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FINAL REPORT
VOYAGER SPACECRAFT
PHASE B, TASK D

VOLUME I

SUMMARY

VOYAGER

GENERAL ELECTRIC

GPO PRICE $ ______

CFSTI PRICE(S) $ ______

Hard copy (HC) 3.00

Microfiche (MF) .65

ff 653 July 65

FACILITY FORM 602

N68-19100 (ACCESSION NUMBER) | 1 (THRU)
(PAGES) | (CODE)
CR-61661 (NASA CR OR TMX OR AD NUMBER) | 31 (CATEGORY)

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16 OCTOBER 1967

FINAL REPORT
VOYAGER SPACECRAFT
PHASE B, TASK D

VOLUME I

SUMMARY

PREPARED FOR

GEORGE C. MARSHALL SPACE FLIGHT CENTER

UNDER MSFC CONTRACT No. NAS8-22603

GENERAL ELECTRIC
MISSILE AND SPACE DIVISION
Valley Forge Space Technology Center
P.O. Box 8555 • Philadelphia , Penna.

VOLUME SUMMARY

The Voyager Phase B, Task D Final Report is contained in four volumes. The volume numbers and titles are as follows:

Volume I	Summary
Volume II	System Description
Book 1	Guidelines and Study Approach, System Functional Description
Book 2	Telecommunication
Book 3	Guidance and Control Computer and Sequencer Power Subsystem Electrical Subsystem
Book 4	Engineering Mechanics Propulsion Planet Scan Platform
Book 5	Design Standards Operational Support Equipment Mission Dependent Equipment
Volume III	Implementation Plan
Volume IV	Engineering Tasks
Book 1	Effect of Capsule RTG's on Spacecraft
Book 2	Applicability of Apollo Checkout Equipment
Book 3	Central Computer
Book 4	Mars Atmosphere Definition
Book 5	Photo-Imaging

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SECTION 1
INTRODUCTION

This volume summarizes General Electric's final reports covering a preliminary design update for a 1973 Voyager mission, a preliminary implementation plan, and five engineering studies related to the Voyager mission. The scope of the General Electric Company effort is summarized as follows:

a. Report on update and optimization of a preliminary spacecraft design for the Voyager 1973 mission; including adaptability and utilization growth to perform 1975-1977-1979 Mars missions.

b. Report on development of a preliminary implementation plan to accommodate the 1973 Voyager mission.

c. Perform Voyager Spacecraft related engineering study tasks and report in the following areas:

   1. Determine the spacecraft design provisions necessary to accommodate the radioactive thermoelectric generators in the Surface Laboratory System for a 1975 Mars mission.

   2. Investigate the applicability of the Apollo Checkout Equipment for the Voyager program.

   3. Investigate the feasibility of a central computer system to replace separate spacecraft systems to accomplish onboard checkout, sequencing, guidance and control, and spacecraft science automation.

   4. Establish theoretical concepts of the Martian atmospheric processes to provide definition required by spacecraft designers.

   5. Investigate the suitability of various photo-imaging systems for use on the Voyager Spacecraft.

Figure 1-1 illustrates the recommended baseline Voyager spacecraft configuration resulting from our systems update studies. The major results of the scope of General Electric studies are as follows:

a. The 1973 Voyager baseline preliminary design has major improvements in science payload which in turn lead to increased data transmission capability and solar array power. Modularity, reliability and adaptability to 1970 decade Mars missions has also been improved. Table 1-1 illustrates some of the basic system update features.

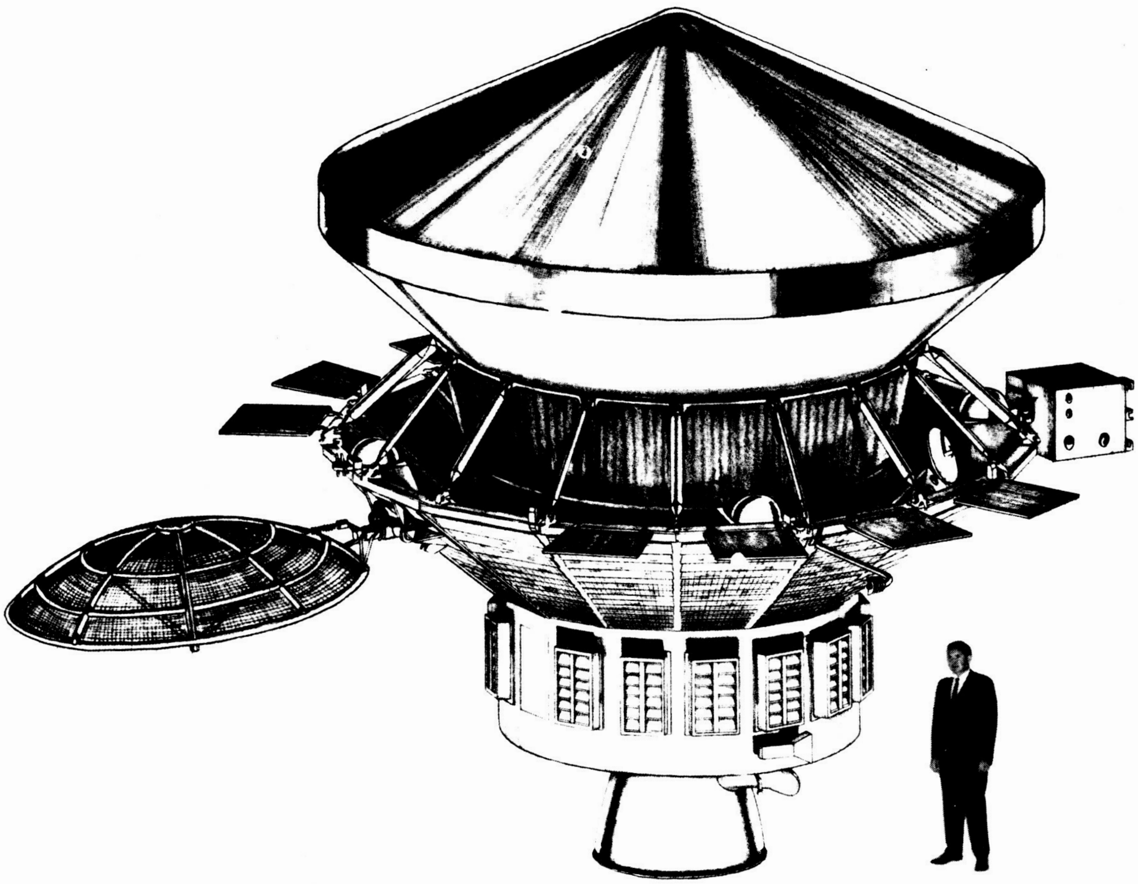

Figure 1-1. Baseline Voyager Spacecraft Configuration

Table 1-1. Comparison of Major Features of Current Spacecraft to Previous Design

	(1971 Mission) Task B	(1973 Mission) Task D
Telecommunications		
High Gain Antenna Diameter (Feet)	7.5	9.5
RF Power Output (Watts)	50	50
Error Control Coding	No	Yes
Data Storage Capacity (Bits)	$6x10^8$	$2.4x10^9$
Maximum Transmission Rate (Bits/Sec)	15,000	40,500
Propulsion		
Orbit Insertion	Solid Rocket	Lunar Excursion Module Descent Engine (LEMDE)
Trajectory Corrections	Monopropellant	LEMDE
Power		
Batteries	Silver Cadmium + Silver Zinc	Nickel Cadmium + Silver Zinc
Solar Array Power (Watts)	650	840
Guidance and Control		
Stabilization References	Sun-Canopus	Sun-Canopus
Reaction Control	High Pressure Cold Gas	High Pressure Cold Gas
Onboard Control (Computer and Sequencer)	Cycled, Special Purpose Computer	Cycled, Special Purpose Computer

b. Preliminary implementation plans indicate that high confidence can be maintained to meet schedule requirements for a 1973 mission. Major facilities needed to produce and test the Voyager Spacecraft either exist or are under development for other programs currently under contract. Further study of NASA/KSC facility requirements and availability is needed.

c. It is basically feasible to plan for the application of radioisotope thermoelectric generators in the 1975 Voyager Surface Laboratory System.

d. Significant portions of Apollo Checkout Equipment-S/C are technically feasible for application to Voyager checkout and launch control.

e. The application of a spacecraft central computer system concept does not appear to be warranted over "separate subsystems" for early Voyager missions application.

f. The results indicate that during a period of high solar activity, Martian atmosphere density at 1,000 kilometers is likely to be three orders of magnitude greater than that derived from the Mariner IV flyby experiments.

g. At least three candidate photo-imaging systems appear to provide suitable methods for enabling the Voyager mission to perform advanced photo-imaging experiments of the Martian surface.

The work reported within this over-all study includes a major part of the Company's effort in preparing for Phase C program competition for the Voyager Spacecraft. Appendix A of this document gives a summary listing of the documents produced under contract NAS8-22603, along with additional reports developed by General Electric and made available to the Marshall Space Flight Center (MSFC).

SECTION 2
SPACECRAFT TECHNICAL SUMMARY

2.1 MISSION OBJECTIVES AND DEFINITION

The Voyager Mars objectives encompass a 1973 mission to (a) place science instruments in orbit about Mars, (b) place science instruments on the surface of Mars, (c) conduct observations of Martian phenomena with these instruments for specified periods of time, and (d) transmit the results of these observations to Earth. To accomplish these objectives, the 1973 mission includes the primary functional areas shown in Figure 2-1. Flight hardware for the mission consists of the launch vehicle system, a Saturn V, carrying two identical planetary vehicles. The planetary vehicle consists of a flight spacecraft and a flight capsule. The flight capsule in turn consists of a capsule bus and a surface laboratory. This flight hardware is shown in Figure 2-2. The launch operation system conducts the launch from Complex 39 at KSC. The mission operation system together with the tracking and data acquisition system provides the ground based control of the flight operation.

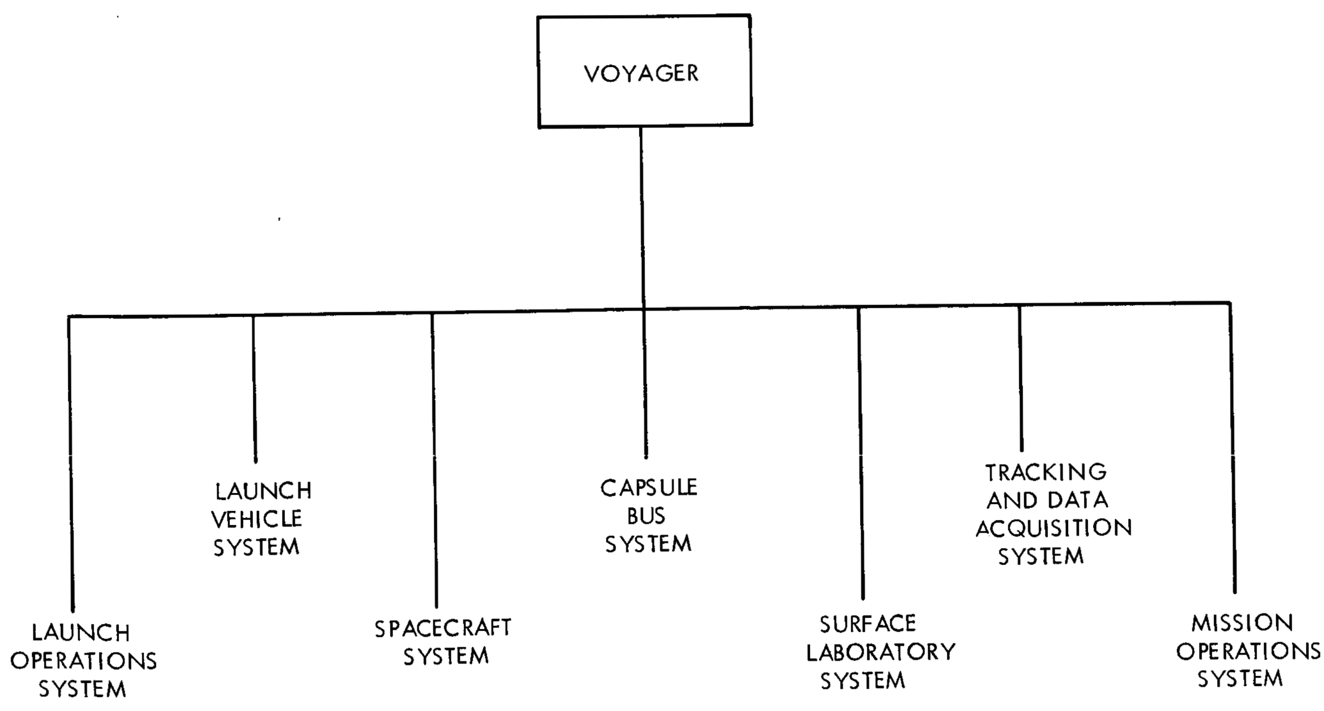

Figure 2-1. Voyager Functional Areas

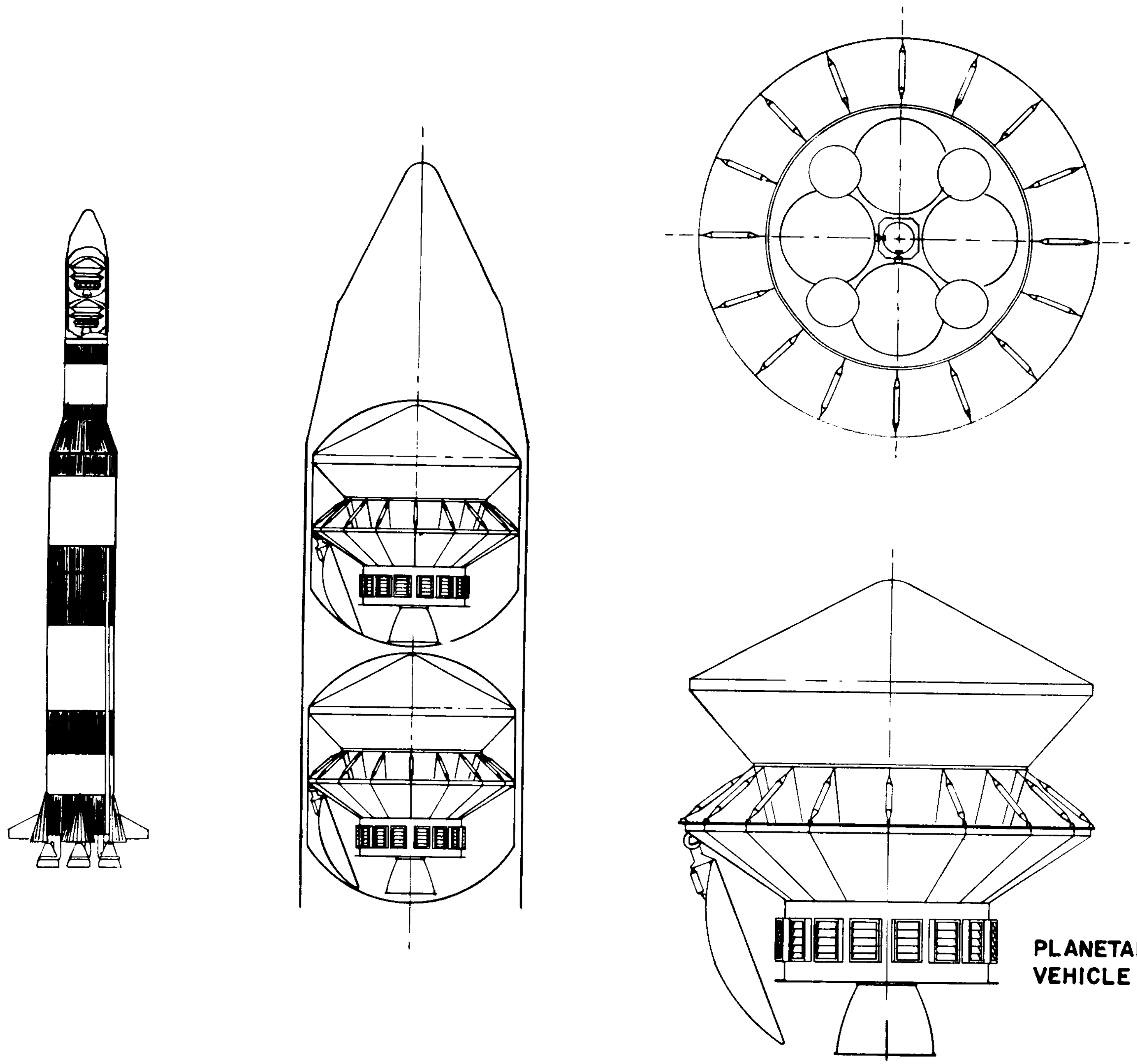

Figure 2-2. Spaceflight Hardware

2.1.1 SPACECRAFT ROLE

The role of the spacecraft within this total mission is two fold:

a. Science Mission - It conducts the orbiting science mission, using instruments such as imaging devices, spectrometers, and radiometers. It must provide the necessary support to these instruments to enable them to acquire the desired data. This includes pointing, sequencing, power and thermal control. It must also provide for transmission of the acquired data back to Earth.

b. Capsule Support - The spacecraft serves as a bus to deliver the flight capsule into orbit about Mars. It supports the flight capsule until separation occurs in orbit by supplying power, sequencing signals, and by transmitting capsule telemetry data. After separation, the relay equipment mounted in the spacecraft relays the capsule de-orbit and entry data back to Earth. Landed data could also be transmitted via this link if desired.

2.1.2 CONSTRAINTS AND GUIDELINES

The basic requirements imposed on the design of the Voyager Spacecraft are specified in the "General Specification for Performance and Design Requirements for the 1973 Voyager Mission," dated January 1, 1967. Mission constraints which have a major impact on the design are:

a. Schedule - The 1973 Mars opportunity places an absolute time constraint on the schedule. This, in turn, demands a design that (1) is a conservative use of flight proven hardware and technology to avoid development difficulties, and (2) allows a maximum amount of parallel fabrication and test to allow early detection and correction of faults.

b. Mission Duration - To accomplish a successful 1973 mission requires a spacecraft operating life in excess of one year. Use of flight proven technology in the design is also of benefit for meeting this requirement. In addition, the design must insure that the entire spacecraft can be adequately tested to detect and remove all defects prior to launch.

c. Planetary Quarantine - The probability of contaminating Mars by Voyager space vehicle borne earth organisms must be kept extremely low. To prevent contamination by the spacecraft, the design, manufacture, and handling of the vehicle must insure a high degree of cleanliness of the external surfaces. In addition, reliable performance must be achieved in the execution of trajectory corrections. Orbit parameters must also be appropriately selected.

Several additional guideline documents have been issued by MSFC which modify or add to the mission specification for purposes of the system update. Major changes affecting the design compared to the Task B design effort carried out in late 1965 are as follows:

a. 1973 Opportunity - The 1973 Mars opportunity is less favorable than the 1971 opportunity in two respects. First, the Earth-Mars distance at encounter has increased from approximately 110 million kilometers for 1971 to 170 million kilometers or greater for 1973. This reduces the capability of a given communication link by about a factor of 2. In addition, Mars is more nearly at aphelion in 1973 so that solar flux is reduced somewhat, reducing the amount of power available per square foot of solar array.

b. Deep Space Network Capability - The capability of the deep space network as defined in JPL Engineering Planning Document 283, revision 2, is more conservatively specified than it was in Task B. The worst case antenna gain and worst case system noise temperature combined result in a reduction of communication capacity of a factor of 2 for a given spacecraft design.

c. Spacecraft Propulsion - The spacecraft propulsion system is based on use of the lunar module descent engine for the system update. The Agena engine and the Titan transtage engine are potential alternates. Use of a liquid propulsion system for orbit insertion instead of the solid rocket proposed by General Electric has a marked effect on the system design. The system configuration changes quite drastically as well as the dynamic properties of the vehicle from a control point of view.

d. Flight Capsule - Flight capsule weights defined for this study vary from 5,000 to 7,000 pounds compared to 3,000 pounds for the 1971 mission. Two cases have been considered: Case A in which the capsule weight is constant at 5,000 pounds, and Case B in which the capsule weight is 6,000 pounds in 1973 and 7,000 pounds thereafter.

e. Magnetic Cleanliness - The requirement for magnetic cleanliness of the overall planetary vehicle has been removed. This allows greater freedom in making design choices that favor the use of flight proven hardware to maximize reliability.

In addition to changes imposed by the change in mission year and changes in guidelines, the system was also updated to reflect changes in technology that have occurred since the Task B design in late 1965. Examples of this are the inclusion of error control coding and improved data storage capability, both of which will be discussed later. Finally, results from the Task C studies carried out during the period of April 1966 through July 1967 have been incorporated where appropriate.

2.2 SPACECRAFT CONFIGURATION

The baseline configuration resulting from the system update is shown in Figure 2-3. In arriving at this configuration, many configuration concepts were developed reflecting the new guidelines. These were then compared from the standpoint of the mission constraints and competing characteristics listed in Table 2-1. Major reasons for selection of the configuration shown in Figure 2-3 were improved ability to meet the fixed schedule and better reliability assurance.

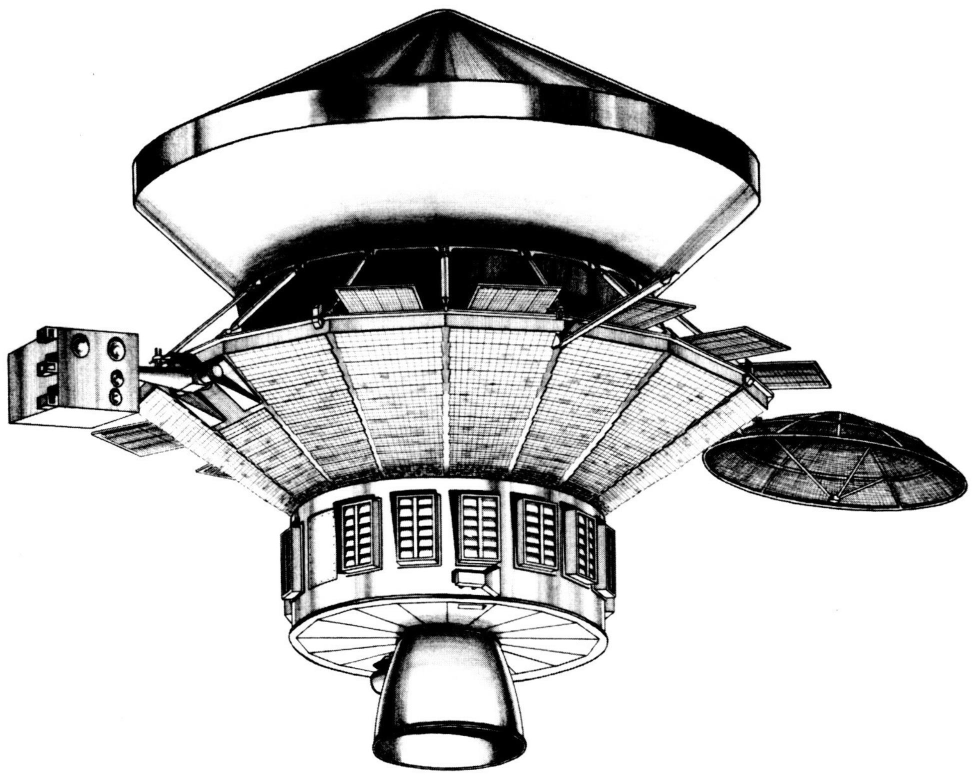

Figure 2-3. Baseline Configuration

The spacecraft consists of three major modules as shown in Figure 2-4.

