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1. Page 2-7/8, Table II-I.

    a. Telecommunications. Move dot opposite "Launch Antenna" from column a to column b.
    b. Propulsion. Change "22-55 Pound Thrust" to "25-55 Pound Thrust".

2. Page 2-19, Line 3. Change "subcommunication" to "subcommutation".

3. Page 2-24, Line 9. Change "10" to "10^{-5}".

4. Page 2-25, Line 1. Change "PCM" to "PSK"

5. Page 2-77, Table II-7. In second column change 3769.0 to 3469.0.

6. Page 2-91, Paragraph 2.3.8, Line 6. Change "0.5" to "0.2".

7. Page 2-93, Paragraph 2.3.13, Lines 8 and 9. Change "544" to "533"

8. Page 2-95, Figure II-22. Delete the part of Canopus arrow that is within Mar's diameter.

9. Page 2-101, Figure II-25. Delete the words "Stimili Control" on line from box labeled "Subsystem No. 'N"OSE" to box labeled "Data Processing." Add a line marked "Stimili" from box labeled "Subsystem No. 2 OSE" to box labeled "No. 2".

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JULY 30, 1965

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VOYAGER SPACECRAFT SYSTEM PRELIMINARY DESIGN

REPORT SUMMARY

VOLUME F

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for

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<u>TEAM MEMBERS</u>

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Texas Instruments

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TRW Systems Group

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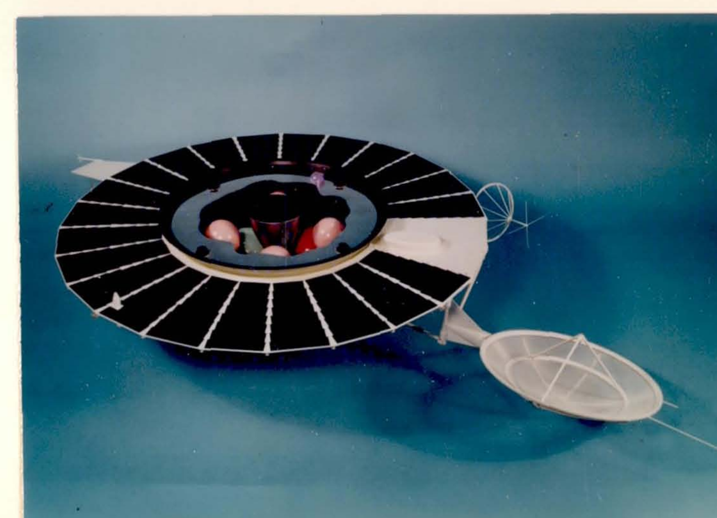
VOYAGER '71 FLIGHT SPACECRAFT PREFERRED DESIGN

LAUNCH CONFIGURATION

CRUISE CONFIGURATION



MARS ORBITER CONFIGURATION

SECTION I

INTRODUCTION

The following pages provide an overall summary of the General Electric report on the Voyager Spacecraft System Preliminary Design Study. This volume is divided into a Technical Summary and an Implementation Plan Summary. The Technical Summary discusses the approach taken to the Spacecraft system design, and briefly describes the 1969 and 1971 recommended spacecraft designs and the alternatives considered. The Implementation Plan Section summarizes the plans to reduce the preliminary design to flight hardware that were prepared during the Phase IA study.

In its proposal for the Voyager Phase IA study, dated February 22, 1965, General Electric identified three major requirements which were considered essential to the successful accomplishment of the program. These were:

a. Achievement of the necessary long-life reliability with a high degree of confidence

b. Strict schedule control to meet a fixed launch window

c. Effective management of the Project and Spacecraft system to achieve the above requirements within the established cost.

These requirements, within the limits of compliance with the Design Objectives and Constraints and the need for achieving design and operational flexibility, became the overriding criteria throughout the conduct of the General Electric study. The study results summarized in this volume are intended to convey the extent to which these requirements have been considered. The Design Status Summary (see Table II-1 in the following section) provides a basis for evaluating the risk inherent in the proposed program approach with respect to reliability, schedule, and flexibility.

The recommended designs for the 1969 and 1971 Spacecraft, the OSE and the implementation plans represent the results of an intensive effort during Phase IA by a team of over 250 experienced General Electric and subcontractor engineering and management personnel.

This team has taken into consideration and fully utilized the experience gained: (1) in performing over 250,000 engineering man-hours of studies sponsored by JPL, NASA and General Electric directly related to the Voyager requirements, and (2) from research and development since 1959 in long-life spacecraft such as Nimbus, OAO and a series of Classified Military Satellite Programs. In addition, and most important, the experience of JPL (and Motorola and Texas Instruments) in the successful Ranger and Mariner programs has been utilized.

Appendix I to this volume lists all the documents that constitute General Electric's Voyager Spacecraft System Preliminary Design Study Report.

SECTION II

TECHNICAL SUMMARY

1.0 VOYAGER DESIGN APPROACH

1.1 FLIGHT SPACECRAFT TASKS

The primary objective of the Voyager program is to perform experiments on the surface of and in orbit about the planet Mars during the 1971, 1973 and subsequent opportunities, in order to obtain information about the existence and nature of extra-terrestial life, the atmospheric surface, and body characteristics of the planet, and the planetary environment.

In fulfilling this objective, the tasks of the Flight Spacecraft are to:

a. Act as a ferry for the Flight Capsule, providing it with power during the transit phase, supplying adequate guidance and the proper separation attitude to allow deflection of the Capsule onto the desired impact trajectory, providing separation commands to the Capsule at the appropriate time, and transmitting, to Earth, Capsule data from lift-off until Capsule impact on Mars.

b. Accommodate the Spacecraft Science Payload, deliver it into an orbit about Mars, and provide it with the required environment, including power, thermal control, minimum electrical and magnetic interference, and proper orientation of the instruments.

c. Maximize the amount of data returned to Earth over the mission duration from the Spacecraft Science Payload.

1.2 MAJOR DESIGN CRITERIA

In designing the Flight Spacecraft to carry out these tasks, many trade-offs are required to achieve an optimum design. The major criteria used in choosing between design alternates are reliability, schedule and flexibility, as discussed below.

1.2.1 RELIABILITY

Particular emphasis is placed on simple and conservative design approaches. Wherever possible, the Voyager design takes advantage of the equipment and techniques developed and the experience gained in the Ranger, Mariner C, and Mariner R designs. Parts, materials, and processes which have demonstrated a history of reliability are used, unless alternates are clearly needed to meet minimum system performance criteria.

Within the weight restraint, functional redundancy is used to provide full capability of critical spacecraft functions despite part or component failures. Useful performance of all critical spacecraft functions will also be possible by back-up modes. Spacecraft functions considered critical include: (1) spacecraft-to-earth communications, (2) continuous sun line attitude control, (3) continuous temperature control, (4) power conversion and regulation, and (5) operation of the earth-to-spacecraft communications and command link.

1.2.2 SCHEDULE

Since the Mars opportunities place absolute constraints on the Project schedule, all design concepts selected must provide assurance that the development can be carried out successfully within the allotted time. As in the case of reliability, this criterion is satisfied by selection of simple and conservative design approaches, and by making use of equipment and techniques that exist from other successful spacecraft programs.

1.2.3 FLEXIBILITY

Since the specific Spacecraft Science Payload is undefined for 1971, and since it will vary from opportunity to opportunity, flexibility in accommodating this equipment and variable mission objectives must be provided.

In addition, the Flight Capsule and the Launch Vehicle are not well defined at this time so that flexibility in accommodating variations in these interfaces is important.

In addition to the three major criteria above, other important considerations in the design choices include cost, magnetic cleanliness and weight.

1.3 RESULTS OF THE DESIGN STUDY

The results of the Voyager Design Study are the recommendation of a preferred design for 1971, discussion of several alternate approaches within the constraints of the Mission Specification, and the evaluation of the 1969 mission and spacecraft design. These results are documented in Volumes A through E. This Technical Summary presents this data in greatly abbreviated form. The following data is presented:

a. Functional description of each of the 1971 Spacecraft Subsystems. The major features, performance parameters, and some of the alternate approaches are discussed.

b. Physical description of each of the major 1971 Spacecraft Assemblies. This description indicates the location of each subsystem and the gross characteristics of the configuration.

c. Description of the 1971 mission sequence and flight operation of the Spacecraft.

d. Description of the 1971 Operational Support Equipment required for assembly, handling, testing, and preparation of the spacecraft for its mission.

e. Discussion of the 1969 Spacecraft mission and design configuration.

Shown in the frontispiece are four views of a model of the 1971 Voyager Spacecraft. The main features of the Spacecraft are identified in Figure II-1.

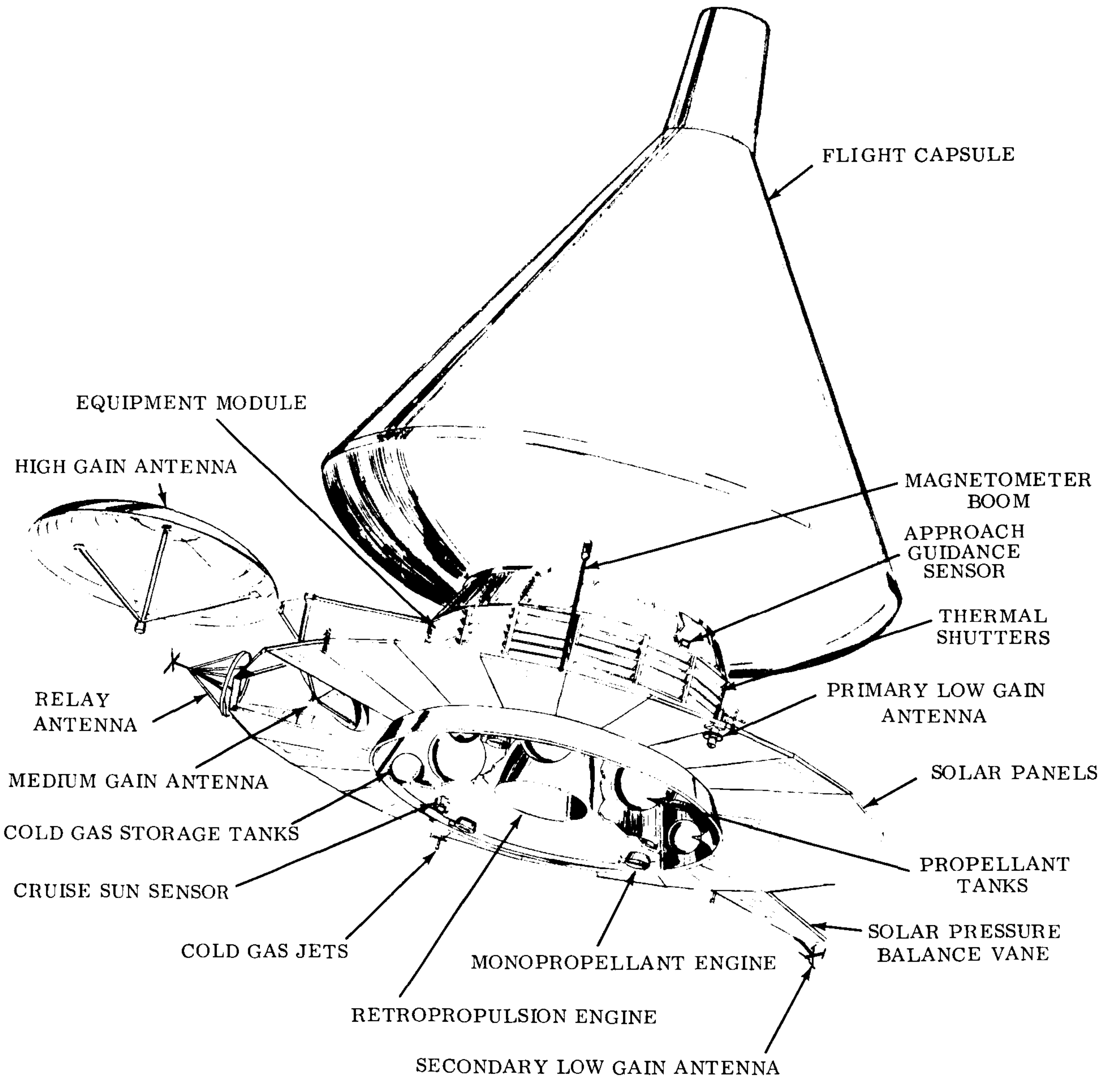


Figure II-1. Features of 1971 Voyager Overall Flight Spacecraft

An evaluation of the recommended design for the Voyager Spacecraft in terms of the major design criteria of reliability, schedule, and flexibility is presented in Table II-1. This Table lists examples for each subsystem of the hardware development status (which affects both reliability and schedule), the redundancy and back-up modes (which affects reliability) and of the provisions for flexibility. Added details are given in each subsystem functional description.

Table II-1. Evaluation of Voyager Spacecraft Design Against Major Design Criteria

SUBSYSTEM	HARDWARE DEVELOPMENT STATUS: ITEM	RELATED PROGRAM	a*	b*	c*		REDUNDANCY AND BACK-UP MODES	MISSION FLEXIBILITY
TELECOMMUNICATIONS	Transponder	Mariner C	●				Three Transponders	
	Power Amplifier (50 W)				●	Apollo (20W)	Three Power Amplifiers	
	Relay Detector			●				
	Relay Receiver			●				
	Tape Recorder				●	JPL Transport (10^8)	Three Tape Recorders	Maximum data storage (6 x 10^8 bits)
	~~Magnetic Core Memories~~			●				
	Data Encoder Subsystem				●	Integrated Circuits	Three Analog-to- Digital Converters	Buffer storage for low-data-rate science data and for Eng. & Capsule data during maneuvers
	Command Detector	Mariner C	●				Three Detectors	
	Command Decoder				●	Integrated Circuits	Dual Decoders	
	Primary Low Gain Antenna	Mariner C	●					
	Secondary Low Gain Antenna			●			Secondary Low Gain Antenna is Back-up to High Gain Antenna during Maneuvers	
	Relay Antenna			●				
	Launch Antenna	Mariner C	●					
	High Gain Antenna			●				Maximum bit rate (8533 1/3 bps)
	Medium Gain Antenna	Mariner C	●				Medium Gain Antenna is back-up to High Gain Antenna at encounter	
PROPULSION	Retropropulsion							1975 and 1977 fly-by missions can be accommodated by removal of retropropulsion
	Thrust Chamber				●	Saint and Apollo Subscale Tests		
	Fuel Tanks			●				
	Oxidizer Tanks			●				Liquid bipropellant allows variability in total impulse
	Pressurization Tanks			●			Pressure Isolation Valves	
	Regulator			●			Redundant Regulators	
	Valves			●			Redundant Shut-off Valves	Tolerant of lateral center of mass offset
	Midcourse Propulsion							Low-thrust monopropellant allows minimum impulse for guidance corrections
	Thrust Chamber				●	22-55 Pound Thrust (Throttleable)		
	Propellant Tank	Mariner C	●					
	Pressurization Tank			●			Isolation Valves for dormant periods	
	Throttling Valve			●				Thrust vector control insensitive to longitudinal location of center of mass.
	Jet Vane and Actuator			●				
	Regulator			●			Redundant Regulators	
	Valves			●			Quad-redundant Valves	
GUIDANCE AND CONTROL	Sun Sensors	OAO	●					Planet Pointing Capability
	Canopus Star Sensor	Mariner C	●				Inertial Control as back-up to Optical Sensors	• Covers sunlit side plus 10° before and beyond terminator
	Integrating Gyros	Mariner C, OAO Class. AF Sat.	●					• Directed to local vertical or to selected targets
	Accelerometer	LEM, Minuteman	●				C&S back-up for Engine Cut-off	
	Approach Guidance Sensor				●	Ames Contract		• Has rates less than 0.01 mr/sec.
	Attitude Control Electronics	Mariner C, OAO Surveyor	●				Triple redundancy / majority voting for attitude control electronics	• Relatively insensitive to orbit parameters
	Autopilot Electronics	Mariner C, Surveyor	●					
	Cold Gas Jet Subsystem	Nimbus, Mariner C, Surveyor	●				Dual Cold Gas Jet Subsystems as in Mariner C	
	Antenna Actuators	Mariner C, Gemini	●					
	Scan Platform Actuators	Mariner C, Gemini	●				Dual Series Solenoid Valves	Control system relatively insensitive to vehicle inertias
	Mars Vertical Sensor			●		ATD Dither Scan		
	Jet Vane Actuators	Mariner C	●				Redundant Jet Vanes for roll control during maneuvers	
	Throttling Valve Actuators			●			Back-up register for Spacecraft turns.	Three axis control capability during Sun or Canopus occultations
ELECTRICAL POWER	Batteries				●	Separator Thickness	Three batteries with excess capacity	Power Distributed as AC
	Main Buck Regulator			●			Dual Main Buck Regulators	
	Charge Regulator			●				Batteries sized for three-hour occultation with less than 60% discharge
	400 cycle Inverter	Mariner C	●				Dual 400-cycle Inverter	
	2400 cycle Inverter	Mariner C,	●				Dual 2400-cycle Inverter	
	Synchronizer	Mariner C	●					
	Power Switching Logic			●				
	Solar Panels			●			16% minimum margin for Solar Panels	
							Frequency standard backup to C&S	
TEMPERATURE CONTROL	Thermal Shutters	Mariner C, Nimbus	●				Allows partial failure of shutter Control	Allows maximum change in Internal Dissipations
	Thermal Shutter Controls	Nimbus	●				Redundant Shutter Actuators	
	Contingency Heaters	Class. AF Sat, Mariner C	●				Redundant Gyro Heaters	
	Insulation	Nimbus, Class. AF Sat, Mariner C	●					Tolerant to Abnormal equipment dissipations
	Coatings	Nimbus, Class. AF Sat.	●					
CONTROLLER AND SEQUENCER	Controller and Sequencer			●			Triple redundancy/ majority voting	255 command storage; updated by ground command
							Sequence Timer provides redundancy for master timer	Minimizes need for ground command
							Ground Command provides back-up for all critical functions	Ample commands for Science control
							Three memories which can be loaded separately	Allows variations in orbit sequencing
PYROTECHNIC	Pyrotechnic Controller	Mariner C	●				Dual electronics	Two events can occur in close sucession
	Explosive Devices	Class. AF Sat, Mariner C	●				Shorted bridge wire can be tolerated	
							Dual Bridge Wires in Squibs	
							Redundant arming switches	

*NOTES:

Column a: Modification of Flight Hardware
Column b: Standard Design Approach
Column c: New Development

2.0 1971 SPACECRAFT PREFERRED DESIGN

2.1 FUNCTIONAL DESCRIPTION

The recommended design for the 1971 Voyager Flight Spacecraft contains the following subsystems:

a. Spacecraft Science Payload which collects the desired data.

b. Telecommunication Subsystem consisting of:
   1. Data Handling and Storage Subsystems which processes both engineering and science data.
   2. Spacecraft Radio which provides the capability for two-way communication between spacecraft and earth.
   3. Relay Radio which provides for receiving capsule transmissions during entry.
   4. Command Subsystem to provide capability for ground based control of spacecraft functions.

c. Propulsion for accomplishing trajectory corrections during the transit phase and for inserting the Flight Spacecraft into a Martian orbit.

d. Guidance and Control Subsystem consisting of:
   1. Attitude Control which provides three axis stabilization of the spacecraft.
   2. Cold Gas Jets which provide torques on the vehicle.
   3. Autopilot Subsystem which controls vehicle attitude during engine firings.
   4. Articulation Subsystem which points the high gain antenna to earth and the science instruments to the planet Mars.
   5. Approach Guidance which makes measurements relative to Mars so as to improve navigation accuracy.

e. Power Subsystem which supplies necessary electrical power to all operating equipment.

f. Temperature control to provide a suitable thermal environment for all equipment.

g. Controller and Sequencer which provides on-board control of spacecraft functions.

h. Pyrotechnic Subsystem to accomplish all explosively actuated events.

i. Structure to provide support for all equipment.

The following sections summarize the design approach for these subsystems and the major considerations in arriving at the recommended design.

2.1.1 SPACECRAFT SCIENCE PAYLOAD

While the specific Science Payload is not defined, the types of instruments likely to be employed and the requirements imposed on the spacecraft are well known. Payload instruments have been categorized in the Mission Guidelines, JPL Project Document No. 46, according to the general objectives of the investigations as: (1) Planetary Observations and (2) Planetary - Interplanetary Environment Observations. They are further divided physically into: (1) Primary Sensor and Directly Associated Hardware and (2) Remote Hardware - Data Automation Equipment (DAE). Provisions that have been made in the Spacecraft Design to accommodate the Science Payload are described briefly in the following paragraphs. Figure II-2 shows the location of the Science Payload on the 1971 Flight Spacecraft.

The Primary Sensors and Directly Associated hardware for Planetary Observations will be mounted in a Scan Platform that is normally oriented to the Mars local vertical. Orientation is achieved through three gimbals: two are commanded to a position such that an axis is erected normal to the orbit plane, and the third rotates the scan platform about this axis under control of a horizon sensor. Viewing of Mars over the sunlit side and up to 10 degrees before and beyond the terminators is possible. Since each gimbal can be controlled by command, viewing of other than the sub-spacecraft point can also be easily accomplished.

Other orientation systems involving fewer gimbals were investigated and are described in Volume B. Depending upon the specific science that is carried, some reduction in complexity may be possible with a rather small penalty in scientific mission value. The three gimbal system is recommended at this time since it provides the greatest flexibility for accommodating scientific requirements.

Other features of the Scan Platform are:

a. A total of 5 cubic feet of volume is available for instruments.

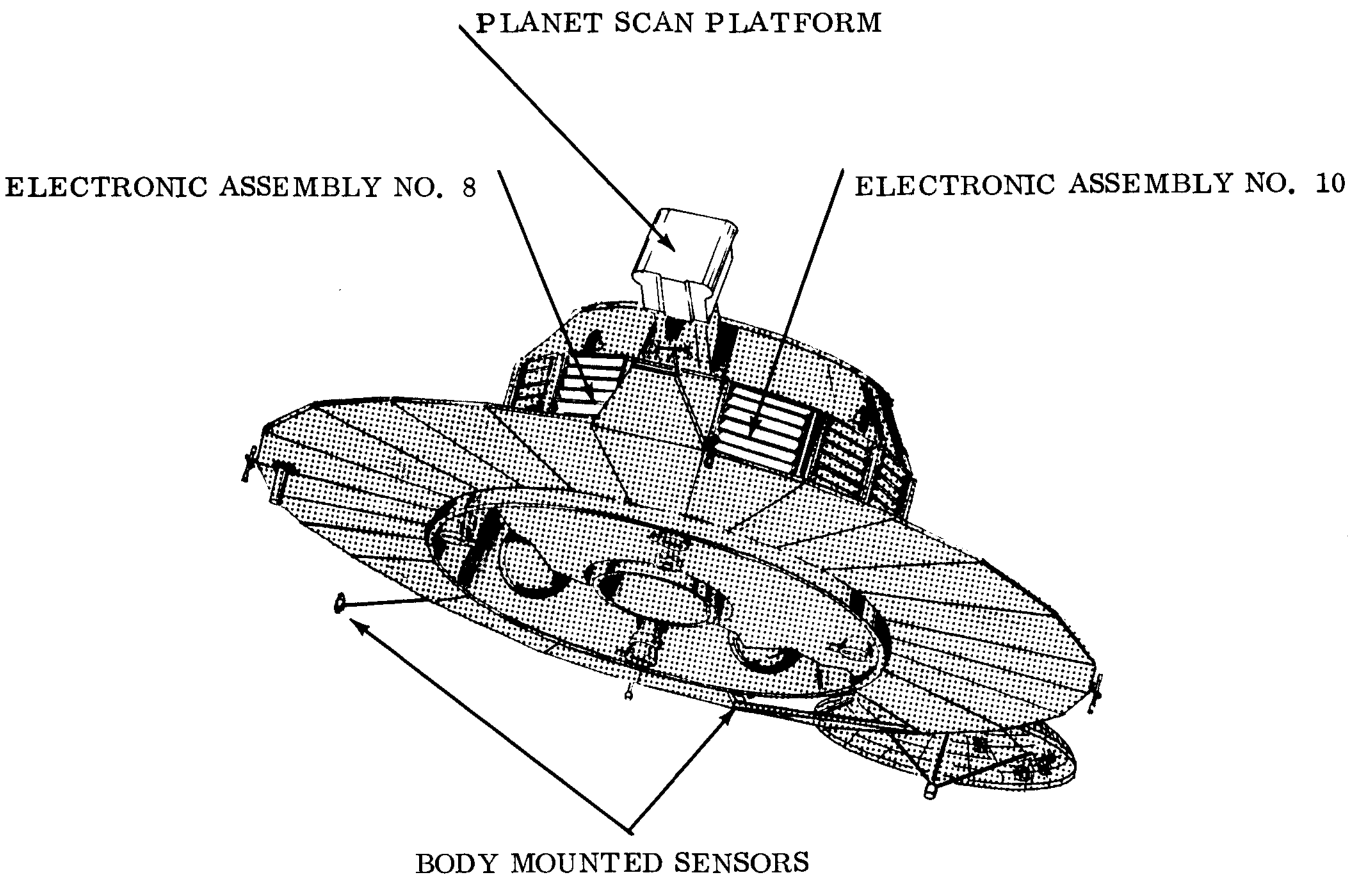


Figure II-2. Location of the Science Payload on the 1971 Voyager Flight Spacecraft

b. It is located on the shady side of the solar array so that it is not exposed to sunlight. This minimizes the possibility of reflected light interfering with optical measurements and provides a capability for close temperature control.

c. Pointing error to the desired direction will not exceed 1.8 degrees (31.4 mr).

d. Motion of the gimbals is produced by stepping motors. When not being stepped, the angular rate is the vehicle deadband rate which will not exceed 0.006^{o}/second (0.01 mr/second). A signal is provided to the DAE to inhibit picture taking, if desired, when the gimbal position is being stepped.

Primary Sensors and Directly Associated Hardware for Planetary - Interplanetary Environment Observations will be located in various places on the Spacecraft. This category includes such experiments as magnetometers, energetic radiation detectors, which can be attached to the Spacecraft Bus body. The following considerations have been made to provide for these sensors:

a. The Vehicle environment has been made as non-interfering as possible. The magnetic field of the Spacecraft has been made as small and as stable as possible. Care has been taken in the design of electrical equipment to minimize the possibility of electromagnetic interference. No sources of nuclear radiation exist in the Spacecraft.

b. Ample space for locating these instruments is available.

The Remote Hardware - Data Automation Equipment is located within the Spacecraft Bus. Two equipment bays providing a total volume of 4.2 cubic feet are provided. No other Spacecraft equipment is mounted in these two bays.

The primary electrical interface between the Spacecraft Bus and the Science Payload occurs with this equipment. The following has been provided for in the design to insure flexibility in dealing with a variety of science payloads:

a. Power is distributed as a 2400 cps square wave with a transformer-rectifier (T/R) at the user end to generate the desired voltage levels. This avoids re-design of the basic Spacecraft Power subsystem to accommodate varying user requirements.

b. A total of 34 or more discrete commands to the DAE have been provided for control of the Science Payload. These commands can be provided at a fixed time in each orbit, or the timing can be varied by ground command. In addition, quantitative commands can be sent to the DAE.

600 million.

c. A bulk storage capacity of 6×10^8 bits is provided to accomodate all the data that can be transmitted at a maximum data rate in one orbit.

d. Buffer storage is provided for slow data rate science for temporary storage during periods when bulk storage data is being transmitted.

The orbit selected for the nominal design has a periapsis altitude of 3000 km and an apoapsis altitude of 25000 km with a period of 19.3 hours. The inclination to the Mars equator is 40^o. However, the Spacecraft design has been made as independent of the specific orbit selected as possible to allow changes based on mission requirements. Examples are:

a. The power system can tolerate Sun occultations up to 3 hours without exceeding 56 percent depth of discharge of the batteries.

b. The control system can tolerate occultation of the Sun or Canopus without loss of stabilization.

c. The sequence timer which controls orbital operations has a maximum duration of 72 hours.

Major changes in the orbital parameters may require relocation or modification to the Scan Platform to optimize viewing. No other changes would be required.

2.1.2 TELECOMMUNICATION SUBSYSTEM

The 1971 Voyager Telecommunication System represents a logical extension of the techniques and equipment designs developed and demonstrated by JPL on the Ranger and Mariner programs. The increased size of the Voyager spacecraft has permitted significant increases in storage capacity, and in reliability, through equipment redundancy. The increased size of Voyager, coupled with the increased capability of the DSIF, permits a large increase in transmission capability. Extensive use of integrated circuits for digital functions has reduced the weight of many elements of the Voyager Telecommunication System, compared to their counterparts in the Mariner System.

The Telecommunication Subsystem performs the following general functions:

a. Telemeters the following types of data to the DSIF stations:

   1. Planetary scan instrument data
   2. Planetary-interplanetary environment data
   3. Capsule data
   4. Spacecraft engineering data
      a) Operational support
      b) Design verification
      c) Failure diagnosis

b. Detects and decodes commands from the DSIF stations to the recipient Spacecraft subsystem on the Spacecraft and Capsule for the control of:

   1. Operation and calibration of the DAE and Science Payload
   2. In-flight operation of the spacecraft and capsule subsystem such as:
      - Interplanetary trajectory corrections
      - Updating antenna and Canopus Sensor pointing sequences
      - Updating science data collection sequencing
   3. Correction of failures

c. Provides the tracking transponder used to determine the relative angular position, velocity and range of the Spacecraft from the stations of the DSIF to enable the Space Flight Operations Facility to compute the parameters of the trajectory.

The equipment required to perform these functions is represented by the block diagram of Figure II-3 and by the Spacecraft diagram (showing equipment location) of Figure II-4. The Telecommunication Subsystem consists of four major hardware subsystems:

a. Data Handling and Storage Subsystem
b. Radio Subsystem for Communication with the DSIF
c. Radio Relay Subsystem for Communication with the Flight Capsule
d. Command Subsystem

General Electric's subcontractors for the above subsystems were Texas Instruments, Inc. for the Data Handling and Storage, and Motorola, Inc. for the Radio, Radio Relay and Command.

2.1.2.1 DATA HANDLING AND STORAGE SUBSYSTEM

The Data Handling and Storage Subsystem consists of the data encoder (commutator, analog to digital converters, data selector and subcarrier modulator) and the data storage subsystem (buffers and tape recorders). As shown on the block diagram (Figure II-3), the subsystem accepts both analog and digital data from the engineering subsystems (280 inputs), from both high (50,000 bps) and low (100 bps) data rate science sensors, from science flare data, and from the Capsule (10 bps) either hard line prior to Capsule separation or via the relay radio subsystem after separation.

The Data Handling and Storage Subsystem provides this data in the form of a composite binary data signal combined with a pseudo-random noise (PN) waveform and a reference sub-carrier to the Radio Subsystem for transmission to Earth. Data rates range from 3-1/3 bits per second (bps) used for the transmission of engineering data through a low gain antenna during maneuver turns, up to 8,533-1/3 bps for the transmission of planetary scan instrument data combined with non-scan science and engineering data during orbital operations. Formats for data transmission are selected by command as appropriate to the mission phase.

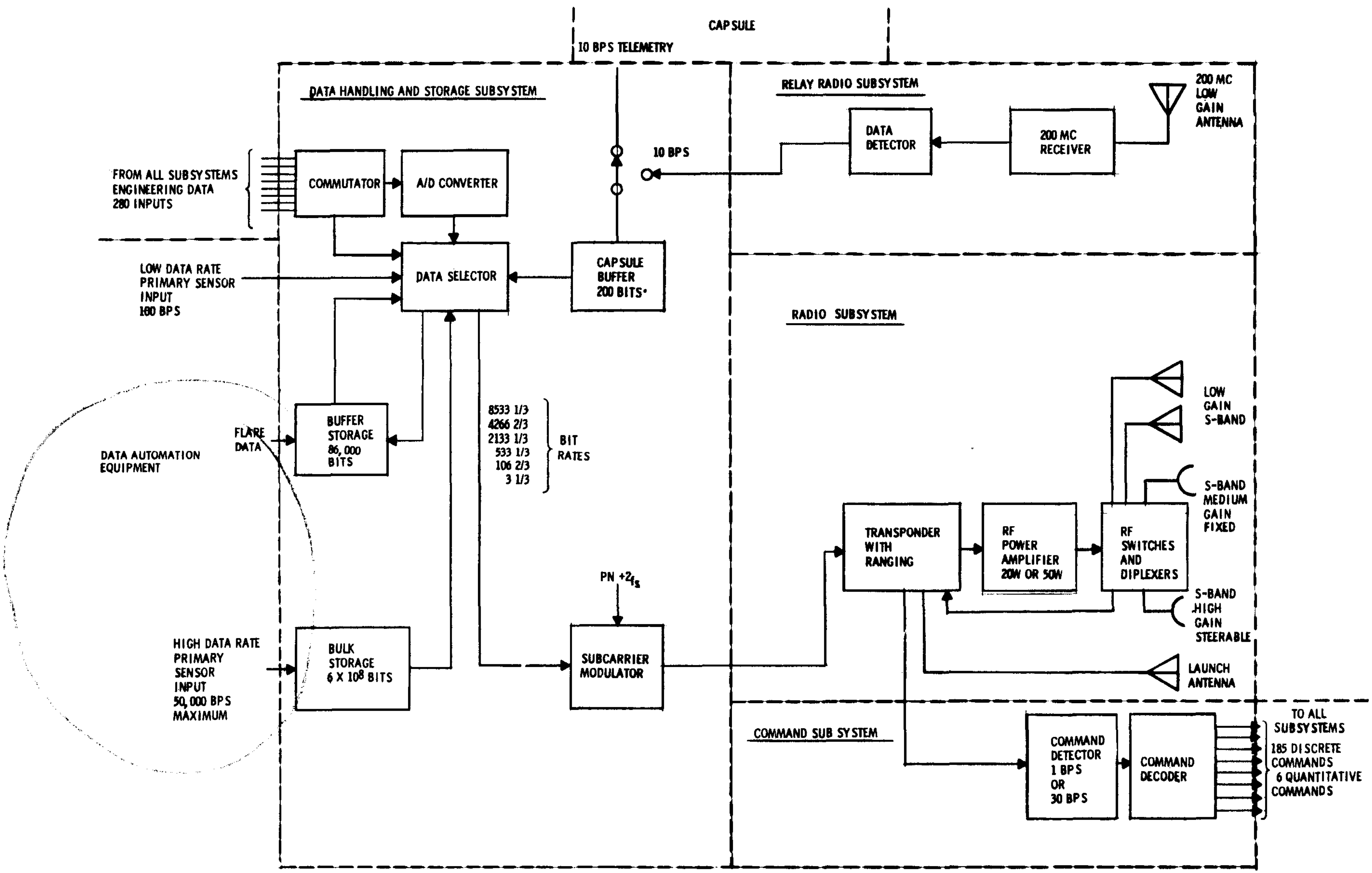


Figure II-3. Flight Spacecraft Telecommunication Subsystem Block Diagram

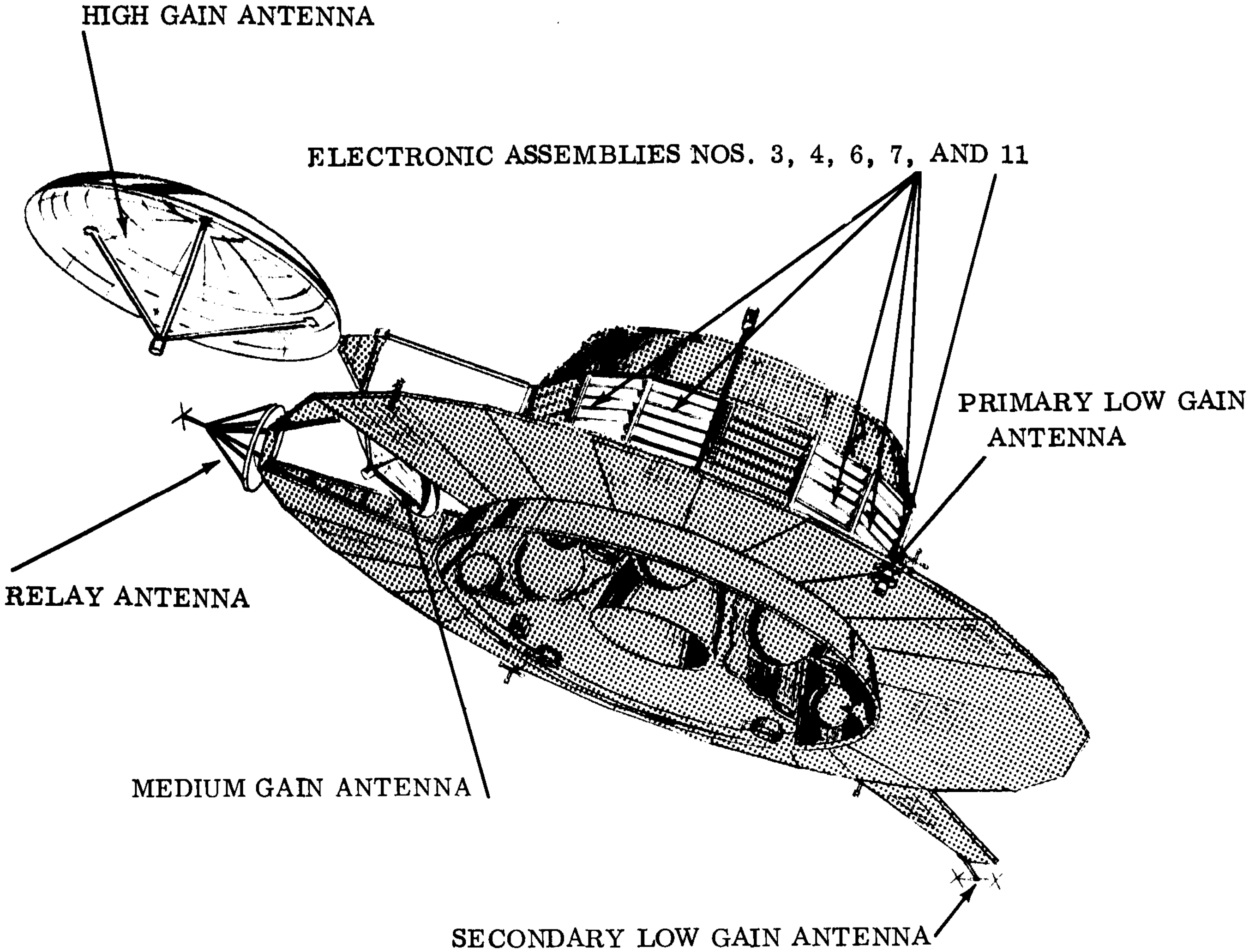


Figure II-4. Location of Telecommunication Subsystem on the 1971 Flight Spacecraft

The Data Encoder Subsystem commutates and encodes the engineering signals into seven bit binary words. The commutator has an addressable high speed deck with two levels of fixed ratio subcommunication. This design permits formats to be changed by command with a minimum of serial switching elements. The commutator design incorporates a high degree of failure isolation which makes block redundancy unnecessary to protect against part failures. Three analog to digital converters are provided. They are connected in parallel and selected with commands by switching power to any one of them. Digital accumulators are provided for conditioning and encoding pulse type inputs into seven bit words. The encoded digital and analog samples are combined and converted into a serial NRZ binary signal. Non-scanned, planetary and interplanetary body located science instruments have their data encoded by the DAE into a serial NRZ digital signal. This data is transferred in real time to the data encoder. The digital data is combined with the encoded analog data to form the composite binary data signal. The Data Storage Subsystem allows real time data to be stored prior to transmission and permits synchronization of different data formats. For example, after separation, Capsule data is collected from the Capsule Relay Radio Subsystem via the Spacecraft Data Storage Subsystem. Small buffer registers in the storage subsystem allow the Capsule relay data to be synchronously formatted with the Bus real time data.

