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VOYAGER DESIGN STUDY

VOLUME I

DESIGN SUMMARY

Prepared Under Contract NAS W-696

for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

OFFICE OF SPACE SCIENCES

WASHINGTON, D.C.

[redacted]

GENERAL ELECTRIC

MISSILE AND SPACE DIVISION

Valley Forge Space Technology Center

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VOLUME SUMMARY

The Voyager Design Study report is contained in six volumes, an appendix, and subcontractor reports. The volume numbers and their titles are as follows:

Volume No.

I — Voyager Design Summary

II — Mission and System Analyses

1. Mission Analysis
2. Parametric System Performance
3. Voyager Systems
4. Reliability

III — Subsystem Design

1. Communications
2. Television
3. Radar
4. Guidance and Control
5. Propulsion
6. Power Supply

Appendix (Classified)

IV — System Design

1. Entry/Lander
2. Orbiter

V — Sterilization

VI — Program Development Plans

Separate Reports from the following Companies are also included:

Aerojet-General Corp.
Barnes Engineering
Bell Aerosystems Co.
Conductron Corp.
Electro-Mechanical Research Inc.
General Electric Co.
  Light Military Electronics Dept.
General Precision Inc.
Hazeltine

North American Aviation Inc.
  Autonetics Division
  Rocketdyne Division
Radio Corporation of America
Rocket Research Corp.
Texas Instruments Corp.
Thiokol Chemical Corp.
  Elkton Division
  Reaction Motors Division

[redacted]

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FOREWORD

Early in the Voyager study General Electric recognized that the size and complexity of the Voyager Program would inevitably require the participation of a broad segment of industry in the design, development, and manufacture of major components and subsystems. To permit the GE system study to take advantage of the capabilities of industry and to ensure that the GE conceptual design incorporated the best ideas and designs available, a series of briefings were held during the first half of 1963 for a number of companies able to contribute to the program. Agreements were negotiated with some of these companies under which General Electric provided information and technical data on the overall Voyager system, and the individual companies then developed their own ideas for the design of the component or subsystem of interest to them. All of this associated work was on a completely unfunded, non-exclusive basis.

Progress meetings were held during the course of the studies to review the subsystem design and to update the system data on which the designs were based. Each of the separate subsystem studies submitted is based on a system design which is close to the final Voyager System recommended by GE, although no attempt was made to accommodate all of the system revisions as they were made. The Missile and Space Division of General Electric also conducted its own studies in all subsystem areas and was not dependent on the work of any other company for the successful completion of its Voyager study. However, the work done by the associated companies has proved to be of much value and in some instances is reflected in the GE final report, appropriately attributed to the proper company.

The work reported shows a very strong interest by industry in the Voyager Program and is tangible evidence of the detailed design available to substantiate the Voyager system recommendations.

The following companies have submitted reports, as shown, on their non-exclusive unfunded studies to General Electric, and five copies of each plus a reproducible has been submitted to NASA as a part of the General Electric Co. final report on its Voyager Study:

- Aerojet-General Corporation, "Voyager Orbiter Propulsion", "Voyager Propulsion Analysis (Lander Portion)"
- Barnes Engineering Company, "Approach Guidance Subsystem"
- Bell Aerosystems Company, "Voyager Orbiter Propulsion"
- Conductron Corporation, "Voyager Radar Subsystem Experiments"
- Electro-Mechanical Research, Inc., "Digital Television Subsystem for Project Voyager"
- General Electric Company, Light Military Electronics Department, "Voyager Radargrammetry"
- General Electric Company, Light Military Electronics Department, "Voyager Guidance and Control"
- General Precision Incorporated, "Study Report of Approach Guidance System for the Voyager Spacecraft"
- Hazeltine Corporation, "Engineering Study Report on TV Camera Subsystem for Mars-Venus Voyager Missions"
- North American Aviation Inc., Autonetics Division, "Autonetics Studies of the Voyager Mission"
- North American Aviation Inc., Rocketdyne Division, "Voyager Orbiter Propulsion"

- Radio Corporation of America, "Report on: S-Band and VHF Design Considerations for Project Voyager"
- Rocket Research Corporation, "Voyager Subliming Solid Control Rocket"
- Texas Instruments Incorporated, "Proposal for Voyager Telecommunications and Data Handling Equipment"
- Thiokol Chemical Corporation, Elkton Division, "Proposal Study for Voyager Spacecraft Orbital Adjustment and Lander Insertion Motors"
- Thiokol Chemical Corporation, Reaction Motors Division, "Voyager Orbiter and Lander Propulsion"

SECTION 1.0 INTRODUCTION

Presented in this report are the results obtained from a conceptual design study for a Voyager spacecraft to perform Orbiter/Lander missions to Mars and Venus during opportunities from 1967 through 1975. To place the study in proper perspective, the following paragraphs are extracted from the NASA Request for Proposal No. 10-929, Voyager Design Studies, dated 5 March 1963:

> "Scientific exploration of Mars and Venus is a major objective of NASA. Preliminary exploration of these planets is being accomplished by the Mariner spacecraft. Initially, Mariners sized for the Atlas-Agena launch vehicle are being used for fly-by missions, and will be followed by Mariner B spacecraft sized for the Atlas-Centaur launch vehicles. Entry capsules for Mars and Venus are planned for the Mariner B spacecraft."
>
> "In recognition of the need for larger unmanned spacecraft, studies are underway for Mars and Venus spacecraft sized for Saturn class launch vehicles. It is anticipated that the increased weight lifting capability of Saturn vehicles will be used in the planetary program for: (a) more difficult planetary missions, (b) more scientific instrumentation, (c) greater data collection and transmission rates, and (d) longer lifetime operation than possible in the Mariner class spacecraft. In order to obtain the scientific data required of Mars and Venus, Voyager will be designed for orbiter and lander missions."