The propulsion module contains the thrust chamber assembly from the Apollo lunar module, the propellant storage and feed system, and the pressurant storage and feed system. The support module provides the interface between the over-all planetary vehicle and the launch vehicle shroud. Major spacecraft equipments mounted to the

Table 2-1. Configuration Evaluation Criteria

A. CONSTRAINTS

1. Quarantine

- Ability to clean
- Cold gas impact on separating bio-barriers
- Capsule line-of-sight to spacecraft
- Debris caused by moving parts in vicinity of capsule (louvers pinpullers, etc.)

2. Minimum Schedule Risk

- Modularity
- Manufacturing and test schedule contingency
- Assemblability
- Analyzability (to avoid surprises late in the development cycle)
- Logistics
- Development difficulties
- Accessibility to electronics
- Accessibility to propulsion
- Minimum interface interactions

3. LV and Launch Period

- Weight

B. COMPETING CHARACTERISTICS

1. Probability of Success

- Testability
- Thermal performance
- Number of deployments
- Shroud and spacecraft separation
- Autopilot control
- Equipment locations (sensors, attitude control jets, antennas)

2. Perform Mission Objectives

- Antenna size
- PSP viewing
- Solar array power

3. Future Mars Missions

- Interface with capsule RTG
- Adaptable to spacecraft RTG
- Mission flexibility
- Growth in propulsion, antenna, power, PSP

4. Cost

- Design
- Manufacture
- Test
- Compatibility with available facilities
- Logistics

5. Added 1973 Capability

- PSP growth
- Antenna growth

6. Other Planets

- Meteoroid protection for Jupiter mission
- Solar array temperature for Venus mission

support module are the solar array panels, various communication antennas, the attitude control reaction system, and the planet scan platform (PSP) housing the science instruments that require pointing to the planet. The electronic module contains all spacecraft electronic equipment such as supporting electronics for the various science instruments, telecommunications, guidance and control, and electric power.

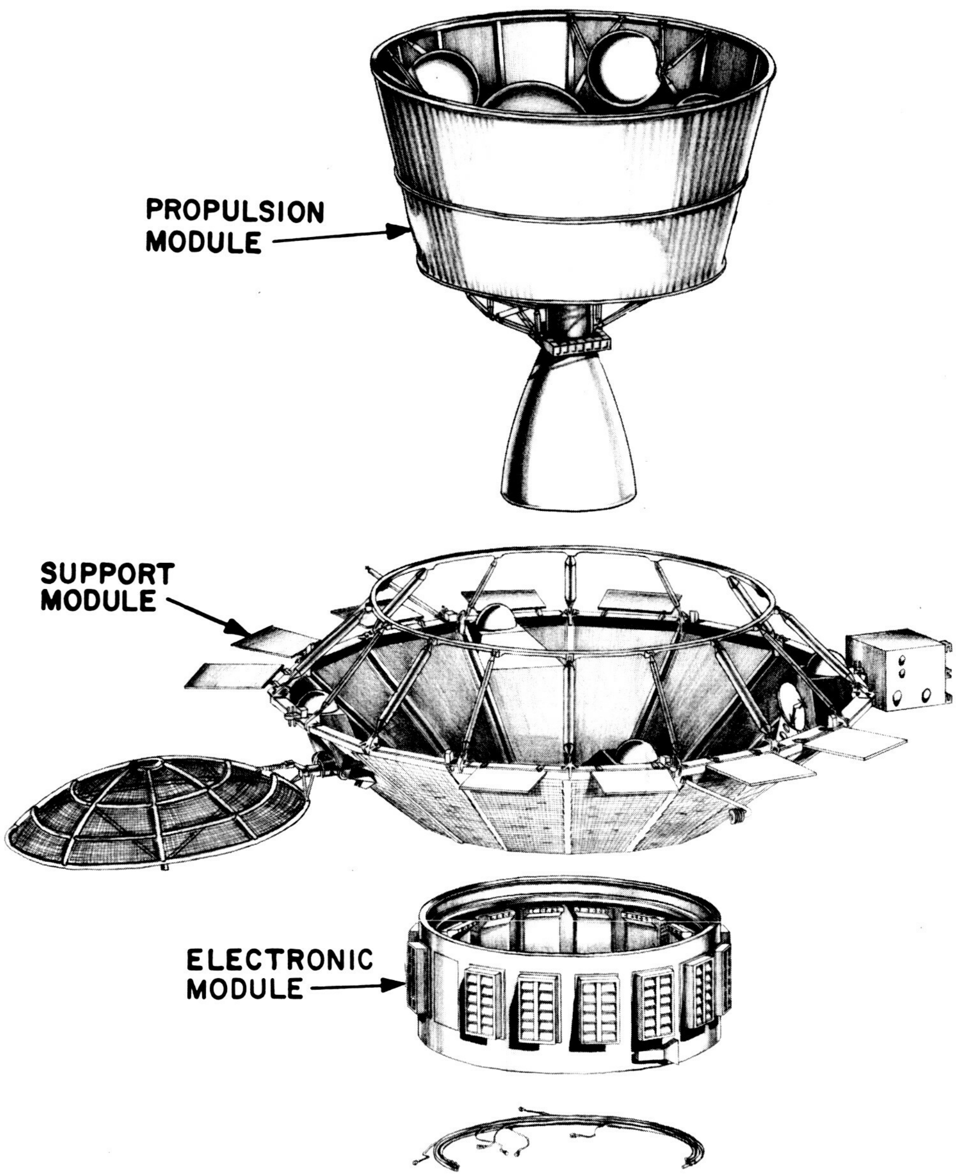

Figure 2-4. Modular Construction

These three modules can be individually assembled and thoroughly tested prior to assembly into a total spacecraft. The interfaces between modules have been kept relatively simple to insure that prior testing is not invalidated at assembly. For example, the propulsion module can be assembled and completely tested and then mated to the support module with only a simple mechanical and electrical connection required. No plumbing is disturbed so that introduction of potential leakage is avoided. This ability to assemble and test major elements in parallel leads to much earlier detection and removal of defects in contrast to a design in which much of the testing must be done in series. Since access is better at the module level, tests can be more thorough, insuring that all equipments are exercised and providing better capability for detection of defects. Either the propulsion module or the electronics module can be mated to the support module first.

Following total system assembly, access to equipment is much better for this configuration than for others considered. This also provides a capability for more thorough testing and allows removal and replacement of defective equipment with minimum disturbance of other hardware.

This configuration is described in detail in Volume II of this report along with other configuration concepts with which it was compared. In addition to modularity and access, key features of the design include the following:

a. PV Support - Location of the support point for the planetary vehicle near the top of the spacecraft greatly eases the problem of separation of shroud and spacecraft compared to a support point near the base of the spacecraft. Minimizing the axial separation between the support and the capsule maximum diameter allows higher tip-off rates without interference during separation. In addition, supporting the total planetary vehicle more nearly at its center of mass leads to better dynamic properties.

b. Propellant Tankage - The propellant tanks have been sized for 1973 through 1979 opportunities assuming growth of the over-all planetary vehicle to the full capability of the Saturn V launch vehicle.

c. Antenna Size - A 9.5-foot antenna can be accommodated within the allowable spacecraft envelope. Increased size is possible with some increase in total planetary vehicle length and weight.

d. Solar Array - Total projected solar array area, both fixed and deployable, is 270 square feet, providing an array output of 840 watts. Increased area is possible with the addition of more deployable panels or utilization of the fixed area at the bottom of the electronic module.

e. Weight - The burnout weight of the spacecraft is 5,299 pounds, With a 7,000-pound flight capsule and sufficient propellants to impart 1.95 kilometers per second velocity change, the total planetary vehicle weight is 24,500 pounds.

2.3 MAJOR TRAJECTORY AND ORBIT CONSIDERATIONS

Many conflicting desires and constraints arise in the selection of trajectories and orbits for a Voyager mission. The ability to achieve the desired capsule landing site, provide suitable conditions for the orbiting experiments, achieve short communication distances and trip times, and the over-all sizing of the propulsion system are involved. In addition, selections must be made considering the overriding constraint of planetary quarantine. Orbital parameters to be discussed in the following sections are indicated in Figure 2-5. They include selection of periapsis and apoapsis altitude, and selection of the orientation of the line of apsides relative to the terminator.

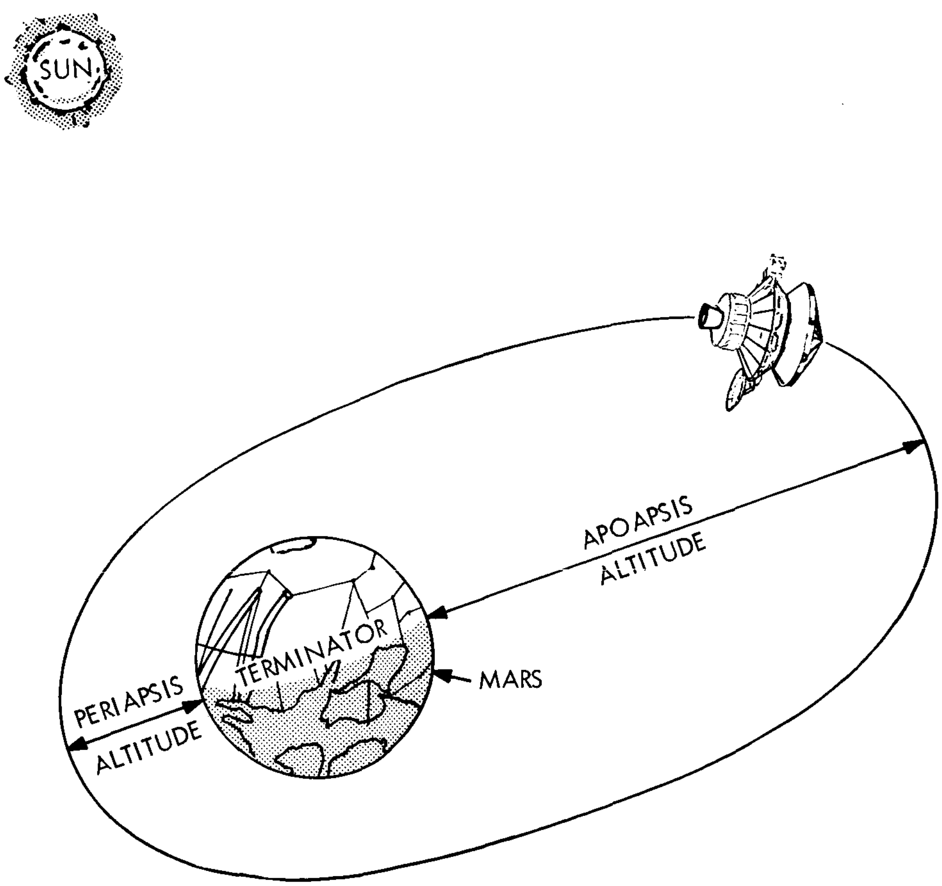

Figure 2-5. Significant Orbit Parameters

2.3.1 PERIAPSIS ALTITUDE

Both from the standpoint of the flight capsule and the orbiting science, a low periapsis altitude is desired. The capsule benefits because of smaller de-orbit propulsion requirements and because of shorter capsule-to-spacecraft communication distances during the period when data is being relayed through the spacecraft. The orbiting science can, of course, achieve higher surface resolution for a given optics design at lower periapsis altitude.

The limitation on minimum periapsis altitude is the quarantine constraint. This constraint can be completely removed only by sterilization of the spacecraft, which imposes intolerable penalties on reliability, cost and schedule. The probability of contaminating Mars by ejecta from the spacecraft surface, e.g. loose particles or micrometeorite spall, as a function of periapsis altitude is indicated on Figure 2-6. Two conditions are shown - one for clean manufacture and one for normal manufacture.

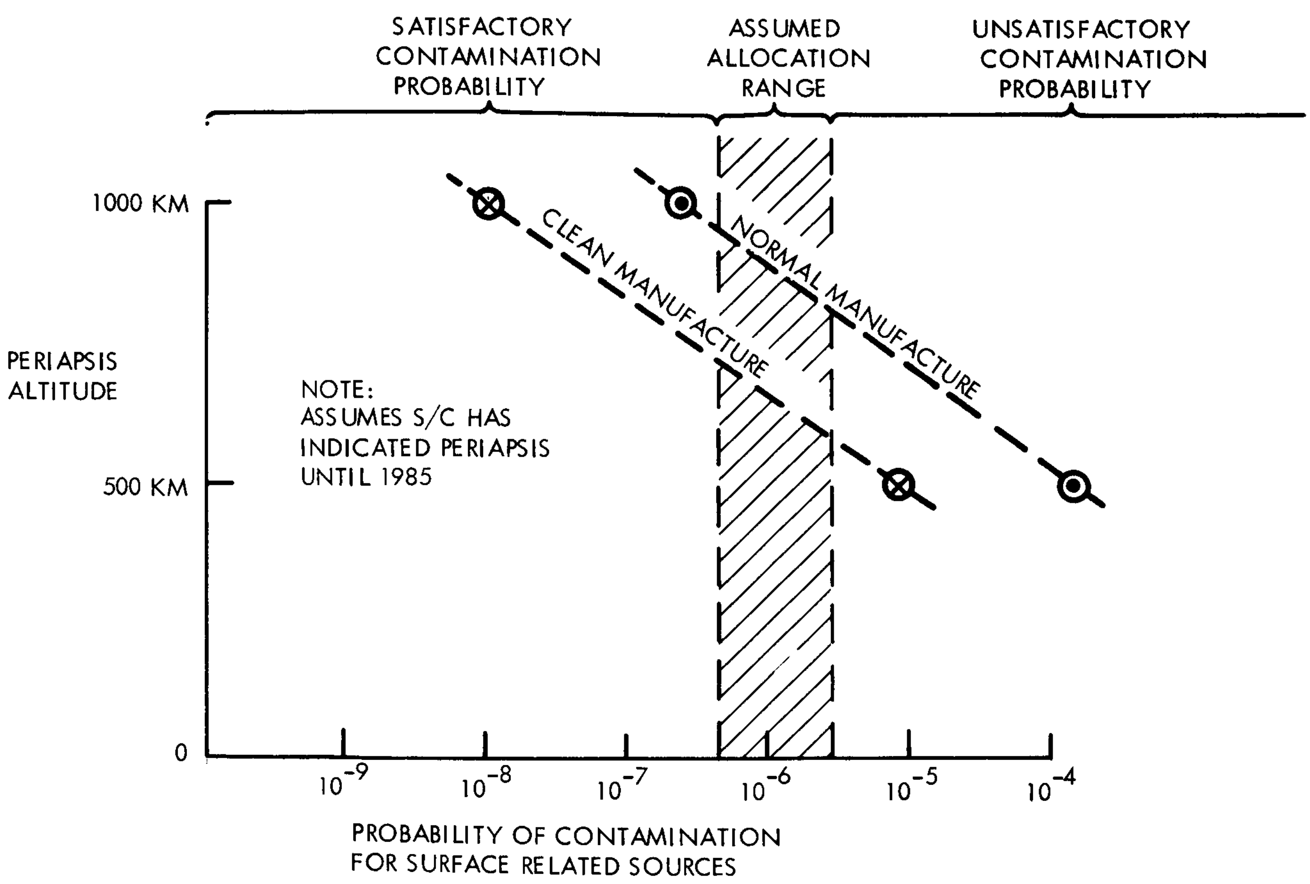

Figure 2-6. Periapsis Altitude Limitation

For the assumed probability to be allocated to this source, a minimum periapsis altitude on the order of 700 kilometers can be tolerated for a clean spacecraft. Guidance analyses have been conducted as described in Volume II which indicate three sigma uncertainties in periapsis altitude of approximately 300 kilometers. Therefore, a nominal periapsis altitude of 1,000 kilometers is recommended at this time.

2.3.2 APOAPSIS ALTITUDE

The effect of apoapsis altitude is somewhat less critical. Lower altitudes, therefore, short orbit periods, are more favorable for mapping coverage in that the available planet surface is overflown in a shorter time period. However, low apoapsis altitudes become quite expensive in terms of propellant requirements as shown in Figure 2-7. From a mapping standpoint, it is probably desirable to have an orbit period which is synchronized with the rate of rotation of Mars to provide contiguous ground surface traces each day or every few days. The number of orbits per Martian day is indicated on Figure 2-7. An orbit period which yields three orbits per Martian day was tentatively selected as being within reasonable propulsion requirements.

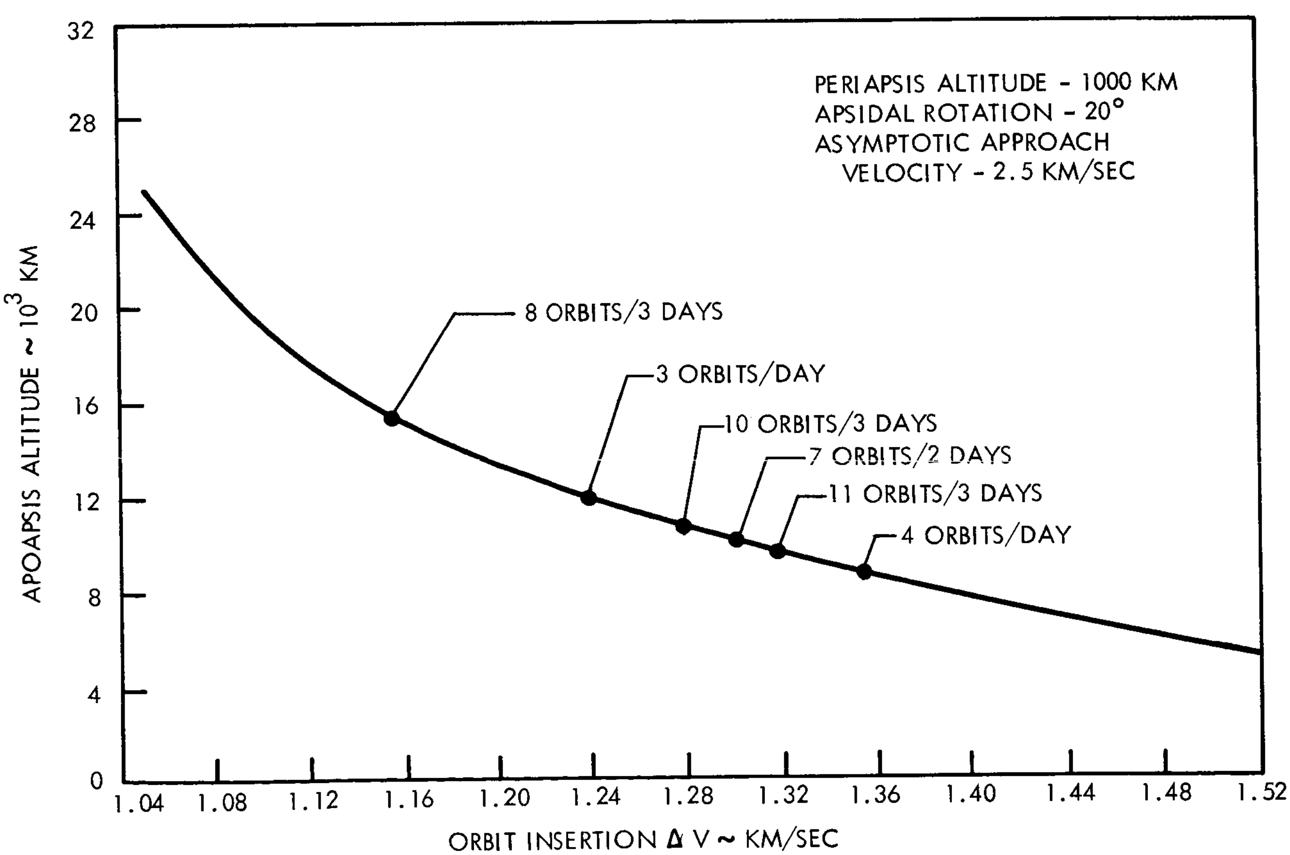

Figure 2-7. Apoapsis Altitude Selections

2.3.3 ORIENTATION OF LINE OF APSIDES

The initial location of the orbit periapsis relative to the terminator is also important. The flight capsule impact area will be near periapsis to minimize de-orbit velocity requirements and communication range during relay operation. For good lighting contrast for descent television pictures, the capsule would prefer a landing site near a terminator. The morning terminator is probably preferable from the capsule standpoint since a full day of operation can be achieved before the first Martian night plus a direct communication link to Earth can be established immediately after impact. The orbiter desires are opposed to this since orbital motion and seasonal progression is such that the periapsis soon moves to the dark side of the planet if it is initially located near the morning terminator. In addition, Figure 2-8 indicates the effect on propulsion requirements of the location of periapsis for various years and trajectory types. Achievement of a morning terminator location is substantially more expensive. A posigrade orbit with periapsis near the evening terminator has been tentatively selected.

Figure 2-8 also indicates that type 1 trajectories are available in all years which satisfy the minimum launch period of 20 days and yield reasonable propulsion requirements. Since type I trajectories are much more advantageous from a standpoint of communication distance and trip time, they have been selected for all years.

The propulsion requirements arising from the trajectory and orbit selection studies, in terms of total impulse, thrust level, and impulse accuracy, can be satisfied by the lunar module descent engine. These studies are reported in greater detail in Volume II of this report.