In addition to the Capsule buffer storage, two other types of data storage are provided by the subsystem. Medium capacity storage is provided by three 28,665 bit magnetic core memories. High capacity bulk storage is provided by three 2×10^8 bit magnetic tape recorders.

The medium capacity buffers collect engineering and non-scan science data during playback of the magnetic tape recorders in orbital operations. During gaps in the scan data, the buffer dumps the engineering and non-scan science data for transmission.

These buffers also store engineering and capsule data during maneuver turns for later transmission, and store the capsule entry data as a back-up to the real time transmission of this important data. These buffers may also be used to store DAE science flare data.

A high capacity storage subsystem is used to temporarily store the DAE scan-instrument data. The storage capacity was sized to accommodate all the data that can be transmitted at the maximum data rate in a normal orbit. For a nominal 19.3 hour orbit, 5.9×10^8 bits may be transmitted at 8533 1/3 bps. A total capacity of 6×10^8 bits in the form of three 2×10^8 bit tape recorders is provided. This capacity exceeds by 60 percent the capacity required to continuously accept the 50,000 bps data from the typical science package described in the Mission Guidelines for the normal two hour period of observation of the illuminated half of the planet. Thus, the full data recovery requirement can be met even if one of the recorders fails.

The Data Handling and Storage Subsystem occupies Electronic Assemblies 6 and 7 of the Spacecraft Equipment Module. The total weight, including mounting structure, is 112.6 pounds.

2.1.2.2 RADIO SUBSYSTEM

The Radio Subsystem consists of transponders, power amplifiers, S-band antennas, and r.f. switching and diplexing.

The transponder phase modulates the telemetry subcarrier signal onto the S-band carrier. The transponder r.f. source is derived either from an internal crystal controlled oscillator, or from the phase lock command receiver, when the receiver has acquired a ground transmitted signal. Three transponders are provided, selectable by ground command switching the power supply. Each transponder is passively coupled to at least two power amplifiers through hybrids. The power amplifiers are also selectable by command and include a 20-watt traveling wave tube (TWT), and two 50-watt power amplifiers which may be either of the TWT or electrostatically focused klystron (ESFK) type. One of the 50-watt amplifiers may also be operated at a 20-watt level. The 50-watt capability is provided to maximize the data transmission capability but requires some development. Both the TWT and ESFK have already been qualified for space use at a 20-watt power level. The extension to 50 watts appears to be a reasonable next step. The primary power required can be supplied within the overall weight constraint, and the thermal dissipation requirements can be met with straightforward techniques.

The three power amplifiers may be connected to any of the four main antennas:

a. 7.5-foot parabolic reflector high gain antenna
b. Mariner C type elliptical reflector medium gain antenna
c. Mariner C type primary low gain antenna
d. Skewed dipole secondary low gain antenna

The high gain antenna is the largest rigid parabolic reflector antenna which could be stored for the selected vehicle configuration. It provides a peak gain of 32.5 db which, for a pointing error of one degree, is degraded only 0.7 db. With the 50 watt power amplifier, a data rate of 8533-1/3 bps can be obtained at a range of 290×10^6 km.

This antenna is deployed at separation from the Launch Vehicle, and is pointed to Earth by a Controller and Sequencer program which may be updated by command. It can be rotated approximately $\pm$ 20 degrees about the Spacecraft pitch axis and 225 degrees about an axis perpendicular to the pitch axis. This amount of freedom allows the antenna to be aimed to the Earth for any orientation of the thrust (roll) axis during maneuvers. Use of the high-gain antenna during maneuvers requires this large angular freedom in one antenna gimbal axis, as compared to a relatively small angle required for normal cruise or orbital operation. The benefits obtained by providing this capability are several-fold:

a. It provides an excellent verification of proper attitude before committing to Capsule separation or engine firing.

b. It allows direct relay of Capsule data during entry while the spacecraft is in the orbit injection attitude.

c. It allows Doppler tracking during orbit injection, which would be marginal using the low gain antenna.

d. It permits the required storage capacity for the capsule data (10 bps) to be approximately halved (58,000 bits vs. 90,000 bits).

In order to have a back-up capability, a Mariner C type non-deployed, non-steerable antenna is employed. With a peak gain of 23.5 db, this antenna will enable a data rate of 533 1/3 bps to be achieved for a period of up to five months of late cruise and orbital operations.

Additionally, two low gain antennas are provided. The primary low gain antenna is of the Mariner C low gain antenna configuration, and provides approximately hemispherical coverage in the normal sun pointing direction. Should the vehicle lose Canopus reference, this antenna furnishes good coverage for cone angles of up to 100 degrees. The secondary low gain antenna, an array of skewed dipoles, provides a toroidal pattern with the toroidal plane approximately in the ecliptic plane. This antenna furnishes a back-up means of obtaining telemetry data after the Spacecraft has achieved maneuver attitude, and also provides the broadest angular coverage for non-normal attitudes.

The different functions which the radio subsystem must perform in normal operations can be accomplished without any r.f. switching until after the orbit is achieved, when a single r.f. switch is operated to connect a 50-watt power amplifier to the high gain antenna. All prior changes involve only power switching. Switching at r.f. may also be required in the event of failure.

The launch radio subsystem configuration comprises an exciter and a turnstile antenna. A fraction (100 mw) of the output of the exciter is passively coupled to the launch antenna.

Until separation, while the Launch Vehicle shroud is in place, radiation is via a coupler and parasitic antenna on the shroud.

The phase lock receiver employs the Mark I transponder receiver design, except improved sensitivity is obtained through the use of new components in the front end, yielding a threshold carrier sensitivity of -153.6 db. Commands may be received out to a range of 2000 x 10^6 km through the high-gain antenna, and to > 330 x 10^6 km through the primary low gain antenna, using the 100-kw transmitter and the 85-foot DSIF antennas. Under emergency conditions, the 210-foot antenna may be used to obtain an additional 10 db margin. The receiver also provides for doppler tracking and turn-around ranging. All three receivers operate continuously.

The Radio Subsystem occupies Electronic Assemblies 3 and 4 of the Spacecraft Equipment Module and includes the several antennas located on the Solar Array Assembly. The total subsystem weight is 159.0 pounds.

2.1.2.3 COMMAND SUBSYSTEM

The Command Subsystem, consisting of the command detectors and decoders, is based upon, and is an expansion of, the Mariner B and C Command systems. The command detector accepts the PN sync and data subcarriers from the receiver and recovers bit and word timing as well as the command data. Two command rates are available: one sub-bit per second and 30 sub-bits per second, corresponding to 0.5 and 15 command bits per second. The high rate capability is provided for transmitting long programs to the Controller and Sequencer (C & S) and the DAE, and is available as long as the high-gain antenna is functioning. The low rate capability is provided to achieve long range under non-nominal conditions. The sub-bits are compared in the command decoder to verify that no errors exist and, if accepted, are translated to provide up to 246 quantitative and discrete commands.

The PN sync subcarrier for the 1 sub-bit per second rate is identical to that used on Mariner C, and the 30 sub-bit per second rate employs a PN sequence 1/30 as long as the Mariner C sequence, so that the PN bit rate is approximately the same for both systems. Three detectors are provided: one at the 30 sub-bit per second rate that can be switched to any receiver, and two at the 1 sub-bit per second rate that are always connected to the same receivers. The one sub-bit per second detector is the same as the Mariner C design, except that in addition to the sync loop bandwidth of 2 cps employed in Mariner C, a bandwidth of 0.5 cps is provided, which is automatically used after acquisition. The sensitivity of the detector for 10 sub-bit error rate is thereby improved to 16 db referenced to a noise bandwidth of 1 cps, compared to 18.5 db if the 2 cps loop bandwidth only were available.

The command decoder uses integrated circuitry throughout, and is completely redundant. The output isolation switches of both decoders are tied in parallel. The Command Subsystem occupies Electronic Assembly 11 of the Spacecraft Equipment Module. It weighs 38.2 pounds.

2.1.2.4 RELAY RADIO SUBSYSTEM

The Relay Radio Subsystem consists of the antenna and a single preselector and preamplifier which feeds two receiver and detector combinations. It receives Flight Capsule telemetry data and provides it to the Data Handling and Storage Subsystem after the Capsule separates from the Spacecraft. An output selector provides data, bit sync and an indicator of "signal present" from one receiver/detector only. The selector bases its decision on receiver AGC and a detector lock indication. Unless both the AGC and detector lock indications are satisfactory, the other receiver/detector combination will be chosen. Only one combination will be used.

The system operates in the VHF band, nominally at 200 mc. From a link performance standpoint only, a lower frequency would be desirable, but 200 mc was chosen as a reasonable trade-off of link efficiency and antenna size.

A coherent system, employing PCM/FM/PM data modulation, was chosen, and even with an interfering broadband signal producing -150 dbm/cps at the receiver input, a 10 bps link is achieved at 8,000 km range. Bit sync for matched filter data detection is obtained by modulating the subcarrier with a tone whose frequency equals the data rate.

The Relay Radio Subsystem is also located in Electronic Assembly 4 (with part of the Radio Subsystem). It weighs 12.6 pounds.

2.1.3 PROPULSION SUBSYSTEM

The propulsion subsystem recommended for the Voyager Spacecraft consists of a monopropellant hydrazine subsystem for midcourse trajectory corrections and a liquid bipropellant subsystem for retropropulsion for orbit insertion. These two subsystems are shown on the Spacecraft diagram of Figure II-5. The bipropellant subsystem has a single fixed thrust chamber. The monopropellant subsystem has four (4) thrust chambers which are throttled for thrust vector control in the pitch and yaw planes during all maneuvers, including orbit insertion. Roll control is achieved through the use of a single jet vane in each monopropellant thrust chamber. Gimbaling is, therefore, not required for the bipropellant thrust chamber. Simplicity of design and operation is stressed, with redundancy of components specified where such redundancy contributes significantly to the probability of overall mission success. The selected propulsion subsystem can meet all proposed mission requirements and, at the same time, offer the flexibility necessary to satisfy modified mission requirements.

In selecting the propulsion subsystem, a large number of possible approaches were considered. The three major candidates that evolved are shown in Table II-2 and a comparison of their primary features is shown in Table II-3. All three of these approaches are satisfactory. Analysis indicates each can successfully perform the Voyager mission.

The single bipropellant engine for both midcourse corrections and orbit injection is the lightest and least complex (and consequently, the most reliable) propulsion system considered. It was not selected as the preferred design during the Phase IA Study because the relative difficulty of autopilot control and propellant acquisition appeared to require additional development to achieve reliable solutions. This decision is further discussed in Volume B.

The bipropellant liquid engine plus four monopropellant engines was selected as the perferred design because of the following advantages:

a. Location of the center of mass along the roll axis is of no concern from a control standpoint. This allows significant flexibility in dealing with an unknown capsule center of mass location.

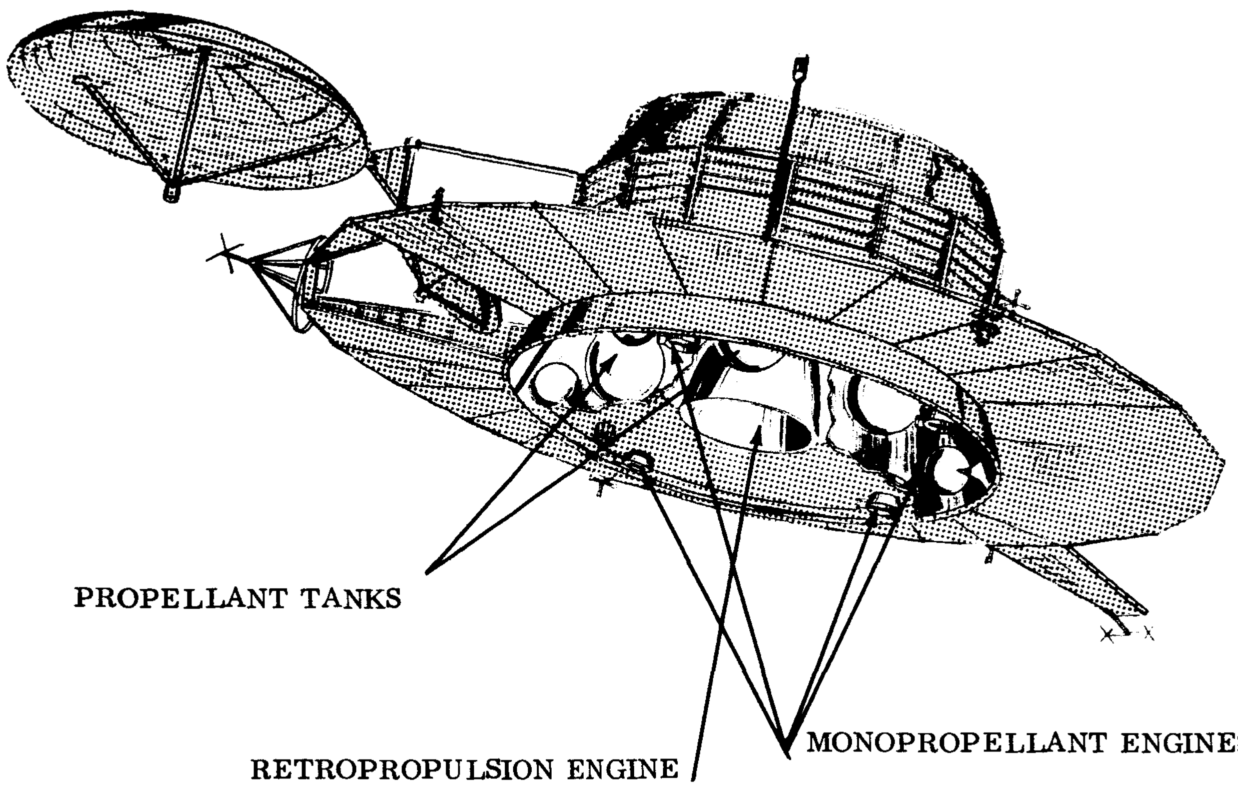


Figure II-5. Location of Propulsion Subsystems on the 1971 Flight Spacecraft

TABLE II-2

MAJOR PROPULSION CANDIDATES

ORBIT INJECTION PROPULSION	MIDCOURSE PROPULSION	PITCH AND YAW CONTROL DURING ORBIT INJECTION	ROLL CONTROL DURING ORBIT INJECTION	PITCH AND YAW CONTROL DURING MIDCOURSE	ROLL CONTROL DURING MIDCOURSE
Bipropellant Liquid Engine	Uses Orbit Injection Propulsion	Gimbaled Engine	Cold Gas	Gimbaled Engine	Cold Gas
Bipropellant Liquid Engine	4 Monoprop-ellant Engines	Throttleable Monopropellant Engines	Jet Vane	Throtteable Monopropellant Engines	Jet Vane
Solid Rocket	4 Monoprop-ellant Engines	Throttleable Monopropellant Engines	Jet Vane	Throtteable Monopropellant Engines	Jet Vane

TABLE II-3.

COMPARISON OF PROPULSION SUBSYSTEMS

	SINGLE BIPROPELLANT CHAMBER	RECOMMENDED DESIGN-BIPROPELLANT PLUS MONO-PROPELLANT	SOLID ROCKET PLUS MONOPROPELLANT
AUTOPILOT DESIGN	Somewhat more complex. May require gain change at capsule separation. Response limited.	Simplest	Same as recommended design with higher response required. Less coupling with propellant slosh.
MISSION FLEXIBILITY	Somewhat better. All propellant in same tanks. Redesign for '75, '77 Flyby.	Good - Can accommodate velocity changes relatively easy. Easily modified for '75, '77 Flyby	Fixed orbit injection capability. '75, '77 same as preferred design.
CONFIGURATION	Sensitive to center of mass location.	Insensitive to center of mass motion along Thrust Axis. Relatively tolerant to motion normal to Thrust Axis.	Better control of center of mass. 2.5g loads imposed during orbit injection. Problem for deployed antennas and experiments.
PROPULSION			
WEIGHT	Least	Highest	
COMPLEXITY	Least	Highest	
PROPELLANT ACQUISITION	N_2O_4 presents a problem. Use screens or N_2 settling jets.	Bladders for monopropellant. Use monopropellant to settle bipropellants.	Bladders for monopropellant.
GUIDANCE ACCURACY	Minimum correction of $\cong 0.5$ m/sec.	Minimum correction of < 0.1 m/sec	Same as preferred design for midcourse.

b. At the expense of some weight, rather wide tolerances on center of mass shifts normal to the roll axis can be accomodated.

c. Obtaining the desired frequency response is easily accomplished.

d. The retropropulsion system is dormant until time for orbit injection.

e. Positive expulsion of the monopropellant is readily provided by bladder.

f. The monopropellant chambers provide accurate velocity increments for trajectory correction.

g. The 1975 and 1977 flyby missions can be performed by removing the large retropropulsion subsystem..

During Phase IA, propulsion studies were conducted under General Electric direction by the following companies:

a. Aerojet General - Liquid and Solid Orbit Injection, Monopropellant

b. Rocketdyne - Liquid bipropellant, Monopropellant

c. TRW/STL - Monopropellant

d. Rocket Research - Monopropellant

e. Thiokol-Elkton - Solid

f. Lockheed Propulsion - Solid

Inputs from these companies have been used in making the system selection and in establishing the mechanization approach for the various components.

The General Electric Company is also participating in an IR&D experimental program with Aerojet-General to determine the long term storage aspects of the proposed solid propellants and component parts. Included in the overall program will be an evaluation of subscale motors.

A description of the monopropellant and the bipropellant subsystems is given in the next two sections.

2.1.3.1 MONOPROPELLANT MIDCOURSE PROPULSION SUBSYSTEM

In Figure II-6, a block diagram of the preferred midcourse propulsion monopropellant subsystem is shown. It is a regulated-gas-pressure-fed system using anhydrous hydrazine (N_2H_4) as the monopropellant. Helium gas is used as the pressurant. The four thrust chamber assemblies are designed to operate over a thrust range of 25 to 55 pounds. Insofar as possible, components are grouped together, and connections are welded to eliminate external leakage. Different functional groups are joined by field brazed joints where welding is not practical. Squib valves are used, where feasible, to eliminate solenoid-operated valves and thus assure higher subsystem reliability.

The helium gas is stored in two 17-inch diameter titanium tanks which are joined to a bank of squib operated gas pressurization and shut-off valves. This bank of valves has four parallel legs with a normally open and a normally closed valve in series in each leg. A three-way squib-operated valve feeds high pressure gas through a normally open port to the primary regulator. This regulator provides regulated gas pressure directly to the four propellant storage tanks. A malfunction signal to the three-way squib valve causes a switchover to the second regulator. A pressure switch senses a high pressure failure of the primary regulator and actuates the three-way squib valve. All of the pressurization components except the tanks, are tray-mounted as a single all-welded unit.

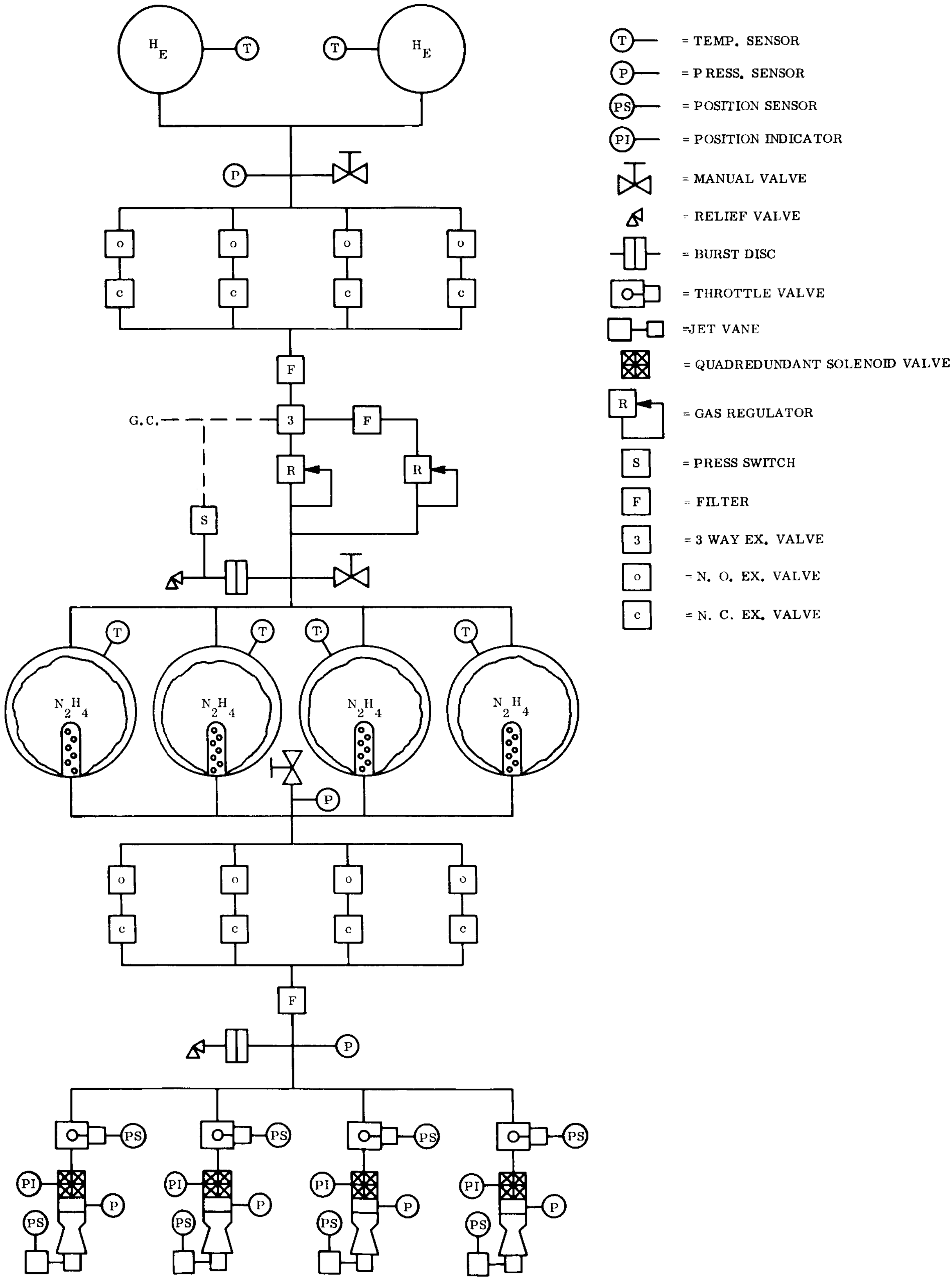


Figure II-6. Midcourse Propulsion System Block Diagram

All four propellant tanks are identical. They are fabricated from titanium alloy and contain butyl rubber bladders which collapse, when pressurized, around a standpipe to assure positive expulsion. All tank discharge lines feed to a common squib valve manifold. The bank of squib valves is similar to those of the pressurization system.

All of these valves and the filter are also tray-mounted and welded together to minimize leakage.

The four thrust chambers are identical units. Each chamber operates over a chamber pressure range of 75-165 psia and a thrust range of 25-55 pounds. Decomposition of the hydrazine is accomplished in a catalyst bed made from Shell 405 catalyst. Decomposed hydrazine at a temperature of approximately 1800°F discharges through the 50 to 1 expansion ratio nozzle to provide the desired thrust. A single torque-motor-operated jet vane in each exhaust jet provides roll control. Thrust chamber operation is initiated and terminated by valves mounted directly on each chamber. Immediately upstream of each quad-redundant valve is a throttling valve capable of modulating the output of each chamber from 25 to 55 pounds.

Selection of the thrust for the midcourse chambers is based upon center of mass and thrust vector uncertainty at the end of retrofire. With the selected thrust level for the retro-propulsion subsystem, and establishing a throttling range of approximately 2:1 (well within the state of the art), the nominal thrust range is 25 to 55 pounds. All chambers nominally operate at the 25 pound level except when making corrections. This will minimize the total hydrazine consumption.

Total weight of the Monopropellant Midcourse Propulsion Subsystem, including fuel, is 677.6 pounds.

2.1.3.2 BIPROPELLANT RETROPROPULSION SUBSYSTEM

The block diagram for the retropropulsion subsystem is shown in Figure II-7. It is a regulated-gas-pressure-fed system, using nitrogen tetroxide (N_2O_4) as the oxidizer, and a

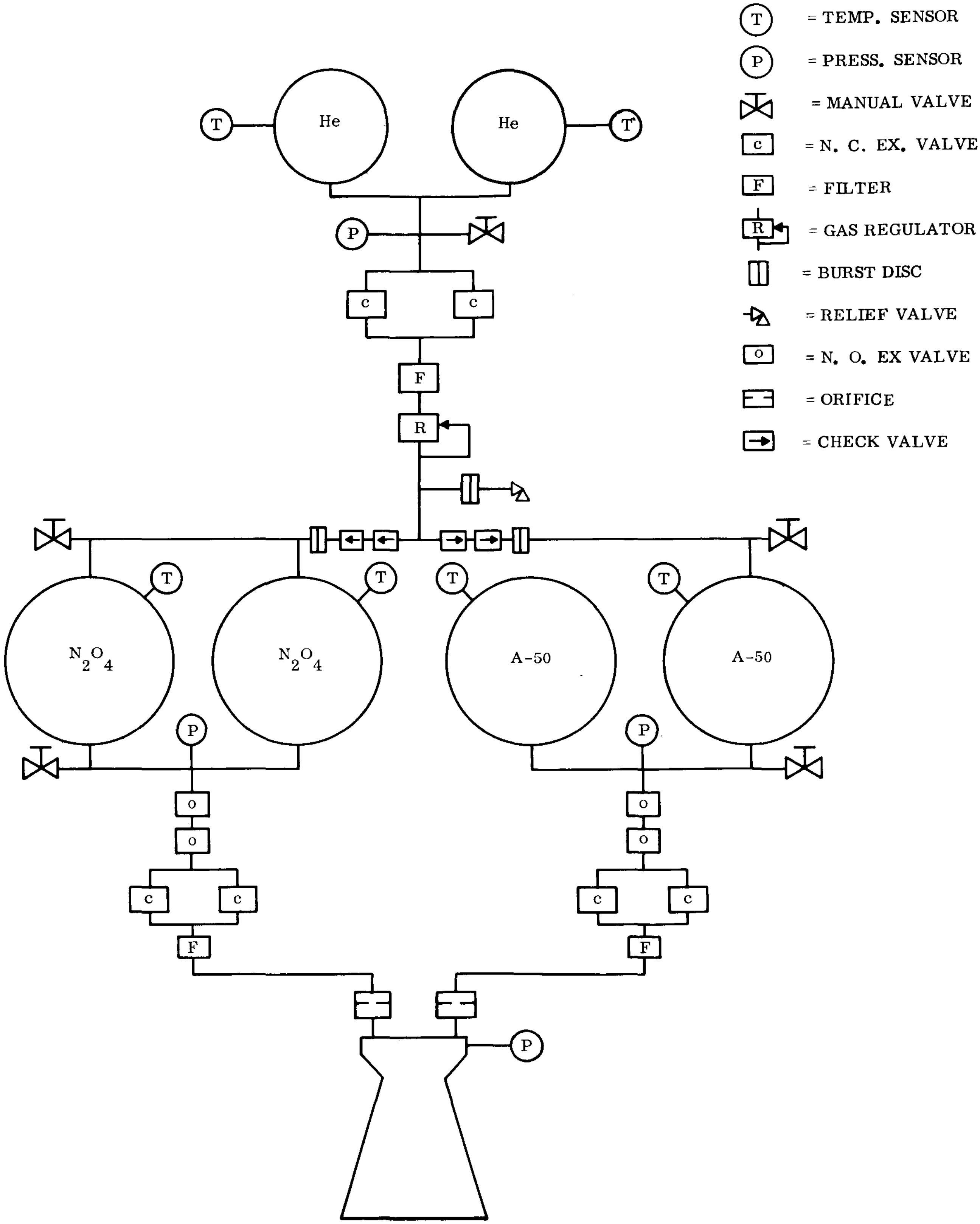


Figure II-7. Retropropulsion Subsystem Block Diagram blend of 50% hydrazine (N_2H_4) and 50% unsymmetrical dimethyl-hydrazine ($[CH_3]_2N_2H_2$) as the fuel. The thrust chamber is a fixed installation using all ablative construction. The thrust is 2200 pounds at a chamber pressure of 100 psia.

The relative simplicity of the design permits the mounting of components into welded functional groups to eliminate external leakage. Functional groups are joined by field brazed joints. Squib valves are used throughout to assure the highest reliability.

Helium gas is stored in two 18.7-inch diameter titanium tanks joined by a common manifold to two normally closed squib valves in parallel. A single stage regulator supplies helium gas to each of the main propellant tanks. A burst disc and relief valve in series are installed downstream of the regulator to protect the system from leakage through the regulator which would overpressurize the propellant tanks. Burst discs are also provided in both the oxidizer and fuel legs of the pressurization system to keep the propellant vapors from mixing or contaminating the regulator, during the nine months storage period. As further protection during and subsequent to the operational period, two check valves in series are installed in each pressurization leg. All of the foregoing valves, filter, regulator and burst discs are tray mounted and welded together to eliminate leakage. A manually operated vent valve is provided in each pressurization leg to aid in filling and emptying of the tanks as required during ground checkout cycles.

Identical spherical tanks, fabricated from titanium alloy, are used for the oxidizer and the fuel. Since propellant settling and acquisition are achieved by firing the Monopropellant Propulsion Subsystem, no positive expulsion devices are required for these tanks. A redundant squib valve network is used for starting and shutting down of the retropropulsion thrust chamber. Two normally open squib valves in series are followed by two normally closed valves in parallel in each propellant leg. Orifices in each side of the injector are used to calibrate all thrust chamber assemblies to identical pressure drops to assure interchangeability. All of these valves, filters and orifices are mounted directly on the thrust chamber and welded together to eliminate leakage.

The selected thrust chamber is an all ablative chamber with an expansion ratio of 60:1. The injector is fabricated from aluminum and uses a conventional doublet impinging injection pattern. Since the required burn time is approximately 316 seconds, the design is well within the present state of the art and thus provides a high reliability potential. Lack of thrust vector control requirements with no need for gimbals, actuators and flexible lines, further enhances the reliability of the unit.

Radiation, regeneration, and ablative cooled thrust chambers were evaluated. Because the overall configuration required a buried installation of the thrust chamber, the radiation cooled approach was eliminated. Ablative cooling has been selected as the preferred method because of its lower weight, reduced complexity, and micrometeoroid resistance.

The thrust level for the retropropulsion engine must fall between the limits imposed by maximum permissible acceleration of the spacecraft, during orbit insertion, and the minimum thrust as fixed by the maximum permissible burn time. One thousand seconds is considered to be state of the art in ablative thrust chamber design. A burn time of 1000 seconds would fix the minimum thrust level at 750 pounds. In addition, for a given total impulse, assuming expansion ratio and chamber pressure remain constant, the thrust chamber weight will increase almost directly with thrust level. Since no gain in performance at higher thrust levels is available to offset the added inert weight, lower thrust levels will decrease the retropropulsion subsystem weight. The thrust level should, therefore, be as low as possible.

A reasonable upper limit on thrust level appears to be 3500 pounds.

There is no existing presently qualified propulsion system in the 750 to 3500 pound thrust range for long duration, deep space operation. The one system in this range which will be qualified within the next year is the LEM ascent engine. It is a 3500 pound thrust bipropellant system. Even though this is a complete propulsion system, the only major component which would have application to Voyager would be the Bell Aerosystems ablative thrust chamber.

Extensive ablative thrust chamber development work has been carried out by several of the major propulsion suppliers at the 2200 pound thrust level on such programs as Saint and Apollo subscale. No engines (or thrust chambers) have been qualified as a result of these programs. However, the work accomplished on thrust chambers have demonstrated durations far in excess of the required 316 seconds (at 2200 pounds).

Ablative chamber development work at other thrust levels, within the desired thrust range, has been of such a lower magnitude, that no other existing hardware appears to warrant consideration.

A weight comparison of thrust chambers at 750, 2200, and 3500 pound thrust levels is given in Table II-4.

Table II-4. Weight Comparison of Thrust Chambers

THRUST (lb)	CHAMBER WEIGHT (lb)	TOTAL ADDITIONAL PROPULSION SYSTEM WEIGHT (lb)
750	100	-98
2200	156	0
3500 (LEM)	212	+111

Although there is a potential weight savings of 98 pounds at the 750 pounds thrust level, there are two significant factors which make the 2200 pound chamber more attractive:

a. The 750 pound thrust level requires approximately a three-fold increase in burning time. Although considered state of the art, the problem of throat erosion must be given close attention.

b. The development effort at the 2200 pound thrust level should be significantly reduced because of earlier design and testing work. Performance and duration have been demonstrated.

While the LEM thrust chamber will be qualified within a year, it adds additional subsystem weight and results in an acceleration level about 60% greater than the 2000 pound level.