The scientific objectives for the Voyager spacecraft are to obtain biological, geophysical-geological, and atmospheric information about Mars and Venus, and to provide the basic knowledge required for subsequent manned missions. The scientific priority is on the detection and characterization of extra-terrestrial life. Because of the much higher probability of the existence of life on Mars, emphasis throughout the study was on the first Mars opportunity in 1969.

The guidelines and constraints for the study were provided by NASA as follows:

a. Parametric studies were to be conducted and one design concept recommended

b. Performance was to be optimized for maximum scientific payload, data collection and transmission, and lifetime with due regard for a high level of reliability in the complete spacecraft

c. Emphasis was to be placed on a spacecraft having a nominal weight of 7000 pounds, capable of being launched by a Saturn C1B with an S-VI upper stage, and to fit within a shroud diameter of 260 inches

d. Consideration was to be given to a spacecraft design having a nominal weight of 3500 pounds, capable of being launched by a Titan III-C, and to fit within a shroud diameter of 120 inches

e. Consideration was to be given to potential compromises in the 7000-pound spacecraft to accommodate later growth to a 60,000-pound spacecraft, capable of being launched by an Advanced Saturn

f. The same basic spacecraft was to be used for all mission opportunities with minimum change. Where requirements for Mars and Venus were in conflict, the design was to be optimized for Mars and compromised for Venus

g. A minimum 30-day launch window was to be considered

h. A complete 210-ft dish DSIF network was to be assumed to be available

i. The mission was to be designed so that the probability of landing one or more viable terrestrial micro-organisms, with either spacecraft or final stage booster, would be less than 10^{-2} for Venus and 10^{-4} for Mars

j. The model atmospheres used were to be those supplied by NASA. These are summarized in Figures 1.0-1 through 1.0-3 and in Table 1.0-1.

It was necessary, during the course of the study program, to study the experiments and scientific equipment for each of the missions in considerable detail in order to make proper decisions in the system tradeoffs required, although such studies were not in the scope of the study program. The results of these additional studies have been included in Volume II.

Late in the study program new information became available to NASA which indicated that the model atmospheres used for Mars might be incorrect. A new model atmosphere was defined by NASA, and a short, parallel study was performed to show what effect, if any, this new atmosphere would have on the system design recommended. The results of this parallel study are included in this report.

TABLE 1.0-1 SURFACE CONDITIONS

	Model Atmosphere	Pressure Lbs/ Ft²	Torr mm	Atm	Temperature °R	°F
MARS	Lower	86.65	31.12	.0628	360	-100
	(original) Mean	180.82	64.94	.1311	450	-10
	Upper	281.88	101.23	.2044	540	+80
	11 Mb	22.97	8.25	.0166	468	+8
	(new) 15 Mb	31.32	11.25	.0227	414	-46
	30 Mb	62.64	22.50	.0454	375	-[illegible]
VENUS	Maximum	15,860		11.50	1512	1052
	Standard	21,150		15.34	1260	800