2.4 ORBITER SCIENCE

The primary motivating force in the design of the Voyager Spacecraft is to return a maximum amount of data to Earth from the orbiting experiments. This requires the spacecraft to operate for a long period of time, but also requires a high rate of return of quality data per unit time.

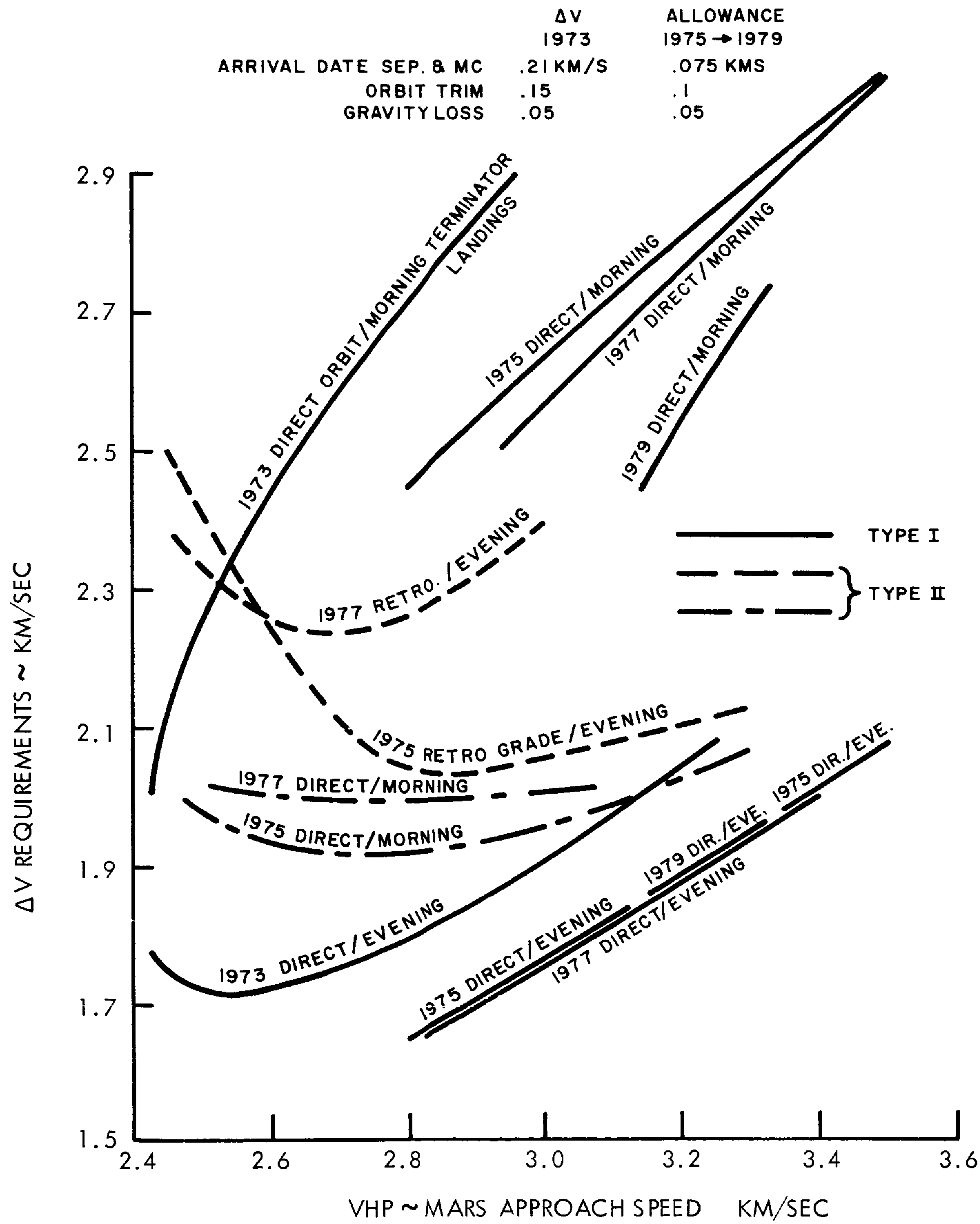

Figure 2-8. Periapsis Location

The baseline science payload used in the system update is shown in table 2-2. This selection agrees with the priorities established by the Space Science Board in their study of Mars exploration. This list of instruments fits within a basic science payload weight of 500 pounds used in this design study. Of the instruments listed, it is felt that most emphasis in 1973 will be placed on the medium resolution photo-imaging experiment whose job it is to map a major portion of the Mars surface at a resolution on the order of 100 meters or better. It is expected that this experiment will place by far the greatest demands on the spacecraft communication capability. The needs of other instruments are discussed in Volume II, and will not be summarized here.

Table 2-2. Baseline Science Payload

1.	Photoimaging
	Medium Resolution TV Camera No. 1
	Medium Resolution TV Camera No. 2
	High Resolution TV Camera
2.	High Resolution IR Spectrometry
3.	Broadband IR Spectrometry
4.	IR Radiometry
5.	UV Spectrometry
6.	Radio Occultation
7.	Celestial Mechanics

2.4.1 DATA TRANSMISSION

The basic communication capability achieved with a spacecraft, for a fixed ground station design, is a function of the transmitted power and the antenna size provided.

Increasing the antenna size incurs weight increases in the antenna itself, its deployment and pointing mechanism, and in the basic vehicle stabilization system due to increased solar pressure unbalance torques and increased pointing accuracy required. Increasing the transmitted power costs weight in the solar array, the thermal control system, and in the radio subsystem itself.

During the system update, an optimization study was performed to determine the most effective method of using weight to increase the communication link capacity. The results of this work are shown in Figure 2-9. The differential weight associated with a change in the power-gain product is shown. At several points on the curve, the optimum antenna size and transmitter power are indicated for a given power-gain product. From a configuration and over-all weight standpoint, as discussed in Volume II, an antenna size of 9.5 feet and a transmitted power level of 50 watts has been selected for the baseline system. With this selection, the data rate achieved as a function of time is shown in Figure 2-10. The lower link capability curve indicates the performance realized under worst case conditions, i.e., all tolerances at their maximum negative value. Under this condition, a data rate of 40,500 bits per second can be accommodated for up to 30 days depending upon the specific encounter date selected. Then as the Earth-Mars distance increases, the data rate is halved for the next 60 days and then halved again to 10,125 bits per second. Assuming a February 1 encounter date, the total data returned is 3 x 10^{11} bits for a 6-month mission.

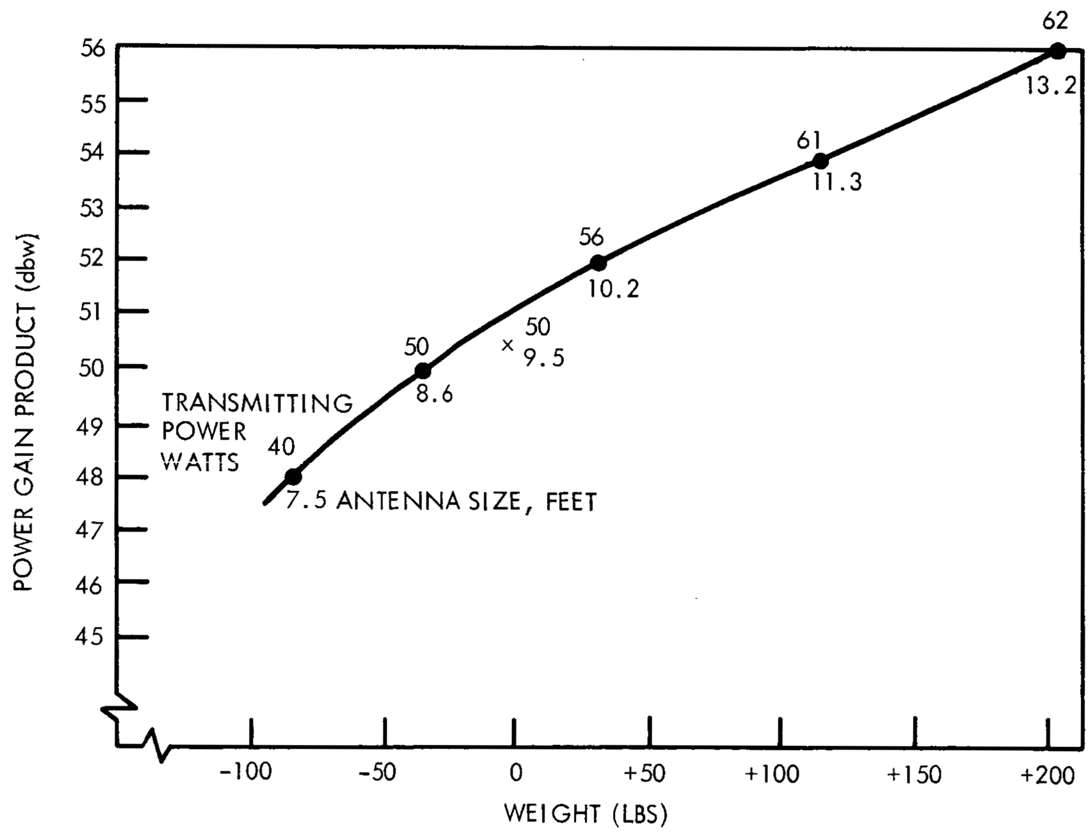

Figure 2-9. Optimum Effective Radiated Power

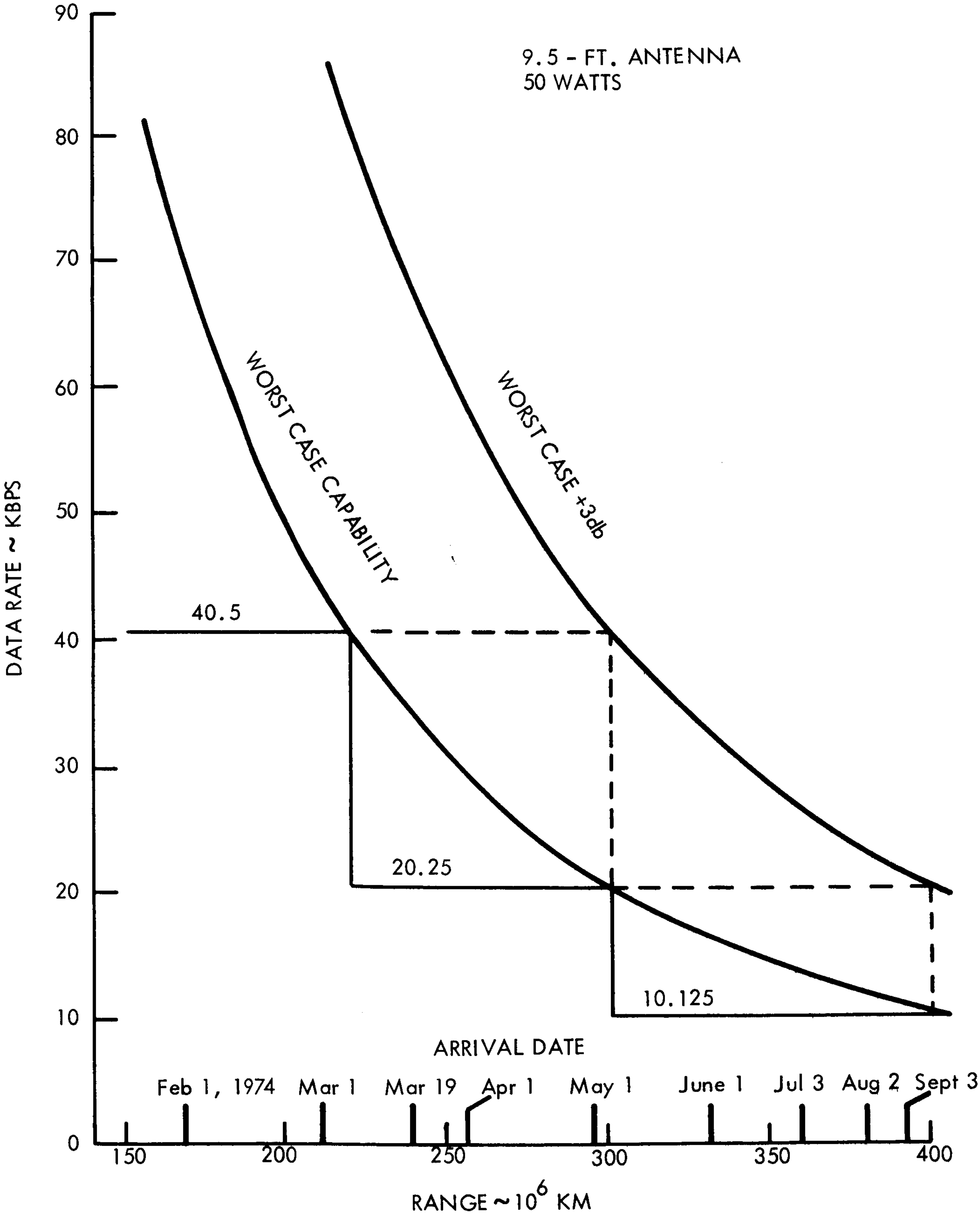

Figure 2-10. Communication Capability Versus Time

It is probable that not all tolerances will be at their maximum negative value. A more likely performance is perhaps indicated by the upper curve of Figure 2-10 which is 3 decibels better than worst case. (The sum of the negative tolerances is approximately 4 decibels.) In this case, the 40,500 bits per second rate can be sustained substantially longer. For the same encounter date of February 1, total data return for a 6-month mission is 4.8×10^{11} bits.

The capability shown in Figure 2-10 assumes efficient utilization of the link. This involves many factors, one of which is the use of error control coding of the data. Based on laboratory work performed by General Electric MSD since 1965, which demonstrates the feasibility of such systems, the use of error control coding has been included in the design. This increases the communication rate for a given power gain product by about a factor of 2 and causes a very slight increase in spacecraft complexity.

2.4.2 DATA STORAGE

The bulk of the science data, from the photo-imaging instruments, is collected over a relatively small portion of the orbit when the spacecraft is at low altitude and the lighting conditions are appropriate. To efficiently use the data rate capability of the spacecraft, this data must be stored and then read out over the total orbital period. To match the 40,500 bit per second capability and the nearly 8-hour orbital period, a total storage capacity in excess of 10^9 bits is required. A total of 2.4×10^9 bits is provided in the baseline system. This is supplied in the form of four recorders: two 1.2×10^9 bit recorders for use with the photo-imaging instruments and two 3.6×10^7 bit recorders for the other instruments. This provides both flexibility and redundancy in the conduct of the scientific mission.

A significant parameter related to the data storage is the achievable "read in" rate Since the photo-imaging experiment produces higher quality data at low altitudes, it is desirable to take pictures rapidly near periapsis. The Vidicon cameras shown in the baseline payload would like read-out rates up to 5×10^6 bits per second. Achieving this rate with state-of-the-art magnetic tape recorders does not appear feasible. Both analog and digital recording techniques have been examined. A rate of 3.9×10^5 bits per second is felt to be achievable with digital machines and 60 khz with analog recording. This limits the amount of surface mapped in a given pass, or the resolution achieved. As described in paragraph 4.5, a useful mapping mission can be conducted within this limitation. Either of these recorders represents some extension of the state-of-the-art and hence, early development work is indicated in this area.

Use of a photographic film system rather than TV cameras and magnetic tape avoids the problem of read-in rate and can provide the necessary capacity for the mission. However, the photo-imaging study (paragraph 4.5) shows several disadvantages for film. Primary problems are (a) difficulty of obtaining intensity resolution and accuracy for the low contrast expected at Mars, and (b) long exposure times which will require image motion compensation that might be avoided with TV. For these reasons, the baseline science payload shows vidicon TV cameras.

2.4.3 INSTRUMENT LOCATION AND POINTING

In addition to returning the data accumulated by the science instruments, the spacecraft must mount those instruments and provide appropriate operating conditions. This requires provision of suitable environments, such as thermal and EMI, and provision of the requisite power and sequencing signals. A major problem is the provision of the necessary viewing requirements and stability during data acquisition.

The basic spacecraft is 3-axis stabilized in celestial space, with two axes using the sun as a stabilization reference and the third axis using Canopus. From this basic platform, the high gain antenna is articulated in two axes to maintain orientation to Earth. The science instruments which must view Mars over a large portion of the orbit period are mounted in the Planet Scan Platform (PSP) indicated in Figure 2-11. This platform has 3-axis articulation relative to the spacecraft with two axes being used to erect a normal to the orbit plane, and the third axis providing rotation about the orbit normal to align the instruments along the Mars local vertical in the normal mode of operation. Maintaining one axis normal to the orbit plane is accomplished open loop by knowledge of the spatial geometry. The third axis control is accomplished by use of a Mars horizon sensor.

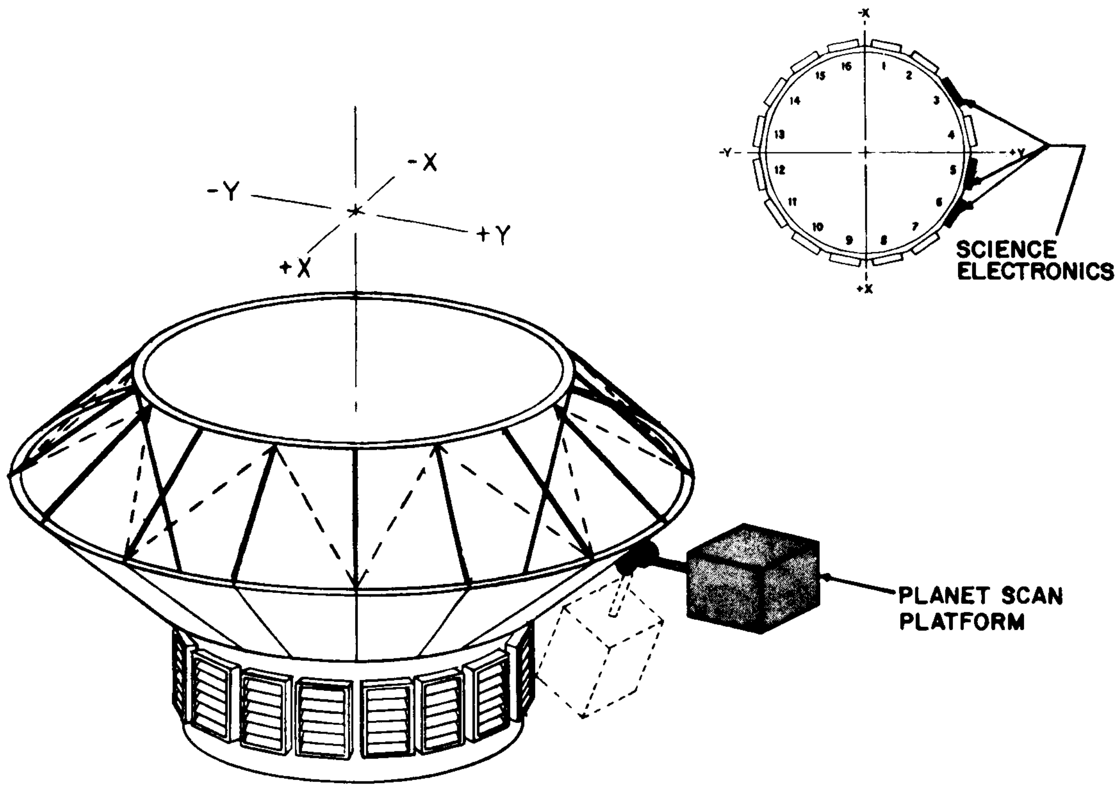

Figure 2-11. Science Equipment Location

Proper location of the PSP and the length of the boom separating it from the spacecraft was a major study during the system update. The range of initial orbits that may be selected and the change in spatial geometry as a function of time after injection makes continuous viewing of the planet without obstruction from some portion of the spacecraft very difficult. Typical results of the study carried out are indicated in Figure 2-12. The curve to the right indicates the percent of all possible orbits which have unobstructed viewing for a specific location of the PSP as a function of boom length. The middle curve indicates the degree of improvement if a second Canopus sensor is used to allow 180 degrees rotation of the spacecraft when viewing is obstructed. The curve on the left indicates the improvement achievable if the aft portion of the capsule bio-barrier, which normally remains with the spacecraft, is removed to prevent obstruction from that item.

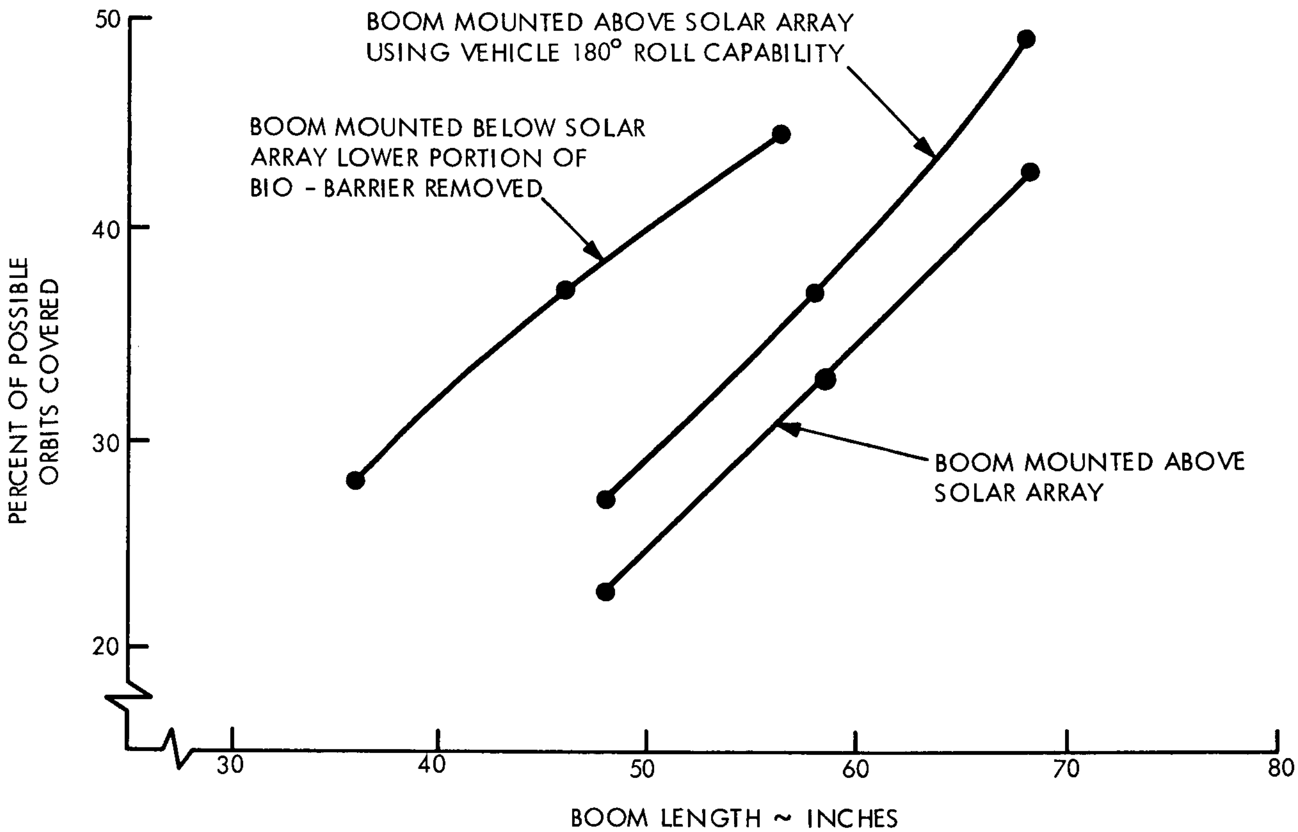

Figure 2-12. PSP Location Trade-Off Study

From such studies, a baseline design was selected which satisfies the 1973 nominal orbit and provides a reasonable degree of flexibility. The 1973 case is shown in Figure 2-13. The shaded portion of the figure indicates the range of cone and clock angles allowed for the orbit plane normal without viewing obstruction. The trajectory plotted within this shaded area shows the motion of the nominal 1973 orbit for a one-year mission. Rotation of the spacecraft is not required nor is removal of the capsule bio-barrier.

