to 5 degrees, and knowledge of the OEC position in its orbit to 20 km and 100 km at periapsis and apoapsis, respectively. For this simple configuration, the normality requirement can be met by either adding an attitude correction capability or inducing partial spin to the OEC at separation. The first approach represents an additional weight increment of 4 pounds. In the second approach partial spinup would provide sufficient stability during the pre-full-spinup phase to ensure meeting the 5 degree normality requirement. Improvement of knowledge of the OEC position on its orbit to the required accuracy can be easily achieved by a number of ways. First, a star sensor (i.e., Canopus) could be incorporated which would provide the desired accuracy. Alternately, an S-band system could be added to the OEC which, by means of ranging from the DSN, could provide the necessary fix on the OEC position to the previously specified accuracies. The use of an S-band would also introduce the capability for up and down links for engineering and scientific data transmission directly to Earth.

The source of possible co-orbital configurations is illustrated in Table 2-2. The merits of the co-orbital approach lie in the simplicity of design and operation which yield a high reliability figure for its operation. In addition, the total weight of the configuration even with the S-band subsystem is substantially below the budgeted figure.

2.2.2 The Orbit Change Concept

The orbit change concept derives from consideration of the possibilities of obtaining additional mapping of the Mars environment by virtue of performing orbital changes. As previously described in the co-orbital concept, the OEC would essentially maintain the same orbit as the Voyager spacecraft, different only by virtue of the small separation velocity effects. With an orbit change capability, the OEC could drop its periapsis, perform small changes in the orbital inclination, or even rotate the line of apsides.

However, changes in OEC orbit with respect to the Voyager orbit introduce new problem areas. Of primary importance is the effect of the new orbit on the communication aspect of the mission. The increased distances necessitate increased power levels. In addition, occultation by the planet would take place between the Voyager and the OEC on a cyclical basis. This then dictates the need for storage of the data, which would then be transmitted at optimum visibility and power availability conditions. A command loop would also be required. However, since a propulsion system would be carried aboard the OEC to perform the orbital changes, stationkeeping or orbit synchronization between the OEC and the Voyager spacecraft would be maintained after the orbit change is performed. This would then allow the OEC to remain within close range of the spacecraft.

As in the case of the co-orbital concept, more than a single version of the orbit change concept can evolve, depending on the desired flexibility and weight allocation. In terms of altitude and orbit determination, the problems remain the same as previously described. Table 2-2 illustrates the possible orbit change system that could evolve. The fundamental differences between the co-orbital and orbit change system are summarized in Table 2-3.

2.3 RECOMMENDED OEC CONFIGURATION

The co-orbital and orbit change mission concepts have been studied, and each has proven to be a viable solution.

The simple co-orbital concept is a relatively light-weight vehicle and is shown to adequately meet mission accuracy requirements. This approach requires precise initial alignment on Voyager to provide an orientation of the configuration spin axis parallel to the ecliptic normal. The location on the Voyager does not appear to affect injection since the launch platform could be properly canted to the desired alignment.

TABLE 2-2. OEC CONCEPT POSSIBILITIES

Concept	Configuration Complements	Added Flexibility (Incremental)	System Weight, pounds	Accuracies (3σ) Attitude Determination, degrees	Accuracies (3σ) Orbit Determination, km	Spin Axis With Respect to ♂ Ecliptic over 6 Months, degrees	Maximum Relay Link Range, km	Minimum Periapsis Altitude, km
1) Co-orbital (basic)	• Baseline scientific bus • Scientific payload • N_2 spinup system • Continuous communication • Sun/Mars sensors • { Non-spinning separation / Partial spin or separation		75	±0.2 sunline ±3.0 spin axis	Periapsis < ~100 Apoapsis .< ~260	< 11.5 < 6.5	3,500	500 (minimum specified for Voyager)
2) Co-orbital (with attitude control system)	Same as (1) plus hydrazine attitude control system	Can orient to desired attitude from random separation attitude	79	±0.2 sunline ±3.0 spin axis	Periapsis < ~100 Apoapsis < ~260	< 5	3,500	500 (minimum specified for Voyager)
3) Co-orbital (with improved sensing)	Same as (2) plus third sensor or additional communication electronics	Can refine orbit determination	82	±0.2 sunline ±0.2 spin axis	Periapsis <20 Apoapsis <100	< 5	3,500	500 (minimum specified for Voyager)
4) Co-orbital (with S-band backup)	Same as (3) plus backup direct Earth communication system	Can operate without Voyager	122	±0.2 sun line ±1.0 spin axis	Periapsis <20 Apoapsis <100	< 5	3,500	500 (minimum specified for Voyager)
1) Orbit-change (basic)	• Baseline scientific bus • Scientific payload • N_2 spinup system • Continuous communication or storage of data • Sun/Mars sensor • Hydrazine attitude control system and orbit change • Non-spinning separation	• Attitude and orbit control • { ΔPeriapsis / ΔOrbit vaccination (limited) • Data storage • Long range communications	110	±0.2 sunline ±3.0 spin axis	Periapsis < ~100 Apoapsis < ~260	< 5	21,500	350
2) Orbit-change (with improved sensing)	Same as (1) plus third sensor or additional communication electronics	Can refine orbit determination	113	±0.2 sunline ±0.2 spin axis	Periapsis <20 Apoapsis <100	< 5	21,500	350
3) Orbit-change (with S-band backup)	Same as (1) plus backup direct Earth communication system	Can operate without Voyager	123	±0.2 sunline ±1.0 spin axis	Periapsis <20 Apoapsis <100	< 5	21,500	350
4) Orbit-change (without stationkeeping)	Same as (3) plus increased propellant system and electronics	Can stationkeep to Voyager	130	±0.2 sunline ±1.0 spin axis	Periapsis <20 Apoapsis <100	< 5	21,500	350
5) Orbit-change (with occulation)	Same as (4) plus occulation experiment and additional power	Can perform occulation experiment	160	±0.2 sunline ±1.0 spin axis	Periapsis <20 Apoapsis <100	< 5	21,500	350

TABLE 2-3. OEC MISSION SPECTRUM

	Co-orbital system	Orbit change system
• Orbit	Essentially same as Voyager spacecraft	Capable of changing orbit/ inclination
• Separation	Carefully controlled for low separation velocity and tipoff	Not critical
• Communication	Continuous short-range line; no storage or command loop required	Long-range link, storage and command loop required; higher power requirements
• Orbit determination	Same technique both cases	
• Attitude control	No active control required; spin-stabilized; attitude based on initial separation effects	Active control available for attitude maintenance, orbit change
• Propulsion	Simple spinup system	Spinup and orbit change systems
• Overall complexity	Simpler with exception of separation velocity conditions/ sensitivities	More complex but substantially more flexible

The importance of maintaining relatively short ranges to Voyager has been intensively studied to ensure the desired flow of experimental data during the mission. Selection of a relatively small separation velocity of 0.1 fps which is commensurate with relative ranges of 3500 km during the mission generates communication power requirements which are within the scope of the OEC mission objectives. A detailed error analysis indicates that ranges of this magnitude can be met with launch windows on the order of $\pm 1/2$ hour at periapsis to as great as ± 1 hour at apoapsis (nominal orbit of 10,000 km x 1000 km). A safety factor of approximately 1/3 km passage of closest approach between Voyager and OEC on the first orbit is designed into the launch window.

Following spinup, OEC operates as an experimental inertial platform, measuring Mars related phenomena and relaying the information via Voyager to Earth.

Whereas the co-orbital mission is qualified as being simple, it is difficult not to say that at the other extreme the orbit change concept is also relatively simple. However, in addition, it is qualified as being flexible and brings to the mission a depth that cannot be provided in the co-orbital concept.

Each of the missions suitably meets the requirements imposed. Selection of a preferred system requires that an additional constraint be imposed; otherwise, the choice would be made based purely on criteria of simplicity or flexibility. This added constraint is the nominal expected weight allotment that could be available on the Voyager 1973 bus. Since the co-orbital configuration can be designed within the 125 pounds allocated to the OEC, then the selection of an orbit change approach to the OEC mission becomes obvious because of the added flexibility and capabilities.

Although this configuration consists of a spacecraft capable of performing orbital changes, its conceptual design allows its operation in a co-orbital mode without reliability degradation. For example, the separation system design approach assumes that the attitude control system would not be used to correct the OEC attitude after spinup. (This approach assumes that a total growth of approximately 10 degrees off the normality of the spin axis to the Martian ecliptic would be permitted.) Therefore, the design of the separation, selection of time lapse between separation and initiation of spinup, and the selected thrust-time curve for spinup are optimized for this requirement. The velocity increment selected is also compatible with the requirement for 6 months of continuous data transmission capability. The spinup system consists of a cold gas blowdown system which operates independently of the attitude correction/orbit change propulsion system. The scientific and engineering data can be transmitted real time, bypassing the tape recorder. The net result is an OEC system operating at a reliability level in the co-orbital mode of operation which is identical to that attainable were the OEC spacecraft designed to operate purely in a co-orbital mode.

2.3.1 Configuration Description

The OEC conceptual design is depicted by an external profile in Figure 2-2. The recommended general arrangement is shown in Figure 2-3. A description of the overall capsule system is given below.

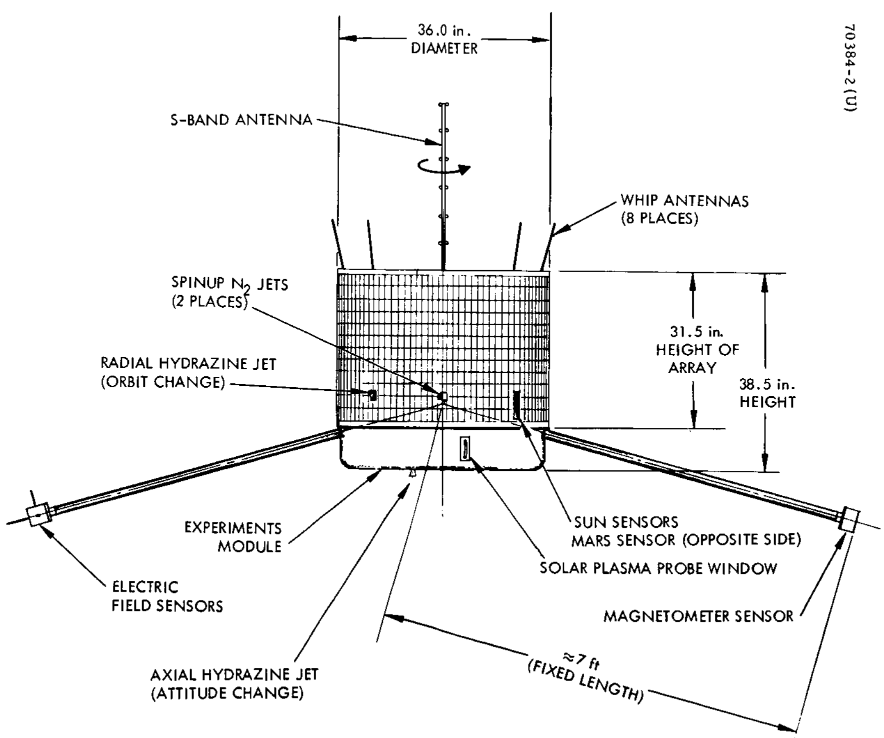

Figure 2-2. OEC Recommended Configuration External Profile

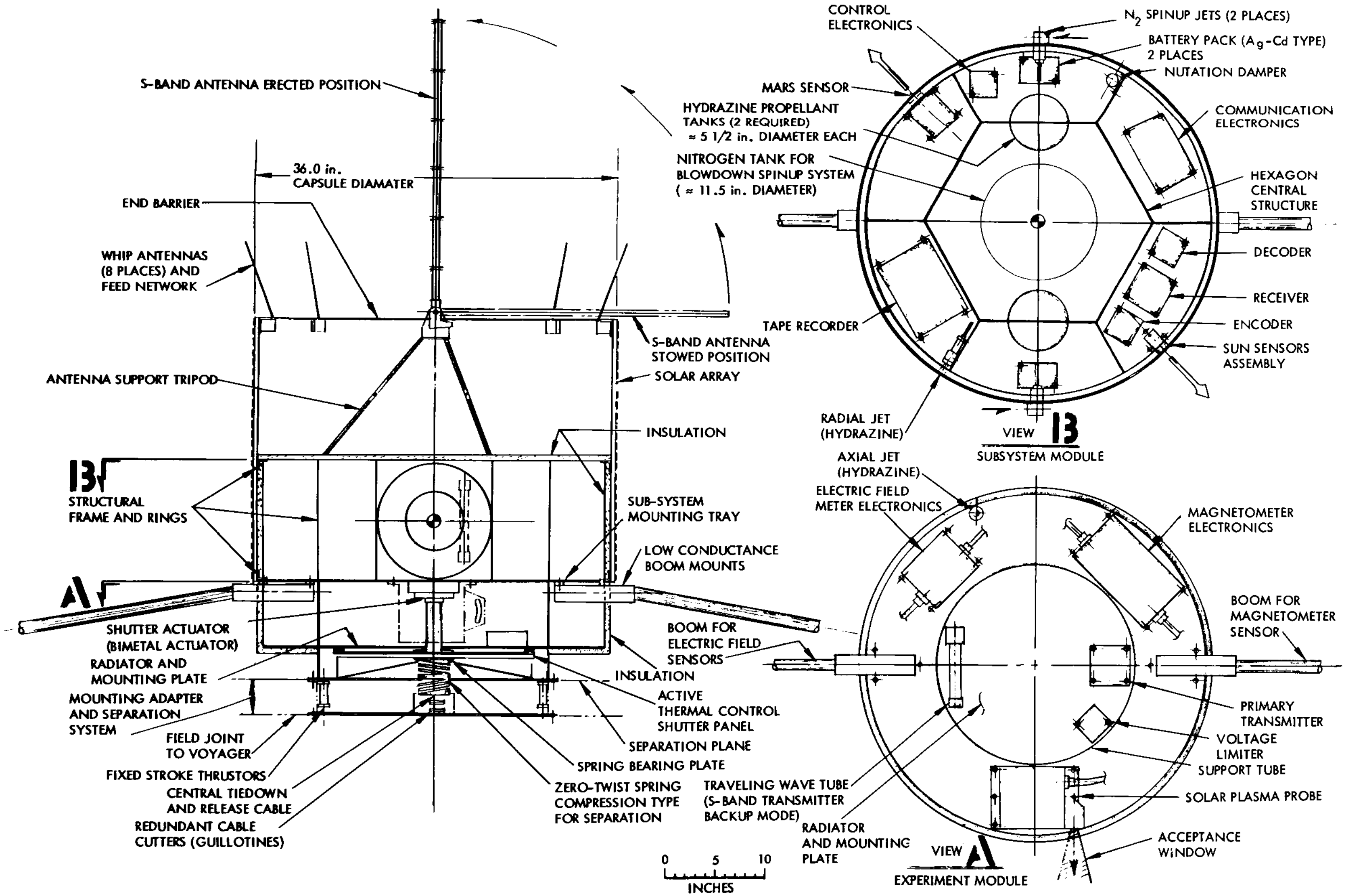

Figure 2-3. OEC Recommended Configuration General Arrangement

The capsule is a cylindrical drum-shaped configuration, the major surface area consisting of solar cell array. Experiments are housed in a cylindrical module below the array. Two radial booms for a magnetometer and electrical field sensor are mounted off the experiment module. At the upper end of the solar array, eight whip antennas are mounted serving the primary communication system; centrally mounted at this end is the deployable S-band antenna. A thin nonstructural thermal end barrier covers the upper end of the capsule and a light-weight tripod mounted to primary structure supports the S-band antenna mount.

Sensors within the capsule demand a clear view through the array, and consequently two small windows are provided, one each for the sun sensor assembly and the Mars planet sensor. In the plane of the capsule's center of mass, two spinup jets are mounted tangent to the cylinder to provide the spinup torque impulse; in addition, for the orbit change capability, a radially oriented jet valve is mounted to thrust normal to the spin axis. To provide attitude and orbit change capability, a single axial jet is located just inboard of the capsule's cylindrical envelope with the thrust vector parallel to the spin axis.

Below the experiment module section is a ring which attaches to the separation mechanism and interface adapter mounted to the Voyager bus structure.

The solar plasma probe requires a clear view to space. A single window is provided in the experiment module's enclosure for a view of the plasma environment normal to the capsule's spin axis.

The design approach pursued was, wherever possible, to maintain a minimum experiment-subsystem interface in terms of packaging. Experiments and associated electronics occupy the lower outboard edge portion of the capsule. The necessary mission supporting subsystems are arranged to occupy equipment bays around a central hexagonal structure. The central portion houses the propellant tanks for the nitrogen blowdown spinup system and dual tanks for the hydrazine system.

To effectively maintain the desired temperature range within the capsule, an active thermal control (ATC) system composed of a bi-metal actuated shutter device is incorporated. The ATC system is centrally located at the base of the experiment module where a non-solar-illuminated radiation corridor to free space is available. High heat dissipating components, such as the transmitters and the power subsystem voltage limiter, have been located on this surface.

The tape recorder and traveling-wave tube amplifier, which are the primary magnetic field contributors, are located as remotely as possible from the magnetometer sensor side of the capsule.

Subsystem and experiment packages are mounted as far outboard on the tray periphery as possible for spin stability consideration. Heavy components such as the batteries, solar plasma probe, propellant storage tankage, etc., are mounted along the transverse axis of maximum inertia, i.e., axis normal to radial booms, so as not to substantially increase the transverse moment of inertia already magnified by virtue of the radial booms and tip mounted sensors.

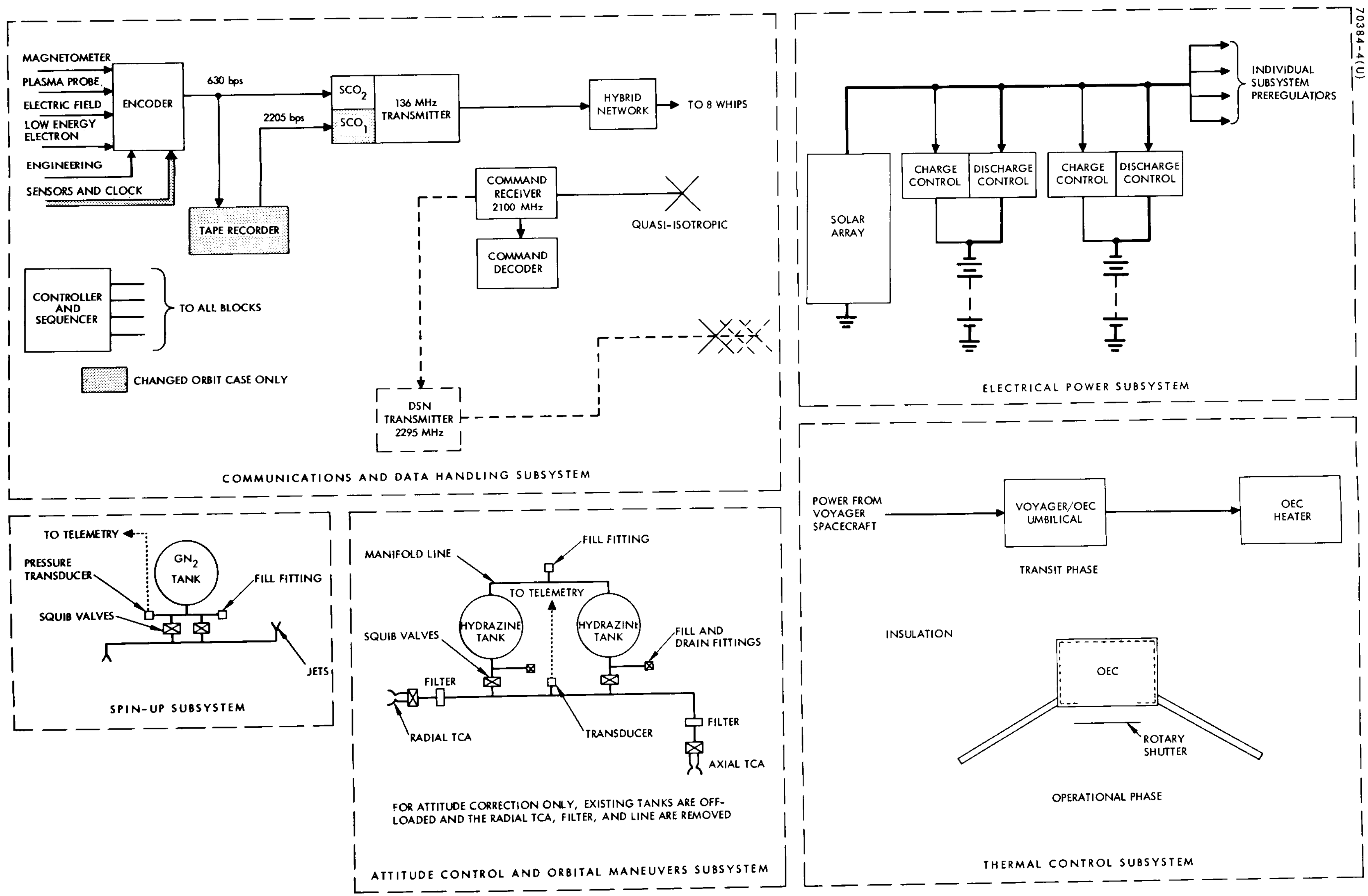

Figure 2-4. OEC Simplified Block Diagram

All surfaces enclosing the subsystems and experiments are insulated from their surroundings, with the exception of the ATC shutter area, so as to minimize the heat losses from the capsule to free space.

Mounting the OEC to Voyager is accomplished by an adapter section that attaches to the Voyager structure. The OEC in turn is mounted to this adapter using multiple pyrotechnic release attachments. A central compression spring of the zero twist type is attached centrally in the adapter and imparts the separation impulse to the OEC.

An itemized weight breakdown of the recommended OEC configuration is presented in Table 2-4. A simplified block diagram for the recommended OEC is shown in Figure 2-4, and an equipment list is shown in Table 2-5. Table 2-6 presents a summary of the OEC performance characteristics.

TABLE 2-4. OEC WEIGHT STATEMENT

		Weight, pounds
Scientific Payload		15.0
Magnetometer and electronics		
Electric field meter and electronics		
Solar plasma probe		
Communication System		17.7
Encoder (2)	4.0	
Transmitter	2.0	
Decoder (2)	2.0	
Receiver	0.6	
Antennas	0.8	
Antenna network	0.3	
Tape recorder	7.0	
Logic electronics	1.0	
S-Band Backup Communication Mode		12.5
TWT amplifier	0.95	
Electronic conversion unit	2.5	
Drive chain	2.15	
Receiver and transponder	3.9	
Diplexer	0.5	
Decoder selection circuits	0.3	
Harness	0.2	
Structure	0.7	
Antenna	1.3	
Power Subsystem		33.2
Solar cell array	13.8	
Batteries	18.0	
Electronics	1.4	

Table 2-4 (continued)

	Weight, pounds		
Limiter	0.3		
Charge boost	0.3		
Controller	0.3		
Discharge circuit	0.5		
Propulsion Subsystems			12.2
N_2 spinup system		4.2	
Propellant	0.5		
Tank	1.2		
Fixed weights	2.5		
Hydrazine attitude control and orbit change system		8.0	
Fixed weights	3.9		
Tanks	0.5		
Propellant	3.6		
Attitude Determination and Controls Equipment			5.2
Sun sensor		0.3	
Mars sensor		2.0	
Nutation damper		0.4	
Sequencer		0.5	
Logic electronics		1.0	
Bracketry and attachments		1.0	
Electrical Harnesses			2.5
Structure			15.0
Fixed radial booms (2)		3.0	
Central frame		4.5	
Rings and bulkheads		4.5	
Brackets and attachments		3.0	
Thermal Control			4.5
Insulation		2.0	
Thermal coatings		0.5	
Radiator panel and actuator		1.3	
Heater		0.7	
Support Adapter and Separation System			5.0
Structure		2.5	
Pyrotechnics		0.2	
Spring mechanism		2.3	
Total Weight			122.8

TABLE 2-5. ORBITAL EXPERIMENTAL CAPSULE EQUIPMENT LIST

Item or Subsystem	Equipment Type*
Communication Subsystems UHF/VHF (relay link)	
Telemetry encoder	Requirements not firm
Transmitter	HS-308
SCO	State of the art design
Whip antennas and diplexer	ATS (Applications Technology Satellite) type modified
Antenna hybrid network	ATS
Command receiver	ATS
Sequencer	State of the art design
Mode control logic	State of the art design
Tape recorder	Types available — various
Communication Subsystem Additional Items for Direct Link (S-band)	
TWT amplifier	Available from communication satellite design
Electronic conversion unit	Surveyor
Drive chain	Surveyor
Receiver and transponder	Surveyor
Diplexer	Surveyor
Command decoder	Requirements not firm
Stacked array antenna	State of the art design
Power Subsystem	
Solar array	State of the art
Batteries (Ag-Cd)	State of the art equipment
Voltage limiter	HS-303A project
Charge boost electronics	HS-303A project

*This tabulation is used simply to indicate the type of equipment that would be required in terms of complexity level. It does not indicate a selected subsystem or component for this mission.

Table 2-5 (continued)

Item or Subsystem	Equipment Type
Battery controller	HS-303A project
Discharge circuitry electronics	HS-303A project
Wiring Harnesses	
Attitude Determination and Control Subsystem	
N_2 cold gas blowdown spinup system	ATS
Sun sensor assembly and amplifier	ATS project
Mars sensor and amplifier	HS-308
Nutation damper	Early Bird
Hydrazine propulsion system	New items
Thermal Control	
Heater	Surveyor
Active shutter device	ATS
Shutter actuator (bi-metal)	ATS
Structure	
Main frame	New item
Booms (fixed)	Comstock and Wescott
Mounting and Separation System	
Adapter	New item
Zero-twist compression spring	Kinemotive Corp., Soehner Spring
Guillotines	Holex Inc.
Explosive bolts	Holex Inc.
Pinpushers	Holex Inc.

TABLE 2-6. SUMMARY OF PERFORMANCE CHARACTERISTICS

• Sun line accuracy determination	<0.25 degree
• Spin axis accuracy determination	≤1.0 degree
• Periapsis position accuracy	<10 km
• Apoapsis position accuracy	<80 km
• Spin axis alignment to ecliptic normal	<5 degrees
• Minimum operating altitude	~350 km
• Disturbance torque attitude errors at 350 km over 6 months	<5 degrees
• Separation velocity increment	~0.1 fps
• Separation distance (no orbit change)	<3500 km
• Baseline orbit change capability	ΔV = 230 fps
• Baseline attitude reorientation capability	θ = 360 degrees
• N_2 spinup system (two jets — blowdown)	130 millipounds thrust
• Hydrazine control system	3 pounds thrust
Axial jet — Attitude control, pulsed mode	0.2262 degree/pulse
Orbit control, continuous mode	1.0 fps/pulse
Radial jet — Orbit control, pulsed mode	0.06 fps/pulse
• Nutation damper time constant	10 minutes

Some of the major advantages provided by the configuration selected are highlighted as follows:

- The modular arrangement approach lends flexibility to the choice of instrument payloads since they are physically isolated from the major capsule subsystems.
- Radial booms mounted from the experiment module eliminate any potential shadow problems on the solar cells.
- Separation schemes are flexible without significant modification to the capsule design.
- The solar array can be a single unit design and need not be fabricated in sub-units.
- The components with high magnetic properties can be located remotely from the magnetometer sensor.
- The propulsion systems tankage can readily be mounted in the plane of the capsule's cg.
- Exhaust impingement by the propulsion system jets is avoided since they are directed and located away from the experiments
- A minimum interface between the experiments and subsystems is required.
- A straightforward spring energy separation scheme is feasible.
- The choice of radial booms allows additional growth capability should additional separation distance of the magnetometer sensor be necessary, stowage envelope permitting.

2.3.2 Subsystem Description

A brief description of each of the OEC recommended configuration subsystems is presented in the following paragraphs.

2.3.2.1 Communications and Data Handling Subsystem

The communications subsystem necessary for accepting commands from Earth for data retrieval and for sequencing vehicle maneuvers is most readily described by the block diagram shown in Figure 2-4. (Tradeoff data substantiating these choices are presented in Volume II, Section 3.0.)

VHF Antenna. The VHF antenna must serve the 136 MHz transmitter with coverage over a 160 degree omnidirectional region. Combined feed line losses and gain should be greater than -6 db with respect to isotropic.

VHF Transmitter. A 136 MHz solid-state transmitter will provide 33 watts of RF power for 55 watts input. Existing designs can be combined in units providing increments of at least 20 watts each. A series phase modulator should be capable of modulating two subcarrier oscillators on the carrier with modulation indices of 0.53 radian and 1.1 radian. These will permit real time data to be transmitter simultaneously with data which have been stored on the tape recorder and will also permit a real time backup mode in event of a tape recorder failure. The power penalty to the subcarrier for stored data is only 1 db.

Encoder. Requirements for the encoder cannot be made firm until answers regarding possible experiment format and desired commutation modes are known. The small number of experiments and the modest data rates — ~630 bits/sec — indicate that an existing encoder or a minor modification of an existing one will be applicable.

Controller and Sequencer. Requirements are not firm for the same reasons stated for the encoder; however, the accuracies and number of sequences outlined for this application appear to be within the capabilities of an ATS type of clock and sequencer and are certainly within the capability of the Surveyor central controller.

Decoder. A suitable demodulator and decoder using 40 percent microminiature circuits is presently being flight-qualified for a larger, more complex satellite. This decoder will be simplified and circuitry for using the Hamming (15, 10) code for double error detection and single error correction will be added. The coding circuitry has been designed using two "flat packs."

Tape Recorder. A tape recorder (Leach model 2200LP) has been designed especially for low power satellite use. A single recorder will be used, fitted with heads for two channels and programmed so that all data can be conserved even if it should be necessary to have two different readout periods per orbit. Data will be entered at 630 bits/sec with tape moving at 0.5 ips. The present tape capacity is 1800 feet, and only 1100 feet is required for the OEC. Readout to read-in ratio is only 4-to-1, so that a single drive motor can be used.

S-Band Transponder. The Surveyor transponder may be used preceded by a tunnel diode amplifier for improved noise figure. The carrier tracking loop should be modified for a $2B_{LO}$ of 20 Hz, and automatic search capability must be added.

S-Band Antenna. A 7 degree omnidirectional antenna using 30 inches for a collinear array must be developed. A circularly polarized element has been designed at UHF and can be scaled for this application.

2.3.2.2 Power Subsystem

All elements of the proposed OEC power subsystem are within the state of the art and based upon successful space flight experience. Designs proposed here utilize the background obtained from flight-proven hardware.

The OEC electrical power subsystem provides all on-board power for each of the using subsystems and experiments. Prime elements of the power subsystem are a cylindrical solar array, two parallel connected silver-cadmium sealed secondary batteries, and two charge-discharge controllers. Power distribution is accomplished through a main bus to the using loads. Pyrotechnic busses can be provided as required and should be connected directly to the batteries.

Primary power is supplied by the solar array during sunlight periods of the orbit for constant loads and for battery recharging. The batteries provide power during eclipse operations and for peak loads exceeding solar panel power output during sunlight operations. The battery charge-discharge controllers provide controlled charge and discharge conditions for each battery.

2.3.2.3 Propulsion Subsystem

Spinup. The modest 33 lb-sec total impulse requirement to spin up the capsule to 60 ± 10 rpm allows the simplest possible nitrogen gas blowdown system to be utilized satisfying weight and envelope restrictions. The thrusters are limited to 130 millipounds in order to avoid undesirable translational effects. Studies conducted leading to thrust level selection are presented in the separation studies included in Volume II, Section 2.4.

Attitude Correction and Orbit Change. The relatively large velocity increment (ΔV approximately 300 fps) precludes a cold gas system for this function. Of the possible choices, only catalytic monopropellant hydrazine is attractive. A 700 lb-sec system was selected for the design presented and is shown to be well within weight and envelope restrictions. Due to the relatively low weight of the hardware, it appears desirable to design the system with the largest possible propellant tanks and offload for missions requiring lower total impulse.

The propulsion systems recommended are presently within the state of the art, so that no serious problems in development are expected.

2.3.2.4 Sensors Subsystem

Two sensors have been selected to provide the information necessary to yield attitude and position measurements for the OEC. These two types of sensors – a sun sensor assembly and a Mars horizon sensor – have been chosen as the two instruments to be integrated into the OEC baseline configuration. A description of each of these sensors is presented in this volume.

A simple method of measuring attitude to the sun from a spinning vehicle is to produce a sun pulse with a slit optics type sensor. The width and orientation of the slit on the vehicle define the width and shape of the sun pulse. A lower limit on pulse width is set by the angular subtense of the sun. By aligning two of these slit fields of view at some preselected angle to one another, it is possible to measure the angle between the satellite spin axis and the sun line.

The horizon sensor contemplated for the OEC mission is simply a horizon crossing indicator operating in the infrared (IR) spectrum. When used in conjunction with a spinning satellite such as the OEC, a signal is produced each time the leading and trailing IR limbs of the planet are crossed. The sensor element itself is sensitive to the different energies received as it passes from space to Mars during a spin cycle. OEC attitude is determined by measuring the time difference of the leading and trailing edge crossings which is proportional to a scanned chord of Mars. The accuracy of this sensor is ± 1.5 degrees for the chord length measurement.

The sun sensor establishes the angle between the sun line and the OEC spin axis within a 360 degree cone of uncertainty about the sun line. By measuring the angle from the spin axis to Mars, the spin axis can be uniquely determined. Because of the inherent stability of the OEC and the mission characteristics, the process of attitude/orbit determination does not require real time operation. Hence, the establishment of the attitude can be accomplished over a number of days. This allows for large collection of raw data indicating within the basic accuracy of the sensors what the attitude is. In order to meet the orbital position requirements, data obtained from the Mars sensor is used in conjunction with ranging from DSN on S-band.

2.3.2.5 Structure Subsystem

The structural frame for the recommended OEC configuration consists basically of the following major sections:

1) Central hexagonal support structure

2) Central mounting tray bulkhead

3) Base mounting/radiator tray

4) Solar array support rings

5) Support tripod for stacked array antenna (S-band)

6) Base support tube

7) End closure bulkheads/thermal barriers

8) Mounting interface adapter incorporating the separation mechanism and providing the OEC-Voyager mechanical interface

The booms are not considered here as part of the primary structural subsystem, but are discussed in detail in the configuration studies material presented in both Volume II and this volume.

The primary supporting structure is the central hexagonal frame to which all members are attached; loads are transmitted and carried through to the base support tube which is attached to the mounting interface adapter. Connected to the hexagon structure is the primary mounting tray bulkhead which extends outboard to support the base of the cylinder solar cell array. At the top of the central frame a lightweight bulkhead provides the upper closure and secondary attachment surface for the solar array. So as to provide a surface and support for the S-band antenna mount and deployment device at the top of the capsule, a tripod is mounted off three corners of the hexagon frame and extends upward to a central point at the top edge of the solar panel. Below the main tray a cylindrical support tube extends downward to the separation flange. Within this tube, near the base, a mounting tray/radiator surface is provided to support the high power dissipation components.

In addition to being the primary support structure, the central hexagon frame serves as the mounting structure for the propulsion systems tankage.

Within the base cylinder of the structure, a circular plate is centrally supported by three light gusset plates and serves as the bearing surface for the separation compression spring.

The proposed structure is considered to be fabricated using nonmagnetic aluminum alloys, with preventive measures taken at the mounting surface to provide an additional oxide coating to minimize any potential solid phase welding (cold welding) occurring at the contact surfaces which might jeopardize successful separation of the OEC. In view of the low compressive stresses and low temperature conditions at the static contact surfaces, solid phase welding is considered to be quite a remote possibility.

2.3.2.6 Thermal Control Subsystem

The recommended thermal design that has evolved from this study is composed of an insulated body which is variably coupled to the external environment at the one end of the capsule. The insulation consists of multilayer mylar blankets, and the variable coupling is achieved with a rotating shutter of the type built and flown as part of the Hughes Applications Technology Satellite program. The shutter, which would be rotated by a bi-metal actuator, will provide a temperature sensitivity of the equipment mounting surfaces inside the OEC of approximately 1°C per watt of internal power dissipation when the shutter is within its operating range. This temperature variation includes the effect of the sun angle uncertainty of ±25 degrees and the seasonal variation in the solar flux from aphelion to perihelion.

The shutter is a 2 square foot area circle with 1 square foot of pie-shaped holes cut in it. The radiator under the shutter is 1 square foot of pie-shaped areas painted white, located so that when the bi-metal actuator is 21°C (70°F) the shutter is "open," i.e., the holes in the shutter are over the white painted pie-shaped areas of the radiator. When the bi-metal actuator is 13°C (55°F) the shutter is "closed," i.e., the pie-shaped white radiator areas are covered by the shutter. The thermal capabilities and the predicted temperatures with this active temperature control system are shown in Table 2-7.

2.3.3 Voyager/OEC Mating and Stowage

Presently three contractors (GE, TRW, and Boeing) are involved with the design phase of the Voyager spacecraft bus. During the feasibility study the designs of each of these contracts were studied to ascertain the interface implications of mounting the OEC on any of the potential design candidates formulated to date. Assessment of all three designs indicates that mounting of the capsule should be restricted to the peripheral volume external to the spacecraft between the Voyager solar array plane at the base and the Lander interface at the forward end. In terms of stowage volume availability, it appears that the OEC recommended configurations could be accommodated by all three designs: the GE concept offers the maximum space, while the Boeing design imposes the tightest envelope for stowage of the OEC due to their proposed solar array stowage and deployment concept.

Figure 2-5 depicts the overall Voyager envelope indicating the areas in which stowage of the OEC might be considered. Figure 2-6 depicts the conceptual arrangement of mounting adapter and separation system.

2.3.4 Operational Characteristics

Prior to discussion of the sequence of mission operation, the effect of the large Mars-Earth communication distance must be introduced. Time delays in the reception of OEC sensor data must be evaluated to ascertain whether any basic real time operational limitations exist.

The relative distance between Mars and Earth is minimum at opposition, 56×10^6 km when Mars is at or near to the perihelion of its orbit. When conjunction coincides with the aphelion of the Martian orbit, the Earth-Mars distance has its maximum value of some 400×10^6 km. Based on these extremes, the time delay associated with communications between these planetary distances lies between 3 and 22 minutes. For the Voyager 1973 mission, a time delay of approximately 15 minutes is assumed. Hence, a minimum round trip time of 30 minutes is required to receive data and transmit a command from Earth to the OEC in orbit about Mars.

The process of real time attitude or orbit determination could be complicated by the time delay. This is of importance largely during a maneuver. Assuming a 15 minute command transmittal delay time, the confirmation of command reception would take a total of 30 minutes. Evaluation of the maneuver requires reception and processing of both the Sun and Mars sensor data taken for a period following the completion of the maneuver. This period depends on the geometry and whether Mars is in the horizon sensor field of view. The time necessary to process this data and establish the proper altitude correction could then take on the order of 2 to 4 hours,

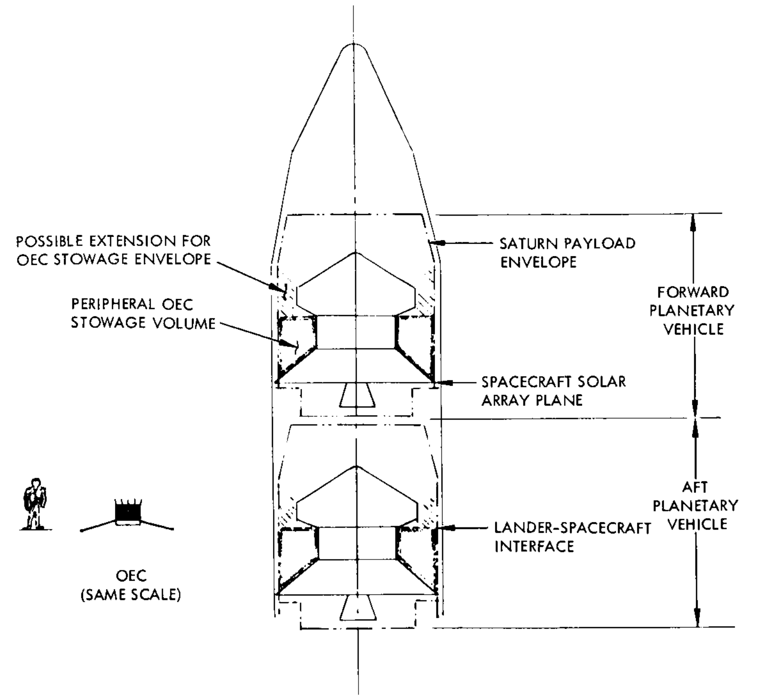

Figure 2-5. Voyager Envelope Considerations for Stowage of OEC

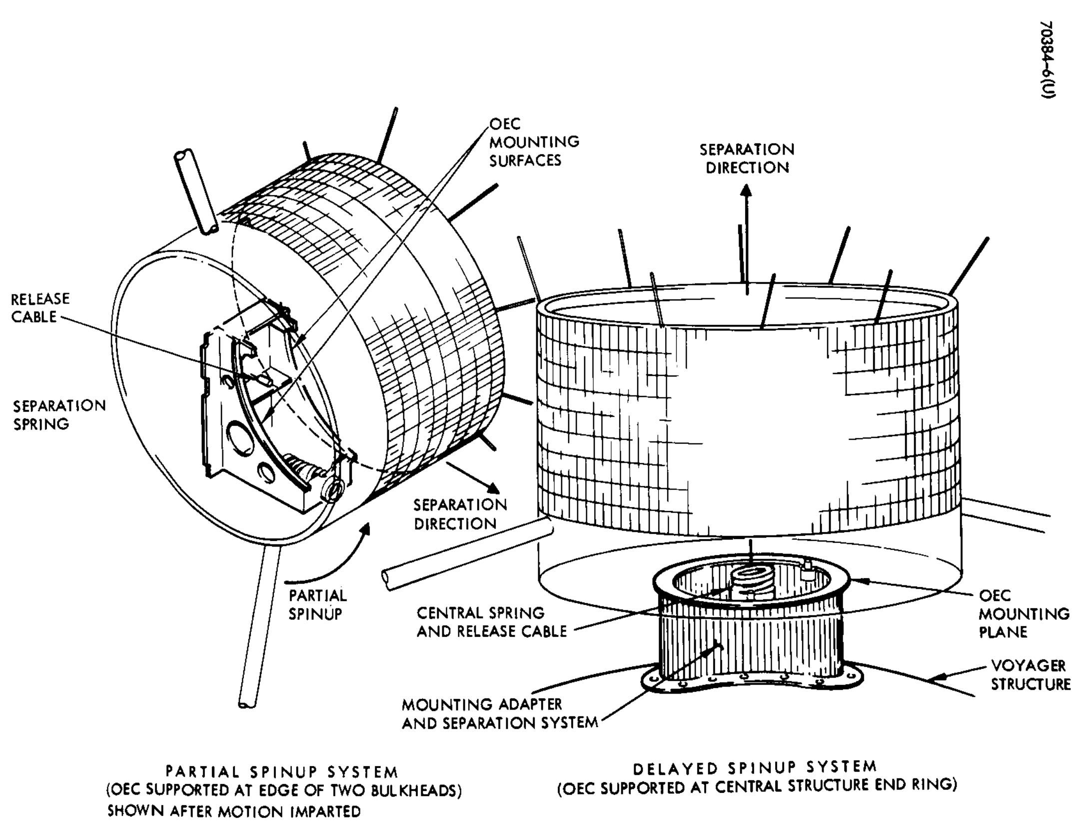

Figure 2-6. Conceptual Arrangement of Mounting Adapter and Separation System

depending to a great extent on the magniture of the orbit maneuver contemplated. To better understand the pseudo-real-time maneuver operation, an example of the attitude-orbit control sequence is developed.

A ΔV of 200 fps is sufficient to lower periapsis of the nominal orbit to 350 km. This requires reorientation of the OEC and thrusting at apoapsis. Procedure for the reorientation maneuver is similar to initial orientation. Since the attitude and orbit are well established prior to this maneuver, the magnitude of attitude correction can be stored in the OEC central sequencer at any time prior to initiation. In the worst case, a change in attitude of 90 degrees might be necessary. The total time to make this maneuver is determined from the total number of axial jet pulses required. A 3 pound axial jet was sized to provide 0.14 degree/pulse.

The total number of pulses required is 640. Therefore, 11 minutes are necessary to perform this change for the nominal OEC spin speed of 60 rpm. Because of the importance of precise attitude alignment prior to a maneuver, the adjustment could be extended over a longer period of time by dividing it into several corrections, each being studied prior to commanding the next.

Upon completion of the attitude correction, the orbit maneuver is initiated. Either the axial or radial jet is applied. If a pulsed radial jet is used, the per-pulse correction of 0.04 fps indicates the execution of 5650 pulses. At 60 rpm. the total time required is 95 minutes.

For this case, the orbit maneuver is conducted at apoapsis. From Section 2.2, Volume II, it is seen that the OEC time spent near the apoapsis (±20 degrees) of this particular orbit is almost 2 hours. The orbit correction is completed and verified within the first orbit. Following confirmation of the change, the attitude is redetermined and corrections to reestablish the spin axis collinear with the ecliptic normal are made.

An identical sequence is followed when using the axial thruster. However, the continuous mode operation must be used. The ΔV added per revolution in continuous operation is 17 times greater than that in the pulsed mode since the pulse spin angle is 21 degrees. Therefore, the maneuver could be completed in just 6 minutes. This is a far more reasonable operation time.

There is one constraint imposed on maneuvers that deserves mention. If the correction is made over several orbits, the attitude to the Sun must be preserved to maintain the solar array at the minimum allowable solar aspect.

Maneuvers may require reorientation of the spin axis normal to, or into the plane of the orbit about Mars. This change is bounded by the inclination of the orbit to Mars which is 20 to 70 degrees. Hence, for low inclined orbits, the attitude adjustment could be as great as 45 degrees. There is then a solar array peak power degradation of 30 percent during this orientation. In this case, a maneuver mode must be established.

TABLE 2-7. THERMAL CAPABILITIES AND PREDICTED TEMPERATURES

	Thermal Capability, °C		Predicted Temperature, °C	
	Non-Operating	Operational	Transit Phase Non-operating	Orbiting Mars, Operational
Batteries (silver cadmium)	-29 to -7 (-20 to 20°F)	4 to 27 (40 to 80°F)	-29 to -7 (-20 to 20°F)**	4 to 27 (40 to 80°F)**
Experiments*	50 to 80 (-58 to 176°F)	-10 to 55 (14 to 131°F)	-34 to -1 (-30 to 30°F)**	4 to 32 (40 to 90°F)**
• Solar plasma probe				
• Electric field meter electronics				
• Magnetometer electronics				
Spacecraft electronics	-32 to 57 (-25 to 135°F)	-18 to 52 (0 to 125°F)	-32 to -1 (-25 to 30°F)**	-1 to 38 (30 to 100°F)**
Tanks and valves for hydrazine	-73 to 60 (-100 to 140°F)	4 to 60 (40 to 140°F)	-34 to -1 (-30 to 30°F)	4 to 32 (40 to 90°F)
Solar panel	-184 to +120 (-300 to 248°F)	-184 to +120 (-300 to 248°F)	-192 to 65 (-314 to 150°F)	-157 to -22 (-250 to -8°F)

*The temperatures shown are the limits presently assumed.
**Mounting surface temperatures.

2.3.5 Sequence of Events

Based on the previous discussions, the sequence of events and the associated time intervals are established. To illustrate the differences between the co-orbital mode of operation and the orbit change mode, two sequences are defined. Table 2-8 and 2-9 list the events. A time history is illustrated for the baseline OEC in Figure 2-7.

2.4 OEC-IMPOSED REQUIREMENTS ON VOYAGER

In the definition of the OEC mission and configuration, of primary importance is the minimization of requirements it would or could impose on the primary Voyager mission or spacecraft. The OEC mission design which was evolved accomplishes this objective. These requirements are briefly discussed below.

Stowage and Release Corridor. The study conducted on the three proposed Voyager spacecraft configurations (GE, TRW, Boeing) revealed that the OEC recommended configuration could be stowed and ejected from the spacecraft without imposing any requirements on the spacecraft configuration in terms of equipment location. In fact, the available envelope from these three spacecraft configurations served as one of the constraints in defining the design envelope of the OEC.

Mounting Surface. The mechanical interface between the OEC and the spacecraft will consist of a mounting surface for the OEC. This mounting surface is defined in detail in Section 4.0 of Volume II.

Thermal Interface. The OEC configuration incorporates a heater for maintenance of the required subsystem temperature during the transit phase. Power for this heater is to be provided by the Voyager spacecraft. The power requirements are quite modest, in the vicinity of 10 to 15 watts.

OEC Separation. A capability would have to be provided on the spacecraft to receive the separation command for the OEC. An umbilical is required between the spacecraft and the OEC to provide initiation of the OEC sequences via the spacecraft.