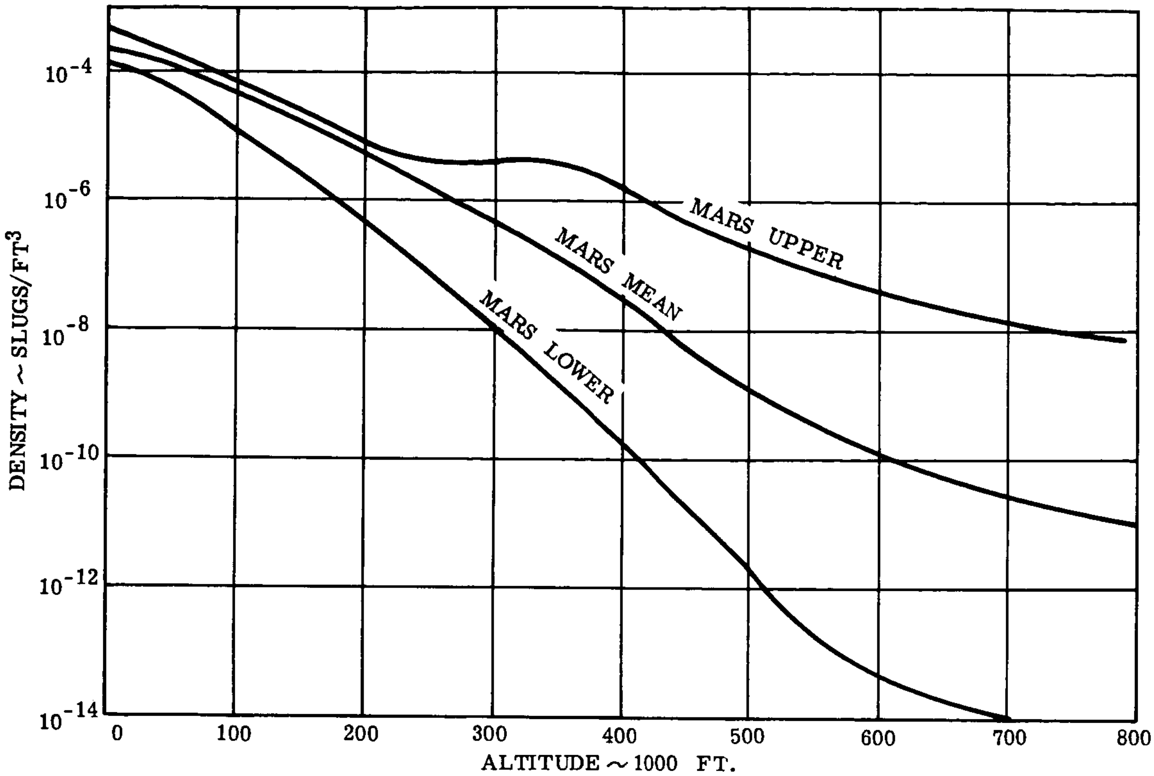

Figure 1.0-1. Mars Atmospheres - Density

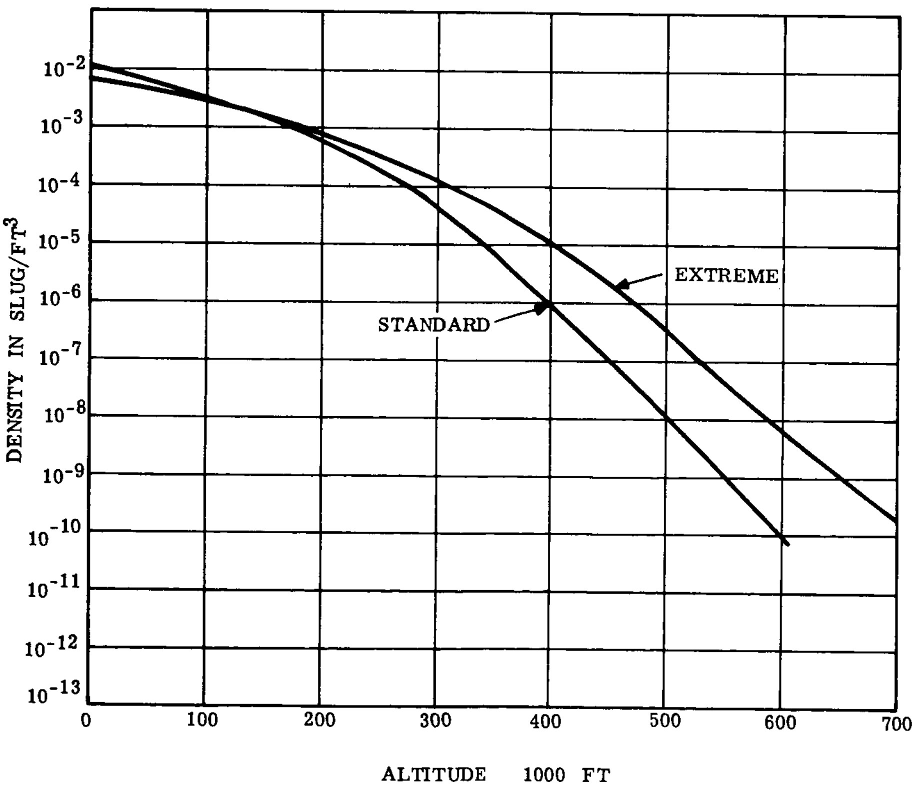

Figure 1.0-2. Venus Atmospheres - Density

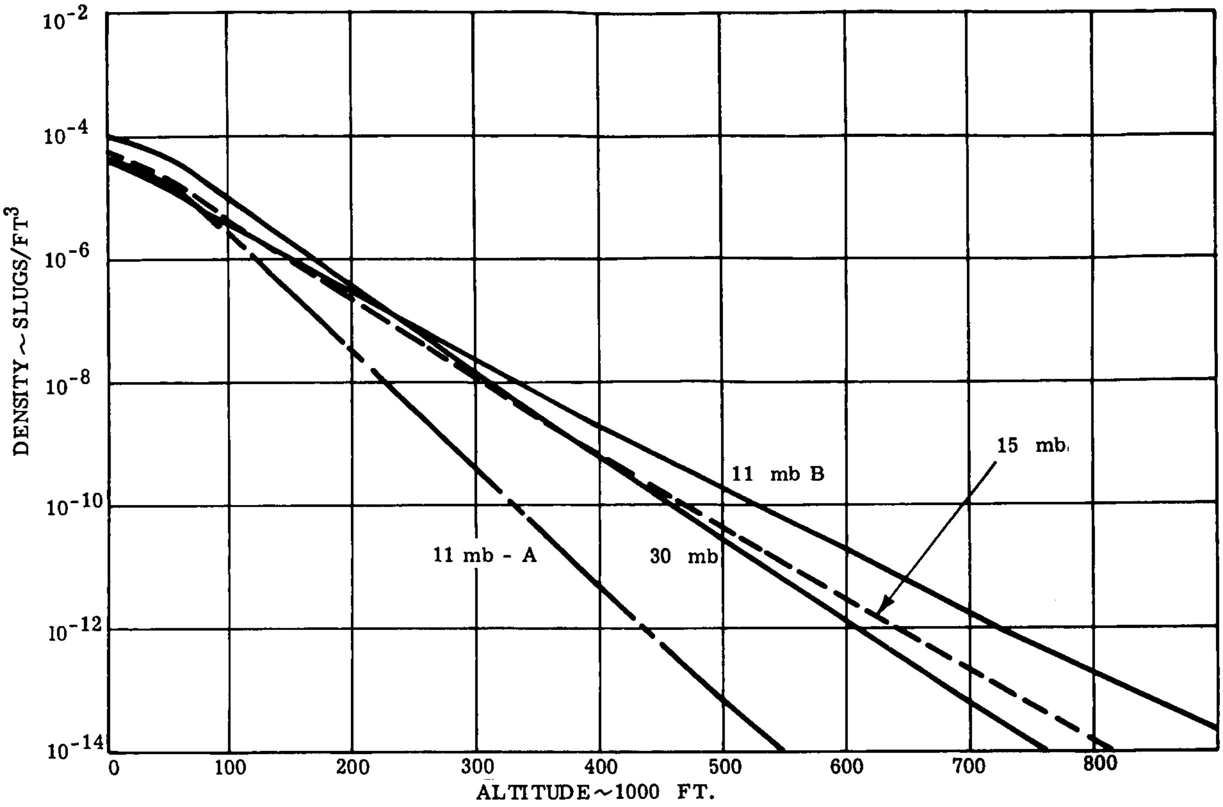

Figure 1. 0-3. Density -Altitude Profile, 15 Millibar Atmosphere

SECTION 2.0 THE RECOMMENDED VOYAGER SYSTEM

2.1 MARS 1961 SYSTEM

The Voyager System recommended in this report is the result of extensive studies during which the following factors were considered: a) the mission requirements, b) the launch vehicle capability, c) the balance between Orbiter and Lander scientific equipment weights and mission values in conjunction with the reliability studies, d) the use of energy-saving eccentric orbits to deliver maximum scientific equipment, and e) the provision for adequate power, guidance and communication capabilities with appropriate application of redundancy and back-up modes at critical junctions in each subsystem.

2.1.1 DESIGN FEATURES

The recommended system, shown in Figure 2.1.1-1, incorporates two identical Landers and an Orbiter with a single propulsion system for midcourse, terminal and orbit injection maneuvers. The main communication link from the Lander to the Earth is through a relay in the Orbiter, with secondary links directly from the Lander to the Earth. All high data rate communication is transmitted from the Orbiter to Earth through an earth-oriented, ten-foot diameter parabolic antenna. The secondary direct link from Lander to Earth utilizes an earth-oriented helix. Solar cells and batteries are the power sources in the Orbiter, primarily because of anticipated restrictions on the availability of sufficient quantities of the desirable radioisotope. However, in the Mars Landers, due to a heating requirement during the Martian nights and the long-life times required, radioisotope thermoelectric generators are the primary power source. The size of these generators is minimized by using a combination of generator and secondary batteries (the latter for high power demands) with the communication links operated on an intermittent basis. High volume data storage is provided by thermoplastic recorders both in the Orbiter and in the Landers.

A star tracker, sun sensors, earth sensors, televised photographs of the target planet against the star background, and two-way doppler tracking are utilized to provide guidance intelligence. The attitude control system utilizes all gas components for simplicity, light weight, and to take the greatest advantage of current state-of-the-art. The single Orbiter propulsion system is a pressurized hypergolic fuel system with a combination of radiation and ablative cooling on the thrust chamber.

2.1.2 SCIENTIFIC CAPABILITY