Thus, from an overall weight and development status viewpoint, the use of a 2200 pound thrust chamber appears to be a logical choice.

The Bipropellant Retropropulsion Subsystem, including fuel, weighs 2791.4 pounds.

2.1.4 GUIDANCE AND CONTROL SUBSYSTEM

The Guidance and Control Subsystem recommended for Voyager consists almost entirely of techniques and equipment whose reliability has been demonstrated on flight systems now in use or in advanced states of development. Examples include the Spacecraft Attitude Control which is based on both the functions and components of Mariner C and on the sun sensors and electronics of the Orbiting Astronomical Observatory, (OAO); the Gas Jet Subsystem which is based on both the Mariner C and Nimbus systems; the autopilot based, in part, on the Mariner C jet vane control; and the Approach Guidance Sensor which is an extension of hardware developments completed for the Ames Research Center.

Voyager requirements extend beyond those of Mariner because of the larger vehicle size and inertia, the somewhat higher accuracy desired, the effects of structural dynamics during autopilot control, the added pointing requirements for the high gain antenna and the planet scan platform, the approach guidance measurement, and the longer life due to the 6 month orbiting phase. Of these requirements, the added life is considered most severe. The design approach for guidance and control has, therefore, emphasized the use of components whose reliability has been flight or test proven, the use of redundancy at both the part and functional level, and the use of alternate or back-up modes of operation. An example, which is further described in Volume B, is the use of integrated circuits in the Control Electronics Subassembly to allow triple redundancy majority voting without a major weight or size penalty.

The Voyager Guidance and Control Subsystem consists of the following elements:

a. Attitude Control Subsystem
b. Cold Gas Jet Subsystem
c. Autopilot Subsystem
d. Articulation Subsystem
e. Approach Guidance Subsystem

The component locations on the Spacecraft are shown in Figure II-8 and the functional relationships are shown in the block diagram of Figure II-9. A brief description of these subsystems is given in the following paragraphs.

2.1.4.1 ATTITUDE CONTROL SUBSYSTEM

This subsystem provides three axis stabilization of the Spacecraft to the Sun/Canopus reference system so that:

a. A coordinate reference is established for guidance corrections.
b. Antennas can be pointed to Earth and instruments can be pointed to Mars.
c. Electrical power can be generated efficiently by the solar panels.
d. Thermal control can be easily maintained.

This subsystem also reorients the Spacecraft to any desired attitude for midcourse connections, Capsule separation, and orbit insertion.

The recommended design uses the Sun and Canopus as external references for stabilization. Two other reference systems for orbital operation were considered. The first stabilizes two axes of the vehicle to Earth and one axis to Mars. The second stabilizes two axes of the vehicle to Mars and the third axis controlled to lie in the orbital plane. The significant features of these two systems and a comparison of them to the preferred design is shown in Table II-5. A more detailed discussion is included in Volume B. While the reduction in articulation that can be achieved by stabilizing two axes to Earth and one to Mars is attractive, the Sun-Canopus system was selected because it can be implemented with proven techniques and hardware and does not have an electrical power penalty.

The attitude control subsystem acquires and stabilizes the Spacecraft to the Sun and Canopus reference from any initial attitude and attitude rate up to 3 degrees per second. Such an acquisition occurs, for example, after separation from the launch vehicle. The technique for acquiring the references is the same as used on Mariner: first stabilize to the Sun in

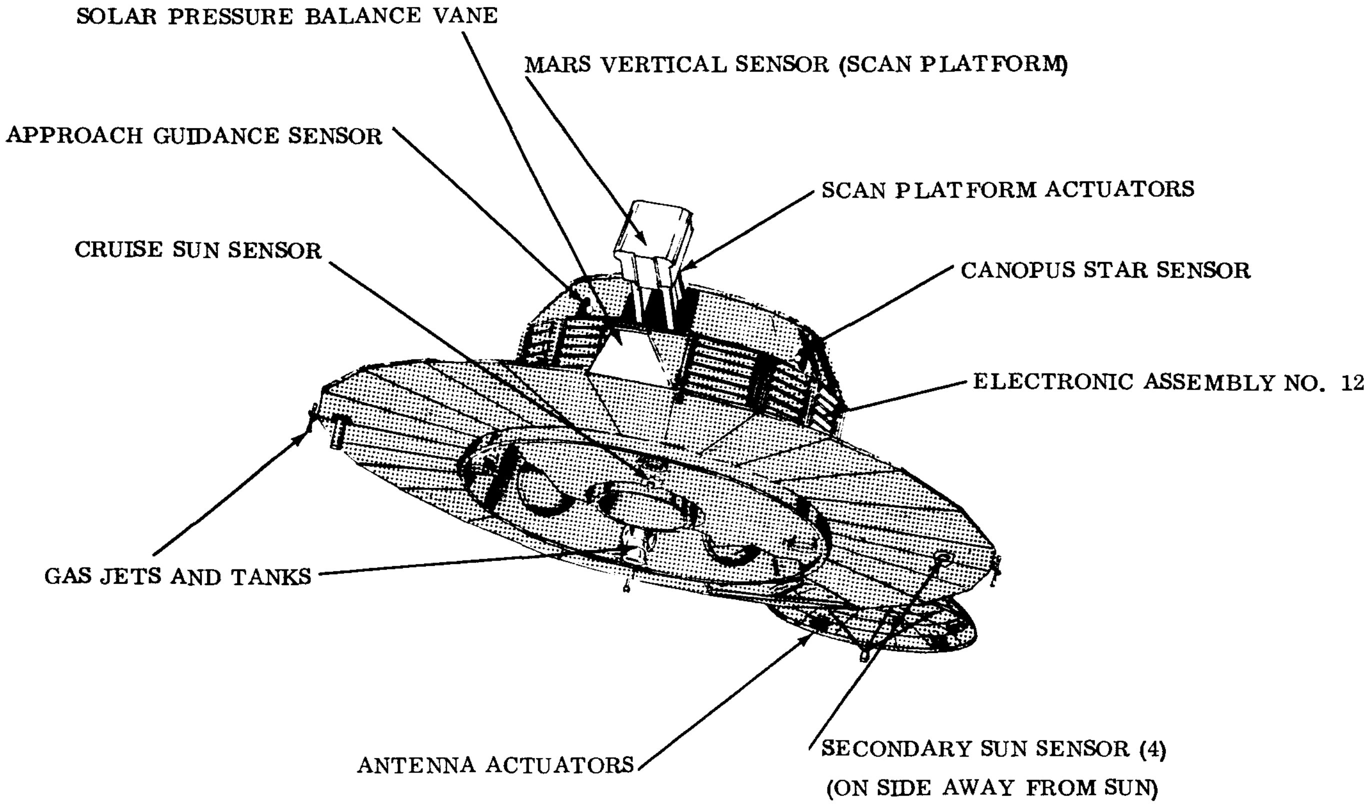


Figure II-8. Location of Guidance and Control Subsystem on the 1971 Flight Spacecraft

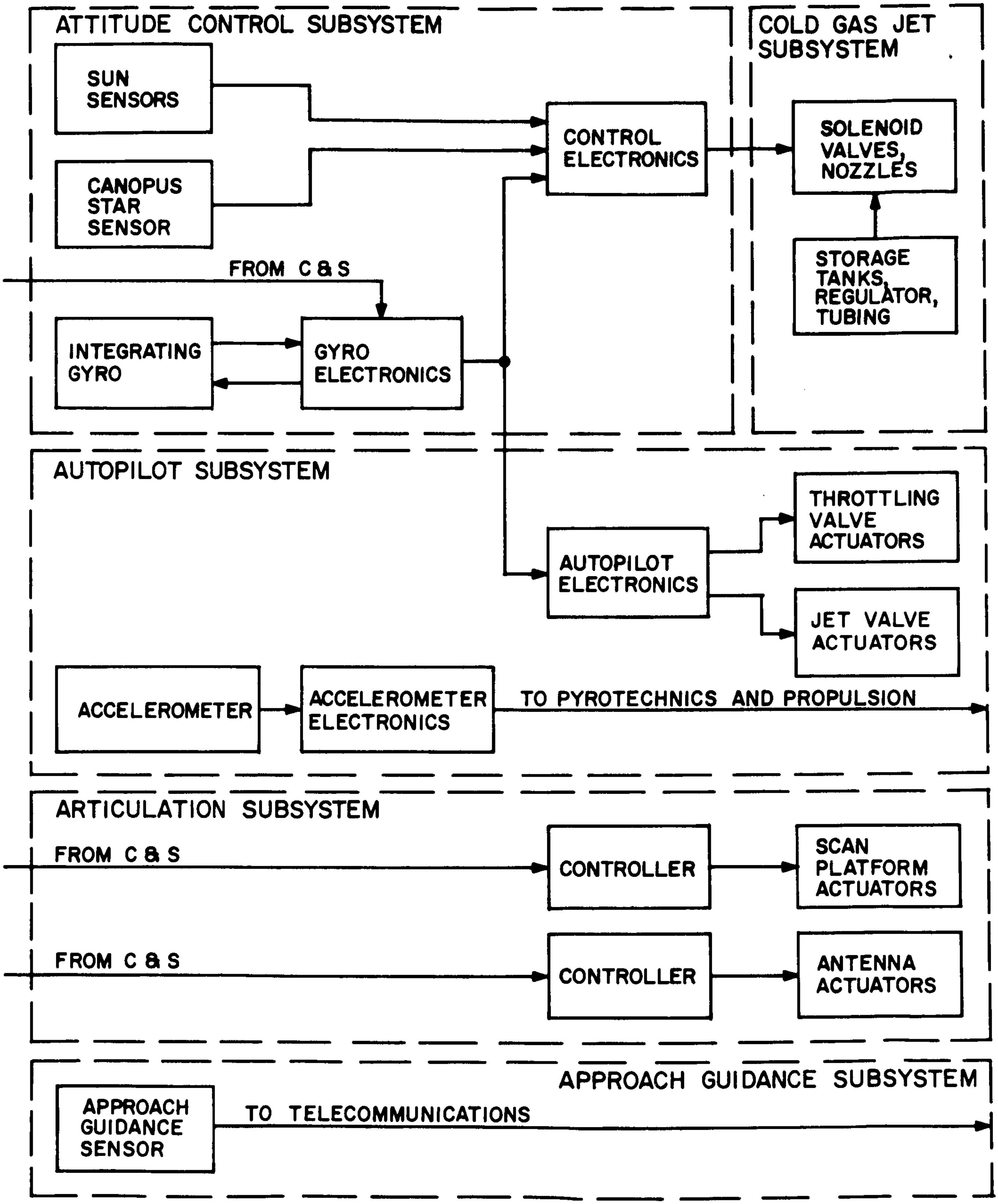


Figure II-9. Guidance and Control Subsystem Block Diagram

TABLE II-5. COMPARISON OF STABILIZATION SYSTEMS

	2 AXIS SUN 1 CANOPUS	2 AXIS EARTH 1 MARS	2 AXIS MARS 1 ORBIT
VEHICLE CONTROL SYSTEM		Large rotations required about Earth line	
ANTENNA POINTING	2 gimbals Programmed motion	Antenna fixed to Space-craft	2 gimbals required Closed loop control Generation of error signals complex
SCIENCE POINTING	3 gimbals 2 motions programmed 1 closed loop	1 gimbal required Vehicle roll provides other axis Instruments rotate about local vertical during orbit	Science fixed to body
POWER		Up to 30° misalignment with sun-power reduction ≅ 15 percent	Dependent on RTG
THERMAL	Best	No major problem - Radiating surfaces can be protected from sunlight	Very difficult - Arbitrary orientation to Sun

two axes, then roll about the Sun pointing axis until Canopus appears in the star sensor field of view. After acquiring these references, the control system, which is now in the cruise mode, maintains the Spacecraft attitude relative to these references with an error of less than 3/4-degree (13 mr).

Spacecraft reorientations are performed by switching control from the Sun/Canopus sensors to body mounted integrating gyroscopes. These are then biased, one at a time, to achieve a constant rate Spacecraft turn, which is maintained for the appropriate length of time to achieve the required position. This procedure is again virtually identical to that successfully used on Mariner. During occulation of the external references, which may occur late in the orbital mission, the attitude control system maintains Spacecraft control by substituting the gyros for the optical sensors in the appropriate loops.

The requirement for gyro operating life is expected to be similar to that of Mariner except for the late orbital phase, when the gyros may be operating continuously. An evaluation of both gas bearing and ball bearing gyros was conducted during the Phase IA study. Although the gas bearing gyro has inherently far longer life capabilities, the ball bearing gyro (Kearfott Alpha series) is recommended for Voyager at this time because its tested life is adequate, it has lower power, and it has a longer development history.

2.1.4.2 COLD GAS JET SUBSYSTEM

High pressure stored gas is used to apply torques to the vehicle at the command of the attitude control electronics. The larger Voyager inertias, number of acquisitions, mission life and disturbance torques result in significant impulse requirements so that consideration has been given to high energy chemical propellants. Measured in terms of percent of total Spacecraft weight, the impulse requirement of the attitude control system is rather small (less than 3 percent of dry spacecraft weight), even if supplied by a cold gas system such as nitrogen. The added complexity of a chemical system is therefore not warranted.

Both nitrogen and freon cold gas systems are satisfactory for Voyager. Freon is recommended because its higher density results in a somewhat lighter subsystem and, more important, smaller gas storage tanks. A comparative analysis is presented in Volume B.

In view of the long mission life, leakage or rapid loss of gas must be prevented. This is accomplished by the use of series solenoid valves in addition to the redundant system approach used on Mariner. In this manner two valves at the same nozzle must leak or stick open to cause a loss of gas in half of the system.

2.1.4.3 AUTOPILOT SUBSYSTEM

The autopilot provides the means of control of velocity vector orientation and magnitude during propulsion operations associated with midcourse correction and Mars orbit insertion. The guidance law used requires that the thrust vector orientation remain fixed in inertial space until terminated when proper magnitude has been achieved. Guidance studies have demonstrated that this simple law is adequate, in that the gravity losses in the Mars orbit insertion are not excessive. Guidance error analysis indicate that the autopilot contributed errors should not exceed about 1 degree in spatial orientation and 1 percent in magnitude, except for velocity increments of 1 meter/sec when 10 percent may be tolerated.

The guidance law is implemented by sensing angular position error and acceleration of the spacecraft and processing the signals to obtain torque error signals and thrust termination signals. Angular position is sensed by three body-mounted rate integrating gyros. Acceleration along the nominal thrusting axis is sensed with a force rebalance accelerometer. Velocity information is obtained by integration of the accelerometer output. A digital reset integrator and counter are used to perform the required integration. The counter is preset with the complement of the desired velocity, and a thrust termination signal is generated when the counter reaches its full count. A timer is used to provide a termination signal as a back-up to the accelerometer. The accelerometer performance is sufficiently better than a timer used alone that fewer midcourse corrections are needed and the Mars orbit accuracy is improved.

The autopilot electronics physically consists of 10 operational amplifiers, compensation networks, a 20-bit shift register and associated logic circuitry located in bay 12. The autopilot also has jet vane actuators located in the expansion cones of each of the midcourse propulsion engines, and throttling valves to control the thrust of each midcourse engine.

2.1.4.4 ARTICULATION SUBSYSTEM

The articulation subsystem points the high gain antenna to Earth and the Planet Scan Platform (PSP) to Mars. In the case of the high gain antenna, two gimbals are provided to achieve Earth pointing to an accuracy of 1 degree. Since the Spacecraft is celestially stabilized and the ephemeris is precisely known, a simple open loop antenna pointing system is employed. By programmed or direct command the gimbal angles are stepped to their required positions to point to Earth. Though it is possible to design a closed loop control system with a sensor on the antenna, the added complexity of such a system does not warrant the small improvement in accuracy possible.

The planet pointing system consists of three gimbals. The first two of these gimbals erect an axis normal to the Spacecraft orbit plane while the third maintains local vertical pointing.

As in the case of the high gain antenna, the orbit plane motion relative to the inertially stabilized Spacecraft will be precisely known, and the first two gimbals which erect the perpendicular can be stepped by command. The last rotation will be performed by sensing the local vertical with a horizon sensor. Commanded positions of this gimbal are possible in the event of the sensor failure or if a fixed orientation is desired.

A two-gimbal system is also possible. In one configuration a first gimbal erects an axis approximately perpendicular to the local vertical and rotation about this axis permits pointing the instruments to the planet. Such a system eliminates one gimbal pivot while maintaining a very high performance in terms of surface observation capability. The performance was evaluated in terms of black and white TV, color TV and IR surface measurements. More specific experiment definition may change the performance parameters of the two-gimbal system. Moreover, even for the experiments evaluated, their value was sensitive to orbit inclination. These reasons have led to the selection of the three-gimbal system which permits a maximum of flexibility.

All gimbals are actuated by means of stepper motors. The output of each stepper motor is geared down such that for each command step the gimbal executes a 1/4-degree change in angle. Between commanded steps the detent torque of the stepper motor holds the gimbal angle against the disturbance torques that are induced by the action of the attitude control. During firing of the engines, the high gain antenna must be deployed; therefore, the stepper motors are put in a stalled mode to hold the gimbal angles against the thrust induced angular acceleration of the engines. In this mode the full stall torque of the motors is available for holding the gimbals. This torque is sufficient to prevent gimbal motion during firing of the engines.

The Mars Vertical Sensor used for controlling the outboard gimbal of the PSP is a dither type horizon tracking device developed by the Advanced Technology Division of American Standard. In this device the projected field of view of two thermistor bolometers are made to dither across the horizon/space interface at diametrically opposed points on the planetary disc. The action of the sensor causes the center of each dither field to lie close to the horizon edge. The bisector of the angle between the center of the two dither fields is the local vertical reference. An alternate sensor that meets the Voyager requirements is the Barnes radiometric sensor.

During the normal cruise mode the Articulation Subsystem orients the high gain antenna to the earth with errors of less than 1.2 degrees (20.9 mr). During maneuvers the orientation error of the antenna will be less than 2.0 degrees (34.9 mr). These errors include the errors of the Attitude Control Subsystem. During the required portion of the mission, the Planet Scan Platform is aligned to the Mars local vertical with an error of less than 1.8 degrees (31.4 mr).

2.1.4.5 APPROACH GUIDANCE SUBSYSTEM

The Approach Guidance Subsystem measures the angles between the Spacecraft line of sight to Mars, Canopus and the Sun during the planet approach phase. These angles are then used as supplementary data to the Earth-based radio measurements in the basic orbit determination system. The Spacecraft equipment makes only the angular measurements, which are telemetered to Earth and used as an input to the orbit determination system. If the angles are measured to an accuracy of 0.1 milliradian, the uncertainty in the impact parameter distance is 34 kilometers as compared to about 120 kilometers possible with the Earth-based radio measurements alone.

These measurements will mainly reduce the positional errors normal to the direction of flight and do little to improve the position along the flight path. It is the miss distance that is of prime interest both from the standpoint of optimizing aerodynamic braking, establishing an accurate touchdown point and Spacecraft orbit. Errors along the flight path affects time of flight which is of secondary interest.

Approach guidance is not a necessity for a successful mission. Earth-based measurements yield satisfactory orbital accuracy. It is planned to include approach guidance in the early missions as a measurement only, to gain experience, and to develop equipment and techniques which may be required on more advanced missions. The measurement equipment consists of a vidicon tube segmented electrically into three areas: one for the Mars image, another for the Sun, and a third for Canopus. Fiber optics is used to focus the three celestial bodies on the appropriate segment of the tube. The electrical deflection circuitry then determines the actual positions on the tube and consequently the angles between these bodies. This tracking technique, though requiring some modifications, has been demonstrated with operating equipment on NASA contract NA52-1087 from Ames Research Center. The required data to be transmitted for each measurement is 160 bits. Measurements are made from approximately 42 hours before encounter until 15 hours before encounter.

The Guidance and Control Subsystem is located in Electronic Assembly 12 of the Equipment Module. The sensors are mounted to the top of the Equipment Module with provisions for adequate fields of view. Total subsystem weight is 216.3 pounds.

2.1.5 ELECTRICAL POWER SUBSYSTEM

The Voyager power supply utilizes photovoltaic/battery energy sources, regulating and converting power to AC for distribution with some power also distributed as unregulated DC. Switching of power for certain users and control of redundant power supply elements is also provided. The basic elements of the power supply are:

a. Photovoltaic Array

b. Secondary Battery (and Charging Circuitry)

c. Voltage Regulator

d. Inverters

e. Switching and Control Equipments

The functional block diagram, showing the interconnection of these elements, is given in Figure II-10. Location on the Spacecraft is shown in Figure II-11.

The solar cells are mounted on a set of 22 identical solar-oriented panels, rigidly attached about the Spacecraft in a flat annular ring. The fixed mounting of the solar panels to the Spacecraft permits full solar orientation of the array without the requirements for any deployment mechanisms, minimizing the area, weight, and complexity of the array. Modularization of the array into identical panels simplifies manufacturing and testing procedures.

The solar cells selected are silicon N/P 1 ohm-cm cells of nominal 11 percent efficiency to air mass zero solar illumination at 85°F. Each panel contains two solar cell strings of parallel and series cells. Each string is diode-isolated from the array bus, and has its own zener shunt regulator to limit the array voltage to 55 volts. The cells are arranged on

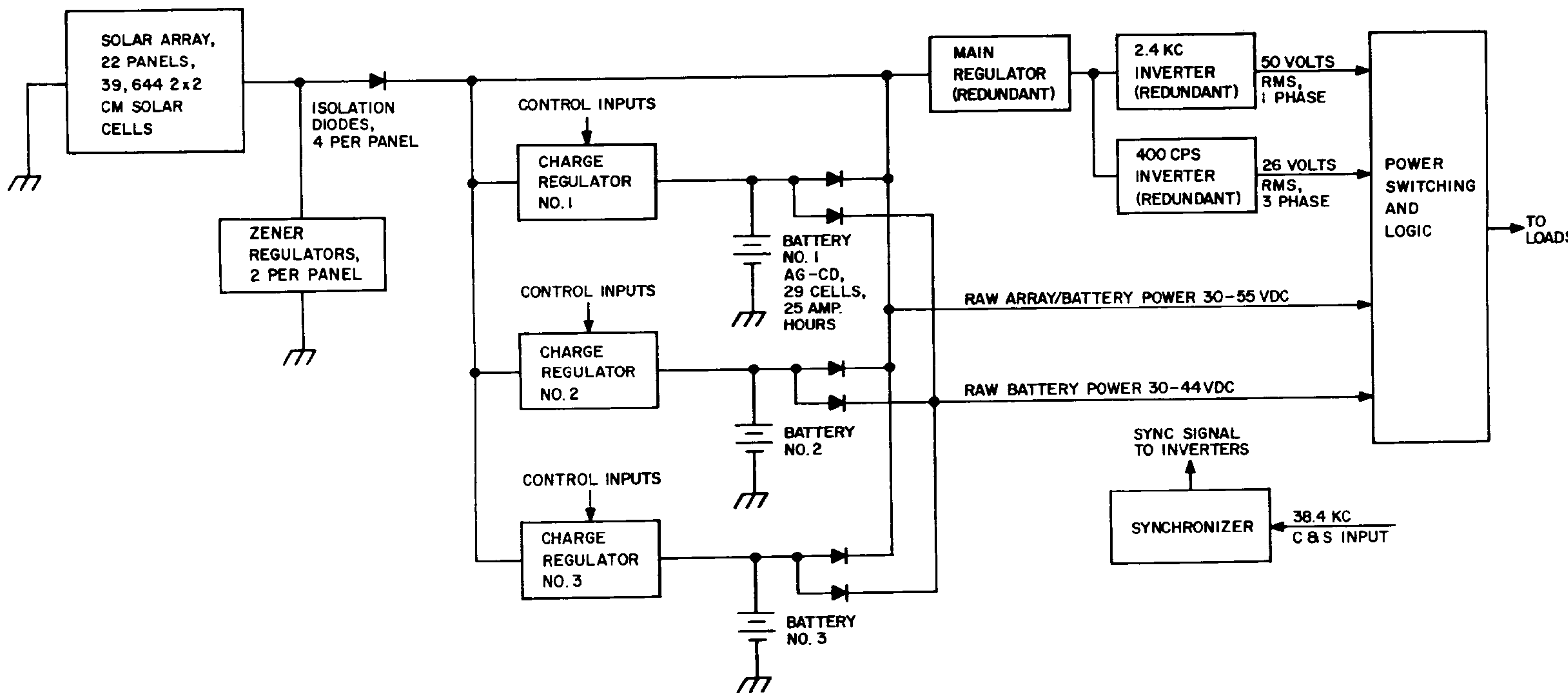


Figure II-10. Electrical Power Subsystem Block Diagram

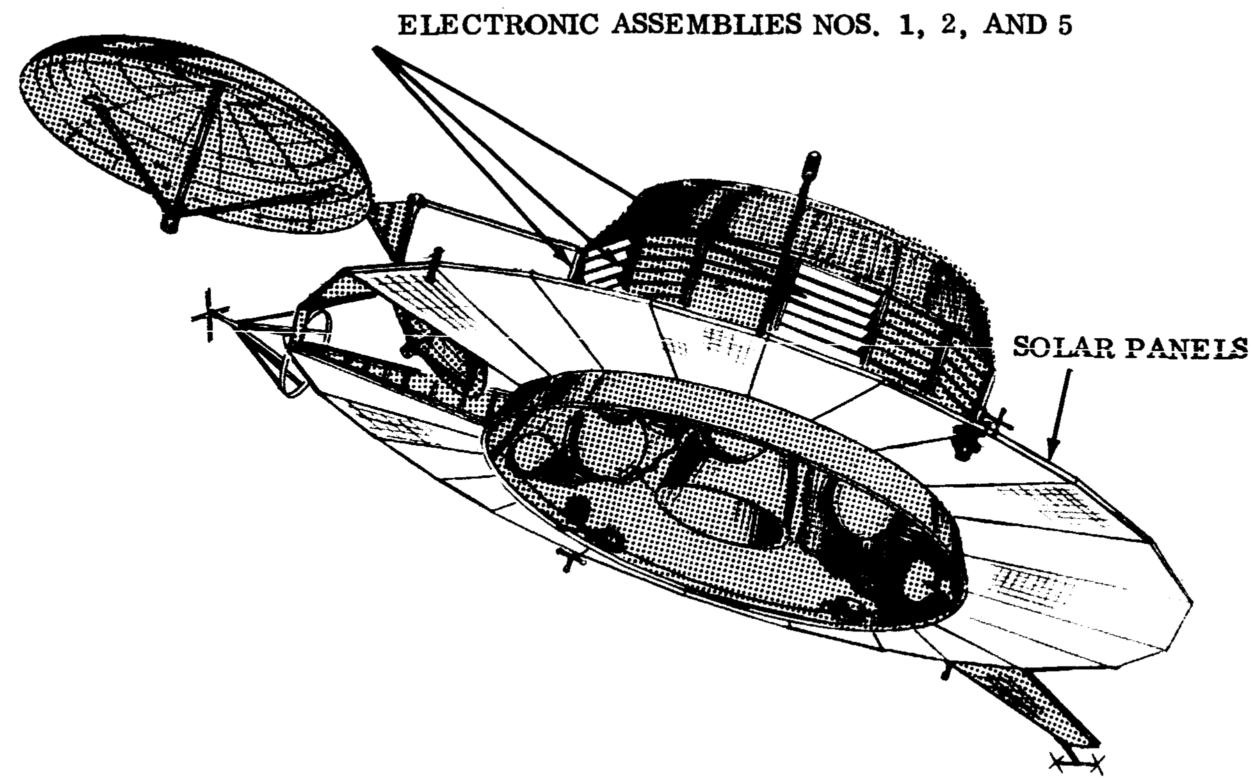


Figure II-11. Location of Electrical Power Subsystem on the 1971 Flight Spacecraft the panels to achieve a high packing factor. Portions of the two strings are located both near to the Spacecraft body and near to the array periphery, to minimize the effects of the large radial panel temperature gradient caused by the blockage of thermal radiation by the Spacecraft and Capsule, thus resulting in virtually the same voltage-current characteristics for the two strings.

The panel construction and cell arrangement are also designed to reduce the effects of induced magnetic fields. Expanded silver mesh is imbedded into the fiberglass front of the panels to form a return current path under each module of cells, thus minimizing the current loop area. Other cabling in the solar array uses twisted wire pairs.

The power output capability of the solar array is shown as a function of time in Figure II-12. This curve is based on a launch date for the 1971 opportunity which results in the greatest Sun-Spacecraft distance at the end of the mission. Superimposed on the array power output profile is the Spacecraft power demand profile. The solid lines in the profile indicate the power levels most likely expected. The dashed lines indicate the increased power requirements during battery charging, as for several hours after a maneuver, or during those Mars orbits experiencing solar occultation.

Also shown in Figure II-12 is the array power output during load sharing. Load sharing occurs when both the array and the battery supply power to the load, and the array voltage is drawn down to the battery discharge voltage. Under this condition, the array will produce less than design power capability. The power supply could possibly "lock" in this condition, permitting battery discharge when the array itself is fully capable of supplying the required power. Since this potential problem is largely confined to the last stages of the mission (see Figure II-12), the simple expedient of load switching is recommended as a solution. The power supply will be brought out of this condition by a momentary interruption of part of the load to reduce the load total to less than the "load-sharing" array power output - - that is, to "turn off" the battery. A knowledge of this load-sharing array power will indicate the mission times and power levels at which the load-sharing might become a problem, and the load switching can be programmed to avoid the condition.

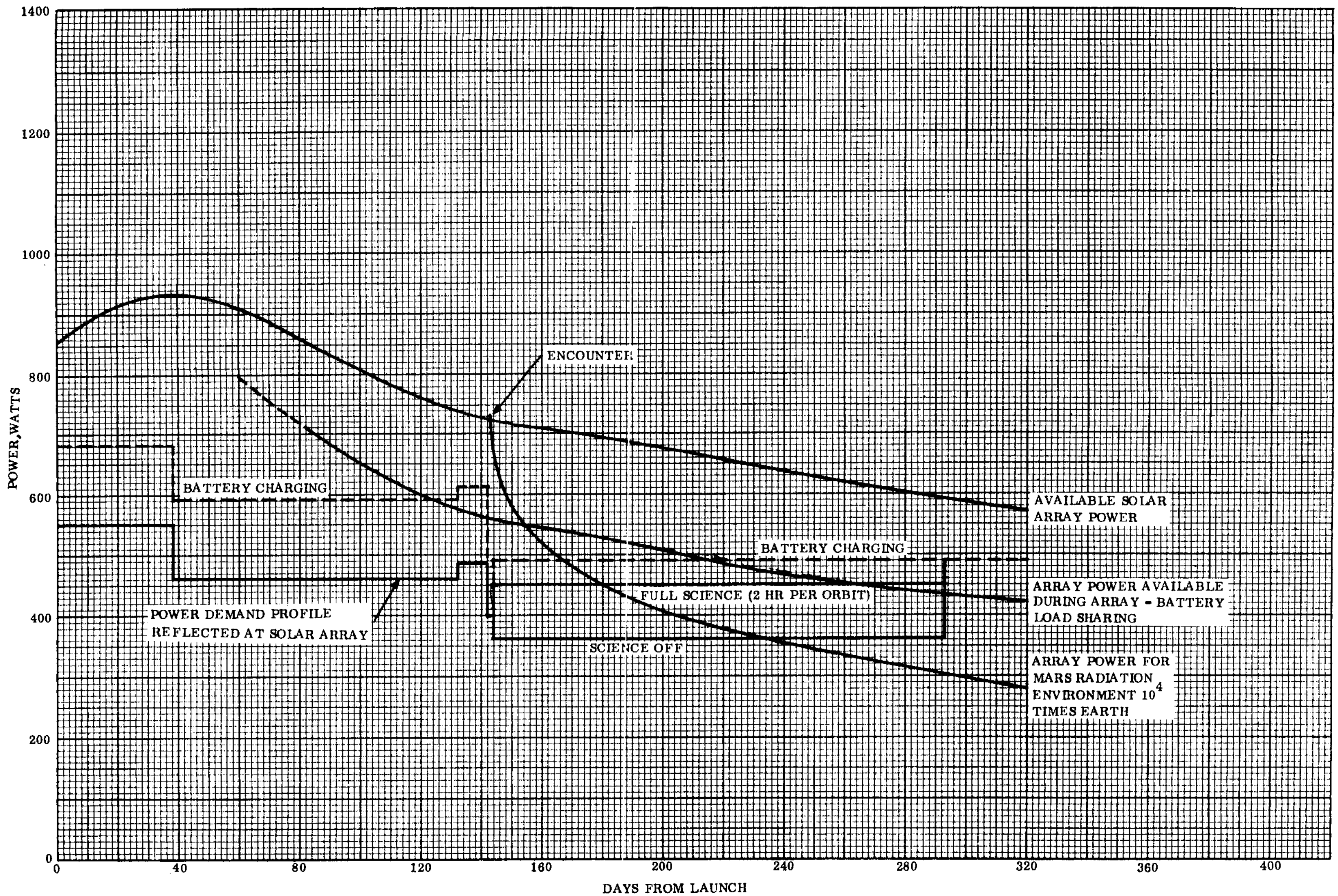


Figure II-12. Power Output Characteristics for Solar Array

Another curve in Figure II-12 shows the decay of solar array power in Mars orbit for the case where the Mars "Van Allen" radiation is 10^4 times that of the Earth. The curve indicates the effects of both the radiation and the increasing Sun-Spacecraft distance as functions of time. The power available decreases very rapidly the first few days in Mars orbit, but sufficient power will be available to operate the Spacecraft and complete Science Payload for 36 days without requiring the use of the battery, and with no degradation of Spacecraft performance. The full Science Payload may be operated for its planned 2 hours per orbit for up to an additional 30 to 40 days, depending on the orbit period, by cycling the batteries. The Science Payload may be operated for at least part of each orbit for up to 87 days from orbit injection, providing there is no eclipse in the orbit. Thus, the effects of an unexpectedly high radiation environment will be to reduce the performance in the latter portion of the Mars orbit phase, when such reduction will have the least effect on overall mission success.

The solar array, as sized with 22 panels, has approximately 16 percent excess capacity to allow for growth in electrical loads. The overall Spacecraft diameter constraint prevents panels larger than the present design from being used, but should array power requirements exceed the capacity of the present array design, up to 10 additional fold-out panels may be used, each the same size as the fixed panels. Thus the array power requirements may be increased by 45 percent before serious design problems are encountered. Adjustments in array capacity as small as 26 watts may be made by this method. Placement of the additional panels is, of course, critical with respect to solar pressure balance of the Spacecraft.

Both solar photovoltaic cells and Radioisotope Thermoelectric Generators (RTG) were considered as prime power sources for the Voyager Spacecraft. The RTG has several potential advantages, of which the most significant are:

a. Improved reliability due to lack of dependence on batteries to carry out a successful mission. Small batteries may still be required if peak loads occur in the Science Payload.

b. The Spacecraft can be designed to have a higher $^m/C_DA$ allowing lower orbit altitudes at Mars without risk of orbit decay in 50 years.

c. Some configuration advantages result from not requiring the large solar array areas.

The RTG source could not be recommended for Voyager primarily because of the uncertainty of fuel availability in sufficient quantities. The only attractive fuel type is Pu238, and while estimates have been made that sufficient quantities can be manufactured in time for Voyager, no positive evidence is available to assure this. In addition, RTG's have disadvantages as follows:

a. Using Pu238 as fuel, the cost probably is much higher than for solar cells.

b. Potential interference with the Spacecraft Science Payload.

c. More restrictive handling procedures for the Flight Spacecraft, although this may be necessary if the Capsule contains an RTG.

Further discussion of the RTG power source is contained in Volume B. Based on the above advantages and disadvantages, the solar photovoltaic cell power source is recommended for Voyager.

Rechargeable silver-cadmium batteries provide power at times when the solar array is not solar-oriented and illuminated. The batteries supply power to the main array/battery bus through isolation diodes, and also provide power through other isolation diodes to a separate raw battery bus for pulse loads such as solenoids, antenna gimbal drives, thrust vector control engine controls, gyro heaters, and to the main voltage regulator fault detector, whose operation is critical.