Antenna Requirements. From the spectrum of Voyager orbits presently covered and for continuous transmission of scientific data from the OEC to Voyager, studies have concluded that a virtually isotropic antenna pattern, i.e., full 4π steradian beam, would be required. Actually, strictly adhering to the present spectrum of Voyager orbits defined, a 160 degree pancake beam could provide continuous visibility under all possible orbital conditions, assuming no OEC or spacecraft orbital plane changes.

Data Storage and Relay Capability. For the primary mode of OEC scientific data transmission, a relay link between the OEC and Voyager is required to accept data transmission at a rate of 630 bits/sec. These data in turn are to be stored by the Voyager Orbiter data handling system to be periodically transmitted to Earth

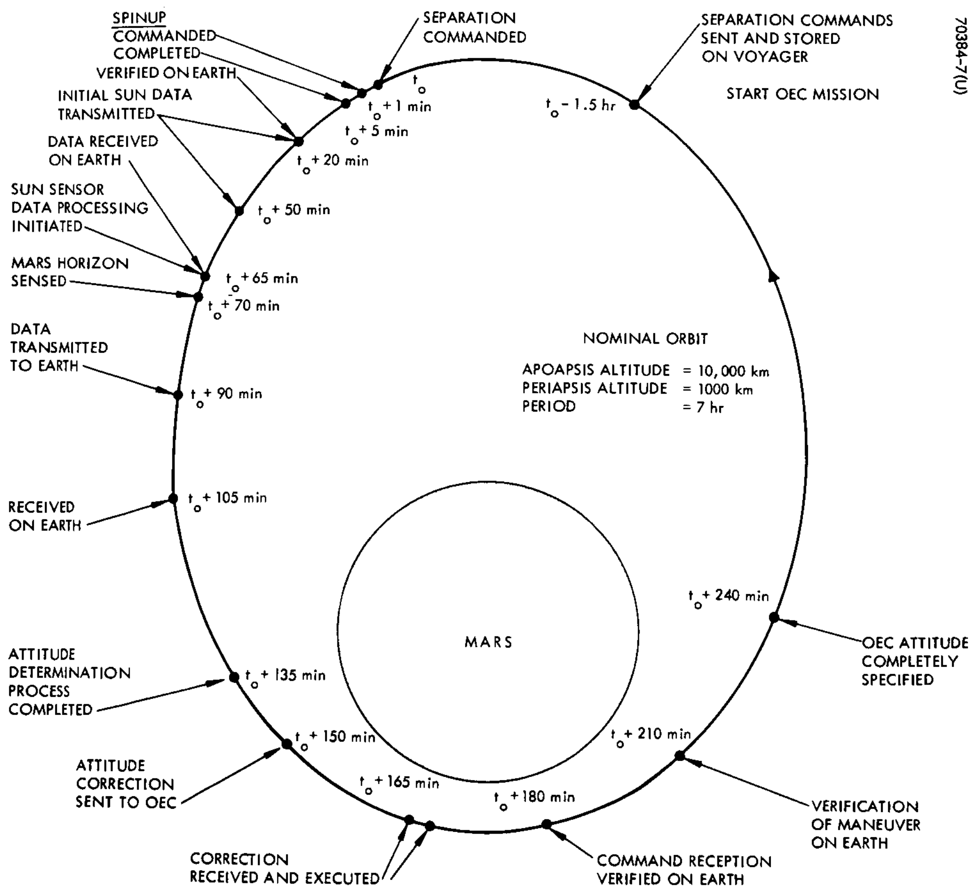

Figure 2-7. Orbital Time History for Baseline OEC

TABLE 2-8. TYPICAL SEQUENCE OF EVENTS - CO-ORBITAL MODE

1) Transit to Mars
   a) OEC temperature controlled using Voyager power
   b) Batteries under continuous trickle charge (NiCd); precharged for (AgCd)

2) Voyager in Mars orbit, landing operation complete

3) Optimum separation time determined (Earth computation)

4) OEC systems activated, internal power connected (Earth command)

5) Separation sequence initiated, timer started (Earth command)
   a) Main structural support of OEC removed (OEC timer)
   b) Power umbilical separated (OEC timer)
   c) Final release, springs push OEC off (OEC timer)
   d) Fire squibs to release spinup gas (OEC timer)
   e) OEC spinup to 60 rpm (automatic consequence of squib firings)
   f) Switch to operational mode (OEC timer)

6) Operational mode
   a) Sample scientific instruments (spin-cycle counter)
   b) Sample sun and Mars sensors (spin-cycle counter)
   c) Sample housekeeping data (spin-cycle counter)
   d) Transmit data to Voyager (continuous—real time)
   e) Control OEC temperature (passive control - power to boom sensors)
   f) Charge batteries (battery controller, as needed)
   g) Provide regulated power (power subsystem)

7) Solar eclipse mode
   a) Switch to battery power (battery controller)
   b) Continue sampling and transmitting (as in item 6)
   c) Control OEC temperature (passive control - power to boom sensors)
   d) Provide regulated power (power subsystem)
   e) Absorb power surge on emergence from eclipse (bus limiters)
   f) Use horizon sensor pulse to activate plasma probe (sensor logic)

8) Data received and stored on Voyager (continuous)

9) Data transmitted from Voyager to Earth (Earth command)

10) Sensor data processed to determine OEC attitude and position (earth computation)

EARTH COMMANDS REQUIRED

Activate OEC (1 on-off)	1 time only
Start separation (1 on-off)	1 time only
Transmit data to Earth (1 on-off)	Depends on Voyager storage

VOYAGER COMMANDS REQUIRED

Relay "activate OEC" (1 on-off)	1 time only - umbilical
Relay "start separation" (1 on-off)	1 time only - umbilical

TABLE 2-9. TYPICAL SEQUENCE OF EVENTS - ORBIT CHANGE MODE

1) Transit to Mars
   a) OEC temperature controlled using Voyager power
   b) Batteries under continuous trickle charge (NiCd); precharged for (AgCd)

2) Voyager in Mars orbit, landing operation complete

3) Optimum separation time determined (Earth computation)

4) OEC system activated, internal power connected (Earth command)

5) Separation sequence initiated, OEC sequencer started (Earth command)
   a) Main structural support of OEC removed (OEC sequencer)
   b) Umbilical separated (OEC sequencer)
   c) Final release, springs push OEC off (OEC sequencer)
   d) Fire squibs to activate propulsion (OEC sequencer)
   e) Spin up to 60 rpm (OEC sequencer)
   f) Activate attitude control system (OEC sequencer)
   g) Switch to acquisition (OEC sequencer)

6) Acquisition Mode
   a) Sample and store scientific instruments (spin-cycle counter)
   b) Sample and store sun and Mars sensors (spin-cycle counter)
   c) Sample and store housekeeping data (spin-cycle counter)
   d) Receive ranging pulse(s) (Voyager command)
   e) Transmit ranging pulse(s) (command decoder to telemetry transpond mode)
   f) Receive transmit data command (Voyager command)
   g) Switch to transmit data mode (command decoder to data and telemetry subsystems)
   h) Switch out of data transmit mode (Voyager command or OEC sequencer)
   i) Assess OEC attitude and provide correction commands (earth computation)
   j) Orientation maneuvers set in OEC sequencer (Earth command through S-band)
   k) Process to stored Sun-spin axis angle (OEC sequencer)
   l) Process about Sun line by stored number of pulses parallel with ecliptic normal (OEC sequencer)

7) Operational Mode
   a) Sample scientific instruments (spin-cycle counter)
   b) Sample housekeeping data (spin-cycle counter)
   c) Transmit data to Voyager
   d) Control OEC temperature (passive control power to boom sensors)
   e) Charge batteries (battery controller as needed)
   f) Provide regulated power (power subsystem)

8) Solar eclipse mode
   a) Switch to battery power (battery controller)
   b) Continue sampling and transmitting data (as in item 8)
   c) Switch off attitude control system (attitude control logic)

Table 2-9 (continued)

d) Control OEC temperature (passive control-power to boom sensors)
e) Use horizon sensor pulse to activate plasma probe
f) Provide regulated power (power subsystem)
g) Absorb power surge on emergency from eclipse (bus limiters)

9) Data (including ranging pulses) received and stored on Voyager Earth command, related to OEC - or Voyager sequencer.

10) Data transmitted from Voyager to Earth (Earth command)

11) Maneuvers, sensor and range data processed to determine OEC attitude and position (Earth computation)
a) Time and direction of the attitude and orbit change impulse determined (Earth computation)
b) Attitude and orbit change impulse duration set in OEC sequencer (Earth command to OEC S-band)
c) Precess to stored Sun-spin axis angle (OEC sequencer)
d) Precess about Sun line by stored number of pulses (OEC sequencer)
e) Fire orbit change engine for stored duration (OEC sequencer)
f) Precess about Sun line, reversing (c) (OEC sequencer)
g) Precess spin axis to perpendicular to Sun line and parallel to ecliptic normal (OEC sequencer, attitude control logic)

EARTH COMMANDS REQUIRED

Orientation maneuvers (2 magnitudes)	
Orbit change impulse duration (1 magnitude)	
Activate OEC (1 on-off)	1 time only
Start separation (1 on-off)	1 time only
Obtain ranging data (1 on-off)	Several times only
Obtain data from OEC (1 on-off)	Every orbit
Transmit data to Earth (1 on-off)	Depends on Voyager storage

VOYAGER COMMANDS REQUIRED

Relay orientation maneuvers (2 magnitudes)	1 time only - umbilical
Relay orbit change impulse duration (1 magnitude)	1 time only - umbilical
Relay "activate OEC" (1 on-off)	1 time only - umbilical
Relay "start separation" (1 on-off)	1 time only - umbilical
Relay "obtain ranging" (1 on-off)	Several times only
Relay "obtain OEC data" (1 on-off)	Every orbit
Command OEC data stop (1 on-off)	Possibly every orbit

via S-band link to the deep space net (DSN). The receiving and storage equipment requirement on the Voyager does not represent a new hardware requirement since the equipment presently planned for data retrieval and storage from the Voyager Lander would be used for the OEC operation. This approach does not interfere with the Lander operations since OEC operations do not begin until the Lander operations are terminated. For the backup OEC communication mode (S-band), there is no requirement for data storage and relay capability of the Voyager spacecraft since the data transmission is directly from the OEC to the DSN in a degraded duty cycle mode.

2.5 OEC DEVELOPMENT: PRELIMINARY ASSESSMENT OF COST AND SCHEDULE

2.5.1 Cost

The findings of the studies presented in Volumes I, II, and III were used as technical requirements and parameters for the preparation of a budgetary cost estimate for planning purposes only. Additional assumptions made in order to establish a basis for costing are listed below. Changes in these assumptions would, of course, cause changes in the budgetary cost made.

1) A total program of approximately 7-1/2 calendar years, as illustrated in the Master Phasing Schedule (Figure 2-8) beginning 1 August 1968.

2) A hardware program of:

   a) Three development models:

      (1) Thermal

      (2) Structural

      (3) Prototype (Y-1) to be delivered 1 June 1970

   b) Three flight models:

      (1) F-1, the first, to be delivered 1 July 1971

      (2) F-2

      (3) F-3

3) Two sets of AGE (aerospace ground equipment)

4) All activities predicated on a Mars launch date of 1 July 1973

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2.5.2 Development Schedule

A Master Phasing Schedule is presented in Figure 2-8 which outlines the major steps required for a development, fabrication, and test program leading to the delivery of three flight spacecraft and one prototype model (Y-1). The prototype model delivery is scheduled for 1 June 1970; the first flight model, for 1 July 1961.

The program plan is based on a Preliminary Design Phase preceding this program. That phase is scheduled for 1 January 1968 to 30 June 1968, followed by a month evaluation period. The preliminary design phase is reccommended and scheduled in order to effect maximum savings of time and costs. The preliminary design can be determined on a small program scale with full attention given to the main concern of design, rather than to the buildup of a major program organization. This phase is therefore recommended as a separate, preliminary program for which no other provisions nor costs are determined herein.

The development program is scheduled for a start date of 1 August 1968. The first flight spacecraft would be ready for delivery 23 months after go-ahead, with subsequent spacecraft at 3 month intervals. Two of the three craft will be used for flight and one as a spare. All three spacecraft will then be in a Voyager/OEC test phase until Mars launch in mid-1973.

The pacing items in the program are the prototype spacecraft design, delivery of the prototype spacecraft (Y-1), and integration of experiments, which may impose severe schedule restrictions on the experimenters. A complete spacecraft (Y-1), in essence, is required 1 year before delivery of the first flight model (F-1).

The spacecraft prototype model Y-1 will be preceded by a thermal (T-1) and a structural (X-1) model, built to correct dimensions and mass requirements with the same actual electrical and mechanical connections as in the flight models. The correct form and fit tests as well as the static and dynamic structure tests performed on these two models will be completed by the time the system assembly and integration activity begins on the Y-1 prototype model.

The necessary time for the various interrelated efforts of the program has been realistically allocated, based on the experience accumulated on past satellite programs. Some time parameters and milestones are imposed by the Voyager program and would otherwise not be recommended for the OEC program. Included in these parameters is the lengthy time span between delivery of the prototype (Y-1) and the first flight model (F-1). Another is the length of approximately 2 years scheduled for Voyager/OEC test.

3.0 SYSTEMS ANALYSIS

The OEC mission is a scientific probe of the presently uncharted environment of the planet Mars. As such, there are many valuable operational modes that deserve careful attention.

A major consideration of this feasibility study was to establish the range of alternative missions that could suitably perform within the specifications outlined in Reference 1.

The OEC mission design, configuration, and performance is the result of intensive analytical studies to:

1) Establish OEC optimum mission design requirements

2) Study the broad orbital characteristics as they pertain to mapping of the Martian magnetosphere

3) Analyze effects of separation and the relative motion between Voyager and the OEC to determine major system parameters

4) Establish an accurate means of attitude and orbit determination for experiment data reduction

5) Specify the most direct form of attitude as well as orbit control to provide desired experimental accuracies and to extend experiment results to other regions of Mars

These are the major areas of study which are necessary to establish the characteristics of the various alternative concepts and to evaluate system performance.

Results of the mission-related studies are summarized in this section. The basic interrelation between the Voyager and OEC in terms of the expected orbits, injection characteristics, and relative motion is extremely important in preparing the OEC for the ensuing 6 months of scientific measurements. Orbit analysis also establishes a detailed relationship between the OEC and the planet in terms of the expected lifetime, eclipsing, and requirements for provision of occultation to Voyager. Basic communications field of view requirements are established in these studies for both the Voyager and the OEC to maintain continuous monitoring of data during periods of transmission.

There are two other major areas of concentration in the systems analysis area. Stabilization of the OEC and both attitude and orbit determination are important to the success of the mission. The manner in which the system is to be stabilized has been specified by Ames Research Center (Reference 1) as a spinning satellite. The characteristics of spin stabilization with regard to meeting the accuracies have prompted detailed analytical treatment of the various external disturbances and a means of attitude control to alleviate any deviations in attitude. The attitude control sensors are in fact the attitude/orbit determination sensors which are necessary to relate the scientific measurements from the experimental payload to the planet Mars.

These studies lead to a performance analysis of OEC. The basic capability of the capsule to perform over the spectrum of missions is determined, with the eventual inclusion of an orbit change capability to increase the experimental mapping features of the OEC system.

3.1 ORBIT ANALYSIS

Several items form the nucleus of the orbit-related studies. In particular, the relationship between the orbits specified in the Voyager Mission Specifications (Reference 2) and the Martian magnetosphere must be analyzed to determine the density and coverage of the magnetospheric map, and to provide a guide to the selection of the most desirable orbits.

Table 2-1 of Section 2.1 summarizes facts about the Voyager spacecraft orbits which are pertinent to the OEC design. The altitudes for the elliptical orbits can be chosen between 500 to 1500 km at periapsis to 10,000 to 20,000 km at apoapsis. The orbital periods associated with this range are shown in Figure 3-1. The period varies between 7 and 14 hours.

In general, the variation of the orbital velocities for these orbits is of the same magnitude; however, the greater distances traveled in the case of higher apoapsis altitude orbits are commensurate with increased periods, as shown in Figure 3-2.

Inclination of the Voyager spacecraft orbits is defined in terms of both a Mars ecliptic and equatorial constraint. This range of inclinations is shown in Table 2-1.

3.1.1 Motion of Orbit in Magnetosphere

A principal objective of the OEC mission is exploration of the Martian magnetosphere, if in fact one does exist. Mariner IV data place upper bounds on the size and extent of this magnetosphere. Figure 3-3 shows a view of the largest magnetosphere consistent with the Mariner result. If the magnetosphere is caused largely by induction (Mars interaction with the solar magnetic field), the magnetopause might be as low as 100 to 500 km at its lowest point. To obtain as complete a picture as possible of the magnetosphere and associated phenomena and to assure its discovery if it exists, it is important that the OEC orbit cover as much of the potential magnetospheric volume as possible during its 6 months operation.

Relating the Voyager orbits to the magnetosphere of Mars is a complex process. There are two factors that cause the orbit to move relative to the Martian magnetosphere. First is the motion of Mars about the Sun; second are changes in the orbit due to planetary oblateness.

By uncoupling the motion, the individual contributions can be identified. Assume for the moment that the OEC is in an orbit about Mars which is fixed inertially. Since the Mars-Sun line rotates at 0.5 deg/day during the 6 month mission, the planet will move through 94.5 degrees about the Sun. An assumed magnetosphere is shown superimposed about Mars in Figure 3-4a. The magnetosphere is a solar-referenced phenomenon; as Mars moves in the ecliptic plane, the magnetosphere rotates to always face the Sun. For an orbit fixed about Mars, the relative (longitudinal) motion between any point in that orbit and the magnetosphere is 94.5 degrees.

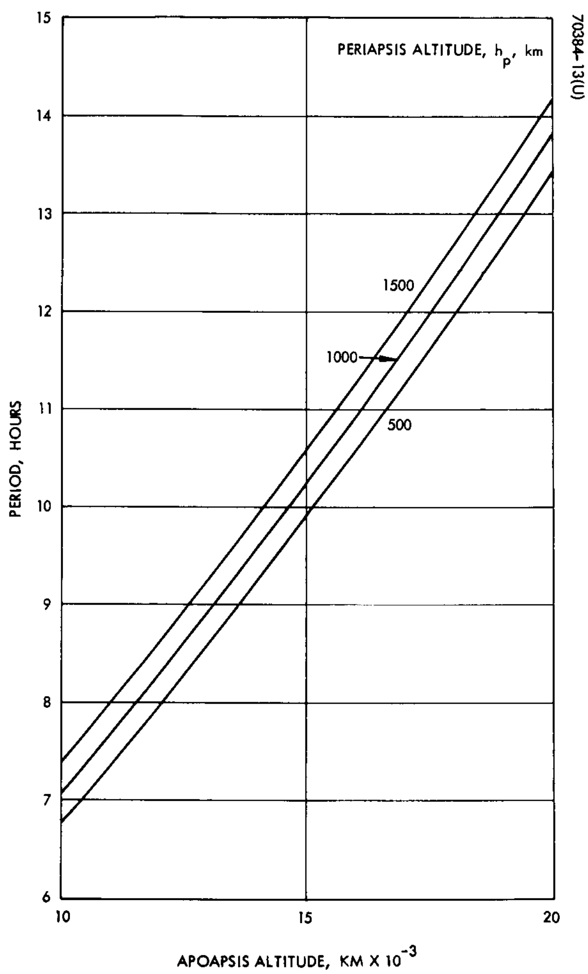


Figure 3-1. Period Versus Apoapsis Altitude

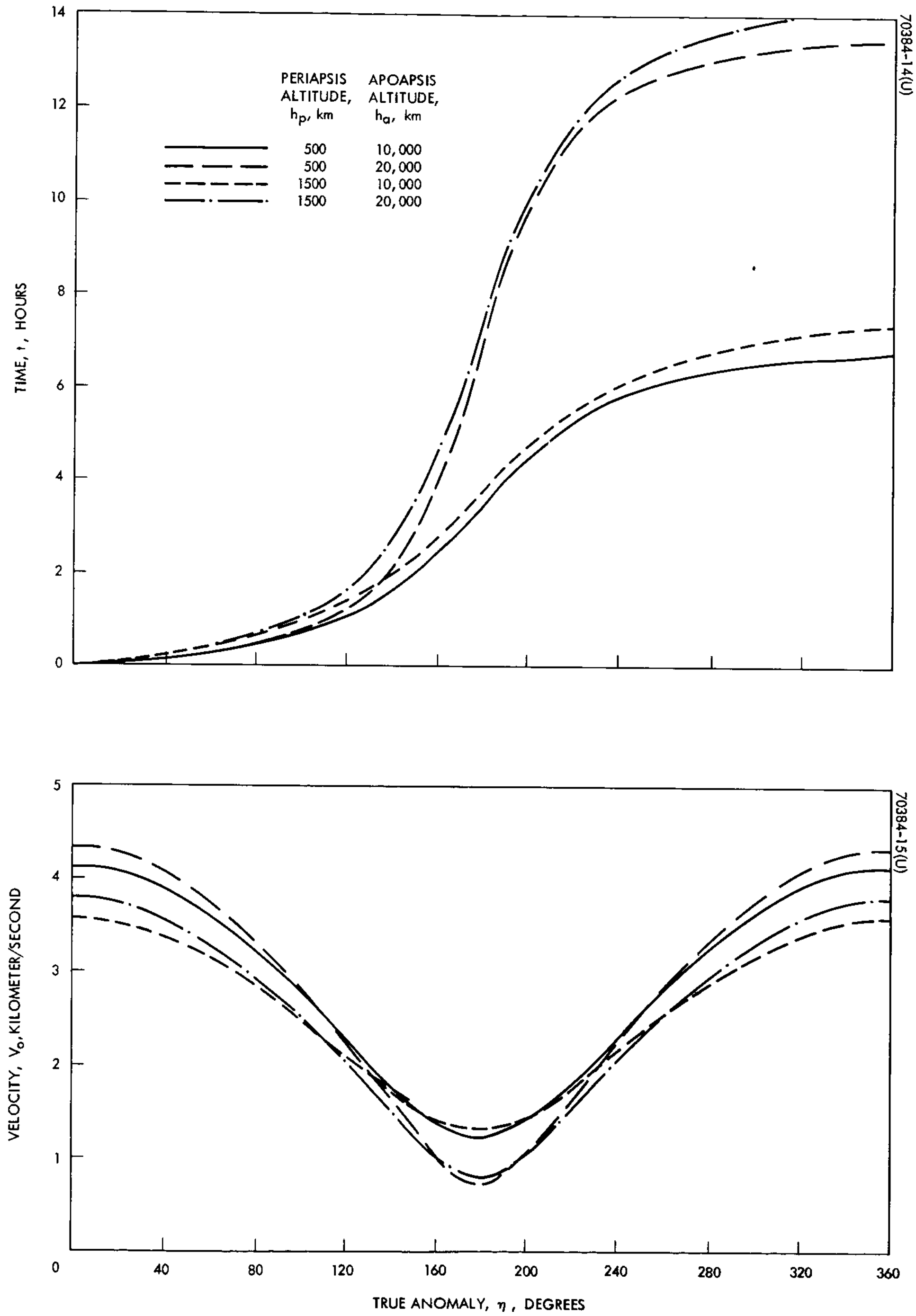
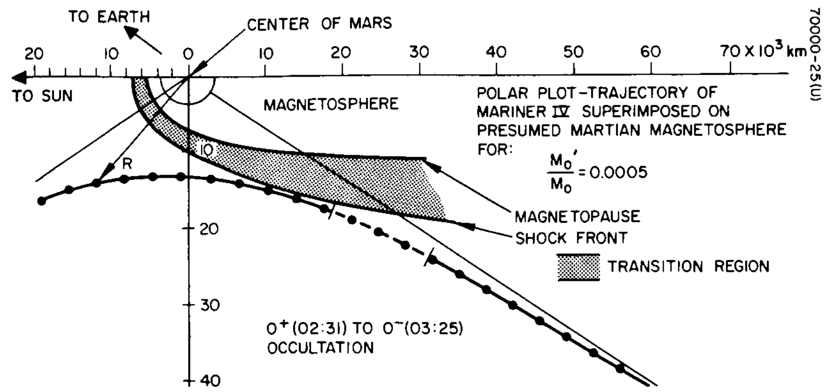


Figure 3-2. Time and Velocity Versus True Anomaly



SUCCESSIVE BLACK DOTS ON TRAJECTORY AT 15-MINUTE INTERVALS

Figure 3-3. Mars Magnetosphere

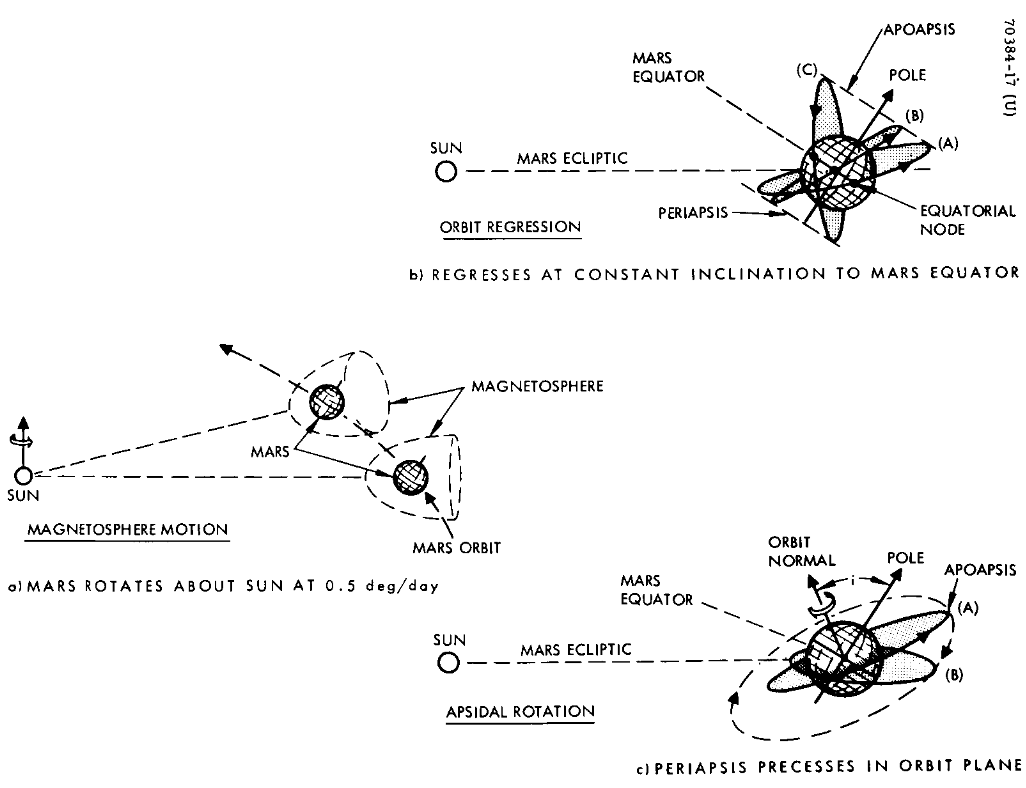


Figure 3-4. Relative Orbit— Magnetosphere Motion

The oblateness of the planet Mars causes the orbit to precess (about Mars) in a known manner. This precession is described in terms of two characteristic motions: orbit regression and apsidal rotation. Orbit regression is defined as a westward precession of the line of nodes about the equatorial plane at a constant inclination to the planet's pole. This is shown in the three orbits A, B, and C of Figure 3-4b. Apsidal rotation represents the motion of the line of apsides (line between periapsis and apoapsis) in the plane of the orbit. This motion is illustrated in Figure 3-4c. The nodal and apsidal precession is plotted as a function of the Voyager OEC orbital inclination (angle i in Figure 3-4c) in Figures 3-5 and 3-6.

The combination of the oblateness effects and the Mars-sunline motion generates a natural mapping of the magnetosphere. As a typical case, consider a 10,000 km x 500 km orbit with the initial longitude of periapsis lined up along the Mars-sunline. As time progresses, the orbit precesses in a clockwise direction, with the apoapsis moving in front of and to the opposite side of the magnetosphere, as illustrated in Figure 3-7. The motion of Mars about the Sun, combined with the orbital precessions about Mars, result in the periapse point traversing through some 230 degrees of heliocentric longitude in the 6 month period, so that data has been gathered over a substantial fraction of the magnetosphere.

Figure 3-8 shows the magnetospheric mapping of the periapse point as a function of the orbit inclination for this 10,000 km x 500 km orbit. The range of inclinations includes all those allowed by the assumed constraints on Voyager orbits (Table 2-1). The latitude-longitude locations of periapse, for a given inclination, trace out a single curve as indicated by the arrows in the figure.

Magnetospheric data is obtained also at points other than orbit periapse, and such data may, in fact, be of significant scientific interest. The orbit sketched in Figure 3-7, for example, is such that apoapsis never enters the magnetosphere, and there is no data obtained at the apoapsis altitude. It is clear that there are many possibilities for different mapping profiles of the magnetosphere, depending on the Voyager orbit selected. In the interest of mission evaluation, and possible Voyager influence, it would be desirable to develop mapping criteria and perform a comprehensive assessment of the relative value of various alternate Voyager (or OEC) orbits. From the data presented here, the conclusion can be drawn that, whatever Voyager orbit is finally selected, the magnetospheric map obtained by OEC will cover a broad region of the magnetosphere spanning at least a 90-degree longitudinal sector.

3.1.2 Orbit Lifetime

As shown later in this section (see 3.5), it is not feasible to significantly change the inclination or position of the line of apsides of the OEC orbit from those of the Voyager orbit. Thus, the only way the OEC can alter its orbit to its scientific advantage is by changes in the altitude of periapse (or apoapse). As indicated earlier, much benefit may accrue to the OEC mission by dropping periapsis as far as possible, consistent with mission constraints; for this reason, an orbit change capability has been included in several of the OEC concepts.

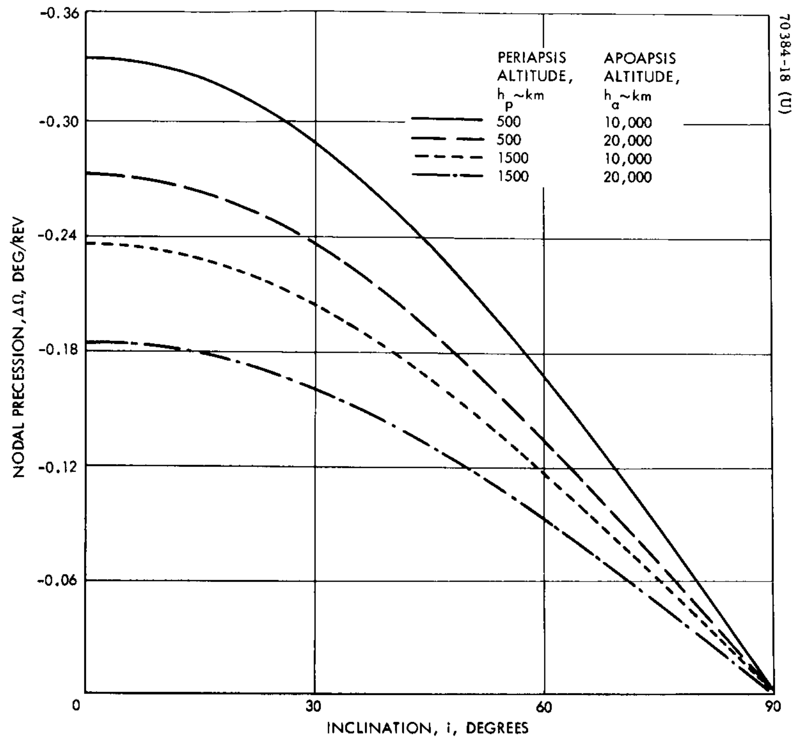


Figure 3-5. Nodal Precession Versus Inclination

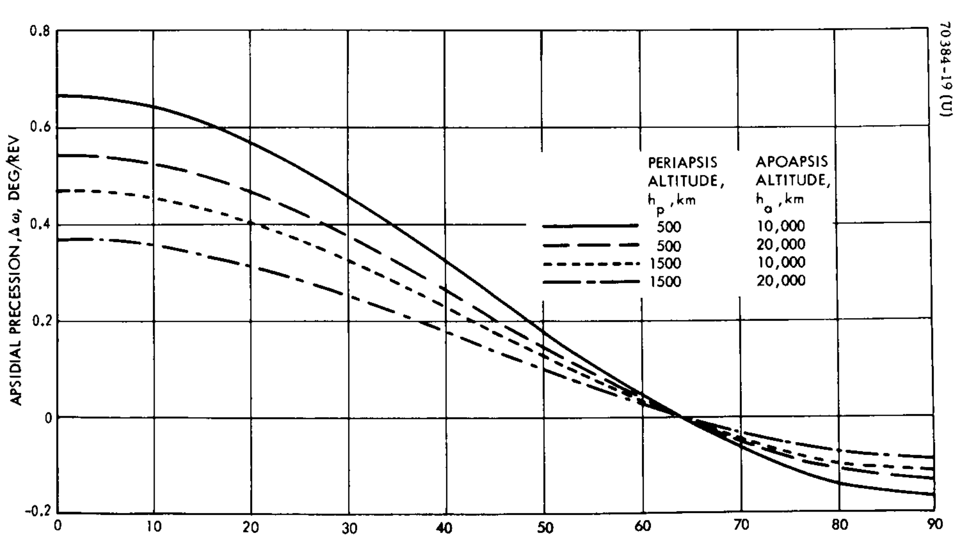


Figure 3-6. Apsidal Precession Versus Inclination

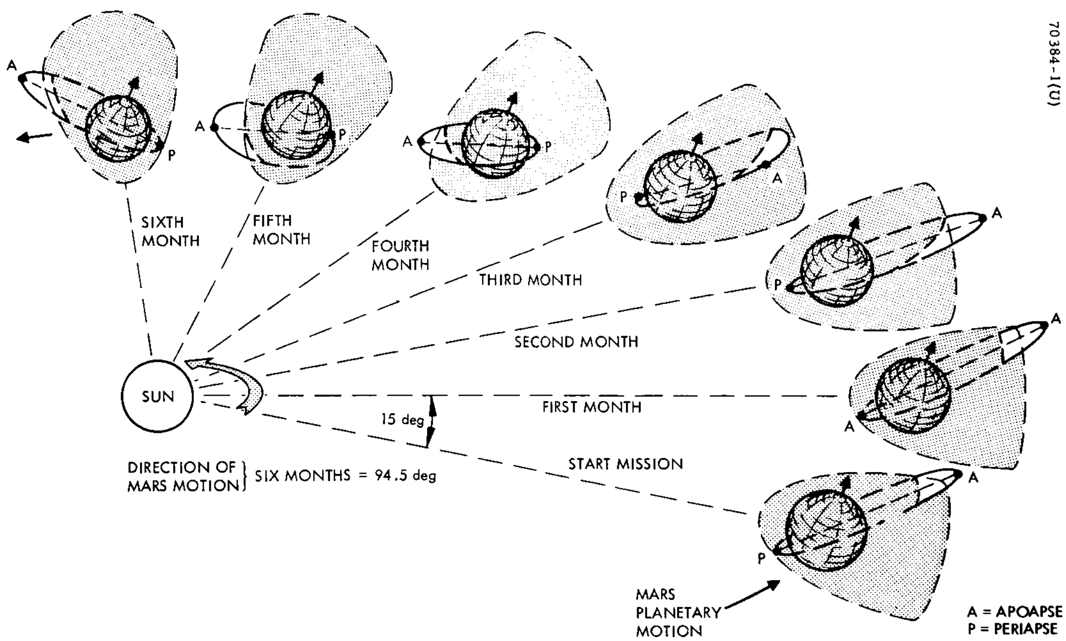


Figure 3-7. Orbit—Magnetosphere Motion

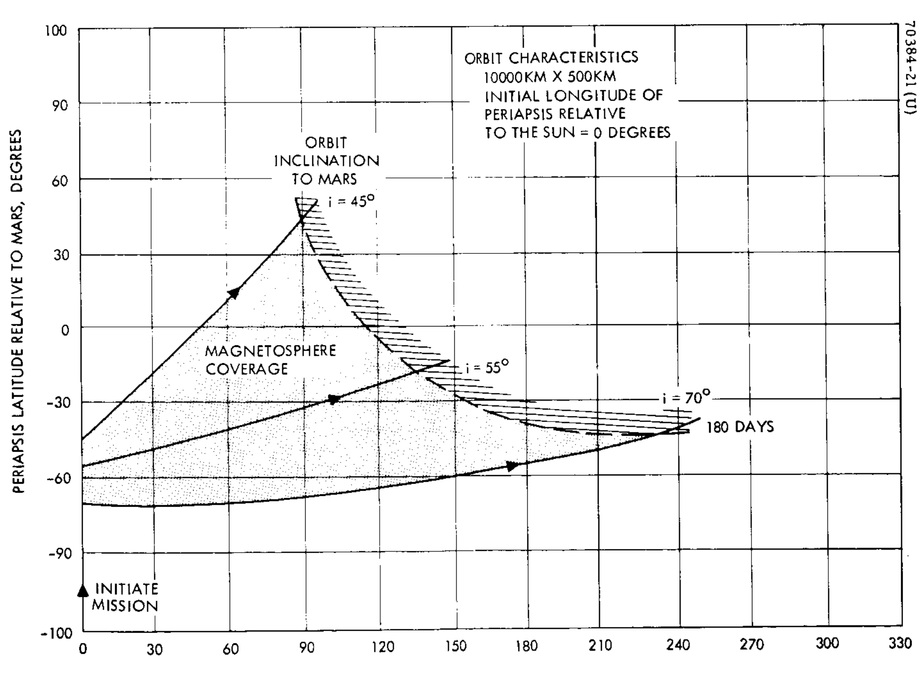


Figure 3-8. Typical OEC Orbit—Magnetosphere Relationship

A study of the atmospheric decay effects on the OEC during the mission was performed to provide an indication of minimum allowable operating altitudes. Ten atmospheric models given by JPL in the Voyager specification (Reference 2) as well as a recent model by D. F. Spencer (Reference 3) were examined; these are listed in Table 3-1.

Figure 3-9 shows the primary results of this study. The atmospheric model labeled VM-9 was found to be the most critical case. With this atmosphere, Figure 3-9 indicates that periapsis altitude should be no lower than 275 km for a 50-year lifetime, or 250 km for a 10-year lifetime. As seen in the curves, the minimum altitude is not sensitive to the initial apoapsis altitudes in the region of 10, 000 to 20, 000 km.

The perturbing effects of the Sun's gravity can be significant in considering lifetime of the OEC, especially in the more eccentric orbits. Preliminary calculations of the solar perturbation effect are reported in Section 2. 2 of Volume II and result in worst case perturbations of periapsis altitude of as much as 50 km in the 6 month period. In cases where these perturbations are large, the largest contributors are periodic with a cycle of half a Martian year — so that the 50 km value should represent a good first estimate of the margin that should be allowed. Detailed computer confirmations of orbit lifetime, including the solar perturbations, should of course be conducted for the specific candidate OEC orbit before final injection into that orbit.

From the data of Figure 3-9, allowing ±50 km for the solar perturbations and an additional margin of safety, 350 km is recommended as a minimum initial OEC altitude. The range of possible periapsis altitudes for OEC over the 6 months, then, is from 300 to 400 km.

3. 1. 3 Solar Eclipse

The duration of the solar eclipses to be experienced by the OEC is an important factor in the thermal control of the OEC and in the design of its power system. The worst possible eclipse period, as shown in Figure 3-10, occurs when the apoapsis of the orbit lies near the ecliptic plane and in the shadow of Mars. For the largest orbit with a 20, 000 km apoapsis, these eclipses could last as long as 2. 7 hours out of a 14-hour orbit period. For the smaller orbits, the worst eclipse is about 1. 6 hours in a 7-hour orbital period.

The constraints placed on Voyager (Table 2-1) require that these "worst case" conditions not be experienced during the first 6 months of Voyager Orbiter operations. Since the OEC may not be placed into orbit until the second month of Voyager orbit (after completion of the landing operation), the possibility that these worst case conditions might occur in the OEC's sixth operating month must be considered. The onset of such long eclipses is illustrated in Figure 3-11 showing the condition where the orbit has apoapse in the ecliptic plane and the Voyager specification (1 hour maximum) is just maintained throughout its 6 months.

In actual fact, the likelihood of this combination of circumstances is low, so that the "nominal" design point for OEC should be the Voyager-specified 8 percent of a period or 1 hour, whichever is least.

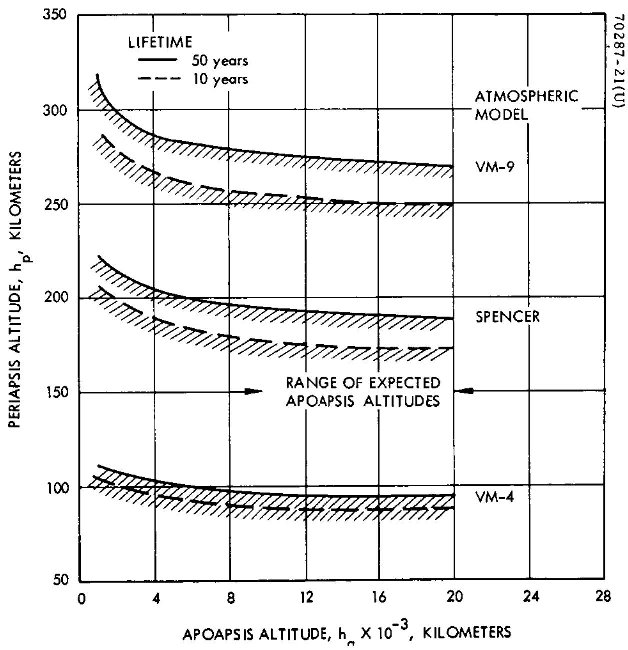


Figure 3-9. Minimum Altitude Limitations

• ECLIPSE PERIOD, < 1 HOUR IN FIRST 5 MONTHS ... POSSIBLY GREATER IN THE SIXTH MONTH

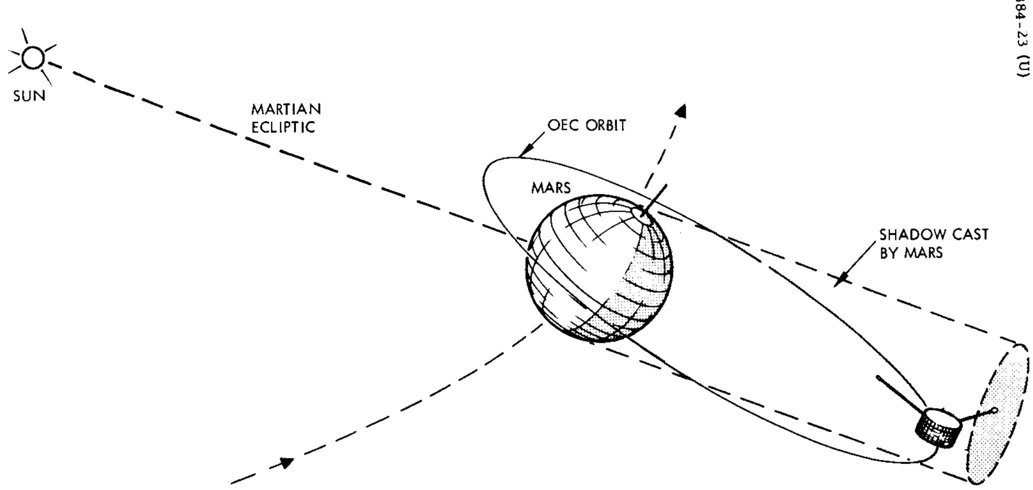


Figure 3-10. OEC Eclipse Studies

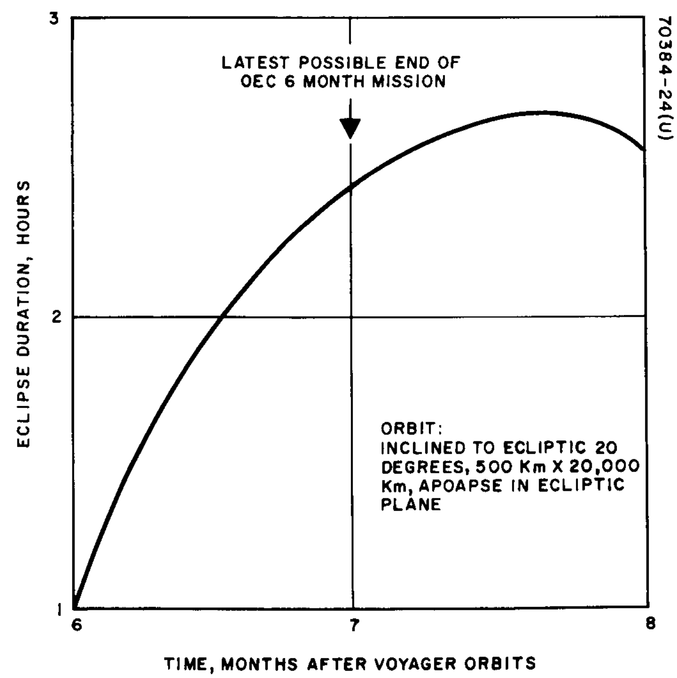


Figure 3-11. Worst Possible OEC Eclipsing

TABLE 3-1. MARTIAN ATMOSPHERIC MODELS

Atmosphere Model	Surface Pressure, millibars	Surface Density, $gm/cm^3 \times 10^{-5}$	Scale Height, km
VM-1	7.0	0.96	14.2
VM-2	7.0	1.85	5.5
VM-3	10.0	1.37	14.3
VM-4	10.0	2.57	5.2
VM-5	14.0	1.91	14.2
VM-6	14.0	3.08	6.1
VM-7	5.0	0.68	14.2
VM-8	5.0	1.32	5.5
VM-9	20.0	2.73	14.2
VM-10	20.0	3.83	6.9
Spencer	6.0	3.00	10.0

3.1.4 Voyager—OEC Orbit Relationship

In the co-orbital mission, the OEC orbit will be almost exactly the same as that of the Voyager Orbiter; a slight difference in orbital period will be present causing the OEC to gradually move away from the Voyager on the same orbit. The distance from OEC to Voyager is a key factor in the OEC design, since it dictates the requirements placed on the OEC primary communication system. Figure 3-12a shows a typical time history of OEC—Voyager range over the 6 months for a co-orbital mission. The range varies periodically over one orbit, while the envelope of these variations increases linearly with time. The rate of increase is in direct proportion to the difference in OEC and Voyager orbit period.

In the case where a periapsis drop is executed (orbit change case), the differences in orbit period are much larger. The range variation has a similar orbital oscillation, but now the envelope of maximum range (during an orbit) oscillates also with a period—for the case shown (Figure 3-12b) of about 20 days. The maximum communication ranges recur periodically throughout the 6 month life. The maximum range is equal to the major axis of the largest of the OEC or Voyager orbits, from 18,000 to 28,000 km for the range of orbits considered.