The scientific capability of the Mars 1969 system is given in Table 2.1.2-1. The emphasis in the payload design has been on biological and geophysical-geological experiments. In this regard the choice for the landing site is Syrtis Major (10° N lat., 285° long.) for one Lander and Pandorae Fretum (24° S lat., 310° long.) for the second since these are two of the more interesting areas from the standpoint of biological exploration. The appearance of Syrtis Major does not change much with seasons; the boundaries are sharp and stable, and it is one of the darkest areas of the planet. Pandorae Fretum, on the other hand, changes considerably with the season with dark color development in spring, deepening with the approach of summer, becoming light again in fall and remaining light in winter.

From many standpoints it would be desirable to land in higher latitude regions. The polar caps never extend down as far as Pandorae Fretum, so that close examination of the white material of the caps, probably frost, is denied. Similarly, the "dark wave" will not be investigated by the Landers. However, in view of the high priority of the life detection and the eventual requirements for choosing sites for manned landing missions, the lower latitude sites seem logical choices for the first opportunity. The landing locations for the later Landers will be determined from the results of the 1969 mission.

TABLE 2.1.2-1. SCIENTIFIC CAPABILITY, MARS 1969 SYSTEM

Each Lander

Biological
- Growth
- Metabolic Activity
- Existence of Organic Molecules
- Existence of Photoautotroph
- Turbidity & PH Changes
- Microscopic Characteristics (TV)
- Organic Gases
- Macroscopic Forms (TV)
- Surface Sounds

Geophysical-Geological
- Surface Penetrability
- Soil Moisture
- Seismic Activity
- Surface Gravity

Atmospheric
- Temperature
- Pressure
- Density
- Composition
- Altitude
- Light Level
- Electron Density

Related Equipment
- Drill, Pulverizer,
- Sample Handling Equipment

Orbiter

Biological

Color Characteristics (TV)	Resolution
Blue	140 M
Green-Yellow	140 M
Red	140 M

IR Spectrum (Sinton Bands)

Geophysical-Geological

Geological Provinces

Stereo Map (TV)	1000 M Hor. 345 M Vert.
NADIR (TV)	1000 M
NADIR (TV)	20 M

- Magnetic Field
- Charged Particle Flux
- Albedo

Atmospheric
- Ionospheric Profile
- IR Emission

Space Environment
- Micrometeoroids
- Magnetic Fields

As indicated in Table 2.1.2-1 a large portion of the Orbiter scientific payload is devoted to various television systems to map the biological and geological provinces. The Orbiter is placed in an eccentric orbit with a perifocal altitude of 1000 nm (± 100 nm) an apofocal altitude of 19,000 nm (+6000 nm, -4500 nm) and an inclination of 55° to the equator. Since the orbit injection point is in the southern hemisphere and the television pictures are taken only around the perifocus to the latus rectum, only about 80 percent of the southern hemisphere is mapped to a maximum resolution of 1000 meters in stereo (345 meter height resolution). The higher resolution black and white and the color pictures are not taken as often, yielding the following number of pictures taken in one complete mapping period of 26 (+9, -10) days:

1000 meter resolution pictures	—	2484
Color sets	—	1196
20 meter resolution pictures	—	1196

One of the primary experiments on the Landers is the panoramic television and the petrographic microscope. The information from these sensors is transmitted to the earth via the Orbiter during the first three months (Orbiter life) and directly thereafter. The cumulative number of television pictures transmitted to the earth from each Lander is:

Time After Landing	No. of TV Pictures
28 hours	125
85 hours	363
3 months	5050
6 months (Lander life)	5585

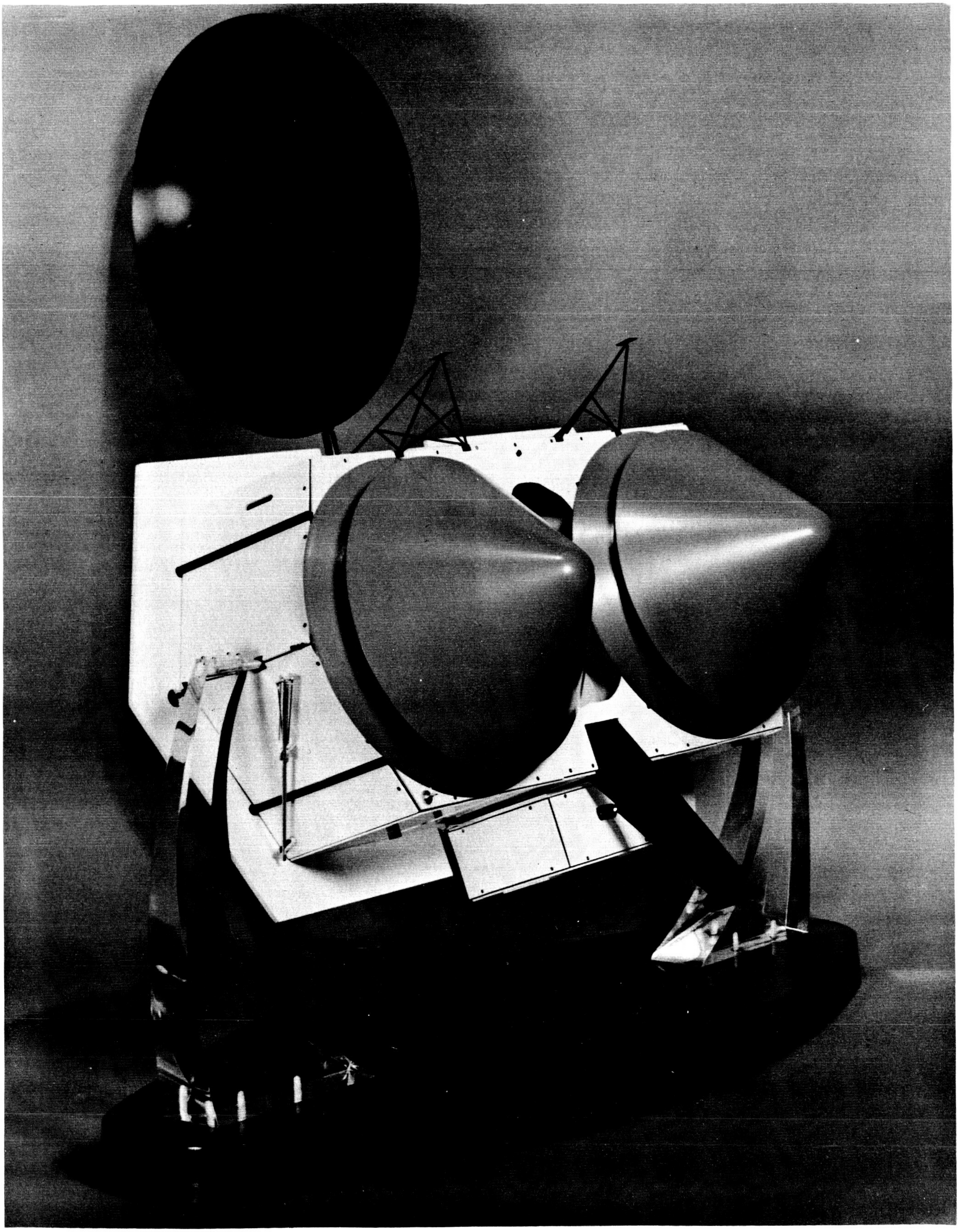

Figure 2.1.1-1. Voyager Spacecraft

2.1.3 MISSION PROFILE

After injection into the transit trajectory, the Voyager spacecraft is separated from the S-VI stage and automatically acquires the cruise attitude; this is confirmed by transmission through the high gain antenna. Midcourse maneuver requirements are computed on the Earth and transmitted to the spacecraft, as is the command for execution. If subsequent tracking data indicates that another midcourse maneuver is required, this can be performed; otherwise, the vehicle proceeds on its way to the planet, periodically transmitting the results of any transit science plus engineering and diagnostic telemetry. When the spacecraft is within 2,000,000 nm of the target planet, an image orthicon camera is used to photograph the planet against the star background. This picture is transmitted to Earth and is used to determine the requirements, if any, for a terminal correction maneuver. This maneuver would be performed approximately 145 hours before encounter (depending upon the magnitude of the error), thus correcting the orbit perifocal altitude to the required degree of accuracy. Additional pictures are transmitted after the maneuver in order to determine the accuracy of the resulting trajectory and to provide information for the computation of the Lander separation requirements.