There are three identical 25 ampere-hour, 29-cell batteries, with a total capacity at end-of-life of 1600 watt-hours. Their capacity when new is 2280 watt hours. The relation between battery energy capacity, maneuver and eclipse loads, and the battery recharge time is indicated in Figure II-13. At least 9 hours are required between consecutive midcourse maneuvers to allow the batteries to recharge.

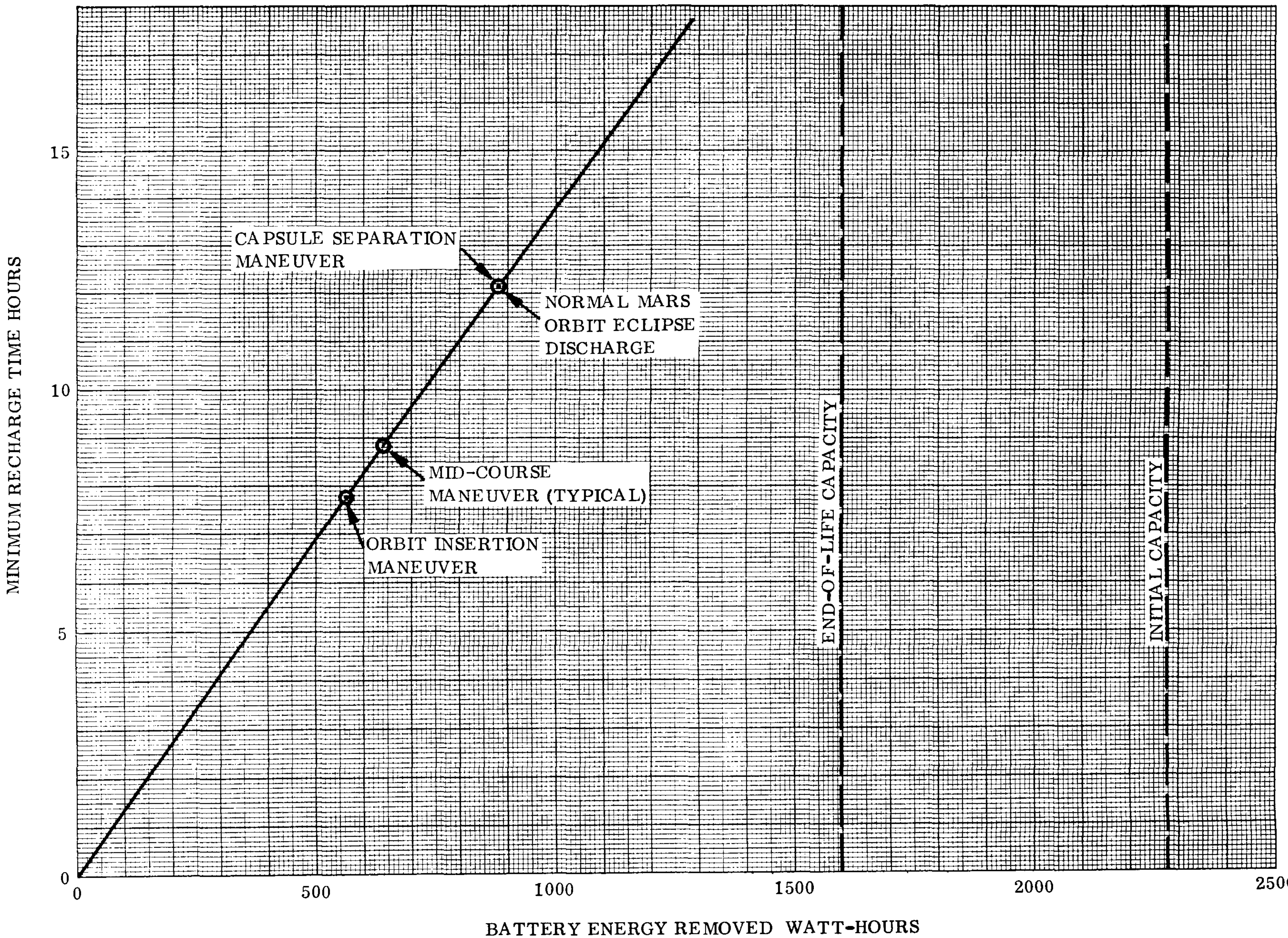


Figure II-13. Battery Energy Utilization

Three rechargeable battery types were considered for Voyager; nickel-cadimum, silver-cadimum and silver zinc systems. The nickel-cadimum battery, though well developed and reliable, was ruled out because of its magnetic properties and, hence, potential interference with the Science Payload. The silver-zinc battery has the highest energy density and would result in the least weight. However, it has limited life cycle capability because of the relatively high chemical activity of zinc in degrading cell separator materials. The silver-cadimum battery is recommended because of its non-magnetic properties, reasonable weight, and its ability to meet the Voyager requirements.

Each battery is charged by a separate charge regulator, a series dissipative type, which varies its impedance to limit the battery charging voltage and current. When all three chargers are in operation, each limits the current into its battery to a nominal 1 ampere, corresponding to the greatest recharge requirements in occulted Mars orbit. If one of the chargers is turned off, the current limits on the other two are adjusted upward to 1.5 ampere; similarly, if only one charger remains in operation, its current limit is set at 3 amperes. Thus, the ability to maintain the original power profile will not be impaired by any reduction in the overall battery charging rate.

Because the mechanisms of battery degradation are not precisely known, provisions are included for adjusting the battery charge voltage limit to allow for short circuit failures of up to three single cells in each battery without exceeding a safe charging voltage of 1.58 volts per cell.

The shorting of cells is principally attributed to the migration of silver particles into the cell separator material; however, all factors contributing to this migration are not fully understood, and conservative design approaches, such as incorporating the additional battery charge control complexity, are recommended.

The information required for modifying the charge voltage limit-setting is derived from telemetered battery voltage, current and temperature information.

Possible abnormal operating conditions may be deduced from the following:

a. Uneven sharing of the battery load during discharge. The battery with a shorted cell will assume a smaller proportion of the load during the initial part of discharge.

b. Excessive battery temperature continued over an extended period.

c. Discrepancy between ampere-hours replaced during charge and those removed during the previous discharge. The accounting of ampere-hours will be required as a function of flight operational support.

A decrease in the voltage limit, by command, will relieve the battery overcharge problem. It will not alter the unequal battery sharing in discharge, but this is not a serious problem in itself as long as end-of-discharge voltages are above specified limits.

Of the three batteries, one is partially redundant, since the Spacecraft can survive in the case of the longest Mars orbit eclipse time with only two of the batteries operative, although at reduced capability.

Some of the unregulated DC power from the array/battery bus is distributed directly to the Radio Subsystem and to the Capsule without further processing. The remainder of the power is regulated and converted to AC. AC distribution of power was selected because: (a) it improves magnetic cleanliness through the elimination of DC current loops; (b) it permits transformer load isolation; (c) it results in overall equipment simplification since a single electronic chopper is used (with DC distribution separate choppers are required for each DC level conversion); and (d) it permits the potential use of equipment already developed for the Mariner 4 Spacecraft.

The main voltage regulator reduces the unregulated voltage appearing at the array battery bus to 28 volts DC $\pm$ 1%. The regulator is a time-ratio-controlled switching buck regulator operating at 90 percent efficiency at full load. Active regulation and input and output filters provide effective isolation of the regulated output from disturbances on the array/batterybus.

A completely redundant voltage regulator is provided, whose use is controlled by a fault detector and switching circuit which is powered directly from the battery. The redundant regulator may also be switched into use by command.

All the power from the voltage regulator is converted to AC in two inverters. Both inverters receive their drive signals from a power synchronizer. The 2.4 kc inverter transforms the regulated DC into a regulated (±2%) square wave AC of 50-volts rms amplitude. This AC is distributed to the user subsystems, each subsystem providing its own transformer/rectifiers for converting the AC power to the required DC voltages. The 2.4 kc inverter is essentially the same as the Mariner C inverter, but is scaled to a higher power output level.

The 400-cps three-phase inverter transforms the regulated DC into a regulated (±2%) three-phase AC at 26 volts rms in a stepped waveform. The inverter consists of three separate switching amplifiers operating 120 degrees out-of-phase, interconnected to produce a stepped waveform. This inverter is identical to the Mariner C design.

The power from the three-phase inverter is used for the tape recorder, the gyros, and perhaps for the scanning instruments in the Science Payload.

Both of the inverters are provided in redundant pairs, each with an on-board fault detector and switching circuit to place the redundant inverter in use automatically. The redundant inverter in each case may also be put into use by command.

The Power Subsystem electronics are located in Electronic Assemblies 1, 2, and 5 of the Equipment Module. Total weight of the subsystem is 414.0 pounds.

2.1.6 TEMPERATURE CONTROL

The purpose of the Temperature Control Subsystem is to maintain all equipment within temperature ranges that will allow reliable performance over the life of the mission. The approach taken has been successfully demonstrated on Mariner by JPL and on Nimbus by General Electric.

All electronic subassemblies, tanks, plumbing and structure are thermally integrated to the maximum extent possible within a superinsulation cocoon. Internal thermal coupling is achieved by the use of: (a) high emissivity surfaces, (b) an open type internal structure, and (c) silicone grease between heat dissipating components and their mounting plates. Advantage is taken of the Spacecraft's Sun-Canopus orientation during the transit and Mars orbit phases by allowing some solar energy to penetrate the normally illuminated surface of the bus superinsulation cocoon to aid in keeping the several enclosed tanks warm. All support structure for appendages such as solar arrays, antennas, the scan platform, retro and midcourse engines, and booms are conductively insulated from the bus structure to reduce heat loss from the bus.

Excess heat is released from within the superinsulated enclosure by means of eleven sets of shutters on the external surface of the equipment bays which control the emittance of the heat rejection surfaces. Each shutter assembly is actuated by a two-phase fluid sensor/bellows/drive rod/return spring arrangement as illustrated in Figure II-14. In selecting the shutter actuation system, consideration was given to three types of actuators. Both bimetallic and thixotropic actuators were considered along with the two-phase fluid that is recommended. A comparison of these approaches is shown in Table II-6.

The two phase liquid-bellows system is recommended for the following reasons:

a. The system is relatively invulnerable to normal and extraneous vibration experienced during the mission lifetime.

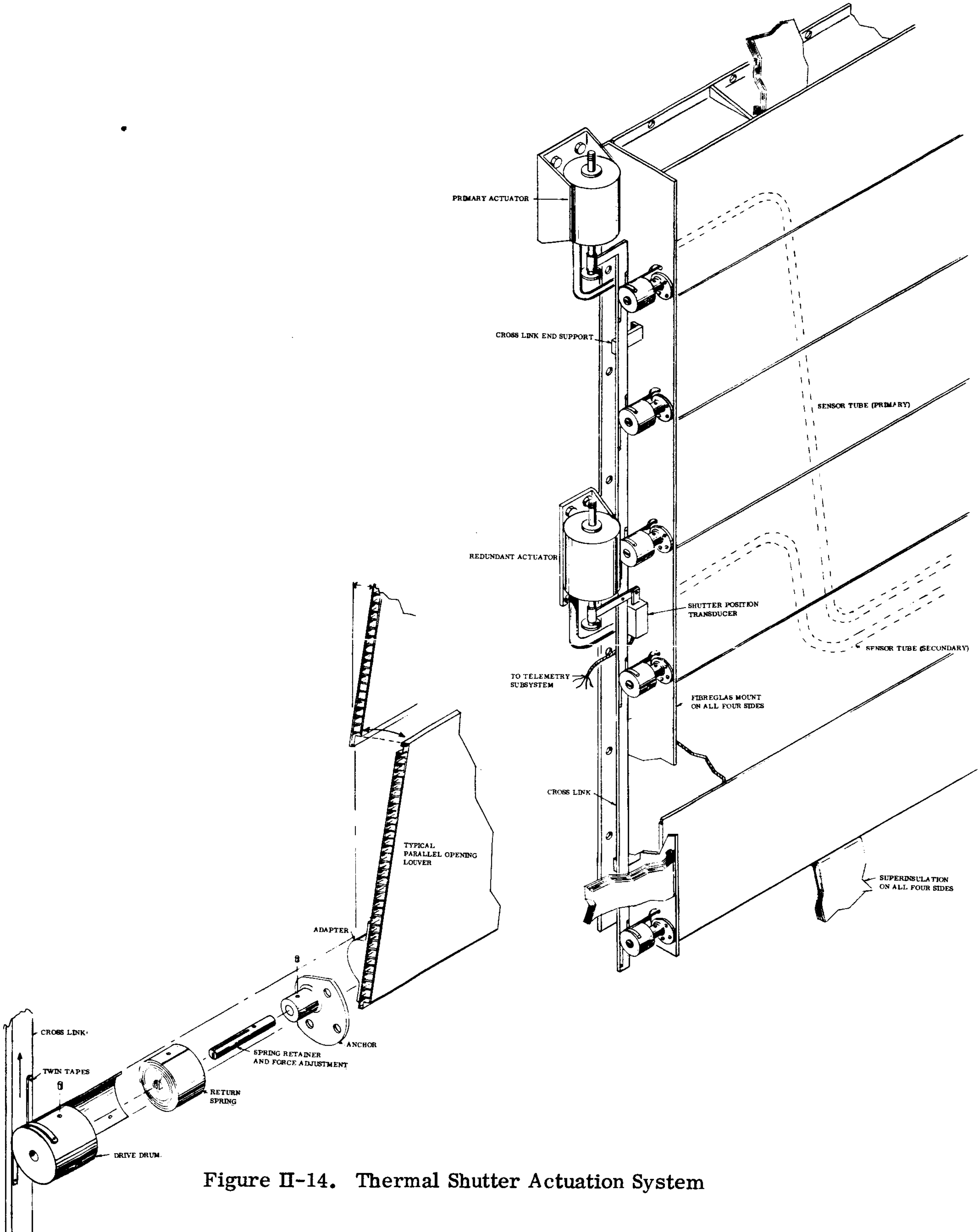


Figure II-14. Thermal Shutter Actuation System

Table II-6. Comparison of Shutter Actuators

	TWO PHASE FLUID	THIXOTROPIC	BIMETALLIC
SERVICE: (AT A GIVEN TEMP. CHANGE)	Moderate force range at large deflections	Unlimited force range at small deflections	Small force at small deflections
STRUCTURAL EFFECTS (LAUNCH LOADS)	Not serious	Not serious	Critical--subject to hysteresis
SHUTTER OPERATION	Gang operation	Gang operation	Individual blade operation
GROUND OPERATION	Pressure compensated bellows to limit ambient pressure effects	As is	As is
REDUNDANCY TECHNIQUE	Twin actuators	Twin actuators	Nor required (adjacent blades assume control)
FAIL SAFE ACCOMMODATE	Not difficult	Not difficult	Difficult - Limited return spring force capability if at all
SPACE IRRADIATION EFFECTS	May be a problem	Probably none	None
DEVELOPMENT	Moderate - Test experience validates fluid selection, filling procedures, sensor tube location and bellows design	Moderate - Bellows and sensor tube design for motion and good thermal contact	Moderate - Trial/error design, structural design critical

Table II-6. (Cont'd)

	TWO PHASE LIQUID	THIXOTROPIC	BIMETALLIC
POTENTIAL PROBLEM AREAS	1. Space irradiation effects 2. Filling procedure critical 3. Superheat control 4. Micrometeoroid puncture/ leakage	1. Pressure relief required for high temperature storage 2. Heat flow to sensor low response slow 3. Motions small--Mechanism design complex 4. Filling procedures critical--No gas entrapment 5. Micrometeoroid puncture/ leakage	1. Structural degradation from vibration, temperature, and cycling 2. Heat flow to sensor may not be predictable or repeatable 3. Fail-safe difficult to accommodate
ADVANTAGES	1. Moderate forces available 2. GE past design/ test data 3. Redundancy incorporation not difficult 4. Fast response to temp. changes 5. Fail safe in closed position easy to accomplish 6. Actuator driven by hottest temp. on mounting plate	1. Little change of hang up due to high forces 2. Probably no irradiation problem	1. Simple-Compact 2. Built-in redundancy/ unless many fail 3. Unaffected by irradiation

b. Temperature sensitivity is believed to be superior to that of competing methods.

c. Redundant actuators and individual shutter return springs lead to high reliability even under certain failure contingency modes.

The bellows actuator employed is a pressure compensated design such that ambient pressure does not influence piston travel. The control fluid is ethyl chloride with its normal operating pressure ranging from 10.8 psia at 40°F to 20 psia at 70°F. The fluid is contained in the beryllium copper bellows that is silver soldered to the housing and to the piston stop. The space between the housing and the bellows is evacuated to approximately 10 microns Hg. No O-rings or gaskets are employed to contain the fluid.

No conventional bearings are employed in the shutter design. Flexure pivots are used at the undriven end while torsion springs integral with the drive drums serve as support at the driven end. The flexure pivot has a restraining ring overhang as a lateral stop to prevent load damage. The torsion spring serves as a restraining force (acting against the actuator force) for positioning the shutter in the closed position. All materials employed are non-magnetic.

The temperature control subsystem will maintain the average operating temperature of the electronic equipment in the bus between 40°F and 70°F. The analysis indicates that for some equipment bays which have essentially constant dissipation, active shutter control is not required to maintain adequate temperatures. The recommended design has shutters on all bays (except the bay adjacent to the Planet Scan Platform), to provide maximum flexibility for accepting equipment changes and to accommodate abnormal conditions in flight.

Total weight of the Temperature Control Subsystem is 120.2 pounds.

2.1.7 CONTROLLER AND SEQUENCER

The primary function of the Controller and Sequencer (C&S) is to permit mission flexibility by providing for the storage and execution of Spacecraft commands. While the Telecommunication Subsystem is capable of providing commands reliably in real time, the exigencies of maneuvers and of orbital operations require the on-board capability for timing and sequencing of stored commands. This function is centralized in the C&S in order to minimize the amount of hardware entailed.

The recommended design for the Controller and Sequencer is a stored program command system in which commands can be stored prior to launch or transmitted to the C&S via the Command Subsystem. In selecting this approach, consideration was given to an on-board computer, ground command only, and a fixed sequence command system. The stored program command system was selected on the basis that it most nearly optimizes the trade-off between required operational support from Earth and minimum complexity in the Spacecraft.

The Controller and Sequencer provides an accurate clock for all the Spacecraft Subsystems, facilitating the synchronization of operations and obviating the need for separate clocks. This is accomplished by the C&S transmitting a variety of cyclic signals, such as the 38.4 kc which governs the 2.4 kc power supply and the one-pulse-per-second square wave which the Guidance and Control Subsystem uses as a clock.

Figure II-15 is a simplified block diagram of the C&S. It has two timers, one for repetitive sequences (Sequence Timer), and the other for non-repetitive commands (Master Timer). By such specialization, the amount of redundant information which is transmitted to and stored in the Spacecraft is reduced. The Master Timer measures the time after launch up to 776 days, in one second increments. Functions which are to be executed at times which can be calculated in advance are associated with the Master Timer. The command to execute is stored in the C&S memory in terms of a digital time label which is compared once a second with the contents of the Master Timer. When the two 26-bit words are

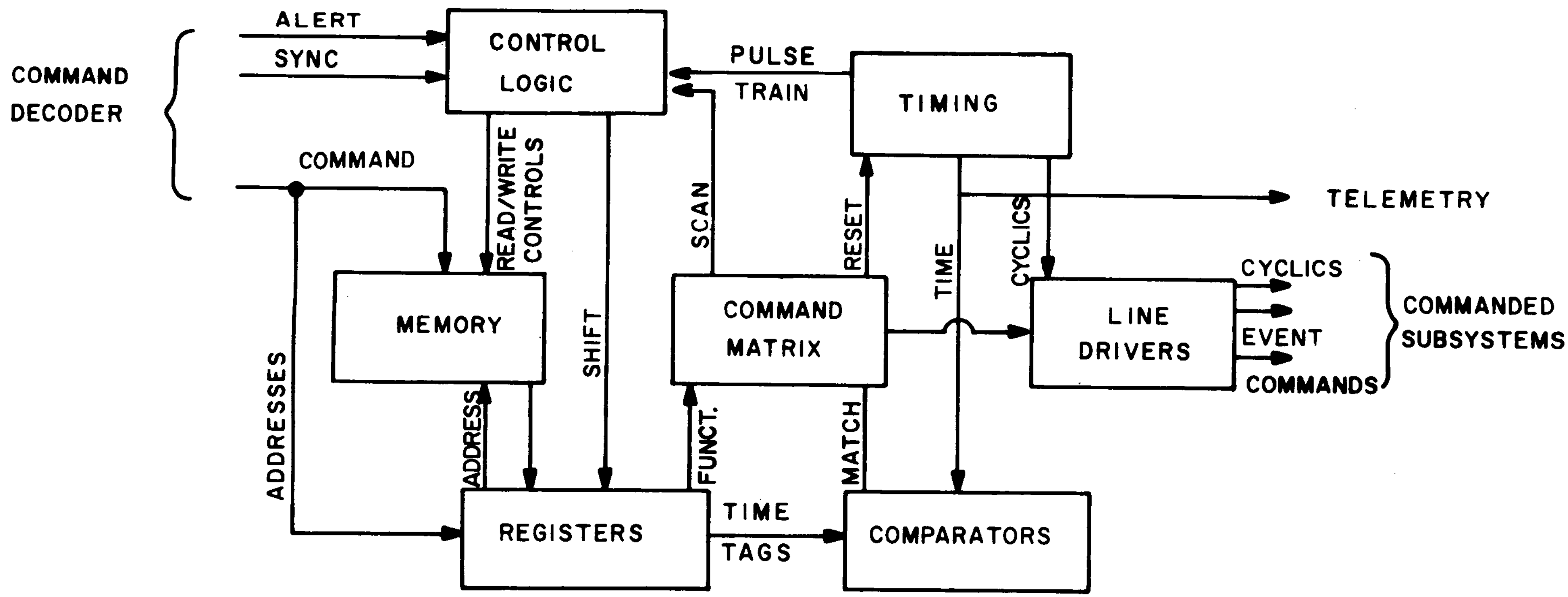


Figure II-15. Controller and Sequencer Block Diagram identical, a command pulse is transmitted over the command line associated with the memory location in which the command word had been stored.

The Sequence Timer runs up to 72 hours, in one second increments, in two modes: for maneuvers it runs through its sequence once and stops; for orbital operations it resets on its last command and repeats the sequence. Commands normally associated with it occur in groups which are related to a base start time, such as the initiation of a maneuver. Since the base start time, which is a Master Timer command, is not necessarily known in advance, it may have to be determined after launch and transmitted via the Telecommunications Subsystem. Sequence Timer command words are of the same length as Master Timer commands; however, the time labels consist of 18 bits and the remaining 8 bits represent the command address. Thus, memory locations associated with the Sequence Timer are not dedicated to particular commands and may be reloaded repeatedly with commands for different functions.

Despite their differences, the Master Timer and the Sequence Timer provide two essentially redundant command channels. Each is used to issue a command whose time label matches its contents. However, the use of one channel to back up the other implies a penalty of additional information through the Telecommunication Subsystem. For instance, if the Master Timer is inoperable, all commands can be issued with "sequence" labels, provided commands are inserted to reset and clear the Sequence Timer.

Because the C&S is central to Spacecraft operations, the philosophy of providing redundancy applies to every element of this subsystem, save only the last stage of the command matrix and the line drivers. (In these two exceptions, a failure affects one command at most.)

The C&S is made up of functional elements, each of which is provided in triplicate; majority voting also in triplicate, ensures that no single failure can be catastrophic. Furthermore, the C&S has considerable overlap of command capability with the Command Decoder, so that each provides a back-up for the other.

The C&S design is quite simple, since it is based upon the conservative use of proven, reliable digital circuits and techniques. It is adaptable for a variety of mission profiles since the control lines are selected by means of binary-coded function addresses, which are decoded by the command matrix. The present package is scaled for the 1969 and 1971 Voyager missions, but it can be used for other missions which require no more capacity than 255 stored commands.

Expansion of C&S capability requires a minimum of redesign, depending on the type of growth. For instance, if it is necessary to double the command storage capacity (without increasing the number of command functions), the memory address portion of the command word is increased by one bit, the address register is lengthened by one bit, and a module is added to each of the redundant memories. Similarly, increasing the number of command functions (without increasing storage capacity) entails incrementing the command matrix and the number of line drivers.

The mission objective is to obtain and relay to Earth information about the Martian environment. The ability to store a large number of commands for delayed execution furthers this objective by permitting operations to proceed despite Earth occultations, ground station down-time and schedules, and other interferences with real-time communications. It makes possible verification of receipt and accuracy of transmission before execution, and it permits accurate timing of closely spaced sequences. Thus the Spacecraft can operate automatically for considerable periods when real-time commands are not adequate.

The Controller and Sequencer is located in Electronic Assembly 12 (along with the Guidance and Control electronics). It weighs 18.0 pounds.

2.1.8 PYROTECHNIC SUBSYSTEM

The Pyrotechnic Subsystem accomplishes nonrepetitive mechanical actions by explosive means. It consists of the following items:

a. Parallel redundant Separation Switches to safe the subsystem during pad and prelaunch checkout and powered flight.

b. A Pyrotechnic Controller which transforms low level electrical command signals into high energy pulses.

c. Electroexplosive Devices which generate a controlled explosive force when electrically initiated.

d. Pin Pullers, Explosive Nuts, and Electrical Disconnect, and a Separation Joint as the mechanical devices which are activated by the generated force when electroexplosive devices are initiated.

The Pyrotechnic Controller receives 2400-cycle square wave AC power through a parallel redundant connection of two electromechanical safe and arm devices both of which are positively locked-out for pad safety, but which are activated upon separation of the spacecraft from the launch vehicle. Separation Switch No. 1 provides immediate electrical continuity from the Electrical Power Subsystem to the Pyrotechnic Controller and enables the Controller and Sequencer and the Guidance and Control Cold Gas Subsystem. These functions are provided by the closure of normally open electrical contacts which carry signal lines from the Controller and Sequencer and the Guidance and Control Subsystem.

Separation Switch No. 2 also enables the Pyrotechnic Controller and the Guidance and Control Cold Gas Subsystem. Separation switch closure initiates a three-minute electronic timer in the Pyrotechnic Controller. At time-out, a power pulse is given to fire the pyrotechnics to deploy the Antennas, enabling the RF link.

Arming of the Pyrotechnic Controller energizes redundant transformer rectifier power supplies which transform down the AC voltage, rectify it, and charge capacitor banks through current limiting resistors. These resistors prevent the initial current drain of the uncharged capacitors from loading down the transformer. Electrical isolation is provided between the primary and secondary of each transformer. Command signals received after a predetermined time from Pyrotechnic Controller arming can turn on discrete semiconductor power switches in the required sequence to accomplish pyrotechnic events. Each semiconductor switch is a silicon controlled rectifier. It is made conductive by a low level gate signal. The magnitude of controlled current is not dependent on the gate signal amplitude over the minimum amount necessary to fire the silicon controlled rectifier, and current can continue to flow after the gate signal is removed. Current flow continues until the capacitor bank is discharged, or until the bridgewire burns open, at which time the silicon controlled rectifier returns to its nonconducting state. The current limiting resistor to charge the capacitor bank must furnish less current than the minimum holding current for the silicon controlled rectifier to insure turnoff in the event that a bridgewire fails to open completely.

Each semiconductor switch delivers electrical energy pulses through a parallel connection of from one to six current limiting resistors, each of which has a series connected electroexplosive device. The bridgewire in the electroexplosive device burns open in less than three milliseconds, effectively opening the circuit.

In the event of a malfunctioning bridgewire, the capacitor would discharge completely, cutting off the semiconductor switch, and allowing the capacitor bank to recharge for subsequent events. Spurious command signals received simultaneous with the arm signal as a result of mechanical disturbances associated with separation are ineffective to cause pyro events, since the capacitors will have accumulated insufficient energy to initiate the electroexplosive device.

A standardized electroexplosive device is recommended. This device is designed to meet all requirements of the national ranges, and is used for operation of pin pullers, explosive nuts, an electrical disconnect, propulsion valves, and as detonators for initiating the Spacecraft separation joint. All electroexplosive devices have various pyrotechnic compounds but have identical electrical characteristics, and are capable of being initiated from a common switching circuit in the Pyrotechnic Controller. These electroexplosive devices use a common cartridge envelope and match head configuration with pressure cartridge mixes to provide either 3000, 5000, or 8000 pounds per square inch pressure, and detonator cartridge mixes to provide one of two different pulses to meet all anticipated Voyager requirements.

Pin pullers are used to function mechanically as locking devices. Gas pressure released on command by an electroexplosive device retracts the piston and releases the locked device for deployment. The pin pullers are capable of functioning with single or redundant electroexplosive devices without release of damaging gases or fragmentation of parts. Pistons are locked in the extended position by shear pins or shear rings to prevent premature retraction of the piston.

The Spacecraft separation joint will use a Sealed Explosive Application for Linear Separation (SEALS) that employs a mild detonating fuse encapsulated in an elastomer tube jacket. The elastomer tube jacket ruggedizes and protects the explosive core against the detrimental effects of handling, installation, and flight environments. This design concept will part a structural ring between the launch vehicle and the spacecraft circumferentially.

Events such as the release of the magnetometer boom and the unlatching of the scan platform will be sensed as having been accomplished by one or more plunger-actuated, miniature switches. These switches are constructed almost entirely of non-magnetic materials. Stainless steels of number 310 or higher are used in place of the more common, but potentially more magnetic, 302, 303, and 304 stainless steels. The switches are bushing-mounted to permit fine adjustment of the height of the switch, and the mounting nuts are safety wired. The switching chambers are evacuated and filled with an inert gas.

The Pyrotechnic Subsystem is completely redundant, including power supply, energy storage, arming device, wiring from the Electric Power Subsystem to the Pyrotechnic Controller, and by dual harness segments downstream of the Pyrotechnic Controller. Two capacitor banks are discharged into the two bridgewires furnished for each event. Failure of either bridgewire or either capacitor bank will not prevent the event from occurring. Either harness segment with its associated electroexplosive device is capable of performing the required functions.

The Pyrotechnic Controller is located in Electronic Assembly 2 (along with elements of the Power Subsystem). It weighs 7.5 pounds. Weight of the electroexplosive devices is 9.9 pounds.

2.1.9 STRUCTURE

The main elements of the structural configuration are as follows:

a. Capsule Support Structure

b. Equipment Module Structure

c. Spacecraft Support Structure

d. Spacecraft Adapter Structure

e. Solar Array Structure

The structure has been made adaptable to a wide range of capsule weights and associated propulsion subsystem weights. It has been sized for the extremes of the 1971 mission with a 2300-pound Capsule and a 3500-pound propulsion subsystem, and to the 1977 mission with a 4500-pound Capsule and only 500-pounds of propulsion. Should there be later perturbations of Capsule and propulsion weights within the limits indicated, the structural configuration will remain unchanged.

Non-magnetic materials are used throughout the spacecraft, specifically aluminum alloys, structural fiberglass, aluminum rivets, and titanium fasteners.

The largest machined part is the Spacecraft adapter which incorporates the field joint between the Launch Vehicle and Spacecraft, and the separation plane. It is a machined aluminum ring, 10-feet in diameter, and is the only major machining problem to be encountered in manufacturing.

The Capsule support structure is a semi-monocoque structure with beaded skins and twelve longerons, six primary and six secondary. Aluminum alloy is used throughout. The six primary longerons are located at the Capsule attachment points. By means of shear lag, the six secondary longerons pick up load so that at the Capsule support structure/equipment module interface all twelve longerons are effective. This interface is a manufacturing joint, with the Capsule support structure being manufactured as a subassembly.

The equipment module is an integrated unit which, to be structurally adequate, requires interaction between the basic structural assembly consisting of the upper and lower rings and longerons, the twelve thermal control shear panels and the packaging assemblies. The upper and lower rings, and the longerons are machined from aluminum alloy. Eight of the twelve longerons have machined mounting pads to which the propulsion module is attached.

The Spacecraft support structure is also an aluminum semi-monocoque structure; it provides the transition structure between the Spacecraft adapter and the equipment module. A manufacturing joint is made at the equipment module/Spacecraft support structure interface. The Spacecraft support and the adapter are manufactured as a single subassembly.

The solar array structure is composed of an annular ring of solar panels. These solar panels are supported by twenty-four spars extending outward from the Spacecraft support structure; they are manufactured from aluminum webs and spar caps. The solar panels are constructed of epoxy fiberglass skins and aluminum honeycomb core bonded together with 0.012 aluminum edge members to form a sandwich panel on which solar cells are mounted. The total structural weight is 433.4 pounds.

2.2 PHYSICAL DESCRIPTION

The configuration for the 1971 Voyager Flight Spacecraft is shown in Figure II-16, physically divided into its five major elements:

a. Capsule Support Cone

b. Equipment Module

c. Spacecraft Support Cone

d. Solar Array Assembly

e. Propulsion Module

The paragraphs which follow describe each of these physical assemblies, the location of the functional subsystems, and the particular design problems of each assembly. A weight summary is given in Table II-7 for both the functional subsystems and the major physical assemblies.

During the design of the Flight Spacecraft, the requirements of ground handling, accessibility, maintenance and repair, and of subsystem and system testing were considered in addition to the functional requirements of the mission and the subsystems. Some of the design features that have been realized are:

a. Removable panels for accessibility to internal equipment.

b. Accessibility to both test and operational harness connectors.

c. Manageable segments for the solar panel assembly for safe, easy replacement.

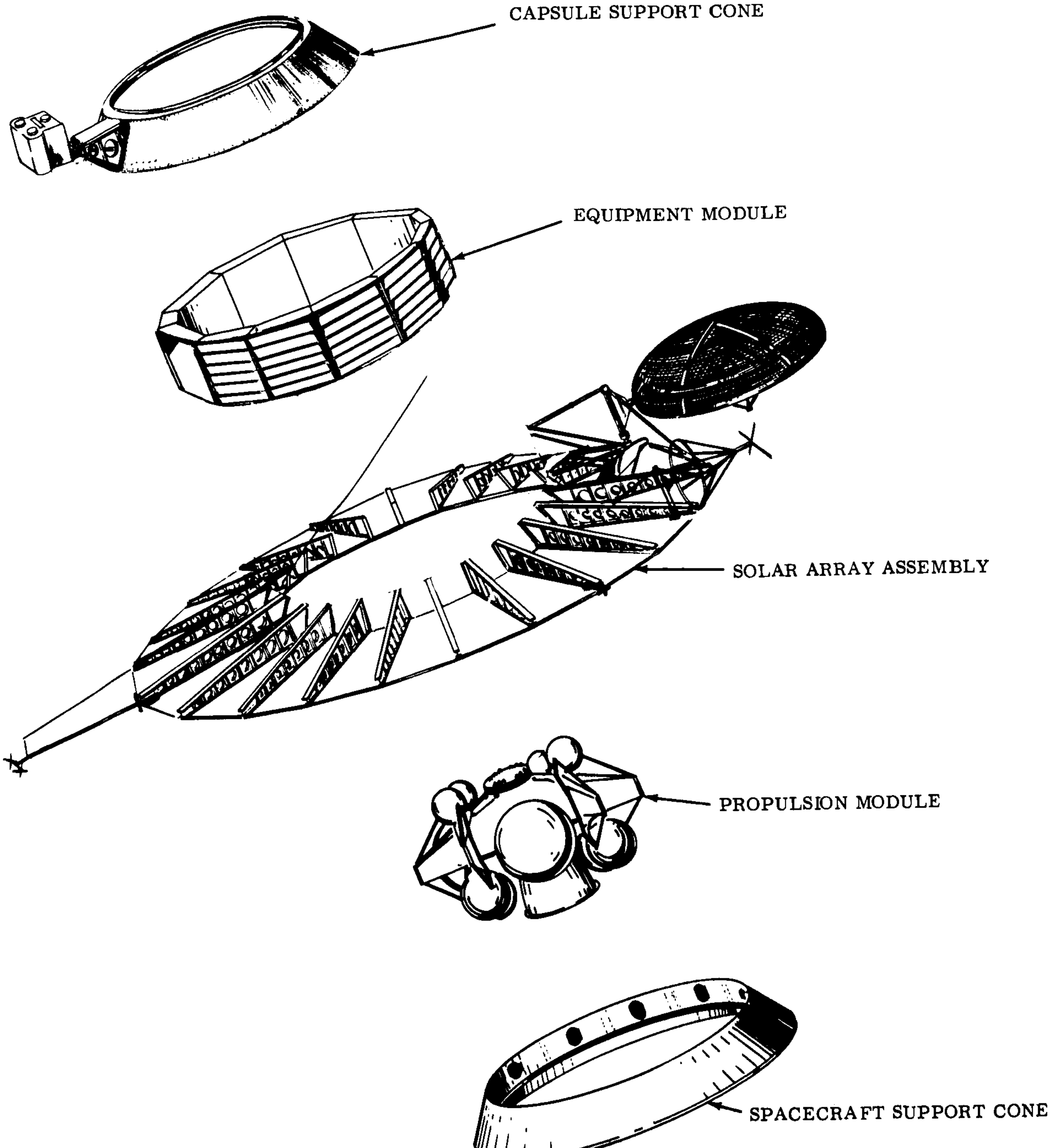


Figure II-16. Exploded View of 1971 Flight Spacecraft

Table II-7.