3.1.5 Voyager—OEC Occultation

Since the Voyager and OEC orbits are coplanar, it is quite likely that Mars will occult the Voyager—OEC line of sight during parts of an orbit. For the co-orbital mission, in which data is transmitted on a real time basis from the OEC, occultation will cause a loss of data. Figure 3-13 illustrates the case in which occultation first occurs when the two vehicles are traveling in the same orbit; geometrical calculations reveal that a periapse separation distance (d of

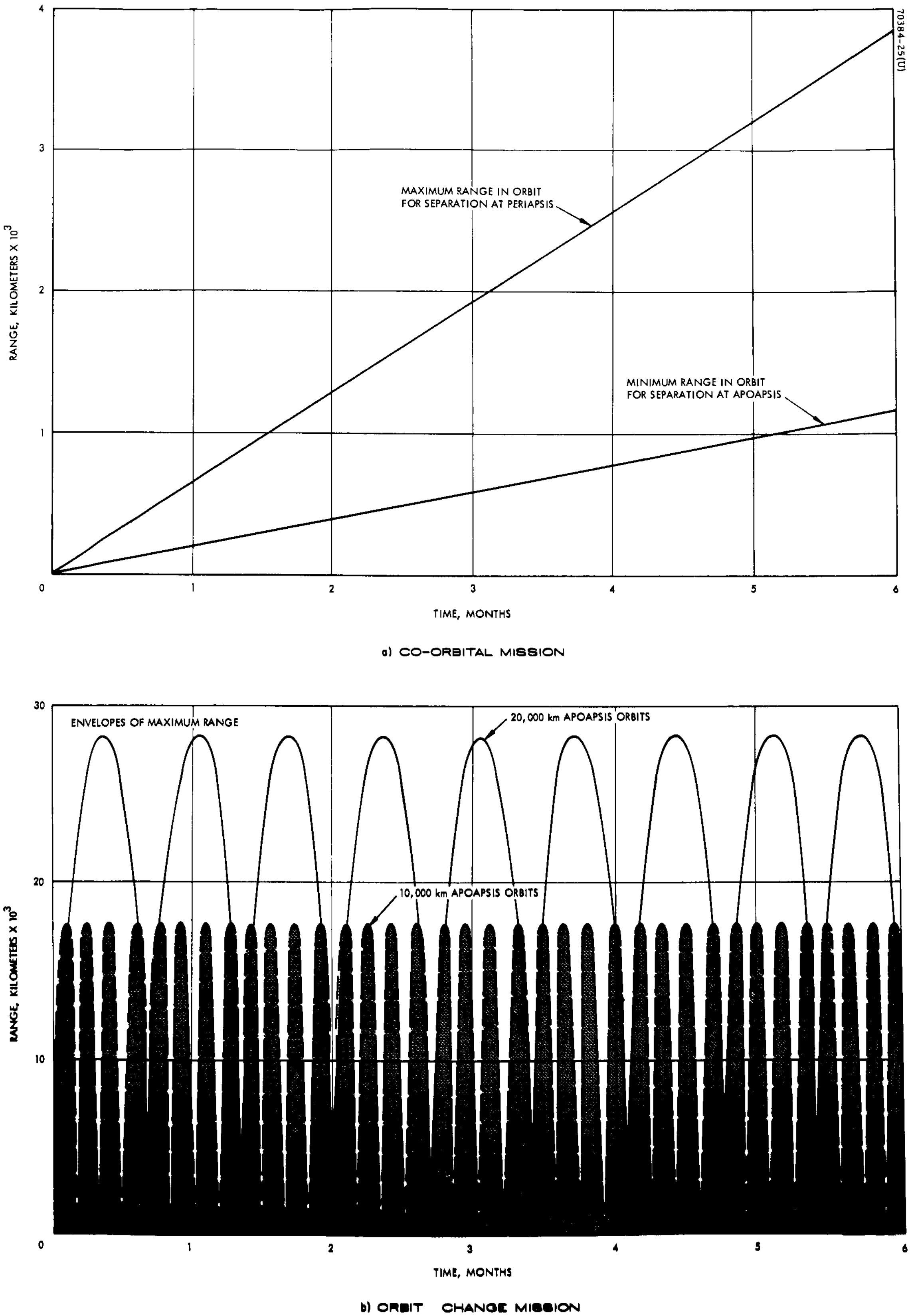


Figure 3-12. Voyager—OEC Range Histories

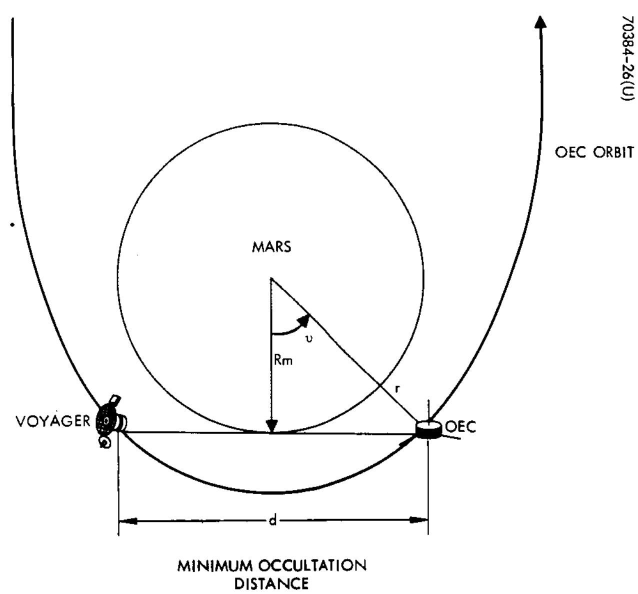


Figure 3-13. Occultation for Two Vehicles in Same Orbit

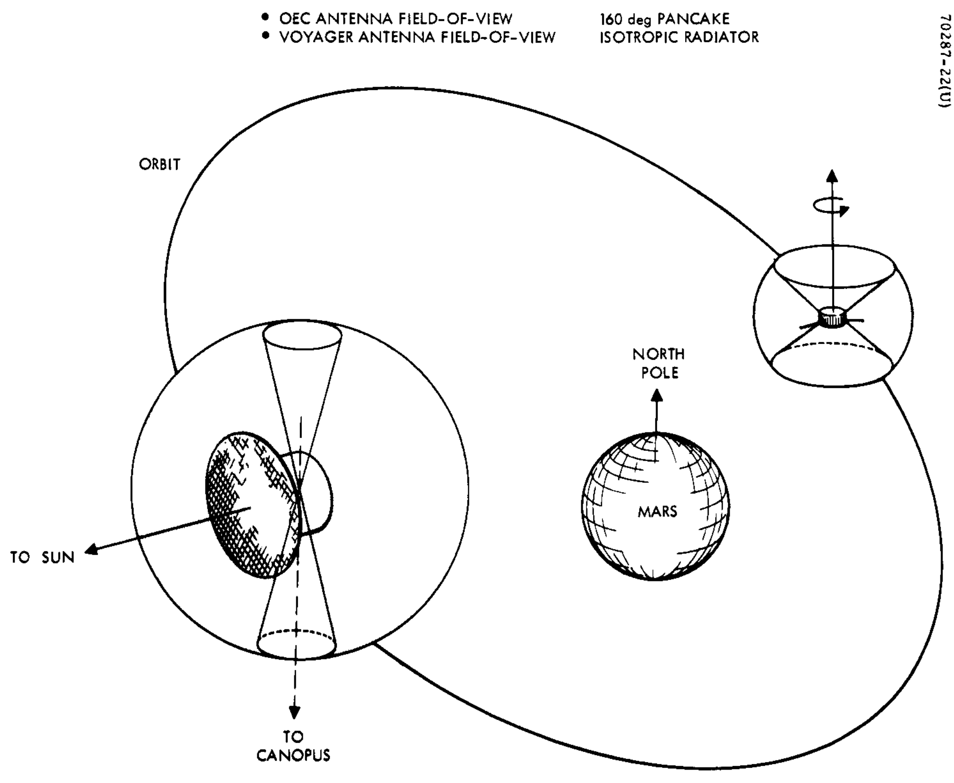


Figure 3-14. Voyager—OEC Antenna Coverage

Figure 3-13) 7000 km* or greater will cause occultation over some part of an orbit. The typical co-orbital case shown in Figure 3-12a stays much closer than this to Voyager and will not experience occultation during its mission lifetime.

One positive aspect of occultation is that an important experiment can be considered — that is, a Voyager—OEC occultation to obtain data on the Martian atmospheric and ionospheric properties. For the orbit change mission (Figure 3-12b), this is, of course, no problem — most orbits will experience occulting of the Voyager—OEC line of sight. For the co-orbital case, a period difference twice that shown in Figure 3-12a would give occultations toward the end of the 6 months — but at the sacrifice of short periods of other data.

3.1.6 Antenna Field of View Requirements

The complex nature of the orbit motion, as well as the generality of possible orbit characteristics, is reflected in the determination of the communication antenna coverage requirements. It is necessary to size the field of view for both the OEC and Voyager to ensure continuity of data transmission.

All missions discussed require similar antenna coverage patterns. The only difference is that continuous data is transmitted in the co-orbital mode whereas data is stored on tape and played back at preselected intervals for the orbit change mission.

The three factors that must be considered in designing the antenna coverage are:

1) Rotation of Voyager as it tracks the Sun and Canopus

2) Orientation of the OEC to Voyager in the nominal orbit

3) Effects of orbit precession

Figure 3-14 illustrates the expected coverage required.

The OEC coverage is necessarily symmetric about the spin axis and requires approximately 160 degrees field of view. The Voyager antenna pattern is more complex to develop. Based on results developed in Section 2.2.9 (Volume II), the Voyager requires an almost isotropic radiation pattern to be able to maintain the OEC in its field of view for any of the possible orbits Voyager might be in. Nulls very near the Canopus-seeking axis may be tolerable.

*The actual values are 7100 to 7500 km depending on which of the Voyager orbits is chosen.

3.2 OEC EJECTION

A key consideration during this study has been the feasibility of separating, or ejecting, the OEC from Voyager in a manner sufficiently well controlled to permit the use of a co-orbital operational mode over a full 6 months of operation. The co-orbital mode may be defined by "real time, continuous transmission of data and minimal, if any, attitude control required." In these terms, then, its achievability rests on whether OEC — Voyager ranges can be maintained sufficiently short to permit real time transmission for reasonable weight and power (region of interest is about 2000 to 4000 km) and whether it is possible at the same time to separate the OEC in or near its desired orientation (spin axis within 5 degrees of normal to the ecliptic plane). The problem is further complicated by the lack of definition of Voyager orbits, so that the question must be answered over the full range of possibilities.

3.2.1 Relative Motion and Separation Parameters

The OEC is injected into an orbit about Mars after the mission of the Voyager Lander is completed. The objective of the injection is to change the orbital period of the OEC orbit from that of Voyager. This is accomplished by making the orbit velocity of the capsule either greater or smaller than that of Voyager by a controlled amount.

The incremental difference in Voyager and OEC orbital periods is determined by the component of separation velocity increment along the orbit velocity vector. Control over this component is exercised by selection of the angle α measured between the separation velocity ΔV and the instantaneous orbit velocity of Voyager, as shown in Figure 3-15. The angle, α, in turn, is chosen by selection of the proper time in the orbit to command separation.

The parallel component of separation velocity is given by

$$\Delta V_p = |\Delta V| \cos \alpha$$

where

$|\Delta V|$ = magnitude of separation velocity increment

α = angle between separationvector and orbit velocity vector

Figure 3-16 shows the dependence of periapse separation distance on the two parameters $|\Delta V|$ and α. The values shown are for the periapse separation after 6 months and correspond to the 6 month point on the "maximum range envelope" like that of Figure 3-12a. The growth of OEC—Voyager range during the mission is linear as in Figure 3-13a. The data plotted in the figure are for injection of the OEC at a particular point in the orbit; the separation distance, for a given $|\Delta V|$ and α, is proportional to the orbit velocity at the point of injection, hence is least for injection nearest apoapsis.

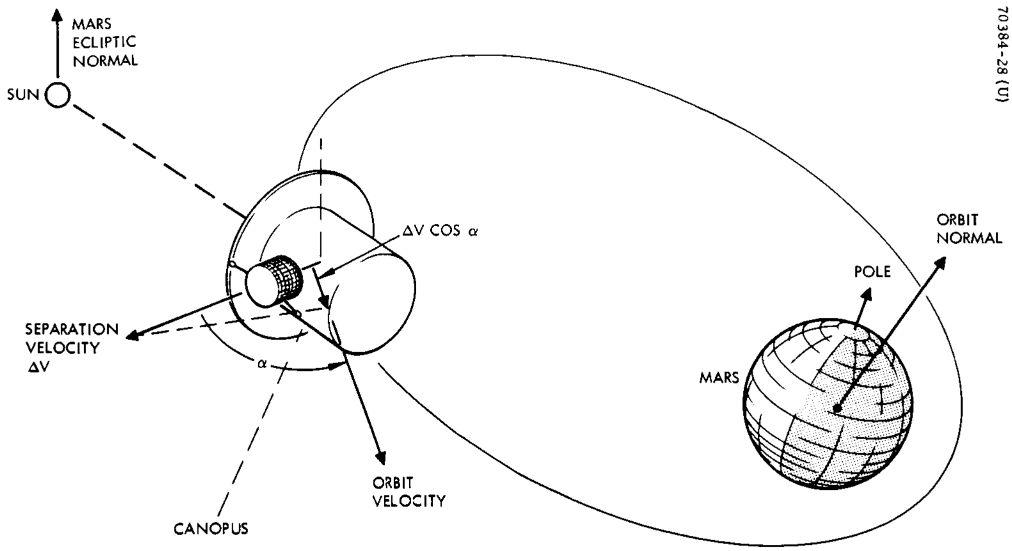


Figure 3-15. Relative Voyager—OEC Motion

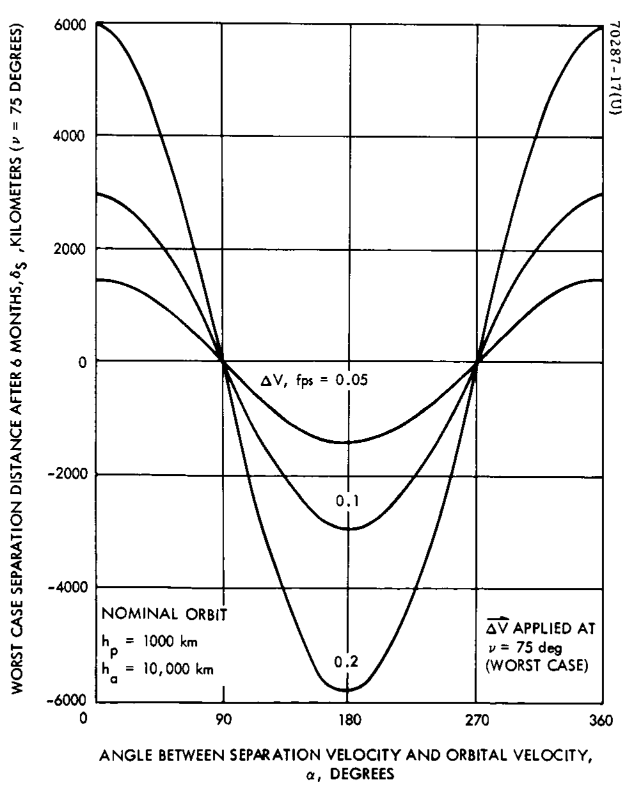


Figure 3-16. Separation Distance After 6 Months as Function of α

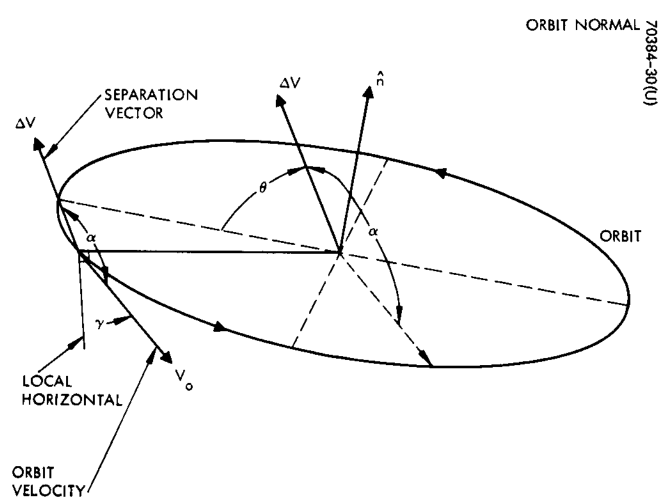


Figure 3-17. Relation of Separation Vector to Its Orbit Plane

3.2.2 Separation Window

The separation time for OEC is chosen on the basis of keeping OEC–Voyager separation over the mission a minimum consistent with assuring positive and permanent separation considering all the errors in imparting the separating impulse. Before discussing the errors in separation, the basic geometrical constraint on the choice of separation conditions will be briefly demonstrated. The detailed analyses of separation constraints and development of the separation windows are reported in Sections 2.3 and 2.4 of Volume II.

Figure 3-17 illustrates a typical orbit and defines the angle θ between the separation vector and the orbit plane. As can be seen, for a fixed inertial (i.e., on Voyager) attitude of the velocity vector ΔV, the angle α lies between

$$\theta \leq \alpha \leq 180 - \theta$$

Separation direction near 90 degrees must be avoided since they do not change the OEC orbit period from that of Voyager (see Figure 3-16); in this case the OEC could pass very close to Voyager during the first orbital revolution.. Since a θ of 90 degrees implies an α of 90 degrees, a limit on the value of θ in the region of 90 degrees must be provided. A detailed discussion of such considerations is presented in Section 2.4, Volume II.

Another important feature seen from the figure is that there are two points in each orbit at which separation with a particular value of (cos α) can be obtained. For a circular orbit, these positions would be 180 degrees out of phase, but for elliptic orbits the flight path angle must also be accounted for. Thus one position of injection is identified in the region of the periapsis and the other near apoapsis. The injections near apoapsis are more desirable since the orbital velocity is less than for periapsis so that the separation is more gradual (shorter ranges) and the period or window is of greater duration.

Ideally, the manner in which the OEC's orbit period is changed from that of Voyager is to add or subtract some small velocity. The injection angle α controls the component of separation velocity along the orbit velocity vector, while the separation mechanism itself controls the magnitude of the velocity impulse, ΔV. Errors in both ΔV and the angle α must be accounted for in providing for the desired injection parameters. Some important sources of errors are:

- Spring errors
- Spring mechanical alignments
- Orbit position uncertainties
- Voyager attitude uncertainties
- Command timing errors

Each of these sources in some way influences the value of the injection velocity component. The variation in the spring characteristics produces errors proportional to the magnitude of velocity increment. Any uncertainty of the

Voyager—OEC orbit creates timing errors since the velocity vector is continually changing direction as well as magnitude. Voyager limit cycle deadband also yields pointing errors. Since the attitude of Voyager is changing, so does the orientation of the OEC separation vector with respect to the orbit velocity vector.

Additional small velocity increments may be imparted to the OEC during its initial spinup, following the actual separation from Voyager. Such errors can be caused by mismatch of thrust between the (coupled) spinup jets or by misalignments of the jets giving forces other than the nominal pure torque couple.

These sources of error are tabulated in Table 3-2 with their rss values taken from the analysis of Section 2.4, Volume II. By far the largest error in angle is contributed by the 50 km assumed Voyager uncertainty — which is certainly a pessimistic value. Important errors in velocity are contributed by the spring tolerances and by the spinup jet mismatch. The latter effect would be reduced by lowering the jet thrust; the values used result from a tradeoff between tipoff errors resulting from slow spinup versus these velocity errors (Section 2.4, Volume II).

TABLE 3-2. SEPARATION AND SPINUP ERRORS

Source	1σ Value	Comment
Spring	10 to 20 percent of ΔV	Probably due to thermal variations.
Mechanical alignment	0.5 degree	Reasonable tolerance.
Orbit position	4.0 degrees	Based on Voyager uncertainty of 50 km. Clearly pessimistic.
Voyager attitude	0.7 degree	Maximum deadband amplitude, vector sum.
Timing errors	Negligible	Execution delays ~ 50 milliseconds.
Spinup jet mismatch	5 percent spinup thrust	
Spinup jet misalignment	<0.5 degree	This error can be reduced to < 0.1 degree with precision alignment techniques.

The variance of the velocity error parallel to the orbital velocity vector is given by

$$\delta V_p^2 = \cos^2 \alpha \left[\delta V^2 + \delta V_a^2\right] + \sin^2 \alpha \left[\Delta V^2 \, \delta\alpha^2 + \delta V_t^{\,2}\right]$$

where

α = angle between nominal ΔV and orbital velocity

ΔV = nominal separation velocity increment

δV = error in nominal ΔV

δV_a = error due to spin jet misalignment

δV_t = error due to spin jet mismatch

$\delta\alpha$ = error in launch angle

where all sources are assumed gaussian distributed independent random variables.

This expression is used to determine the equivalent error in the OEC—Voyager separation range. Algebraically adding the 3σ value of δV_p to the nominal V_p yields the maximum expected range in 6 months, whereas subtracting the two provides a lower bound on OEC—Voyager first orbit passage distance.

The per orbit separation range is given by

$$\delta S = \frac{3Pa\, V_o^2}{\mu} \Delta V_p \pm 3\, \delta V_p$$

where

P = orbit period

a = semi-major axis

V_o = orbital velocity

μ = gravitational constant of Mars

ΔV_p = nominal parallel velocity component

δV_p = error in nominal velocity component

Substitution of nominal characteristics for the 10,000 x 1,000 km orbit into the above equation yields the results shown in Figure 3-18. Both maximum and minimum values are shown. A constraint on the minimum distance of first orbit passage for OEC and Voyager to 1000 feet (0.33 km) was assumed. The maximum range determines the (co-orbital) communication range; in general, distances less than 4000 km are desirable. Cross-plotting these results as a function of the nominal injection angle and ΔV (Figure 3-19) illustrates the boundaries that determine the launch window. For discussion, an upper bound of 3500 km is assumed and yields a reasonable window. A decrease in the window is apparent if lower maximum ranges are taken. Figure 3-20 shows the launch window determination graphically for the 3500 km range limit. There is an optimum selection of separation velocity increment such that a maximum window can be achieved. This point varies somewhat with choice of the maximum range constraint. Assuming δS_{max}=3500 km, the choice of ΔV is 0.073 fps. The window for 0.073 fps is a total 33 degrees. There is a 5-degree decrease to 28 degrees for $\Delta V = 0.1$ fps. The geometric constraint on the injection angle α is $\theta \leq \alpha \leq 180 - \theta$.

For $\Delta V = 0.1$ fps, 40 degrees $\leq \alpha \leq$ 70 degrees for the cos α of Figure 3-20. Thus the minimum value of θ, the angle between the ΔV orientation and the Voyager orbit plane, is

$$\theta_{min} = 40 \text{ degrees}$$

and the range of possible α is 40 degrees $\leq \alpha \leq$ 140 degrees. Thus, there are two available windows:

$$40 \leq \alpha_1 \leq 70 \text{ degrees (increase OEC orbit velocity)}$$

$$110 \leq \alpha_2 \leq 140 \text{ degrees (decrease OEC orbit velocity)}$$

These results are illustrated in a simple example.

Assume that OEC is mounted on Voyager so that the direction of separation makes an angle of at least 40 degrees to the chosen orbit plane as shown in Figure 3-21. As Voyager orbits Mars, the injection angle α, taken equal to 40 degrees at periapsis, increases. The first possible launch window exists from periapsis where $\alpha = 40$ degrees to the point 2 in the orbit where $\alpha = 70$ degrees. During this time the OEC's orbit period can be made greater than Voyager's. Between 70 degrees and 110 degrees there is no window because of the collision constraint imposed. At 110 degrees the window reopens and the OEC orbit period can now be made smaller than Voyager's. This second window remains open up to the time when OEC reaches apoapsis, and $\alpha = 140$. After passing apoapsis the ability to adjust the period is repeated. However, this time the increase of OEC period occurs near apoapsis where 140 degrees $\leq \alpha \leq$ 110 degrees.

3.2.3 Location on Voyager

The results described above indicate the existence of launch windows of the order of 30 degrees in the angle α, centered around the region $\alpha \sim$ 50-60 degrees. As shown earlier, the geometric constraint on the values of α that will be attained during an orbit depends on the angle θ between ΔV and the orbit plane. Since Voyager axes are fixed to the Sun and Canopus, the angle θ will depend on the location of the Voyager orbit relative to the Sun and Canopus (i.e., to Voyager axes) and on the location of the OEC separation direction (ΔV) on the Voyager.

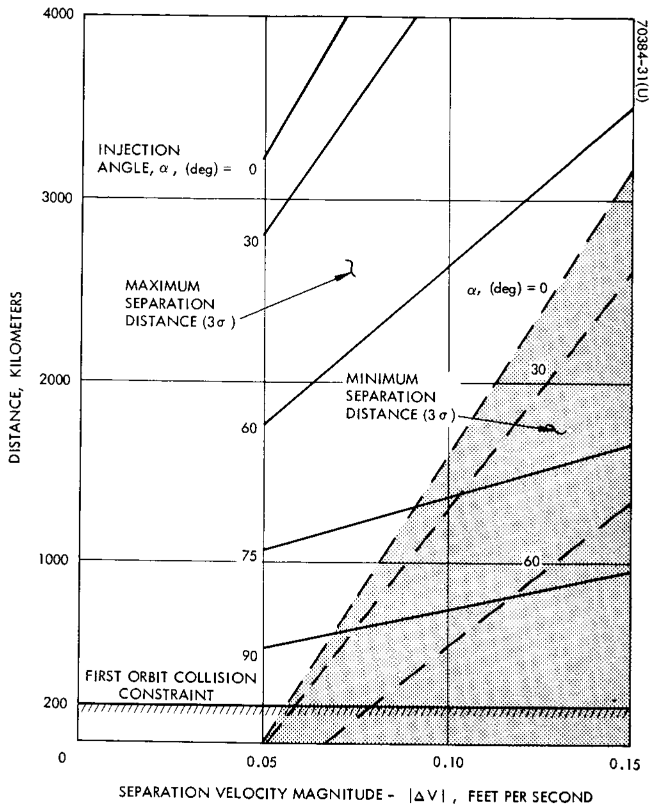


Figure 3-18. Separation Distance as Function of Injection Errors

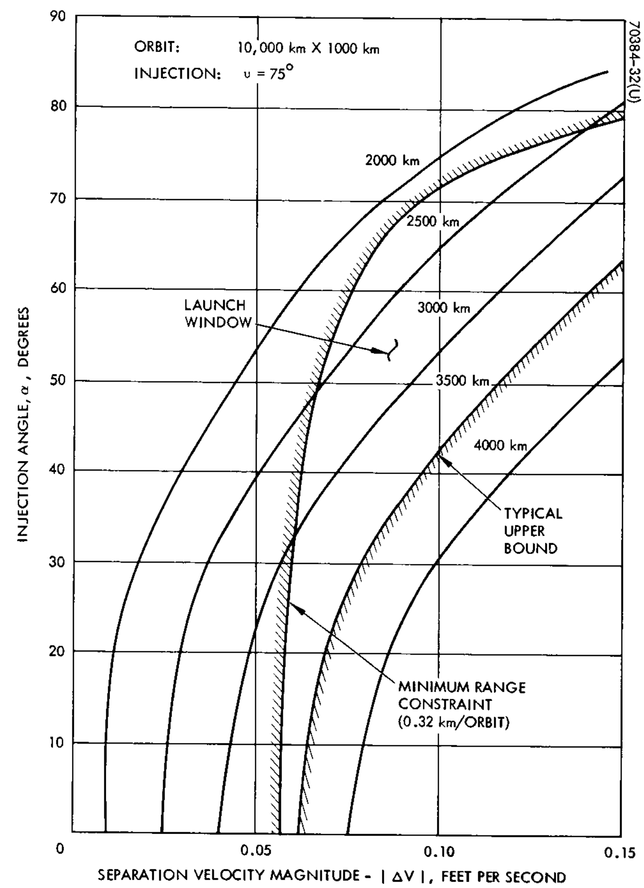


Figure 3-19. Worst Case OEC Separation Launch Windows

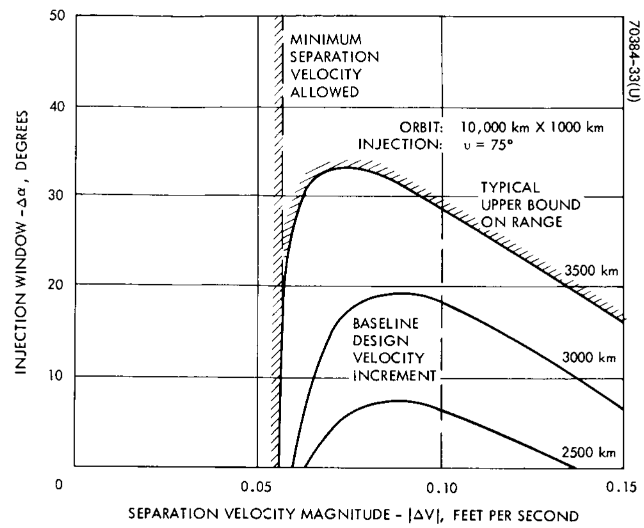


Figure 3-20. Typical OEC Injection Window

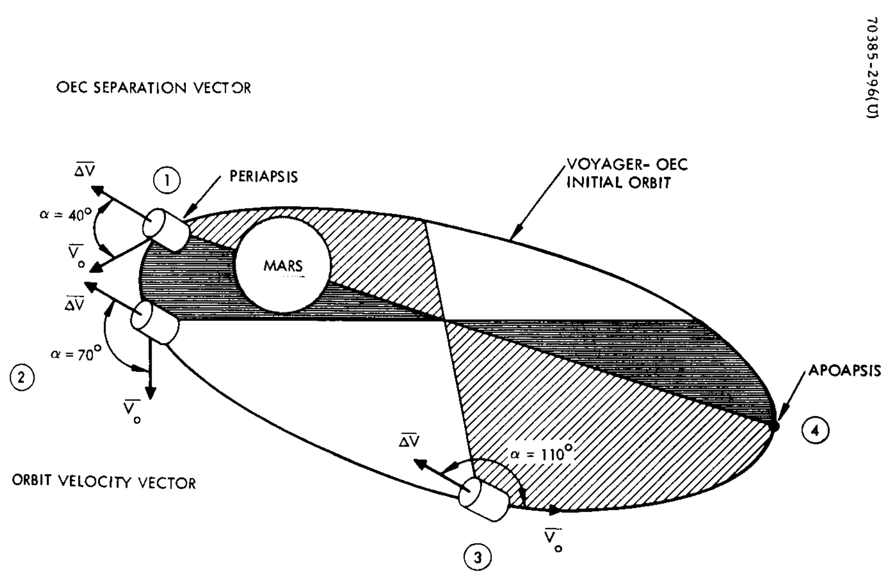


Figure 3-21. Typical OEC Launch Window Characteristic

To evaluate the flexibility with which mounting locations — actually separation corridors — can be chosen on the Voyager, calculations were made and are reported in Section 2.0, Volume II. The possible directions of separation radial to the Voyager (perpendicular to the Sun-pointing axis) are characterized by their angle Γ with Voyager X-axis as shown in Figure 3-22. For a given Γ, then, the orientation of the Voyager orbit in Voyager axes determines the angle between the separation direction and the orbit plane.

If the orbits are characterized in terms of their inclination to the ecliptic plane, $\hat{i}$, then for all possible orientations of the orbit (rotations about the ecliptic normal) the angle θ will lie between fixed limits dependent on Γ. A plot of these limits for $\hat{i} = 45$ degrees and a particular arrival date is shown in Figure 3-23. The shaded area indicates the possible values of θ (for some orbit at $\hat{i} = 45$ degrees) for a given separation direction. To illustrate: at $\Gamma = 0$ degree, the separation direction lies in the ecliptic plane (since X is in the ecliptic for this particular arrival date). In this case, the orbit, inclined 45 degrees to the ecliptic, may be such that it intersects the ecliptic plane along the X-axis (nodal line parallel to X) — in this case X, hence ΔV, lies in the orbit plane and θ is zero. If the orbit plane is rotated 90 degrees so that the nodal line is perpendicular to X, ΔV (along X) will make an angle of 45 degrees with the orbit plane; thus the shaded area in Figure 3-23 extends from 0 to 45 degrees at $\Gamma = 0$.

The crucial point in this analysis is the largest value of θ that might be present with no control over nodal position. For the case illustrated, separation directions from Voyager between -15 degrees to +15 degrees, 75 degrees to 105 degrees, 165 degrees to 195 degrees, and 255 degrees to 285 degrees will guarantee values of θ less than 60 degrees — and permit utilization of a substantial part of the launch window. In the worst case (orbit 45 degrees to ecliptic and arrival at worst time of Martian year) these preferred directions are more narrowly restricted about the four points, 0 degrees, 90 degrees, 180 degrees, and 270 degrees.

Placement of the OEC on the Voyager spacecraft requires a study of the proposed spacecraft configurations. Designs by GE, Boeing and TRW have been examined; they are shown in Figures 3-24, 3-25, and 3-26. The principal factor governing OEC azimuthal location (i.e., the angle Γ) is the location of the external systems. Since the vehicle XYZ coordinates are not necessarily the same as the lines of symmetry for each vehicle, the subsystems are located with reference to the X-axis shown in the figures.

The location of the subsystems, as presently defined in these designs, is tabulated for each vehicle in Table 3-3. For the GE vehicle, assuming the OEC to be separated radially outward (smallest exit corridor requirement), the OEC preferred position is at 0 degree, i.e., on the X-axis. For the Boeing vehicle, two positions are acceptable at 0 and 180 degrees; both of these positions are somewhat cramped by the stowed solar panels. For the TRW vehicle, both the 90 degree and 270 degree positions are available, with the 270 degree location preferred to avoid possible interference with the view of the Canopus sensor.

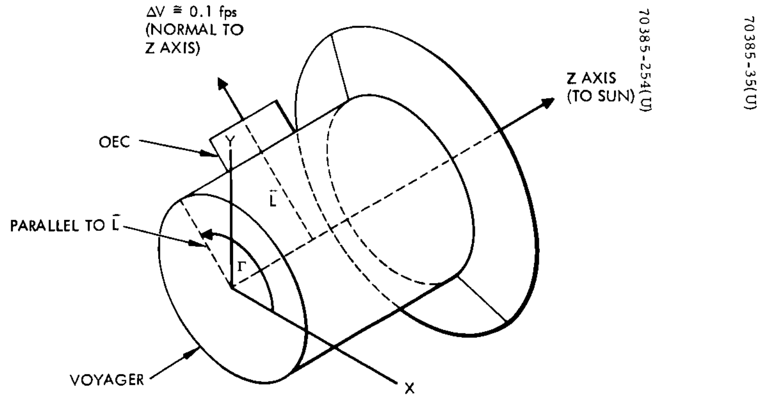


Figure 3-22. OEC Separation Relative to Voyager

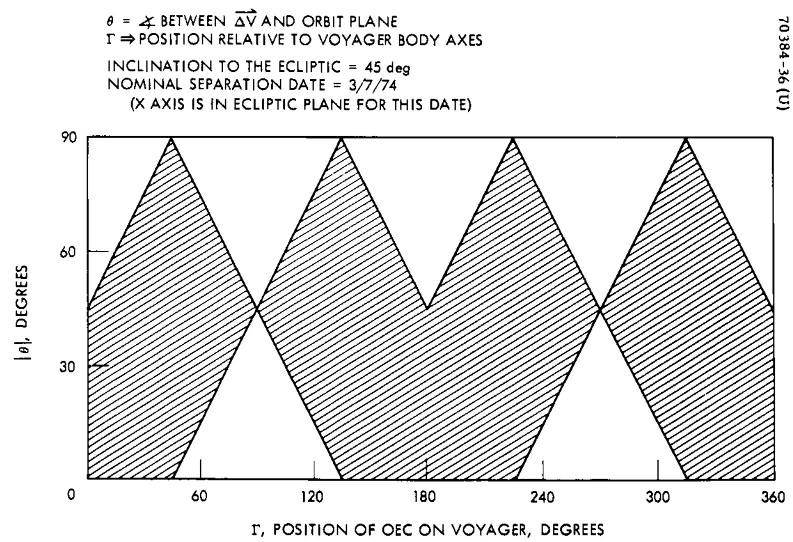


Figure 3-23. Relationship Between OEC Location on Voyager and Orbit Plane

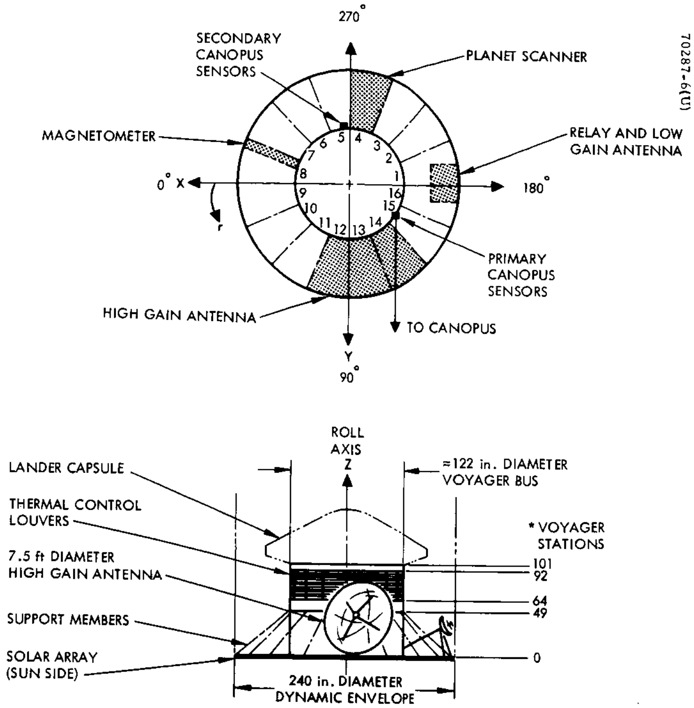


Figure 3-24. General Electric Voyager Spacecraft

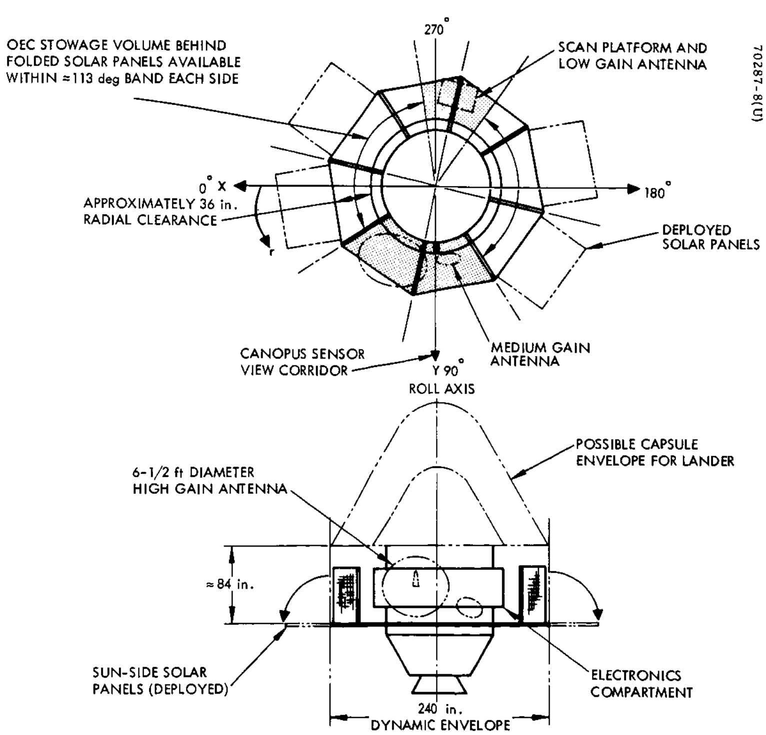


Figure 3-25. Boeing Voyager Spacecraft

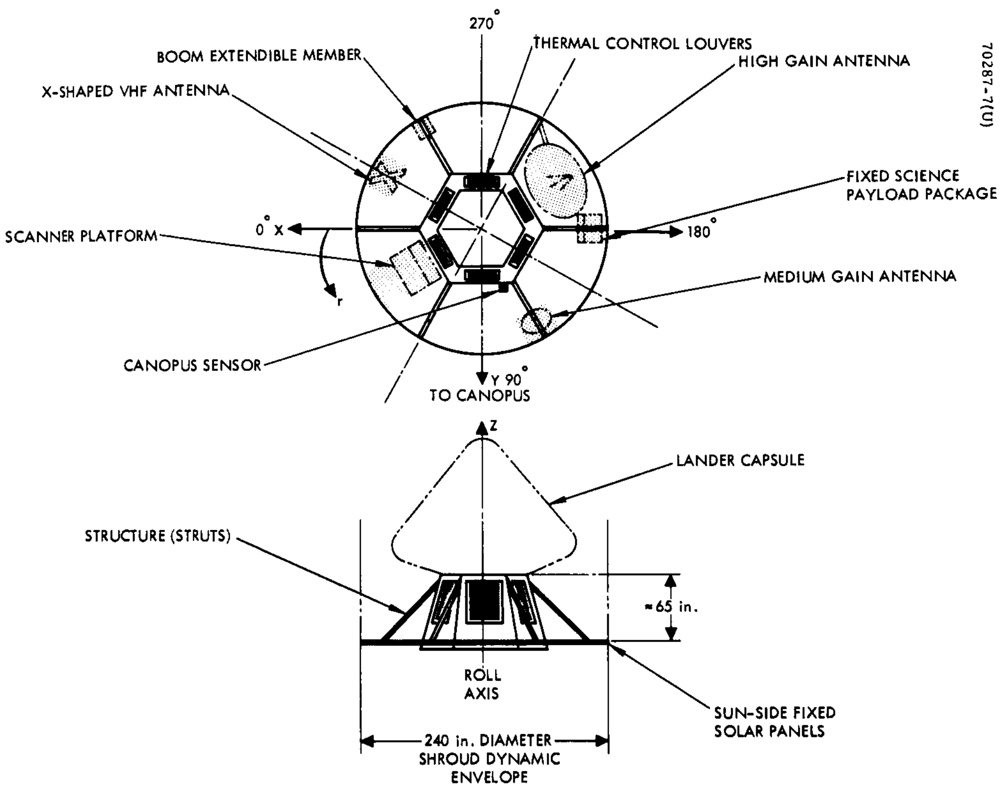


Figure 3-26. TRW Systems Voyager Spacecraft

TABLE 3-3. VOYAGER SPACECRAFT CONFIGURATION SUBSYSTEM LOCATIONS

Configuration	Subsystem	Angle Constraint, degrees
GE	High gain antenna, Canopus sensor	68 to 135
	Relay and low gain antenna	170 to 190
	Planet scanner	247 to 290
	Magnetometer	337 to 342
Boeing	High gain antenna, medium gain antenna, and Canopus sensor	31 to 121
	Scan platform and low gain antenna	234 to 278
TRW	Scan platform	0 to 63
	Medium gain antenna and Canopus sensor	109 to 132
	High gain antenna and science payload	172 to 244
	Boom extendable antenna and X-shaped VHF antenna	

3.2.4 OEC Ejected Attitude and Separation Tipoff

In the simplest mission considered, the OEC would have no attitude control capability at all; its spin attitude through the mission would be that obtained at completion of spinup, changing very slowly due to the (small) disturbances of the solar pressure and the gravitational torques. The Voyager Y-axis (Figure 3-22) always lies in the Sun-Canopus plane; through the Martian year this axis can be as much as 15 degrees away from the ecliptic normal. If, therefore, the OEC is to be oriented normal to the ecliptic after ejection, the separation system must provide flexibility for tilting of the OEC spin axis to account for the location of the ecliptic normal to Voyager's Sun-Canopus coordinates at the expected date of arrival.

Possible modes of separation are illustrated in Figure 3-27. Requirements are generally less stringent for configurations with axially located booms as shown. The detailed configuration interface with separation is treated in Section 6.1 of this volume and Section 4.5 of Volume II.

The discussion in the preceding subsection leading to preferred directions for separation on each of the three Voyager spacecraft designs dictates preferences among the alternates shown in the figure. With both the Boeing and GE spacecraft, separation should be (roughly) in the plane of the ecliptic (along X); hence only the separations "out of the plane of the paper" in the figure will orient the OEC spin axis along the ecliptic normal. With the TRW spacecraft, the separation is at 90 degrees, leading to a choice of separation directions in the plane of the paper.

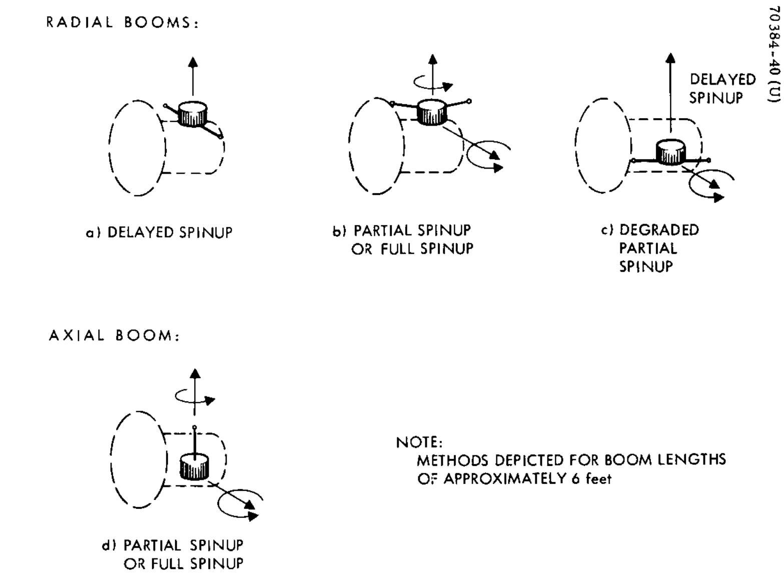


Figure 3-27. Candidate Separation Modes for Fixed Length Radial and Axial Boom Configurations

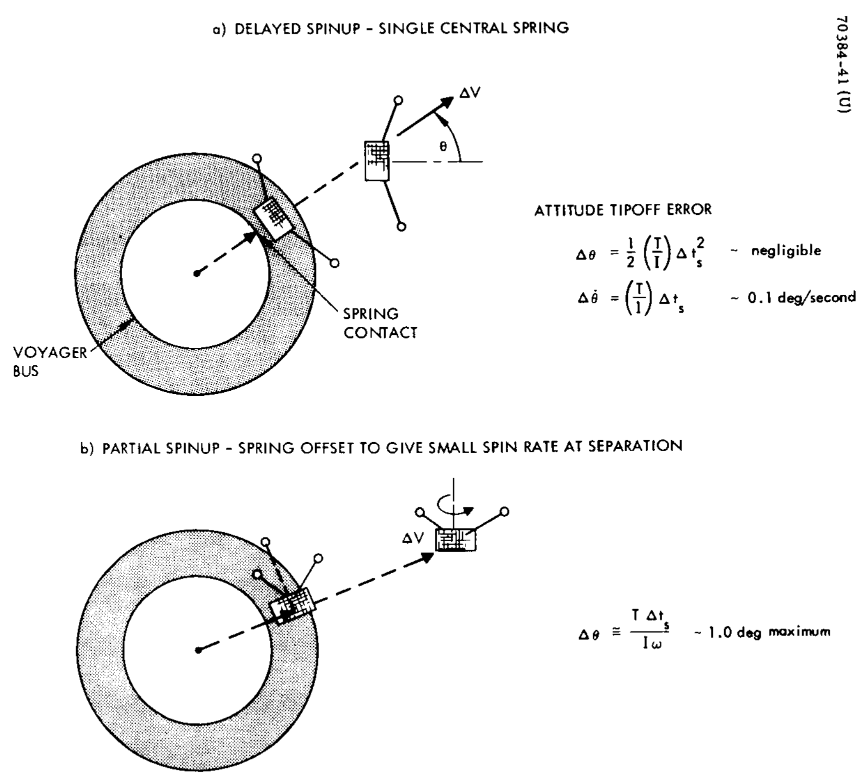


Figure 3-28. Separation Techniques

The simplest separation is a straightforward spring ejection with OEC spinup after achieving adequate clearance from Voyager, as in Figure 3-28a. Nominally, the spring force is through the capsule center of mass; however, due to various misalignments, this force is offset, generating torques about an axis (or axes) transverse to the spin axis. Typical (3σ) accuracies of transverse and angular misalignments of 1/16 inch and 1.0 degree are assumed.

For separation velocities of approximately 0.1 fps, a typical set of spring characteristics would be a 2.1 pound spring force exerted for 0.22 second. With the quoted (3σ) misalignments, this impulse would create a tipoff error of approximately 0.1 deg/sec. By waiting 5 to 6 feet (50 to 60 seconds) for the booms to clear the spacecraft before initiating spinup, a spin axis attitude error of approximately 5 degrees could build up.

Figure 3-28b illustrates another type of separation technique which can prevent these large initial attitude errors. In this alternative scheme, the prime separation force is normal to the spin axis and offset in the plane containing the center of mass. The objective of this form of separation is to impart not only a translational impulse to yield the desired ΔV, but in addition to give a torque impulse providing gyroscopic stability for the 5 or 6 foot coast period. The angular error with this technique results from the transverse torques due to misalignments causing the net OEC angular momentum to deviate from the nominal spin axis. For the same 1/16 inch and 1 degree (3σ) tolerances, a 1 degree attitude error would result with a spin rate of 1 rpm. This spin rate would require a spring offset from the center of mass by about 18 inches.

3.2.5 Voyager—OEC Perturbations

Voyager Perturbations on OEC. The Voyager attitude control concept proposed by each of the three present spacecraft contractors provides operation in a limit cycle mode during the Mars orbital phases. Two operational phases of the Voyager spacecraft mission are of interest for the OEC mission. The predominant factor which establishes the separation initial conditions is the steady-state limit cycle. The inertial mode, which provides Voyager the capability to reorient to any attitude, is of importance because it can be used to orient the OEC in a preferred direction at separation, but is not otherwise treated here.

Because of the near absence of disturbances in space, the optimum method by which active three-axis controlled spacecraft maintain a desired orientation is by the application of the on-off reaction control system. The efficiency of the characteristic limit cycle of this subsystem determines to a large extent the fuel required for the attitude control of the spacecraft. Typically, the attitude control limit cycle is designed to minimize the fuel expended.

For illustration, the parameters associated with the GE Voyager spacecraft will be used in the ensuing discussion. Results obtained are typical of the other designs as well.

The limit cycle is characterized by the attitude deadband and the rate increment. Table 3-4 presents some of the pertinent characteristics of the GE system.