The separation maneuver, planned for 17.8 hours and 150,000 nm before encounter, requires the spacecraft to rotate from the cruise orientation to the direction of the velocity increment to be imparted to the Lander. The Lander is unlatched from the Orbiter and given a small separation impulse by a cold gas system incorporated in the Lander adapter. At a separation distance of 3 feet, cold gas jets are activated and the Lander is spun up and stabilized prior to the firing of the retro motor. Approximately 17 minutes later, when the separation distance has reached 1000 feet, the solid rocket motor is fired to impart the required velocity increment to the Lander. This separation distance is deemed sufficient to preclude Orbiter degradation or disturbance from the retro system. While the first Lander moving away, the spacecraft rotates to the attitude required for the second Lander velocity impulse. The second Lander can be separated as late as 11 hours before encounter and still have enough energy to reach its landing site. Time is therefore available if it is necessary to repeat the orientation sequence of the main vehicle because of an unexpected disturbance from the separation of the first Lander. During the time between separation and entry, the Landers periodically telemeter engineering data back to the Orbiter using the relay link.

In systems which deliver entry vehicles to another planet with relatively unknown atmospheres, a prime consideration is the monitoring of the entry phase down to surface impact by extensive diagnostic and scientific instrumentation. If the retardation system fails or surface characteristics are encountered that are outside the design limits, a maximum amount of information should be obtained up to the failure point so that future Landers can be modified to eliminate such failures.

In order to maintain communications during this critical period, line-of-sight between the Orbiter and Lander must be maintained as the Lander enters and descends to the surface. This is accomplished by a judicious choice of the Lander velocity increment. In addition, a descent radar incorporated in the Mars Landers is used to insure that deployment of the final parachute is delayed until the Lander reaches the altitude of 30,000 feet. The descent time is thereby minimized, which also minimizes the line-of-sight time problem. This reduces the descent time for extensive atmospheric measurements, but since the Voyager Landers emphasize the surface experiments (Mariner B capsules will be designed for atmospheric measurements), this restriction is considered to be reasonable.

While the Landers are proceeding toward the planet, the Orbiter has returned to sun orientation, recharged its batteries and as the Landers approach altitudes of 1,000,000 feet, the Orbiter is commanded to assume the orbit insertion attitude. This eliminates an additional attitude orientation maneuver between the end of the line-of-sight with the Lander and orbit insertion points. Orbital scientific equipment is deployed after orbit insertion and communications are begun on the first orbit so that all accumulated Lander information can be transmitted to earth as soon as possible. A condensed sequence of events is indicated in Table 2.1.3-1.

2.1.4 SYSTEM WEIGHTS

The estimated weight for the Mars 1969 system is given in Table 2.1.4-1.

TABLE 2.1.4-1. MARS 1969 SYSTEM WEIGHTS

Orbiter		
Structure	419	
Harnessing	106	
Power Supply	218	
Guidance & Control	226	
Communications	291	
Thermal Control	87	
Propulsion (Dry)	467	
Diagnostic Instrumentation	30	
Payload (Scientific)	215	
Total	2059	2059 lbs.
Landers		
Heat Shield	90	
Structure	399	
Retardation	159	
Thermal Control	91	
Power Supply	112	
Orientation	58	
Communications	143	
Payload Deployment & Installation	56	
Spin & Separation	42	
Retro Rocket	98	
Adapter & Radiator	47	
Payload (Scientific)	155	
	1450	
Total (2)		2900
Fuel (Orbit Insertion & Midcourse)		2071
Total		7030 lbs.

Table 2.1.3-1
Sequence of Events

	Time to Begin Operation
1. Entry into Transit Mode	(separation from launch vehicle)

Orient to Sun
Deploy High-Gain Antenna
Switch to High-Gain from Omni
Orient to Canopus
Earth verification of Canopus acquisition
Switch to Omni Antenna
Stow High-Gain Antenna

2. First Mid-course Correction .... +1-2 weeks

Commands received from Earth
Orientation of spacecraft to required altitude
Firing of Main Engine

3. Reorientation to the Sun .... Immediately following engine firing

Commands read out by Programmer
Orientation to Sun and verification
Orientation to Canopus and verification

4. Second Mid-course Correction .... +5-8 weeks

5. Reorientation to the Sun .... Immediately following engine firing

6. High-Gain Antenna Deployment .... +187 days

Commands transmitted to spacecraft
High-Gain Antenna pointed to Earth using sensor corrected on programmed angles

7. Terminal Guidance Observation (2×10^6 nm from Planet) ..... +258 days

TV camera turned on by command
TV pictures of planet and background transmitted to Earth

8. Final Trajectory Correction (1×10^6 nm from Planet) .... +264 days

Command transmitted to spacecraft
Switch to Omni
Store High-Gain Antenna
Orientation of spacecraft to required spatial attitude
Firing of Main Engine

9. Reorientation to Sun & Deploy High-Gain Antenna .... Immediately after engine firing

10. TV Pictures taken of Planet and Background

11. Lander Separation (150,000 nm from Planet) .... +269 days

ORBITER

Command transmitted to the spacecraft
Orientation of spacecraft to desired spatial attitude
Physical attachment of Lander to Orbiter broken
Lander is separated
Orbiter is reoriented for second Lander separation
Second Lander is separated