1971 Voyager Spacecraft Weight Summary

FUNCTIONAL SUBSYSTEMS	WEIGHT (LB)	PHYSICAL ASSEMBLIES	WEIGHT (LB)
Spacecraft Science Payload	262.2	Capsule Support Cone	273.6
Telecommunication	322.4		
Propulsion	3769.0	Equipment Module	872.3
Guidance and Control	216.3		
Electrical Power	414.0	Spacecraft Support Cone	250.1
Temperature Control	120.2		
Controller and Sequencer	18.0	Solar Array Assembly	407.9
Pyrotechnics	17.4		
Structure (incl. Harness)	433.4	Propulsion Module	3469.0
Weight Margin	227.1	Weight Margin	227.1
Total Flight Spacecraft	5500.0		5500.0

d. Modular assembly of the propulsion subsystem.

e. Most critical sensors (except sun sensors) are mounted to the equipment module for relatively easy alignment.

f. With the minor exceptions of the Pyrotechnic Controller and the Controller and Sequencer, all functional subsystems are isolated in one or more electronic assemblies (bays). Thus the Power Subsystem electronics occupies three bays.

The Flight Spacecraft as designed may be shipped as one single unit. There is no need to remove equipment for movement and shipping from location to location.

2.2.1 CAPSULE SUPPORT CONE

The capsule support cone provides the interface between the Flight Spacecraft and Flight Capsule. The interface is a simple field joint with the Capsule separation joint being on the Capsule side of the interface. The structure consists of an upper and lower ring, 12 longerons, and 12 skin panels. The skin panels are removable to provide access to test connectors during system testing.

This support cone provides the mounting for the Planet Scan Platform. It also supports the superinsulation cocoon for the upper portion of the spacecraft. Its total weight is 273.6 pounds.

To optimize the mounting of the planet scan platform and provide the view angles required, the lower half of the Capsule bio-barrier will be separated from this cone prior to orbit insertion. While this requires one additional separation event, alternate approaches to locating this platform were less attractive. They required that the scan platform be deployed an extreme distance from the Bus which had the following disadvantages:

a. Long cable runs between the instruments and their associated electronics.

b. Protecting the scan platform during the launch environment was difficult.

c. The scan platform would be exposed to sunlight a portion of the time making thermal control more difficult, adding to the Spacecraft solar pressure unbalance, and making possible reflected sunlight into the optics.

2.2.2 EQUIPMENT MODULE

The Equipment Module houses nearly all of the Spacecraft electronic equipment. It also provides the mounting for the Separation Gas Jet Assembly, the Canopus Tracker, and the Approach Guidance Sensor. The Scan Platform is rigidly attached to the Equipment Module prior to its deployment after Mars orbit insertion.

The Equipment Module has 12 sides, measures 100 inches across the flats, and is 26 inches high. It weighs 872.3 pounds. The basic frame consists of an upper and lower ring and 12 longerons. The electronic equipment is packaged in assemblies (bays) which bolt to the frame and which serve to carry structural loads. This concept of integrating the electronic equipment and the structure has been successfully employed in Ranger and Mariner.

Figure II-17 shows the packaging arrangement of equipment within the Equipment Module; the electronic packaging has been standardized to the greatest possible extent. Three levels of standards are used:

a. Electronic Assembly

Each of the 12 assemblies offers approximately 20-inch by 20-inch by 9-inch packaging volume (including harness). This permits the use of up to 15 subassemblies of the standard size or an appropriate number of special subassemblies. The subassemblies are sandwiched between two plates, as shown in Figure II-18. The inner plate provides a mounting base for the subassemblies as well as a supporting

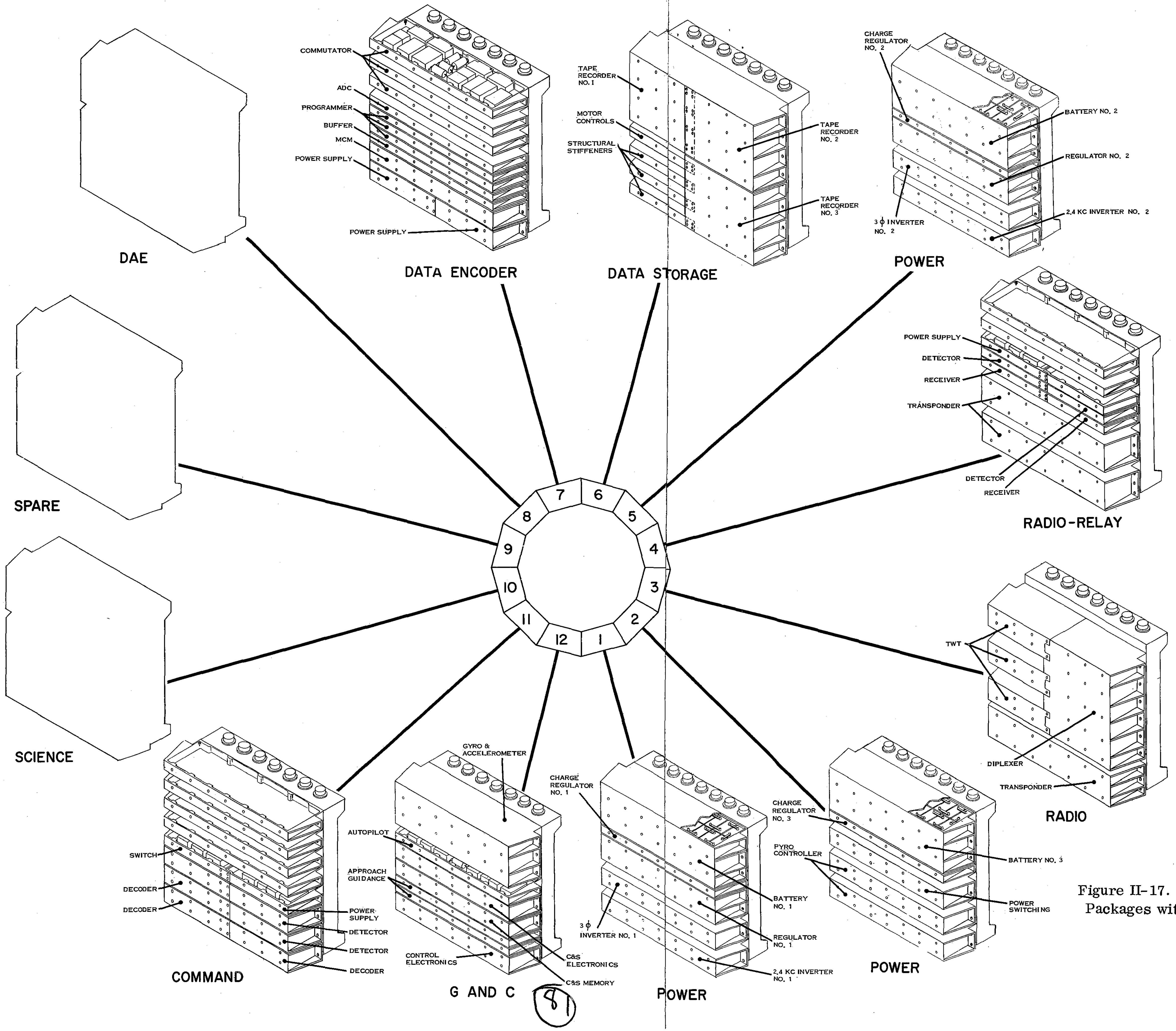


Figure II-17. Arrangement of Equipment Packages within the Equipment Module

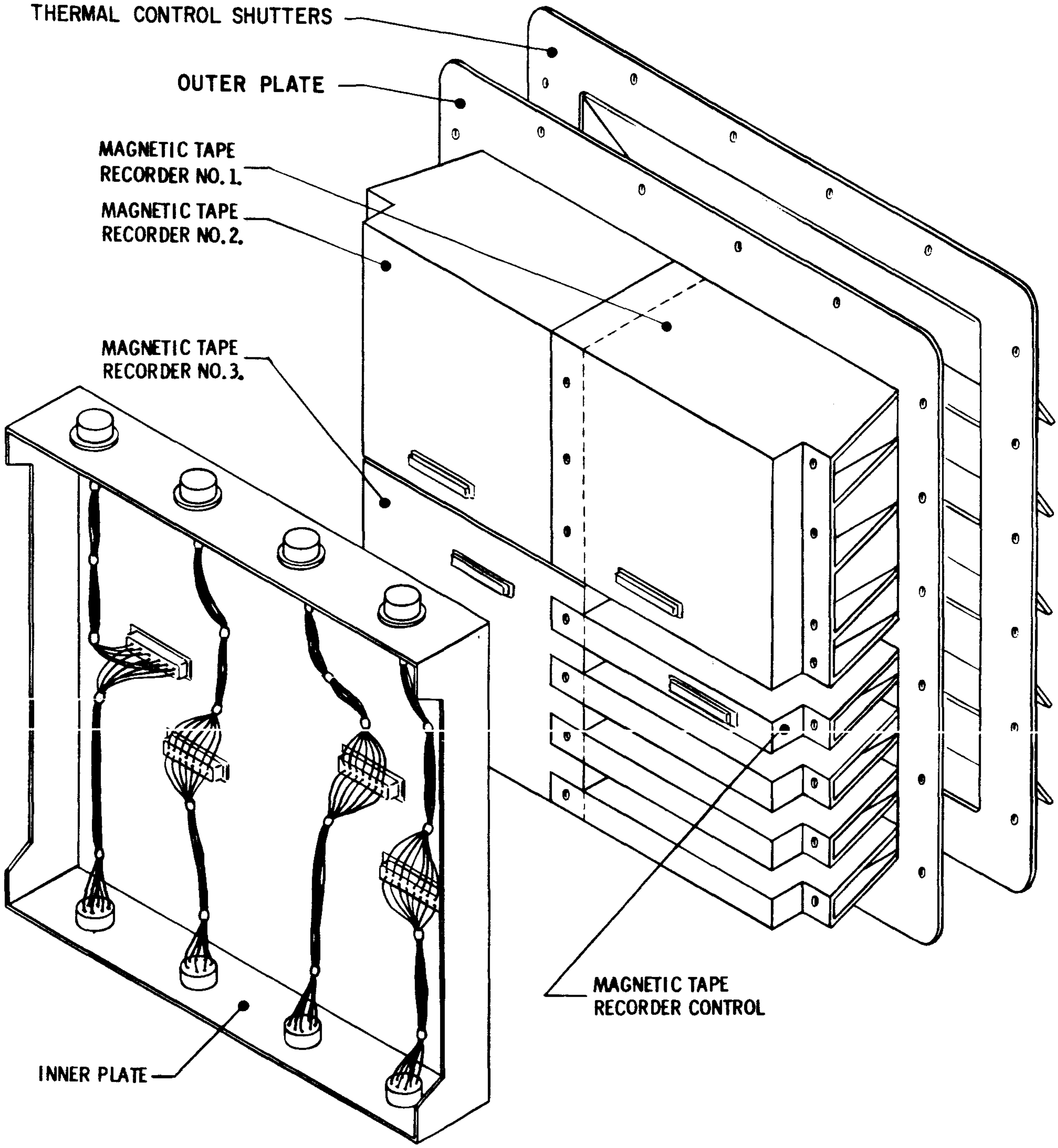


Figure II-18. Exploded View of Electronic Assembly 6 structure for the assembly harness and connectors. The outer plate is a shear plate as well as support for the thermal shutter assembly.

b. Electronic Subassembly
Subassemblies are functional and testable units of each subsystem. They are constructed on a machined subchassis with adequate stiffness to prevent damage due to shock and vibration. The standard dimension is 20-inch by 6-inch by 1.25 inch. To accomodate bulky and non-standard parts such as transformers, chokes and capacitors, and components such as gyros, tape recorders, and radio equipment, the standard thickness is allowed to vary in integral multiples of 1.25 inches.

c. Electronic Modules
Circuits are packaged into cord-wood modules consistent with circuit performance requirements. The module dimensions are standardized to assure maximum utilization of the given subchassis area without sacrificing flexibility. A minimum number of processes and materials will be utilized, to assure more effective control of manufacturing operations.

Considerations of magnetic cleanliness will be extended to the module level through control of part lead materials, interconnections and hardware.

2.2.3 SPACECRAFT SUPPORT CONE

The spacecraft support cone carries flight loads from the Capsule and Flight Spacecraft to the Launch Vehicle interface. The interface again consists of a simple field joint with separation occurring on the Spacecraft side of the interface. Separation in this case is accomplished by an encapsulated mild detonating fuse which separates the conical structure a few inches from the interface plane.

Much of the hardware associated with the Attitude Control Cold Gas Jet Subsystem is located inside this cone, including the tanks, regulators and plumbing. Across the lower position of the cone is a lightweight bulkhead which serves to support the lower superinsulation cocoon and also provides micro-meteoroid protection to the propellant tanks. The Cruise Sun Sensor is mounted to this bulkhead. Total weight is 250.1 pounds.

The solar array structure is mounted to the Spacecraft support cone.

2.2.4 SOLAR ARRAY ASSEMBLY

The solar array structure consists basically of 22 panels and 23 support ribs. A total cell area of approximately 200 sq ft is provided. Other equipment mounted to the solar array support ribs includes the high gain antenna, medium gain antenna, magnetometer boom, UHF relay antenna, primary and secondary low gain antenna, secondary Sun sensors, and cold gas jets with their solenoid valves. A deployable solar vane is located opposite the high gain antenna to balance the solar pressure torque produced by the mesh antenna. Total assembly weight is 407.9 pounds.

2.2.5 PROPULSION MODULE

The propulsion module is designed to be removable as a unit from the Spacecraft. It consists of four spherical tanks for the bipropellants, four spherical tanks for the monopropellants, four pressurization tanks, thrust chambers, and associated components and plumbing. Design of this system as a removable unit allows the system to be assembled and tested without requiring subsequent disassembly for installation into the Flight Spacecraft. The propulsion structure attaches to fittings provided in the Equipment Module. Total weight of the bipropellant and monopropellant systems, including propellants, is 3469.0 pounds.

2.3 MISSION SEQUENCE

Launchings for the Voyager will take place at Cape Kennedy, Florida. Two pads of the AFETR facility will be used giving the capability of launching two Overall Flight Spacecraft in an interval as small as two days. A launch period of approximately 54 days (45 days minimum) is available in 1971 with a minimum daily firing period of two hours.

Only Type I trajectories are considered for the 1971 mission. Trajectories bounded by a launch period of April 30 through June 23, arrival date of November 1 through November 15, and a maximum asymptotic approach speed of 3.5 km/sec are recommended because of the following trade-off considerations:

a. Reasonable propulsion system weight
b. Two hours minimum viewing time of Syrtis Major by Goldstone at Capsule impact
c. Early Earth occultations by Mars
d. Late and minimum Sun occultations by Mars
e. Viewing conditions for scientific experiments
f. A maximum injection energy (C_3) of 18 km^2/sec^2
g. Minimum declination of the outgoing asymptote of minus 33 degrees.

The mission sequence for Voyager is depicted in Figure II-19. The description that follows is referenced to that figure; each paragraph heading corresponds to an illustrated flight status.

2.3.1 LIFT-OFF TO SHROUD SEPARATION

During the launch to injection phase, AFETR tracking and telemetry coverage will be provided for Launch Vehicle and Spacecraft telemetry by DSIF Station No. 71; the telemetry rate is 106-2/3 bps. Telemetry data from the Overall Flight Spacecraft is relayed by the Launch Vehicle during the boost phase. From lift-off to fairing ejection, full Spacecraft telemetry is transmitted using a parasitic antenna located on the fairing; after fairing ejection, communication is from the launch antenna radiating at 100 milliwatts.

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Figure II-19. Mission Sequence

If desired, interplanetary scientific measurements can be initiated immediately after fairing ejection.

2.3.2 HELIOCENTRIC ORBIT INJECTION

After ascent, the Spacecraft is injected into a parking orbit and coasts in this orbit from 2 to 25 minutes, after which the Spacecraft is injected into a heliocentric Mars trajectory by a second burn of the Centaur stage.

2.3.3 CENTAUR SEPARATION

The separation of the Overall Flight Spacecraft from the Centaur is initiated from the Centaur. After separation, the Centaur is backed away from the Overall Flight Spacecraft by employing a retro-rocket thrust of sufficient magnitude to satisfy the planetary quarantine requirement and in a manner to avoid collision with the Spacecraft during Sun acquisition.

2.3.4 SUN AND CANOPUS ACQUISITION; FIRST DEPLOYMENT

Upon separation from the Centaur, the Controller and Sequencer, Attitude Control Cold Gas Subsystem and Pyrotechnics are enabled by dual separation switches. A separation-initiated timer starts the deployment of the antenna, solar pressure balance vane and magnetometer boom, after which the communication link is switched to a low-gain antenna radiating at 50 watts. Sun acquisition is accomplished within 20 minutes after the enabling of the attitude control, and Spacecraft power is then derived from solar energy rather than the on-board batteries. The start of Canopus acquisition is delayed until 1000 minutes after Spacecraft separation in order to calibrate the magnetometer; the Spacecraft turns at a controlled rate about the Sun axis during the period from Sun acquisition to Canopus acquisition. Canopus is acquired within 70 minutes after the initiation of the search; upon acquisition, attitude control is switched to the normal cruise mode.

2.3.5 INTERPLANETARY CRUISE

During the majority of the transit time from Earth to Mars, the Spacecraft remains Sun/Canopus attitude stabilized and transmits continuously at 106-2/3 bps. The transmitted data consists of alternate frames of commutated engineering data, science data and capsule data at a rate of 71, 25, and 10 bps, respectively. When the Spacecraft-to-Earth distance has increased to approximately 15-million km, a 7.5-foot diameter parabolic high-gain antenna is pointed towards Earth and communication to Earth is maintained through the high-gain antenna radiating at 20 watts. The Canopus sensor cone angle and antenna pointing angles will be updated approximately 5 and 250 times, respectively, during the cruise phase, as commanded by the Controller and Sequencer.

2.3.6 MIDCOURSE TRAJECTORY CORRECTION

The cruise phase will be interrupted from one to four times to perform trajectory corrections; one correction will be made within ten days after launch and one correction will be made in conjunction with the capsule separation maneuver. The time and magnitude of the corrections are determined from the trajectory tracking data. Prior to the maneuver, quantitative maneuver commands are sent from Earth and stored in the Controller and Sequencer. Before changing the high-gain antenna to the maneuver orientation, the Earth link is switched to the maneuver mode, which uses a secondary low-gain antenna radiating 50 watts; engineering data are transmitted at 3-1/3 bps and both the capsule data and the engineering data are stored in a magnetic core memory at 13-1/3 bps for later transmission. After the high-gain antenna is orientated, a series of three turns are made by the Spacecraft to obtain the correct maneuver orientation and to point the high-gain antenna toward Earth. The radio is returned to the cruise mode (high-gain antenna and 20 watts), and the stored data is transmitted at 2133 bps. Upon completion of the stored data transmission, telemetry is switched to the cruise mode (106-2/3 bps) and the Spacecraft maneuver orientation is verified. Orientation during the velocity change is maintained by the autopilot.

After engine burning, communication is switched to the maneuver mode, Sun and Canopus references are acquired, high-gain antenna is oriented to Earth, and communication is switched to the cruise mode, in that sequence.

2.3.7 APPROACH GUIDANCE (NOT SHOWN IN FIGURE II-19)

Prior to the separation of the Capsule, improvement in the uncertainty of the Spacecraft position with respect to Mars can be obtained by approach guidance which starts taking measurements when the Spacecraft is approximately 500,000 km from Mars. During the period that the approach guidance is used, the telemetry cruise mode is changed so that the engineering, approach guidance, capsule, and science rates are 51, 27.6, 10 and 18 bps, respectively; however, the number of engineering and science channels remain unchanged with the additional approach guidance channels handled by changing the sampling rate.

At approximately fifteen hours before encounter (200,000 km), the field of view of the approach guidance sensors are encompassed by the Mars image; the approach guidance is turned off and the telemetry returned to normal cruise mode.

2.3.8 FLIGHT CAPSULE SEPARATION

On the basis of the radio tracking and approach guidance measurements, the quantitative commands for the combined Capsule separation and trajectory correction maneuver are received by the Spacecraft from Earth and stored in the Controller and Sequencer. The maneuver proceeds as for the trajectory correction maneuvers. After verification of the maneuver attitude, the capsule is separated from the Spacecraft in a direction opposite to its thrusting direction; one minute after the Capsule release, the Spacecraft is given a 0.5 fps velocity increment to slow it down and allow the capsule to be propelled across the Spacecraft path a sufficient distance in front of the Spacecraft to avoid collision and contamination by Spacecraft attitude control gases. At Capsule separation, the relay receiver in the Spacecraft is turned on and remains on until after capsule impact on Mars. The separation attitude including orientation of the high-gain antenna is maintained for ten minutes to allow the capsule to be monitored during the capsule trajectory injection engine-burn, and to allow the data to be transmitted directly to Earth without storing.

2.3.9 FINAL TRAJECTORY CORRECTION

After the ten minute wait period, Spacecraft communications are switched to the maneuver mode (low-gain antenna, 50 watt, 3-1/3 bps transmitted, 13-1/3 bps stored), and the maneuver procedes as for a standard trajectory correction maneuver, except that the time to obtain the trajectory correction orientation is minimized to conserve battery power.

2.3.10 CRUISE ACQUISITION AND LOWER BIO-BARRIER SEPARATION

After the completion of the maneuver, the Spacecraft is returned to the cruise attitude as per a standard maneuver sequence. The lower portion of the bio-barrier is separated from the Spacecraft eight hours before encounter and continues past Mars on a fly-by trajectory.

2.3.11 MARS ORBIT INJECTION

The trajectory prediction is updated and the quantitative data for the orbit injection maneuver stored in the Controller and Sequencer upon receipt from Earth. The maneuver attitude is obtained similarly to that for trajectory correction maneuvers in sufficient time so that the capsule can be viewed by the Spacecraft from entry to impact. This capsule data as well as engineering data are transmitted to Earth at 106-2/3 bps via the high-gain antenna radiating at 20 watts; in addition, the capsule data is stored for play-back at the completion of the orbit injection maneuver. Sometime before the latest expected capsule impact time, the Spacecraft attitude for orbit injection is verified from Earth. After Capsule impact, the Spacecraft is injected into a nominal 3000 by 25,000 km (as measured from the surface) direct elliptical orbit about Mars with injection occurring in the Southern hemisphere near the evening terminator. The orbit is inclined 40 degrees to the Mars equatorial plane. Injection is accomplished by a bipropellant engine with the four monopropellant engines being controlled by the Autopilot for thrust vector control.

2.3.12 ORBIT ACQUISITION AND PLANET SCAN PLATFORM DEPLOYMENT

After orbit injection, the Sun and Canopus are once again acquired, the high-gain antenna pointed towards Earth, and the buffer data transmitted to Earth at 2133 bps. The scan platform is deployed and commanded to its initial position to point the scan platform instruments at the Mars local vertical. All scientific instrument covers are removed, and the planetary scientific measurements are initiated.

2.3.13 MARS ORBIT OPERATION

In about twice the nominal orbit period of 19.3 hours, the orbit parameters are precisely determined by radio tracking and the scan platform orientation is updated. The orbit parameters may be relayed to the Data Automation Equipment or through the Command Decoder and Controller and Sequencer to the Data Automation Equipment; Spacecraft and possibly science cyclic functions are controlled by the Controller and Sequencer. The scientific data is stored on magnetic tape recorders for non-real time playback and sent real time with engineering data at a combined science and engineering rate varying from 8533 bps to 544 bps. An engineering back-up mode transmitting through a fixed medium-gain antenna radiating at 50 watts has a capability of 544 bps during the early orbit phase.

During portions of each orbit during the early orbit phase, the Earth will be occulted by Mars, providing the capability for making measurements of the Mars atmosphere by the radio propagation method. This occultation is shown in Figures II-20, II-21 and II-22, in which three illustrations of an Orbit Characteristics Model are shown. Figure II-20 is a general view. Figure II-21 shows the orbit as viewed from the Sun. (Occultation of the Sun does not occur for the range of launch period and trajectories dispersions until approximately 150 days after orbit injection.) Figure II-22 shows the orbit as viewed from Earth, showing the occultation. During the occultations, engineering data will be stored in a buffer for later transmission. The reduced radio transmission time may require changes in the rate at which scientific data is obtained by the scientific instruments.

During the last 30 to 60 days of the six-month orbit mission phase, the Sun will be occulted from the Spacecraft by Mars for a part of each orbit, requiring a change in scientific instrument sequencing and Spacecraft control. During long occultations, Spacecraft control is switched to inertial, transmitted power and data rate reduced, and power to the scientific instruments and Data Automation Equipment is reduced to a minimum. Power to operate the Spacecraft during occultation is derived from batteries which are recharged during the sunlight portion of the orbit.

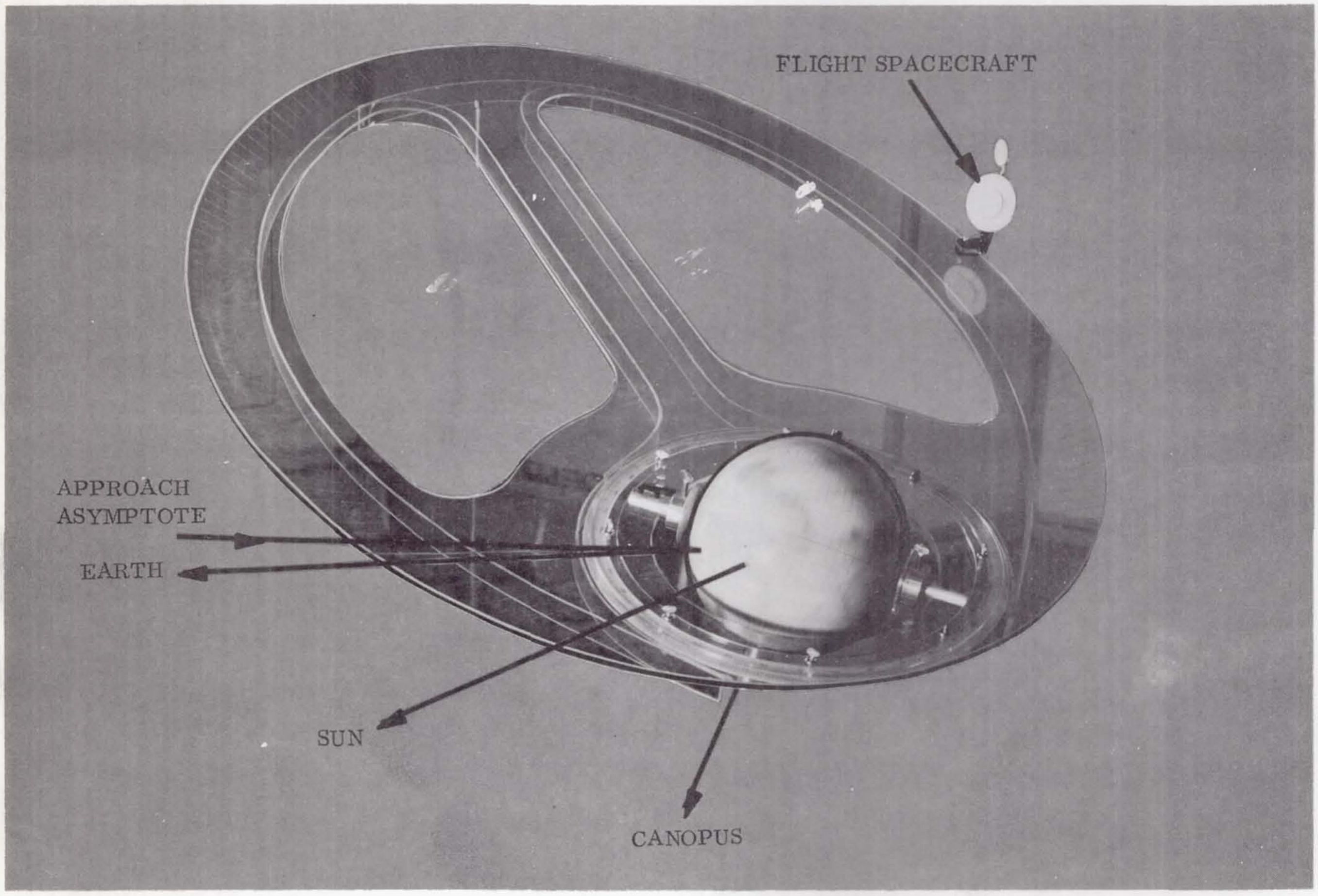


Figure II-20. Orbit Characteristics, General View at Encounter

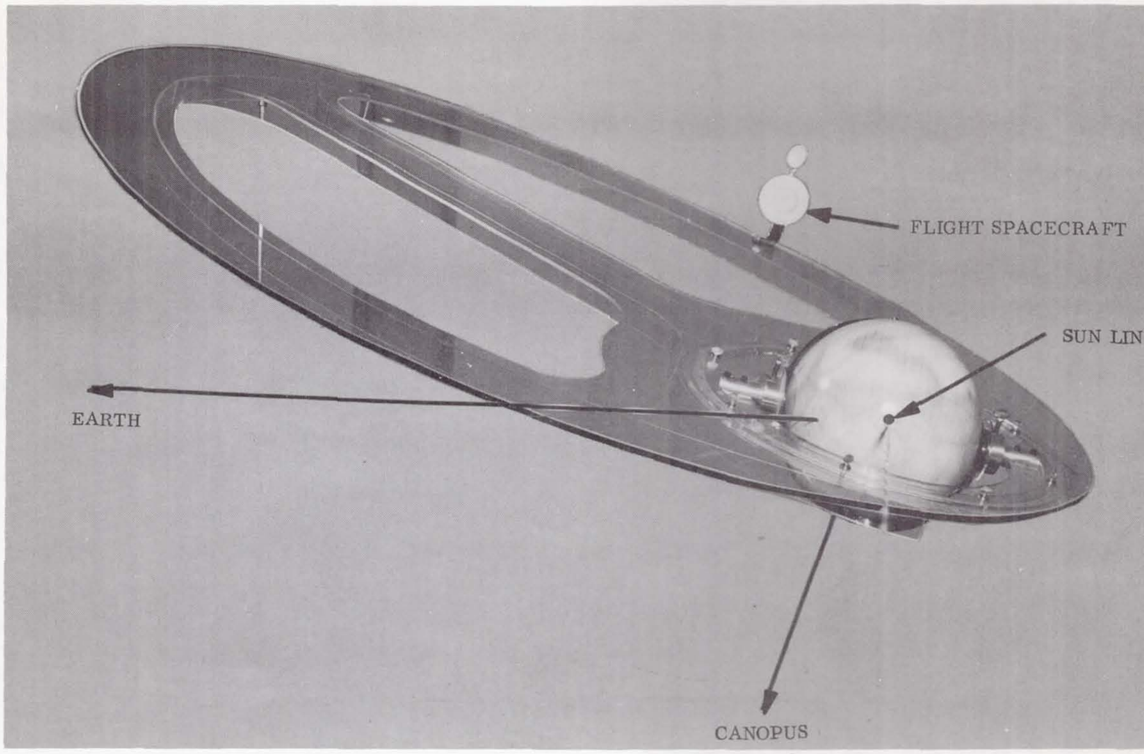


Figure II-21. Orbit Characteristics, Viewed from Sun Line at Encounter

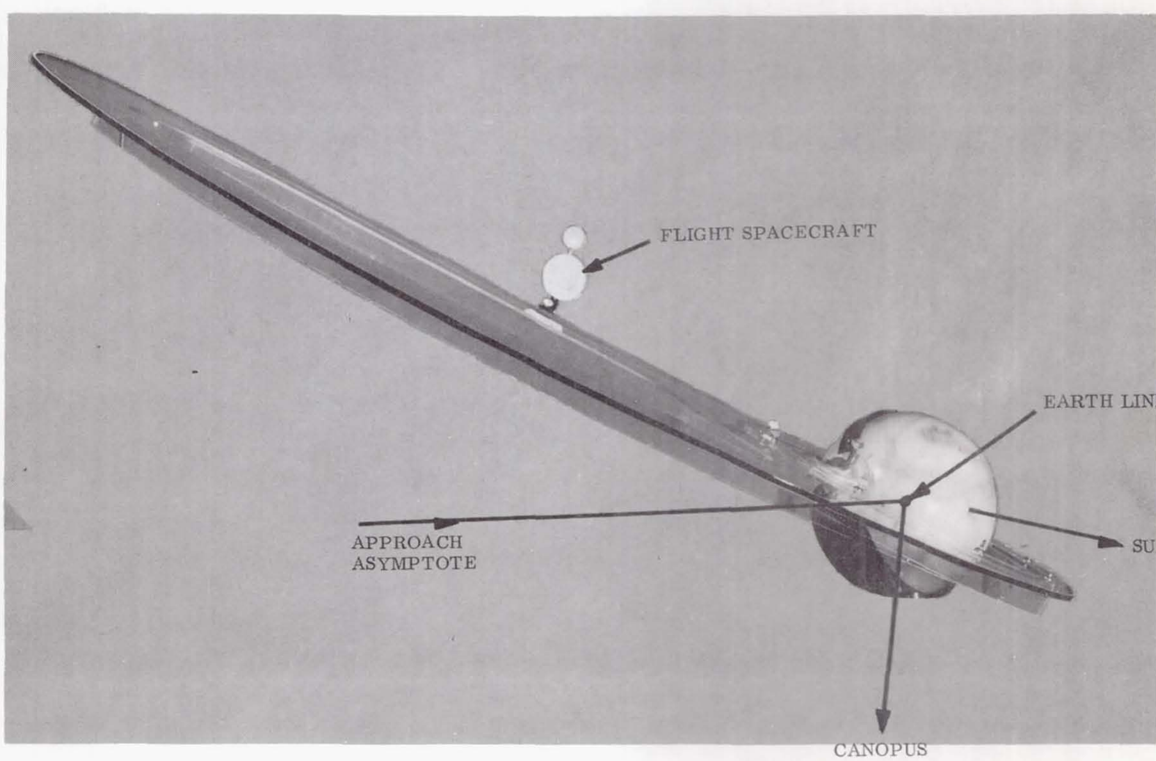


Figure II-22. Orbit Characteristics, Viewed from Earth Line at Encounter

2.4 OPERATIONAL SUPPORT EQUIPMENT

2.4.1 DEFINITION

Operational Support Equipment, (OSE) is defined as being all of the equipment required to assemble, handle, test and prepare the Spacecraft for its mission. In addition, OSE includes the equipment and software required by the Deep Space Network to enable it to control the mission. The OSE is grouped into the following four subsystems:

a. System Test Complex (STC) - The STC is the support equipment configuration used to perform system tests and simulated flights on the assembled Spacecraft. Its parts include OSE which, when removed from the configuration, become the test equipment for the Spacecraft subsystems.

b. Launch Complex Equipment (LCE) - LCE at the launch area controls ground power and monitors for abort criteria during pre-launch. LCE at the Explosive Safe Area is used for final confidence testing of the overall Flight Spacecraft and for support of propellant and pyrotechnic installation.

c. Assembly, Handling and Shipping Equipment (AHSE) - AHSE provides mechanical support.

d. Mission Dependent Equipment (MDE) - MDE is communication components and software needed by the Deep Space Network for Voyager operations.

The OSE primarily supports operations at the Spacecraft Checkout Facility and Launch Area of the Eastern Test Range. The configuration, characteristics and availability of the OSE, however, have been defined with the objective of also utilizing it to support the development, fabrication, and test of the Spacecraft at the contractor's facilities and operations at JPL.