TABLE 3-4. TYPICAL VOYAGER LIMIT CYCLE PARAMETERS

Angular deadband	±8.0 milliradians
Rate increment	$\pm 3.4 \times 10^{-6}$ rad/sec
Control acceleration	± 0.225 mr/sec^2
Valve minimum on-time	30 milliseconds

Assuming a 1 meter moment arm from the center of mass of Voyager to the location of the OEC, the velocity is $\Delta V = 3.4 \times 10^{-6}$ m/sec. This velocity is about four orders of magnitude less than the expected separation velocity of approximately 0.03 m/sec (0.1 fps), so is of negligible importance. Other potential perturbing effects on the OEC are similarly of negligible magnitude.

OEC Perturbations on Voyager. The Voyager limit cycle period is approximately 1.3 hours. It is assumed that this period exists about each of the axes and that a control pulse occurs whenever the deadband is reached. As mentioned above, the limit cycle characteristics are designed to minimize fuel expenditures and still provide necessary attitude accuracies by controlling the spacecraft to point to the Sun and Canopus.

Separation of the OEC from Voyager will not cause a break of lock with the celestial references but will perturb the Voyager from its nominal attitude. The sensor fields of view are sufficiently large to provide continuous signals to the control system. Table 3-5 gives the typical sensor characteristics for the GE configuration, and is expected to be very similar to the Boeing or TRW sensors.

The torque impulse applied to the spacecraft induces an instantaneous rate error. This rate error will be measured by Voyager and a closed loop control signal initiated to pulse the proper jet. The error will be removed by the control torque in about 2 seconds and the Voyager will resume its normal steady state operation.

TABLE 3-5. SENSOR FIELDS OF VIEW AND ACCURACIES

	Field of View, degrees	Linear Range, degrees
Sun sensor	±15	1
Canopus sensor		
Instantaneous	± 4	
Slewing		
Roll	± 2	
Pitch	±15	

There are two additional perturbing effects on Voyager that deserve consideration. First is the translational acceleration added at separation, and second is the decrease in Voyager moments of inertia at separation. Reference 2 establishes the constraints on the allowable perturbations induced on the Voyager spacecraft. In particular, the (unpredictable) translational accelerations must not exceed an average value of 0.6×10^{-7} cm/sec^2 over the mission. In terms of the distance traveled over a 6-month period, this corresponds to an average velocity of 0.95 cm/sec. Assuming a separation velocity of 3.1 cm/sec imparted to the OEC, the acceleration imparted to translate the Voyager is equivalent to a velocity of about 0.13 cm/sec. The Voyager bus is assumed to weigh 1365 kg.

Separation of the OEC from Voyager changes the inertia properties of the spacecraft, influencing the efficiency of the control system operation. Results from GE Voyager design indicate that the decrease in principal moments of inertia due to the capsule-Lander separation will have negligible effect on limit cycle operation. Hence, it is assumed that OEC, which weighs less than 5 percent of the capsule-Lander, will not degrade Voyager steady-state operation.

3.2.6 Summary of Conclusions

The foregoing results permit certain conclusions of key importance to the feasibility of the OEC missions considered here.

Ejection velocity — An ejection velocity of about 0.05 to 0.1 fps appears to adequately satisfy all constraints and provides a separation window of more than adequate duration.

Location of OEC — A good location for OEC on each of the three Voyager spacecraft considered has been identified which permits near-optimum separation.

Communication range — Maximum communication ranges over 6 months can be guaranteed at under 3500 km, meeting all of the ejection and location constraints and including errors in the ejection. This is, of course, premised on the Voyager Orbiter conducting no orbit changes of its own.

Attitude — OEC spin axis attitude can be placed normal to the ecliptic within a few degrees (1 to 3 degrees) if:

1) OEC is tilted on the Voyager so as to be aligned with the ecliptic normal at arrival, and
2) OEC is separated from Voyager in a "partial spin" mode — i.e., with a spin rate of the order of 1 rpm imparted by the separation.

Neither of these techniques has been recommended for the baseline OEC concept described in Sections 2.0 and 5.0 of this volume, since it has attitude control capability. Even without these techniques, and without activation of the attitude control system, the baseline system will be oriented with spin axis within 5 to 20 degrees of the ecliptic normal depending on time of year of arrival; the spin axis will be within 5 degrees (3σ) of normal to the Sun line initially.

Voyager–OEC perturbations — No significant dynamic perturbations of Voyager or OEC on each other should be expected due to or during ejection.

3.3 ATTITUDE AND ORBIT DETERMINATION

The OEC is designed to provide an accurate map of the near environment of Mars. In order to achieve this result, the data received from the scientific instruments must have a reference basis. This reference system must yield an indication of the position to which the experimental measurements refer as OEC orbits the planet, as well as the inertial orientation of the scientific instruments.

The techniques by which this reference is established are known as orbit and attitude determination. The orbit determination process provides a measure of the OEC orbital position as a function of time establishing an inertial (or other) reference for the experimental data.

Orientation of the experiments on the orbit is related to the attitude of the OEC. OEC attitude is established by location of the spin axis of the capsule in inertial space and by the azimuthal position as it spins about this axis. Azimuth is known accurately (±0.25 degree or better) by reference to the once per spin sun pulses.

3.3.1 Requirements for Attitude and Orbit Determination

The OEC accuracy requirements are generated from the expected characteristics of the Mars magnetosphere and the accuracies of the experiments. These requirements have been interpreted in terms of specifications for position determination and vehicle attitude determination. Table 3-6 summarizes the requirements for the OEC.

TABLE 3-6. OEC ATTITUDE AND ORBIT DETERMINATION REQUIREMENTS

Knowledge of spin axis attitude	$< \pm 1.0$ degree (3σ)
Knowledge of attitude to Sun	$< \pm 0.25$ degree (3σ)
Knowledge of orbit position	$< \pm 100$ km (3σ) apoapsis $< \pm 20$ km (3σ) periapsis

In the event that the occultation experiment is included as part of the OEC science, a considerable improvement in position accuracy (to tens of meters) could be required. This possibility should be borne in mind in the following discussion.

3.3.2 Attitude Determination

Attitude determination can be accomplished by measuring the spin axis orientation relative to any of a number of celestial objects. A minimum of two objects is necessary to completely specify the (spin axis) attitude of a satellite.

The available celestial objects for attitude measurement include:

- Sun
- Planet (Mars)
- Stars

Both the Sun and the stars essentially can be treated as point sources, whereas Mars angular subtense varies as a function of closeness to the planet.

The Sun is a natural choice as one of the two references, both because the requirements specifically include Sun—spin axis attitude and because of the simplicity and availability of satellite Sun sensors.

Specific "baseline" sensor designs have been developed for a Sun sensor, Mars sensor, and star sensor for this application; details of these sensor design studies are reported in Section 2.5 of Volume II. The single measurement accuracies calculated for these baseline sensors are listed in Table 3-7.

TABLE 3-7. ATTITUDE SENSOR ACCURACIES

Source	Reference	Angular Accuracy to Reference (3σ), degrees
Sun	Mars-sunline	± 0.5
Star	Starline	$< \pm 0.1$
Mars	Local vertical	$< \pm 1.5$

Since the OEC spin attitude changes only due to disturbance torques at a maximum rate of 0.01 degree per orbit (Section 2.7, Volume II), data from any of these attitude sensors can be smoothed over 10 to 20 orbits to yield accurate OEC attitude. Smoothing of the data over such a long interval will result in accuracies limited only by the static errors in the sensors and in their mechanical and optical alignments. Some quasi-static errors also exist in the Mars sensor operation due to unequal radiation from the edges of the Mars disc; this type of error, too, will be partially smoothed due to the different aspects from which the OEC views Mars as it orbits.

The smoothed accuracies of attitude determination which can be achieved with the two sensor combinations considered are shown in Table 3-8, based on the error budgets of Section 2.5, Volume II. The Sun-Mars sensor combination requires knowledge of orbit position to translate the Mars sensor data (local vertical) to an inertial coordinate system; the accuracy quoted presumes an independent source of orbit determination accurate to the order of 100 km.

TABLE 3-8. ACCURACY OF ATTITUDE DETERMINATION

Sensor	Spin Axis to Sun Angle (3σ), degrees	Sensor Spin Position about Spin Axis (3σ), degrees	Spin Axis Inertial Orientation about Sunline (3σ), degrees
Sun sensor/Mars sensor	±0.2	±0.2	<±1.0
Sun sensor/star sensor	±0.2	±0.2	±0.2

*Presumes orbit known independently to accuracy of about 100 km.

It is seen that either sensor combination meets the mission requirements cited in Table 3-6. The choice between them, then, rests on other factors. All of the sensors considered are relatively simple with no moving parts and solid-state detector elements. An applicable Sun sensor presently exists as space proven hardware from the Applications Technology Satellite program. The Mars sensor considered would be a slight modification of Earth horizon sensors which are similarly presently existing from the TIROS and Hughes HS-308 communication satellite programs. The star sensor design for this study is a new device but is simple and straightforward and would be similar in many ways to the Sun sensor.

An additional factor is the possibility of utilizing the Sun-Mars sensor combination for (degraded) orbit determination in the event of failure in the primary orbit determination equipment. This possibility is discussed briefly later in this section and in some detail in Section 2.5, Volume II.

The various factors are summarized in the tradeoff matrix of Table 3-9. For the baseline (recommended) OEC, the Sun-Mars sensor combination was chosen, primarily because of its orbit determination backup feature.

TABLE 3-9. ATTITUDE DETERMINATION TRADEOFF

Sensor	Accuracy	Weight/Power	Development Required	Comments
Sun sensor/ Mars sensor	Adequate (0.2° to sun 1.0° inertial)	< 2.3 lb 1 watt	Sun sensor existing. Optics modification to TIROS sensor.	Provides backup orbit determination.
Sun sensor/ star sensor	Excellent (0.2° net)	< 4 lb < 1 watt	Sun sensor existing. Star sensor new item, but simple device.	

3.3.3 Orbit Determination

Orbit determination can be effected by several means. Doppler measurements of OEC range and range rate via the Deep Space Net from Earth can provide an orbit determination similar to the accuracies in determining the Voyager orbit.

The steady-state accuracy with which the Deep Space Net can determine OEC orbit position has been quoted conservatively at 1 to 10 km*. Some sources claim accuracies of tens of meters. Whichever estimates are believed, it is clear that DSN tracking can meet OEC requirements. To implement such a scheme, an S-band transponder must be included in the OEC equipment complement; total weight increment for equipment to perform this function on OEC would be about 11 pounds (see Section 5.0 of this volume and Section 2.5 of Volume II).

OEC position can also be determined by measurement of range and range rate from the Voyager bus. The orbit would then be determined with reference to that of Voyager with somewhat less accuracy than that available from direct DSN tracking of the OEC. Brief studies have indicated the feasibility of obtaining Voyager—OEC range data with accuracies of better than 10 km and range rate to an accuracy of 0.01 m/sec or better. According to present Voyager plans, however, there is no provision for Voyager transmission capabilities at the frequencies of interest (136 or 400 MHz) so that this technique would require the addition of a transmitter to Voyager as well as the transpond mode (no significant equipment) to OEC.

It is also possible to determine the position of the OEC autonomously. A planet sensor on the satellite can measure the altitude of OEC as well as its attitude to the vertical; this is described in Section 2.5, Volume II, and in Section 5.0 of this volume.

*Verbal communication from Jet Propulsion Laboratory.

The altitude data, together with the angles measured from Sun and Mars, can yield OEC position in a Mars-Sun system of reference. In such a reference system, there is no way of instantaneously determining azimuthal position about the Mars-sunline — that is, the position so determined is ambiguous in the sense that one orbit cannot be distinguished from a second orbit which is, mathematically, the first orbit rotated about the Mars-sunline. As Mars rotates about the Sun, and the orbit rotates about Mars, this ambiguity can be resolved. These rotations are sufficiently slow that relatively large errors in orbital position in the direction perpendicular to the Mars-Sun-OEC plane are to be expected. The errors are the order of 100 km (3σ) at periapse and 250 km (3σ) or so at apoapse. Positional errors in the Mars-Sun-OEC plane with this technique lie between 10 and 20 km (3σ), meeting the requirements.

Whatever sensor data is used for orbit determination, the raw data must be processed using a "position estimation" computer program. This program uses a model of the OEC orbit with a number of unknown orbit parameters and determines values for the orbit parameters which "best fit" (in a probabilistic sense) the actual data. Thus data gathered over many orbits are used to achieve ultimate accuracies far greater than could be attained from instantaneous measurements. Estimation programs that could be used for OEC are discussed in some detail in Section 2.5, Volume II. One relatively simple program has been simulated on a digital computer and used to assess the orbit determination accuracy of the "baseline sensors". Figure 3-29 shows typical computer results for one case; a large initial position error was assumed and only Mars and Sun sensor data used to determine position in the Mars-Sun-OEC plane. The data shows a position error after eight orbits of about 10 km (3σ). The uncertainty normal to the plane must be separately assessed and depends on whether other sensors are used to remove the ambiguity or not.

3.3.4 Orbit/Attitude Determination Tradeoff

The orbit and attitude determination accuracies achievable with the several sensor combinations considered for OEC are summarized in Table 3-10. Ranging from Voyager is not included in this table; its accuracy is somewhat less than that quoted for the S-band to DSN options and it would require additional equipment on Voyager, so has been rejected for this function.

The Sun and Mars sensors only (option A) do not meet the mission requirement. They do provide a degree of orbit determination which should be potentially useful as a failure mode. Note that, without an independent method of orbit determination, the attitude accuracy is degraded from that cited in Table 3-8.

Adding the S-band communication capability, as in option B, provides not only more than adequate accuracies of both orbit and attitude determination but also yields a backup (degraded) data transmission mode in the event of failure of the link through the Voyager. This is discussed in Section 5.0 of this volume and in Volume II. This option is the recommended combination for the baseline OEC.

Option C provides an autonomous OEC determination system, but without the S-band features. This is the lightest weight, lowest power system of the three options (B, C, D) which meet the mission requirement.

Option D offers the greatest accuracy, but was considered less desirable than B because of the failure mode capability of the Mars sensor for orbit determination.

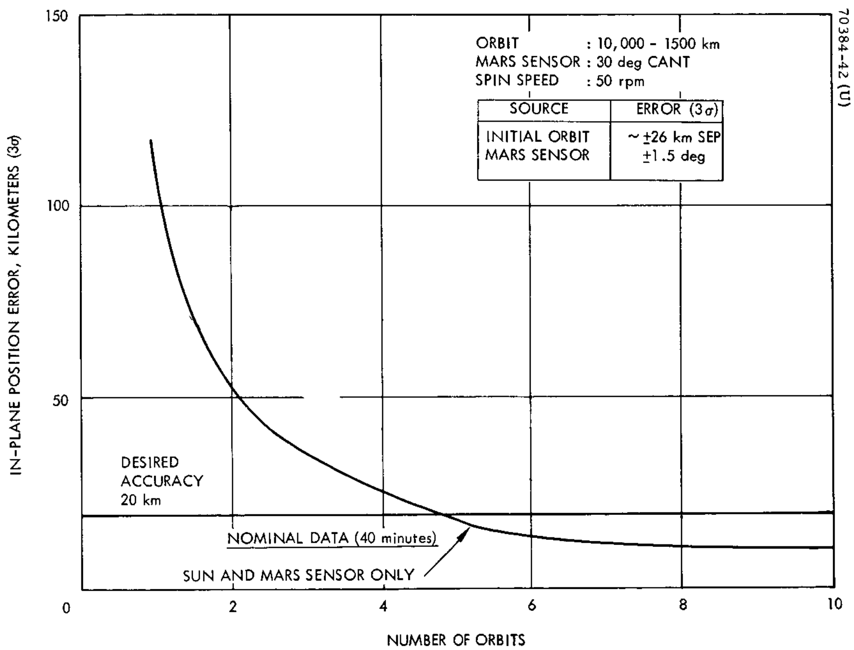


Figure 3-29. Position Accuracy Improvement

TABLE 3-10. ORBIT AND ATTITUDE DETERMINATION ACCURACIES

Sensor	Achievable (3σ) Accuracy				Comment
	Attitude, degrees		Orbit, km		
	Sun	Spin Axis	Periapse	Apoapse	
A ● Sun ● Mars	±0.2	±3	<110 (Mars-Sun-OEC plane only)	<260	Orbit accuracy requirement cannot be met because of positional ambiguity (see Section 2.5, Volume II).
B ● Sun o Mars ● S-band to DSN	±0.2	±1.0	<10	<50	This combination of sensors with S-band available in several OEC concepts outlined in Table 2-1. Occasional fix (once per day) to Earth (or Voyager) required via S-band link to resolve ambiguity in OEC position.
C o Sun ● Mars o Star	±0.2	<±0.2	<10	<50	Star sensor provides additional attitude accuracy and also serves to remove positional ambiguity by precisely locating spin axis location.
D o Sun o Star ● S-band to DSN	±0.2	<±0.2	<10	<50	Star sensor used only to provide increased knowledge of spin axis location. S-band must be used with DSN as prime orbit determination method.

3.4 REQUIREMENTS FOR ATTITUDE STABILIZATION, CONTROL, AND ORBITAL MANEUVERS

A design requirement of the OEC mission specifies that the stabilization system operate over the 6-month lifetime. This is true for the co-orbital as well as orbit change systems. To assess the capsule attitude stability, a study of the external perturbations was conducted. This is necessary to size the attitude control system selected for several of the alternative missions.

Having established the basic mission operation, the total impulse requirements for attitude corrections and attitude changes preliminary to an orbit maneuver must be determined. These requirements along with the velocity increment requirements for changing the orbit of the OEC are established in this section.

3.4.1 Effects of External Disturbances on Attitude

A detailed digital simulation of the three major attitude perturbative effects of solar pressure, gravity torques, and aerodynamic torques are presented in Section 2.7, Volume II. The results are summarized below.

A nominal 10,000 km apoapsis altitude was assumed since the effects at higher altitudes are smaller. Disturbances on OEC at periapsis altitudes of 1000 km, 500 km, and 300 km are evaluated and the results summarized in Table 3-11.

Notice that in general the attitude error per orbit is on the same order of magnitude for each of the sources and amounts to approximately 0.01 deg/orbit when they are root sum squared. At 300 km the errors are slightly larger due to the added influence of the aerodynamic torques. Recall that in Section 3.1 the minimum desirable operating altitude to meet the 50-year lifetime constraint is 275 km (~250 for 10 years). For this reason the effects of attitude disturbances at the lower and more densely packed atmosphere are of no concern.

The capability to remove these errors will be discussed.

3.4.2 Attitude Corrections

Correction of the OEC attitude is necessary for several phases of operation. This system is basic in the orbit change mode and can also be included on the simpler coorbital OEC. The reason for doing so is to ease the initial separation requirements so that the ±5 degree (3σ) attitude alignment to the ecliptic normal can be easily achieved.

In addition, this correction system is used to rotate the capsule into a preferred orientation to measure all components of the Martian magnetic field. This adjustment in attitude can be performed during the initial trimming of attitude following separation or at a later time.

TABLE 3-11. DISTURBANCE TORQUE SUMMARY (DEGREES PER ORBIT)

Source	Attitude Error per Orbit for Periapsis Altitude of: 1000 km	500 km	300 km
Solar pressure	≤0.0048	≤0.0046	≤0.0045
Gravity gradient	≤0.0069	≤0.0081	≤0.0087
Aerodynamic	Negligible	Negligible	≤0.005

- Apoapsis altitude = 10,000 km
- 2 inch CP-cg offset

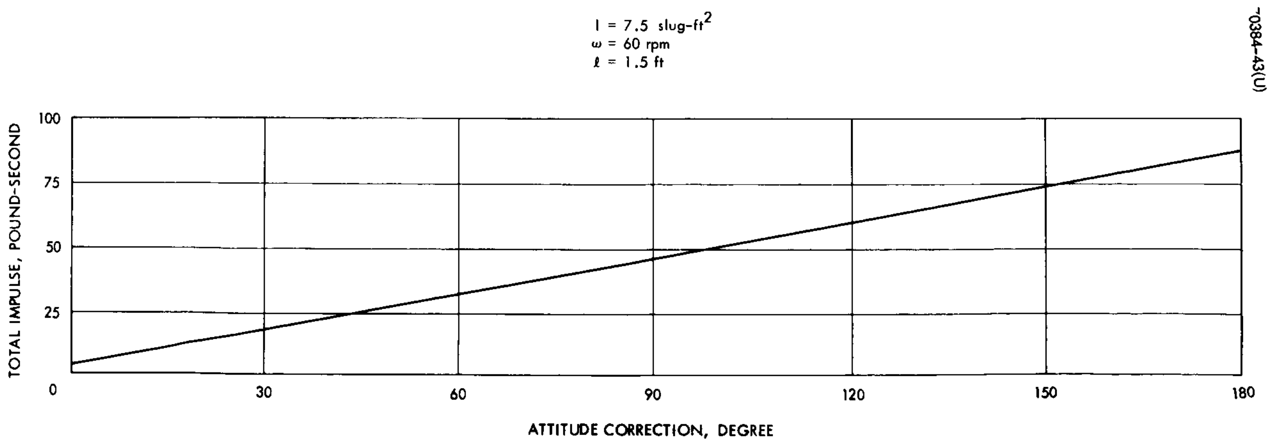


Figure 3-30. Impulse Requirement for Attitude Correction

Finally the correction system is used to adjust the OEC attitude to a new orientation so as to conduct orbit maneuvers.

The requirements for these various corrections can be interpreted in terms of the total impulse necessary to achieve a change. Figure 3-30 illustrates this result in terms of the angular change in attitude and is used to size the attitude correction propellant requirements.

3.4.3 Orbital Maneuvers

The discussion in the preceding sections is pertinent to both the simple co-orbital concept and the orbit change concept. At this point in the discussion, the emphasis is shifted to illuminate the flexibility of the OEC in the orbit change concept.

There are several advantages of the orbit change mission that cannot be realized by the simpler system:

- Capability to perform atmospheric occultation experiments (mother-daughter occultation)
- Provision for adjusting the OEC orbital altitude, inclination or node
- Provision for a stationkeeping mode between Voyager and OEC

The requirements for performing maneuvers are established in the following discussion.. Several orbit change maneuvers are considered: altitude changes, orbit inclination changes, and apse line rotation.

3.4.3.1 Altitude Changes

The velocity requirements to increase or decrease the orbital altitude of the OEC are detailed in Volume II, Section 2.6. The results of this analysis are summarized in the following figures. Figure 3-31 shows the change in the periapsis altitude, Δh_p, as a function of ΔV for several apoapsis altitudes and periapsis altitudes equal to 500 km and 1500 km, respectively. The results show that a capability of 500 fps is more than sufficient to reduce the periapsis altitude to any desired value from the initial Voyager periapsis altitude.

From Figure 3-32, this same velocity capability of 500 fps appears to be sufficient to change the apoapsis altitude by at least 40 percent of the initial value.

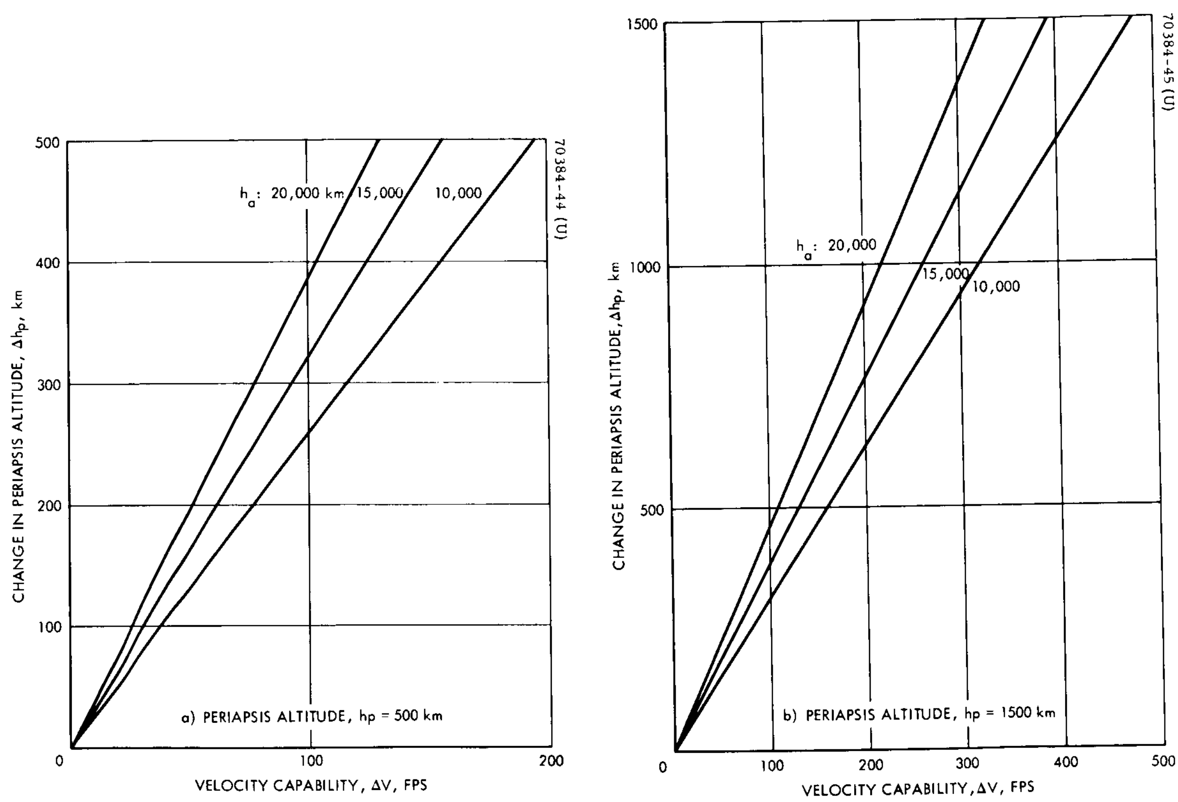


Figure 3-31. Periapsis Change Versus OEC Velocity Capability

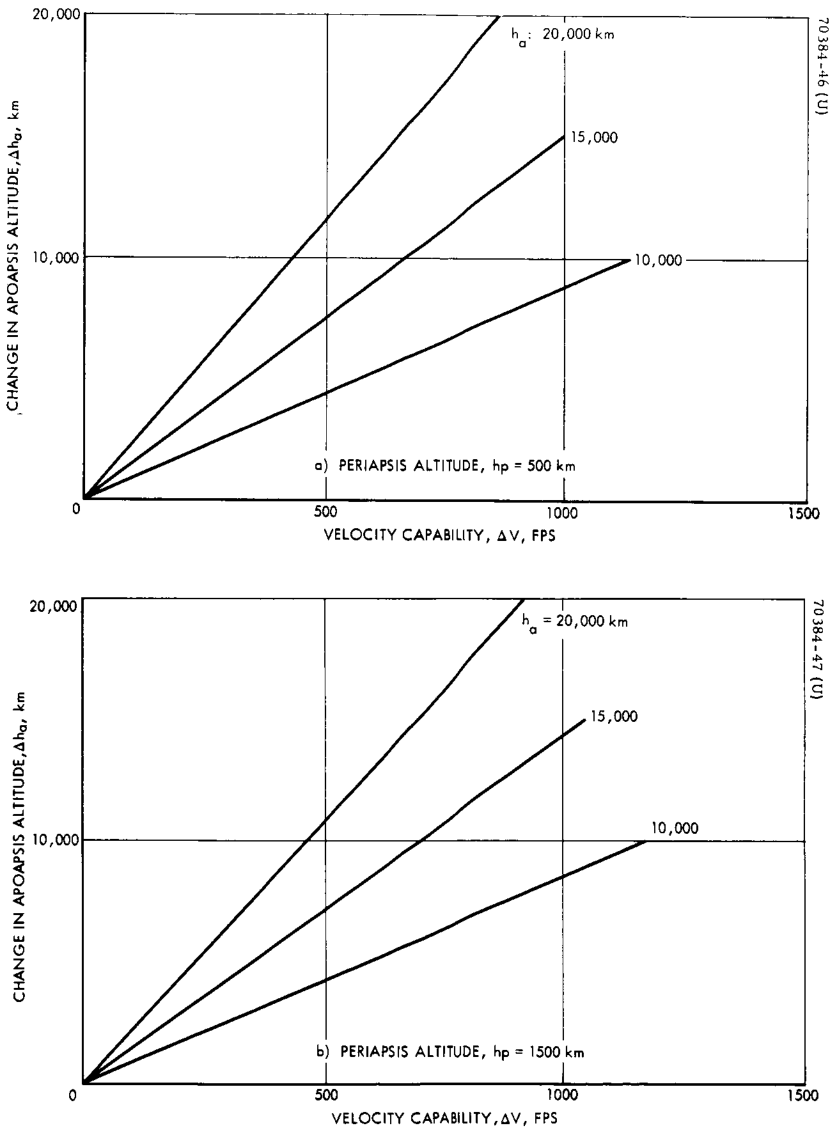


Figure 3-32. Apoapsis Change Versus OEC Velocity Capability

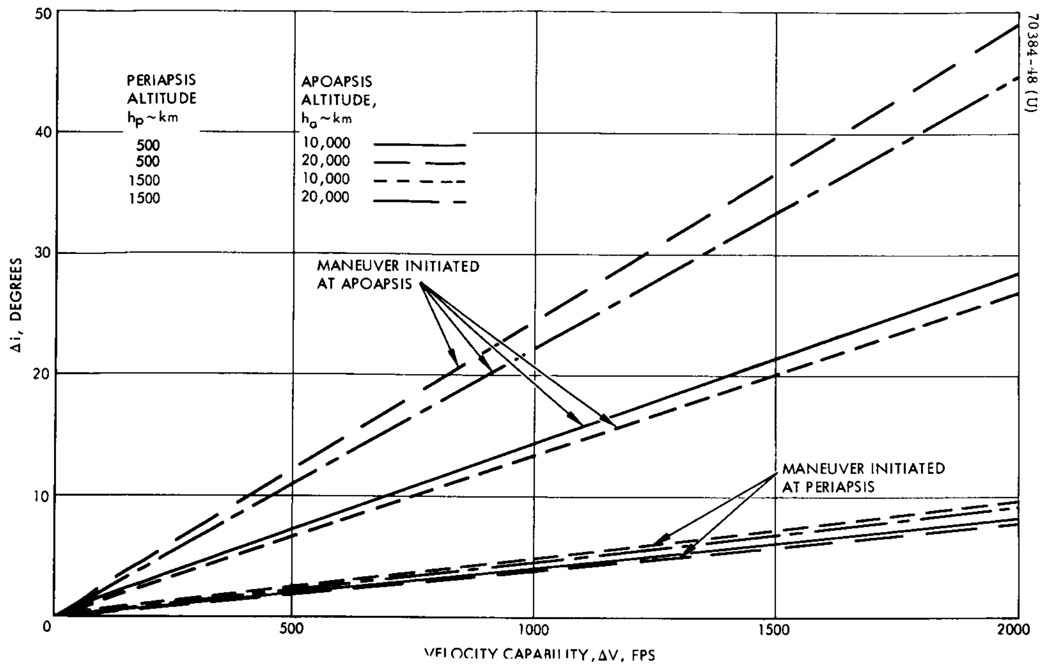


Figure 3-33. Inclination Change Versus OEC Velocity Capability

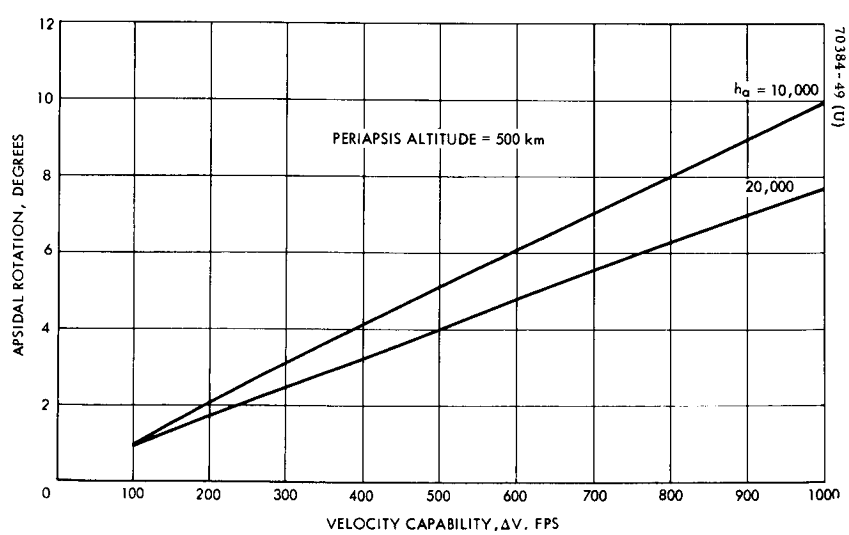


Figure 3-34. Apsidal Rotation Versus OEC Velocity Capability

3.4.3.2 Inclination Changes

Velocity increment requirements for inclination changes are shown in Figure 3-33. Maneuvers initiated from both apoapsis and periapsis are illustrated to show the extremes. The actual selection could be between the two sets of results. It is particularly important to point out that the inclination change obtained depends on the angular location of periapsis relative to the equator (i. e., the argument of periapsis).

The general trend of the results in Figure 3-33 points out the futility in attempting to change inclination. Costs in terms of velocity increment are extremely high. Examining the figure, notice that the larger changes in inclination are developed by maneuvering at apoapsis. For example, a 20 degree change in inclination requires $\Delta V = 1500$ fps for the nominal 10,000 x 1000 km orbit.

3.4.3.3 Apse Line Rotation

From the point of view of the experimenter, it appears that placement of OEC periapsis at the solar subpoint at the beginning of the mission may be desirable. The possibility that the Voyager orbit could not accommodate this requirement prompted the investigation of possibilities to adjust the periapsis position. Control over the location of periapsis is provided by the adjustment of the line of apsides.

Figure 3-34 shows the rotation of periapsis achieved as a function of ΔV expended when the velocity is applied normal to the velocity vector and in the plane of motion at the apoapsis of the orbit. The figure indicates that substantial velocity increments would be required to produce large changes in the location of periapsis. As an example, a velocity increment of 1000 fps, in a 10,000 km by 500 km orbit, will rotate periapsis only 10 degrees. If, however, the apoapsis altitude is decreased to a relatively small value prior to initiation of this maneuver, the capability for rotating periapsis will increase. A combination of this type would require additional satellite reorientation maneuvers, and would therefore increase the complexity of orbital operations. In addition, a ΔV considerably in excess of 1000 fps would be necessary.

Thus it appears that this form of orbit correction is outside the current spectrum of OEC capability.

In summary, it appears that the only logical types of orbit maneuvers are those associated with changing the altitude of the OEC. The number of changes or the magnitude of the adjustment is a function of the weight allotment on the capsule.

3.4.3.4 Voyager – OEC Stationkeeping

The purpose of stationkeeping is to minimize OEC communication ranges by not only ensuring relatively short separation distances but also controlling the relative range between Voyager and OEC. The concept stands out as a viable solution to minimizing power and maximizing the experimental data taken by OEC and transmitted to Voyager.

Figure 3-35 illustrates the manner in which this mission operates. Two orbital maneuvers are required. First, the OEC establishes the proper attitude at apoapsis and maneuvers to a lower periapsis. Without reorienting, the OEC is commanded at the new periapsis to increase its apoapsis by the same amount that periapsis was decreased. In the ideal situation, the new orbit has the same period as the original OEC orbit in which the Voyager remains. By considering a Voyager centered coordinate system, the OEC appears to hover relatively close to Voyager in a manner shown in Figure 3-36. The communications ranges can be kept relatively constant, by the proper maneuvers.

In the actual case, both attitude and orbit trim maneuvers would be required to bring the OEC into the desired geometry and to maintain it. External disturbances such as solar pressure would continually perturb both vehicles from the desired nominal state; hence stationkeeping at some fixed interval of time would be used.

This form of maneuvering is not considered in the baseline mode of OEC operation but does appear to represent a reasonable alternative.

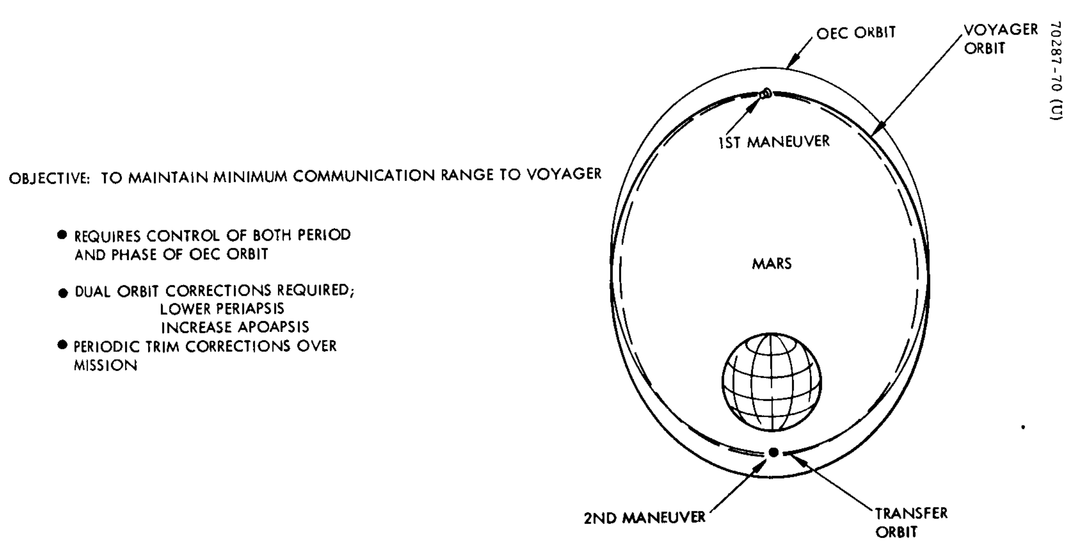


Figure 3-35. Voyager—OEC Stationkeeping

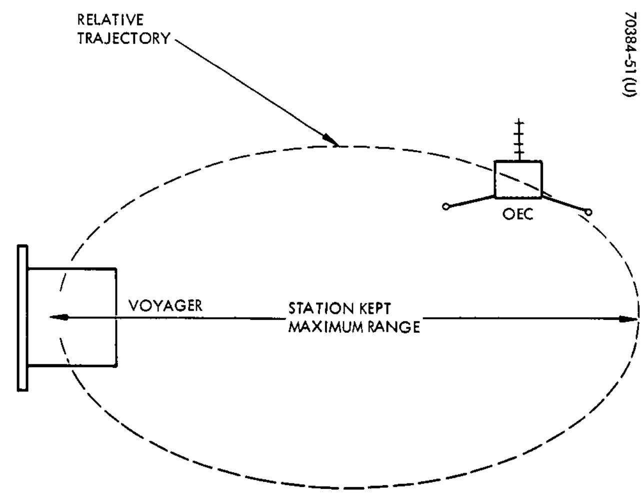


Figure 3-36. Relative Voyager—OEC Motion

3.5 ATTITUDE STABILIZATION, CONTROL, AND ORBIT CONTROL IMPLEMENTATION

The OEC is a spin stabilized satellite of Mars. This method of stabilization is not only the most desirable from a scientific point of view since it does provide a natural spinning platform for gathering field and particle measurements but is also the most desirable from a satellite design point of view. Stabilization as such is passive and is the least expensive and most reliable technique available.

With the addition of an active control capability, the versatility of this system increases dramatically. In its simple mode of operation (co-orbital mission), the OEC is separated, spun up to 60 rpm, and remains in a particular orbit relative to Voyager. By the addition of the active attitude control system, several important additional mission experiments can be carried out.

The desirable features of an active attitude control system are:

1) Capability to initially align spin axis normal to the ecliptic plane for all viable OEC missions.

2) Provision for attitude corrections of external perturbations.

3) Capability of providing additional OEC magnetic field component measurement by reversing the direction of the spin axis attitude.

4) Necessary to provide orientation for orbital maneuvers.

Once the attitude control system is part of the satellite design, the extension to orbit control is rather simple.

Orbit control extends the OEC mission flexibility in many ways; for example:

1) Provision to map the magnetosphere at altitudes very near to the planet.

2) Capability to change apoapsis altitude to investigate the shock boundary in the tail of the magnetosphere.

3) Possibility of performing a stationkeeping mode of operation.

A summary of the basic stabilization concept and the sizing of an attitude/orbit control system is presented next.

3.5.1 OEC Spinup System

Following separation from the Voyager spacecraft, the OEC will be spun up to the predetermined rate. A spinup system is included on the OEC to spin the vehicle to 60 rpm. This stabilization technique provides a gyroscopic "stiffness" and allows relatively large thrust levels to be utilized for velocity and attitude control maneuvers without appreciably disturbing the desired orientation. In addition, this form of stabilization allows for a reasonably long period of time to pass before orientation errors due to natural disturbance torques need be corrected.

Spin will be performed by a nitrogen pressure blowdown system. A thrust level of 130 millipounds is selected based upon considerations in Section 2.4.2, Volume II. A total impulse of 33.3 lb-sec is required to bring the OEC up to 60 rpm.

3.5.2 Nutation Damping*

The undesirable disturbances introduced in the capsule during the spinup such as thrust mismatch or jet misalignment force the vehicle into what is classically called a free precession or nutation. By definition, this is an angular motion of the body axis about the angular momentum vector. As such, the induced nutation angle can be removed by the addition of a passive device which dissipates the ntuational energy via fluid viscosity. This device is called a nutation damper and is located on the vehicle parallel to the spin axis as shown in Figure 3-37. The distance from the spin axis fixes the nutation damping time constant.

Nutation damping for the OEC can be provided by a damper similar to that designed for the Syncom/Early Bird satellites. This type of damper is passive and consists of a fiberglass tube partially filled with mercury. The spacecraft nutation motion results in a buildup of surface waves in the fluid, causing the nutational energy to be dissipated via the fluid viscosity.

The damper similar to that proposed for the OEC is illustrated in Figure 3-38; the damper characteristics are presented in Table 3-12. A damper time constant of 10 minutes appears satisfactory.

3.5.3 Control Considerations

The manner in which the attitude and orbit of a spinning satellite are adjusted is based on laws of rotational dynamics. As shown in Figure 3-39, an axial thruster is required to precess the spin axis. The direction of precession is selected by the position in the spin cycle that the thrust is applied as well

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*Because of current convention at Hughes, the term "nutation" and free precession are synonomous. Forced precession does, however, hold to its classical definition and is considered as "precession" in the text.

TABLE 3-12. OEC DAMPER CHARACTERISTICS

Diameter	0.8 cm
Length	18.0 cm
Fraction filled	0.2569
Distance from center of mass	30.5 cm
Mercury weight	0.03175 kg
Tube, cap weight	0.01814 kg
Total weight	0.05 kg

as the duration. Thrust is commanded about an axis 90 degrees out of phase with the actual desired motion. Performance of a correction requires that both the pulse initiation angle and the total number of pulses be known beforehand. The initiation angle is referenced to the Sun sensor signal.

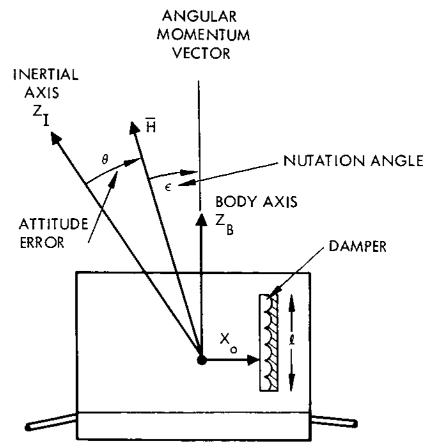
The axial jet thrust selected for the control system operates at 3 pounds over a spin angle of 21 degrees; 60 milliseconds is assumed as a minimum on-time for the jet. Based on these numbers, both the attitude correction and orbit correction per pulse can be determined. The nominal values are

$$\Delta\theta = 0.2262 \text{ degrees/pulse}$$

$$\Delta V = 0.06 \text{ fps/pulse}$$

There are two techniques for correcting the orbital position of a spinning satellite. The first operates in a pulsed mode identical to the axial thruster but is aligned along an OEC radius. Referring to Figure 3-40, the application of thrust is through the center of mass of the OEC, producing a translational motion in the desired direction. The other approach is one already mentioned — that is, to apply the axial thruster in a continuous operational mode. In the continuous mode the thruster provides a $\Delta V \sim 1$ fps/revolution.

There are several differences in these approaches and their applications. First, pulsed operation is inherently less efficient in producing the desired thrust level because of a slight degradation in specific impulse; thus, the continuously operated axial maneuver is more desirable. On the other hand, an orbit maneuver conducted with the radial jet has a built-in safety for an open failure. For example, if the radial jet solenoid fails to close the fuel will be totally expended, and the average motion of the vehicle about the spin axis is cancelled. A similar failure of the axial jet has no degraded mode. Orbit maneuvers with the axial thruster have a slightly greater I_{sp}. Mission performance evaluation indicates that both modes could be necessary to carry out the orbit change mission. Thus, both methods of orbit control are chosen for the baseline since the cost of adding the radial thruster to the system is small and the redundancy afforded the orbit change mission is reasonable.

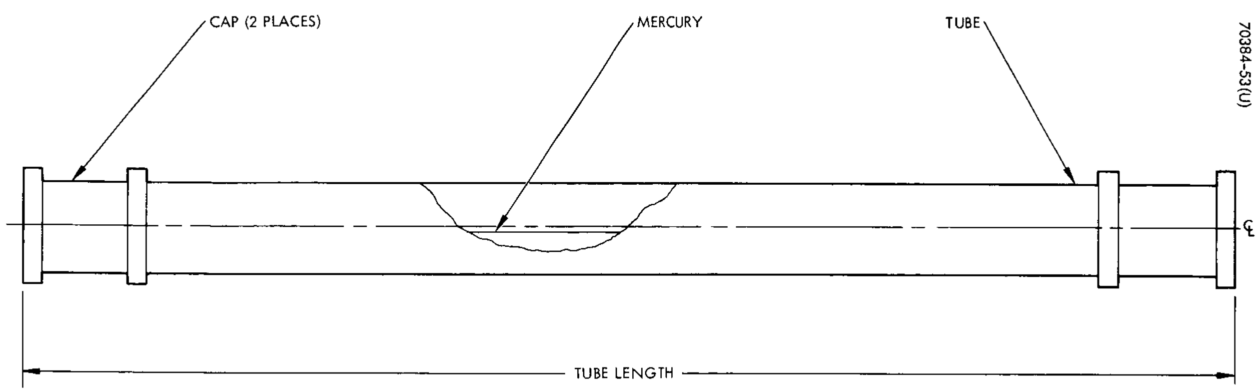


70384-52 (U)

CHARACTERISTICS

MERCURY WEIGHT	0.03175 kg
TUBE WEIGHT	0.01814 kg
LENGTH	18 cm
DIAMETER	0.8 cm
FRACTION FILLED	0.2569
DISTANCE FROM CM	≥30.5 cm
TIME CONSTANT	10 minutes

Figure 3-37. Nutation Damper



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INSIDE DIAMETER	= 0.787 - 0.813 cm
OUTSIDE DIAMETER	= 0.975 - 1.026 cm
LENGTH	= 17.932 cm
TUBE WEIGHT	= 0.088 kg

Figure 3-38. Syncom, Early Bird, and HS-303A Nutation Damper

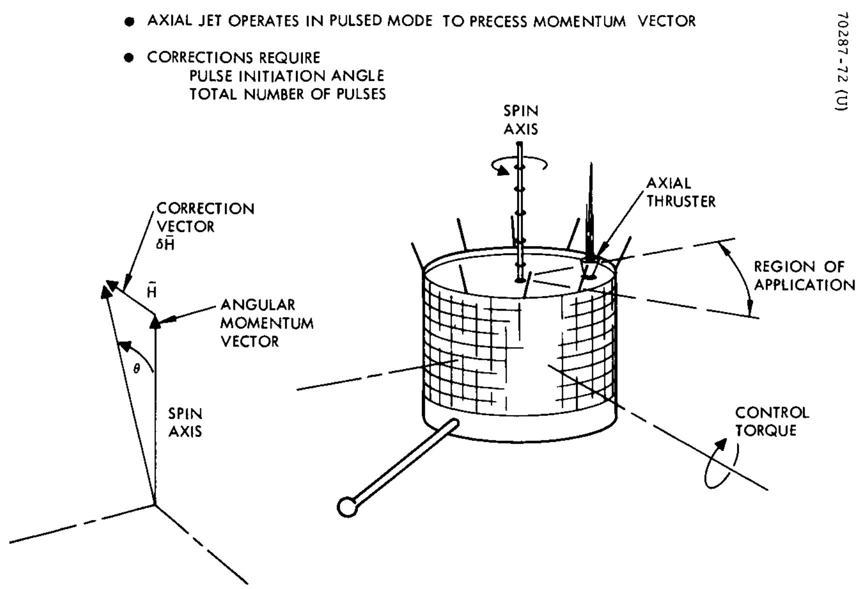


Figure 3-39. Attitude Correction Technique

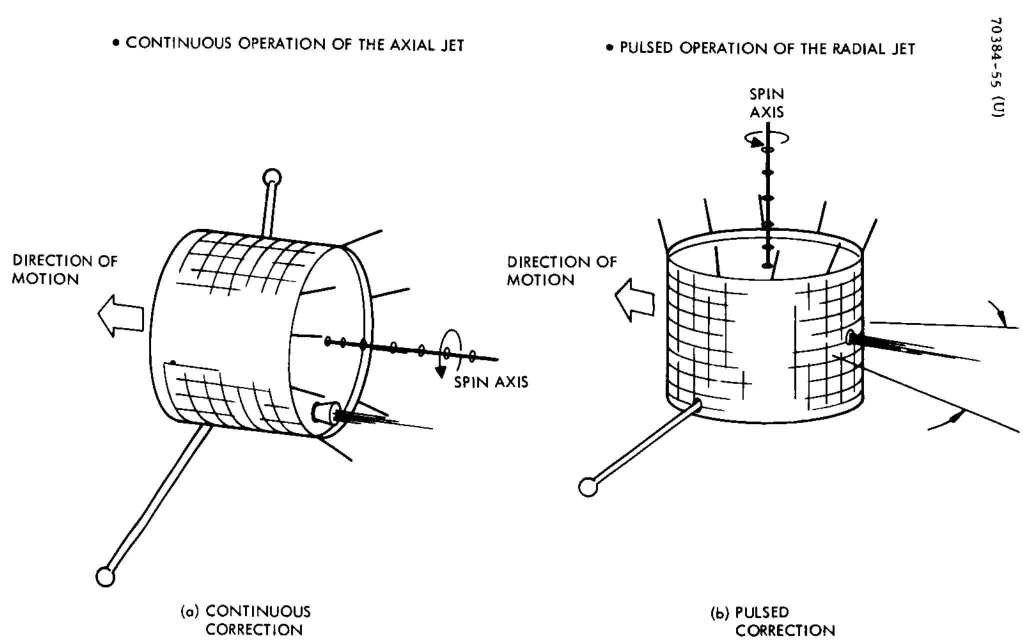


Figure 3-40. Orbit Correction

3.5.4 Performance

A hydrazine propulsion (see Section 5.0) is selected for the attitude/orbit control function. The fuel requirements for the propulsion system are sized by both the disturbances and the maneuvers.