— Orbiter Reorientation to the Sun .... Immediately following Lander separation

— Orbiter Reorientation for Injection into Orbit .... 50 minutes before retro engine firing

Commands transmitted to spacecraft
Switch to Omni Antenna
Stow High-Gain Antenna
Orientation of spacecraft to required attitude

— Firing of Main Engine

— Orbiter in Orbit .... +270 days

Deploy High-Gain Antenna
Deploy Planet Pointing Package
Deploy any other instrumentation required for the specific mission

LANDERS

	Altitude	
— Activate cold gas spin system		Immediately following Lander Separation
— Fire retro rocket		17 min. after separation
— Fire thruster bolts - shed adapter		1 min. before entry
— Entry Phase	1,000,000 ft.	17 hours after separation
Enter blackout	500,000 to 800,000 ft.	
Radar Altimeter on	79,000 to 330,000 ft.	
End blackout	36,000 to 250,000 ft.	
Switch from transit antenna to descent antenna		
Decelerator chute deployed	22,500 to 30,000 ft.	
Main chute deployed in reefed condition		
— Final Descent Phase		
Main parachute disreefed	4,000 to 15,000 ft.	
Impact	0	15-25 minutes from 1,000,000 ft.
— Surface Phase		
Lander orientation and experiment deployment		
Playback descent and initial surface information		
Deploy TV and begin experiments		23 minutes after impact

2.2 ALTERNATE MARS 1969 SYSTEM

Although the major effort of the study was concentrated on the design of a system for the Saturn C-1B launch vehicle, it is recommended that strong consideration be given to a Voyager system based upon the Titan III-C. Preliminary comparisons between the Saturn C-1B with the S-VI stage and the Titan III-C launch vehicle show that an all-Orbiter Titan III-C spacecraft together with an all-Lander Titan III-C spacecraft are more flexible and can accomplish as much or more than the Saturn C-1B system. In addition, preliminary cost comparisons indicate that the Titan III-C systems might be more economical, especially if there were no other applications for the S-VI stage.

Table 2.2-1 shows that the estimated payload for the Saturn C-1B launched Mars 1969 Orbiter could be exceeded by an all-Orbiter launched with a Titan III-C. However, a single launch of the best Lander combination would deliver less total Lander weight to the surface of Mars than a single Saturn C-1B combined Orbiter-Lander.

TABLE 2.2-1. ALTERNATE MARS 1969 SYSTEMS

Launch Vehicle	Titan III		Saturn C-1B
Spacecraft	Orbiter	Bus Lander	Orbiter and Lander
Structure	234	206	419
Harness	70	60	106
Power Supply	218	49	218
Guidance and Control	216	200	226
Communications	276	10	291
Diagnostic Instrumentation	30	30	30
Thermal Control	70	10	87
Propulsion	370	115	467
Orbiter Payload	223	—	215
Orbiting Weight	1707	(bus) 680	2059
Lander Weight	—	2230	(2) 1450
Lander Payload	—	(330)	(155) each
Fuel	1643	90	2072
Total Weight	3350	3000	7030
Orbit (x 1000 nm)	1 x 19	—	1 x 19

In arriving at the estimated system performance for the Titan III-C systems, it was assumed that the different systems given in Table 2.2-1 used the same thirty day launch windows which were chosen to maximize the Orbiter system capability. This was done because of the desirability of having both the Orbiter and the Landers functioning during the same period, thus improving the reliability of the Landers by utilizing a communications relay link between the two in addition to the Lander-Earth direct links. If only a bus-Lander system were flown, a different launch window would be chosen which would add several hundred pounds to the Lander weight capability.

SECTION 3.0 EVOLUTIONARY VOYAGER PROGRAM

The logical development of the knowledge of target planets with succeeding opportunities (and, concomitantly, the expected evolution of sensor and scientific objectives), plus the variations in the injection and orbit insertion energy requirements, preclude a static Voyager program utilizing invariable spacecraft and payloads. Consequently, an Evolutionary Voyager Program was developed. This program is summarized in Table 3.0-1 which shows the Mars opportunities from 1969 through 1975 and the Venus opportunities for 1970 and 1972.

It can be seen from this table that the utilization of an Orbiter is high in the beginning of the Mars program, and diminishes to zero in 1975 where the Orbiter is employed as a fly-by bus to deliver the Landers. The increasing knowledge of the biological and geological provinces of Mars obtained by the Orbiters during the 1969 and 1971 opportunities lead to the investigation of more interesting surface areas by the Landers in 1973 and 1975, perhaps culminating in the detailed survey of possible manned landing sites by a surface rover in 1975.

Two identical Landers were chosen for each of the Mars missions since the results of the reliability analysis indicate that when the total weight allocated to Landers was greater than 1840 pounds, the estimated attainable mission value was maximized by dividing the weight between two identical Landers. For the Venus missions, however, one Lander is better because the high weight of the thermal control system causes a severe reduction in the payload carried by a Lander. This, in turn, causes the minimum weight for a dual Lander system to be higher than the capability of the Voyager spacecraft.

Due to the cloud cover surrounding Venus, the first Orbiter would carry radar mapping equipment and, due to the uncertainty in knowledge of the atmosphere and surface conditions on Venus, a minimum Lander of 525 pounds is utilized. In the 1972 Venus opportunity, the Orbiter is lighter since the interest in radar measurements of the surface will have waned and a large Lander is carried with considerable capability to make geological, atmospheric and possibly biological surveys of the surface environment on Venus.

A detailed listing of the experiments to be conducted during each of the missions is given in Tables 3.0-2 to 3.0-7.