The stability provided by the pointing system is such that angular motion during an exposure period is less than 3×10^{-6} radians. This will allow 10-meter resolution to be achieved with acceptable smear degradation.

2.5 SPACECRAFT OPERATION CONTROL

Control of the flight operation for an interplanetary mission is different in several respects from near Earth mission. The major factor is the round trip communication time for radio signals which may be typically 20 minutes at encounter and increases to nearly 45 minutes at maximum Earth-Mars distance. Primarily

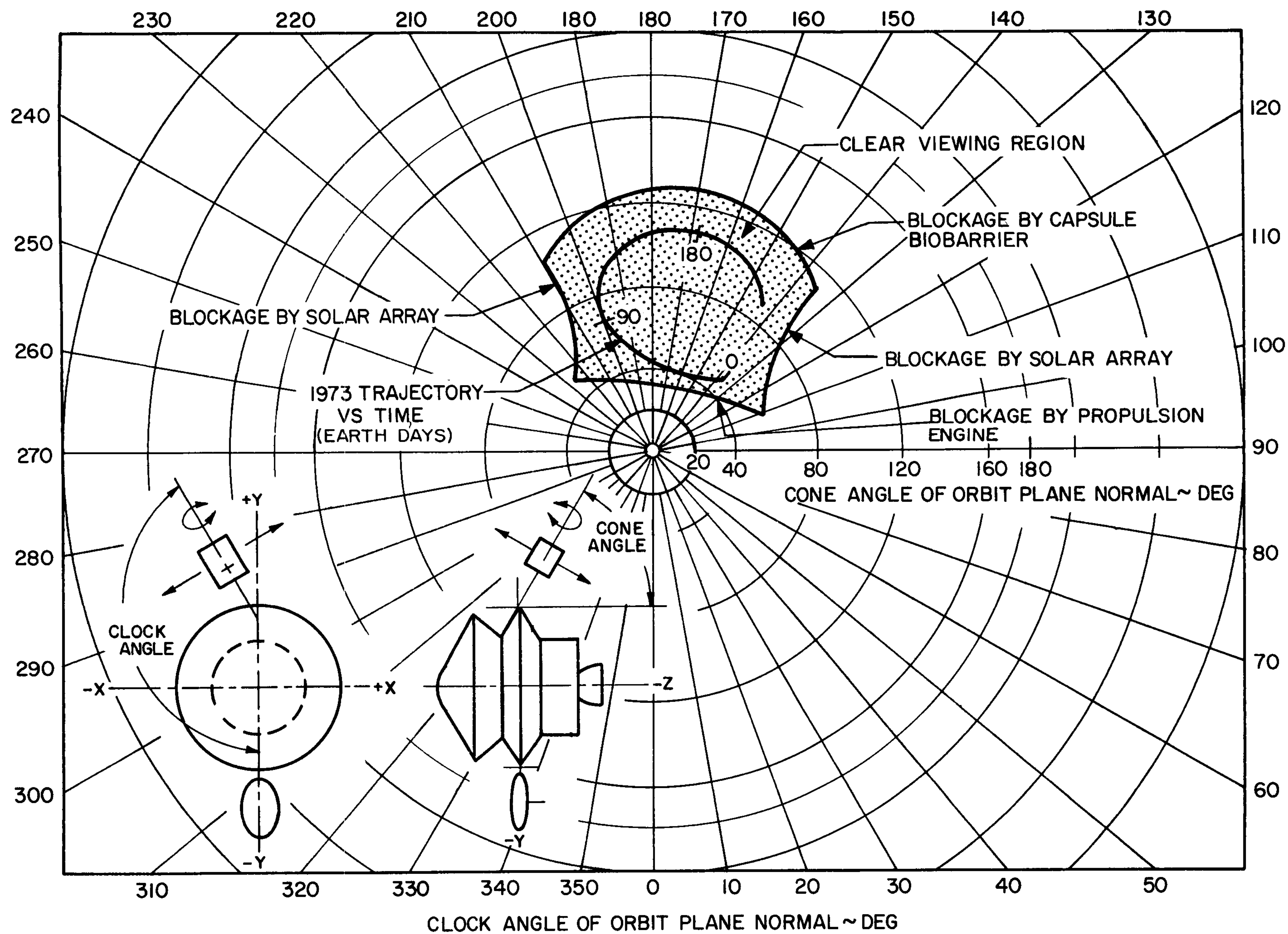

Figure 2-13. PSP Viewing for 1973

because of this delay, it is desirable to make the spacecraft as automatic as possible and use the ground control in a trim mode - to alter programmed sequences as required or to initiate alternate modes of operation in the event of anomalies. This basic operational philosophy and the design implementation of it is essentially unchanged from Task B.

2.5.1 ONBOARD CONTROL

The computer and sequencer provides basic control of the spacecraft operation throughout the mission. It controls such functions as antenna deployment and pointing, deployment of solar paddles, initiation of capsule separation, and pointing of the open loop axes of PSP. It also controls maneuvers and trajectory corrections using information transmitted by ground command specifying the required thrust direction and impulse magnitude. The computer and sequencer is designed with a high degree of reliability to insure that false commands will not be issued and legitimate commands will not be dismissed. As an example, the design is such that no single piece part failure will cause failure of a critical command. Also, the computer and sequencer can tolerate momentary loss of spacecraft power without loss of memory.

2.5.2 GROUND CONTROL

Adequate engineering telemetry data is transmitted to the ground station to allow an assessment of spacecraft performance at all times. This data, together with tracking data and the data returned from the science instruments, provide the basis from which the flight operations personnel assess the mission progress and take action when necessary. They can update programmed sequences stored in the computer and sequencer to accommodate differences in the trajectory and orbit from that planned. They can provide back-up capability for any command issued by the computer and sequencer to essentially accomplish manual control of the mission. Hardware failures which are not immediately catastrophic to the mission can be diagnosed via the telemetry link and corrective action, such as switching in a redundant unit, can be accomplished by ground command. For other failures, such as power system equipment, the round trip communication time cannot be tolerated and onboard sensing and switching is provided.

Command capability is provided throughout the mission except for periods when the spacecraft is occulted from Earth by Mars. Any one of the three frequency addressable command receivers and detectors can be selected. One of the three receivers is connected to the high gain antenna and the other two are connected to lower gain broader coverage antennas.

2.5.3 INSTRUMENT SEQUENCING

Sequencing of individual science instruments must be handled in a way that minimizes problems when instruments are changed, either within a given mission or from mission to mission. The data automation subsystem (DAS), which collects and formats the science data, also provides the detailed sequencing signals. It receives gross timing information from the computer and sequencer, such as periapsis passage or terminator crossing signals, and generates specific commands to the instruments. The DAS sequencing can also be modified by ground command based on the return from various science instruments to maximize the value of the mission. Using the DAS for this function avoids the need for modification of the C&S each time the science changes.

2.6 CONCLUSIONS AND RECOMMENDATIONS

At the conclusion of this system update, some problems still remain. Significant ones are:

a. The effect of motion of the large quantities of liquid propellants on the attitude control system is still not adequatedly understood. Therefore, a basic question still exists regarding the adequacy of the control system. Further analytical work is required in this area, and perhaps a useful flight test can be defined to be carried out as part of AAP or some other flight program.

b. A summary weight statement for the over-all system is shown in Table 2-2. It can be seen that for the later missions, all current guidelines - 20-day launch period, 7,000-pound capsule, spacecraft velocity capability of 1.95 km/sec, and 5,000-pound project contingency - cannot be achieved with the baseline spacecraft. This is not deemed a serious problem since these missions can be performed using less than 1.95 km/sec. Table 2-3 indicates the effect of reduction in velocity capability, launch period, capsule weight, or project contingency on the weight picture. The spacecraft system weight can, of course, be reduced, but at the expense of reduced capability, e.g., science payload, communication capability, reduced margins, or less redundancy.

c. The mission is currently somewhat limited by the achievable read-in rates of reliable tape recorders. This problem should be attacked in an effort to develop more capable data storage methods.

Several positive improvements have been made in the spacecraft design as a result of the current update. Major improvements are:

1. A larger science payload has been accommodated, along with increased power to support this science and its data storage.

2. The data transmission capability of the spacecraft has been improved over the Task B design by a factor of 5. This is attributable to antenna diameter increase from 7.5 to 9.5 feet and to improved transmission efficiency. The chief factor in improving the efficiency is the addition of error control coding.

3. The data storage capacity has been increased from 6×10^8 bits to 2.4×10^9 bits to match this increase in transmission capability.

4. The spacecraft configuration, as previously discussed, has significant advantages over the Task B design. Chief among these are improved modularity and accessibility which provide better ability to manufacture and test the vehicle, thus insuring the required delivery date and the required mission life.

5. The spacecraft is more readily adaptable to all Voyager Mars missions from 1973 through 1979. Propellant tanks and basic structure have been sized to accommodate the largest planetary vehicle that will fit within the Saturn V capability. The solar array has been sized to provide a minimum of 835 watts with Mars at aphelion. Most later Mars missions will not encounter this condition in their first six months, so added power is available. Also, in later missions, the capsule will undoubtedly be powered by radioisotope thermoelectric generators and will not require 200 watts from the spacecraft power source early in the mission. A very similar spacecraft configuration was studied during Task C and it was determined that the spacecraft solar power system could be replaced by RTG's for future missions with a minimum of modifications.

Table 2-3. Planetary Vehicle Weight Summary

	Case A		Case B			
	1973	1977	1973	Reduced Velocity 1977	Reduced Launch Period 1977	Reduced Contingency 1977
Spacecraft Dry Weight	5,013	5,013	5,038	5,038	5,038	5,038
Unuseable Propellant	261	261	276	276	290	290
Spacecraft Burnout Weight	5,274	5,274	5,314	5,314	5,328	5,328
Spacecraft Contingency	264	264	266	266	266	266
Flight Capsule	5,000	5,000	6,000	7,000	7,000	7,000
Velocity Capability (km/sec)	1.95	1.95	1.95	<u>1.835</u>	1.95	1.95
Useable Propellant	9,970	9,832	10,956	10,824	11,750	11,750
Separated Planetary Vehicle Weight	20,508	20,370	22,536	23,404	24,344	24,344
Planetary Vehicle Adapter	126	126	126	126	126	126
Planetary Vehicle Launch Weight	20,634	20,496	22,662	23,530	24,470	24,470
Total Launch Payload (2 Planetary Vehicles)	41,268	40,992	45,324	47,060	48,940	48,940
Project Contingency (2 Planetary Vehicles)	5,000	5,000	5,000	5,000	5,000	3,120
Launch Period (Days)	30	24	26	20	<u>18</u>	20

SECTION 3
IMPLEMENTATION PLAN

Volume III of this report is a preliminary implementation plan for the Voyager Spacecraft program leading to a 1973 launch. The plan contains a description of major test programs required to develop spacecraft subsystems and the over-all spacecraft system, and to integrate the spacecraft with other interfacing Voyager systems. The hardware and test facilities necessary for these test programs are defined, and a top level schedule for the over-all program is developed. The current version of the plan concerns itself exclusively with these major test activities and does not discuss other critical elements needed for a successful project. These elements, such as development of standards, program controls, and quality assurance, will be discussed in the Phase C proposal.

Development of the implementation plan considered the following major factors:

a. The effect of the fixed launch date.
b. The long life requirements.
c. The quarantine restrictions.
d. Program cost.
e. The maximum utilization of existing facilities.
f. Handling and transportation.
g. Integration of the spacecraft with other Voyager systems.

The following paragraphs summarize briefly the key features of the test program, the over-all schedule, and the major facility picture for the spacecraft project.

3.1 TEST PROGRAM

The over-all test program consists of three major phases. These phases and their objectives are:

a. Development Test - Develop designs which will perform the required function with adequate margin under the operating environments that exist.

b. Qualification Test - Demonstrate that the design and the process by which the flight hardware will be produced are satisfactory for meeting the mission requirements.

c. Flight Acceptance Test - Provide maximum assurance that existing and potential defects that could cause flight failures have been removed from the flight hardware.

3.1.1 DEVELOPMENT

To accomplish the objectives of the development phase, basic hardware elements (such as circuits, black boxes or actuators) will be tested to determine their performance capability as a function of the severity of a given environment (such as temperature or input voltage level). In addition, tests at higher levels (subsystem and system) will be conducted to determine the environments that each basic element will see throughout the total mission. Generally, the following testing is required:

a. Breadboard testing of all electronic equipment, and early model testing of mechanical and electromechanical items.

b. Testing of pacakged hardware elements as components, as part of a subsystem, and as part of the over-all system.

c. Testing of representative system models to determine the operating conditions seen by all equipment under major environments such as acoustic noise, EMI, thermal vacuum, and propulsion firing.

d. Development of interface compatibility between flight hardware systems, such as spacecraft-to-capsule and spacecraft-to-launch vehicle, by combined testing with appropriate models.

An integral part of development testing is development of a compatible interface between the spacecraft and its operational support equipment.

3.1.2 QUALIFICATION

Qualification testing is carried out at two basic hardware levels. First, major hardware elements (such as electronic assemblies, thrust chamber assemblies, Canopus sensors) are qualified as end items. Typical hardware elements that are qualified at this level are identified in Figure 3-1. Second, the over-all spacecraft system is subjected to qualification tests.

The rationale for this two-level qualification program is two-fold:

a. Potential problems are uncovered earlier by testing major assemblies, since they can be available in advance of the over-all system. Problems, if they occur, can generally be resolved more quickly and cheaply at this level.

b. The major assemblies can be subjected to a somewhat more severe environment than they might see as part of the overall system qualification. This provides some assurance that spacecraft-to-spacecraft variances will not allow a potentially inadequate assembly to survive qualification and subsequently fail in a flight unit.

In addition to these two basic qualification levels, critical elements mounted in a major assembly will be qualified individually in addition to being qualified as part of the assembly. This will include such items as gyros, tape recorders, batteries, and rf power amplifiers. This is done for the same reasons as cited above.

3.1.3 FLIGHT ACCEPTANCE

Experience at GE-MSD has demonstrated that the adequacy of ground testing of flight hardware is a key element in the success of a spacecraft program. This testing must be thorough and must subject the flight hardware to critical environments as severe as those encountered in flight to provide adequate assurance that defects in the hardware have been removed.

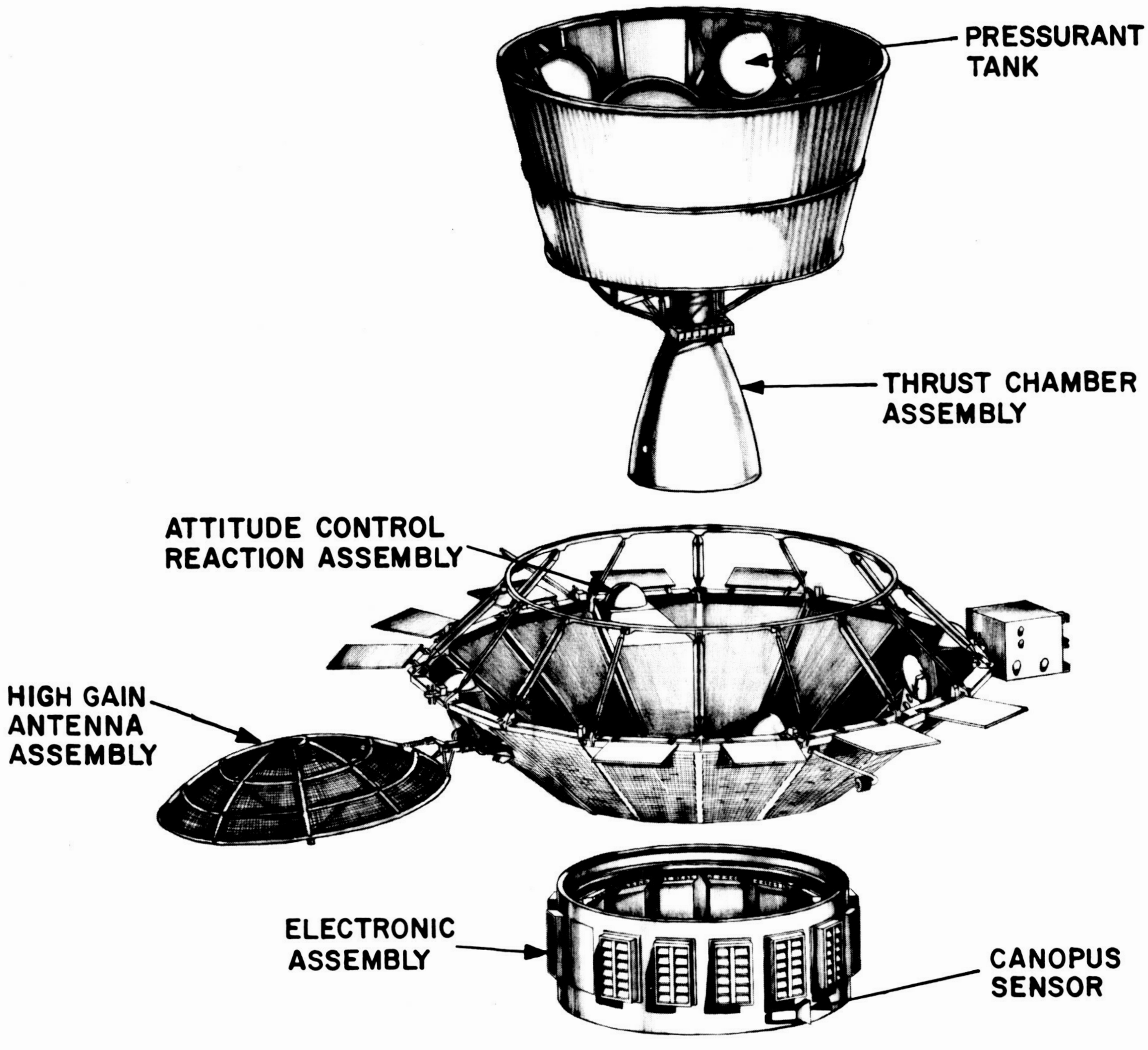

Figure 3-1. Hardware Qualification Level

This practice was followed in the successful Nimbus and Mariner Programs and General Electric has accumulated considerable statistics in support of this approach. One classified program, operated from 1964 to 1967, included a significant number of flights of the same design and had varying acceptance test requirements throughout its life. Vibration acceptance tests were conducted on the early spacecraft system flight units - stopped - and then reinstated on two other occasions. Thermal-vacuum acceptance testing was also conducted on a number of the later flight units.

3.2 PROGRAM SCHEDULE

Development of an optimum schedule for a program such as Voyager with the dual requirements of a fixed launch date and over 15 months of operation required in space is indeed difficult. Development of this schedule must answer such questions as:

a. How much time should be allowed for the basic development phase before entering into the qualification program? A too short development cycle will produce an inadequate design with many problems during qualification. A long development phase minimizes the probability of problems during qualification, but entails the risk that a major problem could be encountered uncomfortably late in the program.

b. How much overlap between phases should be allowed? For example, how much of the system qualification should be completed before start of assembly of the flight units ?

c. How much contingency time should be provided and how should it be allotted? For example, should all contingency time be assigned to the end of the program with planned delivery of the flight spacecraft several months early? Or should several intermediate milestones be firmly established with contingency time provided for each? The former allows more time for thorough testing of the flight article if the contingency time is not used. while the latter focuses management attention more strongly on earlier events in the program.

The overall allocation of time currently recommended by GE-MSD is shown in Figure 3-2. The system definition phase (Phase C) is assumed to extend for a period of 9 months. This phase produces CEI specifications, part I, for all equipment. In addition, development hardware efforts are initiated in critical areas, some of which are discussed in Volume III. The major design reviews indicated in the figure are defined as:

a. System Design Review (SDR) - A review resulting in the definition of the over-all configuration.

b. Preliminary Design Review (PDR) - A review of the system specifications and requirements which will be the baseline for detailed design.

c. Hard Design Review (HDR) - A review of the design at the end of the packaging and detailed design effort.

d. Critical Design Review (CDR) - A review of the design at the end of the developmental phase.

e. First Article Configuration Inspection (FACI) - A formal inspection of flight-type hardware used in the qualification tests.

The SDR and the HDR are additional reviews recommended by General Electric.

A more detailed schedule of the activities during the program is shown in Figure 3-3. This figure shows the time allotment for test programs being conducted during the development, qualification, and flight acceptance phases. Most of the test programs shown are discussed in some detail in Volume III.