Table 3-11 summarized the per orbit disturbance due to each of the perturbative sources. The errors were combined in a root sum square fashion to determine the correction frequency and the total impulse required to correct these attitude errors over the 6 months.

The rss attitude error is given in Table 3-13. Based on these errors, the requirements to maintain the attitude to < 5 degrees are ascertained. A correction frequency of about 6 months (mission lifetime) is necessary for a nominal periapsis altitude of 1000 km for that duration. At the lower altitudes, the increased disturbance level necessitates correcting the attitude more often. It is very important to realize that the 5 degree constraint is only a desirable goal to meet, and that increasing this requirement by 1 or 5 degrees does not affect the solar plasma instrument but increases the complexity to reduce its data. Increasing this requirement by just 1 degree increases the system operation by an additional month. Hence, it is definitely reasonable to operate the co-orbital mission of the OEC for the 6 months with the desired accuracies and beyond that time for up to a year with accuracies of 11 degrees. This is limited only by the OEC subsystems ability to continue to function properly.

TABLE 3-13. EFFECT OF ATTITUDE DISTURBANCE

Periapsis Altitude, km	Attitude Error, deg/orbit	Total Impulse (6 Months), lb-sec	Fuel Weight, pounds	Correction Frequency, months
1000	0.0083	4.1	0.02	~6
500	0.0093	5.0	0.025	<5
300	0.011	6.2	0.03	<4

- Allowable attitude error — 5 degrees
- Apoapsis — 10,000 km
- Spin rate — 60 rpm
- Pulse axial jet
  - Angle — 22 degree/pulse
  - Duration — 60 milliseconds
- Thrust level — 3 pounds
- Correction increment — 0.2262 deg/pulse

TABLE 3-14. TYPICAL ORBIT/ATTITUDE CORRECTION

Maneuver	ΔV for Orbit Change Maneuver from Periapsis, fps		
	1500 km	1000 km	500 km
Initial attitude orientation (360 degrees)	60	60	60
Attitude reorientation (90 degrees)	15	15	15
Lower periapsis to 350 km	355	205	50
Total ΔV	430	280	125
Required fuel, pounds	7.7	5.3	2.5

- Apoapsis — 10,000 km
- Spin rate — 60 rpm
- Axial jet thrust — 3 pounds
- Radial jet thrust (pulsed) — 3 pounds
- Pulse accuracy — ΔV = 0.0599 fps/pulse; Δθ = 0.2262 deg/pulse

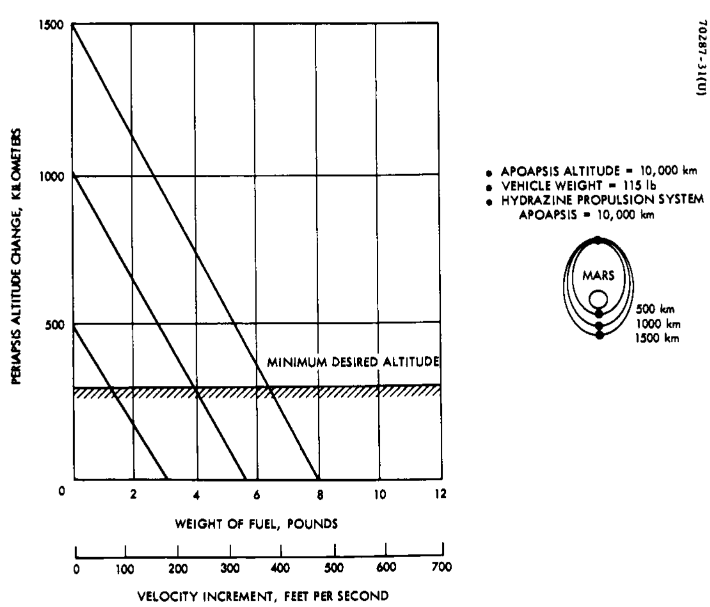


Figure 3-41. Adjustment of Periapsis Altitude

The total impulse requirements are shown in Table 3-13. Assuming the nominal system parameters, the total impulse is found to vary between 4 and 6 lb-sec. Based on the performance characteristics of hydrazine, the fuel weight requirements are shown to be very small, in the range of 20 to 30 millipounds.

Fuel allotments for both attitude and orbit requirements are far greater than those to trim the attitude of the OEC. Typical propulsion system requirements are determined for a sequence in which

1) An initial 360 degree OEC attitude maneuver is required some time after separation from Voyager. (Part of this is to conduct reorientation for magnetic field measurement.)

2) A reorientation in attitude of 45 degrees is required to operate the axial or radial jet.

3) An orbit adjustment to lower periapsis altitude from 1500 km, 1000 km, or 500 km to 350 km is required.

4) Attitude is reestablished along the ecliptic normal by a 45 degree attitude correction.

The results are tabulated in Table 3-14 assuming a 3 pound thrust level as well as the jet characteristics sited earlier. The velocity increment and fuel requirements for the periapsis attitude correction are shown in Figure 3-41. A summary of the results given in the table indicates the minimal fuel requirements to provide attitude corrections, as well as the small increase in OEC system weight. This weight increase accounts for propellant to adjust the altitude to minimum distances from Mars which are commensurate with the 50-year lifetime constraint.

The selected baseline OEC has been designed to provide the attitude and orbit adjustments to maneuver from a nominal 10,000 km x 1000 km altitude to a periapsis altitude of 350 km. As indicated in Table 3-13, a fuel budget of 3.5 pounds (ΔV=205 fps) is necessary.

4.0 COMMUNICATIONS AND DATA HANDLING

The components needed for accepting commands from earth, for data retrieval and for sequencing vehicle events, are regarded as the telecommunications subsystem. Studies that have led to the selection of data links and requirements for these components are presented in Volume II, Section 3.0. A summary of the study results and of a recommended set of components is presented below. Functional requirements have been deduced from the list of nominal experiments and the range of orbits and accuracies outlined in Reference 1.

4.1 SUMMARY OF COMMUNICATIONS STUDIES

Requirements listed in Table 4-1 were used for selecting the range of parameters treated in the study and for determining powers and capacities of the recommended components for an illustrative design. Curves included here and in Volume II may be used to enter perturbations in the mission requirements and deduce new components characteristics.

Two paths were candidates for communication both to and from the OEC: a direct link between OEC and Earth (Deep Space Net) or a relay link through the Voyager. Link calculations showed that the relay path is necessary for transmitting data, and either path is adequate for sending commands to the OEC.

Two alternatives again arose in evaluating the relay link: the continuing short range link offered in the co-orbital case and the long range, intermittent opportunities that typify the orbit change case. The first situation permits use of a low power transmitter with no need for a storage device. These requirements are easily met in the light-weight spacecraft described for the co-orbital case. The second type of mission involves much higher energy for the transmitter as well as data storage capability, and every aspect of the communications subsystem must be optimized to remain within the maximum weight constraints. The latter system has been emphasized in the following discussion because this mission offers opportunity for greater experiments flexibility and because the communications selections are more critical.

4.1.1 Direct Communications Link

The possibility of a direct link to the DSN was first investigated and the energy necessary to send the body of experimental data desired was found to exceed the potential resources of the spacecraft by a factor between 23 and 800. The wide difference in these factors reflects range variation, coding, and minimum-versus-nominal performance which can be seen in Figure 4-1. It was felt that this level of degraded performance could have value as an emergency mode; therefore, an S-band transmitter has been included to permit some data to be retrieved in the event of a

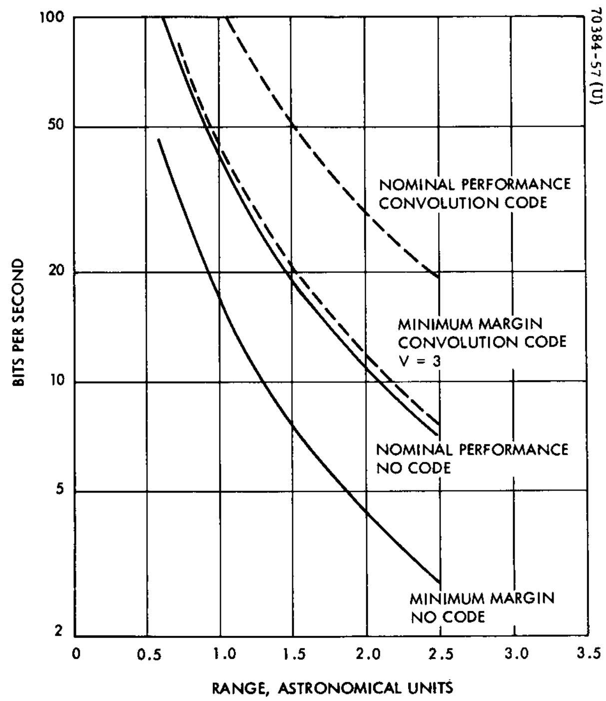


Figure 4-1. Link Capability for 10 watt Traveling-Wave Tube into DSN Versus Range

TABLE 4-1. COMMUNICATIONS REQUIREMENTS

• Sampling rate for experiments	550 bits/sec or 1 frame/spin
Distance between samples	<4.4 km at periapsis, <1.3 km at apoapsis
• Mars sensor data	8 bits/sec, averaged
• Clock-sun sensor	15 bits/sec
• Engineering data	50 bits/sec, averaged
• Data rate	630 bits/sec or 16 megabits in 10,000 km orbit 32 megabits in 20,000 km orbit
• Commands	
Set magnitudes into sequencer	7
Start sequences	4
Select data readout modes	4
Backup capability on-off	16 (32 commands)

failure in the relay link or even if the entire Voyager spacecraft experiences some catastrophe. Following the philosophy of making the vehicle potentially independent of the Voyager, an S-band command receiver was incorporated after a data link calculation showed that a direct DSN to OEC link can support 10 bits/sec (bit error rate 10^{-5}).

4.1.2 Use of Coding

A brief summary of existing coding techniques was conducted with a view to OEC applicability; coding is recommended for both S-band links. The reasons for coding are different for the two links, and reference should be made to Section 3.0 of Volume II for details. In brief, the coding for the command link involves error correction, as well as error detection, so that the rate of command rejection and retransmission is reduced from one in 10^3 to one in 10^6 attempts. For the same energy per information bit, the word error rate is reduced from 10^{-5} to 10^{-10}. A word format of 10 bits with an additional 5 binits is recommended to use a Hamming "distance-four" code implemented for double error correction and single error detection. Convolutional coding is recommended for the emergency mode data link in order to realize a greater amount of data retrieved for the same energy expended. A code using 3 binits for every information bit has been shown to use a modest amount of circuitry, be programmed for decoding on ordinary computers, and permit sufficient

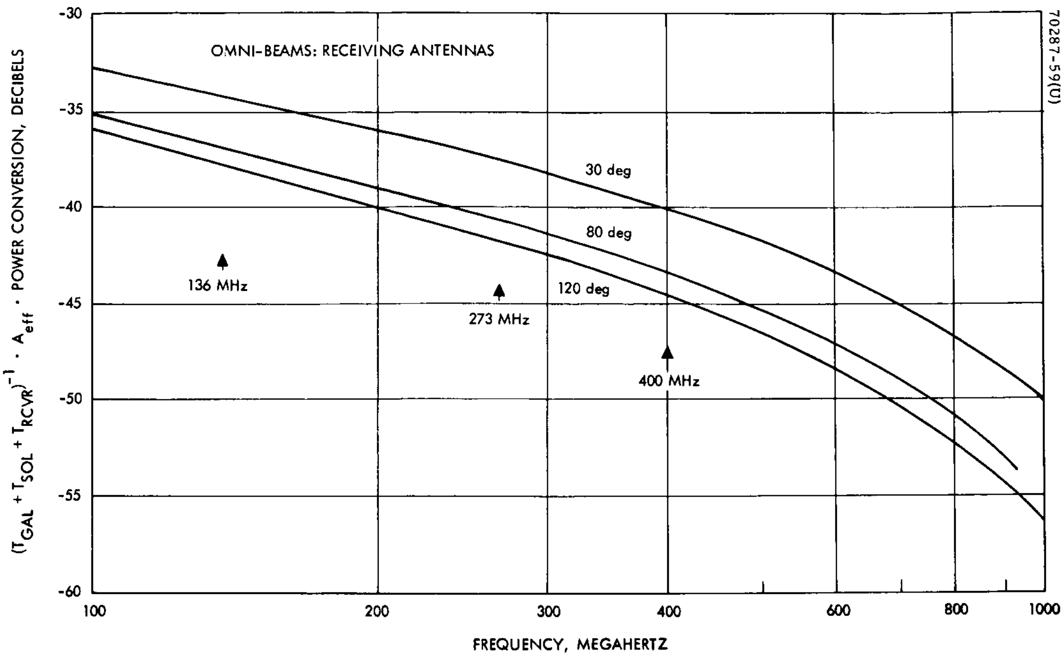


Figure 4-2. Frequency-Dependent Factors Affecting Link Capacity

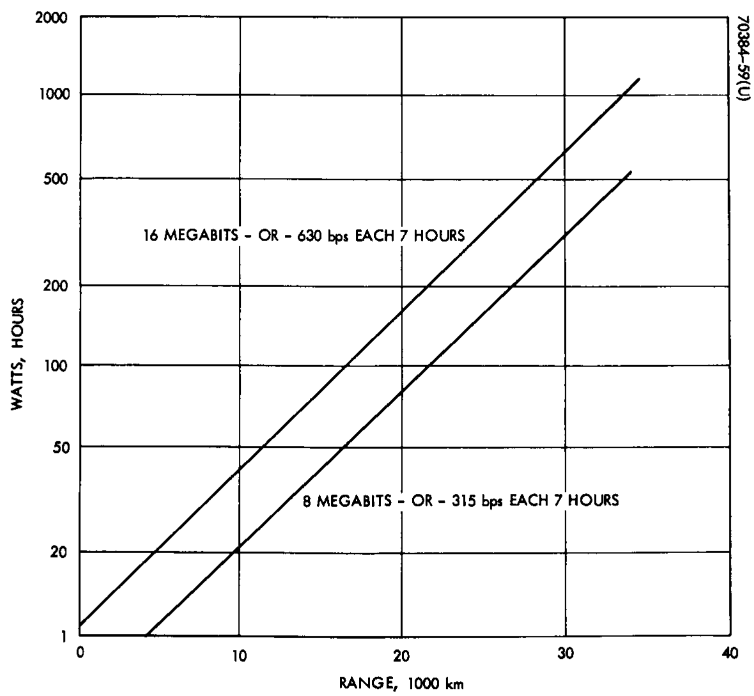


Figure 4-3. Energy Required at 136 MHz to Transmit Various Amounts of Data Versus Range

signal to noise for practical receiver lock-up times. The advantages in bit rate can be read from Figure 4-1.

4.1.3 Selection of Relay Link Frequency

A number of link capabilities were calculated after deciding that the primary data mode must use a relay link through the Voyager. Frequency effects were examined in detail and are summarized in Figure 4-2. The most significant trend of diminishing link capability with increasing frequency occurs because antenna coverage is a fixed quantity for both the OEC and Voyager receiving antenna; hence the effective catchment area diminishes at 6 db/octave. The several other factors tend to cancel each other.

The band at 136 MHz has been recommended for the orbit change mode because this frequency permits the ideal experiments sample rate to be transmitted from 75 to 100 percent of the mission, depending on the orbit and change selected. If the 273 MHz channel were selected, then the ideal sampling rate could be accommodated only 50 to 64 percent of the time for the orbital extremes. Selection of 400 MHz would further reduce the number of orbits for successful sampling to 44 and 56 percent.

4.1.4 Energy Required for Transmitter

Energy requirements for transmitting the entire desired amount of data were calculated and balanced against the solar resources of the vehicle. These requirements are shown in Figure 4-3, in which watt-hours are plotted against range for a system operating at 136 MHz. The two higher frequency channels under consideration (273 MHz and 400 MHz) had such exorbitant requirements that they are not even shown, but may readily be deduced from Figure 4-3.

Bearing in mind that data will be stored in a tape recorder and read out at the most convenient times, it is useful to express the link in terms of energy, since this quantity is independent of rates and times. Having the range over which the desired data can be transmitted, the pertinent orbits were examined to determine how long and how often the two vehicles would remain within the supportable distances.

4.1.5 Orbital Considerations With Respect to Data Readout Opportunities

Relative positions of the two vehicles in orbit were examined to determine when reasonable ranges would occur for substantial time intervals. Even a cursory examination of the orbits reveals that the range increases as the interval increases; therefore, from the energy aspect, it is desirable to send all of the information in a "burst" at the time in each orbit when minimum range occurs. Factors mitigating against this trend are the high power and weight for a transmitter to handle the short term load and the increase in battery weight as the proportion of power from the solar panels becomes less significant. A few configurations of the power supply and transmitter based on actual ranges and intervals served to show that 2-hour intervals for the closest orbit and 4 hours for the largest orbit yielded a practical design and these values are used for the ensuing discussions. The range of practical designs appeared to encompass intervals from one-half to twice those chosen.

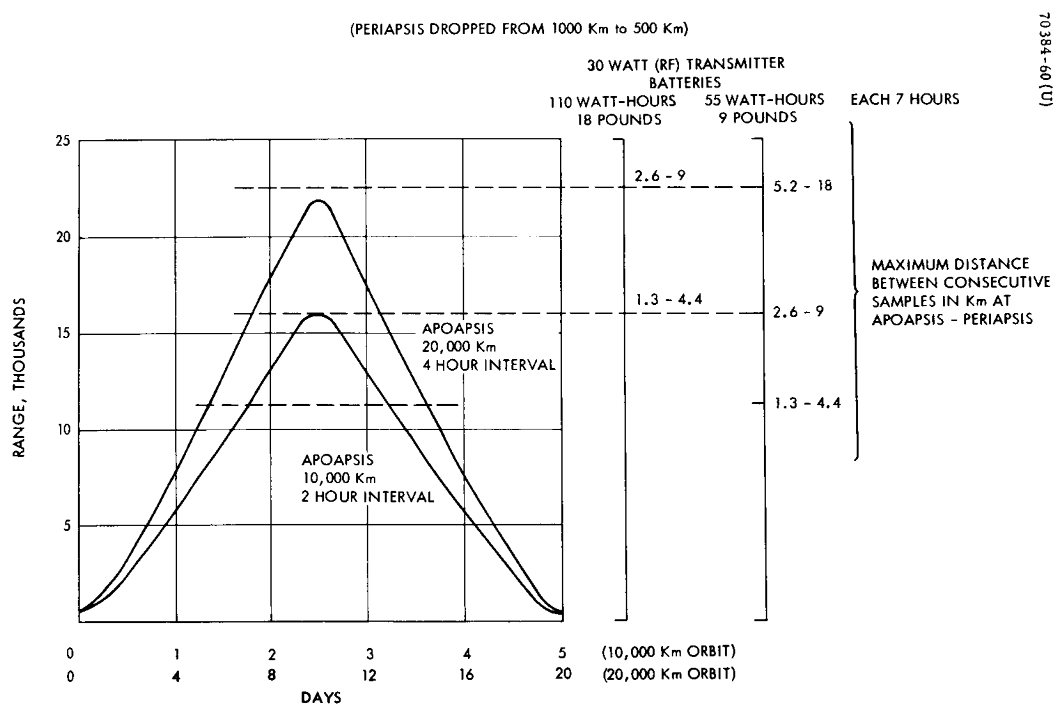


Figure 4-4. Communication Readout Ranges and Sampling Spacing for Typical Orbit Change

Figure 4-4 gives a profile of occurrence of readout opportunities for two extremes of orbit situations. In the upper curve, the two vehicles have been injected in a 20,000 to 1000 km orbit and then the OEC has been dropped to 20,000 to 500 km. The two vehicles will slowly part and then grow closer together since the new OEC orbital period is then 24 minutes shorter than the original orbit occupied by the Voyager spacecraft. The range is shown for four readout periods, and the orbital time is 14 hours. The lower curve shows the same situation for 2-hour readout opportunities with 7-hour orbits. In this way the duty cycle is maintained constant. This corresponds to the fact that the data collected and the energy collected from the sun are both constant for a given interval of time.

The ordinates on the right of the curve represent the amount of data that can be accommodated for particular ranges if batteries are carried that can provide 110 and 55 watt-hours respectively. The two curves marked 1.8 to 4 km indicate that all experiments are sampled every spin cycle. The greater distance represents sampling less frequently, such as every other spin or every fourth spin. This is only one way to use a degraded mode; the experimenter might also elect to maintain the sampling density and restrict the sector of the orbit sampled.

The 110 watt-hour battery, weighing 18 pounds, is shown in the 122 pound OEC. The 55 watt-hour battery would permit a 113 pound OEC. The proportion of time during which a particular data collection rate can be sustained is deduced from the number of days during which the curve remains below the appropriate horizontal line; e.g., the 18 pound batteries always permit maximum sampling in the 10,000 km apogee orbits and 75 percent of the time in the 20,000 km apogee orbits. These curves represent a synodic cycle that repeats periodically with the periodicity shown. More extreme orbital changes result in shorter periods, but the ordinates remain constant.

The 4- and 2-hour readout intervals are consistent with the 33 watt transmitters shown for the recommended configuration.

These power supply weights and capacities are affected severely by eclipse operations. The design philosophy for those orbits is set forth in Section 5.2 of this volume.

4.1.6 Modulation Method

The systems ramifications of a non-coherent FSK on the carrier versus a coherent PCM/4-M using a subcarrier oscillator were evaluated in terms of Voyager equipment complexity and link capability. It is concluded that the more complicated receiver necessary for the coherent modulation method is warranted by the real advantage in link capability. This advantage has the effect of offering a 7.3 db saving in energy. The significance of this value can be emphasized by recalling that the batteries and transmitters proposed in the previous paragraphs represent maximum spacecraft weight and are based on the more efficient modulation method.

An analysis was made of multiple frequency shift keying (MFSK) to assess its value for the emergency S-band data link. This was rejected after the study of frequency drift effects which is included in Volume II.

4.2 COMPONENTS DESCRIPTION

The foregoing studies have produced an illustrative system shown in the block diagram of Figure 4-5. Characteristics for each component are described below, and each function is either satisfied by existing equipment or can be met by scaling existing designs. As indicated in the studies, it is possible to trade VHF transmitter power for time of transmission per orbit; therefore, if a slightly different transmitter should become qualified, it could probably be used. Similar latitude pertains to the S-band transmitter and the tape recorder. The number of operational modes should also be kept open pending orbit selection and direction from experimenters.

VHF Antenna. The OEC-mounted antenna serves the 136 MHz transmitter with coverage over a 140 degree omnidirectional region. Combined feed line losses and gain should be greater than -6 db with respect to isotropic.

VHF Transmitter. A 136 MHz solid-state transmitter will provide 33 watts of RF power for 55 watts input. Existing designs can be combined in units providing increments of at least 20 watts each. A series phase modulator should be capable of modulating two subcarrier oscillators on the carrier with modulation indices of 0.53 radian and 1.1 radians. These will permit real time data to be transmitted simultaneously with data stored on the tape recorder and will also permit a real time backup mode in event of a tape recorder failure. The power penalty to the subcarrier for stored data is only 1 db.

Encoder. Requirements for the encoder cannot be made firm until answers regarding possible experiments format and desired commutation modes are known. The small number of experiments and the modest data rates — ~630 bits/sec — indicate that an existing encoder or a minor modification of an existing encoder will be applicable.

Controller and Sequencer. Requirements for the controller and sequencer are not firm for the same reasons stated for the encoder; however, the accuracies and number of sequences outlined for this application appear to be within the capabilities of an ATS type of clock and sequencer and are certainly within the capability of a central controller such as was used on Surveyor.

Decoder. A suitable demodulator and decoder using 40 percent microminiature circuits is presently being flight-qualified for a larger, more complex satellite. This decoder can be simplified and circuitry for using the Hamming (15, 10) code for double error detection and single error correction can be added. The coding circuitry has been designed using two "flat packs."

Tape Recorder. A Leach 2200LP tape recorder has been designed especially for low power satellite use. A single recorder can be used for the OEC, fitted with heads for two channels and programmed so that all data can be conserved even if it should be necessary to have two different readout periods per orbit. Data would be entered at 630 bits/sec with tape moving at 0.5 ips. The present tape capacity is 1800 feet, and only 1100 feet is required for the OEC. Readout to read-in ratio is only 4 to 1, so that a single drive motor can be used. An estimate of the magnetic effect of the tape recorder is included in Section 6.0 of this volume and Section 7.0 of Volume II.

S-Band Transponder. The Surveyor transponder may be used preceded by a tunnel diode amplifier for improved noise figure. The carrier tracking loop should be modified for a $2B_{LO}$ of 20 Hz, and automatic search capability must be added.

S-Band Antenna. A 7 degree omnidirectional antenna using 30 inches for a collinear array must be developed. A circularly polarized element has been designed at UHF and can be scaled for this application.

Growth Capability. The primary area for communication system growth is that of increased data transmission. As power can be made available, the 20 watt increments can be added to the transmitter with a weight increase of about 5 ounces per increment. Each increment will involve thermal dissipation of 13 watts and prime power of 33 watts.

Added storage capability would be required and is best achieved by adding tape recorders. Adding recorders rather than channels has the advantage that system reliability is enhanced.

The command decoder has roughly twice the command capability presently envisioned.

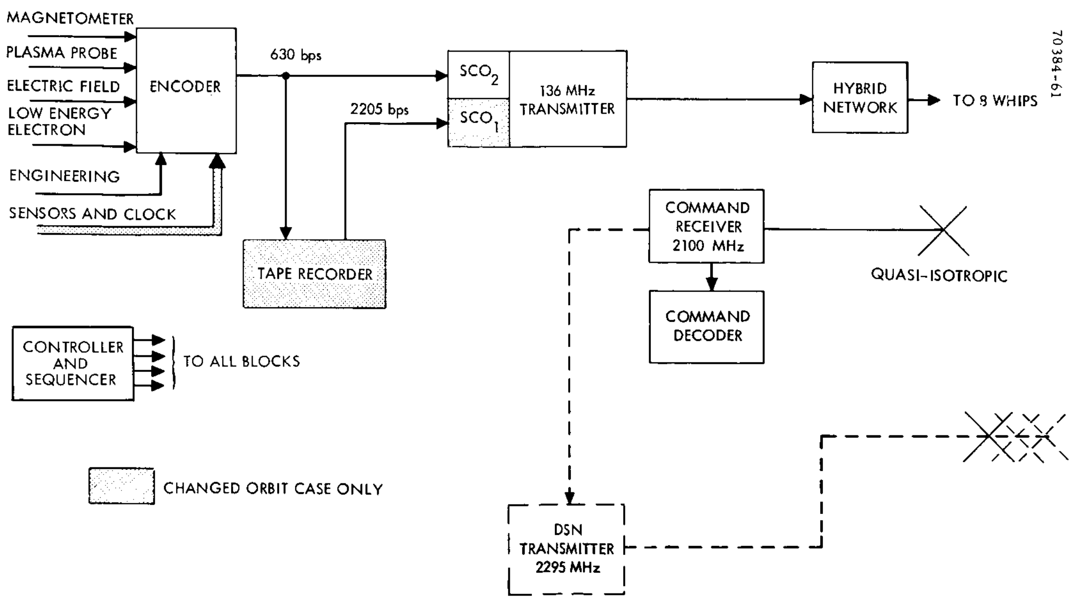


Figure 4-5. Communications Block Diagram

5.0 OEC CONFIGURATION AND SUBSYSTEMS

5.1 OEC CONFIGURATION STUDIES

5.1.1 General

The OEC baseline configuration which is discussed in detail in this section was formulated as a result of a number of configuration iterations necessitated by the results and conclusions arrived at throughout the study. Certain key areas had to be investigated and their ramifications affecting the OEC configuration fully evaluated before a feasible design concept could be formulated. Of significant importance in establishing an OEC configuration were the studies conducted in the following interrelated areas:

1) Voyager Spacecraft Designs - There are presently three Voyager spacecraft contractors (GE, TRW, and Boeing), each having evolved a conceptual design of a 1973 Voyager spacecraft bus. An evaluation had to be conducted to establish the constraints imposed by each candidate design on the configuration selection of the OEC.

2) OEC Geometric Relationships - In addition to the space envelope restrictions imposed on the OEC configuration by Voyager, it was necessary to establish the compatibility of spatial and size relationships in three major areas: the solar cell array, the volume and orientation requirements of the experiments, and a preliminary assessment of capsule subsystems spatial requirements.

3) Boom Configurations - The requirement that the OEC carry a magnetometer necessitates that the capsule be designed to accommodate booms to mount the magnetic sensor a sufficient distance from the capsule magnetic field. Studies were carried out to establish not only the effects of the boom(s) placement on the capsule but also the type of boom system best suited for the mission.

4) OEC Moments of Inertia Implications - Attitude stabilization of the OEC was specified as a passively spin stabilized system. For spin stabilization, the moment of inertia of the capsule about the spin axis must be greater than that about the transverse axes. Thus, the OEC configuration studies had to include the investigation of boom(s) and capsule geometric and mass distribution relationships.

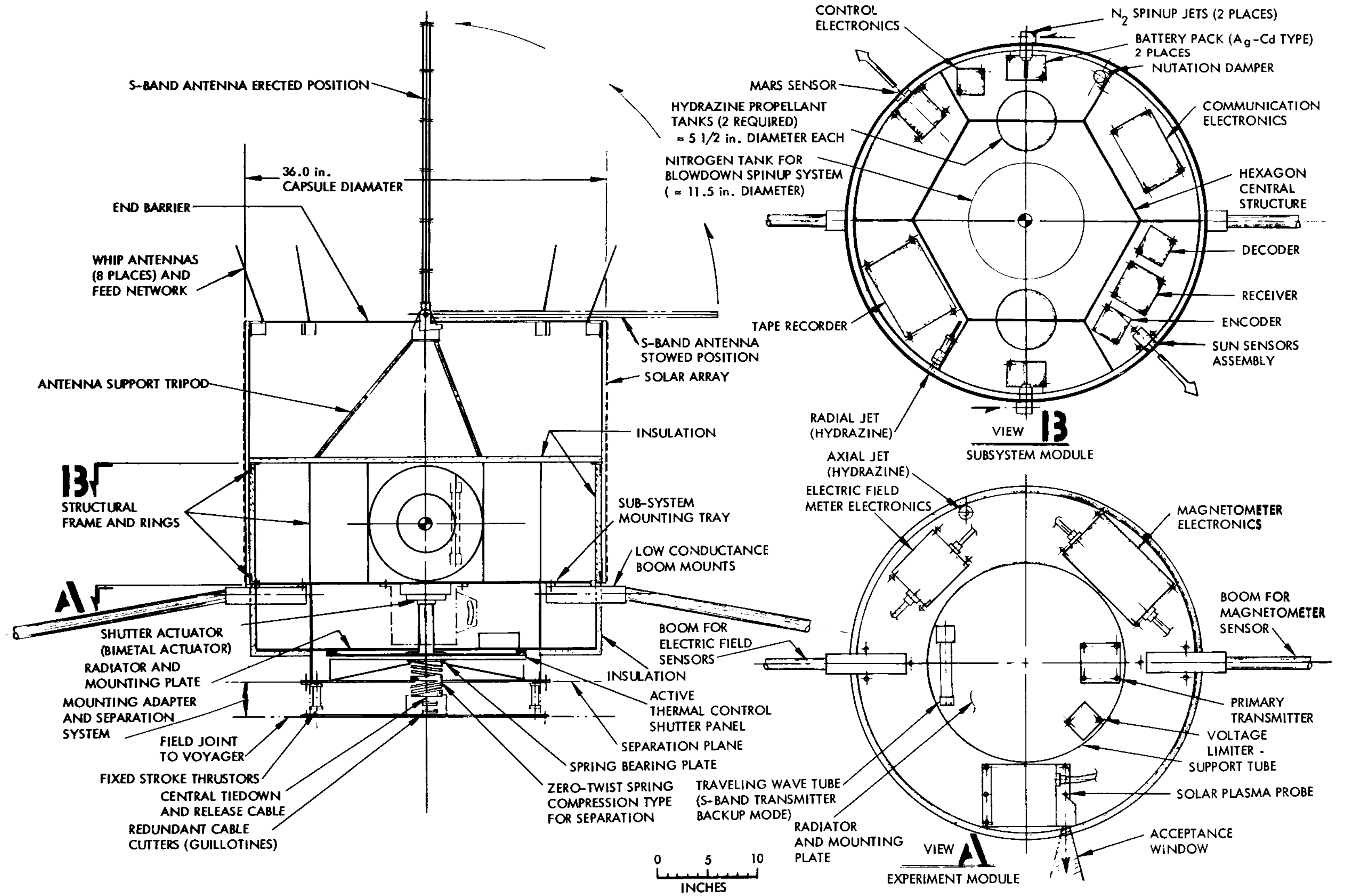


Figure 5-1. OEC Recommended Configuration General Arrangement

5) Mounting and Separation Concepts - Another aspect of configuration selection was the study of feasible and simple mounting and separation concepts that could be adapted to or easily integrated with the basic design approach adopted for the OEC configuration.

A summary of the findings of these studies, including a complete description of the baseline configuration selected, is presented here. A detailed discussion of the tradeoff studies is presented in Section 4.0 of Volume II.

OEC system requirements stipulated by the Ames Research Center at the beginning of the study were a major consideration in the design approaches considered for the OEC. The system requirements that significantly affect the OEC configuration development are:

1) The OEC is to be carried aboard the 1973 Voyager spacecraft bus in transit to the planet Mars. In orbit about the planet, the capsule is to be separated from the bus for its normal orbital mission operation and scientific data gathering.

2) The lifetime of the OEC is stipulated to be 12 months while attached to Voyager and 6 months in operation in orbit about Mars.

3) The capsule is required to be a spin stabilized vehicle at 60 rpm ±10 rpm.

4) The capsule's magnetic field is limited to 0.25 gamma in the vicinity of the magnetometer sensor.

5) The nominal scientific payload is specified to consist of a magnetometer, a solar plasma probe, and an electric field meter. The total weight of the science equipment is stipulated at 15 pounds and would demand a constant power source of 10.0 watts.

6) The design goal for the capsule system is established at 75 to 125 pounds.

7) The capsule's spin axis is required to be oriented normal to the ecliptic plane.

5.1.2 OEC Recommended Configuration

As a result of this feasibility study, a conceptual design of the OEC has evolved and is depicted by the external profile in Figure 2-2; the recommended general arrangement is shown here (Figure 5-1).

The capsule is a cylindrical drum-shaped configuration, the major upper portion consisting of the solar cell array. Below the solar array extends an additional cylindrical length to accommodate placement of the experiment packages. Two radially oriented booms are mounted diametrically opposed off the lower experiment module section. A magnetometer sensor and an electric field antenna are mounted at the ends of the booms. The booms are slightly non-perpendicular to the cylinder axis and drooped away from the solar array to eliminate shadowing on portions of the solar cells. At the upper end of the solar array are mounted eight whip antennas serving the primary communication system; centrally mounted at this end, the S-band system communication antenna is deployed 90 degrees from a folded position to be aligned with the cylindrical axis, or spin axis, of the OEC. Since only a thin non-structural thermal end barrier covers the upper end of the capsule, a lightweight tripod centrally supports the S-band antenna mount. The tripod is supported off the central hexagonal primary structure.

Within the envelope enclosed by the thin wall solar cell array, the capsule's supporting subsystems are housed. Attitude determination sensors within the capsule demand a clear view through the array, and consequently two small windows are provided, one each for the sun sensor assembly and the Mars planet sensor. The propulsion systems chosen to be part of the OEC baseline design require that in the plane of the capsule's center of mass, two spinup jets be mounted tangent to the cylinder to provide the spinup torque impulse; in addition, for the orbit change capability, a radially oriented jet valve is mounted to thrust normal to the spin axis. To provide attitude and orbit change capability, a single axial jet is located just inboard of the capsule's cylindrical envelope with the thrust vector parallel to the spin axis, although offset from it, to provide a torque capability about the transverse axis.

Protruding slightly below the experiment module section is a smaller cylindrical structural tube and attachment ring which attaches to the separation mechanism and interface adapter mounted to the Voyager bus structure.

For the baseline scientific payload specified, the solar plasma probe requires a clear view to space, with the requirement that the instrument's acceptance window scan the plasma primarily in the ecliptic plane. To satisfy this requirement, a single window is provided in the experiment module's enclosure for a view of the plasma environment normal to the capsule's spin axis.

The design approach pursued was, wherever possible, to maintain a minimum experiment-subsystem interface in terms of packaging; consequently, the configuration selected provides for experiments and associated electronics to occupy the lower outboard edge portion of the capsule. Above this section, the necessary mission supporting subsystems are arranged to occupy equipment bays around a central hexagonal structure. The central portion of this section houses the propellant tanks for the nitrogen blowdown spinup system and dual tanks for the hydrazine system provided for attitude correction and orbit change capability. The propellant tanks are commonly mounted in the transverse plane of the capsule's cg.

To effectively maintain the desired temperature range within the capsule, thermal control studies (Volume II, Section 6.4) indicated the need for an active thermal control system composed of a bi-metal actuated shutter device. The active thermal control system is centrally located at the base of the experiment module where a non-solar-illuminated radiation corridor to free space is available. The mounting plate covered by the shutter panel serves as an ideal location to mount the high heat dissipating components; therefore the transmitters, both for the primary and backup communication mode, and the power subsystem voltage limiter have been located on this surface.

The components which are the primary magnetic field contributors in the capsule such as the tape recorder and the traveling-wave tube amplifier have been purposely located as remotely as possible from the magnetometer sensor side of the capsule.

In order to most benefit the spin axis moment of inertia, the subsystem and experiment packages are mounted as far outboard on the tray periphery as possible. To reduce the transverse moment of inertia along an axis perpendicular to the booms, the subsystem packages and experiments are mounted on a common tray at the module interface. Heavy components such as the batteries, solar plasma probe, and propellant storage tankage are mounted along the transverse axis of maximum inertia, i.e., axis normal to radial booms, so as not to substantially increase the transverse moment of inertia already magnified by virtue of the radial booms and tip mounted sensors.

All surfaces enclosing the subsystems and experiments are insulated from their surroundings with the exception of the active thermal control shutter area, so as to minimize the heat losses from the capsule to free space.

The OEC is mounted to Voyager by a short adapter section attached to the Voyager structure at a field joint. The OEC is mounted to this adapter at the separation joint using multiple pyrotechnic release attachments. A central compression spring of the zero twist type is attached centrally in the adapter and imparts separation impulse to the OEC. Briefly, separation is accomplished as follows. The electrical umbilical between the OEC and Voyager is disconnected, explosive bolts release the separation joint followed by fixed stroke pinpushers (thrusters) that gap the separation plane contact faces. A central cable holds the OEC against the end of the separation spring. Redundant guillotines sever the central tiedown cable and allow the spring to push the OEC away from Voyager with minimum tipoff disturbances.

An itemized weight breakdown of the recommended OEC configuration is presented in Table 5-1. From the breakdown, the minimum system weight is shown to be 75.2 pounds for a pure co-orbital mission capsule without attitude correction capability. An increase in system weight by 3.5 pounds provides attitude correction capability to the OEC, bringing the total system weight to 78.7 pounds. To provide the capability for orbit change maneuver with the inherent increase in system weight due to increased data handling and power subsystem requirements brings the OEC weight to 110.3 pounds. Adding to this weight the additional complexity of an S-band backup communication system yields a total baseline OEC system weight of 122.8 pounds.

TABLE 5-1. OEC WEIGHT STATEMENT

			Weight, pounds
• Scientific Payload			15.0
Magnetometer and electronics			
Electric field meter and electronics			
Solar plasma probe			
• Communication System			9.7
Encoder (2)	4.0		
Transmitter	2.0		
Decoder (2)	2.0		
Receiver	0.6		
Antennas	0.8		
Antenna network	0.3		
• Power Subsystem			15.1
Solar cell array		9.2	
Batteries		4.5	
Electronics		1.4	
Limiter	0.3		
Charge boost	0.3		
Controller	0.3		
Discharge circuit	0.5		
• Attitude Determination and Controls			8.4
N_2 propulsion (spinup only)		4.2	
Propellant	0.5		
Tank	1.2		
Fixed weights	2.5		
Sun sensor		0.3	
Mars sensor		2.0	
Nutation damper		0.4	
Sequencer		0.5	
Logic electronics		1.0	
• Electrical Harnesses			2.5
• Structure			15.0
Fixed radial booms (2)		3.0	
Central frame		4.5	
Rings and bulkheads		4.5	
Brackets and attachments		3.0	

Table 5-1 (continued)

			Weight, pounds
• Thermal Control			4.5
Insulation		2.0	
Thermal coatings		0.5	
Radiator panels and actuator		1.3	
Heater		0.7	
• Support Adapter and Separation System			5.0
Structure	2.5		
Pyrotechnics	0.2		
Spring mechanism	2.3		
Minimum OEC system total weight (for co-orbital mission without attitude correction capability)			75.2

Minimum OEC system total weight is increased as itemized below for the following additional capabilities:

1) Initial Attitude Correction Capability (Spin Axis Orientation)			
Hydrazine attitude control system			3.5
Fixed weights		2.6	
Tankage		0.5	
Propellant (offloaded)		0.4	
Additional weight			3.5
OEC system weight			78.7
2) Orbit Change Capability and Additional Communication Requirements			
Hydrazine attitude control and orbit-change-system (includes capability 1 above)			8.0
Fixed weights		3.6	
Tankage		0.5	
Propellant		3.9	
(3.5 pounds for 200 fps ΔV + 0.4 pound for 90 degree re-orientation)			
Tape recorder			7.0
Logic electronics			1.0
Power subsystem			18.1
Solar array		4.6	
Batteries (Ag-Cd)		13.5	
Attachment bracketry and harness			1.0
Additional weight			35.1
OEC system weight			110.3

Table 5-1 (continued)

		Additional Weight, pounds
3) S-Band Backup Communication Mode		
Electronic components		12.5
TWT amplifier	0.95	
Electronic conversion unit	2.5	
Drive chain	2.15	
Receiver and transponder	3.9	
Diplexer	0.5	
Decoder selection circuits	0.3	
Harnesses	0.2	
Structure	0.7	
Antenna	1.3	
Additional weight		12.5

5.1.3 Advantages of Recommended Configuration

Some of the major advantages provided by the configuration selected are as follows:

- The modular arrangement approach lends flexibility to the choice of instrument payloads since they are physically isolated from the major capsule subsystems.
- Radial booms mounted from the experiment module eliminate any potential shadow problems on the solar cells.
- Separation schemes are flexible without significant modification to the capsule design.
- The solar array can be a single unit design and need not be fabricated in sub-units.
- The components with high magnetic properties can be located remotely from the magnetometer sensor.
- The propulsion systems tankage can readily be mounted in the plane of the capsule's cg.
- Exhaust impingement by the propulsion system jets is avoided since they are directed and located away from the experiments.
- Minimum interface between the experiments and subsystems is required.
- A straightforward spring energy separation scheme is feasible.
- The choice of radial booms allows additional growth capability should additional separation distance of the magnetometer sensor be necessary, stowage envelope permitting.

5.1.4 Summary of Configuration Tradeoff Studies

5.1.4.1 Voyager Spacecraft Design Consideration

During the feasibility study the primary candidate designs of each of the three present contractors for the Voyager spacecraft bus (GE, TRW, and Boeing) were studied to ascertain the interface implications of mounting the OEC. Assessment of all three designs indicate that the capsule mounting should be restricted to the peripheral volume external to the spacecraft between the Voyager solar array plane at the base and the Lander interface at the forward end. In terms of stowage volume availability, it appears that the OEC recommended configurations could be accommodated by all three designs: the GE concept offers the maximum space, while the Boeing design imposes the tightest envelope for stowage of the OEC due to their proposed solar array stowage and deployment concept.

Figure 2-5 (Section 2.0) depicts the overall Voyager envelope indicating the areas in which stowage of the OEC might be considered. Figures 3-24, 3-25, and 3-26 (Section 3.0), show the GE, Boeing, and TRW Voyager designs, with the space interference locations indicated on each.

5.1.4.2 OEC Geometric Relationships

In evolving the recommended configuration, an assessment had to be made of the geometric relationship between the solar array size and basic volume requirements of the experiments and subsystems. In essence, the solar cell array defines the basic capsule envelope. Once this envelope was defined, the internal arrangement of the capsule could be formulated and developed. Alternative OEC configuration arrangements investigated are summarized in Figure 5-2. Both integrated (experiments and subsystems share envelope) and modular (compartmentized) configurations were investigated. The modular fixed height design (B on the figure) is recognized as the baseline configuration; it is discussed in more detail in the configuration studies in Volume II, Section 4.0.

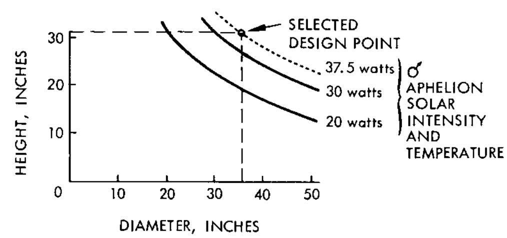
5.1.4.3 Boom Configurations

Three approaches to mounting booms on the capsule were considered in this feasibility study. The alternatives considered are depicted in Figure 5-3. The recommended configuration incorporates the two radially oriented drooped booms utilizing the fixed (non-deployable) type design. Of the four types of boom designs listed on the figure, the best choice in terms of minimum weight, accuracy of sensor position and attitude, reliability, and design simplicity is afforded using the fixed boom concept.