2.4.1.1 TESTING PHILOSOPHY

The testing philosophy which was adapted to guide the OSE design has the following significant aspects:

a. To use equipment and equipment configurations successfully applied to similar spacecraft, rather than approaches requiring more development and hence having more risk and reliability hazards.

b. To depend more upon the capabilities and judgment of the engineering personnel who will use the OSE than upon sophistication and capability within the OSE.

c. To use manual control of OSE in all aspects except those where overriding considerations, based upon the flight hardware or mission, establish the desirability to the using engineers of having automatic or computer controlled capability.

2.4.2 DESIGN GUIDELINES

The design guidelines used for the Voyager OSE result from having as a principle objective, use of the Mariner-C OSE design approach and OSE components. Insofar as the System Test Complex is concerned, the guidelines are:

a. To enable the cognizant subsystem engineer to control the testing of his subsystem. Hence, the STC concentrates all practicable test control and decision making at the consoles of the subsystems.

b. To have system tests simulate actual mission configurations as far as practicable. The principle sources of test stimulation in a system test are, accordingly, the flight program of stored commands and the commands issued to the Radio Subsystem by the OSE. The analysis and evaluation of the performance and capability of each of the flight subsystems is accomplished by monitoring the flight subsystem through telemetry and hard wire test points. This approach is a proven, low risk approach, successfully implemented in the Mariner C program.

c. To maximize compatibility between the System Test Complex and the DSN. In cases where compatibility is not necessarily consistent with the most efficient or economic design approach, the compatibility criteria was considered to be dominant, and was used. An example of this is the use of digital computers in the System Test Complex. In order to assure complete compatibility with DSN procedures, and to use proven approaches, the recommended Voyager System Test Complex includes two digital computers utilized in a manner which duplicates the DSN planned method of (1) telemetry decommutation, and (2) command generation and verification. These functions might be more economically or more efficiently mechanized through using a single digital computer with the proper characteristics, if one were available.

d. To preserve procedural and functional identity with the Mariner C System Test Complex in the recommended Voyager System Test Complex. Several STC components having a high degree of physical analogy with Mariner C counterparts can be used, if available, for modification and integration into the Voyager STC.

e. To minimize MDE and establish MDE-STC compatibility. In configuring the STC, the mission-dependent hardware and software is used. This leads to the decision to include in the STC configuration, some of the general purpose equipment planned for the DSN, so that the MDE software could be used identically in the STC.

f. To simplify and minimize the OSE needed in the Explosive Safe Area and the Launch Area. The loading and verification of the flight program into the Spacecraft are pre-launch functions allocated to the STC, as is the evaluation of TLM data. The STC, although physically remote, performs these support functions, while the extent and complexity of LCE is kept small.

g. To simplify the design of AHSE items, to be consistent with electrical OSE guidelines. The most obvious impact has been in the recommended design of the Spacecraft Handling Fixture. The recommended approach to testing the Guidance and Control System, after it has been integrated with the spacecraft, is to use static testing instead of dynamic testing. This recommendation was based upon in-house experience in testing of other spacecraft which indicated the low value of dynamic testing after the subsystem has been tested dynamically during development testing and once it was assembled with the spacecraft. The recommended Spacecraft Handling Fixture is therefore a simple fixture to hold the spacecraft in several orientations, rather than a complex high precision equipment such as an air bearing.

2.4.3 DESIGN APPROACH TO STC

The STC, from the point of view of scope and complexity, is the principal item of OSE for the Voyager. Before delineating specific design features of its components, the general design approach to be taken was considered. Three alternative configurations were considered ; these configurations are illustrated by the simplified block diagrams shown in Figures II-23, II-24 and II-25.

Figure II-23 shows the design approach to the STC which was implemented on the Mariner C program. Its features are:

a. The cognizant subsystem engineer, through the use of his subsystem OSE, is the key to the analysis and evaluation of the performance of his subsystem during system testing.

b. The subsystem OSE in the STC, when augmented by some auxiliary equipment, can test the functionally isolated subsystem. The OSE used in the System Test Complex can, therefore, be the same OSE used to test the subsystem before it is integrated into the Spacecraft.

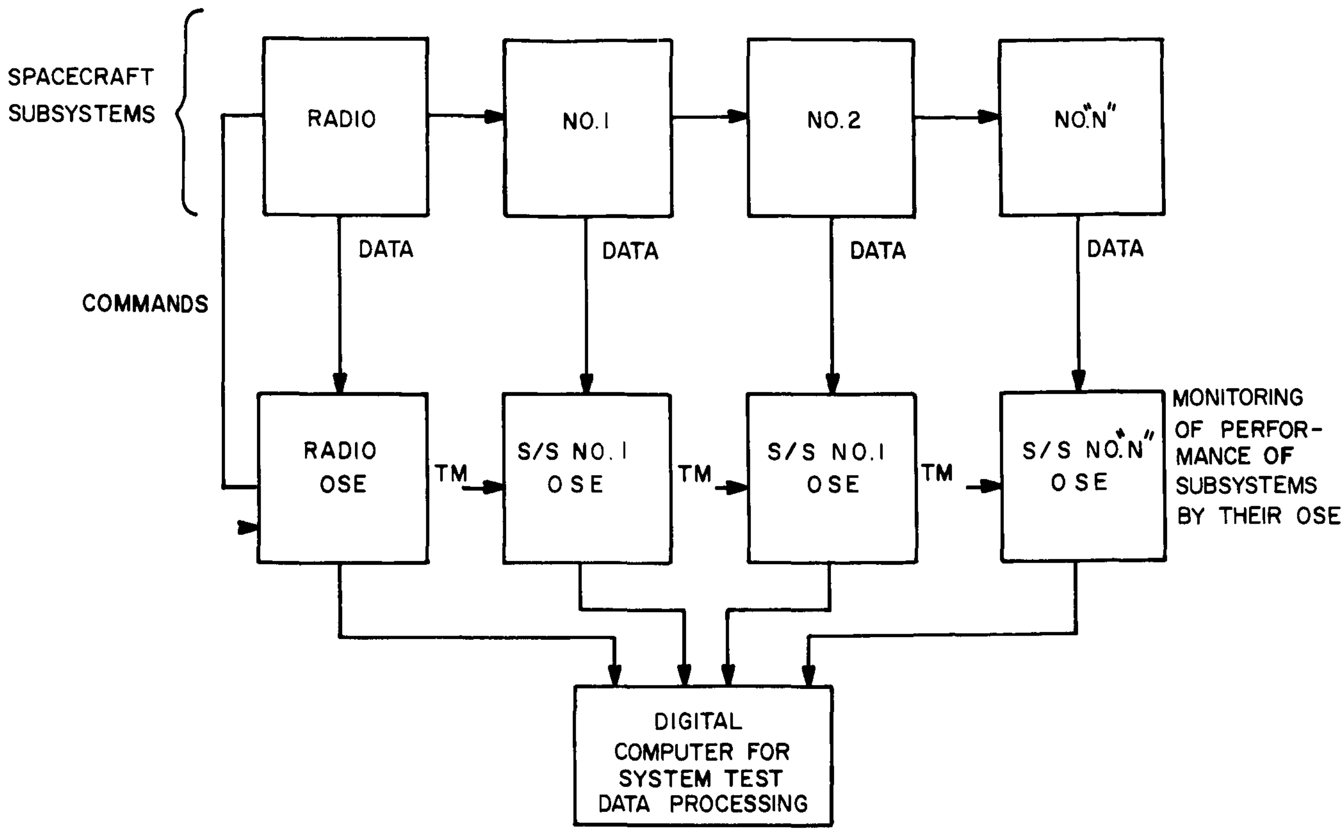


Figure II-23. Mariner C, STC Block Diagram

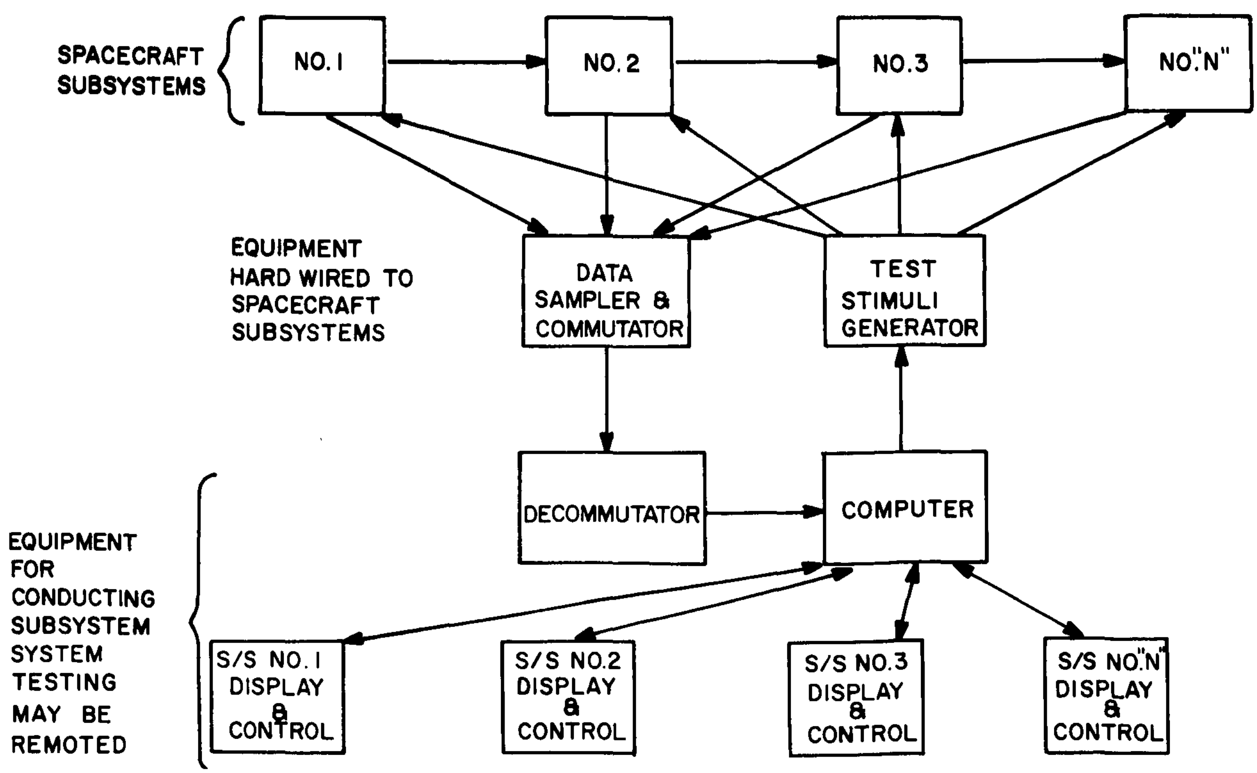


Figure II-24. Apollo System Test Equipment Block Diagram

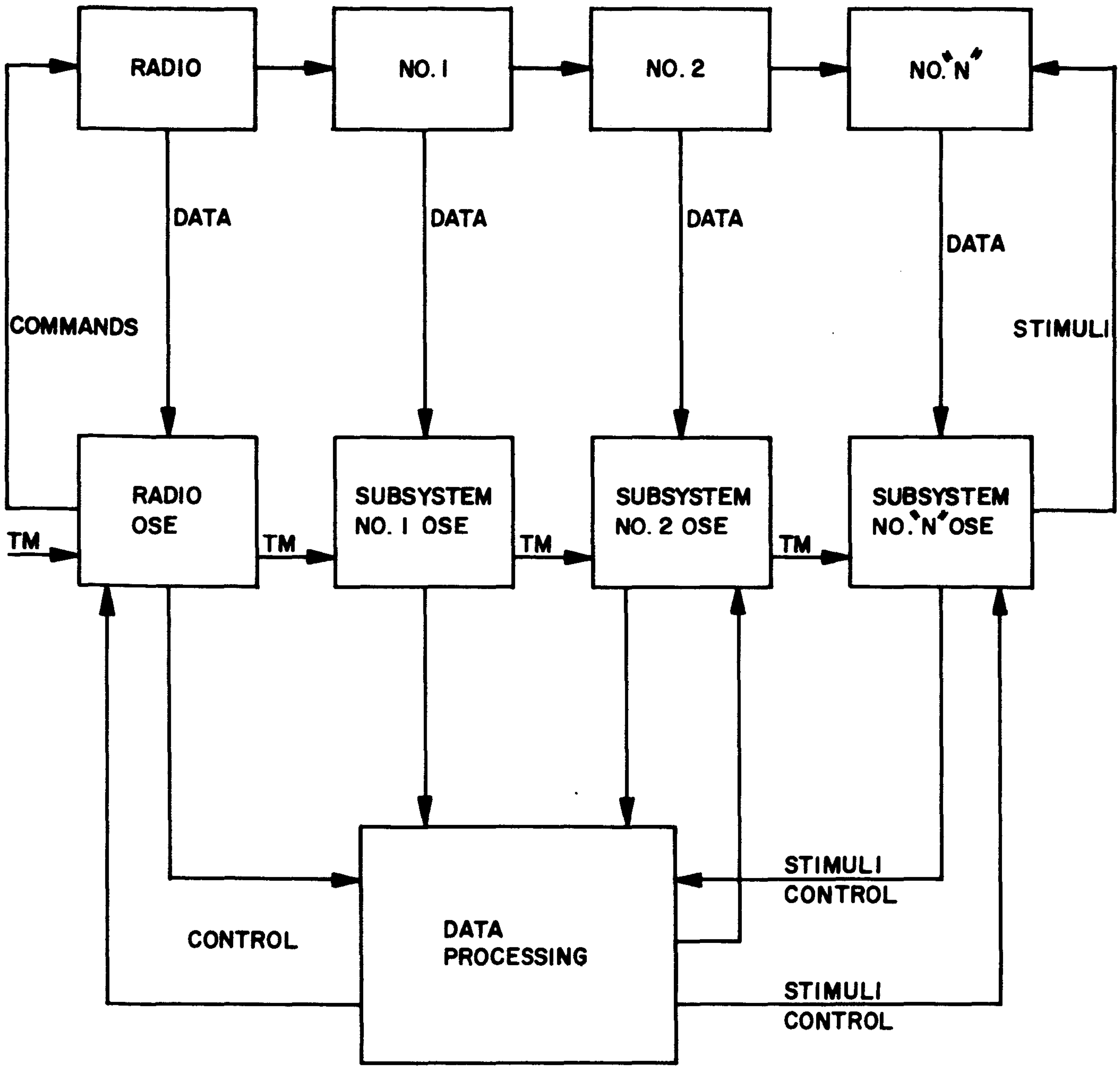


Figure II-25. Voyager STC Block Diagram

c. The OSE for the various subsystems feed data to a central processing system which provides integrated records and displays for the system test.

Figure II-24 shows the design approach to system and subsystem testing implemented on the Apollo Program. Its features are:

a. Data monitoring and control of stimuli to initiate test sequences are exercised by the cognizant subsystem engineers from consoles which may be remote.

b. Test Data and Stimuli control are pre-processed, with the information in either direction being in digital form.

c. Only the commutation and sampling and the stimuli generation need be in intimate proximity to the Spacecraft. This setup then permits most of the equipment otherwise required for the launch complex to be eliminated, the same control and display consoles being used for system test and launch control.

Figure II-25 shows the STC design recommended for Voyager. This recommendation resulted from considering the approach used on the Mariner C and the approach used on Apollo, and the requirements and objectives peculiar to the Voyager Program. It is essentially the same approach used for Mariner C. The Mariner C STC design is extended to suit the Voyager situation in two respects:

a. In order to conduct system tests, certain types of stimuli must be applied externally, as the approach of having one subsystem stimulate the other is not completely sufficient to yield a simulated flight mission; an example of this is the stimulation of sensors for Sun and Canopus. In the Voyager STC, therefore, the Data Processing function, at the proper time in the mission sequence, controls the input of signals into the G & C to give Sun and Canopus angle positions.

b. During system testing, the Voyager mission sequence should be controlled and coordinated. This function, as shown in Figure II-25, is allocated to the Data Processing. It consists of such control as C & S reset control and TM format and data rate control.

The physical configuration recommended for the Voyager STC is very similar to the Mariner C STC. The control aspects of system testing are of such scope that they can be handled by the same processing equipment (Computer Data System) used for test data processing in the case of Mariner C.

The recommended Voyager STC is characterized by the utilization of Mariner C equipment, Mariner C testing approach slightly extended to fit Voyager, and the use of equipment which permits each mission-dependent equipment and mission-dependent procedure to be duplicated as a normal STC function.

3.0 1969 SPACECRAFT

3.1 MISSION SELECTION

A test flight in 1969 is postulated as a means of increasing the probability of conducting a successful operational mission in 1971; it was clearly stated that this study should not consider any requirement to satisfy scientific objectives in the 1969 flight test. Further, in accord with JPL direction, the flight test objectives considered here are limited to those which are important to the success of the Flight Spacecraft and its supporting equipment and procedures. Test objectives for the 1971 Flight Capsule and 1971 Experimental Payload, both considered GFE to the Spacecraft contractor, were not considered in selecting the mission concept described in this report.

Obviously, a broadening of the scope of objectives to include overall program concerns, such as Capsule and Launch Vehicle objectives, would emphasize different mission selection considerations, and very possibly lead to alternate conclusions.

The process of mission selection must consider the trade-offs between the mission value achievable in terms of satisfying engineering test objectives, the mission cost, and the effects upon the 1971 program. The first step is to define specific test objectives for the Spacecraft flight so that meaningful comparisons can be made between alternative missions in terms of the number and value of test results that can be obtained, and the timeliness of the answers in terms of their contribution to the 1971 Spacecraft development. A list of such objectives was prepared (see Appendix I to Volume D) considering the following as the general objectives of the 1969 Spacecraft flight test:

a. Demonstrate specific 1971 Voyager spacecraft components, subsystems, and system interactions in a flight test environment involving both planetary orbiting and deep space cruise.

b. Verify the test, launch, and operational procedures planned for the 1971 Voyager operational mission.

c. Demonstrate the adequacy of the Operational Support Equipment (OSE) to be used in conjunction with the 1971 flight.

d. Exercise the interfaces within the program; for example, the interface between the Spacecraft and the DSN, or the interface between the Spacecraft contractor and the Jet Propulsion Laboratory.

Consideration of the detailed engineering objectives leads to the conclusion that almost all of the advantage of the 1969 test flight is in tests of the specific hardware to be flown in 1971; there are no concepts proposed that need further flight verification before their use in the 1971 Spacecraft program. Hence, the approach adopted in the mission selection was to maximize the amount of 1971 Spacecraft equipment that can be flown without modification in 1969.

A number of mission alternatives were evaluated for test value early in the study, including earth orbiting flights, direct ascent to Mars fly-by, lunar orbits and others. Two conclusions were drawn from this study: First, that a large share of the mission test value is associated with use of the main retropropulsion system. This would include such items as demonstration of engine operation per se, and system interaction effects such as autopilot operation with engines firing and effect of plumes upon the Spacecraft. This conclusion in turn indicates a strong desire to have the test mission begin with an Earth-orbiting phase, since the Atlas/Centaur is unable to deliver both basic Spacecraft bus and retropropulsion to an escape trajectory.

The second conclusion is that a flight to Mars does increase the value of the engineering test, but only by a small amount. Specifically, engineering tests of the Mars vertical sensor, the approach guidance sensor, and additional measurements of the magnetically trapped radiation are considered of value in improving the probability of Spacecraft success in 1971. However, these tests were judged to add only about 10 percent to the engineering value of the flight test. Since this change is well within the range of uncertainty of the subjective ratings attached to the relative importance of different test objectives, it was necessary to invoke other considerations as dominant in selecting between two principle mission types: a Earth orbit to Mars fly-by mission, using the retropropulsion system to provide the energy to eject the Spacecraft onto the Mars transfer orbit; or the same mission flown after the Mars opportunity, as an Earth orbit to deep space flight.

Three other considerations were invoked to select between these prime alternatives; these considerations were mission difficulty, cost, and schedule considerations. From the mission difficulty point of view, the Earth orbit to deep space mission is slightly preferable for two reasons. The Earth orbit to Mars mission imposes more trajectory penalties than the deep space shot. For example, energy constraints limit the altitude of perigee to about 200 miles. This imposes several operational problems for tracking the Spacecraft from the DSIF. Second, the weight capability for the Mars fly-by case is somewhat marginal. This will result in making changes to the Spacecraft just to save weight, (e.g. reduction of redundancy), which will reduce the desired similarity between the 1969 and 1971 missions, and increase the program cost.

The second aspect considered was the program cost difference. The difference in cost for these two alternatives was estimated to be 5 percent less for the deep space mission; too small a difference to influence the decision significantly.

The final aspect considered was program schedule. From this standpoint, a distinct preference exists for the later launch date, which defines the mission as Earth-orbit and deep space. There are several reasons for this. The desire to test the main propulsion in space, after a storage period, leads to a requirement for the Spacecraft to have a target launch date more than a month earlier than required for a direct flight to Mars. This is an additional burden upon an already demanding schedule. The overall effect of a Mars flight is to advance the date by which hardware detail design is completed by several months compared with the schedule considered optimum for a 1971 operational flight. This will require either much more detail design to be done during Phase IB, thus partially defeating the intended planning concept of this process, or or else require the Spacecraft design and testing to be accomplished at such a pace that the risk of serious error is greatly magnified.

Further, because the 1969 Test Spacecraft must be released for procurement so early in the development cycle of the 1971 Flight Spacecraft, many inevitable design improvements will not be factored into the 1969 design. Not only does this increase the risk of non-instructive failure in the 1969 vehicle, it further dilutes the desired similarity between the two flight articles.

On the other hand, a flight date of September, 1969, has relatively little effect upon the optimum 1971 program. The 1969 Flight Spacecraft assembly and test preceeds assembly and test of the PTM (Proof Test Model) by only a few months. This not only avoids early schedule acceleration but also paves the way for the 1971 PTM in terms of training and experience. The net effect to the program is similar to building additional copies of the PTM, except for the differences imposed by the choice of Launch Vehicle, and flying it instead of putting it into a ground thermal vacuum life test. In addition, a flight date of September is early enough so that flight test data is useful for any required 1971 Spacecraft modifications.

In summary, of the factors considered in selecting a mission for the Atlas/Centaur that best compromises an engineering test of the Spacecraft and program considerations, the later flight is preferred for these reasons:

a. The difference in engineering test value of a Mars fly-by versus a deep space shot is too small and too subjective to be decisive

b. The cost difference is too small and uncertain to exert much influence on the answer

c. Mission flexibility slightly favors a deep space shot

d. Schedule considerations strongly favor a later flight.

This does not mean that the earlier flight date for a Mars mission cannot be satisfied; it can, but it will require some acceleration of the program and result in less similarity between the 1969 Test Spacecraft and the 1971 Flight Spacecraft.

3.2 DESCRIPTION OF SPACECRAFT

The mission sequence consists of an Atlas-Centaur launch into eccentric Earth orbit for a period of several weeks. After Earth orbiting tests are complete, the Spacecraft Propulsion System is operated to eject the Spacecraft from Earth into a heliocentric orbit which will cause the Spacecraft to achieve Earth-vehicle and vehicle-Sun ranges comparable to those expected in the 1971 Voyager mission. Two Atlas/Centaur Launch Vehicles with extended Surveyor shrouds are assumed, with launch operations conducted from one pad of Complex 36 at AFETR. The separated Spacecraft weight, including retro-propulsion, is 5150 lb. Figure II-26 shows the 1969 Spacecraft.

The first launch is to be scheduled for early September. In the event of Launch Vehicle failure to achieve orbit, a capability should be provided to make a second launch within approximately one month. If the first launch is successful, the second flight will be postponed for several months, pending results of the first flight. If a Spacecraft flight failure occurs on the first Spacecraft, a "fix" would be applied to the second, and launch made as quickly as possible thereafter, considering pad availability and on-pad operations. If in-flight failure does not occur within the first few months of flight, JPL program management must elect either to launch the second Spacecraft to obtain additional flight experience, or to cancel the launch, use the second Spacecraft for additional ground testing, and return the Launch Vehicle to inventory.

The 1969 Voyager Test Spacecraft will be functionally very similar to the 1971 operational Flight Spacecraft, except that the GFE Flight Capsule and Spacecraft Science Payload (including DAE) will not be carried. The basic 1971 Spacecraft Equipment Module, 1971 Propulsion Module and Planet Scan Platform will be flown essentially unmodified, but the solar array, Planet Scan Platform and antennas will be stowed and deployed differently because of the reduced diameter of the Atlas/Centaur Launch Vehicle shroud. The solar array and the high-gain antennas are redesigned because of the limited volume within the shroud. A comparison of the two Spacecraft configurations is shown in Figure II-27.

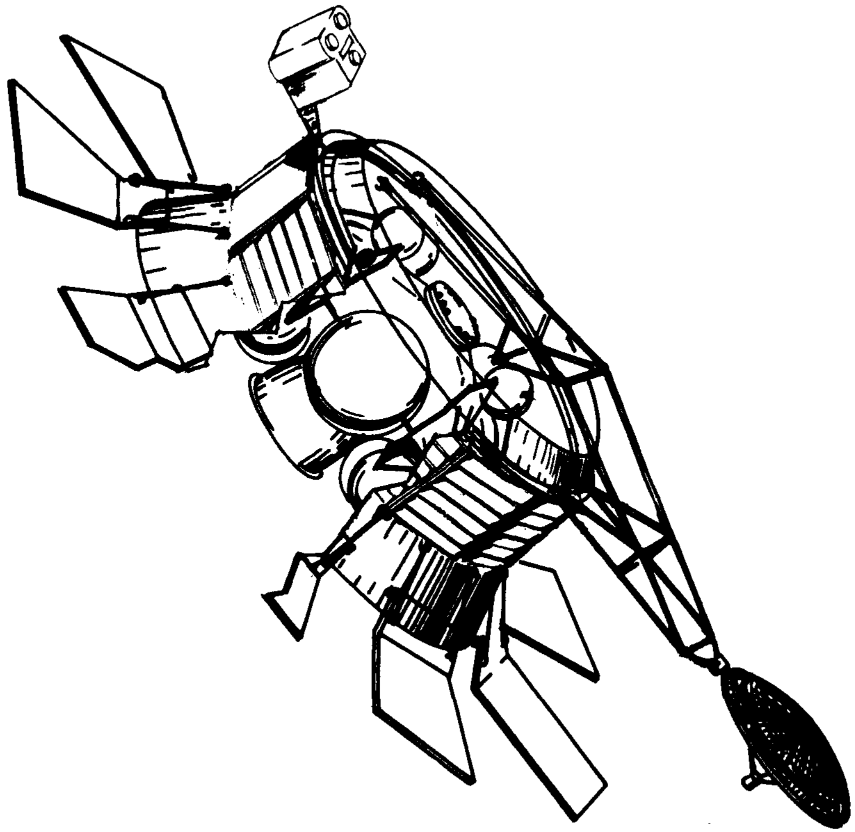
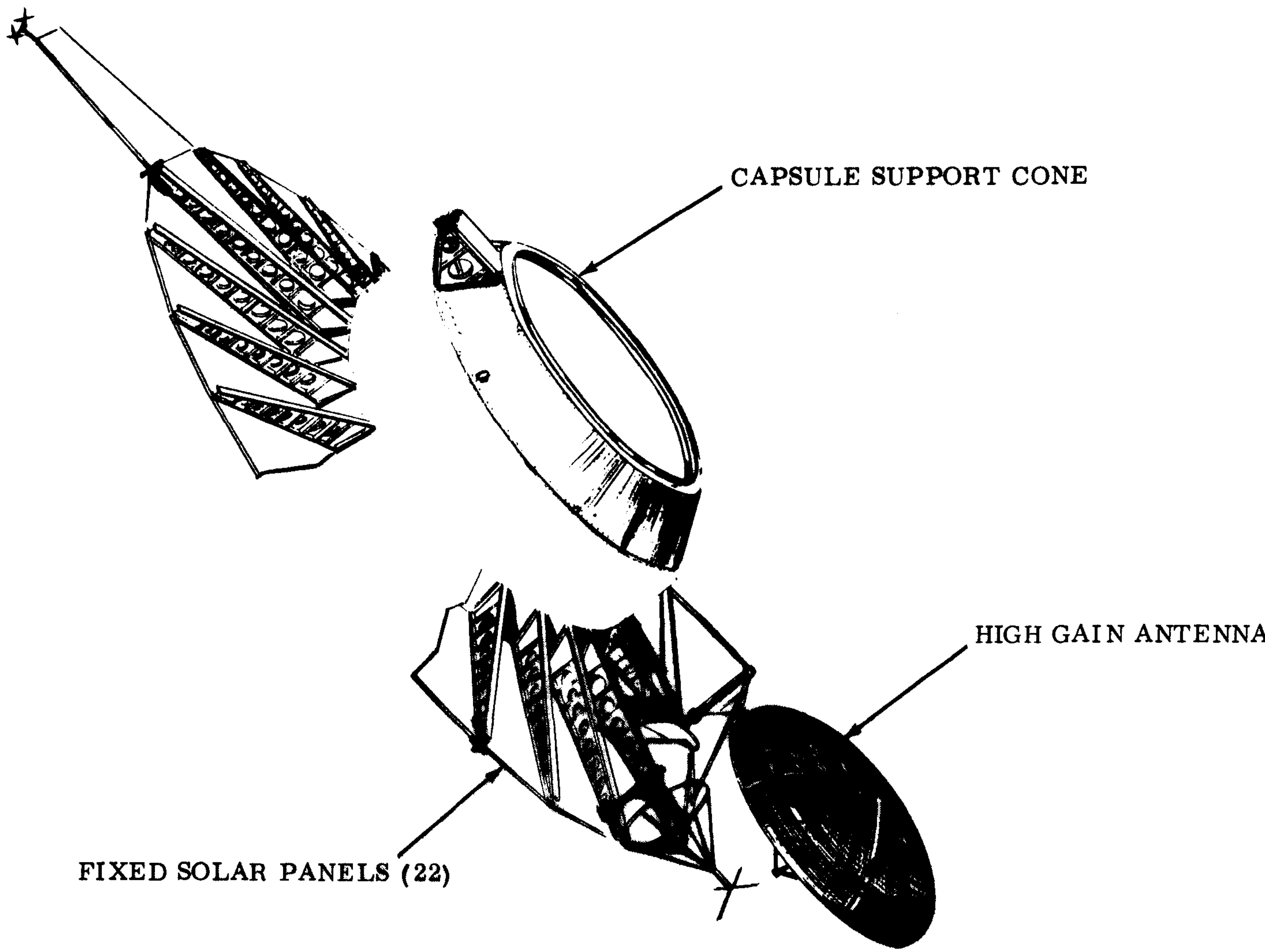
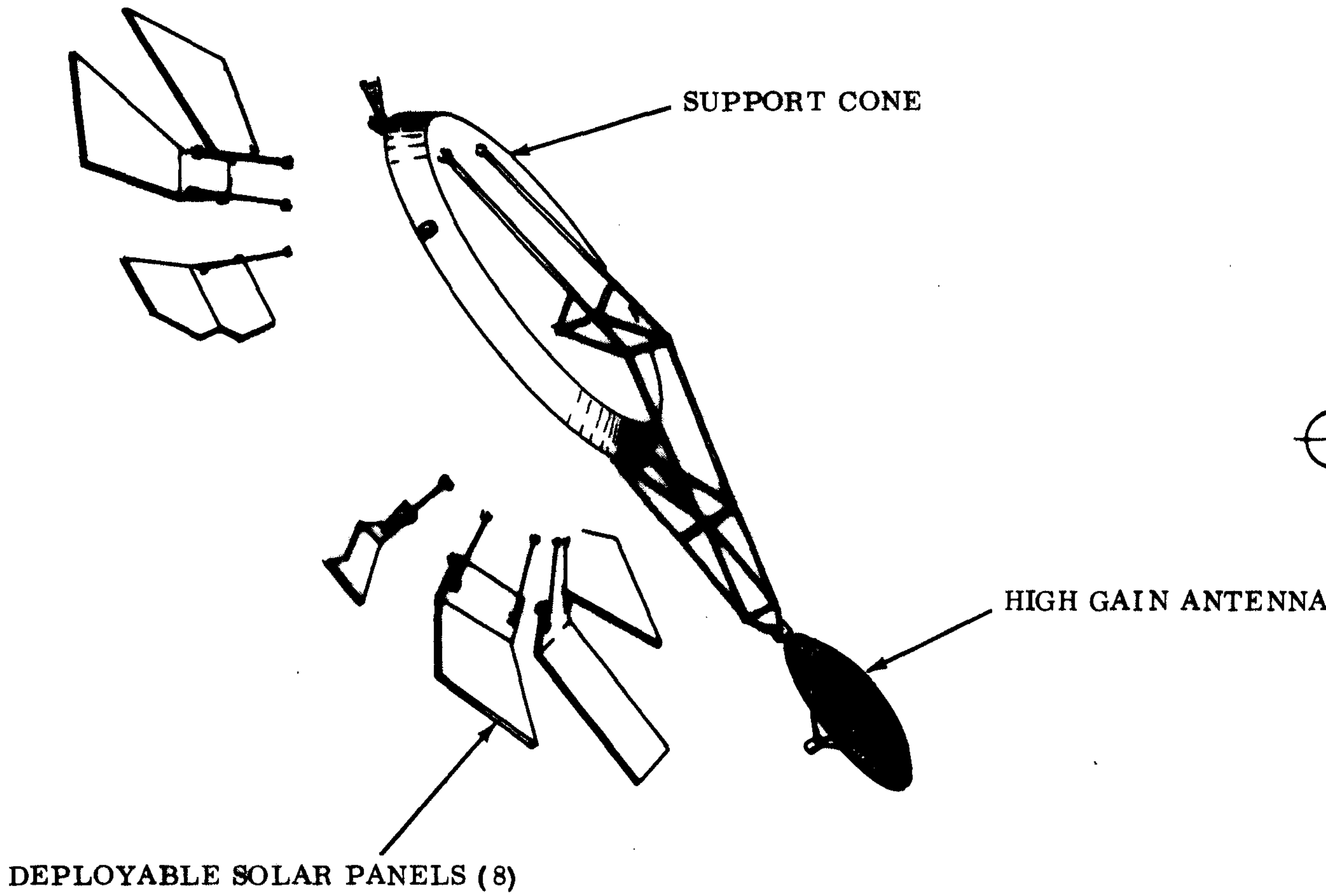


Figure II-26. 1969 Voyager Flight Spacecraft


CAPSULE SUPPORT CONE
HIGH GAIN ANTENNA
FIXED SOLAR PANELS (22)


SUPPORT CONE
HIGH GAIN ANTENNA
DEPLOYABLE SOLAR PANELS (8)

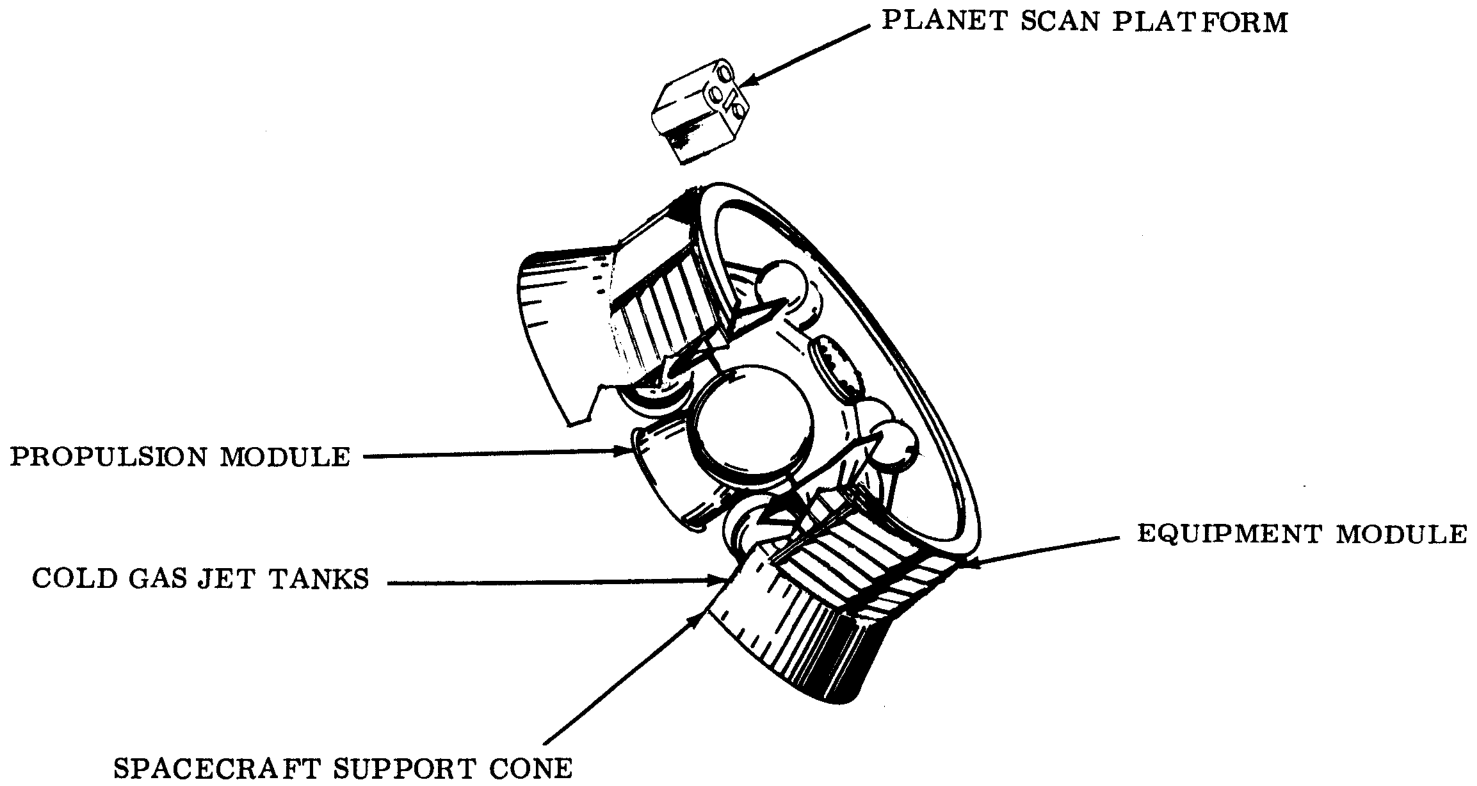


Figure II-27. Comparison of 1969 and 1971 Voyager Flight Spacecraft

Provision has been made in the Spacecraft design for a nominal experiment payload to be included on a non-interference basis if this is desired by JPL and NASA management. Fifty pounds of weight are allocated for this purpose; space is available in bays 8 and 10 and in the Planet Scan Platform, 16 watts of power are allocated during most of the mission, and 10 bits per second of channel capacity is available except during maneuvers and special engineering tests.