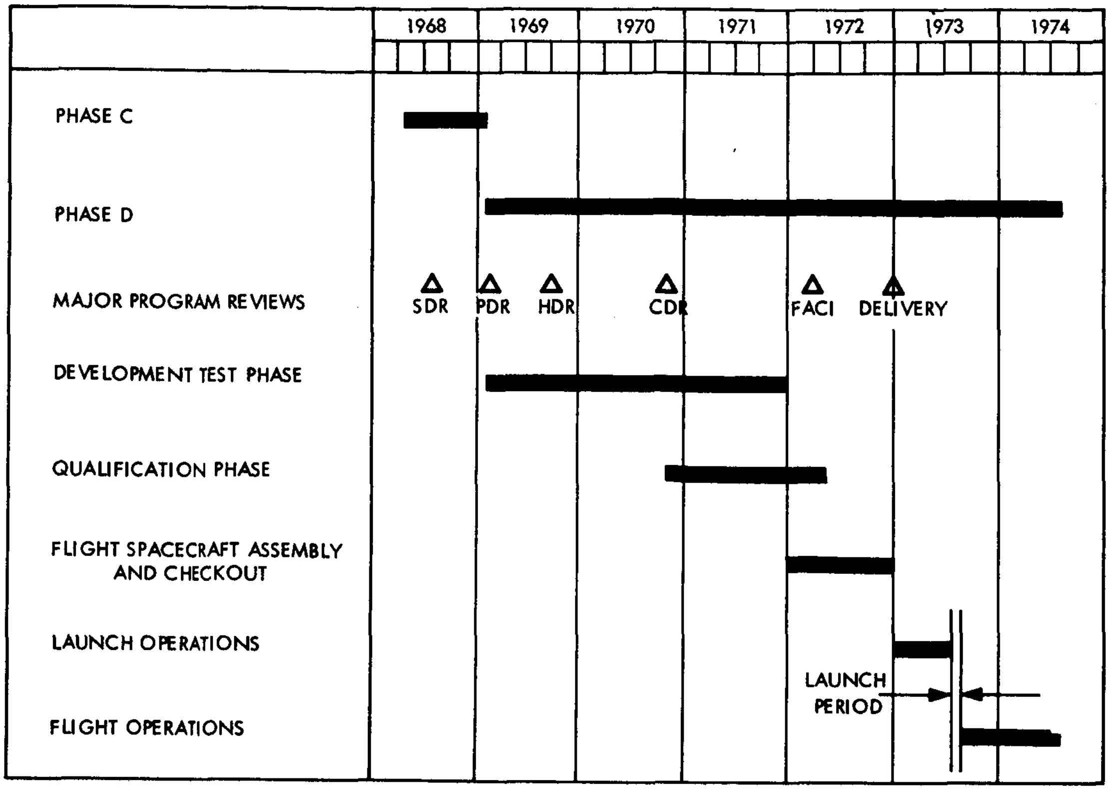

Figure 3-2. Over-all Program Phasing

In developing this schedule, particular attention was paid to those subsystems involving major test hardware items. These included development of the structure, propulsion, and thermal control subsystems, each of which is discussed in Volume III. To generate this schedule, each required step in each major test program was defined to the level of detail shown in Figure 3-4, the test sequence for the engineering development model.

3.2.1 SCHEDULE ASSURANCE

Confidence in adhering to this schedule, and hence meeting the fixed launch date, comes from the following considerations:

a. The probability of time consuming development difficulties is minimized by a design which makes maximum use of flight proven hardware and technologies. A preliminary assessment of the General Electric Voyager design in this regard is shown in Figure 3-5. The small number of items requiring new development is of major importance to insuring the schedule will be met.

b. A maximum amount of parallel testing is planned in the development phase of the program to provide the earliest possible achievement of a design to meet the mission requirements. For example, three packaged units of each hardware element will be fabricated to allow essentially parallel testing at the black box, subsystem, and system levels. This allows early

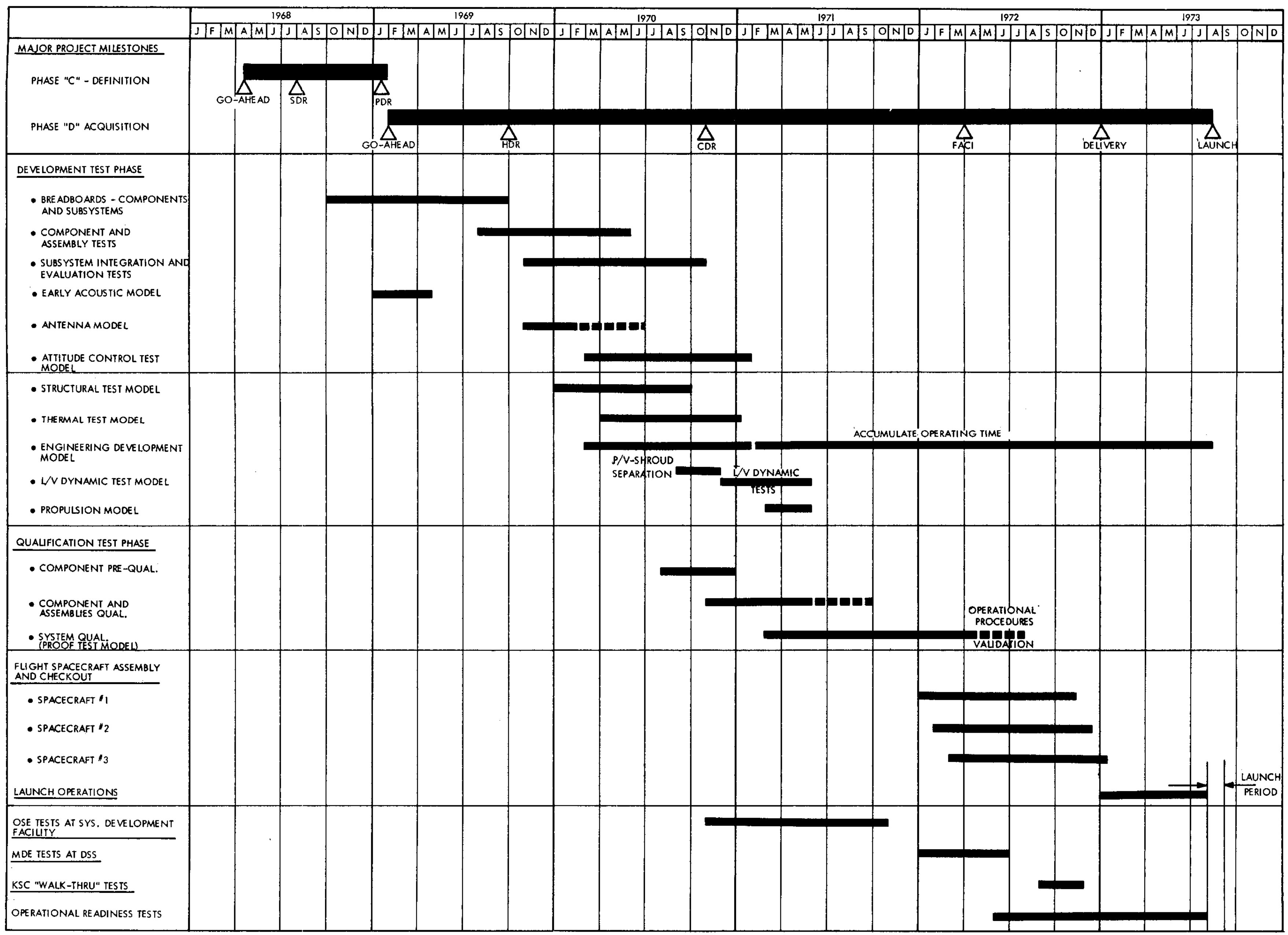

Figure 3-3. Summary Test Schedule

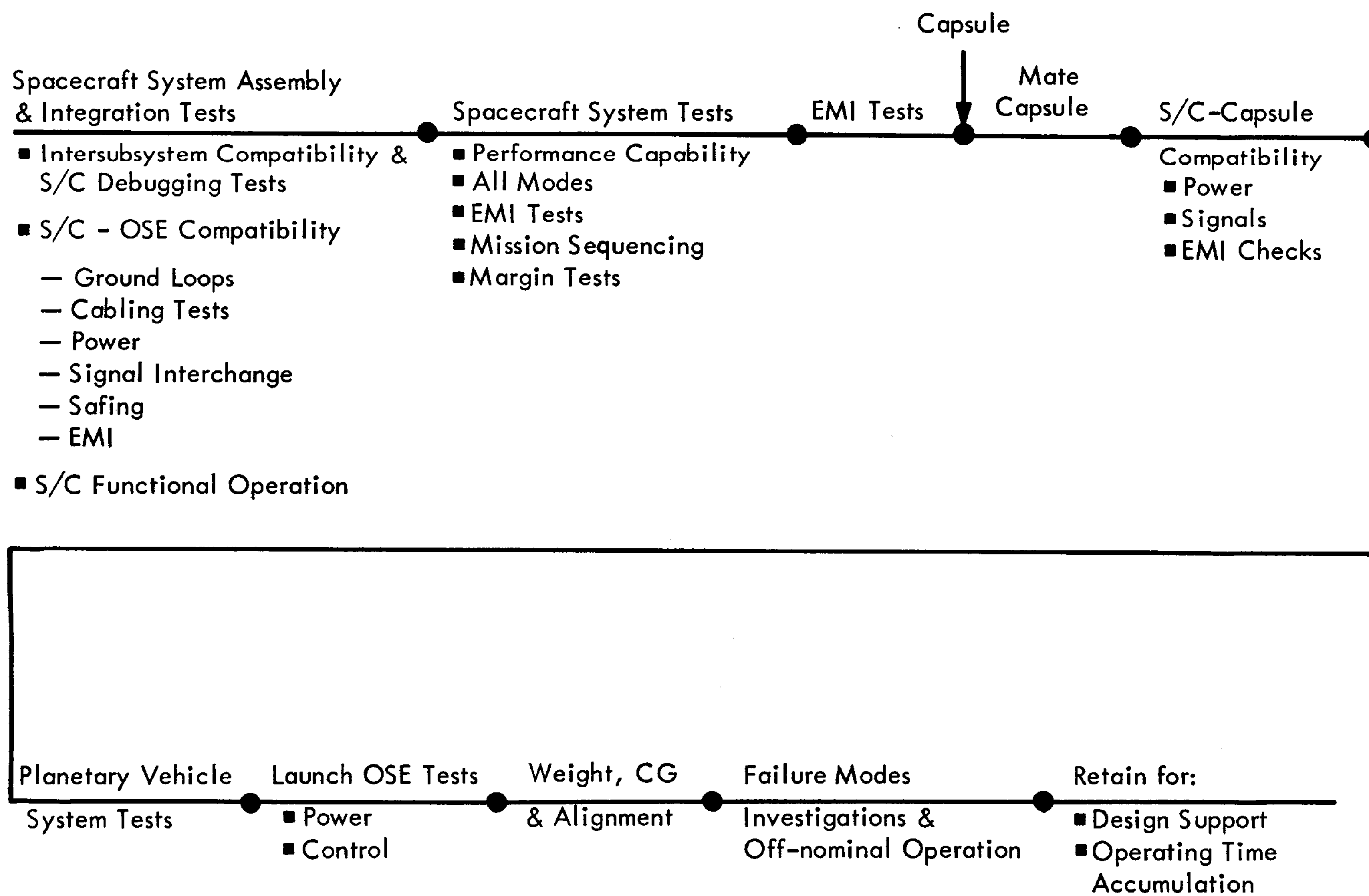

Figure 3-4. Flow Plan - Engineering Development Model

SUBSYSTEM	ITEM	PREVIOUS APPLICATION	MODIFICATION OF FLIGHT-PROVEN HARDWARE	STATE-OF-THE-ART DESIGN APPROACH	NEW DEVELOPMENT
Telecommunications	Transponder	Mariner C, '69	X		
	Power Amplifier (50W)	Apollo (20W)			X
	Tape Recorders	JPL Transport (10^8)			X
	Data Encoder Subsystem	Mariner '69		X	
	Command Detector	Mariner '69	X		
	Command Decoder	Mariner '69		X	
	Maneuver Antenna			X	
	Broad Coverage Antenna			X	
	Medium Gain Antenna	Mariner C	X		
	Relay Antenna			X	
	High Gain Antenna			X	
	Data Automation Subsystem	Mariner C, '69		X	
Propulsion	Propulsion	LEM, Apollo, Class. A. F. Satellites	X		
	Propellant	LEM	X		
	Chamber	LEM	X		
	Nozzle	LEM	X		
	TVC	LEM		X	
	Propellant Tank	LEM		X	
	Pressurization Tank	Saturn SI B		X	
	Regulator	LEM	X		
	Valves	LEM	X		
	Propellant Motion Control				
	. Start Tanks	Agena		X	
	. Screens				X
	. Baffles	Titan, Saturn	X		
Guidance and Control	Sun Sensors	OAO	X		
	Canopus Star Sensor	Mariner C, Mariner '69	X		
	Integrating Gyros	Mariner C, OAO Class. A. F. Satellites	X		
	Accelerometer	LEM, Minuteman	X		
	Attitude Control Electronics	Mariner C, OAO, Nimbus, Surveyor	X		
	Autopilot Electronics	Mariner C, Surveyor	X		
	Cold Gas Jet Subsystem	Nimbus, Mariner C, Surveyor	X		
	Antenna Actuators	Mariner C, Gemini	X		
	PSP Horizon Sensor	None		X	
	PSP Actuator	Catalog Item	X		
Electrical Power	Batteries				X
	Main Regulator			X	
	Charge Regulator			X	
	400 Hz Inverter	Mariner C	X		
	2400 Hz Inverter	Mariner C	X		
	Synchronizer	Mariner C	X		
	Power Switching Logic			X	
	Solar Panels			X	
Temperature Control	Thermal Shutters	Mariner C, Nimbus	X		
	Thermal Shutter Controls	Mariner C	X		
	Contingency Heaters	Class. A. F. Satellites Mariner C	X		
	Insulation	Nimbus, Class. A. F. Satellites Mariner C	X		
	Coatings	Nimbus, Class. A. F. Satellites	X		
Computer and Sequencer	Computer and Sequencer			X	
Pyrotechnic	Pyrotechnic Controller	Mariner C	X		
	Explosive Devices	Class. A. F. Satellites, Mariner C Saturn S-2 Polaris Pershing Sprint	X		

Figure 3-5. Development Status Chart

determination of the performance capability of each element, as well as determination of the environment it will see and its compatibility with other equipments. As a second example, three system models (of varying degrees of completeness) are planned to allow parallel determination of thermal, mechanical, and electrical environments that will be seen by the functioning equipment. While this approach seems somewhat costly, it is an extremely wise investment to attempt to uncover basic difficulties very early in the program.

c. The design briefly described in Section 2 has been heavily biased by manufacturing and test considerations. The modular design which allows assembly and test of major spacecraft elements in parallel, with relatively simple interfaces at final assembly, is a major factor in schedule assurance.

d. Contingency time has been incorporated in the schedule shown in Figure 3-3. While not specifically identified, each major test provides some time for failure, fix, and retest, within the schedule shown for that item. For example, it is felt that delivery to KSC by January 1973 provides at least two months contingency in the required operations leading to a launch which will occur no earlier than August 1973. The proposed method of allocation and control of this contingency time will be described as part of the Phase C proposal.

3.2.2 PROGRAM UNCERTAINTIES

Some aspects of the Voyager project are not well defined at present and hence alternatives may be required to the basic program described in Volume III. Typical areas are:

a. The capsule design is not well known and hence the complexity of the spacecraft-capsule interface is not clear. The program described in Volume III assumes first mating of flight spacecraft and flight capsule at KSC, assuming a relatively simple interface. Increased complexity could change this approach and impact the overall schedule, test plans, logistics, and facility requirements.

b. From a potential mission value standpoint, it is often desirable to delay selection and design of specific science instruments as late as possible to enable use of the latest technology for providing a more capable instrument. From a probability of success standpoint (both meeting the fixed launch date and successful flight operation), it is desirable to define the spacecraft science instruments early so that they are handled much like other equipment in the development and qualification cycle. Much detailed scheduling work is expected to be required to arrive at the optimum approach for each specific instrument.

3.3 FACILITIES

Because of their potential cost and time effects, the major facility requirements for development, qualification and production of the spacecraft have been examined in some detail.

Based on data from past GE-MSD spacecraft programs, the cleanliness requirement imposed by the quarantine constraint can be met without imposing major facility needs. Tents, which control carefully the environment immediately adjacent to the spacecraft coupled with a reasonable degree of cleanliness in the surrounding area, will suffice. This approach has been demonstrated on past programs to provide results as good as more expensive, permanently installed clean rooms since the major source of contaminate is not the ambient air but the vehicle itself and operations personnel.

The major environmental facilities needed for system testing - acoustic, thermal vacuum, vibration - either exist or are projected for other programs currently on contract.

The KSC facilities to support launch operations are not well defined and additional study of this area is recommended. The Manned Spacecraft Operations Building is certainly adequate to support Voyager operations if available. The Spacecraft Checkout Facility used for Ranger and Mariner may be suitable but Voyager's size may make this a marginal facility.

The facilities available for handling hazardous materials and a planetary vehicle assembly area remain to be defined. The facility for assembly of the overall planetary vehicle will be strongly influenced by the need to handle a sterile capsule with relatively large quantities of propellants aboard.

SECTION 4
SUMMARY OF ENGINEERING TASKS

4.1 SPACECRAFT DESIGN PROVISIONS TO ACCOMMODATE A RADIOISOTOPE THERMOELECTRIC GENERATOR (RTG) IN THE SURFACE LABORATORY SYSTEM (1975)

4.1.1 OBJECTIVE

The application of radioisotope thermoelectric generators (RTG's) is being considered. for the prime power source for the Voyager Surface Laboratory System (SLS) in 1975. The objective of this study was to examine the impact of the RTG's upon the spacecraft, and to determine the necessary spacecraft design provisions for compatibility. A secondary objective was to examine the impact of RTG's used in both the SLS and the spacecraft.

4.1.2 SCOPE

RTG's are characterized by elevated temperature operation and the emission of nuclear radiation. It was the interaction of these characteristics with the spacecraft that was the main concern of this study. Other aspects of the study dealt with the potential reduction of spacecraft power requirements since the RTG powered capsule (SLS) is self-sufficient from a power standpoint.

Representative spacecraft and capsule configurations shown in Figure 4-1 were first selected to serve as the basis for thermal and nuclear radiation analyses. These particular configurations were selected to introduce rather than avoid potential interaction problems. For example, the spacecraft electronics are located close to the spacecraft capsule interface to aggravate thermal and nuclear interactions.

The capsule configuration also reflects this approach by confining the heat rejected from the RTG's to a narrow zone, creating possible conditions of excessive temperatures. The underlying thought was that if analysis showed the practicality of the worst case approaches, greater margin would be possible with judicious designs. The planetary vehicle thus defined consisted of a solar array powered spacecraft and an RTG powered capsule.

The case with an RTG powered spacecraft was considered by circumferentially locating eight 75-watt RTG's on the lower side of the solar panels as shown on Figure 4-2.

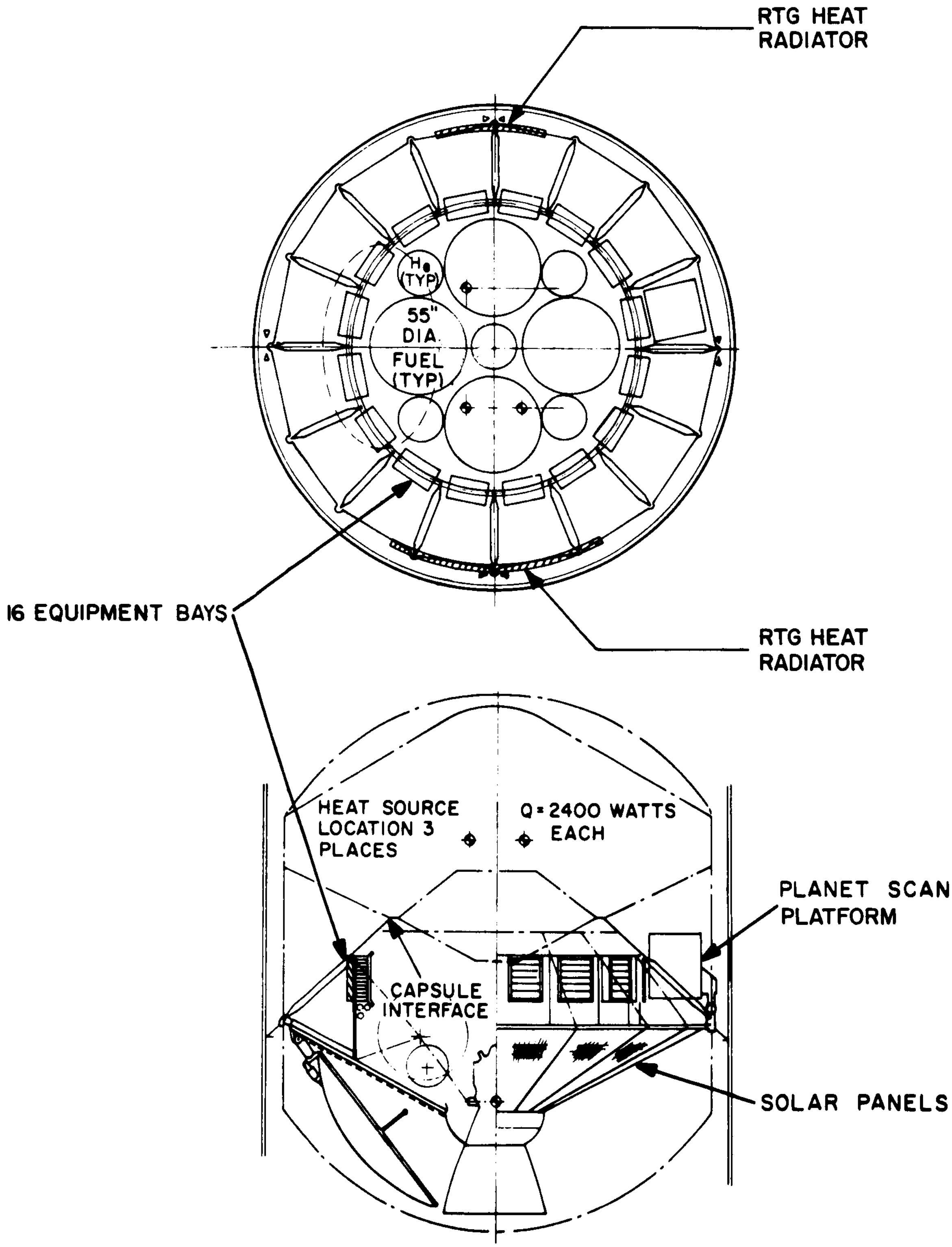

Figure 4-1. Planetary Vehicle Configuration (Solar Powered Spacecraft)

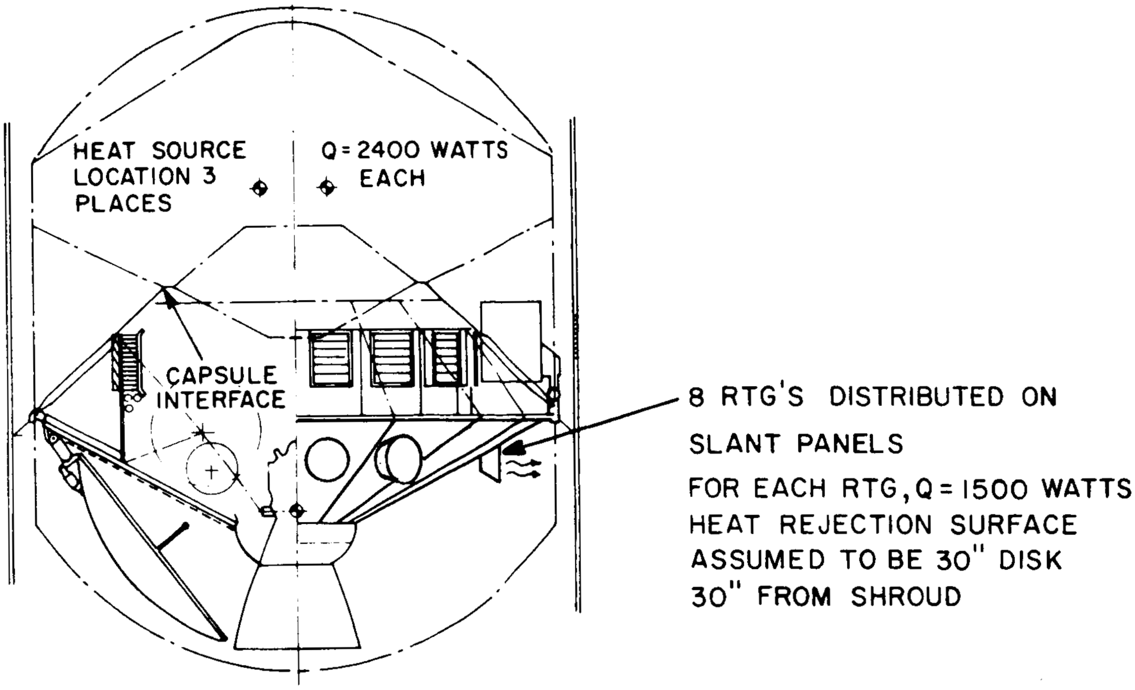

Figure 4-2. Planetary Vehicle Configuration (RTG Powered Spacecraft)

4.1.3 RESULTS

4.1.3.1 Thermal Interaction

For the solar powered planetary vehicle defined, the inside shroud wall will be about 260 degrees F. in the vicinity of the capsule RTG radiators during steady state prelaunch conditions. Because the spacecraft is remote from this elevated temperature zone, effects on the spacecraft are slight and will not result in a substantial increase in shroud cooling requirements.