5.1.4.4 Effects of Booms on Capsule Moments of Inertia

For spin stabilization of the OEC to be effective, the moment of inertia of the capsule about the spin axis must be greater than the largest value of transverse inertia. Based on experience (Volume II, Section 4.4) with many spin-stabilized

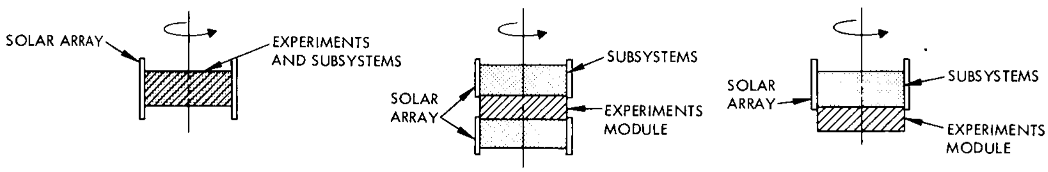
- SOLAR ARRAY REQUIRED SURFACE AREA DEFINES BASIC ENVELOPE
- EXPERIMENTS AND SUBSYSTEMS REQUIRE PACKAGING VOLUME OF 4 cubic feet (MINIMUM)
- POSSIBLE PACKAGING ALTERNATIVES



INTEGRATED PANCAKE DESIGN

MODULAR FIXED HEIGHT DESIGN A

MODULAR FIXED HEIGHT DESIGN B



FEATURES:
- HIGH PACKAGING DENSITY
- OPTIMIZES ROLL INERTIA
- STRUCTURAL WEIGHT PENALTY
- DESIGN FREEDOM IN ARRANGEMENT
- INTERFACE IMPLICATIONS
- GOOD FOR AXIAL BOOM
- RADIAL BOOM SHADOW PROBLEM

FEATURES:
- STACKING IMPLIES MINIMUM HEIGHT (≈ 20 in.)
- EXPERIMENT FLEXIBILITY
- SUBSYSTEMS NOT IN cg PLANE
- SOLAR ARRAY IN TWO SECTIONS
- RADIAL BOOM SHADOW PROBLEM
- MINIMUM LENGTH AXIAL BOOM
- DUAL INTERFACE WITH EXPERIMENTS

FEATURES:
- SINGLE SOLAR ARRAY
- SUBSYSTEMS AT cg PLANE
- ELIMINATES RADIAL BOOM SHADOW
- SINGLE INTERFACE WITH EXPERIMENTS
- MAGNETIC COMPONENTS FARTHEST FROM SENSOR

Figure 5-2. OEC Configuration Alternatives

AXIAL BOOM | RADIAL BOOM | RADIAL DROOPED BOOM

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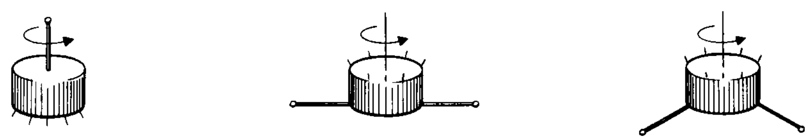


FEASIBLE BOOM DESIGN CONCEPTS

TYPE	WEIGHT (POUNDS)	SENSOR POSITION ACCURACY	DESIGN COMPLEXITY
FIXED	3.0	EXCELLENT (< 1 deg)	SIMPLE
HINGED	6.0	GOOD	REQUIRES RELEASE ACTUATION AND LOCKING
S.T.E.M. (DEPLOYABLE)	9.0	FAIR (TORSIONALLY POOR 3 deg MINIMUM)	SPACE QUALIFIED BUT MORE COMPLEX
TELESCOPIC	6.0	GOOD	SIMILAR TO HINGED TYPE

Figure 5-3. OEC Boom Configuration Alternatives

satellites, a ratio of 1.10 for roll to pitch inertia has proved to be desirable, although lesser values can be tolerated if necessary. Figure 5-4 shows a family of curves for boom length as a function of capsule cylindrical diameter for different design approaches, i.e., axial booms and radial boom considering both integrated and modular capsule design approaches. It is important to note that for the axial boom concept a definite size penalty in terms of required capsule diameter results for small increases in boom length. An axial boom in the order of 7 feet requires a capsule diameter in excess of 4 feet, which would impose a fairly severe stowage volume problem on the Voyager spacecraft. A prime consideration in the selection of radial booms for the OEC baseline configuration is their growth potential, with longer or heavier booms having little effect on capsule configuration design.

5.1.4.5 Mounting and Separation Concepts

As restricted by the Voyager spacecraft stowage volume constraints, the possible OEC positions and separation directions are shown on Figure 5-5. For the baseline, the separation direction is taken as collinear with the capsule spin axis, where the capsule is mounted with its spin axis normal to Voyager. A simple spring separation concept as depicted in the right-hand sketch of Figure 5-6 is proposed. Due to the extremely low forces used for separation, this energy with a spring stressed to extremely low levels should offer no problems in spite of the long stowage time (12 months) considered in this study. The concept selected may also be modified slightly as indicated by the second sketch on the figure. In this manner, the OEC is stabilized at the start of the separation phase by imparting a small partial spin rate with the separating impulse.

Figure 3-27 (Section 3.0) depicts the separation modes investigated during the study. Of importance in considering the candidate separation modes is the clearance corridor that must be available on the Voyager spacecraft to ensure collision-free separation of the OEC. The axial boom approach requires the minimum corridor with or without early spinup. For the radial boom cases, the delayed spinup concept has been chosen to ensure that clearances are available at the extreme tips of the radial booms. Two hinged radial booms, stowed alongside the capsule, would permit the use of radial booms with early spinup and a minimum clearance corridor, at the cost of complexity.

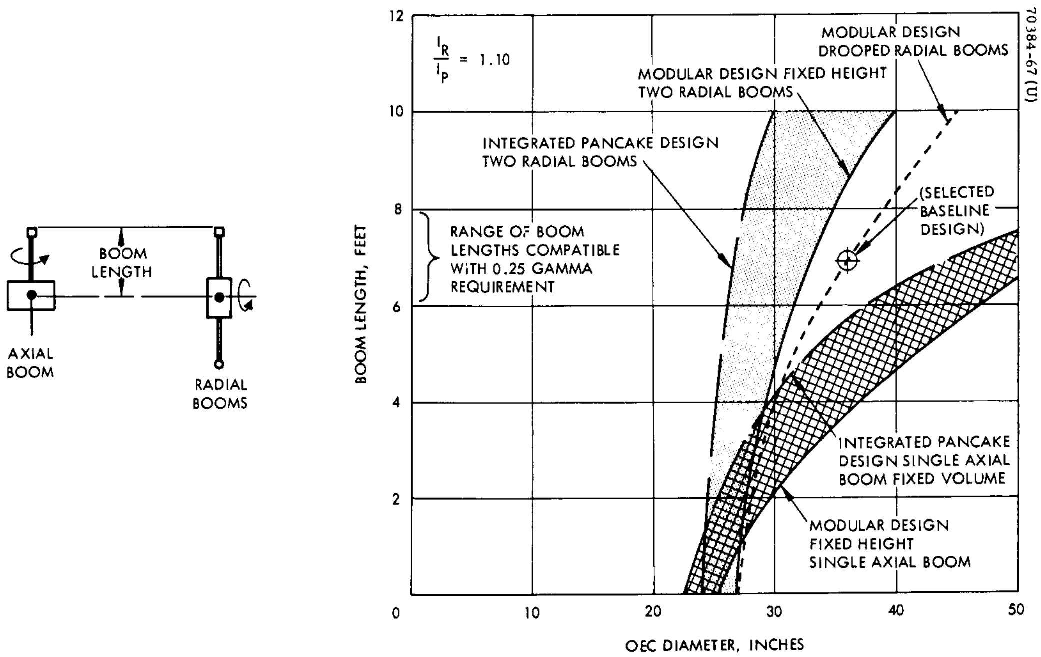
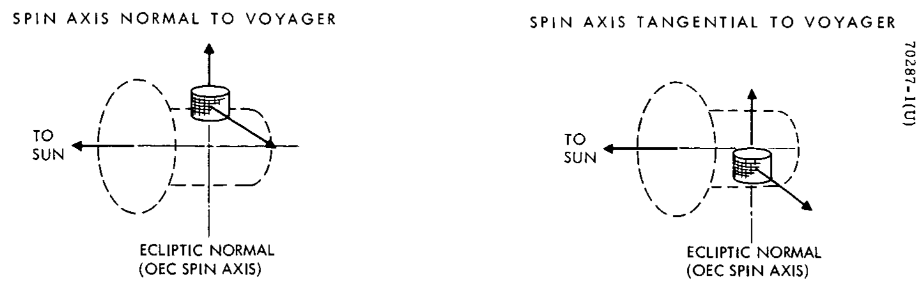


Figure 5-4. Boom Implications on Inertia Ratios



FEASIBLE SEPARATION CONCEPTS

TYPES	SEQUENCE	REMARKS
• DELAYED SPINUP	RELEASE/EJECT	DESIGN SIMPLICITY/ATTITUDE ERRORS BUILD UP
• PARTIAL SPINUP	RELEASE/EJECT AND SPIN	DESIGN SIMPLICITY/REDUCED ATTITUDE ERRORS
• FULL SPINUP	SPIN/RELEASE/EJECT	INCREASED DESIGN COMPLEXITY/CLEARANCE PROBLEMS/ NEGLIGIBLE ATTITUDE ERRORS

Figure 5-5. Spectrum of Separation Modes

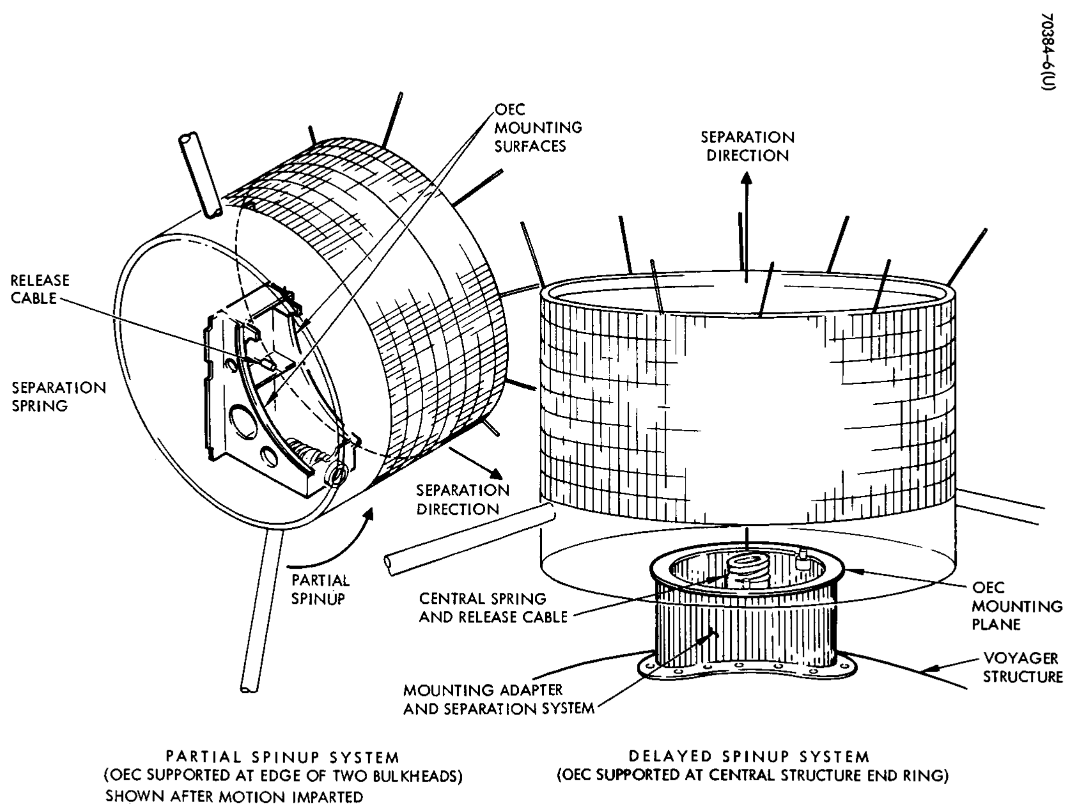


Figure 5-6. Conceptual Arrangement of Mounting Adapter and Separation System

5.2 POWER SUBSYSTEM

5.2.1 General

The OEC power subsystem requirements encompass a spectrum of profiles necessitated by the presently specified envelope of Voyager missions (orbits) and contemplated OEC missions. Of significant impact on the study guidelines are the extremely low temperature environments the solar array will be exposed to in transit and during possible eclipse periods and the minimization of magnetic contamination of the capsule affecting the type of battery chosen for the mission. In addition, OEC orbit change capabilities impose peak power demands on the system which reflect weight penalties to the overall system. Discussed here are the evolved general requirements and a broad description of the contemplated system components. Presented also are parametric data resulting in system definition for various profiles of power requirements. Key areas that require further development and testing are low temperature solar panel survival and silver-cadmium battery technology improvement. The growth implications of the power subsystem can be readily evaluated by the parametric data developed during this study and presented in detail in Section 6.1 of Volume II.

5.2.2 Requirements

The OEC missions considered during this study encompassed a wide variety of possibilities that directly affected the power subsystem performance requirements. At one end of the spectrum is the simplest of co-orbital missions where the scientific data is continuously being acquired and transmitted to the Voyager spacecraft for all phases of each orbit including eclipse periods. In such a mode of operation, the power demands range between 16 and 19 watts of constant power. For the orbit change mission, the possibility of occultation by Mars of the OEC and Voyager necessitates storing the data acquired by the OEC for transmission during those phases of the orbits when communication visibility is not interrupted. The power required on a constant basis for the scientific equipment and scientific bus equipment is in the order of 19 watts, with peak demands of an additional 55 watts to operate the transmitter during the phases that data can be transmitted to Voyager. Table 5-2 shows a breakdown of the OEC average power requirements considered in this study.

The range of possible Voyager orbits and constraints on the Voyager mission involving eclipse durations also play a major role in the selection of the power subsystem performance parameters. For this study the specified range of Voyager orbits of 500 to 1500 km periapsis and 10,000 to 20,000 km apoapsis yields orbital periods ranging from 7 to 15 hours for the minimum and maximum orbits respectively. For eclipse periods, the constraints are that during the first month of the in-orbit operation no eclipsing can occur for the Voyager, although the possibility exists that the OEC may not be separated from Voyager during this period, depending on completion of the Voyager Lander capsule operations. Voyager specifications stipulate that for the next 5 months the eclipse periods not exceed 1 hour or 8 percent of the orbital period, whichever is less. If OEC operation were to be initiated upon the termination of Voyager's first month of operation in orbit, then its sixth month of operation would fall into a period where eclipsing could exceed the above constraints.

TABLE 5-2. OEC POWER REQUIREMENTS

Item	Average Constant Power, Watts
Experiments	10.0
Tape recorder	3.0
Encoder	2.0
Phase lock loop	0.5
Receiver and decoder	3.5
Total	19.0 plus transmitter*

*Additional power of 55 watts required periodically for high power transmitter for orbit-change system, OR additional power of 57 watts required periodically for S-band transmission

Studies of eclipse periods for the extreme orbits specified were conducted and are discussed in detail in Section 2.2 of Volume II. The findings indicate that for the 7-hour orbit the eclipse period can rise to 1.5 hours and for the 15-hour orbit the eclipsing can rise to 2.65 hours during this last month of operation. Clearly, if the OEC mission were to be initiated almost simultaneous with the initial Voyager in-orbit operation, the OEC would be subjected only to the eclipse durations imposed on the Voyager and never to the maximum eclipse time posed above.

Realizing the large eccentricity in the Martian heliocentric orbit, during the 6 months operation, the OEC will not encounter the extreme environments typical of both Mars perihelion and aphelion. Arrival dates specified for the Voyager mission were investigated and found to be in the proximity of aphelion where solar intensities are minimum for the cycle.

5.2.3 Performance

The power requirements for the OEC orbit change system have been utilized in development of parametric data for the power subsystem. In this operational mode, average power requirements are a constant 19 watts during both sunlight and eclipse periods. Transmitter power has been assumed as 55 watts (Volume II, Section 6.1) for power system parametric calculations.

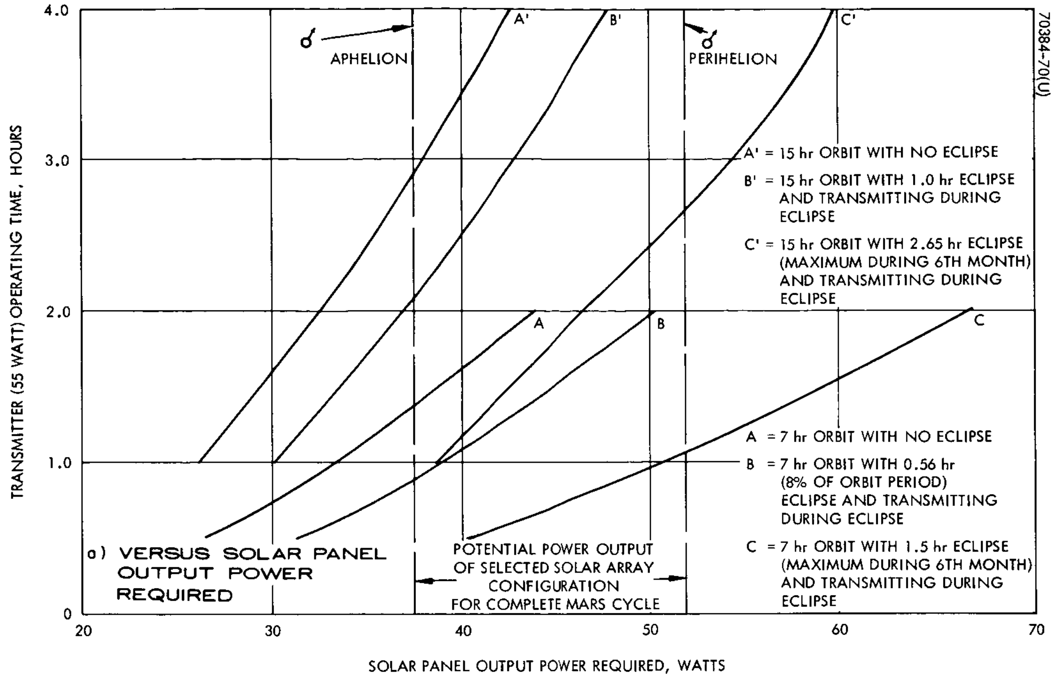
The broad spectrum of missions considered for the OEC during this feasibility study necessitated that the data presented cover the range of power subsystem performance parameters for the possible boundary conditions. Based on the total weight limitation imposed on the OEC, a weight allocation for the power subsystem was placed on the solar array and batteries considered in the recommended configuration. The solar array size that could be accommodated based on studies of stowage space availability was also a factor in selection of a power subsystem. For the recommended configuration, a weight budget of 32 pounds was established for the power subsystem. The 36-inch-diameter by 31.5-inch-long cylindrical array can provide between 37.5 and 52 watts at Mars aphelion and perihelion respectively and would weigh 13.8 pounds, allowing approximately 18 pounds for batteries aboard the OEC.

For the orbit change mission, Figure 5-7a shows a family of curves for both the 7-hour and 15-hour orbits for various eclipse times as related to transmitter operating time and solar panel output power required. The selected solar array power output for the baseline configuration is also indicated for the Mars orbit extremes. Figure 5-7b shows the same family of curves plotted to indicate transmitter operating time versus battery energy requirements. Figure 5-8 relates energy required by the transmitter to range between the OEC and Voyager for the specific data rates considered. Range versus time for the two extremes of orbits considered here is shown in Figure 4-4 (Section 4.0).

Evaluation of the data presented in the figures establishes that performance boundaries certainly exist, but to design the power system for these conditions would impose severe penalties in terms of size and weight and therefore does not seem a rational approach.

Examining the data shows that if the power system were to be capable of transmitting at the full data rate of 630 bits/sec for maximum possible ranges (≈17,000 km) in the 7-hour orbit which requires 110 watt-hours (Figure 5-8), 2 hours of transmission time would be required. Considering also the worst case eclipsing of 1.5 hours, 2 hours of transmission time would require a solar array capable of providing in excess of 67 watts (Figure 5-7a, curve C). Hence the approach taken in view of the large spectrum of mission possibilities was to select a power system consistent with the weight and envelope constraints for the OEC. During the major portion of the mission (first 5 months), it can be established that for the longest orbit period a minimum of 2.1 hours of transmission time (115 watt-hours) can be realized (Figure 5-7a, curve B') intersection with minimum solar panel power line at Mars aphelion. From Figure 5-8, this energy relates to approximately a maximum range of 17,000 km. Hence for the days that the range between OEC and Voyager exceeds this maximum, degraded data transmission is imposed. The phasing of the orbits causes ranges in excess of the maximum distance to occur approximately 30 percent of the time. During these periods at least half of the data is retrivable or sampling would have to be reduced. The natural Mars cycle, as the mission passes Mars aphelion, will tend to enhance the data transmission capabilities due to increasing power availability and to reduce the degraded transmission phase of 30 percent. Examining the lesser orbit case yields similar conclusions.

The preceding discussion does not take into account the possibility of performing "stationkeeping" in orbit "synchronization" between Voyager and OEC. If this were to be done, the range could be substantially reduced, permitting maximum data transmission at all times during the 6 month mission.



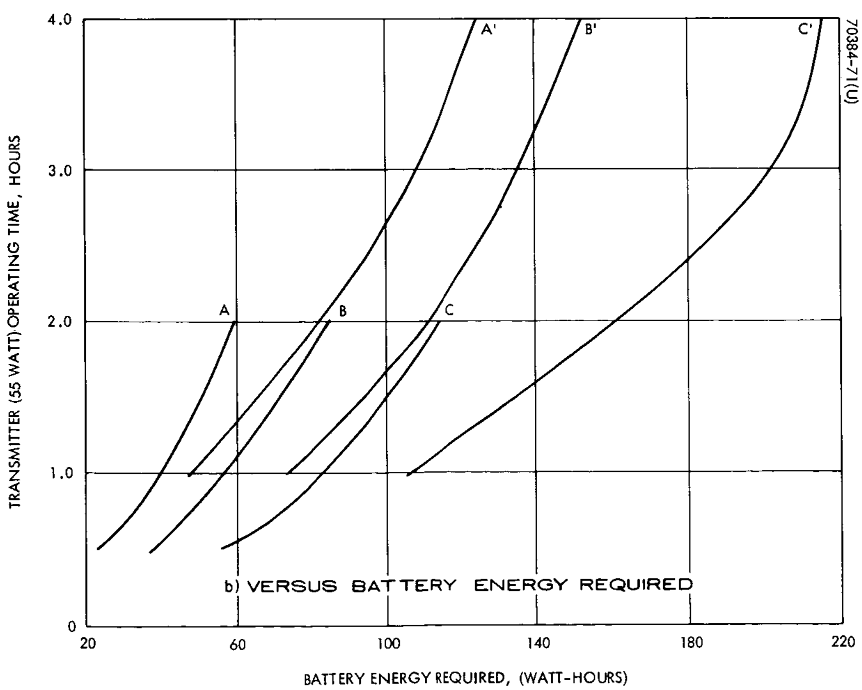


Figure 5-7. Transmitter Operating Time

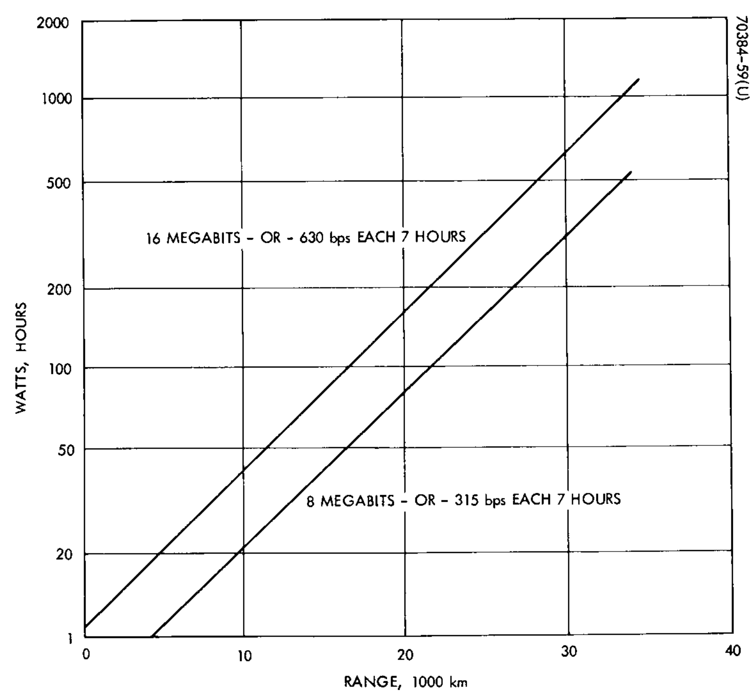


Figure 5-8. Energy Required to Transmit Various Amounts of Data Versus Range

5.2.4 Subsystem Description

All elements considered for the OEC power subsystem are within the state of the art and based on successful space flight experience. Designs considered here utilize the background obtained from flight-proven hardware.

The OEC electrical power subsystem provides all on-board power for each of the using subsystems and experiments. Prime elements of the power subsystem are a cylindrical solar array, two parallel connected silver-cadmium sealed secondary batteries, and two charge-discharge controllers. Power distribution is accomplished through a main bus to the using loads. Busses for single function (one-shot) or pyrotechnic devices can be provided as required and could be connected directly to the batteries.

Primary power is supplied by the solar array during sunlight periods of the orbit for constant loads and for battery recharging. The batteries provide power during eclipse operations and for peak loads exceeding solar panel power output during sunlight operations. The battery charge-discharge controllers provide controlled charge and discharge conditions for each battery. A schematic block diagram is shown in Figure 5-9.

A description of the solar cell array, batteries, and control electronics follows.

5.2.4.1 Solar Panel

Figure 5-10 and Tables 5-3 and 5-4 depict the size, performance, operational life, construction, and environmental capabilities of solar panels developed by Hughes since 1960 for spacecraft applications. In addition to the flat panels developed for Surveyor, the cylindrical panels developed for communication satellites bound both size and power requirements applicable to the OEC mission. The operational and test temperature limits shown in Table 5-3 represent the design requirements and environmental test limits successfully demonstrated in solar panel qualification. The data presented in parentheses indicates panel specimen test levels.

Figure 5-11 presents the fabrication details of Hughes-developed solar panels and illustrates typical coefficients of thermal expansion of materials used in solar panel fabrication. Communication satellites have utilized a fiberglass faced aluminum honeycomb structural member or substrate. Solar cell bonding has been accomplished using a modified epoxy adhesive system, permitting close matching of the expansion coefficient of the composite substrate and silicon solar cells. The fiberglass faced substrate design results in 1) an increased low temperature capability, 2) a light-weight solar panel, and 3) a solar panel capable of serving as a primary spacecraft structural member. As shown in Table 5-3, the fiberglass faced substrate design has been tested on solar panel segments successfully to temperature extremes of -320°F.

The solar panels for the first four Surveyor spacecraft used titanium faced aluminum honeycomb substrate and an RTV adhesive system for solar cell bonding. The ability of the Surveyor solar panel to withstand extreme environmental

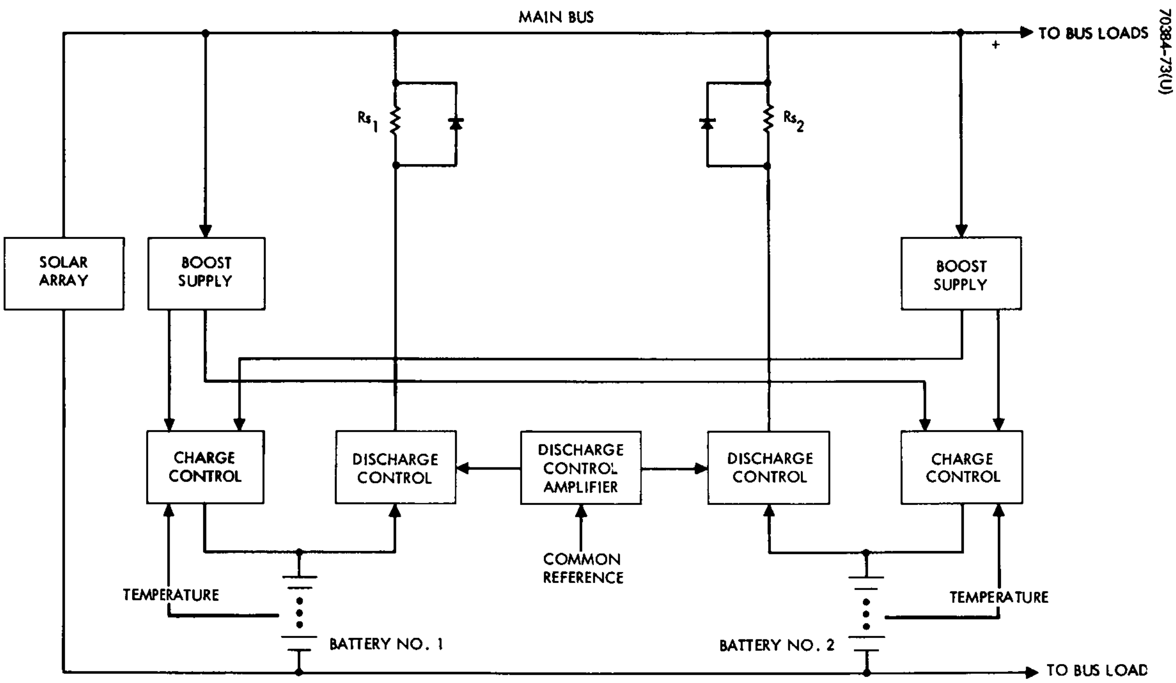


Figure 5-9. Charge-Discharge Control Block Diagram

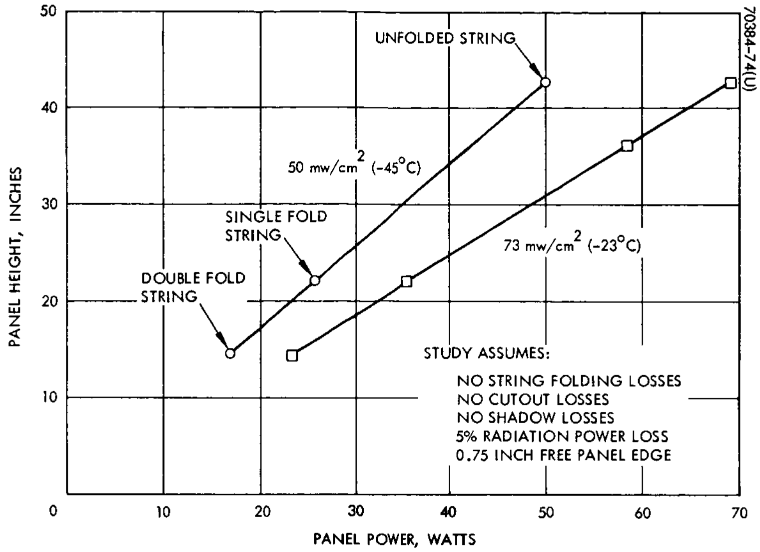


Figure 5-10. Solar Panel Power Output Versus Height

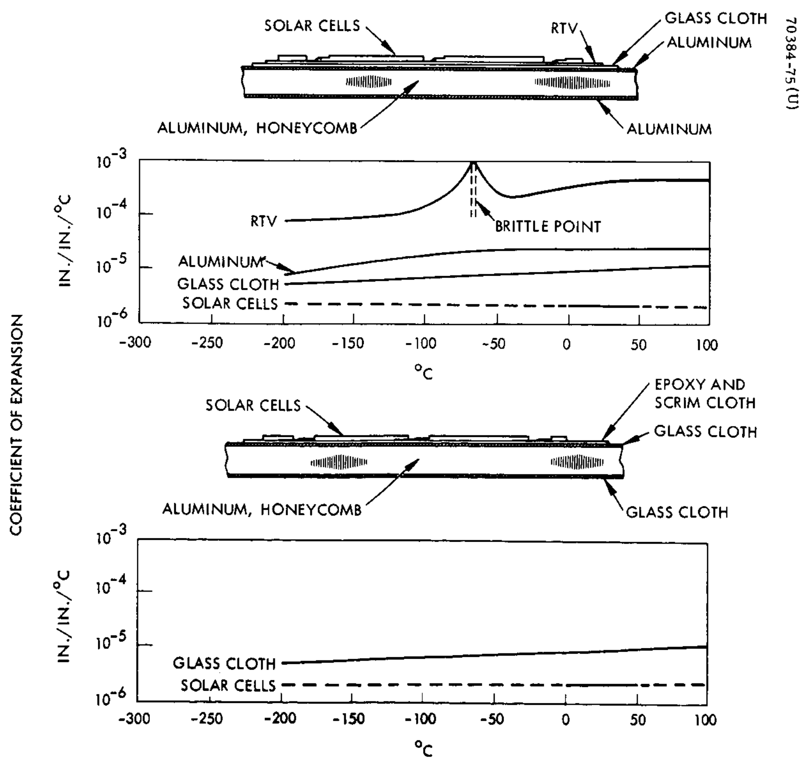


Figure 5-11. Comparison of Solar Panel Fabrication Techniques

TABLE 5-3. SOLAR PANEL PERFORMANCE DATA
(denotes specimen test)

Project	Panel Diameter, inches	Panel Rated Power, watts	Duration of Space Operations as of 7-6-67, days	Temperature Limits, °F Operating	Temperature Limits, °F Test
Surveyor	Flat	101.9	210	-300	-270 (-320)
Syncom 2	28	30.0	1441	-80	-150
Syncom 3	28	33	1051	-80	-320
Early Bird (HS-308)	28	46.5	811	-80	-120 (-320)
Intelsat (HS-303A (3))	56	104	176	-150	-80 (-290)
ATS-A	56	139	91	-200	-200 (-250)
ATS-B	57.7	175	212	-80	(-250)
HS-308	111.3	1050	Design phase	-190	-200 (-320)

TABLE 5-4. TEST DATA ON HUGHES ADHESIVE SYSTEM

Program	Temperature, °F	Number of Temperature Cycles	Remarks
Advanced Syncom	+250 -250	500	Coupon (typical panel segment) 15°F/minute
Advanced Syncom	+250 -320	25	Hot plate to LN_2 immersion
Syncom 3	150	35	Panel (typical quadrant of cylindrical array) 20°F/minute
ATS	+250 -250	5000	Coupon (typical panel segment) 16°F/minute

exposure was vividly demonstrated by the continued performance of SC-1 after several lunar nights. Final communication was lost with the spacecraft after 210 days of successful lunar operation sustained by the solar panel.

A cylindrical solar panel configuration is considered optimum for use on the OEC based on Ames specified design criteria, simplicity, reliability, and operational considerations. Based on the preferred OEC configuration and range of potential power requirements, solar panel performance and weights have been estimated for a 36-inch-diameter cylindrical substrate varying in length from 14 to 36 inches, as shown in Volume II, Section 6.1.3. The estimates are based on actual panels developed for communication satellites, which encompass the size of the OEC solar panel.

Based on the use of passive thermal control techniques, the solar panel will be exposed to temperatures as low as -195°C (-314°F) during transit and operational temperatures ranging from -22° to -157°C (-8° to -250°F) while in orbit. Solar panel systems have been tested within the predicted temperature ranges and have demonstrated their ability to survive repeated cycling to liquid nitrogen temperature. Degradation due to low temperature exposure can be prevented through selection of materials and processes for solar panel fabrication.

The electrical performance of a 36-inch-diameter cylindrical solar panel at Mars perihelion ranges from 27.5 watts to 58.5 watts for solar panel lengths varying from 17 to 36 inches. The electrical performance figures at aphelion for the respective panel lengths decrease in range of 20 to 42.5 watts. The estimated weight, including all attachments and vertical connectors, is 7.4 pounds for the 17-inch panel and 15.7 pounds for the 36-inch panel. These estimates are based on utilization of a fiberglass-faced aluminum honeycomb structure and 2 x 2 cm shallow diffused silicon solar cells with 0.006 inch quartz coverslides. The 2 x 2 cm solar cells are recommended for both economy and increased performance resulting from fewer interconnections compared to 1 x 2 cm cells.

Design practices have been developed for minimizing the effects of RF and magnetic fields and have been successfully applied in space programs. Several thermal control coatings and techniques have been developed and utilized for passive thermal control of solar panels.

The studies performed indicate that a solar panel can be designed and qualified to the mission environmental requirements that will provide successful OEC operation during the illuminated periods of the orbit.

5.2.4.2 Batteries

The difficult task of producing reliable batteries for space flight has been resolved by the use of four principal techniques:

1) Design battery capacity to adequately meet the mission load requirements.

2) Integrate battery design with the battery charge controller design.

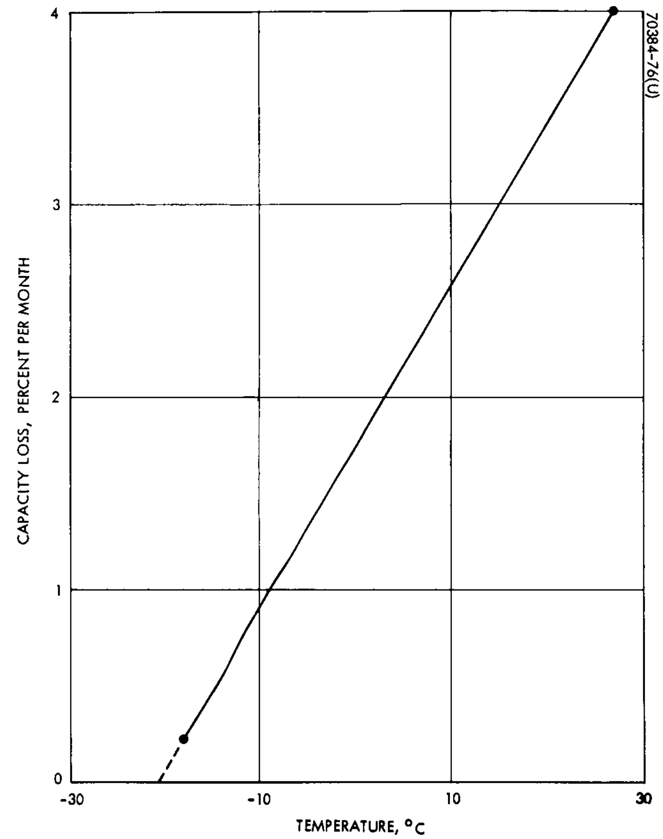


Figure 5-12. Estimated Open Circuit Stand Loss for Silver-Cadmium Batteries

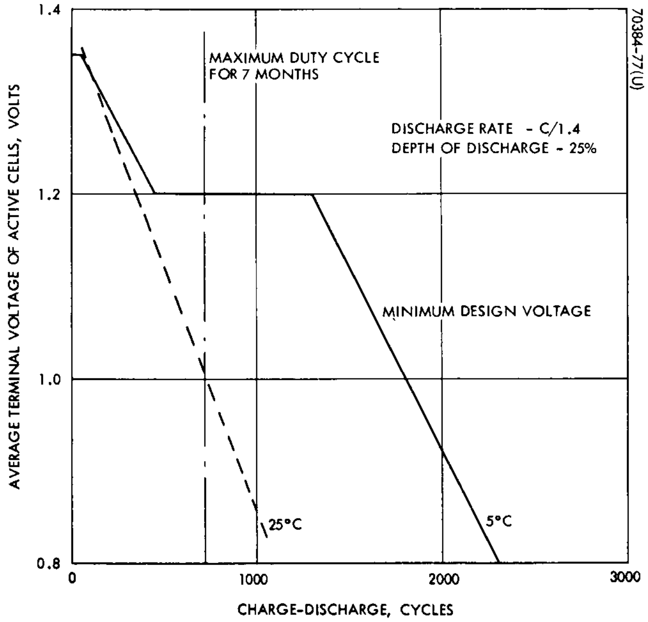


Figure 5-13. Average Silver-Cadmium Cell Voltage Versus Cycles as Function of Temperature

3) Maintain a close working relationship with the cell vendors and place particular emphasis on in-process controls and quality tests.

4) Perform extensive test programs to evaluate the product and determine performance limitations.

Table 5-5 lists the nominal characteristics of batteries commonly used in various space programs.

The silver-cadmium battery has been selected for the OEC mission because of its non-magnetic nature, demonstrated cycle life, good low temperature storage capability, and relatively high reliability. The energy density of silver-cadmium falls between silver-zinc and nickel-cadmium batteries. The energy density of silver-cadmium cells is temperature-dependent and discharge-rate-dependent. Silver-cadmium batteries must be charged in a manner to prevent high overcharge rates. The charge rate tentatively selected for the OEC application is one-tenth the ampere-hour rating of the cell size selected. This is a conservative value and under more precisely defined operational and environmental conditions can be optimized.

Two silver-cadmium batteries of the required capacity will be used in parallel to optimize reliability of the energy storage system. Depth of discharge is limited to 25 percent to assure maximum cycle life and voltage capability over the planned mission operating period.

For the OEC mission it is planned that the batteries be fully charged immediately prior to launch and stored in an open circuit condition at a temperature of $0 \pm 20^{\circ}F$ during the transit portion of the mission. Estimated stand loss of capacity during storage in transit is shown in Figure 5-12 and is less than 0.5 percent per month at $0^{\circ}F$ ($-17^{\circ}C$). For a 12-month transit period, stand loss would be approximately 5 percent of initial battery capacity, assuring capability to begin the orbital mission phase without recharge.

In Figure 5-13 cycle life versus cell voltage as a function of battery operating temperature is shown. It is estimated that the batteries will experience approximately 620 charge-discharge cycles in the 7-hour orbit mission, which is within the demonstrated cycle life capability of silver-cadmium batteries to date.

TABLE 5-5. NOMINAL BATTERY CHARACTERISTICS

Battery Type	Nominal Voltage	Watt-Hours per Pound	Watt-Hours per Cubic Inch	Relative Lifetime, years
Silver-zinc	1.5	30 to 80	4 to 5	1
Silver-cadmium	1.1	15 to 30	2 to 3	2
Nickel-cadmium	1.2	10 to 12	1 to 1.5	3 to 5

Operating temperature during charge-discharge orbital operations is planned to be $60^{\circ} \pm 20^{\circ}F$ ($15^{\circ} \pm 10^{\circ}C$). A temperature sensor will be located in each battery package providing a signal to the charge control circuitry to preclude charging at higher than normal operating temperatures.

5.2.4.3 Charge-Discharge Control

The function of the battery charge-discharge control (Figure 5-9) is to sense the charge condition of the battery and the voltage on the main bus to determine the mode of operation. The Ag-Cd batteries require a current-limited charge operation with an upper voltage limit cutoff.

When the battery is at its lowest capacity, approximately 1.0 volts per cell, the battery receives a maximum charge current through the charge control from the array. As the battery charge and voltage increase, the reduced difference voltage between battery and charger causes saturation of the current limit control followed by a decrease in charge current. When the battery terminal voltage increases to approximately 41.9 volts, the charge control is cut off. With the charge control cut off, the series rectifier in the controller becomes back-biased, thus turning off all charge current into the battery. During an eclipse mode, the battery alone supplies all power to the system. As the battery discharges, the cycle reverses.

Since the battery terminal voltage will decrease after removal of the charge voltage, the charge control circuit will have a sensing voltage dead band. The magnitude of this dead band will be optimized to preclude on/off cycling of the charge control resulting from the normal battery voltage drop to essentially open circuit voltage.

To prevent overcharging the battery at temperature extremes, a thermal sensor is located within the battery package which turns the battery charge control off if the battery temperature reaches $95^{\circ}F$. This condition will remain until the battery temperature decreases to $80^{\circ}F$. In addition, the voltage control circuit in the charge controller will contain a temperature compensating network that will adjust the charge voltage to prevent battery overcharge at elevated temperatures.

5.3 PROPULSION SUBSYSTEMS

5.3.1 General

The studies conducted have demonstrated that the OEC propulsion requirements, established from OEC systems analyses, can be satisfied utilizing a simple cold gas (nitrogen) blowdown spinup system and a monopropellant hydrazine system for attitude maneuvers and orbit changing capabilities. A preliminary definition of each of the two suggested systems is presented here in diagram form, including a weight breakdown of each system. The selection of the specific components comprising each system is a task to be accomplished once a specific OEC mission is defined; i.e., once maneuvers, orbit selection and orbit change requirements become fully established. The recommended OEC propulsion systems, considering the conceptual overall OEC system, do offer potential growth capabilities, the only penalty being slight additional weight to the systems. The propulsion systems recommended are presently within the state of the art, so that no serious problems in development are expected.

The OEC propulsion weight budget (established at approximately 12 pounds) is liberal enough to allow emphasis on simplicity and reliability in system selection. Although some weight saving over the selected systems would be possible, the alternative choices would result in substantially lower reliability without commensurate gain in performance. The preferred systems are described briefly in the following discussion of general requirements, followed by a detailed description of each design.

5.3.2 Requirements

5.3.2.1 Co-orbital Mode

Spinup of the capsule to 60 $\pm$10 rpm is the only propulsion function required. The thrusters are limited to 130 millipounds in order to avoid undesirable translational effects. Studies conducted leading to thrust level selection are presented in the separation studies included in Volume II. The modest 33 lb-sec total impulse requirement allows the simplest possible nitrogen gas blowdown system to be utilized, meeting weight and envelope restrictions.

5.3.2.2 Attitude Correction

The worst possible attitude error would be 90 degrees, since alignment of the spacecraft in either direction along the desired axis is acceptable. Rotation of the spinning spacecraft through a 90 degree angle requires approximately 75 lb-sec. A simple blowdown nitrogen gas system is acceptable but is not the recommended choice since an orbit change capability is seriously contemplated for this or other OEC missions. For this reason, an offloaded hydrazine orbit change system is recommended.

5.3.2.3 Orbit Change

The relatively large velocity increment (ΔV approximately 300 fps) precludes a cold gas system for this function. Of the possible choices, only catalytic monopropellant hydrazine is attractive. A 700 lb-sec system was selected for the design presented and is shown to be well within weight and envelope restrictions. Due to the relatively low weight of the hardware, it appears desirable to design the system with the largest possible propellant tanks and offload for missions requiring lower total impulse.

5.3.3 Subsystem Descriptions

5.3.3.1 Spinup Subsystems

A number of possible candidates were rejected because of obvious reliability or development problems with little or no weight savings in a system of small total impulse. These include:

- Bipropellants
- Monopropellants
- Electrical propulsion
- Subliming solid
- Resistojets

The systems that appeared attractive enough for detailed analysis were:

- Cold gas
- Conventional solid rocket motors
- Vaporjets

A comparison of these systems is shown in Table 5-6. Detailed studies associated with the propulsion selection are presented in Volume II, Section 6.2.

A simple nitrogen cold gas system weighs only about 4 pounds and is the obvious choice in all other categories. Although it might be possible to reduce the weight by choosing a different propellant gas, i.e., one of the freons, the long history and extensive data for nitrogen allows a shorter, less expensive development program.

A diagram of the system is shown in Figure 5-14. Two parallel squib valves control system operation. Pending further reliability analysis, it probably would be possible to eliminate one of the valves since this type of valve contains redundant squibs and has demonstrated extremely reliable operation.

TABLE 5-6. SPINUP SYSTEM COMPARISON

System	Weight, pounds	Reliability	Cost	Schedule, months
Cold gas	4.1	High	Low	6 to 9
Solid	1.0	High	Medium	12 to 18
Vaporjet		Good	High	18 to 24
Electrical	5.6			
Equilibrium	4.2			

Order of Preference:

- Cold gas - Acceptable weight, best in other categories.
- Solid - May be useful emergency backup. Too high thrust, possible plume contamination.
- Vaporjet - No obvious advantages. Technology less developed.

A maximum thrust of 130 millipounds can be obtained with a thruster having a 0.020-inch-diameter throat if the operating pressure is limited to 240 psia, as shown in Figure 5-15. It would be possible to reduce throat diameter and increase chamber pressure but it becomes difficult to control manufacturing tolerances for smaller throats. From Figure 5-16 it is found that tank diameter and weight are acceptable. Titanium is the preferred material, but aluminum would be acceptable. The best choice would be a tank which is already available, if one can be found that satisfies these requirements.

A summary of component and propellant weights is presented as part of Table 5-7.

5.3.3.2 Orbit Change Subsystem

The simplest, most reliable system capable of accomplishing the orbit change mission within the weight allocation is a monopropellant. Hydrogen peroxide systems have been used by Hughes on a number of satellites with good success and meet most requirements of this mission. However, peroxide has a decomposition rate high enough to require venting of gas during the transit from Earth to Mars. The problem of gas venting in zero gravity combined with lower performance for peroxide makes hydrazine the best propellant choice.