The description of the 1969 Test Spacecraft can best be done by comparison with the 1971 preferred design to show differences. Despite every effort to minimize changes between 1969 and 1971, the use of the volume-limited Atlas/Centaur imposes a number of significant design variations. Use of the Spacecraft in Earth orbit in order to accommodate a propulsion test introduces other variations, although these appear to be of less significance than the diameter imposed changes.

The 1969 Test Spacecraft differs from the 1971 Spacecraft in the following areas:

a. No Capsule, Bio-Barrier or Lander Support Cone.

b. Eight deployable solar panels rather than 22 fixed panels.

c. A 3 ft 9 in. rather than a 7 ft 6 in. antenna; deployment means and gimbal structure are modified also. No medium-gain antenna is used.

d. Less Science Payload. This includes the body mounted sensors, the scan platform sensors, and the electronics in bays 8 and 10.

e. Added diagnostic telemetry. This adds to the electronics in bay 8 and to the harness.

f. Different mounting provisions for the high-gain antenna, the scan platform, and the solar panels.

g. Two batteries rather than three. The battery in bay 1 was chosen for removal since this bay is identical to bay 5. The 1971 bay design will, therefore, be flight tested.

h. Delete the Separation Δ V Motor.

i. Add a Tip-off Motor in order to perform tip-off rate tests.

j. Add four pivot joints to the Attitude Control Propulsion lines to nozzle assemblies located at the ends of the solar panels.

The commonality of 1969 and 1971 spacecraft is shown by Table II-8.

The major design effort involved in preparing the 1969 Test Spacecraft is in the design of the deployable solar panels and new high-gain antenna, the mechanisms and structures for the deployment of solar panels, high-gain antenna, and planet scanner; and in the design of 1969 added diagnostic equipment. In the design, a serious effort will be made to provide spacecraft mass properties and structural dynamic response sufficiently similar to the 1971 Spacecraft to avoid any requirements to modify the autopilot and Attitude Control Subsystem for the 1969 test flight. Additional study will be required to demonstrate that this goal can be met. If it cannot, considerable additional analysis of these subsystems will be required, and the value of the flight test of these items will be reduced. The configuration is sufficiently different that the thermal analysis must be repeated, and the flight test will not be a very satisfying demonstration of the thermal performance of the 1971 spacecraft, although it will provide a good check of the adequacy of the thermal analysis and test procedures. Because of the different interfaces and mission, the system analysis and integration effort must be duplicated for the 1969 flight. Other detailed differences between the Spacecrafts, not immediately apparent, will undoubtedly develop during the design because of the different Launch Vehicle and mission profile. This will increase the engineering effort required, and reduce the test value by some indeterminate amount.

Table II-8. Equipment Comparison of the 1969 and 1971 Spacecraft

Equipment Common to Both 1969 and 1971 Spacecraft		Equipment Used For 1971 Only	Equipment Used For 1969 Only
Name	Modification		
Scan Platform	Delete Science; Change Support	Capsule Support Cone	Support Cone
Equipment Module		Solar Panel Assemblies (22)	Solar Panel Assy's. (8)
Power Bay 1	Delete Battery and Charge Regulator	Science Sensors	Diagnostic Sensors
Power Bay 2		7 1/2 ft Antenna	3 3/4 ft Antenna
Radio Bay 3		Medium Gain Antenna	
Radio Bay 4		Flight Capsule	
Power Bay 5		Bio-Barrier	
Tape Recorder Bay 6			
Data Encoder Bay 7			
DAE Bay 8	Delete Science; Add Diagnostic Electronics		
Spare Bay 9			
Science Bay 10	Delete Science		
Command Bay 11			
G & C Bay 12			
Cold Gas Jet Subsystem	Delete Δ V Motor; Add Pivots & Tip-off Motor		
Spacecraft Support Cone			
Attitude Control Sensors			
Low Gain Antennas			
Thermal Control			
Pyrotechnic Devices	Add Panel Deployment Squibs		
Harness	Add Diagnostic Harness		
Spacecraft Adapter			

3.3 SATURN/CENTAUR BOOSTER

The use of a Saturn/Centaur launch vehicle in 1969 would remove all the undesirable aspects of the 1969 Test Flight resulting from differences in the Spacecraft. The 1969 and 1971 Spacecraft could be identical except for minor design modifications that result from the test flight or from subsequent ground testing. A 2300-pound "Capsule Simulator" can be carried to provide an adequate demonstration of the compatibility of this major interface. All of the 1971 Science Payload can be carried if it is available in time for the test flight.

Maintaining identity between these two vehicles has the following major advantages:

a. The design and development effort is much more efficient in that all design personnel pursue a single design. Duplication of major ground tests such as the Structural Test Model, Thermal Test Model, and Engineering Model Spacecraft is not required. The cost saving from this is substantial.

b. All flight tests are truly representative of the 1971 mission. Questions regarding the adequacy of structural tests, autopilot tests, thermal tests, and deployment tests no longer exist. If a deficiency is uncovered in the flight, it is truly a deficiency of the 1971 Spacecraft design and not of a modified version.

For these reasons, it is strongly recommended that a Saturn/Centaur be considered as the Launch Vehicle for the 1969 Test Flight.

While it has not been analyzed in depth, the Saturn IB vehicle without the Centaur upper stage might be an attractive alternate to the Atlas/Centaur. With the large shroud available, the modifications to the Spacecraft required from a volume standpoint could be avoided. A mission as proposed for the Atlas/Centaur involving an earth orbiting phase followed by use of the retropropulsion system to achieve a deep space trajectory is well within the energy capabilities of the Saturn IB.

Use of the Saturn IB only, provides some cost reduction compared to Saturn/Centaur and removes any question of Launch Vehicle availability in time for the 1969 Test Flight.

SECTION III

IMPLEMENTATION PLAN SUMMARY

1.0 INTRODUCTION

The fundamental project management requirements which must be satisfied to successfully accomplish the Voyager Spacecraft Project are:

a. Achievement of the necessary long-life reliability with a high degree of confidence.

b. Strict schedule control to meet a fixed launch window.

c. Effective management of the Project and spacecraft system to achieve the above within the established cost.

In carrying out the Phase 1A study, General Electric's activities were geared to two primary objectives:

a. To arrive at a conservative, flexible spacecraft design which could: (1) accommodate a variety of spacecraft and lander science payloads, mission profiles, and trajectories, (2) adapt to subsequent missions, and (3) accept technology improvements.

b. To formulate an overall implementation plan which would provide the highest possible confidence in achieving the project management requirements stated above.

The sections that follow summarize the revisions that have been made to the Voyager schedules and implementation plans presented in General Electric's Phase IA proposal.

2.0 SCHEDULE

2.1 1971 SPACECRAFT SYSTEM

In updating the 1971 Spacecraft System Schedule, primary consideration was given to the critical importance of meeting the fixed launch period window. The factors considered most important were: (1) the need for providing allowance throughout the entire schedule to solve problems as they arise, and (2) to develop a work breakdown structure and schedule that can be readily and easily measured.

Figure III-1 shows the schedules recommended by General Electric for the 1971 Spacecraft System Program, and at the bottom of this figure, for the 1969 Flight Test Program. The latter schedule will be discussed in the following section. The guidelines used in developing the 1971 system schedule were those listed in the Voyager Mission Specification, plus the following:

a. An early release of development hardware (first hard design) consistent with reasonable time to convert functional specifications into drawings (six months) is desirable to permit early start of development testing.

b. Subsystem compatibility should be verified in a system test model before release of TA and PTM hardware.

c. Each flight vehicle will be processed through two cycles at the launch site -- the first cycle being a dry run.

Table III-1 provides a detailed analysis of the assumptions made and the rationale on which the 1971 schedule is based. To show how provision has been made throughout the schedule to permit corrections to be made as they arise, the charts also show potential problem areas, the effect they have on the schedule, and the corrective action that can be taken.

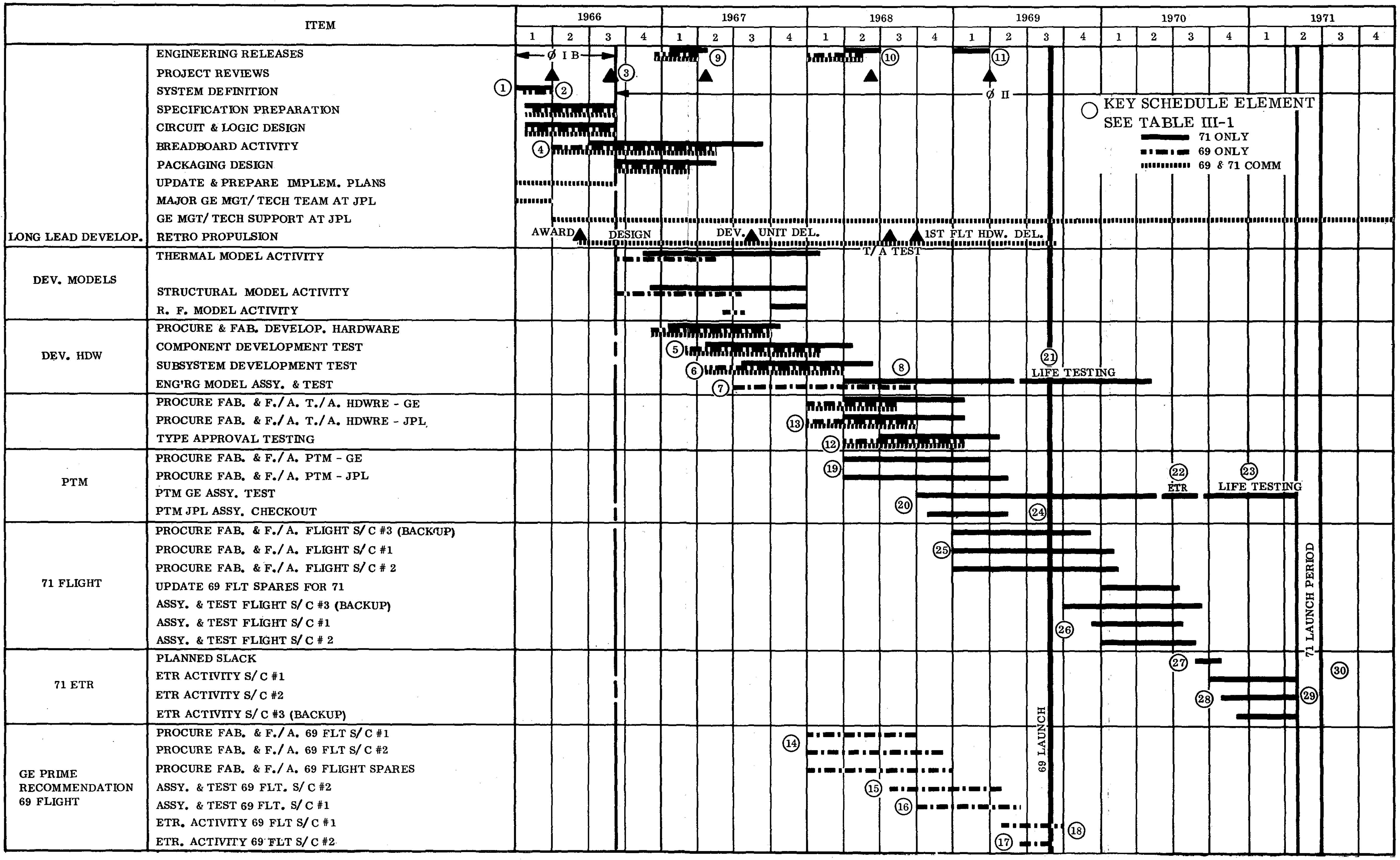


Figure III-I. Recommended Schedule for the 1971 Spacecraft System Program and the 1969 Flight Test Program

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Table III-I. Spacecraft Schedule Rationale

KEY SCHEDULE ELEMENT	ASSUMPTION	RATIONALE	TRADEOFFS		
			ALTERNATE ASSUMPTION (OR POSSIBLE PROBLEM AREAS)	EFFECT ON BASELINE SCHEDULE	POSSIBLE SCHEDULE RECOVERY
1. Initiation of Phase IB	Start Date Jan. 1, 1966 Technical Requirements of '69 and '71 missions will be firm.	Present JPL Planning	Phase IB Start Delayed	Phase IB Period would be shortened by amount of delay	Not Applicable
2. JPL/GE Agreement on System & Subsystem Functional Descriptions and Phase II "Baseline" Project Plan	Key GE Voyager personnel will be in residence at Pasadena until "Baseline" agreement is reached; approximately Mar. 30, 1966.	A period of technical and management transfusion to establish a complete meeting of the minds between key JPL and GE personnel is desirable.	None Recommended	Not Applicable	Not Applicable
3. Advance ordering for Phase II	Continuity of Project effort is maintained between Phase IB and Phase II. Release of long-lead parts and materials immediately at beginning of Phase II	Interpretation of present JPL planning	Hardware commitments not permitted due to delay in start of Phase II.	Certain development subassembly, subsystem and system testing starts will be delayed one-for-one.	1. Reduce "stagger periods between release and test of development hardware, T/A and PTM hardware, and flight hardware. 2. Reduce slack time (higher risk).
4. Breadboard Activity	Phase IB funds will be available for breadboard testing on selected subsystems as required to define system.	Interpretation of present JPL Planning	Breadboard activity not permitted during Phase IB.	An approximate 3-month delay in release to fabrication of long development hardware.	Development test period/ for long life development hardware would be shortened by 3 months, increasing the possible rework of '69 flight S/C hardware.
5. Component Development	Flt. hardware design releases will not be made until after sufficient engineering development to assure performance under ambient, vibration and thermal environments.	Experience indicates that premature release of designs for flight hardware creates costly and time consuming delays late in the schedule.	Make earliest possible design releases.	1. Might expedite schedule. 2. Much higher schedule risk because of probability of more frequent test failures later in the schedule.	1. Not Applicable 2. Expedite required redesign and rework.
6. Subsystem & Model Development	Sufficient testing will be performed on subsystem basis to assure component compatibility and performance under selected environments. Examples: Structural Dynamic Tests Thermal Testing OSE Compatibility Antenna Range Tests Propulsion Hot firing Shroud & adaptor compatibility Separation Tests	Past experience - particularly in the early verification of thermal and dynamic environment.	Unable to accomplish all tests as planned.	1. Risk of significant delay if technical failures are encountered in any of these tests late in program.	1. Combine tests (elimination is not recommended) on a common model. 2. Plan alternate approaches as back-up for possible major problems.
7. '69 Configuration Development Spacecraft	A complete model will be assembled for system performance evaluation, environmental testing and duplication of anomalies of the '69 flight.	Assures subsystem elect. and mechanical compatibility prior to release to fabrication of '69 T/A and Flight hardware.	Eliminate '69 engineering model and proceed to assembly of '69 flight S/C. (Not Recommended)	1. Risk program delay due to technical problems in flight S/C assembly and test.	1. Expedite Required Redesign and Rework
8. '71 Configuration Development Spacecraft	The subsystem development test units and the '71 structural test model will be assembled into the '71 system S/C configuration for '71 system performance evaluation	Assures subsystem elect. and functional compatibility prior to release to fabrication of '71 T/A and PTM hardware	Tradeoffs vary from: 1. Update '69 engineering model to '71 configuration 2. Eliminate '71 engineering model and rely on commonality of 69/71 until PTM is avail.	1. Shortened '69 Devel. S/C Testing adds risk to '69 program schedule due to problems encountered in '69 flt. S/C assembly & test after engineering model has been updated to 71 configuration. 2. Risk to '71 program schedule due to lack of system experience prior to T/A Testing and PTM assembly and test.	1. Expedite Required Redesign and Rework. 2. Shorten T/A, PTM Test and '71 Assembly and Checkout cycles by using 2 or 3 shifts.
9. Releases to Fab. of Development hardware	Development hardware common to both '69 and '71 configuration will be procured and fabricated from formally released drawings with changes documented by CN's using GE internal change notice procedures. No CCB approvals required.	Standard GE/MSD practice assures documentation of all changes and internal hardware control.	None Recommended	Not Applicable	Not Applicable
10. Releases to Fab. of '69 T/A and Flt. hardware and '71 T/A and PTM Hardware	T/A, PTM and '69 Flt Hdw. will be procured and fabricated simultaneously from formally released drawings using GE internal Change Notice. Formal CCB approvals required.	Standard GE/MSD practice permits parallel fabrication and testing with some cost savings. Results of subsequent testing will be factored into the '71 flight Hdw. release.	None Recommended	Not Applicable	Not Applicable
11. '71 Flt. Hdw. Releases	The '71 Flt. S/C will be fabricated from Production drawings. JPL approval required before CCB action.	Interpretation of JPL requirements, early design freeze and tight config. control requiring JPL approval for all changes.	None Recommended	Not Applicable	Not Applicable
12. GE Type Approval Testing	'69 Hdw. to be type approved by GE prior to '69 Flt. For those items that change substantially for the '71 configuration additional T/A Hdw. will be fabricated to '71 configuration and T/A tested.	Assures confidence in Hdw. under over stress requirements prior to '69 Flt. and '71 Flt.	Delay all type approval testing until '71 T/A and PTM Release. (Not Recommended)	Risk to '69 program schedule and Flt. success due to problems encountered during Flt S/C assembly and test and actual Flight.	Expedite Required Redesign and Rework.

Table III-I. Spacecraft Schedule Rationale (Cont'd)

KEY SCHEDULE ELEMENT	ASSUMPTION	RATIONALE	ALTERNATE ASSUMPTION (OR POSSIBLE PROBLEM AREAS)	TRADEOFFS EFFECT ON BASELINE SCHEDULE	POSSIBLE SCHEDULE RECOVERY
13. JPL Type approval Hardware	A complete set of T/A Units will be fabricated, flight acceptance tested and sent to JPL for type approval.	In accordance with JPL requirement. Independent testing gives added assurance of reliability of design.	Non Recommended	Not Applicable	Not Applicable
14. '69 Flight Hdw.	Acceptance test on a sub-assembly basis prior to assembly in the system	Standard GE/MSD practice assures performance under vibration and thermal/vacuum environments	None Recommended	Not Applicable	Not Applicable
15. '69 Flt. Spacecraft #1 (Back up)	Spacecraft #1 to be used as a back-up vehicle for '69 flight. Will be launched as 2nd '69 Flt. if req'd.	Assembly, Flight Acceptance Testing, Compatibility & Operational procedures are exercised on a back-up Flt. test vehicle prior to use on the '69 Flt. S/C #2 The Prime '69 launch vehicle	Fly first '69 Flt. model.	Could improve schedule by one month with added risk to flight by flying S/C #1 without walk-thru of an additional S/C.	Not Applicable
16. '69 Test Flight Spacecraft #2	Acceptance test prior to shipment	Standard GE/MSD practice assures performance under vibration and thermal/vacuum environments	None recommended.	Not Applicable	Not Applicable
17. ETR Activity '69 Flight Test	S/C #1 available at ETR 4 months before launch for walk-thru. S/C #2 at ETR 3 months before launch	To assure a double walk-thru in the limited time prior to Flt. Assure back-up S/C is avail. for Flt. in case of emergency	Allow more time at ETR because of use of Centaur Launch vehicle	Would squeeze '69 schedule on a one-for-one basis.	Could allow '69 launch date to move out approximately two months.
18. '69 Launch Period	Will be a deep space probe after approximately two months in Earth orbit. Nominal launch date of 1st Flt is Sept. 1, 1969.	See Volume D -Appendix 1	See Volume D - Appendix 1	See Volume D - Appendix 1	See Volume D - Appendix 1
19. GE & JPL PTM Procurement and Fabrication	Release for Fabrication of both PTM's simultaneously	PTM must be built to final '71 Flt. configuration per JPL requirements and GE/MSD standard practice.	None Recommended	Not Applicable	Not Applicable
20. Proof Test Model: 1 PTM at GE 1 PTM at JPL	Full PTM system testing at GE; independent test program at JPL.	Basic flight assurance test program; higher reliability for mission success due to independent testing	None Recommended	Not Applicable	Not Applicable
21. Engineering Model Life Testing	'71 Engineering model is suitable for system life testing.	Considerable life testing data can be gathered prior to assembly of '71 Flt S/C	None Recommended	Not Applicable	Not Applicable
22. '71 ETR Walk-Thru	The GE PTM will be utilized for ETR Walk-thru then returned to GE for more life testing and '71 Flt. anomaly simulation and testing.	Experience has shown walk-thru testing on other than Flt. S/C very prudent to schedule maintenance	Skip walk-thru with PTM and depend on double walk-thru of each S/C	Would allow for more system life testing prior to launch	Not Applicable
23. PTM Life Testing	System life testing can be conducted on GE PTM after completion of ETR Walk-thru	Considerable life test data from a Flt. Qual S/C can be gathered on PTM prior to the '71 launches.	Conduct System Life Testing on JPL PTM in Parallel with GE PTM Testing.	Would allow for more system life testing prior to launch	Not Applicable
24. S/C-DSN Compatability Check	In addition to a S/C simulator previously suppl. by GE, the JPL PTM will be sent to Goldstone for DSN compatibility check prior to '71 launch.	Represents least interruption to overall program while accomplishing necessary compatibility check.	Use only the simulator to conduct DSN-S/C compat. check.	Would improve overall schedule situation at an increased risk to the mission.	Not Applicable
25. Procure and Fabricate '71 Flt. Hdw.	Release to Fabrication of all (3) '71 Flt. S/C delayed until Elect. & Mech. S/S Compat. has been proved on PTM	Per JPL requirements and GE-MSD Standard Practice	None Recommended	Not Applicable	Not Applicable
26. '71 Flight S/C	Acceptance Test prior to shipment to assure performance under vibration and thermal/vacuum environ. Backup S/C #3 to precede Flt S/C #1 and #2	Standard GE-MSD Practice. Processing backup S/C 1st provides a source of spares tested to system level if failures occur in the Flt S/C.	None Recommended	Not Applicable	Not Applicable
27. Planned "Slack Period"	Plan for the unexpected by allowing a 2 month slack period over that required for field pre-launch operation.	Experience has indicated this is desirable to accommodate unforeseen events during space-developement programs with fixed launch periods.	Insert program slack earlier in critical areas. (Not Recommended)	Not Applicable	Not Applicable
28. '71 Launch Operations	a) 1st Flight Spacecraft available at ETR 6 mos before launch. b) 6 months required in field for dry run and final processing of two Flt. S/C. c) Backup S/C not given dry run through facility.	Launch experience. Provides necessary time to solve unplanned problems with minimum risk to fixed launch period	No S/C Dry Run	Would improve schedule by Two months at an increase in Risk to Fixed Launch Period	Not Applicable
29. '71 Launch Period	Type I Trajectory used provides launch period from May 4, 1971 to June 23, 1971.	Type I Trajectory provides shortest trip time and communication distances. Launch period is best compromise of all constraints.	None Recommended	Not Applicable	Not Applicable
30. Mission Support	Will continue at GE, JPL, ETR, etc. thru last encounter.	JPL may desire continued availability of GE team	None Recommended	Not Applicable	Not Applicable

The schedule is based on a preliminary work breakdown structure which provides clearly defined and easily measurable work packages. Frequent milestone points are defined for each work package, thus permitting early detection and corrective action for any deviation from the schedule. Details on the preliminary work breakdown structure can be found in the Schedule and Cost Plan.

2.2 1969 FLIGHT TEST SPACECRAFT SYSTEM

Three primary factors were evaluated and analyzed in selecting the program for the 1969 flight test. (These are discussed in detail in Volume D - Appendix I, along with others such as cost and mission difficulties.)

a. An assessment of the specific engineering test objectives that could be satisfied in a deep-space shot versus a Mars fly-by.

b. The compatibility of the 1969 schedule to the 1971 schedule and program requirements.

c. The required availability of 1969 flight test results for best utilization in the 1971 development program.

The following conclusions were reached as the result of this analysis:

a. A deep-space flight can provide nearly all of the significant Spacecraft engineering test data (excluding data required for the Capsule) that can be obtained to maximize the success of the 1971 mission (more than 90 percent of the obtainable engineering data can be acquired from a deep-space shot).

b. A flight test launch in September of 1969 provides maximum compatibility with the 1971 development schedule and provides engineering flight test data at an optimum period in the program.

The JPL Voyager Phase IA work statement specifies that the objective of the flight test program is to achieve improved probability of 1971 mission success. It was reiterated at the May 21, 1965 JPL contractors' briefing that the flight test is intended to support only the Spacecraft requirements, and that it has no scientific mission. On this basis, General Electric recommends a deep-space test flight launched in September of 1969.

2.3 ALTERNATE FLIGHT TEST PROGRAMS

In the event that obtaining scientific data on Mars becomes an objective in 1969, a launch to meet the Mars window is entirely feasible. Figure III-2 presents a schedule that would permit such an objective to be achieved. However, this results in several significant implications in terms of the stated objective in the Phase IA work statement:

a. An expansion in the work scope of Phase IB would be desirable to provide for additional breadboard work and earlier release of selected long lead critical hardware; e.g., power amplifiers, tape recorders, propulsion, etc.

b. A number of required tasks would have to be overlapped and carried out more in parallel; e.g., development system test, type approval, and Flight Spacecraft and assembly and checkout. This overlap is considered entirely feasible although less desirable.

c. An earlier design freeze of the 1969 Flight Test Spacecraft would be required which is likely to result in less commonality between the 1969 and 1971 spacecraft.

The use of the Saturn IB/Centaur or the Saturn IB alone for the flight test program is clearly preferable to the Atlas/Centaur. Cost and availablility permitting, the use of these boosters would permit the design of the 1969 Flight Test Spacecraft and the 1971 Spacecraft to be essentially identical and thus maximize the return of useful engineering data. This would also permit a significant reduction in the required engineering design effort. A test flight to Mars using the Saturn IB/Centaur would also permit a launch date as late as the end of April, 1969, thus providing greater schedule compatibility with the 1971 program.

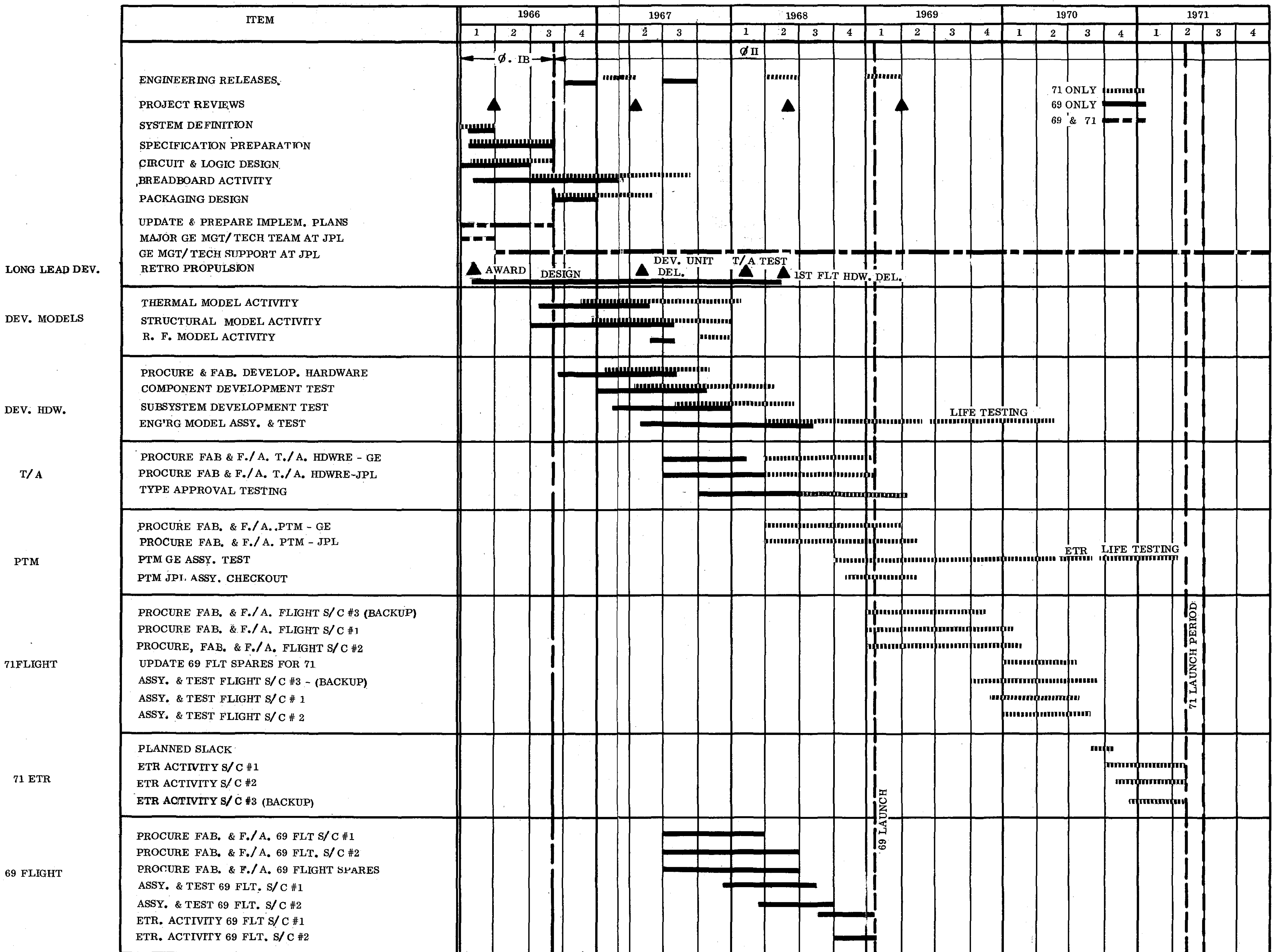

Figure III-2. Early 1969 Flight Schedule

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3.0 OVERALL IMPLEMENTATION PLAN

To emphasize significant features, schedule and implementation plans presented in the five volumes of the Final Technical Report are summarized below in a chronological manner. Work flow and integration and control activities, which are part of the individual management and project plans, are incorporated in the discussion. The purpose and relationship of the various plans and plan elements are illustrated in Figure III-3.

3.1 PHASE IB

General Electric will establish the Voyager Project Manager, his staff, and 50 to 60 systems and subsystems design, project, reliability, quality assurance, and manufacturing engineers and planning personnel in Pasadena at the start of the Phase IB. This team, including Motorola and Texas Instruments engineering and management personnel, will work in conjunction with the JPL Voyager Team to establish the mission definition and Spacecraft system design definition baseline. The Project Manager will expand the Pasadena team to an estimated 150 people as rapidly as the scope of the task becomes defined. The basic task is expected to consist of updating the Phase IA Functional Description documents and Implementation Plans submitted by General Electric, based upon JPL preference from their in-house Voyager study work, prior flight experience, and Phase IA evaluation. The establishment of this first baseline, which should conclude in a JPL/GE Project Review sign-off of the revised Functional Descriptions and Implementation Plans, is expected to require three to five months.

Other key actions to be accomplished during this period are:

a. A propulsion system and propulsion system supplier will be selected and a development contract initiated.

b. The Interface Integration Board will be established, hold its first meeting, establish Interface Control Working Groups, and begin preliminary definition of interface requirements.

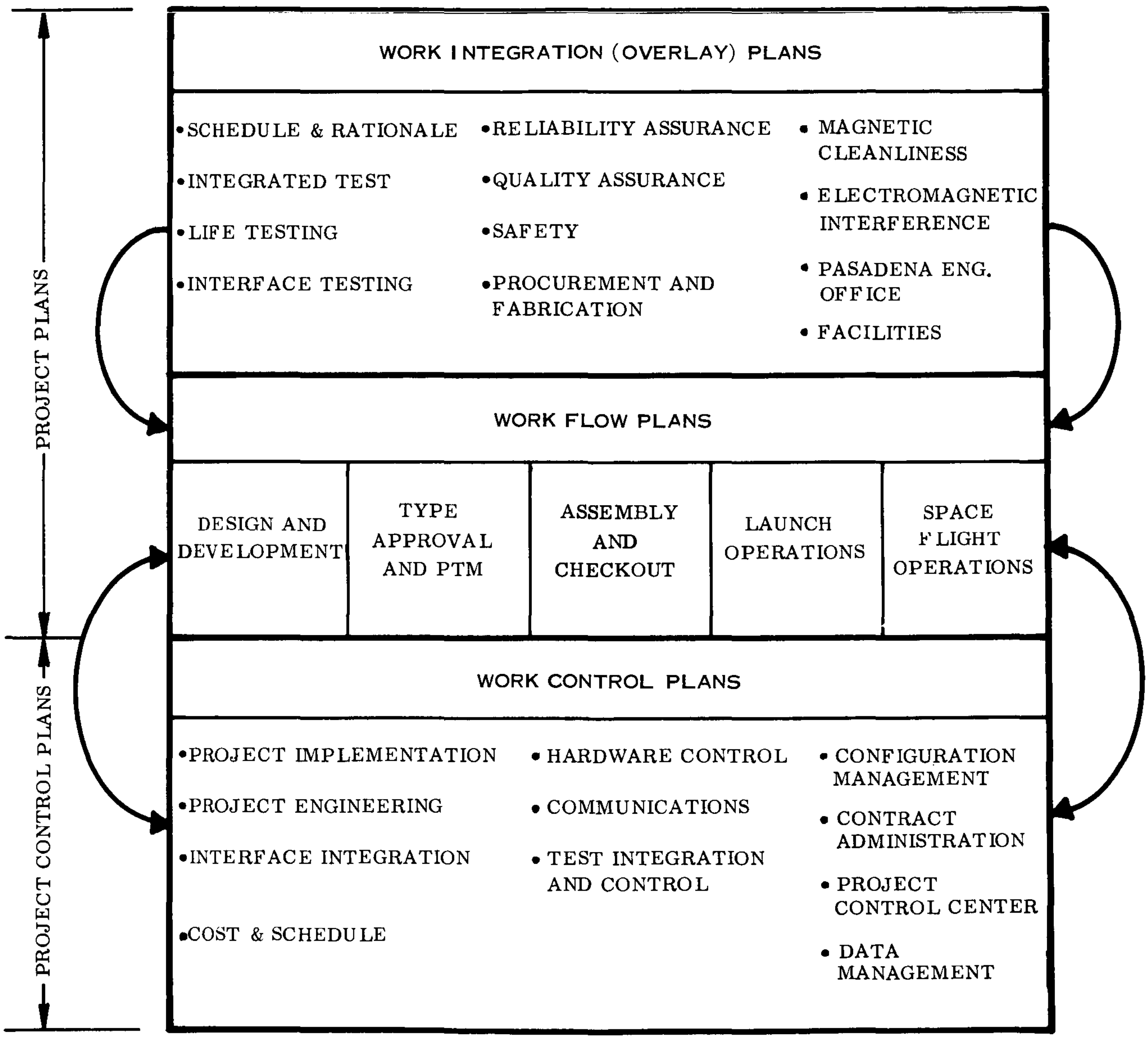

Figure III-3. Implementation Plan Relationships

Once the baseline is established, the major portions of the technical and management team will return to their home offices to continue system design. A basic team of 25-30 engineering, planning and support personnel to provide mission analysis, system design, and management planning support desired by JPL will remain in Pasadena, either on JPL premises, if desired, or at a nearby Pasadena Engineering Office.