Spacecraft RTG's would result in an additional zone of elevated shroud temperature at about 220° F. and would require a further increase in shroud cooling.

During the launch phase, the shroud zone near the capsule RTG radiators would rise to 420 degrees F. as shown on Figure 4-3, about 100 degrees higher than if RTG's were not used. With an upper limit on permissible shroud temperature estimated to be around 300 degrees F., some method of removing heat from the shroud wall appears necessary. The heat of vaporization of several pounds of water contained in appropriate cooling coils would suffice to remove the required heat from the shroud. After shroud separation, there are no appreciable RTG thermal effects on the spacecraft.

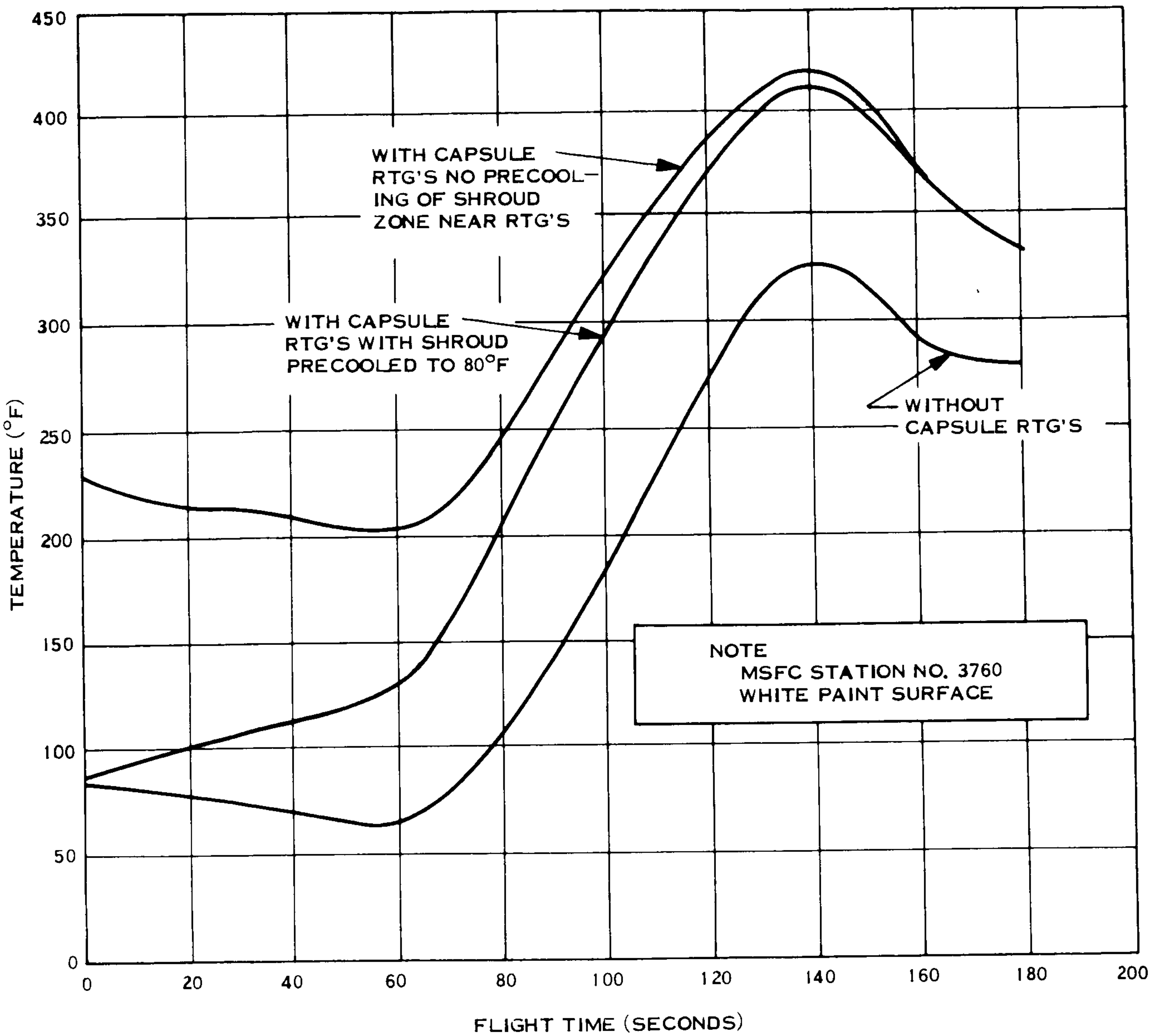

Figure 4-3. Temperature History of Outside of Shroud Opposite Capsule RTG Radiating Plate

4.1.3.2 Nuclear Radiation Interaction

Radiation sensitivity levels were determined for threshold effects, moderate effect, and severe damage for spacecraft piece parts, materials and science instruments. The spacecraft radiation environment was determined by mapping the gamma and neutron levels throughout the planetary vehicle. Total integrated dose rates for the mission were then established. Considering first the spacecraft parts and materials (exclusive of the science instruments), the effect of gamma radiation is negligible. Neutron radiation produces some threshold and moderate effects. These effects can be overcome by proper part selection and derating, circuit design, and perhaps in a few cases, by local shielding.

The combined effect of RTG's in the spacecraft and capsule is only slightly higher than that due to capsule RTG's alone.

Nuclear radiation interaction with the science payload will be in the form of dynamic interference, i.e., inability of certain instruments to distinguish between the natural particles of interest and those resulting from RTG emission. Of the nine science instruments defined as baseline, only the ultraviolet spectrometer will be seriously affected by the RTG environment. The problem created is the reduction of signal-to-noise ratios and the ability to extract useful information from target areas of weak ultraviolet emission. The two infrared spectrometers of the baseline instruments will also be affected but to a much less degree. Other potential science instruments were examined. It was found that some of these would experience dynamic interference depending on their sensitivity requirements and their effects of radioactivity induced in spacecraft materials. In general, it was found that this radioactivity would decay to insignificance within several hours after the separation of the RTG powered capsule. The use of shielding for the sensitive instruments can be avoided if instrument operation can be delayed until after capsule separation; this is not the case with an RTG powered spacecraft.

In reviewing the effects on potential photo-imaging systems, it was found that 40 to 130 pounds of shielding may be required to limit fogging of photographic film to acceptable levels. An uncertainty exists as a result of limited data on the effects of neutrons on film.

4.1.3.3 Mission Effects

The RTG powered capsule is self-sufficient thereby removing demands for 200 watts of spacecraft power (up to time of separation). However, solar occultations late in the spacecraft mission and the associated need for battery charging power preclude significant reduction in spacecraft solar array area.

Mission advantages do appear likely in the period of up to one month between planetary vehicle Mars orbit insertion and capsule separation and deorbit. With an additional

200 watts of power available to the spacecraft during this period, it is possible to consider such improvements as:

a. Increasing the science data return.

b. Selection of optimum Mars orbits which may include earlier solar occultations.

c. Use of RTG capsule power as an emergency backup to spacecraft battery power during maneuvers.

4.1.4 CONCLUSIONS AND RECOMMENDATIONS

RTG power in the SLS or in both the SLS and spacecraft can be effectively accommodated by the Voyager Spacecraft. The major problems are with the nuclear radiation interactions with the science payload instruments. In this respect, it is recommended that future studies be planned to assess problems and solutions for effective utilization of candidate science instruments for the Voyager Spacecraft.

4.2 APPLICABILITY OF APOLLO CHECKOUT EQUIPMENT TO VOYAGER OSE

4.2.1 OBJECTIVE

The basic objective of this engineering task was to determine the applicability to the Voyager program of the checkout system and support equipment now in use on the Saturn/Apollo program. The specific objectives were to determine the extent of applicability of the checkout systems at KSC to Voyager system test and launch control, the nature of any incompatibilities, and the modifications required to resolve any such incompatibilities. An additional objective was the identification of Saturn/Apollo mechanical and fluid ground equipment used at KSC which could be used or modified for use on Voyager.

4.2.2 CHECKOUT SYSTEMS

Two checkout systems are operational at the KSC Saturn/Apollo facilities - the Saturn ESE (Electrical Support Equipment) and the Apollo ACE-S/C (Acceptance Checkout Equipment - Spacecraft). Of these, only ACE has a significant degree of applicability. This is attributable to the fact that as the ESE system is sized for the Saturn application, it is more comprehensive than needed for Voyager; moreover, its use is assumed to be pre-empted by the Saturn V launch vehicle used for the Voyager mission and for any other in-process Apollo type missions.

Most of ACE (four systems of which are currently operational at KSC) is located in the Manned Space Operations Building (MSOB) at Merritt Island; some elements of the system, however, stay with the Apollo spacecraft through its ground mission flow at KSC, as shown in Figure 4-4.

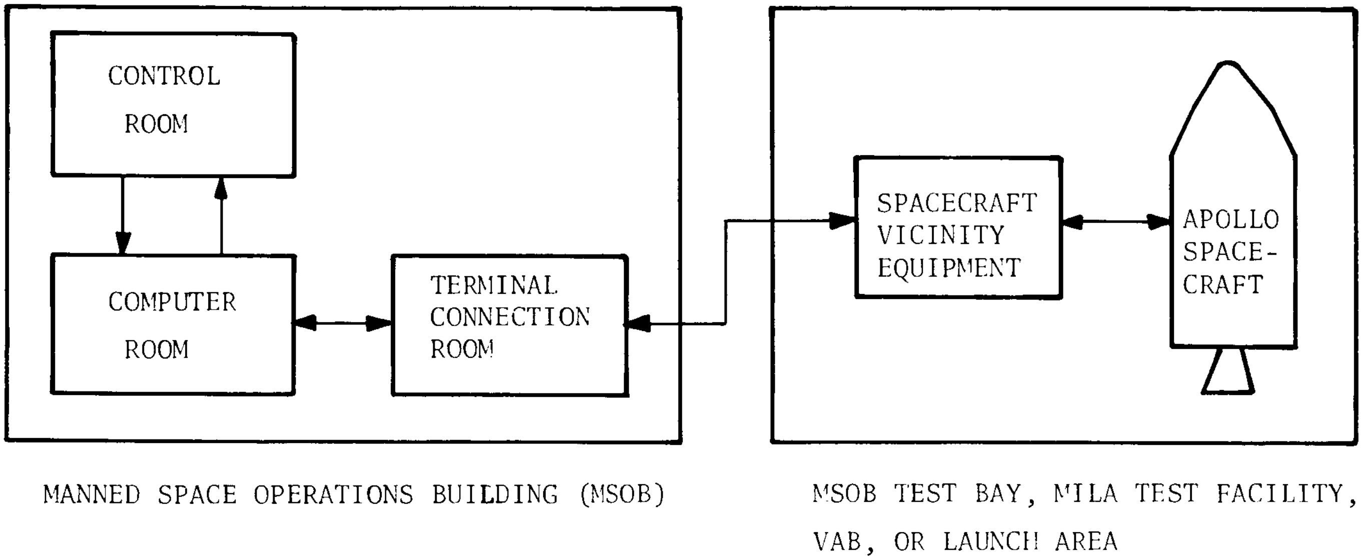

Figure 4-4. Location of ACE at KSC

The heart of the system, the computers, controls and displays located in the MSOB, was generally found to be applicable to Voyager checkout; the spacecraft vicinity equipment, however, for the most part must be unique to Voyager.

More specifically, operators' controls and displays, located in the MSOB control room, can be adapted to Voyager on a one-for-one (one ACE for one Voyager spacecraft) basis. Most of the required control and display reconfiguration can be accomplished through test programming, as these controls and displays are general purpose in nature. An optional modification would involve relocation of switches, lights and meters on the panels to facilitate test operations.

The ACE Computer Room, which contains a complex of two CDC 160G computers, two Radiation 540 decommutators, and other peripheral equipment, is apparently applicable, in its existing form, to Voyager, with modification to accommodate potential Voyager decommutation incompatibility and trend testing. The preferred approach, in the event Voyager spacecraft telemetry would use asynchronous medium and low speed telemetry decks, would be to have ACE's decommutator handle OSE telemetry while the downlink computer handles the spacecraft telemetry decommutation. Various approaches were established to implement 10-bit accuracy (which would allow for more sensitive trend detection) with minimum impact on the present 8-bit system.

The terminal connection room of the MSOB does not appear to need any modification to be applied to Voyager.

The present ACE system software packages exist in two parts - an essentially system oriented package for computer implementation of operator generated commands, data analysis and display functions, and an adaptive program for automated testing. The majority of the former package is applicable to Voyager; the adaptive test programs, however, are written for specific spacecraft and the Apollo routines will not be suitable for Voyager.

Although ACE spacecraft vicinity equipment is less susceptible to "universal" use than control room and computer room equipment, there is the possibility of utilizing certain Saturn/Apollo components in implementing Voyager vicinity equipment. Examples of these are the Saturn DDAS (Digital Data Acquisition System) as a 10-bit PCM equipment and the Apollo baseplate units as standard switching modules.

4.2.3 SUPPORT EQUIPMENT

Approximately 100 items of Apollo fuel, oxidizer and gas service equipment, and handling equipment have been identified as capable, with various degrees of modification, of being adapted to Voyager. Most of these items are fluid and fuel service equipment used for LEMDE. Additionally, due to location, size, and weight similarity between LEM and Voyager, many items of LEM handling equipment can be utilized with a minimum of modification.

4.2.4 CONCLUSIONS AND RECOMMENDATIONS

This study has established the technical feasbility of applying significant portions of ACE-S/C to Voyager checkout and launch control. (Refer to OSE Requirements of the Voyager System Description - Volume II for an evaluation of alternative OSE concepts from an operational and implementation point of view.)

A program decision, however, will additionally require consideration of:

a. Program priorities for use of the ACE systems at KSC.

b. Over-all program costs to make required modifications and to provide the necessary sets of equipment at the factory as well as at KSC.

c. Projected performance of ACE in 1973, 1975, etc., versus development of a state-of-the-art and Voyager-optimized system.

Insofar as LEM fluid service and handling equipment utilization is concerned, technical feasibility (with a minimum of modification of many of the selected items) appears established. Decisions for utilization would, therefore, be primarily dependent upon the availability of this equipment for Voyager.

4.3 CENTRAL COMPUTER STUDY

4.3.1 OBJECTIVE

The purpose of the study was to assess the relative merits of a centralized (central computer) approach versus a decentralized (separate subsystems) approach in implementing the many digital data processing functions required by the Voyager Spacecraft. Typical functions are: decode command, format telemetry data, provide sequencing signals.

During the course of the study, many facets of both approaches were explored. In our consideration of the central computer approach, excellent contributions were made by the Federal Systems Division of IBM and appear as appendixes to our Milestone Report VOY-D3-TM-22.

4.3.2 STUDY APPROACH

Comparison of a separate subsystem implementation with a central computer approach must answer such questions as:

a. Does one approach or the other result in a substantial decrease in the total hardware required, and hence provide a higher reliability potential as well as lower size and weight? This could definitely be the case if several of the separate subsystems required a basic arithmetic unit which could be replaced by a single, more capable arithmetic unit in the central computer.

b. Does one approach offer an advantage in flexibility to accommodate changes from mission to mission, or within a mission? For example, using a separate subsystem to process science data confines changes to that subsystem when the science payload is changed from mission to mission. On the other hand, a sufficiently capable central computer might accommodate such a change through programming changes only.

c. Is one approach inherently more testable, leading to improved schedule assurance as well as better capability to detect and remove defects in the hardware? This is basically a question of the relative difficulty of producing and testing one relatively complex device compared to several simpler ones.

This study used a method of numerical evaluation to indicate the relative desirability of adopting a centralized computer approach. Comparison criteria ranged from the foremost item of reliability through flexibility, testability, weight, power, size and design difficulty. The method includes all decision factors considered amenable to numerical evaluation. The numerical approach forces a quantitative assessment of the degree to which each implementation satisfies the above criteria. The main features of the evaluation for each competitive implementation are:

a. For each function, the competing implementations are evaluated for each comparison criteria, resulting in a numerical index of performance.

b. The performance indexes obtained from (a) above for each function are weighted, in accordance with the relative importance of each function to the mission, and summed to give the total performance index for each of the competing implementations.

The process is systematic so that the various factors are exposed for review and modification as desired. The relative sensitivity of the total performance index to changes can be readily determined.

4.3.3 NUMERICAL EVALUATION

Comparisons of competitive implementations have been made for functions taken from the Voyager Task B Study as listed below:

a. Process Ground Commands
b. Provide G&C Logic and Switching Control
c. Process Telemetry Data
d. Initiate Time-from-Launch Functions
e. Initiate Computed Functions
f. Initiate Time-to-Go Functions
g. Initiate Periodic Orbit Functions
h. Provide C&S Data to Telemetry Subsystem
i. Provide Occultation Signals
j. Provide Data Storage Capability
k. Perform PSP Gimbal E Local Vertical Tracking
l. Provide Control of Experiments
m. Process Scientific Data

These comparisons were made for three degrees of applied redundancy to explore its effect. The intermediate grouping adds examples of data compression, error correction coding, and approach guidance. The highest complexity grouping adds an onboard check-out example and substitutes a more complex data compression example. The results of the numerical evaluations are shown in Figure 4-5.

4.3.4 CONCLUSIONS AND RECOMMENDATIONS

The results of the numerical analysis as shown in Figure 4-5 indicated a small margin in favor of the decentralized subsystem approach to processing functions.

When subjective judgment based on experience is added to the numerical results, it reinforces the conclusion that the Voyager Spacecraft, as presently conceived, does not warrant changing from the separate subsystem implementations to a central computer. The functions previously listed are not individually complex - none of them require multiplication, for example. Thus, sharing of a central computer is unlikely to effect a significant reduction in the total equipment required.