A diagram of a hydrazine system suitable for the OEC mission is shown in Figure 5-14. The two thruster configuration is identical to that used on a number of operational satellites and would be operated in the same way. However, it is noted that only one (axial) thruster is mandatory for both attitude control and orbit change. The radial thruster shown might be used to advantage by reducing the

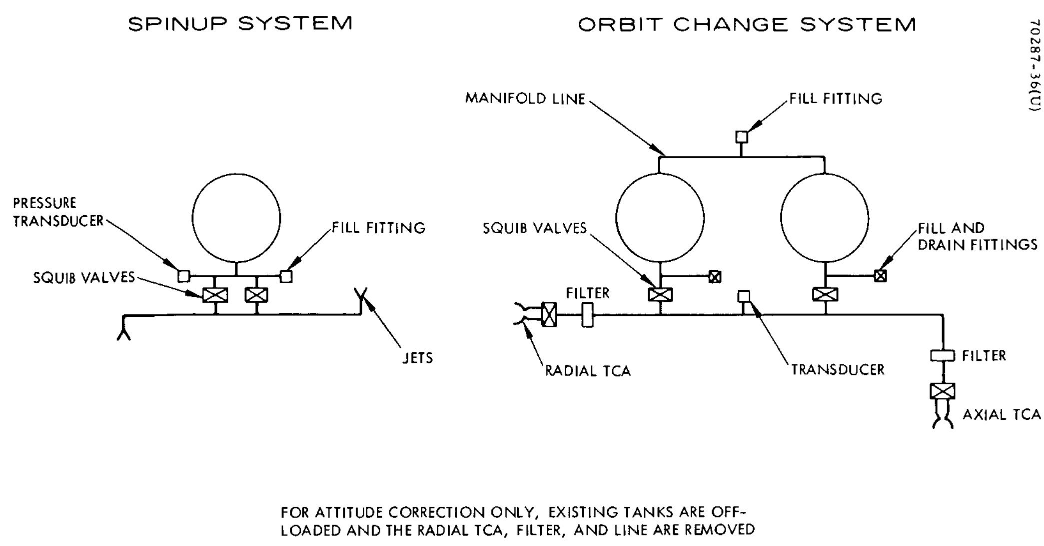


Figure 5-14. Spinup and Orbit Change Systems

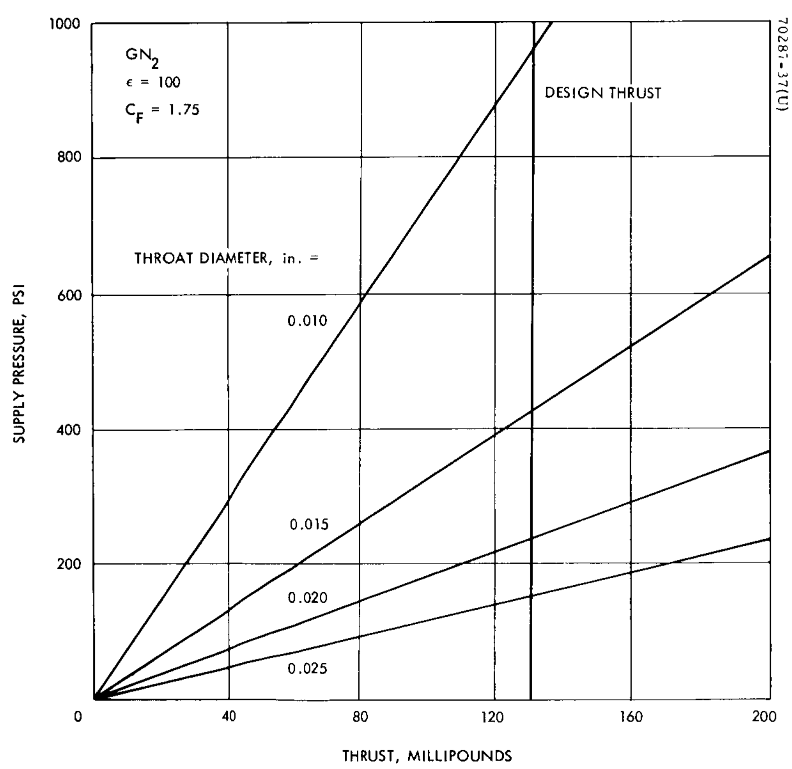


Figure 5-15. Thrust Versus Supply Pressure and Throat Diameter

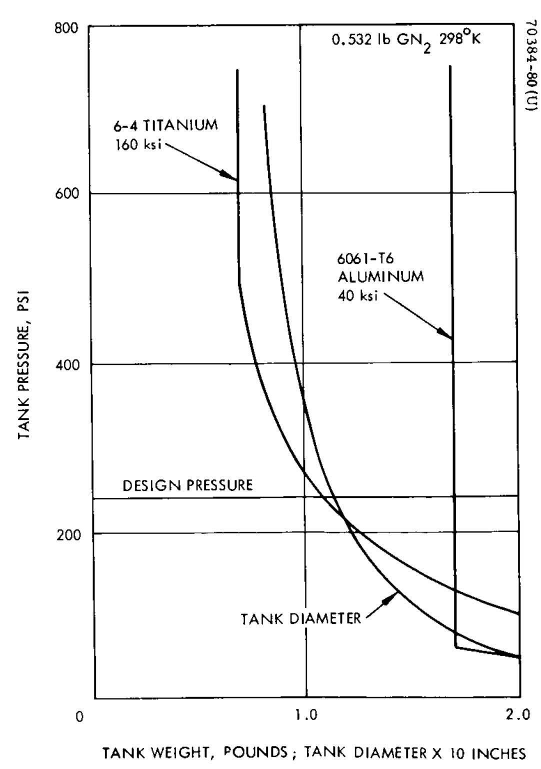


Figure 5-16. Nitrogen Spinup System

TABLE 5-7. SUMMARY OF RECOMMENDED PROPULSION SYSTEM

Item	Nitrogen Spinup		Orbit Change System: Complete		Orbit Change System: Attitude Control Only		Alternate GN_2 Attitude Control	
	Number	Weight, pounds	Number	Weight, pounds	Number	Weight, pounds	Number	Weight, pounds
Squib valves	2	0.6	2	0.6	2	0.6	2	0.6
Solenoid valves			2	1.2				
Quad								
Single					1	0.2	1	0.4
Thruster	2	0.6	2	1.0	1	0.5	Included in valve	
Filter			2	0.2	1	0.1		
Manifolds and brackets		1.0		1.2		0.8		0.5
Instrumentation		0.3		0.3		0.3		0.3
Total fixed weight		2.5		4.5		2.5		1.8
Propellant		0.5		3.0		0.3		1.0
Tanks	1	1.1	2	0.5	2	0.5	1 or 2	1.4
Total system weight		4.1		8.0		3.3		4.2

angle through which the spacecraft must be turned in preparation for orbit change or for periodic orbit synchronization with Voyager; alternatively, it could be replaced by a second axial thruster for redundancy. Two axial thrusters, each equipped with a dual valve, would be roughtly equivalent in reliability and weight to a single thruster with a quad valve. Note that squib valves are used to isolate the propellant within the tanks until just before the system is used. By this scheme, the valves are exposed to propellant for a much shorter time and leakage during transit is precluded. All fill and drain fittings are sealed after propellant and pressurant loading.

Tank weight and diameter are shown in Figure 5-17. The design weight indicated is a minimal impulse case, but it is seen that the capacity of the system can be increased considerably without much increase in tank weight. As previously indicated, it is recommended that an existing tank which would fit within the required envelope be found or a new tank designed for maximum envelope. Total impulse capability is given in Figure 5-18 where it is seen that the design example will deliver between 600 and 700 lb-sec, depending on the amount of pulse operation. Total system weight is shown in Table 5-7.

As previously indicated, the orbit change system could easily incorporate the spinup function by addition of two thrusters. However, this would be considered undesirable for the following reasons:

1) A system that is required to turn off after a specified amount of propellant has been expended is inherently less reliable than a gas blowdown system. Also, the reliability of the primary spinup system should be high enough that there would be no incentive to provide redundancy at the expense of extra weight and some reduction of orbit change reliability.

2) The zero or low gravity condition prior to spinup would require an additional device to guarantee propellant feed at the beginning of operation.

3) There is little commonality of components. All components except the propellant tank would be additional.

5.3.3.3 Attitude Control Subsystem

The orbit change system described is fully capable of all attitude control functions and could be offloaded to provide only this capability. Table 5-7 shows the total weight of such a configuration with only one thruster.

The most attractive alternative for attitude control alone would be the simple nitrogen system also shown in Table 5-7. Tank and system weight is shown over a range of total impulse in Figure 5-19. It is seen that the nitrogen system is compettitive in weight with the offloaded hydrazine system.

If growth to orbit change were not a factor, it would be cheaper and easier to build the nitrogen system. However, since it is anticipated that orbit change will be a requirement for some of the OEC missions, it would not be necessary or desirable to build an independent system for attitude control.

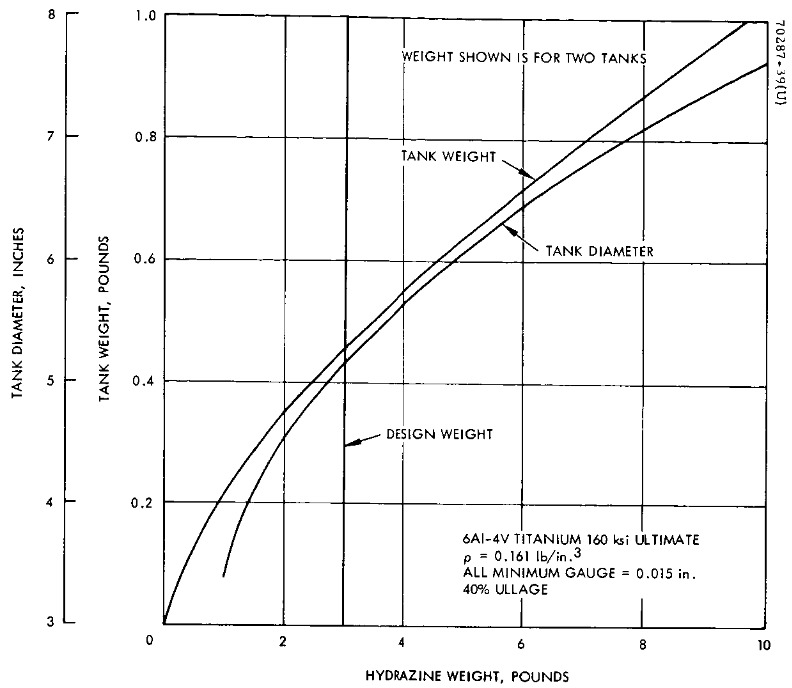


Figure 5-17. Hydrazine Tank Weight and Diameter

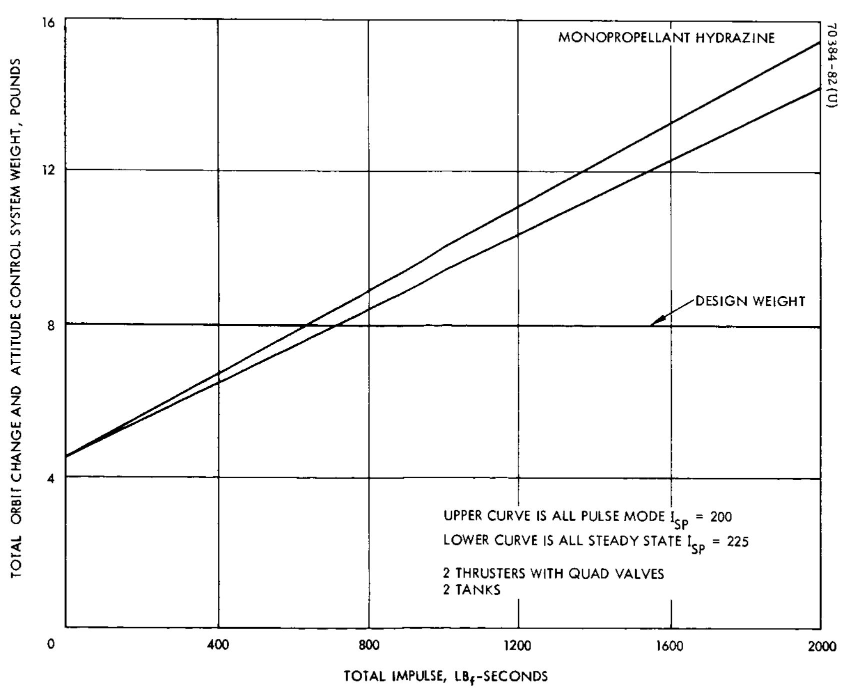


Figure 5-18. OEC Orbit Change and Attitude Control System

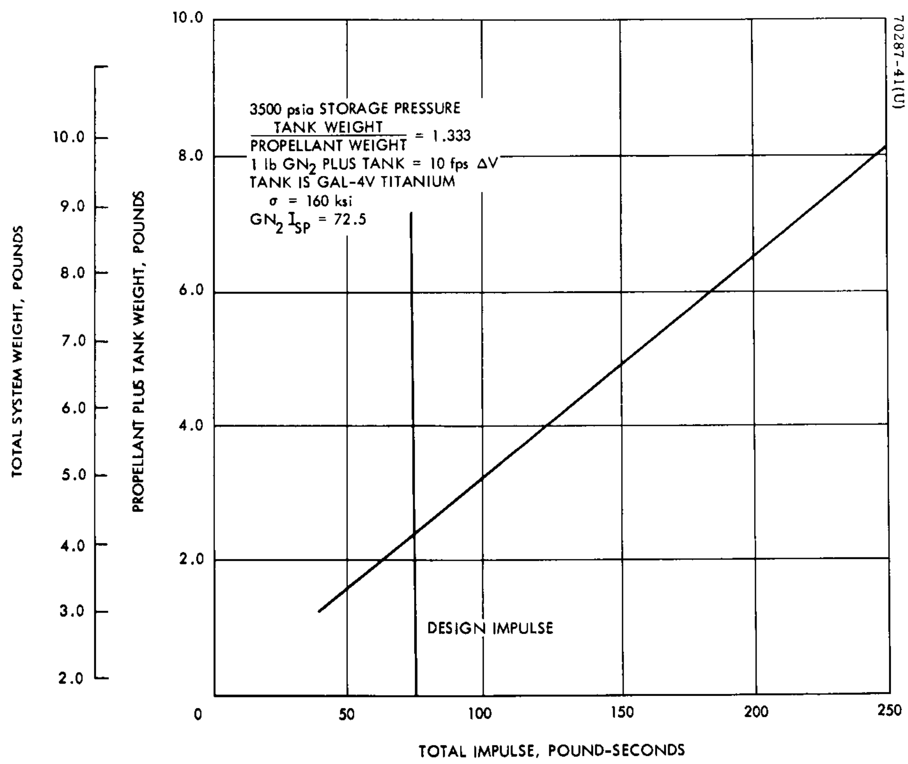


Figure 5-19. Alternate GN_2 Attitude Control System

5.4 STRUCTURE SUBSYSTEM

5.4.1 General

This feasibility study did not include any detailed structural analysis since no definitive static and dynamic environment has yet been established for the OEC. With the advancement of the Voyager spacecraft design and overall system environment definition, a meaningful study of the OEC structure could be implemented. Structural considerations for this feasibility study were limited to establishing a conceptual framework that satisfied equipment and subsystem arrangements, thermal control considerations, and attachment and separation techniques selected. Based on the group of nominal experiments considered and the results of other subsystem tradeoff studies, a structural arrangement was developed that satisfies envelope, power, packaging, thermal control and mounting and separation constraints.

A brief description of the OEC structure is presented in the following paragraphs.

5.4.2 Structure Description

The structural frame for the recommended OEC configuration consists basically of the following major sections:

1) Central hexagonal support structure
2) Central mounting tray bulkhead
3) Base mounting/radiator tray
4) Solar array support rings
5) Support tripod for stacked array antenna (S-band)
6) Base support tube
7) End closure bulkheads/thermal barriers
8) Mounting interface adapter incorporating the separation mechanism and providing the OEC-Voyager mechanical interface.

The booms are discussed in detail in the configuration studies material presented in both Volume II and in this volume in Section 6.1.

The primary supporting structure is the central hexagonal frame to which all members are attached and loads are transmitted and carried through to the base support tube, which is attached to the mounting interface adapter. Connected to the hexagon structure is the primary mounting tray bulkhead which extends outboard to support the base of the cylindrical solar cell array. At the top of the central frame a lightweight bulkhead provides the upper closure and secondary attachment surface for the solar array. So as to provide a surface and support for the S-band antenna mount and deployment device at the top of the capsule, a tripod is mounted off three corners of the hexagon frame and extends upward to a central point at the top edge of the solar panel. Below the main tray a cylindrical support tube extends downward to the separation flange. Within this tube, near the base, a mounting tray/ radiator surface is provided to support the high power dissipation components.

In addition to being the primary support structure, the central hexagon frame serves as the mounting structure for the propulsion systems tankage.

Within the base cylinder of the structure a circular plate is centrally supported by three light gusset plates and serves as the bearing surface for the separation compression spring.

The proposed structure is considered to be fabricated using nonmagnetic aluminum alloys, with an additional oxide coating at the mounting surface to minimize any potential solid phase welding (cold welding) at the contact surfaces that might jeopardize successful separation of the OEC. In view of the low compressive stresses and low temperature conditions at the static contact surfaces, solid phase welding is considered quite a remote possibility.

5.4.3 Structure Weight

An assessment of the weight of the proposed structure, excluding bracketry, attachments, and separation adapter system, is as follows:

Central hexagon structure	1.7
Central mounting tray	2.1
Base tray	0.6
Array support rings	1.7
Support tripod	0.2
Base tube	1.9
End bulkhead	1.4
Total	9.6 pounds

A budget of 15 pounds has been established for the OEC structure and attachment (including 3.0 pounds for two radial booms plus an additional 2 to 3 pounds for bracketry). Satellite structures have been built that weigh less than 10 percent of the total system weight; therefore the allocation of 12+ percent for the OEC structure, based on the baseline configuration weight of 122.8 pounds, seems adequate and conservative. The complete support adapter and separation mechanism was estimated at 5.0 pounds, which is also felt to be quite conservative.

5.4.4 OEC Moments of Inertia

In order to maintain a desirable ratio of roll to transverse moments of inertia for inherent spin stabilization of the OEC, the recommended configuration was developed with this criteria as one of the prime considerations. Calculated values of moments of inertia for the orbit configuration OEC are as follows:

$$I_{roll} = 7.5 \text{ slug-ft}^2$$

$$I_{transverse\ (max)} = 6.8 \text{ slug-ft}^2 \text{ (axis normal to booms)}$$

$$I_{transverse\ (min)} = 2.0 \text{ slug-ft}^2 \text{ (axis parallel to booms)}$$

A minimum ratio of roll to transverse inertia of 1.10 results. The values noted above consider two fixed radial booms extending 7 feet from the spin axis of the OEC, each supporting a sensor weighing 1 pound at the outboard tip of the boom. The boom members were considered at 1.5 pounds each.

5.5 THERMAL CONTROL

5.5.1 General

The thermal environment and thermal control requirements during transit and operation including eclipse conditions for the OEC were analyzed. Alternate means of satisfying thermal requirements were investigated. The basic assumptions used in the analysis are as follows:

1) The OEC is to be carried to Mars on the Voyager spacecraft and could remain on Voyager for the first 4 weeks orbiting Mars after a transit period of 12 months (maximum).

2) After separation from Voyager, the OEC is to be spunup to 60 rpm and will orbit Mars at an altitude of 300 to 1500 km at periapsis to 10,000 to 20,000 km at apoapsis.

3) The solar direction assumed in the thermal analysis was 90 ± 25 degrees from the spin axis, a highly conservative value compared to the desired ± 5 degrees from normal required by the experiments.

4) The reflected solar flux from Mars was neglected in this thermal analysis.

5.5.2 Thermal Design Description

The recommended thermal design (Figure 5-20) that has evolved from this study is composed of an insulated body which is variably coupled to the external environment at the one end of the capsule. The insulation consists of multilayer Mylar blankets and the variable coupling is achieved with a rotating shutter of the type built and flown as part of the Applications Technology Satellite program. The shutter, which would be rotated by a bi-metal actuator, will provide a temperature sensitivity of the equipment mounting surfaces inside the OEC of approximately $1^{\circ}C$ per watt of internal power dissipation when the shutter is within its operating range. This temperature variation includes the effect of the sun angle uncertainty of ± 25 degrees and the seasonal variation in the solar flux from aphelion to perihelion.

The shutter is a 2 square foot area circle with 1 square foot of pie-shaped holes cut in it. The radiator under the shutter is 1 square foot of pie-shaped areas painted white that are located so that when the bi-metal actuator is $21^{\circ}C$ ($70^{\circ}F$) the shutter is "open"; i.e., the holes in the shutter are over the white painted pie-shaped areas of the radiator. When the bi-metal actuator is $13^{\circ}C$ ($55^{\circ}F$) the shutter is "closed", i.e., the pie-shaped white radiator areas are covered by the shutter. The thermal capabilities and predicted temperatures with this active temperature control system are shown in Table 5-8.

The weight penalties associated with the thermal control of the OEC are minimal. The end of the spacecraft in this study is 0.024 inch thick aluminum sheet (or aluminum honeycomb sandwich with 0.012 inch thick facesheets) to prevent large internal temperature gradients. The weight, 2.4 pounds, would be

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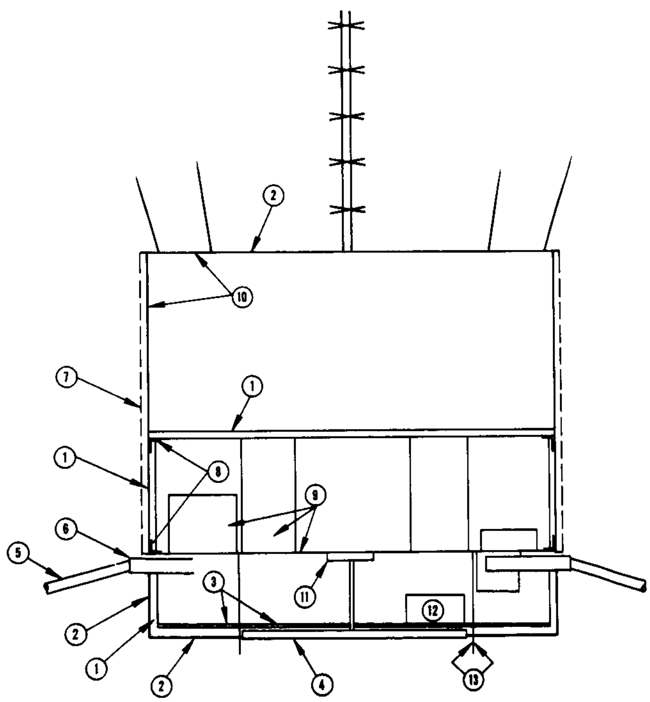

(1) INSULATION, 30 LAYERS OF 1/4 MIL THICK ALUMIZED MYLAR

(2) OUTER COVER SHEET, 1 MIL KAPTON FILM WITH 1400 - 3000 ANGSTROMS OF ALUMINUM ON BOTH SIDES, 6000 ANGSTROMS OF SiO ON THE OUTSIDE ($\alpha = 0.12$, $e = 0.16$)

(3) ALUMINUM SHEET 24 MIL THICK (OR ALUMINUM HONEYCOMB CORE SANDWICH WITH 12 MIL THICK FACE SHEETS). PAINTED BLACK 1 MIL THICK ON INSIDE SURFACE

(4) ROTARY SHUTTER FOR TEMPERATURE CONTROL

(5) BOOM, ALUMINUM TUBE

(6) PLASTIC TUBE FOR LOW CONDUCTANCE BETWEEN INSIDE OF SPACECRAFT AND ALUMINUM BOOM

(7) SOLAR PANEL

(8) PLASTIC MOUNTS FOR LOW HEAT LOSS TO SOLAR PANEL

(9) BLACK PAINT 1 MIL THICK ALL INSIDE SURFACES EXCEPT EQUIPMENT MOUNTING SURFACES, INSULATION, INSIDE OF SOLAR PANEL, AND INSIDE OF END BARRIER

(10) VAPOR DEPOSITED ALUMINUM ON INSIDE OF SOLAR PANEL AND END BARRIER ($e = 0.04$)

(11) BI-METALLIC ACTUATOR FOR TEMPERATURE CONTROL SHUTTER

(12) HIGH POWER UNITS (PRIMARY TRANSMITTER, VOLTAGE LIMITER, AND S-BAND TRANSMITTER TWT)

(13) POLISHED ALUMINUM OR VAPOR DEPOSITED ALUMINUM ON ATTACHMENT RING

Figure 5-20. Recommended Thermal Design

TABLE 5-8. THERMAL CAPABILITIES AND PREDICTED TEMPERATURES

	Thermal Capability, °C		Predicted Temperature, °C	
	Non-Operating	Operational	Transit Phase Non-operating	Orbiting Mars, Operational
Batteries (silver cadmium)	-29 to -7 (-20 to 20°F)	4 to 27 (40 to 80°F)	-29 to -7 (-20 to 20°F)**	4 to 27 (40 to 80°F)**
Experiments*	50 to 80 (-58 to 176°F)	-10 to 55 (14 to 131°F)	-34 to -1 (-30 to 30°F)**	4 to 32 (40 to 90°F)**
• Solar plasma probe				
• Electric field meter electronics				
• Magnetometer electronics				
Spacecraft electronics	-32 to 57 (-25 to 135°F)	-18 to 52 (0 to 125°F)	-32 to -1 (-25 to 30°F)**	-1 to 38 (30 to 100°F)**
Tanks and valves for hydrazine	-73 to 60 (-100 to 140°F)	4 to 60 (40 to 140°F)	-34 to -1 (-30 to 30°F)	4 to 32 (40 to 90°F)
Solar panel	-184 to +120 (-300 to 248°F)	-184 to +120 (-300 to 248°F)	-192 to 65 (-314 to 150°F)	-157 to -22 (-250 to -8°F)

*The temperatures shown are the limits presently assumed.
**Mounting surface temperatures.

divided between thermal control and structures weight. The insulation and cover sheet weight would be 2.46 pounds. The paint for the inside of the ends of the spacecraft and internal packages would be 0.31 pound. The rotating shutter type of active temperature control (bi-metallic strip, linkage, shutter, etc.) would be approximately 0.94 pound. The electrical heater and heater control electronics for the transit phase is 0.2 pound.

5.5.3 System Performance

The capsule internal steady state temperature as a function of internal power dissipation is shown in Figure 5-21.

The solar panel minimum temperature after a 2.65 hour eclipse is -157°C (-250°F). The interior of the OEC has a mean temperature drop of only 2°C in the 2.65 hour eclipse because the minimum internal power dissipation (16 watts) is almost equal to the heat losses to space from the insulated section of the capsule.

The OEC interior temperatures with the primary transmitter operating 4 hours each 15-hour orbit are shown in Figure 5-22a. The temperatures of the capsule interior structure modes, not shown in these figures, are within the required temperature limits. The capsule interior temperatures satisfy the equipment requirements although the battery mounting surface maximum temperature reaches the maximum allowable temperature.

The spacecraft interior temperature with the S-band transmitter operating at 47 watts dissipation for 2 hours each 7-hour orbit is shown in Figure 5-22a. The constant power dissipation was assumed to be 19 watts; later analysis indicated this value to be 16 watts.

The temperatures in Figure 5-22 would be higher if the solar angle were 115 degrees (sunlight on the radiator) rather than 90 degrees. This would cause the battery to be overheated. Lowering the temperature range at which the shutter opens and closes by 10°C and possibly reducing the duration of S-band transmitter operation should keep the battery temperatures within allowable limits.

5.5.4 Boom Considerations

The sunlight will heat the drooped radial booms on one side more than the other. The resulting temperature gradient will cause deflection. However, the angular deflection of the sensors on the ends of the booms can be kept within 0.12 degree (12 percent of the maximum allowable) by using non-magnetic aluminum instead of fiberglass booms. Thermal gradient effects are negligible in the case of a boom mounted along the spin axis.

5.5.5 Transit Phase Considerations

During the transit phase of the mission where the capsule is shaded by Voyager, the batteries require an environment of -29 to -7°C (-20 to 20°F) for optimum storage. The electrical power required from Voyager to keep the inside

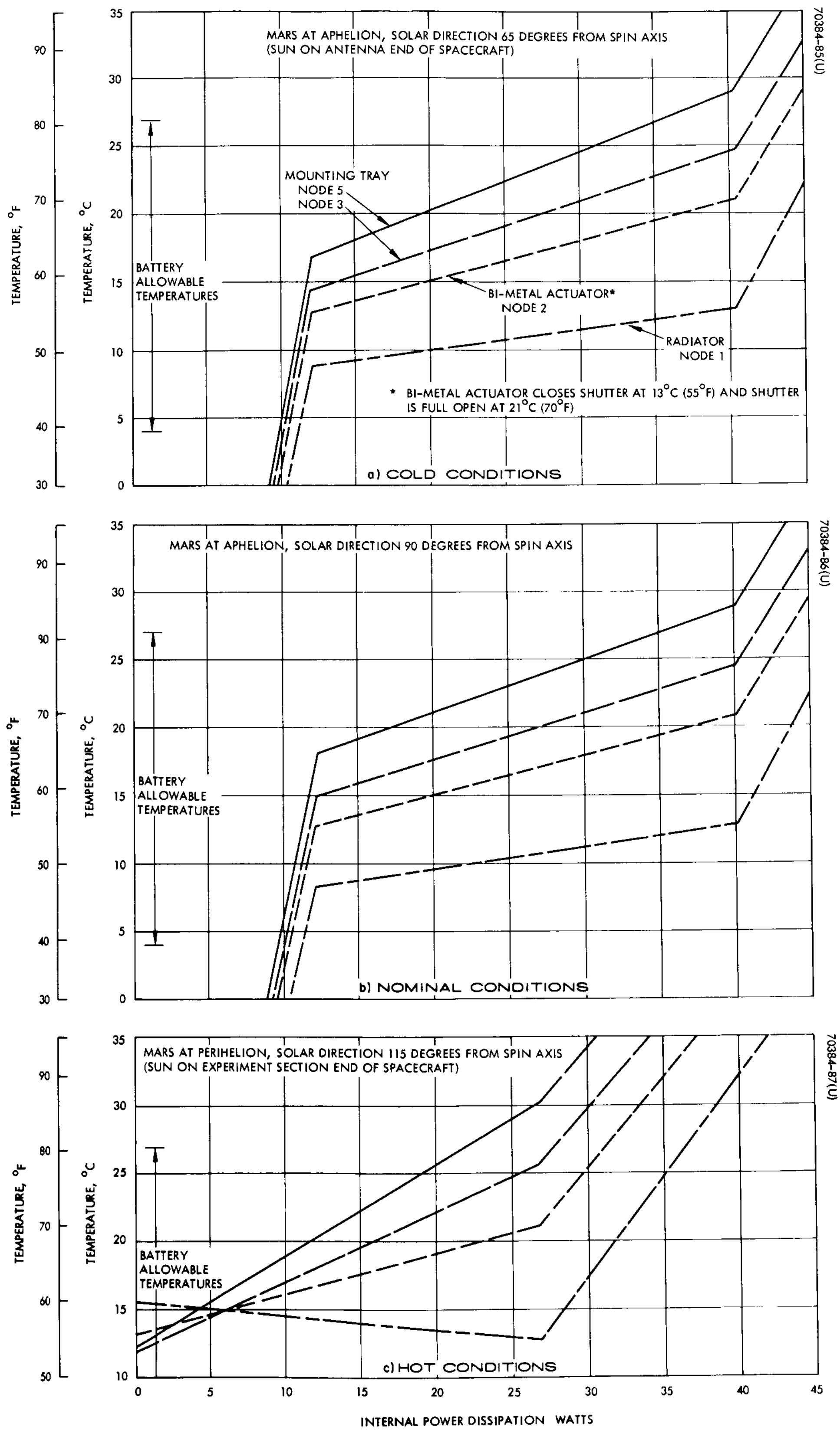

Figure 5-21. Spacecraft Steady State Temperatures

of the OEC at -29 to -7°C is only 9 watts and can be provided using a small thermostatically controlled heater. The minimum solar panel temperature in transit is -192°C (-314°F). The solar panel can withstand temperatures at least as low as -184°C (-300°F) and with technology improvements probably can be designed to withstand -192°C, the minimum predicted temperature.

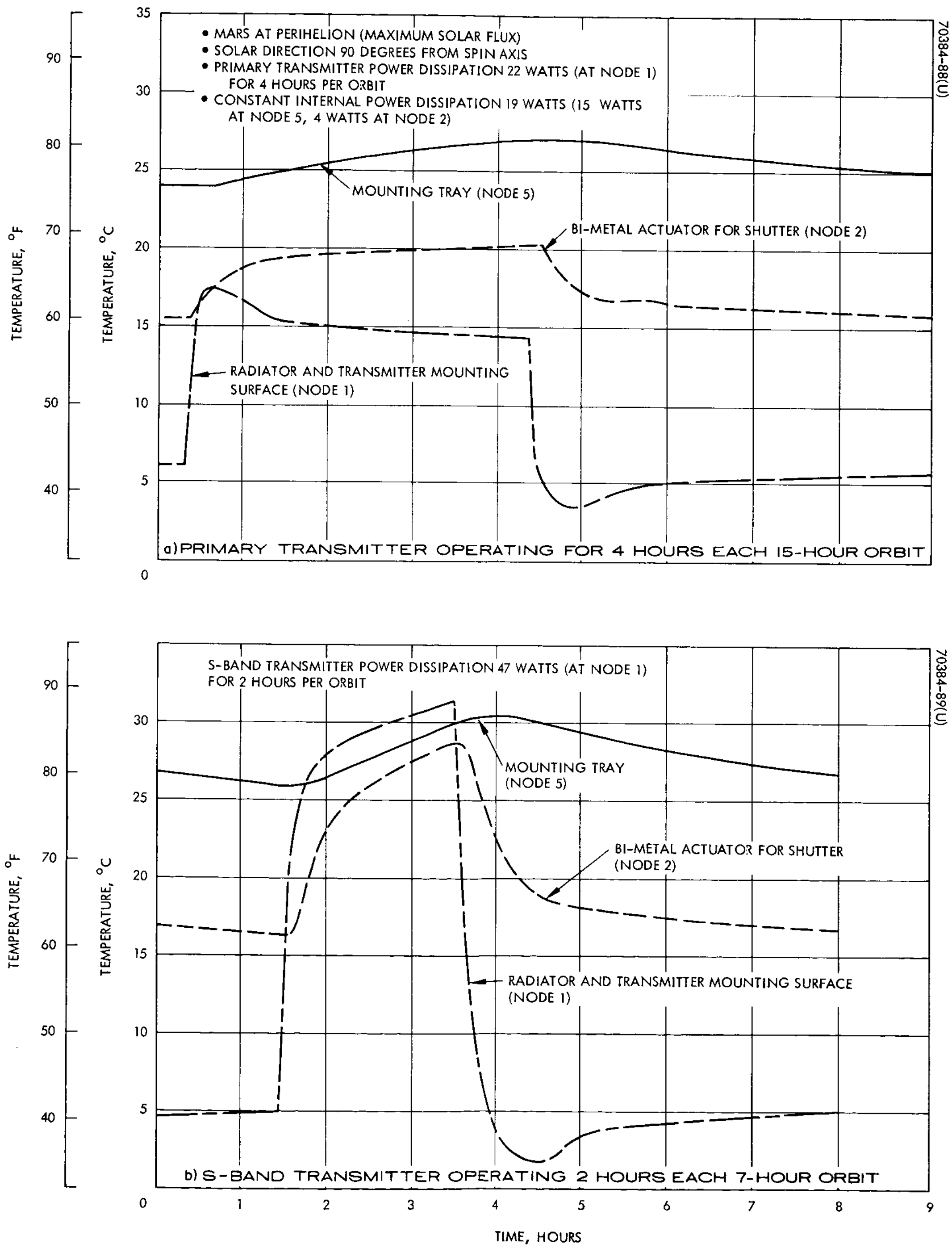

Figure 5-22. Spacecraft Interior Temperatures

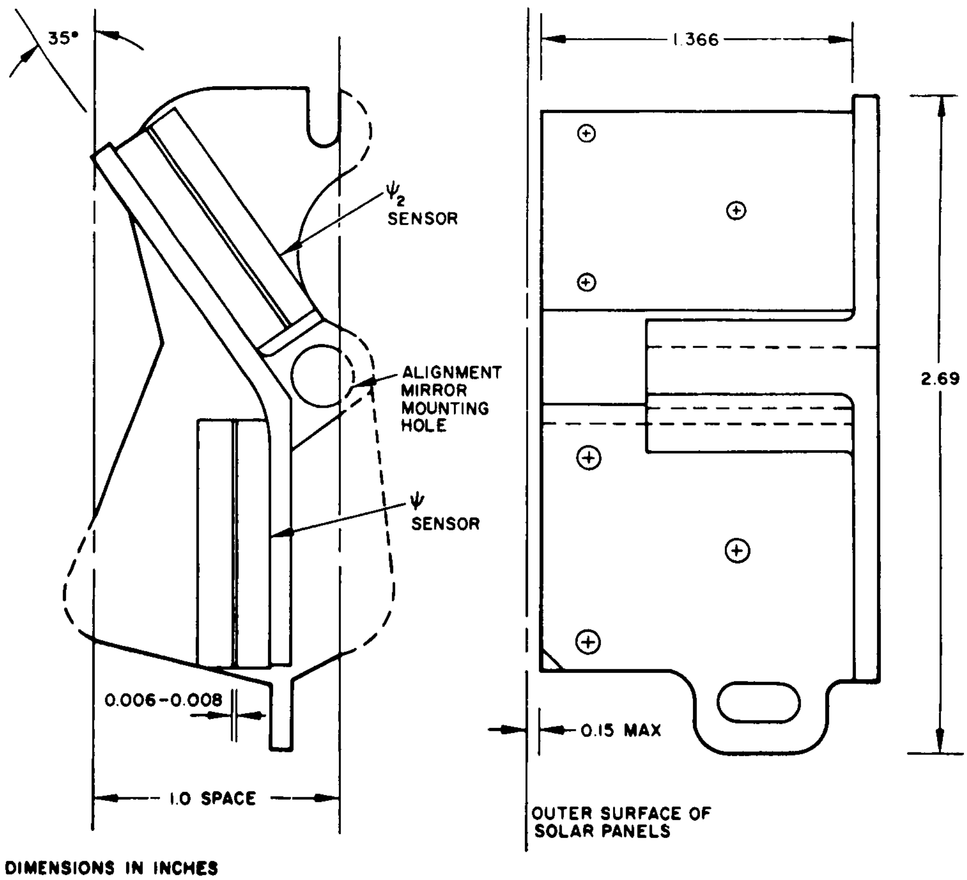

Figure 5-23. Sun Sensor Assembly

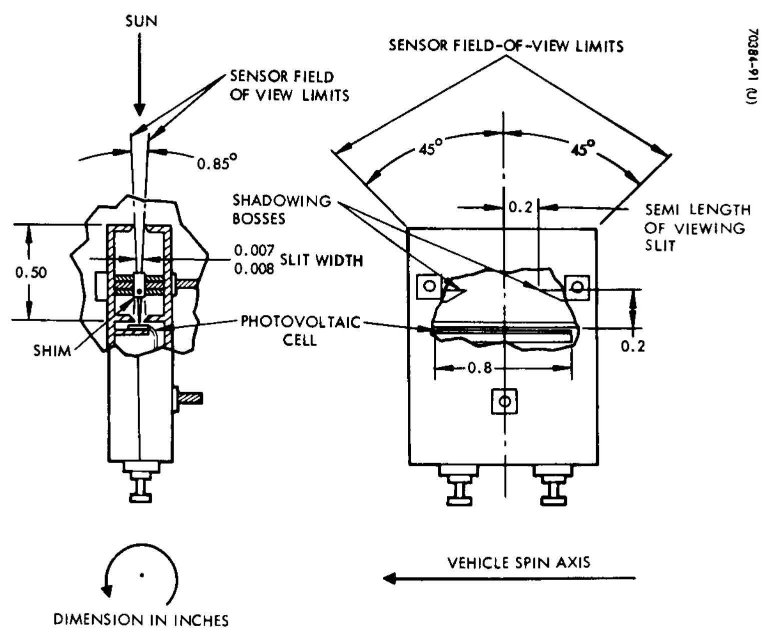

Figure 5-24. Schematic Diagram of Sensor Unit

5.6 SENSORS SUBSYSTEM

5.6.1 General

Two sensors have been selected to provide the information necessary to yield attitude and position measurements for the OEC. These two types of sensors – a sun sensor assembly and a Mars horizon sensor – have been chosen as the two instruments to be integrated into the OEC baseline configuration. A description of each of these sensors is presented in this section.

A simple method of measuring attitude to the sun from a spinning vehicle is to produce a sun pulse with a slit optics type sensor. The width and orientation of the slit on the vehicle define the width and shape of the sun pulse. A lower limit on pulse width is set by the angle subtended by the sun. By aligning two of these slit fields of view at some preselected angle to one another, it is possible to measure the angle between the satellite spin axis and the sun line.

The horizon sensor contemplated for the OEC mission is simply a horizon crossing indicator operating in the infrared spectrum. When used in conjunction with a spinning satellite such as the OEC, a signal is produced each time the leading and trailing IR limbs of the planet are crossed. The sensor element itself is sensitive to the different energies received as it passes from space to Mars during a spin cycle. OEC attitude is determined by measuring the time difference of the leading and trailing edge crossings, which is proportional to a scanned chord of Mars. The accuracy of this sensor is ± 1.5 degrees for the chord length measurement.

The sun sensor establishes the angle between the sun line and the OEC spin axis within a 360 degree cone of uncertainty about the sun line. By measuring the angle from the spin axis to Mars, the spin axis can be uniquely determined. Because of the inherent stability of the OEC and the mission characteristics, the process of attitude/orbit determination does not require real time operation. Hence, the establishment of the attitude can be accomplished over a number of days. This allows for considerable collection of raw data indicating within the basic accuracy of the sensors what the attitude is.

5.6.2 Sun Sensors Description

The sun sensor assembly consists of two identical sensor units mounted on a precision aluminum bracket (see Figure 5-23). A schematic view of a sensor unit is shown in Figure 5-24. The shims placed between the two aluminum sensor halves at the three screw locations determine the width of the sensor slit. The width is controlled within 0.007 to 0.008 inch.

The sensor assembly is a small rugged package that is well able to withstand typical launch and orbital environments. The cell is bonded to the sensor housing with an epoxy cement-fiberglass combination that protects the cell from any damage due to thermal expansion effects in addition to securely holding it in the proper position. The width of the viewing slits is very stable once they are adjusted, due to the mechanical strength of the sensor halves and the use of metal shims. Each sensor unit consists of an n-on-p silicon photovoltaic cell, a load resistor, and two clam-like aluminum shells. A narrow gap between the clam shells defines the narrow, fan-shaped field of view of the sensor. When the sunline and the sensor field of view coincide, the silicon cell is illuminated and an output pulse produced. When the spin axis is aligned along the ecliptic normal, both sensors receive the solar energy at the same time. However, if the vehicle is tipped in either direction, there is a time difference between ψ and ψ_2 pulses as indicated in Figure 5-25.

The ψ and ψ_2 pulses are telemetered to Voyager (after appropriate pulse squaring), and then to Earth for ground processing. The angle between the spin axis and sunline (ϕ) can be determined to ± 0.5 degree on a per pulse basis. By ground and in-flight calibration of the actual unit, and by smoothing the data over a number of measurements, the ϕ angle uncertainty can be reduced to approximately ± 0.2 degree (3σ).

Selection of the inclination angle between the two fields of view is based on considerations of pulse width, scan time, and accuracy. An angle of 35 degrees represents an optimum angle of inclination. Hence

$$\cos\phi = \sin(\psi - \psi_2) \cos 35^\circ$$

The angle ϕ is determined using this equation. Figure 5-26 depicts the sun sensor geometry. The plane of one fan-shaped field of view is nominally parallel to the spin axis.

The electrical output signal from the sensor is a function of the input energy from the sun that falls on the cell, the load resistor, and the diode loading effect of the unilluminated area of the cell. A typical sensor output pulse is shown in Figure 5-27. This width, plus the 21 arc minute angular size of the sun at Mars distance, results in a nominal ψ pulse width of approximately 1.25 degrees. (Due to the 35-degree inclination of the ψ_2 sensor, its nominal pulse width is 1.25/cos 35 degrees = 1.53 degrees.)

As shown in Figure 5-24, the length of illuminated cell area is approximately 0.4 inch. Thus, the nominal illuminated area is 0.4 x 0.0075 inch = 0.003 square inch. The defined field of view of the sensor is ± 45 degrees from the normal to the cell surface. At angles greater than 45 degrees to the cell, the illuminated area (and consequently the sensor output) drops off sharply. The complete sensor assembly weighs approximately 0.15 pound.

5.6.3 Horizon Sensor Description

The horizon sensor contemplated for the OEC mission is simply a horizon crossing indicator operating in the infrared spectrum. When used in conjunction with a spinning satellite such as the OEC, a signal is produced each time the leading and trailing IR limbs of the planet are crossed. Figure 5-28 illustrates how the sensor is used. The sensor element itself is sensitive to the different energies received as it passes from space to Mars during a spin cycle. OEC attitude is determined by measuring the time difference of the leading and trailing edge crossings which is proportional to a scanned chord of Mars.

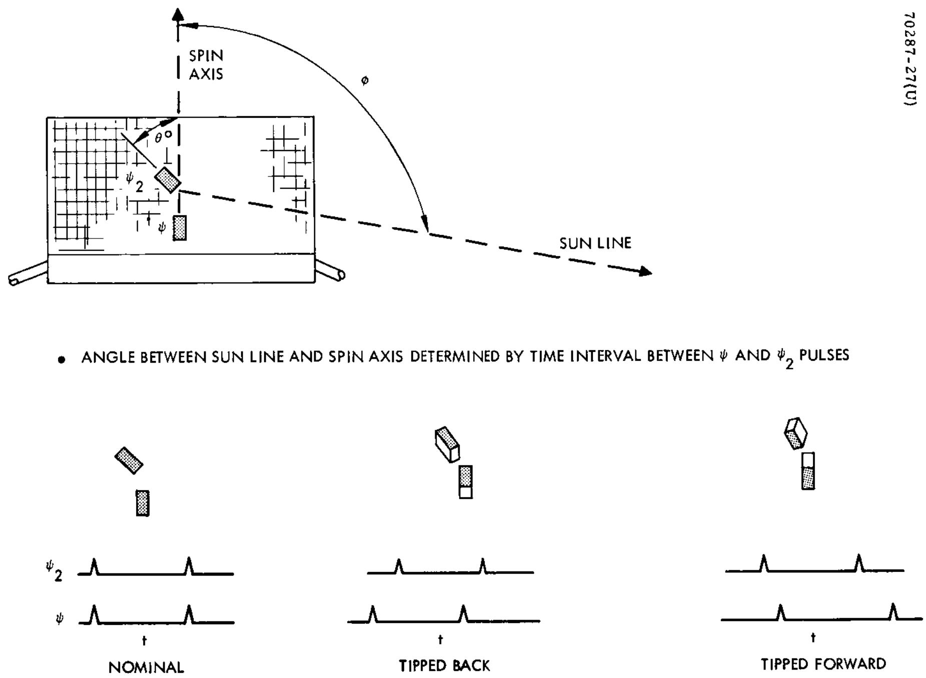

Figure 5-25. Spacecraft Attitude as It Affects Relative Position of ψ and ψ_2 Pulses

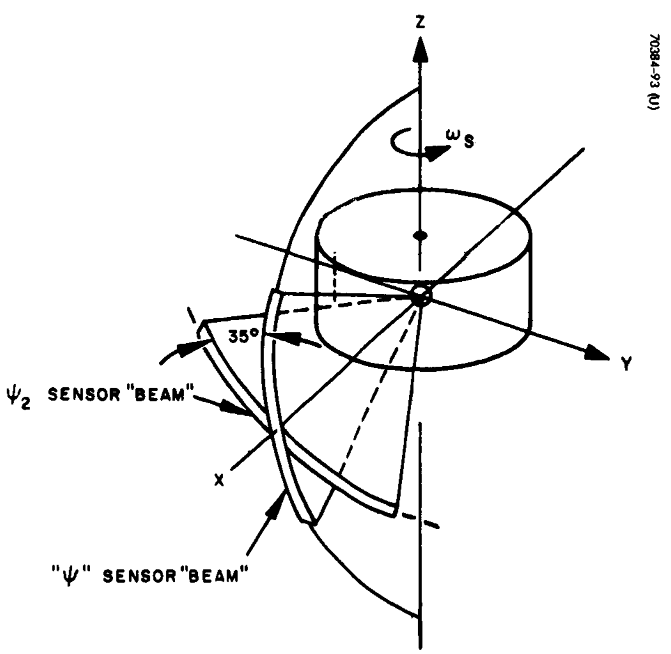

Figure 5-26. Sun Sensor Geometry

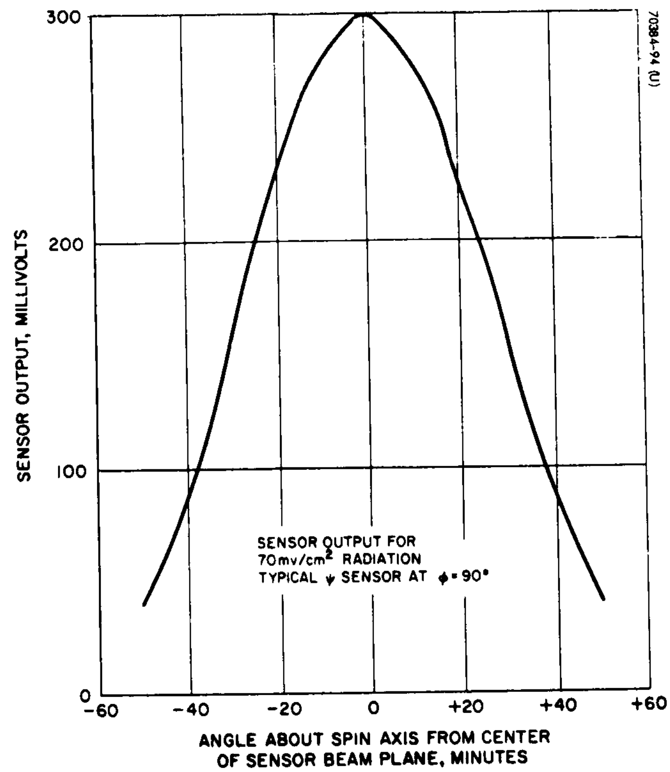

Figure 5-27. Typical Sun Sensor Output

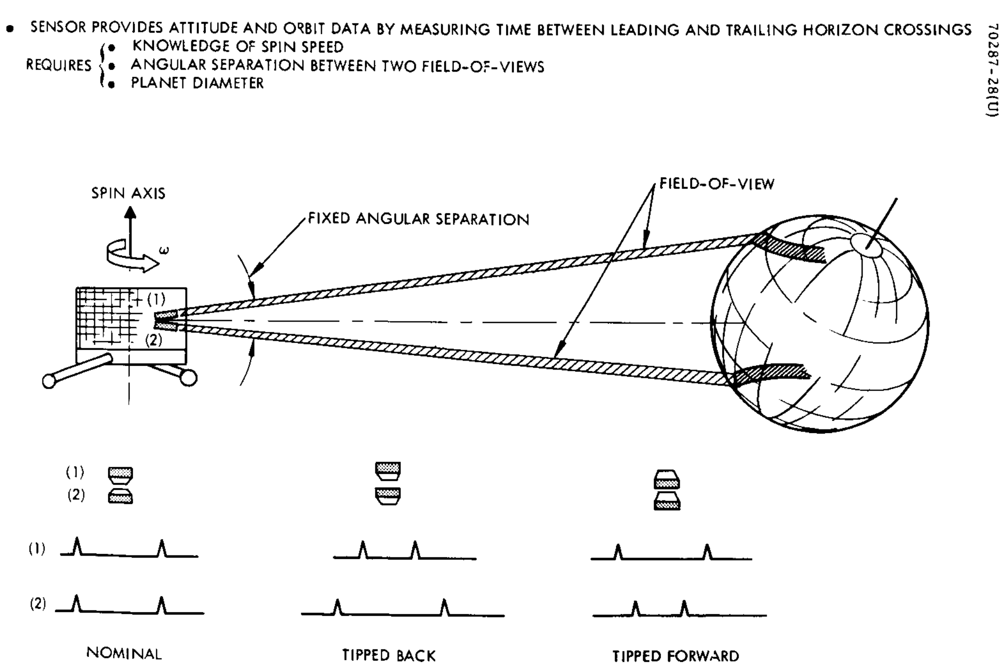

Figure 5-28. Application of Mars Sensor

Processed attitude data is used to update the initial OEC orbital information on a periodic basis. By the use of two horizon indicators inclined to one another at some fixed angle, and with knowledge of the spin speed, the altitude of the capsule above Mars can be ascertained and used to establish an updated orbit.