The objective of the balance of work to be completed during Phase IB in support of JPL, both in Pasadena and at Valley Forge, is definition of the Functional Specification Baseline. This baseline will consist of Functional Specifications for the 1971 and 1969 spacecraft and OSE Systems, and a complete Schedule and Implementation Plan for Phase II, prepared by GE and approved by JPL before the end of Phase IB.

Parallel and supporting activities to be carried out during this latter part of Phase IB are:

a. Assignment of additional required personnel by name and assigned task according to the established Project manning plan as work becomes more definitive.

b. Testing of selected subsystem breadboards in support of subsystem definition.

c. Updating and JPL approval of the preliminary make-or-buy list, identification of major subcontractors, and establishment of a list of approved vendors for significant hardware items, based on performance records at JPL and GE.

d. Continued analysis in depth in areas which have significant potential influence on the design to further identify design requirements which must be factored-in early. Some of these include reliability; subassembly, subsystem and system test (Integrated Test Plan); quality inspection; producibility; and launch and space flight operations.

e. Expansion of the Project Control Center established in Phase IA to fully support project planning, progress measurement, analysis and reporting, status display, and configuration management activity.

f. Expansion of configuration management activity to have all elements in place, and ready or functioning. This includes:

   1. Software and hardware identification systems compatible with JPL and all major subcontractors

   2. Computer programs for the configuration identification and status system

   3. All remote centers

   4. Configuration Control Board

   5. Document control system

g. Early establishment of the Integrated Test Board to guide the revision and updating of the preliminary Integrated Test Plan in conjunction with JPL, with a goal of JPL approval by the end of Phase IB.

h. Initiation of formal configuration control activity to process and approve necessary changes to the first established baseline.

At the conclusion of Phase IB, a Project Review will be conducted with JPL chairing the activity. This review should include all associate contractor/agencies and have the following objectives:

a. Compare subsystem and system functional specifications with mission and Project requirements.

b. Clarify and ensure inclusion of all significant design criteria, characteristics, and restraints which influence design requirements.

c. Assure completion and adequacy of the necessary documentation, including specifications, and management plans such as Reliability, Procurement, Configuration Management, Project Control.

d. Establish a thorough, unified understanding and JPL endorsement of the next course of action.

3.2 PHASE II

The final management action of Phase IB should be the establishment of the JPL/GE approved Phase II statement of work, incentive provisions and contract which will complete the Functional Specification Baseline and mark the beginning of Phase II.

The approved Project Implementation Plan will be the governing document for overall implementation and control of the Project. Those plans not in effect will be activated to assure that planning, resources, and work are initiated when required.

System functional specifications for both the 1969 and 1971 spacecraft and OSE configurations will be converted to electrical design and package design, supported by rapid expansion of the breadboard test activity. Breadboards will be replaced with "three-dimensional" models and full scale mock-ups to conduct EMI and thermal dissipation tests, and harness and packaging evaluation. All design and development activity for both the 1969 and 1971 Spacecraft and OSE configurations will be controlled by the same Project functional and design engineers, with additional support provided for those activities peculiar to 1969. The design, fabrication, and test development cycles for the thermal, structural, and RF models will also commence at the start of Phase II.

Long-life reliability achievement will start with the design and development cycle, and continue throughout the total project. Commencing with engineering hardware, reliability standards constraints such as use of approved parts, materials, and processes, inclusion of RFM analysis, application of worst case design approach and derating factors, and use of established standard circuit approaches will be applied. Other development parameters, such as producibility, safety, subsystems and system test requirements, interface requirements and all other factors that have significant influence on the spacecraft and OSE design will continue to be analyzed in depth to assure total requirement consideration in the design.

Necessary changes to the Functional Specifications, which may develop from the hardware design activity or in preparation of detailed design specifications, will be processed by the Configuration Management Office. These will be submitted to JPL for approval as an extension of the configuration control activity (software) initiated in Phase IB.

During the design and development cycle, design reviews of the spacecraft and OSE, down to component levels, will be conducted by the Reliability Section. Functional organizations, technical consultants, and JPL, as desired, will participate in these reviews to assure soundness of the design, conformance with the work integration (overlay) plans, and to preclude downstream problems in procurement, producibility and test.

Conclusions and recommended actions will be published as minutes of each meeting held, and distributed to JPL, the GE Project Manager, his staff, and Project Control Center. The recommended actions will become part of the Project Action Item List requiring follow-up and must be answered by the responsible design engineer.

Review by the Project Engineer, Reliability Engineer, and JPL provides the "check and balance" for obtaining answers which meet the overall Project requirements.

Management reviews at all levels of project activity to assess technical integrity and progress against plans will be continuous throughout Phase II. Review activity begins with working-level Project integration meetings conducted by Project Control on a day-to-day basis.

Overall Project status review meetings will be conducted weekly by the Project Manager. The project staff will present and review all aspects of technical, schedule, and cost progress, and develop action items. The senior JPL resident will be invited to attend these meetings. Concentrated technical reviews of critical designs, overall subsystems, and the overall system will be held as extensions of these meetings. Top Division and Company technical specialists will be called to attend and review special areas of interest. This provides a mechanism for the Project Manager, in addition to his day-to-day informal discussions with his engineering staff, to detect a developing technical problem. It is then the Project Manager's responsibility, with advice from his engineering managers, to identify, secure, and apply the many special capabilities within the Division and Company. In addition to its own extensive internal resources, the Missile and Space Division is currently the largest internal "customer" of such Company Laboratories as the General Electric Research Laboratory, Advanced Technologies Laboratory, and the Electronics Laboratory. Some of these and several other Department special capabilities, as shown on the list of acknowledgements, contributed to this study effort.

At the General Electric corporate level, the Vice President and General Manager of the Missile and Space Division and his staff will review Project performance against JPL Voyager requirements and plan on a semi-monthly or monthly basis. Significant problems and actions and any additional assistance needed will be considered and appropriate action taken.

Scheduled Technical Direction (T/D) meetings with JPL are recommended to provide top-level, total-project review and guidance on a monthly basis. Documented agreements and action items placed on General Electric, other system contractors/agencies, and JPL, will be the primary output of these meetings.

At the appropriate scheduled point in the 1969/1971 design cycle when development hardware (other than long lead items) is ready for release to procurement, a major Project Review will be conducted, chaired by JPL. All aspects of the design, including mission requirements, external interfaces, producibility considerations, and breadboard test results will be reviewed. Also to be reviewed is the implementation status of such plans such as Development Test, Type Approval and Proof Test Model Test, Assembly and Checkout, Launch and Space Flight Operations, and the effectiveness of the Quality and Reliability Assurance efforts. The desired result is the JPL approval and acceptance of the development hardware design (Stage 3 release) and agreement to proceed with procurement.

Intensive management attention will be given to the procurement activity, following a number of guidelines established in the procurement plan including:

a. Adherence to JPL-approved make-or-buy list.

b. Reliability and Quality Assurance requirements in the General Electric contract imposed to the maximum practical extent down through all subcontractor/vendor levels having design responsibility, and second tier as a minimum for fabrication.

c. A subcontract manager established for each significant subcontract who will be responsible for direction, monitoring and review.

d. Subcontractor T/D meetings at GE monthly, and major subcontractors included in JPL T/D meetings.

Engineering hardware will be procured and fabricated for subassembly and subsystem testing. This will consist of hardware common to both 1969 and 1971 spacecraft and OSE configurations, and hardware peculiar to each. Engineering will conduct and evaluate these tests.

Two development system models will be provided - one for the 1969 configuration and one for the 1971 configuration - primarily because of the overlap in the system development test requirements, and the need for an engineering system to support the 1969 assembly, checkout, launch, and flight anomaly evaluation activity. System Test and Field Operations will conduct this and all system tests.

The 1971 structural model, and engineering component and subsystem test hardware will be used to the maximum possible extent in the assembly of the 1971 development system.

The Integrated Test plan that will be followed includes subassembly, subsystem and system hardware evaluation under ambient and full environmental (type-approval simulated) conditions.

During development system testing, interface tests between booster, lander, and science instrumentation development models and the spacecraft bus will be performed to verify mechanical fit, electrical and EMI compatibility, etc.

Demonstration of long-life reliability will be heavily emphasized throughout the Project by life testing, starting with breadboards and progressing to all assembly levels of development hardware to begin the critical assessment of life capability. Use of the dynamic mission equivalent (DME) approach in the test procedure is planned.

Failures encountered will be reviewed by the Failure Analysis Review Board and corrective action follow-up provided. Design action is the responsibility of Engineering. Implementation of the design action is planned, directed and integrated by the Project Engineer responsible. In addition, Reliability Assurance will incorporate the life implications of the failure data into its continuous overall assessment of the system life capability. All design changes will be formally documented by Engineering, reviewed by the Configuration Control Board, and given complete distribution.

The Configuration Control Center, presently linked with major subcontractors, will be extended to include JPL and the Pasadena Engineering Office during Phase IB. This link is by desk-side equipment which provides remote updating and interrogation capability to keep the total spacecraft system configuration information in the GE data bank current.

JPL will be kept informed on a daily basis of progress and significant events. Rapid communications will be assured by telephone contact between GE and JPL Project Managers and at all other levels of the organization; in-house JPL representatives; the Pasadena Engineering Office, which will provide close liaison service; and TWX and Data Fax equipment. Weekly reports of schedule and manpower status (progress summary and significant variance information only), as well as monthly detailed progress reports and Quarterly Summary Reports of technical performance, cost and schedule progress against plan, with significant actions highlighted, will be submitted. The Project Control Center provides visual display of status, current action items, and plans, for the overall project activity.

Achieving Type Approval quality before the start of fabrication of the T/A hardware is an objective of the development phase of the Project. Progressive formality and rigidity in inspection and test procedures to identify and correct, in each succeeding subassembly, all less-than-flight-quality conditions in processes, materials, parts, and workmanship will parallel the hardware development effort. This formality and rigidity will be invoked to assure production of flight quality hardware during the T/A procurement and fabrication cycle, both at GE and at all subcontractors.

Prior to the release of T/A hardware for procurement, a major Project Review by JPL will be conducted with the same objectives as those stated for Stage 3 release. With the release of T/A and PTM hardware, full configuration control procedure will be invoked, and all changes to this Qualification Baseline, both hardware and software, must be approved by the Chairman of the Configuration Control Board. JPL and all Project functions are represented on this full-time Board. The Board Chairman, who reports to the Configuration Management Office in Project Control, has authority to approve all changes that fall within contract scope and current Baseline definition. A summary of all significant Board action will be transmitted to JPL, and reviewed with the Project staff in the weekly Project Review Meetings. All out-of-contract-scope changes will be referred to the managers of Project Control and Business Management to determine with JPL the need for an Engineering Change Proposal.

Two sets of T/A hardware and appropriate spares will be procured for both the 1969 and the 1971 program. Type Approval testing will be performed on one set of hardware at GE; JPL will be provided with the second set with the objective of an integrated but independent assessment.

A Proof Test Model is not considered justifiable for the 1969 test flight. Instead, earlier introduction of the first of two Flight Spacecraft is planned, with extended system environmental testing on this first Spacecraft. This Spacecraft would then be used for a Kennedy Spaceflight Center (KSC) walk-through, and become the back-up flight unit. Subsequent to 1969 launch activities, the OSE will be returned to GE for updating to the 1971 configuration to the fullest extent practical, as shown in Figure 4-1, Volume A, Section V.

Two PTM units will be provided for the 1971 mission, one to be tested at GE and one to be delivered to JPL. General Electric will provide technical support and equipment as required by JPL to maintain the Spacecraft and OSE in a current configuration. Upon completion and JPL approval of the GE PTM test, the Spacecraft will be committed to extended life test using the DME concept until it is required at KSC for a walk-through. The JPL PTM is proposed as the interface test unit for the DSN at Goldstone after test at Pasadena.

Prior to releasing all flight hardware for procurement, a Project Review by JPL will be conducted with the same objectives as those stated for the Stage 3 and T/A releases. Three sets of flight hardware plus spares for 1971 will be procured at the designated points in the schedule.

The System Test teams that conducted the development system model test and the 1971 PTM test, augmented by engineering personnel from other functional areas, will be assigned to individual teams with a designated senior test director for each flight spacecraft. Each team will be responsible for the assembly and flight acceptance testing of their assigned Spacecraft, and will process it from initiation in-house through launch.

At the completion of flight acceptance testing for each spacecraft, a thorough review of all documentation associated with the fabrication and test of that Spacecraft and attendant OSE will be conducted by the Integrated Test Board (ITB). This review is to assure that documentation is available and in order, Spacecraft configuration is in accordance with that documentation, manufacturing deficiencies have been corrected, failures have been analyzed and corrected, testing has been conducted in accordance with approved testing requirements, and that test results have been recorded and verified. Prior to delivery of the three 1971 flight spacecraft to KSC, a buy-off will be conducted by JPL and GE Project management for the purpose of evaluating the results of the ITB activities, and to approve the Spacecraft for shipment to KSC.

At the launch site, General Electric will support JPL in the overall launch preparation process. A technical team will precede the spacecraft to KSC to support JPL in facility preparation and readiness effort. The Spacecraft Test Teams will be responsible for all Spacecraft processing activity in support of JPL.

A team of systems and design engineers will be assigned to JPL in support of the space flight operations preparation, the flight activity, and post flight analyses as desired.

4.0 PROJECT ORGANIZATION

4.1 INTRODUCTION

Further study of Voyager requirements during Phase IA led to consolidation of the functions reporting to the GE Voyager Spacecraft Project Manager, and provides a more manageable span of control. This structure, shown in Figure III-4 for Phase IB and II, provides a more homogenous grouping of activities and responsibilities, and has been designed to interface as closely as possible with the JPL organization.

4.2 OPERATIONAL CONCEPTS

Principal concepts which will govern the overall management and operation of the project organization are:

a. JPL will be integrated into the project functions to the maximum extent desired - membership on review and control boards, invitation to attend meetings, office space in the Project area, open access to all office areas and the Project Control room.

b. The entire project must be responsive and readily adaptable to direction. Greater knowledge, increased capability, and better understanding contribute to a more successful program.

c. Management plans and controls are considered tools which: (1) provide guidelines and boundaries within which mature individuals can exercise maximum use of their experience and ingenuity in performing assigned responsibilities, and (2) provide the means for a common understanding between JPL and GE from upper management through all levels of both operations.

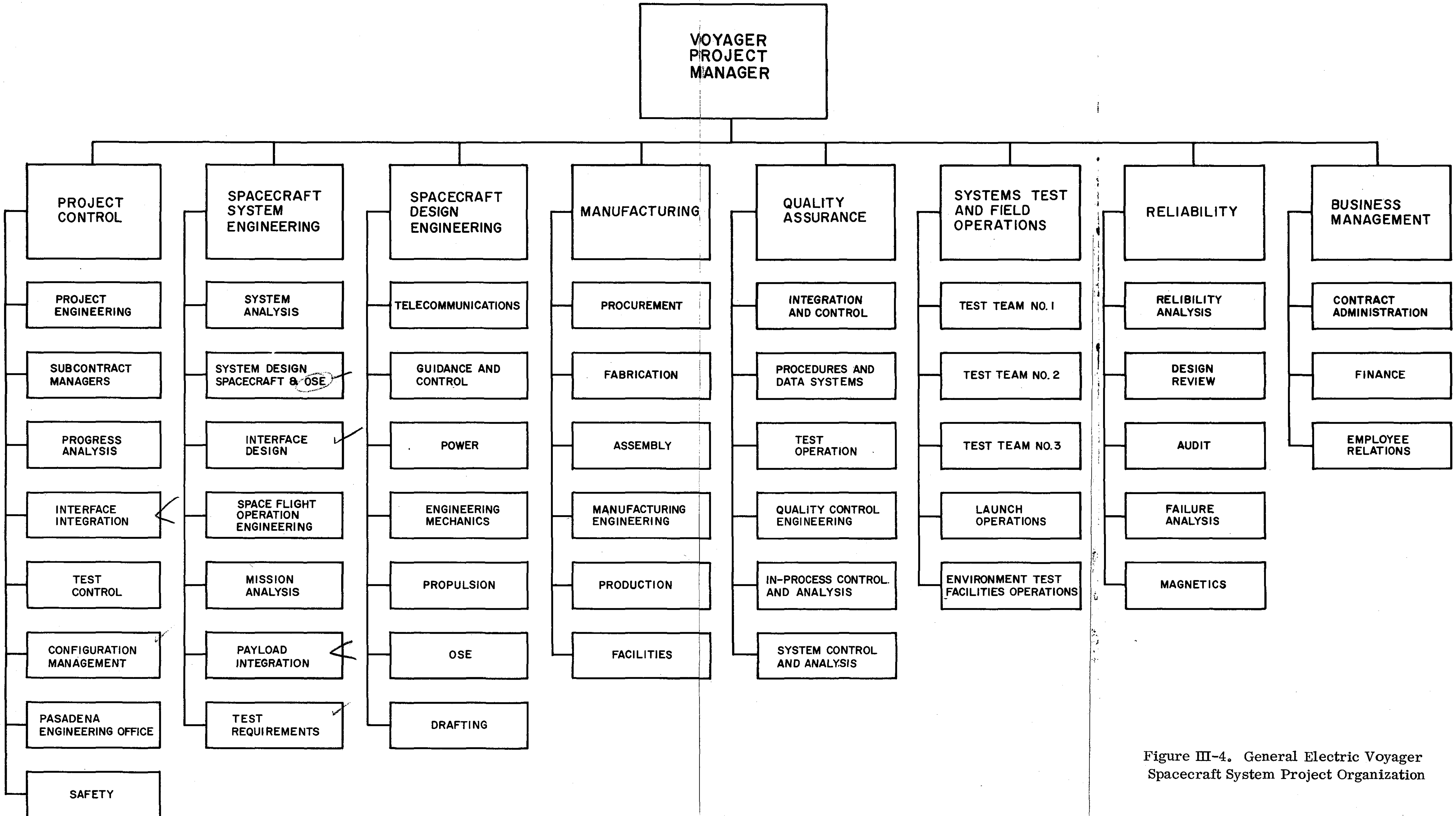

Figure III-4. General Electric Voyager Spacecraft System Project Organization

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(3)

4.3 STRUCTURE AND LOCATION

The Voyager Project organization is vertical with all personnel performing full time on the Project reporting administratively and functionally to the Project Manager. All Project personnel are, and will be, located in the same office area adjacent to the assembly and test area to provide minimum communication-line distances between personnel while promoting maximum total responsiveness to Voyager requirements and Project Manager direction.

4.4 ORGANIZATION DESCRIPTION

The following paragraphs briefly describe the function of the Project Manager and each group reporting to him, together with the key responsibilities and relationships which are considered of most importance.

4.4.1 PROJECT MANAGER

The Project Manager is responsible for successful fulfillment of all Voyager Spacecraft Project objectives and contractual requirements and has no other responsiblities. He will be assigned, by the General Electric Company, the authority, personnel and facilities required to fulfill the following responsibilities:

a. Meeting General Electric's commitments to JPL on the Voyager Spacecraft Program.

b. Identification and establishment of the required resources including personnel and facilities needed to meet the Project requirements.

c. Preparation and implementation of the required program to fulfill technical, schedule and cost commitments.

d. Communication of Project progress against plan, key problem areas, and assistance required to JPL management and the General Electric Executive Office.

4.4.2 PROJECT CONTROL

In a project of the magnitude of the Voyager Spacecraft System, with its multitude of complex interfaces, program measurement and control becomes an extremely important consideration. For this reason, a Project Control Section has been established reporting directly to the Voyager Project Manager.

To provide overall task management, Project Engineers will be assigned responsibility for major "work package" tasks with the authority to carry out project direction, task planning, activity integration and schedule and cost management. In this regard, their responsibility for assigned tasks is similar to that of the Project Manager for the overall project. The operation of the project control section is geared to support the Project Engineer with progress information, current status of work, cost versus plan, focus of developing trouble spots, subcontract progress, and hardware status and assignment. The primary point of contact for the JPL Cognizant Engineers will be the Project Engineers. However, the Project Engineers are also responsible for assuring that the JPL Cognizant Engineers have ready access to the design and systems engineers as well as other specialists for detailed discussions.

Major elements of the Spacecraft System will be subcontracted; therefore, effective management and control of subcontractors is essential to the success of the program. All subcontracts will be managed by Project Control. A subcontract manager will be assigned responsibility for each major subcontract and will have a team representing concerned sections (Procurement for contract administration, Engineering, Quality Assurance, Reliability, Legal, etc.) for a particular subcontract. His responsibility includes integration of all related management functions - vendor selection, work statements and specifications, negotiation, progress review, technical direction, design review, and hardware acceptance.

The Project Control Operation is responsible for assuring the implementation of the interface requirements established by the systems engineering section in conjunction with JPL. The Interface Integration Engineer in project control will serve as the GE Chairman of the Science Interface Control Working Group in support of JPL. In addition, the Interface

Integration Group will provide personnel to serve on the Interface Control Working Groups chaired by JPL, e.g., launch vehicle, capsule and the DSN.

The Integrated Test Board, chaired by the Test Control Engineer of Project Control, will assure that the Integrated Plan is prepared, reviewed and approved by JPL, and will monitor, review, and approve the ITP and all changes to it. The Integrated Test Board is made up of representatives of all concerned Project functions - Engineering, Reliability, Quality Assurance, Project Control, System Test and Field Operations, Safety and JPL if desired.

Effective configuration and data management is considered an essential requirement in the Voyager Spacecraft Project. The Project Control Section has responsibility for this activity which includes: establishment and operation of the data bank for configuration identification (the hardware-software numbering system, the computer programs, the remote update centers, and the communication link with all elements - JPL, subcontractors, etc.); chairmanship of the Change Control Board which reviews and approves all hardware and software changes after formal review procedure is invoked; and provision of the single source of parts lists used for procurement and assembly of hardware.

A Pasadena Engineering Office, reporting to the Project Control Manager, will be established at/or near JPL to assist and support JPL in all Project communications. General Electric will locate key members of its engineering and management team at this office, during Phase IB, in order to facilitate supporting JPL in the preparation of the Project specifications and implementations plans.

4.4.3 SPACECRAFT SYSTEMS ENGINEERING

Systems Engineering will be responsible for definition and establishment of the Spacecraft and OSE systems design in compliance with JPL technical direction. A well integrated systems design is achieved through the establishment of a competent centralized systems group with the authority to define and integrate the requirements for the system design. This includes supporting JPL in performing mission analysis, defining the spacecraft and OSE system concept, and performing operational systems analysis for space flight operations. Support is also provided to JPL in defining the interfaces: science, lander, launch vehicle and DSN, integrating the science interface, and supporting the JPL integration effort in the other interface areas.

A key responsibility of Systems Engineering is to apportion reliability to the subsystem and component level, working in conjunction with Design Engineering, in order to assure optimum allocation of risks across the total Spacecraft system.

To assure a Spacecraft OSE system design concept which is consistent with the space vehicle system as well as with the capsule, DSN and other external interfaces, Spacecraft Systems Engineering has the responsibility to define the OSE system specifications.

To assure that both system views and hardware views are taken by highly qualified homogeneous groups, and that conflicts between system requirements and equipment performance are directly visible to the Project Manager, Systems Engineering and Design Engineering have been organizationally separated.

4.4.4 SPACECRAFT DESIGN ENGINEERING

The Spacecraft Design Engineering section has the responsibility for the design of the Spacecraft hardware for each subsystem in compliance with the requirements established by Systems Engineering. The design engineer is responsible for this activity from project inception to launch. Design responsibility for all OSE hardware which interfaces (electrical, mechanical, thermal, RF) with Spacecraft equipment is organizationally centralized within an OSE Design group in the Design Engineering Section. OSE design engineers for individual OSE subsystem elements will be physically located with their Spacecraft subsystem design counterpart for maximum integration. To assure subsystem and system compatibility, OSE tasks placed on other internal Project Sections, such as Quality Assurance Engineering for STE, or on outside subcontractors, will be accompanied by design specifications prepared by OSE Design which prescribe specific design approaches, standards, functional and physical interface characteristics, etc.

4.4.5 MANUFACTURING

The Manufacturing Section has direct responsibility for providing all Voyager Spacecraft and OSE hardware. The Procurement Operation prepares and implements the "Make-or-Buy" plan which is approved by the Voyager Project Manger. The overall management and direction of subcontractors rests with the Project Control Section and the Manufacturing Procurement Operation supports these activities by administering all contractual matters.

Other Manufacturing functions provide all necessary resources for the in-house hardware fabrication, starting with raw material receiving through storage in bonded stock of completed subassemblies. They also assign to the system test teams, which are under the direction of System Test and Field Operations, the necessary technicians to assemble each spacecraft. Manufacturing will be represented on the Design Review Board, the Reliability Board, the Change Control Board, and the Material Review Board.

The establishment of Manufacturing facilities requirements, and the implementation of these facilities, is the responsibility of the Manufacturing Section. This section will also implement the facilities requirements established by other sections such as Systems Engineering, Systems Test and Field Operations, and Quality Assurance.

4.4.6 QUALITY ASSURANCE

To assure the conformance to design specifications for all flight hardware, spares and OSE, a Quality Assurance Section will report directly to the Voyager Project Manager. This function has been established as an independent group because it will not only provide the measurements required to establish conformance to specifications, but will also establish the quality requirements that must be met if reliable long-life hardware is to result. Quality Assurance has the authority to reject hardware if requirements and specifications are not being met. They are responsible for the conduct of type approval and acceptance testing up to the subassembly level. Key tasks to be performed by this section include:

a. Integration of quality considerations during the design and development phase.

b. Implementation of the vendor control plan.

c. Configuration control and traceability to the parts level.

d. Conduct of in-line quality measurements and evaluations during the procurement and manufacturing cycle.

e. Failure analysis, reporting, corrective action, and follow-up, including chairmanship of the Failure Analysis Review Board.

4.4.7 SYSTEM TEST AND FIELD OPERATIONS

Providing technical competence and continuity of experience for the conduct of all system level testing, from the initial in-house development tests to launch, were key requirements in determining the organization structure. To meet these requirements, an independent System Test and Field Operations section reporting to the Project Manager was established. Its responsibility includes planning, direction, and evaluation of all system tests, e.g., engineering system model tests, environmental model tests, proof test model tests, system interface tests, and Flight Spacecraft acceptance tests and launch preparation.

An assembly and test team, headed by a senior test director reporting to the System Test and Field Operations Manager, will be assigned to each Spacecraft. These basic teams will be augmented by systems, design, project, quality assurance and manufacturing engineers to utilize important knowledge and experience available and to provide the capability to expand and retract efficiently with test requirements.

The team is thoroughly familiar with the Voyager Spacecraft when it formally starts its activity at the beginning of the assembly phase. It provides technical direction and conducts subsystem tests as the assembly progresses, in order to provide the important continuity between the assembly experience and the actual system test. Three complete teams and test leaders are required to process three Flight Spacecraft essentially in parallel. A pool of Voyager experienced personnel can be drawn upon to supplement these teams if extended coverage is required. Each team remains together when formed and proceeds with the assigned Spacecraft from start of system assembly through launch operations. The assembly and test team concept will also be applied to the Engineering Test Model and the PTM Spacecraft.

4.4.8 RELIABILITY

Because it is the most critical key problem in the entire Voyager Spacecraft program, the reliability function has been set up reporting directly to the Voyager Project Manager. The section will be responsible for preparation, overall implementation and direction of the Voyager Reliability Program. Key elements of its activities include:

a. Reliability analyses, studies and investigations during all phases of the program, from initial hardware specifications through flight operations. This includes the establishment of reliability objectives, figure-of-merit analyses, parts/materials/processes and standards definition.

b. Chairmanship of the Design Review Board which will be responsible for organizing, conducting and reporting on technical design reviews.

c. The audit of all activities of the program to assure that all procedures, practices and activities are compatible with long-life reliability.

d. Granting qualification status to TA and PTM hardware.

e. Operation of the Risk Appraisal of Programs System (RAPS) if further experience in its use on other programs supports its effectiveness (see Volume A, Section V).

4.4.9 BUSINESS MANAGEMENT

The primary responsibility of Business Management will be the administration of all matters pertaining to the contract. In this regard, one of the most important aspects in a program of the magnitude of Voyager is the maintenance of technical flexibility and, at the same time, complete compliance with contractual provisions. Business Management will interface closely with JPL Procurement and with the GE Voyager Project Control Section to assure that contractual paper work keeps pace with the work activity. The establishment of well-defined work statement and effective and workable incentive provisions will be another key responsibility of this section.

4.5 MANNING PLANS

The in-place Phase IA team will be augmented during expansion in Phase IB by transfer of experienced personnel from other areas of the Missile and Space Division to the Voyager Spacecraft Project. Planned phase-out of work on existing Missile and Space Division programs will provide the necessary personnel to accomplish all Phase IB tasks. The Division's total employment of 17,000, of which 4,000 are engineers, will provide a pool of experienced personnel for the expansion required during Phase II of the Project. Design experience and implementation planning capability can be maintained during this growth by use of these personnel who have been working on programs with similar requirements to Voyager.

4.6 RELATIONSHIP OF IMPLEMENTATION PLANS TO ORGANIZATION

Figure III-5 shows the relationship of the Voyager implementation plans to the organization. This chart indicates the organizational element responsible for management (seeing that the plan is prepared and implemented) and approval of each plan.

MANAGEMENT PLAN	C I I NO.	MANAGEMENT (PREPARATION & IMPLEMENTATION) AND APPROVAL RESPONSIBILITY								
		PROJECT MANAGER	SPACECRAFT SYSTEM ENGINEERING	SPACECRAFT DESIGN ENGINEERING	MANU-FACTURING	SYSTEM TEST & FIELD OPERATION	PROJECT CONTROL	QUALITY ASSURANCE	RELIA-BILITY	BUSINESS MANAGE-MENT
WORK INTEGRATION (OVERLAY) PLANS										
SCHEDULE AND RATIONALE	VB110VP001		A	A	A	A	M	A		A
INTEGRATED TEST	VB110VP002		A			M	A		A	
LIFE TESTING	VB110VP008		A	A		A	A	A	M	
INTERFACE TESTING	VB110VP009		A	A		M	A		A	
RELIABILITY ASSURANCE	VB110VP010		A				A		M	
QUALITY ASSURANCE	VB110VP011						A	M	A	
SAFETY	VB110VP012				A	A	M	A	A	
PROCUREMENT & FABRICATION	VB110VP014				M		A, M (sub-cont)	A	A	
MAGNETIC CLEANLINESS	VB110VP015			M			A	A	A	
ELECTROMAGNETIC INTERFERENCE	VB110VP016	ALL PLANS		M		A	A	A	A	
PASADENA ENGINEERING OFFICE	VB110VP017						M			A
FACILITIES	VB110VP018		A		M	A	A	A		
WORK FLOW PLANS										
DESIGN AND DEVELOPMENT	VB110VP003		M	A			A		A	
T/A AND PTM	VB110VP004		A	A		M(PTM)	A	M(T/A)	A	
ASSEMBLY AND CHECKOUT	VB110VP005				A	M	A	A	A	
LAUNCH OPERATIONS	VB110VP006		A		A	M	A	A	A	
SPACE FLIGHT OPERATIONS	VB110VP007		M			A	A		A	
WORK CONTROL PLANS										
PROJECT IMPLEMENTATION	VB120VP012	A					M			
PROJECT ENGINEERING	VB120VP001						M		A	
INTERFACE INTEGRATION	VB120VP002		A			A	M			
COST AND SCHEDULE	VB120VP003		A	A	A	A	M	A		A
HARDWARE CONTROL	VB120VP004	APPROVE			A		M			
COMMUNICATION	VB120VP005						M			A
TEST INTEGRATION AND CONTROL	VB120VP006		A			A	M		A	
CONFIGURATION MANAGEMENT	VB120VP007		A	A	A	A	M	A	A	A
CONTRACT ADMINISTRATION	VB120VP008						A			M
PROJECT CONTROL CENTER	VB120VP009						M			
DATA MANAGEMENT	VB120VP010						M			A

M - MANAGEMENT
A - APPROVAL

Figure III-5. Plan Implementation Matrix

APPENDIX

APPENDIX I

LIST OF DOCUMENTS CONSTITUTING THE GENERAL ELECTRIC VOYAGER PHASE IA FINAL REPORT

VOLUME A

VOLUME A

Book 2 of 4

APPENDIX I (CONT)

VOLUME A

Book 3 of 4

APPENDIX I (Cont'd)

VOLUME A

(Book 4A and 4B of 4)

VOLUME A

(Book 4A and 4B of 4)

APPENDIX 1 (CONT)

VOLUME B

(Book 1 of 3)

SYSTEM ALTERNATES

VOLUME B

(Book 2 of 3)

TELECOMMUNICATIONS

Section CII NO.

IV ALTERNATE SUBSYSTEM MECHANIZATIONS CONSIDERED (TELECOMMUNICATIONS)

VOLUME B

(Book 3 of 3)

G&C, C&S, PWR, MECH, PROP, PLANS

APPENDIX I (CONT)

VOLUME B - APPENDICES (SECTION IV)

Appendix	I	Propulsion (Technical) - Liquid Systems Studies, Rocketdyne
Appendix	II	Propulsion (Technical) - Spherical Solid Propellant Rocket Motor (Plus Classified Supplement), Thiokol
Appendix	III	Propulsion (Technical) - Technical Data and Program Plan, Aerojet Liquid
Appendix	IV	Propulsion (Technical - Solid Rocket Retromotor (Plus Classified Supplement), Aerojet Solid
Appendix	V	Propulsion (Technical) - Hydrazine Monopropellant Midcourse Correction System, TRW Systems
Appendix	VI	Propulsion (Technical) - Voyager '71 Monopropellant Hydrazine Midcourse Propulsion System, Rocket Research
Appendix	VII	Propulsion (Technical) - Solid Retro-Propulsion System for Voyager (Plus Classified Supplement), Lockheed

VOLUME B - APPENDICES (SECTION V)

Appendix	I	Propulsion (Implementation Plan) - Rocketdyne
Appendix	II	Propulsion (Implementation Plan) - Spherical Solid Propellant Rocket Motor, Thiokol
Appendix	III	Propulsion (Implementation Plan) - Technical Data and Program Plan, Aerojet Liquid
Appendix	IV	Propulsion (Implementation Plan) - Solid Rocket Retromotor, Aerojet Solid
Appendix	V	Propulsion (Implementation Plan) - Hydrazine Monopropellant Midcourse Correction System, TRW Systems
Appendix	VI	Propulsion (Implementation Plan) - Voyager '71 Monopropellant Hydrazine Midcourse Propulsion System, Rocket Research
Appendix	VII	Propulsion (Implementation Plan) - Solid Retro-Propulsion System for Voyager, Lockheed

VOLUME C

1971 VOYAGER SPACECRAFT OPERATIONAL SUPPORT EQUIPMENT

APPENDIX I (CONT)

VOLUME D

PREFERRED DESIGN FOR THE 1969 SPACECRAFT

VOLUME D (CONT)

APPENDIX I (CONT)

VOLUME E

1969 VOYAGER SPACECRAFT OPERATIONAL SUPPORT EQUIPMENT

VOLUME F

REPORT SUMMARY