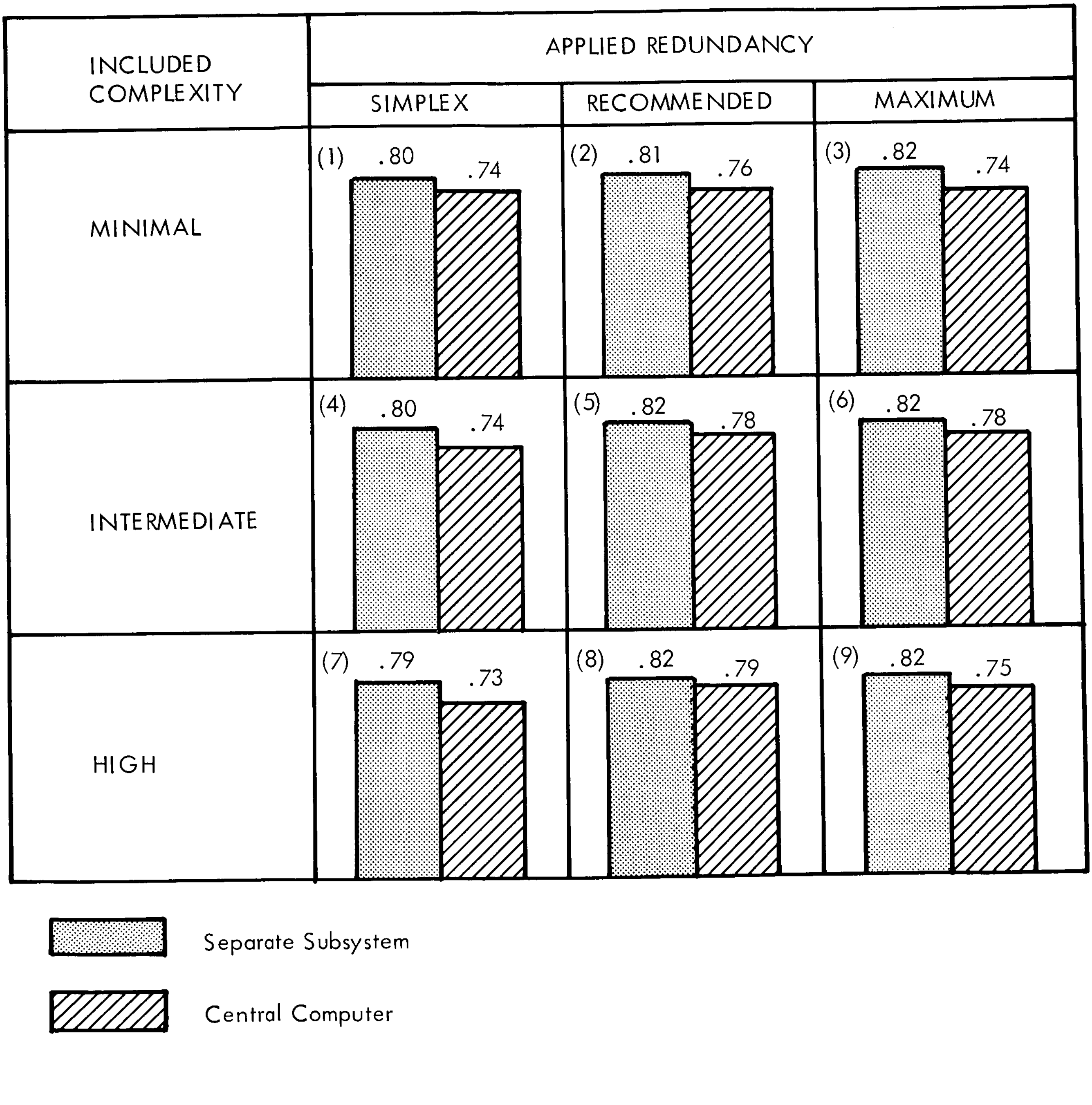

Figure 4-5. Results of Numerical Evaluations

A change to a central computer approach has program effects which are not easily evaluated numerically, but which may be serious. For instance, the functional independence of each subsystem (i.e., telemetry, command, etc) is greatly decreased, thereby making the early subsystem tests less conclusive. More importance is then attached to total spacecraft system testing which, of course, is always later in time.

The data processing requirements of a Voyager Spacecraft are considerably less than contemporary computers can provide. On the other hand, for a Voyager application, computer failure rate data indicates the need for an order of magnitude improvement in mean time to failure of contemporary computers.

Accordingly, it is recommended that presently anticipated near term Voyager data processing functions be implemented by the "separate subsystems" approach. The relatively small margin by which the separate subsystem approach is numerically favored suggests that for future Voyager missions in which functions of significantly increased complexity may be contemplated, the question of choice can logically be reopened.

4.4 ATMOSPHERE DEFINITION STUDY

4.4.1 OBJECTIVE

The objective of the atmosphere definition study was to provide definition of the Martian atmospheric characteristics required by spacecraft designers. The categories of interest are: physical data, atmospheric parameters near the surface (<1 meter), entry atmosphere structure (<100 km), outer atmosphere structure (>100 km), clouds and haze, and chemical kinetics and composition. Emphasis was placed on both the characteristics and the chemical kinetics and composition of the outer atmosphere.

4.4.2 SCOPE

The atmosphere definition task consisted of utilizing existing Martian atmosphere guides and updating these to incorporate the more recent information. This included the findings based on the Mariner IV data, the spectroscopic data obtained during the recent opposition, theoretical studies, and empirical studies. Empirical methods were developed to provide an estimate of parameter values required when the available information was inadequate or totally lacking. In addition, a theoretical analysis of the chemical kinetics and the composition of the atmosphere was made in order to gain a better understanding of the probable time-space characteristics of the upper atmosphere.

4.4.3 KEY CONCLUSIONS

a. The atmosphere density at altitudes in excess of 500 kilometers is likely to exhibit diurnal (day-night) variations of one order of magnitude, and solar cyclic variations of three orders of magnitude. Since a period of high solar activity is expected in 1969, it is anticipated that the density profile based on the 1969 Mariner flyby experiments may yield density values which are as much as three orders of magnitude greater than those deduced from the Mariner IV data. It should further be noted that 1973 closely corresponds to 1965 in terms of solar activity, both being periods of relatively low solar activity.

b. The chemical composition of the atmosphere is dependent on chemical kinetics, solar flux and diffusion. Because the thermal balance or temperature of the atmosphere is dependent on the composition, as is the average molecular weight, the chemical structure is of prime importance for a realistic determination of the upper atmospheric density.

c. A limited sample of observed motions of white and yellow clouds may provide a means of assessing the horizontal and vertical circulation characteristics. For example, the motion of the low altitude yellow dust clouds, (Figure 4-6) when compared with the motion of the higher altitude yellow projections (Figure 4-7) suggests a poleward flow at upper levels with an equatorward flow at low levels.

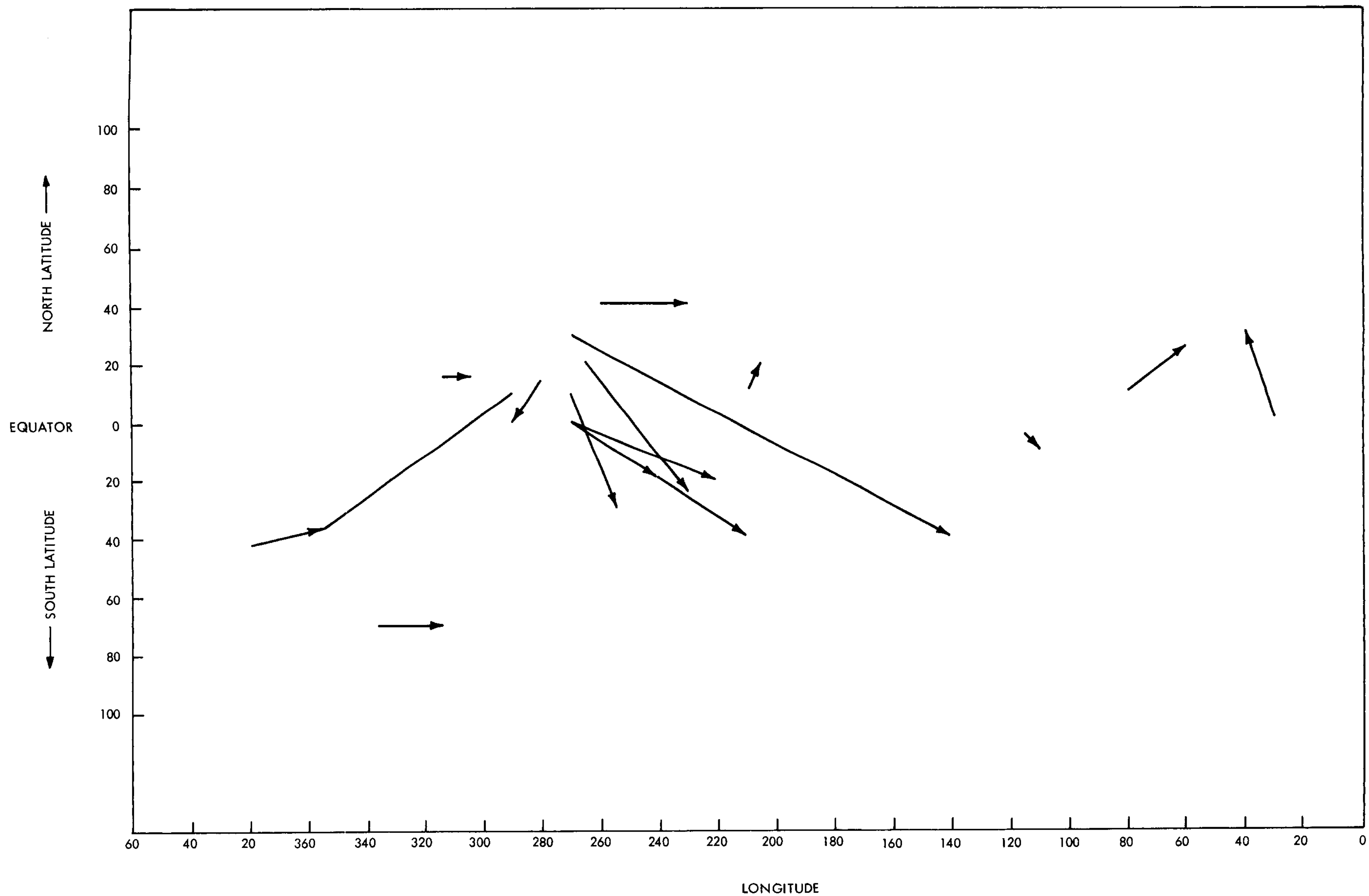

Figure 4-6. Motion of Low Altitude Yellow Dust Clouds

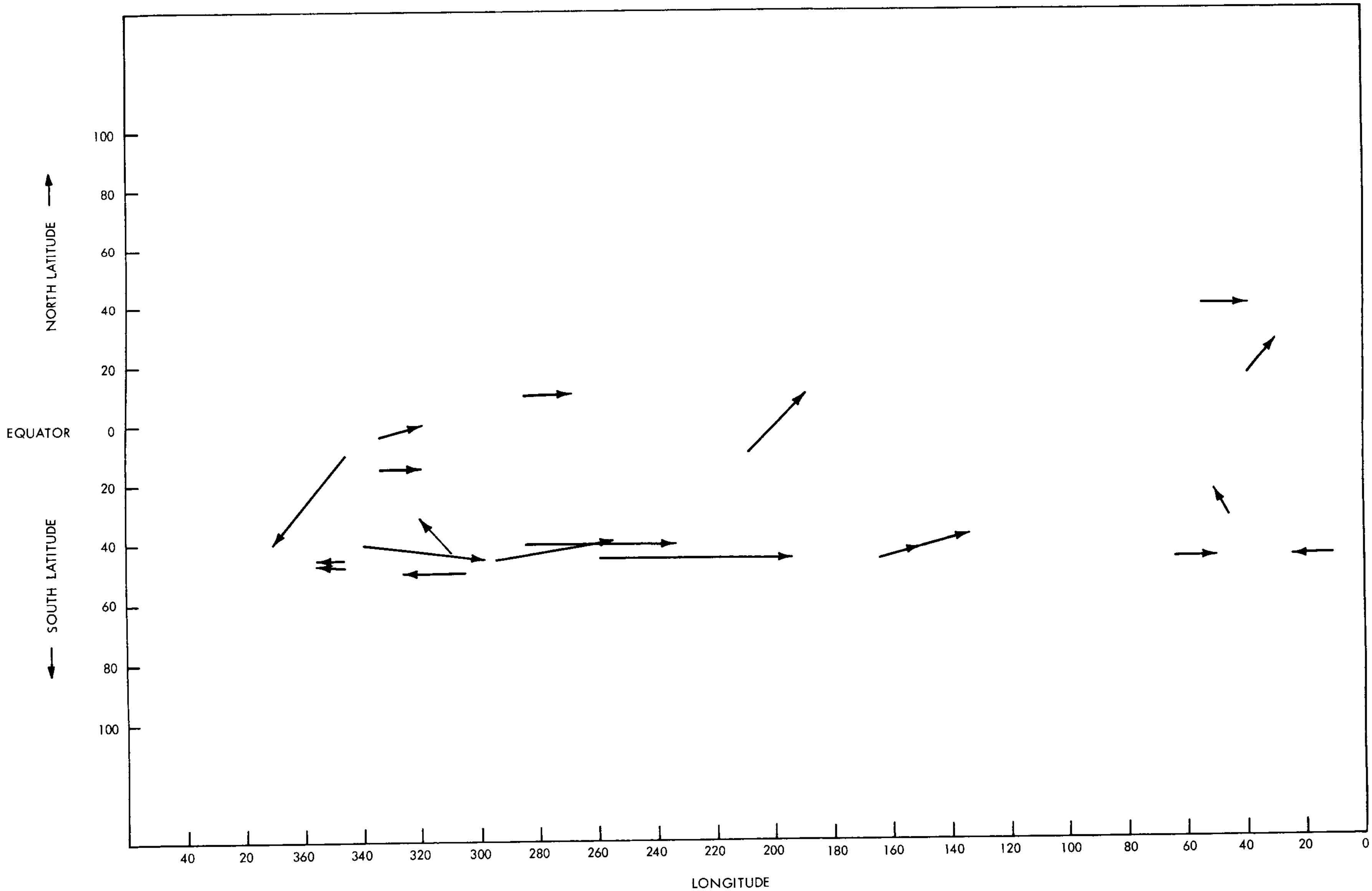

Figure 4-7. Motion of High Altitude Yellow Projections

4.4.4 RECOMMENDATIONS

a. A standard and complete set of physical data values, giving means and extremes, should be adopted for use throughout the Voyager program.

b. Studies of the chemical kinetics of the upper atmosphere of Mars should be extended to include additional species, such as hydrogen and helium. Results of these extended studies should be compared to the empirically derived upper atmosphere temperature and density, in order to test the relative validity of the latter. Such coupling of the two approaches would provide greater confidence in the derived variations of the upper atmosphere density as a function of solar activity.

4.5 PHOTO-IMAGING STUDY

4.5.1 OBJECTIVE

The broad objective of the photo-imaging study is to investigate alternate methods, such as the use of television sensors or photographic film, to perform photo-imaging experiments of the Martian surface from an orbiting spacecraft. For each photo-imaging system considered, the quality of the data returned must be established, and the relative impact of the alternate systems on the mission and system design must be assessed.

Many of the desirable scientific missions, such as topological, areological, meteorological, biological, and studies of both the planet and its satellites' gross characteristics, were evaluated to pinpoint the critical photo-imaging system requirements. Design requirements for spatial and intensity resolution, lighting conditions, optimum spectral bands, multispectral and stereo operation were studied.

4.5.2 CHARACTERISTICS OF MARS

All photo-imaging systems are affected by the visual characteristics of the planet Mars. It is expected that topographic mapping will have to be performed under low contrast conditions, i.e., intensity resolution of about 0.5 percent will be required. This is strongly indicated by the Mariner IV results. The low contrast is partly due to the eroded surface condition with its resultant small slopes in addition to scattering in the atmosphere which tends to decrease contrast and make it less sensitive to solar illumination angle. Not unexpectedly, the spectral requirements for sensing are for longer wave lengths, towards the red.

These characteristics affect the design and selection of the specific photo-imaging experiment and constitute a very severe requirement for some of the candidates considered.

4.5.3 MISSION CONSIDERATIONS

The photo-imaging experiment places many demands on the over-all mission design, primarily concerning the selection of orbit parameters and the desired lighting conditions. In summary, these demands are:

a. Periapsis Altitude - All candidate systems benefit from low periapsis altitude since the surface resolution obtainable with a given set of optics improves directly with nearness to the surface. The increase in surface speed of the sub-spacecraft point and its impact on image motion compensation requirements is minor. The minimum allowable periapsis altitude is limited by planetary quarantine as described in Section 2.

b. Orbit Inclination - Studies of the effects of orbit selection on the ability to map the planet have been carried out. An example of this work is shown in Figure 4-8, which is a typical answer to the question, "How fast can you map the Martian surface?". This curve shows the data taking opportunity which each candidate system has when placed in a nominal 1,000 by 12,000 kilometer orbit with a 40-degree inclination. This orbit permits mapping of about 32 percent of the total Martian surface in one month's time. However, this rate of mapping is dependent on the sensor field of view (10 degrees) and on the capability of the data handling system to record and transmit the equivalent amount of data at a rate fast enough for the spatial resolution desired.

c. Orbit Period Control - For mapping of the planet, it may be desirable to synchronize the orbit period with Mars' rotation such that overlapping ground traces are produced. For an orbit period that is 1/3 of a Martian day, orbit period control should be on the order of 0.5 minute. This is within the capability of the orbit determination and correction process.

d. Planet Areas of Interest - One task is to find areas on the planet's surface that look promising for future biological exploration. The expected concentration of such areas between +10 and -40° latitude suggest that a 40-degree inclined orbit would permit mapping most of them. Prospective soft lander sites should additionally be investigated at high spatial and intensity resolutions. Diurnal and seasonal variations, such as the advance of the wave of darkening, determine at what intervals the mapping of specific areas should be repeated.

e. Lighting Conditions - The orbit selected must provide suitable lighting conditions for collection of high quality imaging data.

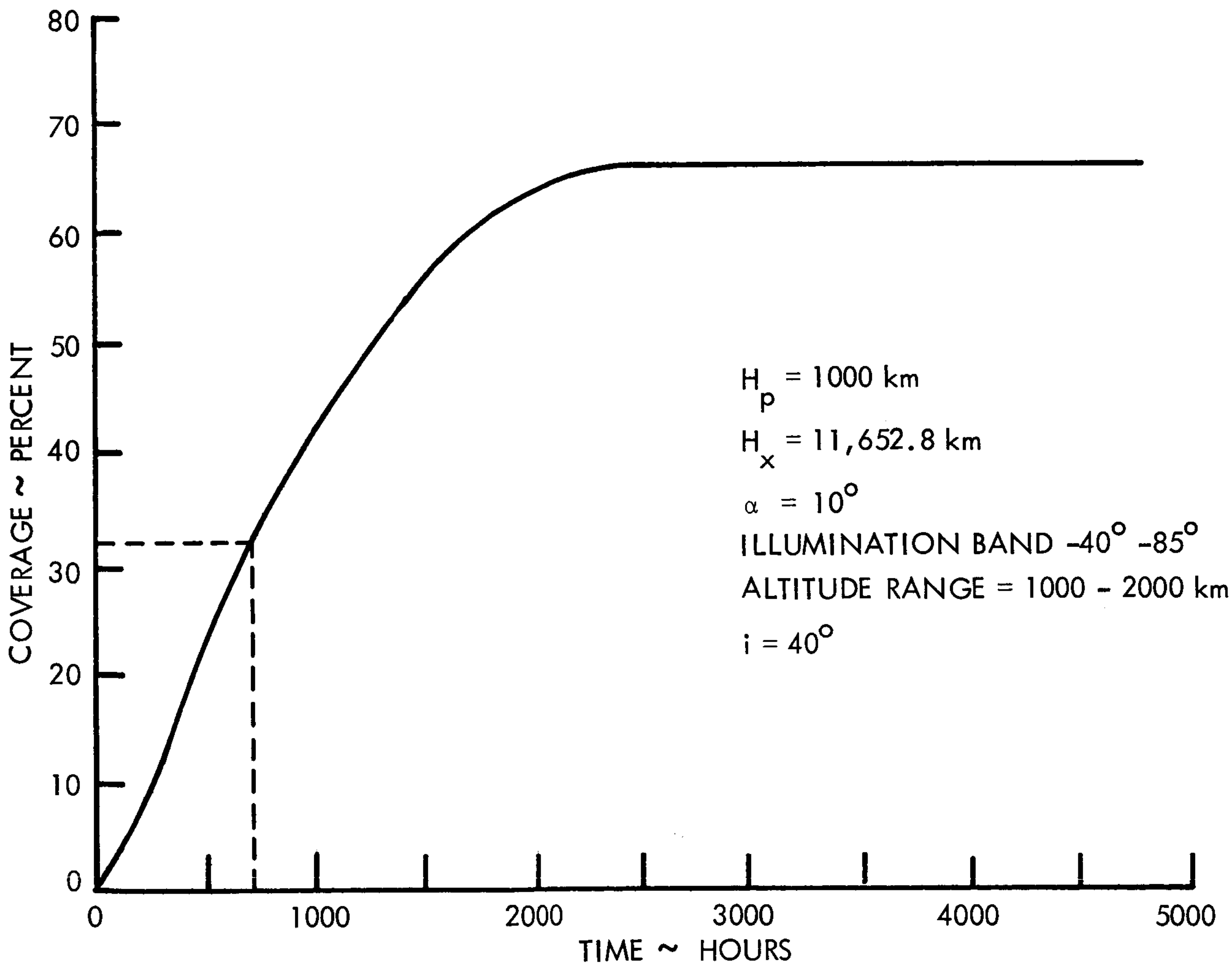

Figure 4-8. Mapping Coverage

In general it can be stated that the above considerations are not affected by the specific choice of a particular photo-imaging system. An orbit that is suitable for one candidate will be equally suitable for another.

4.5.4 SPACECRAFT DESIGN CONSIDERATION

Contrary to the above statement, the requirements imposed on the spacecraft system design can be strongly influenced by the choice of photo-imaging experiment. Major areas that will be affected are discussed briefly:

a. Data Handling and Transmission - Two problems exist. The first is the limitation on the data recording rate of available and predicted tape recorders. This limits the field of view (width of the mapping strip) for a given resolution capability (or conversely, the resolution for a given field of view) for those systems requiring auxiliary data storage. The second problem is the limitation on the data transmission rate that is set by the possible antenna size, power capability, and pointing accuracy. This limits the total area that each candidate photo-imaging system can map during one orbit.

   An example of these problems is shown in Figure 4-9. This curve is drawn for a given rate of mapping (i.e., for the same mission as shown in the preceding Figure 4-8) and illustrates the constraint on resolution made by the record rate and the transmission rate. If it is desired to improve the resolution from 100 to 30 meters, an order of magnitude increase in both data recording rate (to 400 kilohertz) and in data transmission rate (to 200 kilobits per second) is required.

   Our conclusion concerning the data handling problem is that the 100 meter resolution requirement for mapping is in reasonable agreement with the data handling capability of the baseline spacecraft. A transmission rate in excess of 30 kilobits per second is achievable, and a record rate of 60 kilohertz may be possible with analog recorders. Choosing these parameters permits, for the 100 meter resolution mission, a data collection period of up to 29 minutes per orbit for the baseline 8.2-hour period, 1,000 by 12,000 kilometer orbit. The data transmission time is for the remainder of the orbit period.

b. Stability Requirements - The exposure time required by various instruments can affect the stability required of the planet scan platform to insure acceptable smear. In addition, to produce a non-distorted map using a line scanning technique requires more accurate pointing than does an imaging instrument.

c. Other Parameters - The spacecraft is obviously concerned with over-all weight, power, and environmental control required by the candidate systems. In particular, the spacecraft is sensitive to weight and power dissipation in the planet scan platform. This platform is articulated with respect to the spacecraft, yielding mechanisms that are sensitive to total platform weight. It has variable orientation to the sun, leading to a difficult thermal problem, and hence sensitivity to thermal dissipation.