This form of horizon sensing for attitude or orbit determination has been applied on such Earth orbiting satellites as TIROS and the Hughes HS-308 (currently being built).

The Mars sensor system assembly is shown in Figure 5-29. Two narrow-beam IR sensors in a common housing assembly are arranged with one unit aimed 21 degrees above and the other 21 degrees below the sensor centerline.

Each of the two sensor units has three mounting bosses that define a plane whose relationship to the sensor optical axis is constructed to be within 0.05 degree. These mounting bosses in turn mate with accurately located bosses on each sensor assembly housing. The housing for each unit incorporates alignment references (such as an accurate hole and slot) that accommodates an alignment fixture. The alignment fixture has mirrors that permit the use of autocollimation techniques during the final assembly alignment of the vehicle. The weight of the two sensor units is less than 3 pounds.

The sensor consists of a coated germanium optical system, a multilayer interference filter, an immersed thermistor bolometer detector, and processing electronics. The detector element is masked to precisely define the sensor field of view limits. The most fragile part of the sensor is the IR telescope, including the optics, filter, and detector. The front part of the sensor in which the telescope is located will be supported to ensure that resonance amplification is nil.

Operating altitudes ranging from 500 km at periapsis to 20,000 km at apoapsis were considered in sizing the cant angle between the two horizon sensors.

For the above altitude range, the angle β is constrained to operate between $17 \leq \beta \leq 102$ degrees where the angle is defined with respect to the planet's actual disc.

An optimum sensor angular separation of approximately $\beta = \pm 21$ degrees is selected. A nominal OEC spin speed of 60 rpm is assumed to determine Mars scan time. The scan angle is 30 degrees and the duration is 83.3 milliseconds. These values vary with the OEC orbit position or altitude.

Various types of indicators could be considered for this task, but most visible light indicators have difficulty contending with diurnal effects. By operating in the infrared spectrum, these effects are avoided. Many successful satellite designs implementing infrared horizon sensing technology have proven the concept in an Earth environment. The differences between the IR characteristics of Earth and Mars requires adaptation of such a sensor to the expected Mars environment. The available data on spectral information pertaining to the Martian atmosphere is used to design the instrument. Much of the available data falls within a 7 to 13 micron portion of the spectrum.

Because of the OEC's wide range of possible attitudes, the sensor will at the same time scan the Mars poles. Because of the lack of sufficient atmospheric detail, a conservative design approach is to account for the worst case temperature variation.

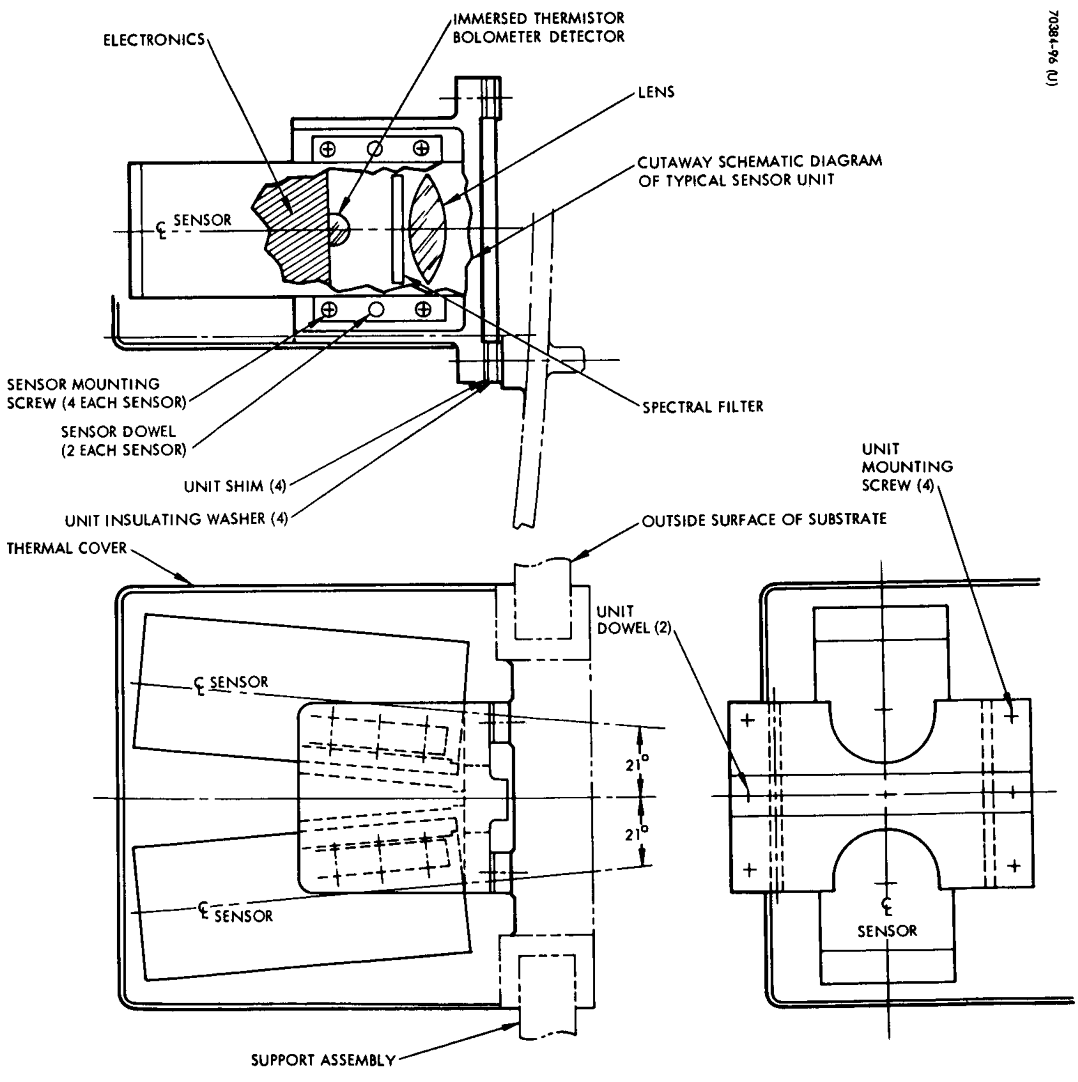

Figure 5-29. Mars Sensor Assembly

6.0 MAGNETIC CLEANLINESS

The desire to perform accurate measurements of the very weak Martian magnetic field is a prime motivation for the OEC mission whose feasibility is the subject of this study. In general, the achievability of a particular level of magnetic cleanliness is not susceptible to paper analysis — magnetic cleanliness is achieved only by a thorough and painstaking program of controlling the design, parts, materials, processes, and techniques backed by a comprehensive testing program. Feasibility, then, can only be demonstrated on the basis of past performance — and rests on the implicit assumption of magnetic controls equaling or surpassing in effectiveness those used in past programs.

6.1 OVERALL SPACECRAFT FIELDS

The OEC spacecraft magnetic fields are required to be less than 0.25 gamma (1 gamma is 10^{-5} gauss) at the location of the (boom-mounted) magnetometer. One perspective on the feasibility of achieving this low level is gained by a comparison with magnetic contamination levels achieved on comparable spacecraft. Table 6-1 gives a comparison with Pioneer and IMP spacecrafts, both comparable in size and complexity to OEC. It can be seen that the OEC magnetic requirement, while severe, appears feasible.

TABLE 6-1. OVERALL COMPARISON OF MAGNETIC CLEANLINESS

Spacecraft	Weight, pounds	Boom, inches	Field
Pioneer VI	140	83	0.58γ (TRW data)
IMP I	140	82	0.8γ
IMP F and G	~150	82	0.25γ (specification)
OEC	123	84 (can go to ~96)	0.25γ (specification)

Magnetic Budgets — Detailed Estimate. In order to further assess the difficulty of meeting the OEC requirement, preliminary estimates of the field contributions of individual units comprising the OEC equipment have been made. Volume II lists data on various pieces of equipment obtained from Pioneer and Mariner programs. Most of the units relevant to OEC are comfortably under the specification levels. The traveling-wave tube (for S-band communication) and the tape recorder fall in the category of "special cases" with permanent magnetic characteristics so that the total field contribution from these units is largely that in the de-permed state. The contributions to the net OEC field from these two items are substantial with the tape recorder a distinctly critical item.

In the baseline OEC configuration, the magnetometer is mounted 7 feet from the spacecraft center. In the preliminary layout of equipment arrangement within the spacecraft, some attempt was made to locate the primary magnetic offenders as far away as possible from the magnetometer. Table 6-2 lists the principal units and assemblies in the OEC and tabulates the estimated magnetic field contributions of each at the magnetometer. The first entry in the table serves as a comparison for a typical subsystem by specification on the IMP F and G spacecrafts. It is evident, as expected, that the tape recorder is the prime offender, even when placed as far as possible from the magnetometer.

The preliminary budgets given in Table 6-2 yield a "worst possible field" of 0.6γ when added together arithmetically; if the individual field contributions are root sum squared, a "best possible field" of 0.24γ results. Excluding the two "special cases" (tape recorder and TWT), the values are 0.4γ arithmetic and 0.17γ rss'd. If the magnetometer were placed 8 feet from the spacecraft center, these values would decrease to 0.45γ arithmetic, 0.17γ rss with tape recorder and traveling-wave tube and 0.28γ arithmetic, 0.12γ rss without these two items.

These results do not provide assurance that the 0.25γ goal set for OEC will be met; as always in a spacecraft program, many compromises are required in the interest of cost, schedule, or particular interface and performance criteria. The preliminary estimates, however, show that both on the basis of previous spacecraft achievements and on the basis of data on the particular units, the 0.25γ goal should be regarded as a feasible one.

6.2 ACHIEVEMENT OF MAGNETIC CLEANLINESS – MAGNETIC CONTROL

As is evident from the preceding discussion, the achievement of the OEC magnetic goals is no simple task. To meet the specified levels, the elimination of magnetic contributions must be pursued throughout the design, development, and assembly of the OEC. Basically, a magnetic control program must be adopted that will have as its prime objectives:

- Total elimination of permanent magnets and ferromagnetic materials from the spacecraft except where absolutely required
- Careful control, supplemented by testing, of materials, processes, and components to ensure that the spacecraft contains an absolute minimum of magnetically permeable material

TABLE 6-2. PRELIMINARY MAGNETIC BUDGET

Item	Distance, inches	Field	Comments
Reference for typical subsystem	84	0.06γ	Taken from specification for IMP F and G
Tape recorder	97	0.17	Critical time — consistent with Mariner 67
Control jets (2)	96, 99	0.02 each	Measured on ATS
Communication electronics	72	0.10	Consistent with Pioneer VI and IMP specifications
Power electronics	80	0.074	Consistent with Pioneer VI and IMP specifications
Magnetometer electronics	75	0.06	Consistent with Pioneer VI and IMP specifications
Other electronics	90	0.05	Consistent with Pioneer VI and IMP specifications
Traveling-wave tube	90	0.05	Pioneer VI
Structure, harness, etc.	84	0.08	
Arithmetic sum		0.62γ	
RSS sum		0.24γ	

- Careful control and test to ensure that design of electronic units, wiring harnesses, and connectors, etc., produce fields well below critical levels in all their possible operating modes

- Arrangement of units within the spacecraft so as to minimize the contaminating field at the magnetometer

Criteria for the design of magnetically clean spacecraft have been developed on the basis of the experience gained in implementing (successfully) such programs as IMP and Pioneer. This clearly pertinent information should be maximally utilized in the OEC program, supplemented by results of testing for items specific to OEC.

A magnetic controlplan must be an integral part of the overall spacecraft hardware planning and must contain at least the following items:

1) Control organization

2) Specifications

3) Procedures and tests

4) Fabrication and assembly

5) Handling

To implement a strict magnetics control program, an organization should be formed to have a responsibility on a parallel with the spacecraft design, quality assurance, and experiment integration functions. This group would have responsibilities for

- Maintaining and augmenting, as appropriate, a Magnetic Requirements document based on best available information and continually updated to take account of data generated by test. Close liaison with NASA personne should be maintained to assure maximum utilization of experience with magnetically clean design and fabrication techniques acquired during the IMP, Pioneer, and other programs.

- Reviewing and approving circuit design, layout, and selection of materials and components to assure compliance with magnetic requirements. Approval of exceptions to these requirements or of new or unusual materials or techniques should be granted only after consultation with the NASA project office; such approvals, with supporting data, should be carefully documented.

- Reviewing and approving procurement specifications and vendor processes and facilities to ensure compliance with magnetic requirements. Certification and on-site surveillance of vendors should probably be accomplished in conjunction with the overall quality control function, monitored and reviewed by the Magnetic Control Organization.

- Formulation and implementation of a plan for magnetic inspection of parts and components. It is likely that many (if not all) parts will require 100 percent magnetic inspection (50 gauss de-perm followed by measurement after 25 gauss perm and after subsequent 50 gauss de-perm) Establishment of criteria for selection of magnetically acceptable parts should be closely coordinated with NASA.

- Formulation, coordination, and implementation of a plan for magnetic testing of subassemblies. Tests should be performed as early as possible in the development cycle to permit the identification and rectification of any trouble areas. All questionable items or techniques should be proven by testing unless adequate assurance of magnetic acceptability can be otherwise provided.

A preliminary outline for an OEC Magnetic Control Plan is presented as a part of Volume III.

7.0 OEC RELIABILITY

7.1 RELIABILITY ASSESSMENT

The goal set for the OEC is a 0.75 reliability over a 12-month transit period, stowed on the Voyager spacecraft, followed by a 6-month orbit operational period. This requirement is relatively modest as space systems go and appears to be well within the present state of the art. Hughes satellites of comparable or greater size and complexity than the OEC have presently accumulated 11 satellite years in space, with all of those that were placed in their specified orbits still fulfilling their intended functions. The earliest of these satellites, Syncom 2, has been in orbit for over 3 years. Thus, the feasibility of achieving the OEC goal can be demonstrated by analogy to other space systems.

Another approach to the demonstration of the feasibility of achieving the desired OEC reliability is through a block diagram reliability assessment using the specific OEC equipment with failure data estimated or extrapolated from experience with similar units. Such an assessment, reported in Volume II, has been performed yielding a prediction of 0.78 for the baseline OEC over its 18-month life (6 months in orbital operation, 12 in stowed condition). This prediction should be considered pessimistic since detailed failure mode analysis and determination and incorporation of optimum redundancies have not been performed in this feasibility study.

The baseline OEC has been configured to have a number of capabilities not absolutely essential to meeting a minimum mission function. Such capabilities include the orbit change propulsion and the tape recorder for data storage. This same spacecraft operated in its simplest mode (real time data transmission, no propulsion except spinup) yields a reliability of 0.92 for the 18 months.

Reliability block diagrams for the baseline OEC and for the same spacecraft operated in its simplest ("co-orbital") mode are shown in Figures 7-1 and 7-2. Table 7-1 lists the sources from which part types, parts counts, and/or life data were taken for the OEC equipment reliability estimates. The dormant failure rates were derived on the basis of studies recently performed by the Martin Company and published by Rome Air Development Center.*

*D.F. Cottrell et al, RADC-TR-66-348, "Dormant Operating and Storage Effects on Electronic Equipment and Part Reliability," October 1966.

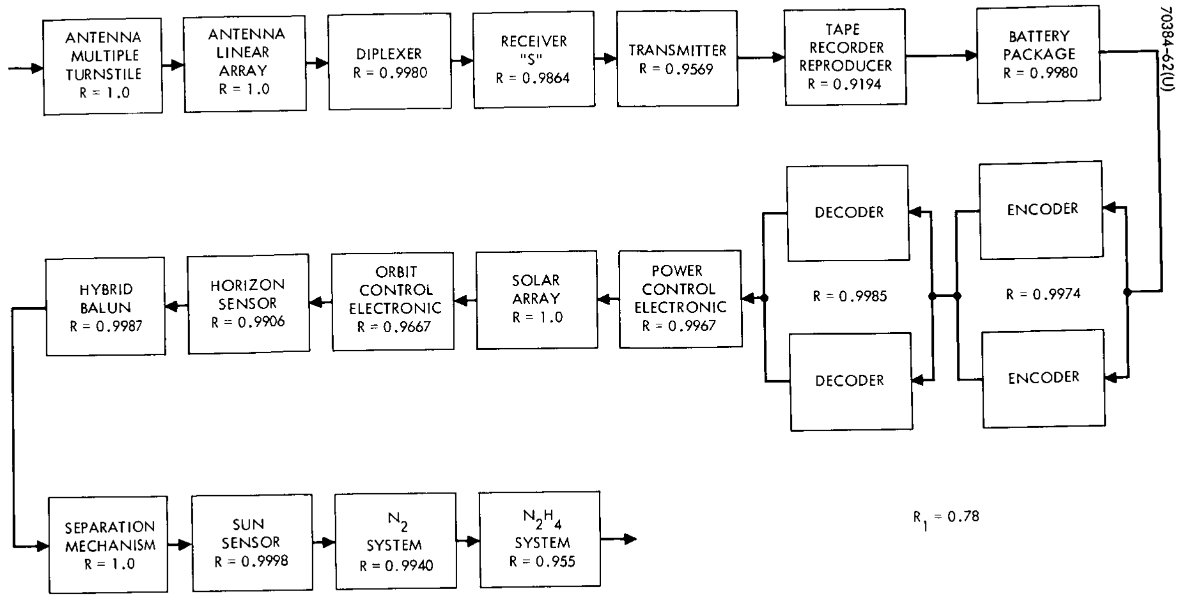

Figure 7-1. Reliability Block Diagram for Primary Communication Mode With Orbit Change

High speed data transmission rate capability

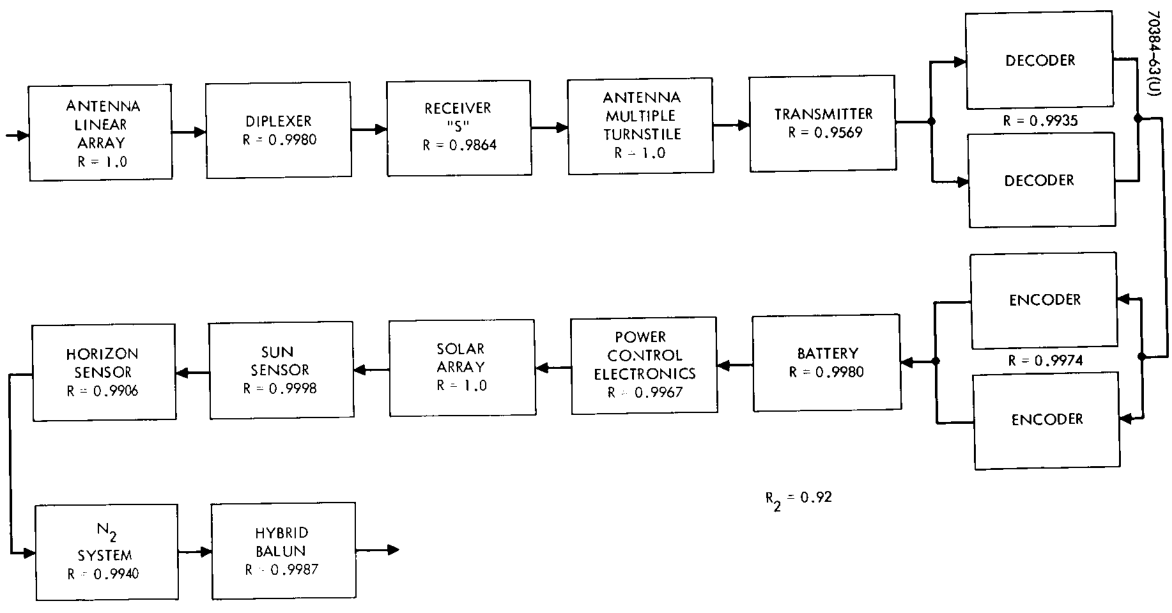

Figure 7-2. Reliability Block Diagram for Primary Communication Mode Without Orbit Change

Continuous data transmission capability

TABLE 7-1. OEC RELIABILITY ESTIMATES

Unit	Failure Rate, percent per 1000 hours		Reliability	Reference Source
	On	Off		
Receiver	0.1541	0.0136	0.9920	Drawing 457210-100
Transmitter	0.8338	0.0847	0.9569	Power amplifiers 3080013 and 475220-101
Decoder	0.57514	0.1616	0.9613	ATS drawings
Encoder	0.6267	0.2849	0.9488	IDCSP/A
Tape recorder reproducer	1.60	0.16	0.9194	Performance data-space
Orbit control electronics	0.3876	0.1904	0.9667	IDCSP/A
Horizon sensor	0.1775	0.0177	0.9906	IDCSP/A
Sun sensor	0.0064	0.0006	0.9998	IDCSP/A
Diplexer	0.0404	0.0020	0.9980	ATS drawings
Power control electronics	0.0877	0.0076	0.9967	IDCSP/A
Hybrid balun	0.0245	0.002	0.9987	ATS drawings
Battery package	—	—	0.9980	Performance analysis
N_2 spinup system	—	—	0.9940	JPL Contract 951720
N_2H_4 propulsion system	—	—	0.955	JPL Contract 951720
Solar array	—	—	1.0	Performance studies
Antenna — 8 whips	—	—	1.0	Performance analysis
Receiver — S-band	0.2580	0.0250	0.9864	Drawing 231900 — less transponder
Transmitter — S-band	1.4000	0.1424	0.9286	Drawing 263220 — less transponder
Diplexer — S-band	0.0244	0.0024	0.9987	Drawing 231872
Antenna — linear array	—	—	1.0	Syncom

8.0 CONCLUSIONS

The findings of the technical studies conducted reveal that the OEC mission is feasible and that the recommended configuration is capable of the following:

1) The recommended configuration is capable of supporting the operation of the baseline scientific payload. This payload is that described in this report as to weight, power, viewing, accuracy, data rate, magnetic cleanliness, etc., requirements. In addition, other payloads could be incorporated if the weight budgets were to be increased. Such additional payload could be, for example, the equipment required to perform mother-daughter occultation with the Voyager spacecraft.

2) The configuration meets all the requirements stipulated by the Ames Research Center not only as to scientific payload parameters but also with regard to compatibilities and constraints on the Voyager mission. The recommended configuration can be stowed in any of the three proposed 1973 Voyager spacecraft configurations (TRW, GE, Boeing). The perturbing forces associated with separation have negligible effects on the spacecraft and the requirements aa to lifetime in orbit, operational life, and reliability can be met.

3) The OEC can operate in different modes of data acquisition and transmission. All of the commands to the OEC will be sent directly to the OEC from Earth via S-band. The initial umbilical ejection, separation, and initiation of OEC sequences will be commanded via Voyager spacecraft. During its normal operation, the OEC will continuously gather scientific data and relay it real time to the Voyager spacecraft where it will be stored until ready for transmission to Earth. This mode of transmission will be limited by the power levels on board the OEC However, sufficient power is available to transmit continuously for 6 months if no orbit changes are effected. In the event of orbit changes, occultation and distances would not allow continuous transmission of full data rates. In this case, the data may be stored on board the OEC until geometry opportunities allow the data transmission. Another option available to the OEC for continuous data transmission after orbital changes are effected is that of "stationkeeping" or orbit synchronization with the Voyager spacecraft. The OEC will have a backup capability for transmission of engineering data and limited scientific data via S-band directly to Earth. These data transmissions are limited by available power levels in the OEC However, it does give the OEC the freedom of operating independently from the Voyager spacecraft.

4) The OEC is capable not only of performing attitude corrections and changes such as a 180 degree spin axis precession for spacecraft magnetic field calibration but also of making orbital changes. These orbital changes may be periapsis drop, orbit inclination, or line of apsides rotation. The 122.8 pound recommended configuration carries sufficient fuel to permit a periapsis drop from an initial 1000 km to 350 km. This lower level is dictated by the lifetime in orbit requirement of the Voyager mission.

5) The OEC can survive the orbital Mars environment. The thermal control has been designed to provide survival to all of the OEC subsystems for a full Martian cycle. This thermal control consists of both passive and active (rotating shutters) techniques. During the transit phase heaters are used which draw power from the Voyager spacecraft. The solar array has been designed to provide adequate power levels after prolonged exposure to the radiation environment.

6) The OEC mission imposes a minimum of requirements on the Voyager mission. These requirements are briefly described below.

Stowage and Release Corridor. The study conducted on the three proposed Voyager spacecraft configurations (GE, TRW, Boeing) revealed that the OEC recommended configuration could be stowed and ejected from the spacecraft without imposing any requirements on the spacecraft configuration in terms of equipment location. In fact, the available envelope from these three spacecraft configurations served as one of the constraints in defining the design envelope of the OEC.

Mounting Surface. The mechanical interface between the OEC and the spacecraft will consist of a mounting surface for the OEC. This mounting surface is defined in detail in Section 5.1.4.5 of this volume.

Thermal Interface. The OEC configuration incorporates a heater for maintenance of the required subsystem temperature during the transit phase. Power for this heater is to be provided by the Voyager spacecraft. The power requirements are quite modest and in the vicinity of 10 to 15 watts. Although the OEC presently shows this requirement, an alternate feasible approach is available in which insulation for the OEC would be provided.

OEC Separation. A capability would have to be provided on the spacecraft to receive the separation command for the OEC. An umbilical is required between the spacecraft and the OEC to provide initiation of the OEC sequencer via the spacecraft.

Antenna Requirement. From the spectrum of Voyager orbits presently covered and for continuous transmission of scientific data from the OEC to Voyager, studies have concluded that a virtually isotropic antenna pattern, i.e., full 4π steradian beam, would be required. Actually, strictly adhering to the present spectrum of Voyager orbits defined, a 160 degree pancake beam could provide continuous visibility under all possible orbital conditions assuming no OEC or spacecraft orbital plane changes.

Data Storage and Relay Capability. For the primary mode of OEC scientific data transmission, a relay link between the OEC and the Voyager is required to accept data transmission at a rate of 630 bits/sec. These data in turn are to be stored by the Voyager Orbiter data handling system to be periodically transmitted to Earth via S-band link to the Deep Space Net (DSN). The receiving and storage equipment requirement on the Voyager does not represent a new hardware requirement since the equipment presently planned for data retrieval and storage from the Voyager Lander would be used for the OEC operation. This approach does not interfere with the Lander operations since OEC operations do not begin until the Lander operations are terminated. For the backup OEC communication mode (S-band), there is no requirement for data storage and relay capability of the Voyager spacecraft since the data transmission is directly from the OEC to the DSN in a degraded duty cycle mode.

Finally, the remaining problem areas must be assessed. The studies conducted do not reveal problem areas that could affect the mission feasibility. However, mission flexibility is grossly dependent on weight allocation to the OEC. Although an OEC mission can be conducted with a nominal gross weight of 75 pounds including 15 pounds of scientific payload, in order to introduce the flexibilities previously discussed, the OEC nominal weight is close to the upper limits under consideration, i.e., 125 pounds.

If other scientific payloads are to be considered in addition to the previously described nominal payload, additional weight must be allocated to the OEC or some of its flexibilities removed such as the orbit change capability, data storage, or S-band backup mode of operation.

There are critical areas in the design of the OEC such as the separation system and the communications and data handling subsystems. However, the design requirements associated with these subsystems are not classified as problem areas.

9.0 REFERENCES

1. "Specification for an Orbital Experiment Capsule Study," NASA Ames Research Center Specification No. A-12506.

2. Performance and Design Requirements for the 1973 Voyager Mission, General Specification for, Jet Propulsion Laboratory, SE 002 B3001-1B21, 1 January 1967.

3. D. F. Spencer, "Our Present Knowledge of the Martian Atmosphere," Jet Propulsion Laboratory, Contract No. NAS 7-100, presented at AIAA/AAS Stepping Stones to Mars Meeting, 28-30 March 1966.

10.0 GLOSSARY OF TERMS

Acquisition	The process of locating the orbit of a satellite or trajectory of a space probe by properly pointing an antenna or telescope, to allow gathering of tracking or telemetry data.
Active	Performing a dynamic function as "active thermal control" in contrast to "passive."
Aerographic	Description of air or atmosphere.
Adapter	The flange, or extension, on one portion of a vehicle providing the means of fitting another portion to it.
Albedo	The albedo of a celestial body is the ratio of the total amount of sunlight reflected from the body in all directions, to the amount that falls on the body.
Ambient	Condition of the environment surrounding a body in motion but undisturbed or unaffected by it, as in "ambient air," or "ambient temperature."
Angstrom	A unit of length, used chiefly in expressing short wavelengths. Ten billion angstroms equal one meter.
Aphelion	The point which is farthest from the sun on the orbit of a celestial object orbiting the sun.
Apoapsis (apofocus)	On the orbit of an object, the point which is farthest from the body orbited.
Apse-Line	Line between periapses and apoapsis.
ARC	Ames Research Center
AREO	Combining form of <u>Ares</u> (Mars) as in "areography" (geography of Mars).
Argument of Latitude	The angle measured in the orbit plane in the direction of motion from the ascending node to the object in orbit. It is numerically equal to the sum of the argument of perifocus and the true anomaly.

Glossary of Terms

Argument of Perifocus	The angle measured in the orbit plane in the direction of motion from the ascending node to perifocus.
Ascending Node	The point at which an object's orbit crosses the reference plane (usually the ecliptic) from south to north.
Astronomical Unit (A.U.)	Semi-major axis of earth's orbit about the sun.
Atmosphere	The envelope of air surrounding the earth; also the body of gases surrounding any planet or other celestial body.
Attitude	Orientation of a space vehicle as determined by the inclination of its axis to a frame of reference.
Attitude Control System (ACS)	A system within the flight control system to maintain the desired attitude of a vehicle.
ATS	Application Technology Satellite.
Axial Jet	The nozzle or thruster assembly of a propulsion system oriented parallel to the spin axis to provide an impulse abou a transverse axis.
Backup Item	In Research and Development programming, an additional item under development to perform the general functions of another item under development. The item may be seconda to an identified primary item, or a parallel development to enhance the probability of success in performing the genera function.
Ballistic Coefficient	(W/C_DA) A design parameter indicating the relative magnitude of inertial and aerodynamic effects, used in performan analysis of objects which move through the atmosphere.
Baseline Configuration	Recommended configuration of the OEC vehicle evolved by tradeoff studies.
Biological Sterility	A condition of complete absence of viable organisms.
Blowdown Spinup System	A propulsion system designed to provide the spinup torque via the release of all of the stored gas in "one time" operation.
Boom	A member used to support or extend a sensor some finite distance from a vehicle.

Glossary of Terms

Boost	A descriptive term which defined the use of rocket propulsion, either solid or liquid propellant types, during initial climb, liftoff, and first phase of the propelled flight.
Boost Phase	The portion of the powered flight period beginning at liftoff of the Saturn V Launch Vehicle and ending with injection of the spacecraft into a Mars trajectory.
Booster	The entire propulsion system, including all stages of an aerospace vehicle, but not including those propulsive elements which are a part of the payload. The booster comprises the propulsion system and the structure. Herein used to identify rocket engines of the Saturn V.
Bus	A vehicle designed to house and support equipment, i.e., "scientific bus."
Cable Cutter	A pyrotechnic device used to sever cables as a method of release.
Cable Harness	Wires and cable so arranged and tied that they may be inserted, connected, or removed after disconnection, as a suit. Sometimes called wiring harness.
Capsule	The OEC, a scientific bus.
Checkout	A sequence of operational and calibrational tests performed to determine the condition and status of a system or any portion thereof.
Cold Welding	Solid phase welding, the adhesion or cohesion of metals occurring in a vacuum.
Command	A signal that initiated or triggers an action in the receiving device.
Communication Satellite (COMSAT)	A satellite designed to reflect or relay radio or other communication waves.
Configuration	An arrangement of components or subsystems comprising a system.
Constraint	A physical, mechanical, electronic, metallurgical, thermal, or other limitation placed upon a design or an action under which a specified approach or procedure must be followed.
Control System	A system is a space vehicle or spacecraft that serves to maintain or change its attitude.

Glossary of Terms

Co-orbital System	Referring to a mode of operation between the OEC and Voyager where for the life of the mission no occulation occurs and continuous relay capability exists.
Dead Band	A regime in which there is no response to an input signal.
Diplexer	A device permitting an antenna system to be used simultaneously or separately by two transmitters.
Doppler Shift	The change in frequency with which energy reaches a receiver when the source of radiation (or a reflector of the radiation) and the receiver are in motion relative to each other. The Doppler shift is used in many tracking and navigation systems.
DSIF	Deep Space Instrumentation Facility.
DSN	Deep Space Net
Dynamic Envelope	A surface defining a volume that provides adequate clearance accounting for displacements occurring as a result of vibrations.
Earlybird	A Hughes-built communication satellite.
Eccentricity	Ratio of distance between foci (of ellipse) to length of major axis.
Ecliptic	The plane of the earth's orbit around the sun, inclined to the earth's equator by about 23° 27'.
Electric Propulsion	The generation of thrust by acceleration of a propellant with some electrical device, such as an arc jet, ion engine, or magnetohydrodynamic accelerator.
Electromagnetic Radiation	Energy (propagated through space or through material media) in the form of an advancing disturbance in electrical and magnetic fields existing in space or in the media. Also called simply "radiation."
Ephemeris (pl Empemerides)	A tabular statement of the positions of objects in space at specified intervals of time. A standard yearly reference used by the US is "The American Ephemeris and Nautical Almanac," issued in Great Britain as "The Astronomical Ephemeris."
Epoch	An instant of time or a data selected as point of reference.

Glossary of Terms

Error Analysis	A study or evaluation of cumulative errors inherent of a system or method of operation.
Field Joint	The junction between the OEC and the Voyager spacecraft.
Ground Support Equipment (GSE)	All ground equipment that is part of the complete system and that must be furnished to ensure complete support of the system.
GSFC	Goddard Space Flight Center, a branch of NASA located in Greenbelt, Maryland.
Guidance	The process of directing the movement of an astronautical vehicle or spacecraft, with particular reference to the selection of a flight path or trajectory.
Guillotine	A pyrotechnic device used to sever a line or cable.
Gyro	A device utilizing the angular momentum of a spinning rotor to sense angular motion of its base about one or two axes at right angles to the spin axis. Also called "Gyroscope."
HAC	Hughes Aircraft Company.
Heliocentric	Orbiting about the sun as a central body.
HS	The prefix given to Hughes Satellites standing for Hughes Space System Division.
Human Engineering	The art or science of designing, building, or equipping mechanical devices or artifical environments suitable to the anthropometric, physiological, or psychological requirements of the men who will use them.
Hybrid Network	Network employing 3 db couplers with output signals in phase quadrature, for power division, isolation, and phasing to antenna elements.
Hydrazine	A propellant, N_2H_4.
Inclination	The angle by which the orbital plane of an object in space is inclined to the plane of reference (usually the equator in geocentric work, or the ecliptic in heliocentric work).
Injection	The process of putting an artificial satellite into orbit. Also the time of such action.

Glossary of Terms

Interface	A common boundary between one component, system, etc., and another. Used especially when these require mating.
Ion	An atom or molecular group of atoms having an electric charge. Sometimes also a free electron or other charged subatomic particle.
Isotropic	Having the same properties in all directions.
Jet	A nozzle or thruster assembly.
Lander	A capsule intended to survive entering a planet's atmosphere and land on the planet for the purpose of scientific investigation.
Launch (noun)	The process or action of sending off or placing into dynamic flight an aerospace vehicle, probe, or the like. It is a process performed upon a vehicle, requiring a time period of some duration. In this respect, it differs from a liftoff which is performed by the vehicle and occurs at a particular instant in time.
Launch Vehicle	Any device that propels and guides a spacecraft into orbit about the earth or into a trajectory to another celestial body. Often called "booster."
Launch Window	An interval of time during which a rocket can be launched to accomplish a particular purpose.
Limit Cycle	A periodic oscillation with fixed amplitude and frequency.
Line of Apsides	Line between periapses and apoapsis.
Line of Nodes	The intersection of an orbit plane and a reference plane.
Magnetic Cleanliness	Pertaining to the minimizing of the magnetic properties of a system; techniques used to insure low magnetic fields for a spacecraft.
Magnetopause	The outer boundary of the areomagnetic cavity.
Magnetosphere	The areomagnetic cavity.
Mean Anomaly	The angle through which an orbiting body would move in a specified period of time if it moved at it's mean angular rate.

Mega	A prefix meaning multiplied by one million as in "megacycles."
Modular	A technique of construction or design utilizing compartmentation or building blocks.
Modulation	Alteration of amplitude or frequency of a wave in accordance with input signal variations.
Module	A self-contained unit of a spacecraft serving as a building block for the overall structure.
Multiplexer	A mechanical or electrical device for sharing of a circuit by two or more coincident signals.
Nodes (ascending and descending)	Points of intersection of an orbit with the reference plane (usually ecliptic or equator).
Nose Fairing	A jettisonable covering, or shroud, usually about the payload and/or other upper portion of a space vehicle, designed to reduce aerodynamic drag and to protect the enclosed volume from aerodynamic heating and loading during passage through the atmosphere.
Nutation	Torque free motion of the spin axis about the angular momentum vector is termed "free precession" or nutation.
Nutation Damper	A passive device that dissipated nutation energy via fluid viscosity.
Occultation	State of being hidden from view by intervention of celestial body.
OEC	Orbital Experiment Capsule.
Orbit Change System	Referring to a mode of operation between the OEC and Voyager where the OEC can inject itself into orbits different from the basic Voyager orbit, thereby allowing occultation to occur.
Oxidizer	A rocket propellant component, such as liquid oxygen, nitric acid, fluorine, and others, which will support combustion when in combination with a fuel.
Parameters	A set of quantities defining a system.

Glossary of Terms

Payload	The portion of an aerospace vehicle designed specifically to house and transport cargo, scientific instrumentation, and ancillary equipment, the exclusive purpose of such portion being to accomplish the mission objectives. Also, the weight of such portion of the vehicle.
Periapsis (perifocus)	An orbiting body's point of nearest approach to the surface of the central body.
Perihelion	The point of nearest approach to the sun of a solar orbit.
Perturbation	A disturbance in the regular motion of a body, as the result of an additional force to those causing the regular motion.
Photon	According to the quantum theory of radiation, the elementary quantity or "quantum" of radiant energy.
Pinpusher	A pyrotechnic device producing a thrust on a piston rod shaft to actuate a mechanism or impart an impulse.
Plasma	An electrically conductive gas comprised of neutral particles, ionized particles, and free electrons, but which, taken as a whole, is electrically neutral.
Precession	Motion of the angular momentum vector under a constantly applied torque is termed "forced precession" or precession.
Proton	A positively-charged subatomic particle having a charge equal to the negative charge of the electron, but of 1837 times the mass; a constituent of all atomic nuclei.
Radial Jet	The thruster or nozzle assembly of a propulsion system oriented normal to the spin axis of a satellite.
Real Time	The theoretically simultaneous acquisition of the knowledge of an event and the occurrence of the event. Because simultaneity is not possible in practice, the interval between occurrence of the event and the receipt, by the requesting person, of the knowledge of the event, must be no greater than that of the fastest electronic communications means.
Recommended OEC Configuration	The OEC configuration evolved during the feasibility study best suited to satisfy overall mission goals.

Glossary of Terms

Regression	Orbital plane changes of an artificial satellite about the polar axis of a planet due to the ablateness of the planet.
Reliability	Fractional probability of accomplishing all of the functions required for success of a given task or mission, within a specified time.
Restraint	A physical, mechanical, electronic, metallurgical, thermal, or other limitation placed upon a design or an action under which the possibility of following certain specified approaches or procedures is denied.
Roll	The movement of a missile or aerospace vehicle about its longitudinal axis.
Satellite	An attendant body that revolves about another (primary) body; especially in the solar system, a secondary body, or moon, that revolves about a planet. A man-made object revolving about a celestial body, such as a spacecraft orbiting about the earth.
Separation	The action, time, or place that the OEC is disconnected from the Voyager.
Separation Plane	That point or series of points (usually thought of as a plane) at which separation occurs.
Shock Front	The outer boundary of the transition region external to the magnetopause.
Shutter	A device used in active thermal control techniques to change the exposed surface properties of radiating surface by means of slits covering a surface.
Solar Array	An arrangement of photovoltaic cells used as a primary power source on space vehicles.
Solar Constant	The rate at which solar radiation is received on a surface perpendicular to the incident radiation and at the earth's mean distance from the sun, but outside the earth's atmosphere. $G = 442$ Btu/hr ft^2 or 1.94 cal/min cm^2.
Solid Phase Welding	Cold welding, the adhesion or cohesion of bare (oxide-free) metals in a vacuum.
Spacecraft	An artifical body designed to transport a payload and operating essentially or exclusively outside the earth's atmosphere.

Glossary of Terms

Specific Impulse	Impulse content per unit weight of propellant.
Spin-table	A mechanism to which an object can be mounted and spun up to a desired speed.
Spinup	The act of spinning a vehicle by means such as thrusters.
Squib	A small electrical pyrotechnic device used to fire the igniter in a rocket, or for some similar purpose. Not to be confused with a detonator.
Stationkeeping	Maintaining a fixed position relative to the body being orbited.
Sterilization	Application of methods aimed at the attainment of biological sterility.
Substrate	The supporting structure of a solar panel.
Sun-line	A line between a space vehicle or celestial body and the sun.
Superinsulation	Alternate layers of reflective material and insulation in near-vacuum, assembled around an object to inhibit heat flux to or from the object.
Syncom	Hughes-built communication satellite employed in synchronous earth orbit.
System	An arrangement of entities or equipment especially integrated to perform a specific function or functions, e.g., propulsion system, guidance system, ground support system, flight control system, major system.
Telemetry	The science of measuring a quantity or quantities, transmitting the measured value to a distant station, and there interpreting, indicating, or recording the quantities measured.
Terminator	The line dividing the illuminated and unilluminated part of a planet's disk.
Thrust	The amount of driving force, measured in pounds, exerted on a missile or space vehicle by its jet or rocket engines or other propulsive force.

Glossary of Terms

Tracking	The process of following the movement of a satellite or rocket by radar, radio, and/or photographic observations, generally for the purpose of recording its trajectory or for improving the reception of signals from the body.
Trajectory	The path traced through space by a space vehicle, probe, or the like, which is propelled the entire distance, or part of it; a flight profile.
Transit	The non-powered phase of the trajectory between injection and the target, or between injection and the initiation of the terminal phase of the maneuver. Incourse (midcourse) maneuvers usually occur during the transit portion of the flight.
True Anomaly	An angle in the orbit plane measured from the perifocus to the object in the direction of motion of the object.
Umbilical	Any one of the servicing, controlling, or testing electrical lines between the OEC and Voyager.
Vis-viva Energy	The energy at injection which is designated C_3. This energy shall be defined as: $$C_3 = V^2 - \frac{2GM}{r}$$ where V is the velocity (relative to the geocentric); r is the geocentric distance at injection; and $GM = 3.986032 \times 10^{14}$ meters^3/sec^2.
Voyager	The interplanetary spacecraft to be launched by a Saturn V vehicle to orbit the planet Mars.
Whip Antennas	Quarter wavelength unipole radiation.
Zero-twist Spring	A precision ground type of spring fabricated to provide accurate pure compression.

11.0 NOTATIONS

SYMBOLS AND ABBREVIATIONS

a	Semi-major axis of ellipse
$\bar{a}$	Unit vector in spin axis direction
Ag	Silver
C	Centigrade temperature scale (degrees)
Cd	Cadmium
cm	Centimeter
D	Separation distance
db	Decibel
D_t	Tube inside diameter
F	Force; also Farenheit temperature scale (degrees)
FF	Fraction filled
g	Local acceleration of gravity
GN_2	Gaseous nitrogen
h	Altitude from surface
h_a	Apoapsis altitude
h_p	Periapsis altitude
hr	Hours
i	Orbital inclination to Martian equator
i_e	Orbital inclination to ecliptic

i_p	Angle between orbital plane and terminator plane
I	Mass moment of inertia
I_{sp}	Specific impulse
I_{ω}	Spin axis moment of inertia
km	Kilometer
kw	Kilowatt
ℓ	Length
Lb	Pound (mass)
Lb_f	Pound (force)
$\vec{m}$	Unit vector from spacecraft to Mars center
M	Mass
min	Minute
mlb	Millipound
mv	Millivolt
MHz	Megahertz
Ni	Nickel
P	Orbit period
R	Rankine temperature scale (degrees); also reliability; also resistance
R_m	Mars radius
rad	Radians
RF	Radio frequency
$\vec{S}$	Unit vector from spacecraft to sun
t	Time

T	Torque; also duration
V	Velocity
X	Body axis; also distance
Y	Body axis
Z	Body axis
Zn	Zinc
$\sim$	Proportional sign
[ ]	Dimension
$\vert\ \vert$	Numerical equality without regard for sign
$\approx$ or $\rightarrow$	Approaching equality
$a > b$	a greater than b
$a < b$	a less than b
∞	Infinite

GREEK LETTERS

α	Angle between separation velocity vector and orbit velocity vector; also Mars-OEC-sun angle
β	Angular separation of sensors
Γ	Angular position on Voyager
γ	Flight path angle
δ	Infinitesimal increment
Δ	Finite difference
ϵ	Nutation angle; also cant angle between sensor beams; also angular misalignment
η	Angle between spin axis and sensor optical axis

θ	Attitude error angle; also angle between separation vector and orbit plane
μ	Gravitational constant of Mars
ν	True anomaly
π	3.1416
ρ	Density
σ	Stress
φ	Angle between spin axis and spacecraft sunline
ψ	Sun sensor pulse designation
ω	Spin rate