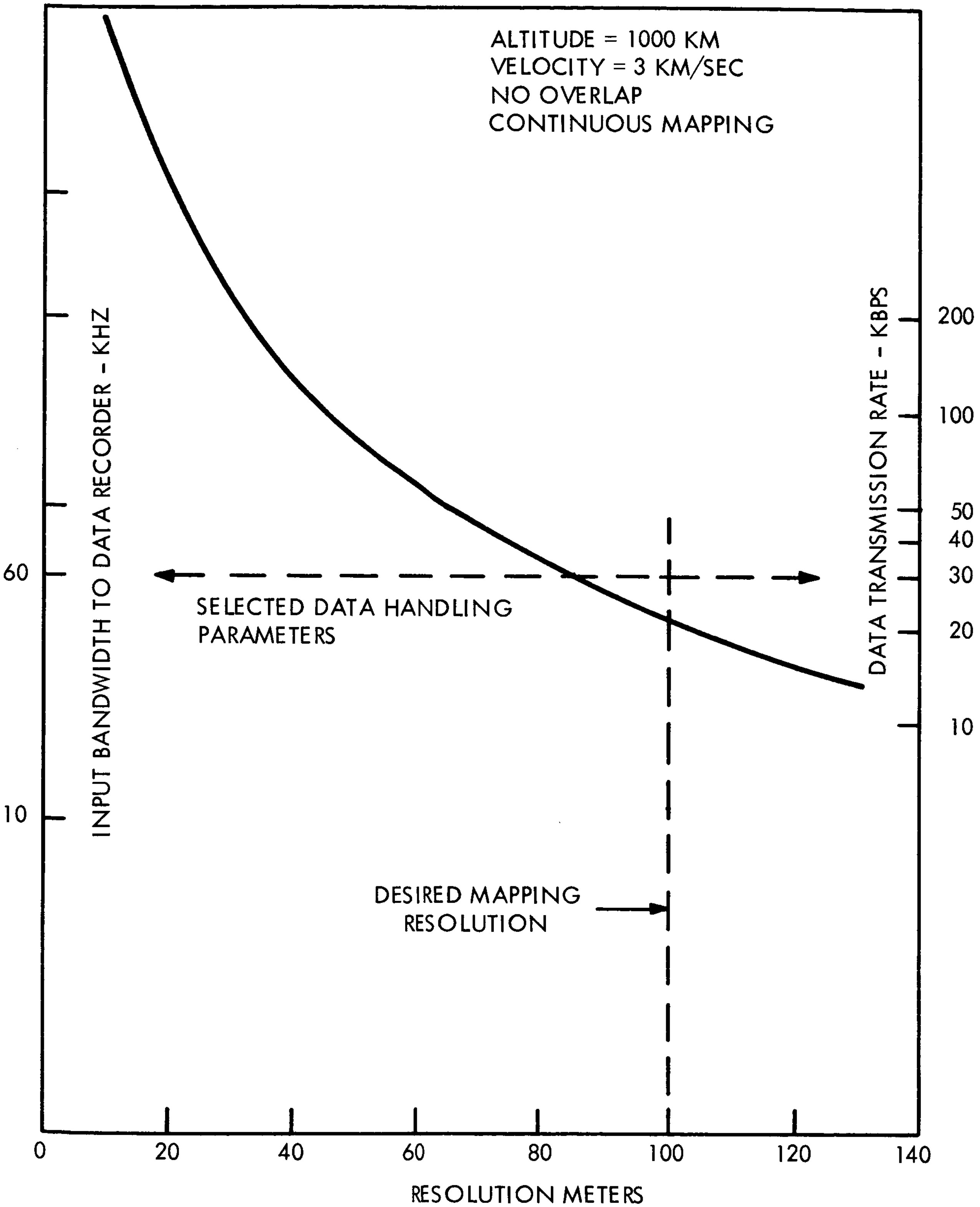

Figure 4-9 . Data Handling Requirements

4.5.5 CANDIDATE PHOTO-IMAGING SYSTEMS

A television system, a photographic film system, and an optomechanical scan system have been selected for the candidate photo-imaging systems. The three systems were selected by eliminating many other attractive candidates because of state-of-the-art limitations and data storage and transmission rate constraints. The selected systems are representative of many possible photo-imaging systems so that the comparison results will have general application.

To allow comparison, each candidate has been designed to meet the same set of requirements. Both 100-meter mapping and 10-meter spot coverage are included. For example, the effective field of view of each photo-imaging sensor has been made equal so that each can map the surface at the same rate. To accomplish this, three television sensors and three optomechanical scanners are used, with their fields of view arranged as shown in Figure 4-10, to be equivalent to the single photographic camera whose large image format permits a larger field of view for the same spatial resolution. Therefore, the three systems have equal resolutions and mapping coverage rate so that comparisons of their total weight, volume, and power can be made on a common basis.

4.5.5.1 Television

The television system consists of vidicon sensors, camera optics, electronic controls, and a tape recorder. Three of the vidicons are used for the 100-meter resolution mapping; the fourth is used for the 10-meter reconnaissance mission.

The RCA Return Beam Vidicon has been selected for the television photo-imaging system because of its good sensitivity, unusually high spatial resolution, long storage capability, high signal-to-noise ratio, large signal output, and rugged construction. An analog magnetic tape recorder has been selected for this photo-imaging system.

4.5.5.2 Photographic Film

The photographic film system consists of cameras, high definition silver halide film, a film processor, and a readout scanner. The selected film is Kodak SO-243, similar to that used on the Lunar Orbiter. It is a very fine grain aerial photography film that provides a large number of grains in each resolution cell. The film is fairly insensitive so that longer exposure times are needed. This feature, however, also

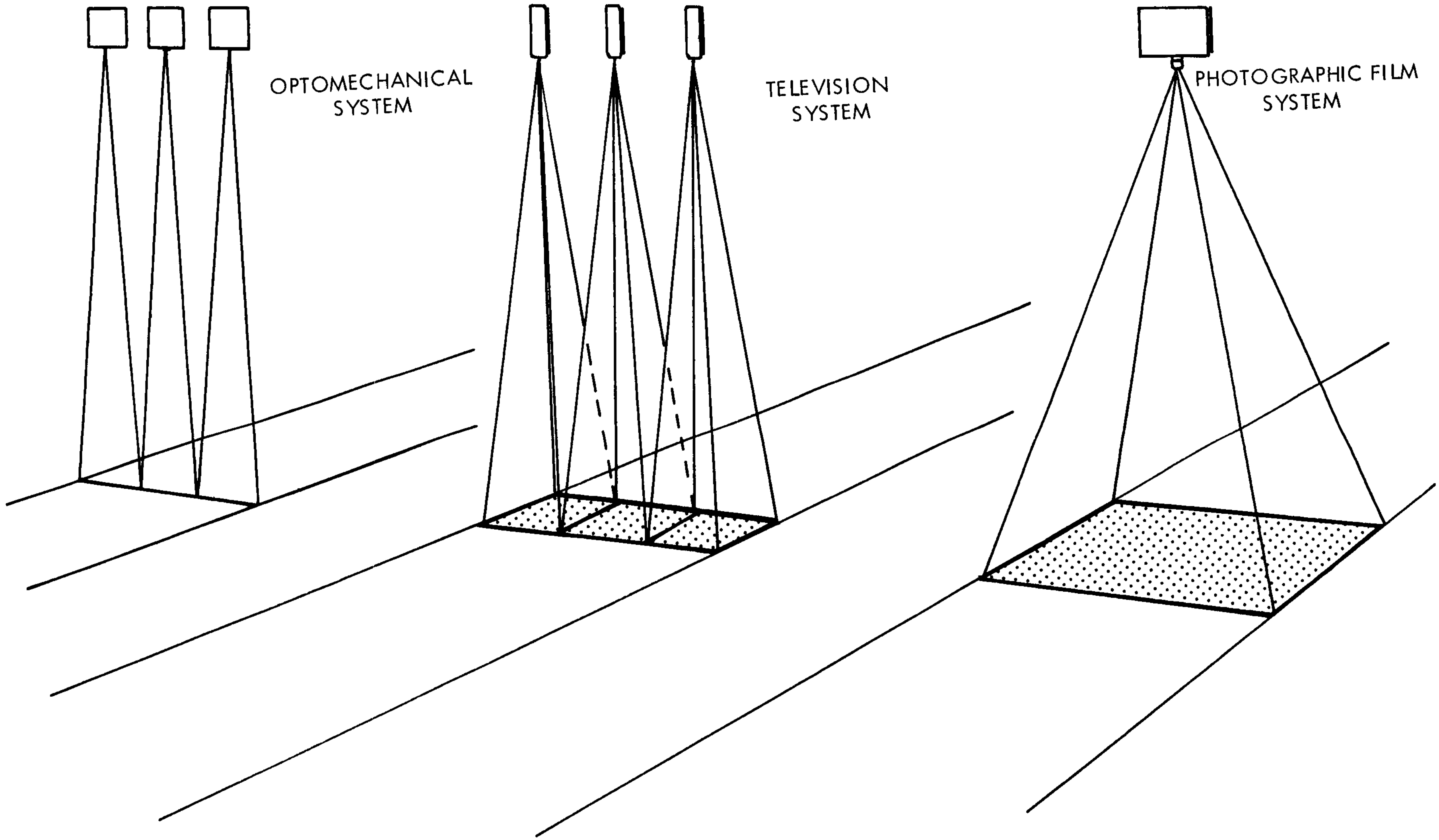

Figure 4-10. Equivalent Coverage of the Three Systems

reduces the fogging due to particle radiation. For the 1973 mission, only moderate shielding is required for film protection.

Two lens systems are used for the single camera. One is used for the mapping mission; the other for reconnaissance.

4.5.5.3 Optomechanical Scan

The optomechanical scan system consists of imaging optics, scanning wheel assemblies, photomultiplier detectors, electronics, and a tape recorder. In contrast to the other two systems which take a series of adjacent photographs to form a total map, this concept produces the map line by line. The optical image is scanned in a direction perpendicular to the direction of ground travel and the resultant signal variations are stored in the tape recorder. Each image line overlaps the preceding trace a small amount. The timing and size of these lines is adjusted to the spacecraft velocity so that a continuous map, with no gaps, results. This technique is analogous to the manner in which the electron beam or the flying spot scanner reads out the image formed by the Vidicon or the photograph.

4.5.6 COMPARISON OF PHOTO-IMAGING SYSTEMS

4.5.6.1 Performance

All three candidate photo-imaging systems meet the requirements for the 100-meter medium resolution mapping mission. Only the television system and the photographic film system can fully meet the 10-meter reconnaissance mission requirements. The optomechanical scan system can achieve a resolution of 10 meters but at the expense of a low signal-to-noise ratio such that a 4 percent contrast is required. It can achieve a 30-meter resolution at 2 percent contrast.

The spatial resolution performance as a function of contrast is also a major difference between film and television. Due to non-uniformities, film will have intensity distortions that limit the resolvable intensity differences it can sense. Also, particle radiation will tend to fog the film, again reducing the lowest contrast it can sense. Television will also have tube surface non-uniformities. However, it is expected that these can be calibrated and will therefore not limit the resolvable intensity. Television also can be limited by the accuracy of the tape recorder.

However, for a low contrast scene, the technique of removing the average background level from the signal reduces this error source to a very small level. The indication at this time is that film will be limited to an intensity resolution of 2 percent while television can achieve 1 percent or lower values. Additional work is required to demonstrate feasibility of attaining the 0.5 percent intensity resolution requirement.

4.5.6.2 Reliability

Our evaluation of the three candidate systems indicates that image motion compensation (IMC) for the low contrast Martian surface is perhaps the most complex and difficult development required. An attempt to increase the optics size, and therefore reduce the exposure time, to minimize the IMC indicates, for the 10-meter film system, that a weight penalty of nearly 300 pounds is needed to reduce the exposure by only 50 percent. Other areas of complexity are the optomechanical scanning wheels for the scan system, the tape recorder for the television system, and the film processor.

Redundancy can be applied to the TV or optomechanical scan system for less weight than the film system. Some redundancy exists in the designs shown due to use of multiple instruments.

A back-up mode that protects against tape recorder failure can be devised for the systems which use this storage means. This method degrades the resolution capability of the television or the optomechanical scan system so that the data rate output is low enough for direct transmission to Earth. The degradation required is a factor of 4, i.e., a resulting resolution capability of 400 meters. The change can be made simply; for example, by reducing the line scan rate and increasing the scanning aperture of the optomechanical scan system, or by reducing the tube scan rate and the number of scan lines of the television system.

4.5.7 IMPACT ON SPACECRAFT

A comparison of the candidate systems relative to their impact on the spacecraft is given in Table 4-1. The magnitude of the differences is not so large that any candidate system can be eliminated. An estimate of the weight impact on the baseline spacecraft of the three 100-meter candidate systems when used with either 10-meter film or TV system is shown in Tables 4-2 and 4-3. The values shown are the increase

Table 4-1. Relative Impact on Spacecraft

	Film	Television	Optomechanical
Telecommunication			
Transmission Rate	←	Same	→
Data Storage	Part of Instrument Repeated Readout Possible	Pushes Recorder State-of-the-Art	Pushes Recorder State-of-the-Art
Attitude Control			
Random Motions During Exposure	←	Same	→
Pointing Accuracy			Some Distortion in Pictures
Environmental Protection			
Thermal	Power Required for Temperature Control of Film		
Radiation	≈ 25 Pounds Shielding No RTG Up to 140 Pounds with RTG in Capsule		

Table 4-2. Weight Impact of Each Candidate Photo-Imaging System (Weights Include Structure, Power, and Propulsion)

10 Meter Systems \ 100 Meter Systems	Television	Photographic Film	Opto-Mechanical Scan
Television	428 pounds	611	189
Photographic Film	944	507	816

Table 4-3. Weight and Volume on the Planet Scan Platform

10 Meter Systems \ 100 Meter Systems	Television	Photographic Film	Opto-Mechanical Scan
Television	200 pounds 8.8 Cu. ft.	222 15.2	150 17.4
Photographic Film	352 18.9	212 15	302 26

in the total spacecraft weight, including the effects on structure, power, and propulsion as well as the actual weight of the candidate system.

The lightest system is the combination of optomechanical scan for 100-meter resolution and television for 10-meter resolution. The heaviest is television for 100-meter resolution and film for 10 meters. It is of interest that any combination of film and television is heavier by at least 100 pounds than the use of either for both resolution missions.

4.5.8 CONCLUSIONS AND RECOMMENDATIONS

The following five items are the major conclusions and recommendations of the photo-imaging study.

4.5.8.1 Mission Requirements

Mapping of most of the planet at 100-meter resolution and spot reconnaissance at 10-meter resolution seem to constitute a good compromise between what can be attained and what is required to satisfy topological, areological, biological, and landing site selection requirements. In addition, a small number of pictures at 1,000-km resolution and a few frames of the entire planet would help determine cloud patterns and planet gross features. While polar regions are of interest, most biologically attractive regions can be covered from a 40 degrees inclined orbit. Topographic mapping will have to be performed under low contrast conditions i.e., intensity resolution of about 0.5 percent will be required. Long wave lengths, towards the red, are recommended for use.

4.5.8.2 Mapping Coverage

For 100-meter resolution mapping the data storage rates limit the field of view of the television and optomechanical scan systems to about 10 degrees. Furthermore, the data transmission rate limits the total area that each candidate photo-imaging system can map during one orbit.

The baseline orbit, while not optimum for photo-imaging, will permit medium resolution mapping of about 32 percent of the total Martian surface within one month's time if a 10-degree field-of-view sensor is used. The baseline orbit is required to satisfy a whole series of orbit constraints, including those designed to enhance capsule descent and landing conditions.

4.5.8.3 Candidate Systems

Based on performance, reliability, availability, and impact upon the spacecraft, the three candidate systems, television, photographic film, and optomechanical scanning, can perform useful photo-imaging experiments of the Martian surface from an orbiting spacecraft. The chief performance differences have been found to be in the sensing of very low contrast scenes, such as are expected on Mars. The lack of uniformity and potential radiation damage are major questions in the low contrast performance of film. Similarly, the design of the optomechanical scan system has a marginal signal output for the high resolution mission.

4.5.8.4 Impact Upon Spacecraft

Evaluating all impacts, such as weight, power, and attitude control accuracy, and expressing them in terms of the total planetary vehicle weight increase above the baseline design, the lightest system is the combination of a medium resolution optomechanical scanner and a high resolution television system. This is due to the simplicity of the scanner when it is used for a medium resolution application, and the low platform weight of the television system.

Among individual systems, the heaviest is television for 100-meter resolution and film for 10 meters. Any combination of film and television is heavier by at least 100 pounds than use of either for both resolution missions.

4.5.8.5 Reliability

Reliability is the major problem of the candidate photo-imaging systems. Although each candidate design was based on work of a previous flight program (Mariner, Nimbus, Lunar Orbiter and ATS), potential unreliabilities are present in each. For example:

a. Television - tape recorder; Vidicon
b. Photographic Film - radiation effect on film; processor
c. Optomechanical Scanner - scanning mechanism
d. General - electromechanical devices; image motion compensation

An image motion compensation sensor for the low contrast Martian surface is the most complex and difficult development required.

The use of three television cameras and of three optomechanical scanners makes possible complete loss of one camera or scanner with the only penalty a reduction in the rate of mapping. On the other hand, if tape recorder failure occurs, the television cameras and optomechanical scanners can scan slower to produce pictures of poorer resolution (400 meters) which can then be telemetered directly back to Earth. These degraded modes of operation could lead to a high degree of realization of mission objectives despite a major element failure.

APPENDIX A

This final report is submitted under terms of Contract NAS 8-22603 under paragraphs A and B, Article I - Scope. Key documents related to the over-all study are listed below:

PROGRAM PLANS

- "Program Plan Voyager Spacecraft System Task D Study" dated July 7, 1967
- "Program Plan Voyager Spacecraft System Phase C Preproposal Activities" dated July 7, 1967

MILESTONE REPORTS

DOCUMENT NUMBER	TITLE
VOY-D-TM-16	Baseline Science Payload Definition
VOY-P-TM-1	Over-all Test Approach Defined
VOY-P-TM-2	Implementation Plan Supplement Material
VOY-P-TM-3	Trajectories and Orbits Defined
VOY-P-TM-4	Propulsion Requirements Defined
VOY-P-TM-5	Test Program Schedules Prepared
VOY-P-TM-6	Test Flow Plans Prepared
VOY-P-TM-7	Test Facility Requirements Defined
VOY-P-TM-8	Preliminary MDE Requirements
VOY-P-TM-10	Application of Opto Electronic Solid State Devices to the Voyager Computer and Sequencer Subsystem

MILESTONE REPORTS (CONTINUED)

DOCUMENT NUMBER	TITLE
VOY-P-TM-11	Level 1, 2, 3 Schedules Voyager Spacecraft System - Preliminary
VOY-P-TM-12	Antenna and Solar Array Sizing Trade Study
VOY-P-TM-13	Propulsion Requirements
VOY-P-TM-14	OSE Concepts Voyager Spacecraft System
VOY-P-TM-15	Application of Large Scale Integrated Circuits to the Voyager Computer and Sequencer
VOY-P-TM-16	Preliminary MDE Requirements Voyager Spacecraft
VOY-P-TM-17	Voyager Mission Analysis and Trajectory Selection
VOY-P-TM-18	OSE Evaluation Criteria
VOY-P-TM-19	Use of Non-Destructive Memory Devices in the Voyager Computer and Sequencer Subsystem
VOY-P-TM-20	Auxiliary Thruster Requirements
VOY-P-TM-21	Interface Test Requirements
VOY-P-TM-22	Response to a Negative Parity to the Memory Readout in the Voyager Computer and Sequencer Subsystem
VOY-P-TM-23	ETO Decontamination Effects on Materials and Systems
VOY-P-TM-24	Circuit Analysis of Signetic Devices
VOY-P-TM-25	Voyager Harness Design Study
VOY-P-TM-26	Review of MSFC PPD-600 Vol. I Preferred Parts List and Design Guidelines for Application to Voyager Project

MILESTONE REPORTS (CONTINUED)

DOCUMENT NUMBER	TITLE
VOY-P-TM-27	Computer and Sequencer Response to Central Power Fault
VOY-D1-TM-3	Lander Interface Definition
VOY-D1-TM-6	Planetary Vehicle Definition
VOY-D1-TM-7	Planetary Vehicle Mapping
VOY-D1-TM-8	Thermal Analysis
VOY-D1-TM-9	Mission Effects
VOY-D1-TR-2	Spacecraft Design for Capsule RTG Interaction Study
VOY-D2-TM-1	Preliminary Voyager/ACE Spacecraft System Definition
VOY-D2-TM-15	Voyager ACE Trade Study
VOY-D3-TM-5	Central Computer Study Function to be Included
VOY-D3-TM-10	Central Computer Study Evaluation Criteria and Methods
VOY-D3-TM-14	Preliminary Indication of Relative Merit Central Computer Study
VOY-D3-TM-22	Central Computer Study Indication of Relative Merit

MILESTONE REPORTS (CONTINUED)

DOCUMENT NUMBER	TITLE
VOY-D4-TM-4	Survey of Present Knowledge of Martian Atmosphere with Application to Spacecraft Design
VOY-D4-TM-17	A Preliminary Estimate of the Solar Cycle Variation of the Atmosphere of Mars
VOY-D4-TM-18	On the Latitudinal and Seasonal Variation of the Martian Atmosphere Below 100 Kilometers
VOY-D5-TM-2	Photo-Imaging Study
VOY-D5-TM-11	Combined Photo-Imaging Mission Specification
VOY-D5-TM-12	Parametric Data Requirements of the Photo-Imaging System
VOY-D5-TM-13	Equipment Performance Data for the Photo-Imaging System
VOY-D5-TM-19	Candidate Photo-Imaging Systems

FINAL REPORT VOLUMES

VOLUME I	SUMMARY
VOLUME II	1973 VOYAGER SPACECRAFT DESCRIPTION
VOLUME III	IMPLEMENTATION PLAN
VOLUME IV	ENGINEERING TASKS