The Orbiters and Landers summarized in the Evolutionary Voyager Program are all based on the weight capabilities of the Saturn C-1B launch vehicle with the S-VI upper stage. The availability and capability of the Titan III-C launch vehicle were reviewed and the characteristics of an Evolutionary Voyager Program utilizing this booster are summarized in Table 3.0-8. Single Landers are indicated in this program because the dual/single Lander crossover point had not been firmly defined at the time the Titan III-C portion of the study was completed. Nevertheless, it can be seen from the two tables that the payload capabilities of the Titan III-C systems are substantially equivalent to those of the Saturn C-1B. In addition, the Titan III-C Orbiter system has the capability of a circular orbit in 1971 which would permit obtaining a television map with constant resolution and a much greater coverage than the Saturn C-1B Orbiter.

TABLE 3.0-1. EVOLUTIONARY VOYAGER PROGRAM

	MARS				VENUS	
	1969	1971	1973	1975	1970	1972
Total Weight (lbs.)	7030	7320	6000	5500	7260	7350
Orbiter (lbs.)	2058	2100	1400	1335 (Fly-by Bus)	2145	1800
Payload (lbs.)	215	223	77	61	137	61
Mission	S. Hemi Map (Stereo) Color Characteristics Ionospheric Profile Particles and Field Planetary Emission	N. Hemi Map (Stereo) Color Characteristics Ionospheric Profile Particles and Field Planetary Emission	Upper Atmosphere Composition Particles and Field	Particles and Fields Planetary Emission	Radar Map, Particles and Field, TV of Clouds, Ionospheric Profile	Particles and Field of Clouds Ionospheric Profile
Landers (2) (lbs.)	1450 (each)	2000	2000	2000	525 (single)	2600 (single)
Payload (lbs.)	155 (each)	255	255	255	60	210
Mission	Biological Analysis, Forms-(TV)-Micro and Macro, Geological Atmospheric	Biological Analysis, Forms-TV-Micro and Macro, Geological, Ionospheric Profile, Atmospheric	Upper Atmosphere Characteristics, Biological Analysis, Micro and Macro Forms (TV) Geological Ionospheric Profile Atmospheric	TV Survey of Possible Manned Landing Areas (Surface Rover) Micro Forms (TV), Geological, Atmospheric	Descent Radar or TV, Surface Hardness, Macro, Forms-(TV), Atmospheric	Micro and Macro Forms (TV) Geological, Ionospheric Profile, Atmospheric
Orbit (nm)	1000 x 19,000	1000 x 19,000	200 x 9000	-	1000 x 4300	1000 x 73[illegible]0
Inclination	55^o	45^o	53^o	-	68^o	68^o
Life						
Orbiter	3 mo.	3 mo.	10 days	-	3 mo.	3 mo.
Landers	6 mo.	6 mo.	6 mo.	6 mo.	10-30 min.	6.5 hrs.

TABLE 3.0-2. MARS 1969

I. Landers During Entry or Descent

1. Temperature
2. Pressure
3. Density
4. Composition
   a. Mass Spectrometer
   b. Gas Chromatograph
5. Altitude (Radar Altimeter)
6. Electron Density (Langmuir Probe)

II. Landers on Surface

1. Temperature
2. Pressure
3. Density
4. Composition of Atmosphere
   a. Mass Spectrometer
   b. Gas Chromatograph
5. Wind Speed & Direction
6. Television - 2 Cameras
7. Precipitation
8. Surface Sounds
9. Light Level Indicator
10. Surface Penetrability
11. Soil Moisture
12. Seismic Activity (1-axis seismometer-seismograph)
13. Surface Gravity
14. Radioisotope Growth Detector
15. Turbidity & PH Growth Detector
16. Multiple Chamber Growth Detector
17. Photoautotroph Detector
18. Microscopic Analysis
   a. Atmospheric Aerosols
   b. Surface Materials
   c. Biological Materials
19. Drill
20. Pulverizer
21. Sample Handling Equipment

III. Orbiter

1. IR Multichannel Radiometer
2. IR Spectrometer
3. Magnetic Field
4. Television (multi-color stereo)
5. Cosmic Dust
6. Ionospheric Profile: Radio Propagation Experiment
7. Ionization Chamber and GM Tube Assembly
8. Solar Multichannel Radiometer
9. Polarimeter (Skylight Analyzer)
10. Sferics
11. X-ray Flux from Sun

TABLE 3.0-3. MARS 1971

I. Landers During Entry or Descent

1. Temperature
2. Pressure
3. Density
4. Composition
   Mass Spectrometer
5. Altitude
   (Radar Altimeter)
6. Radiometer
   1215 Å (Lyman α)
   1026 Å (Lyman β)
   972 Å (Lyman γ)
   584 Å (HeI)
   304 Å (HeII)
   1445 Å - 1500 Å
   2500 Å - 3000 Å
7. 8446 A Radiometer

II. Landers on Surface

1. Temperature
2. Pressure
3. Density
4. Composition
   a. Mass Spectrometer
   b. Gas Chromatograph
5. Wind Speed & Direction
6. Television - 2 Cameras
7. Surface Sounds
8. Polarimeter (Skylight Analyzer)
9. Surface Penetrability
10. Seismic Activity
    1-Axis Seismometer
11. X-ray Diffractometer
12. α-Particle Scattering
    Sensor
13. Thermal Diffusivity of Ground
14. Electrical Conductivity of
    Ground
15. Microscopic Analysis
    a. Atmospheric Aerosols
    b. Surface Material
    c. Biological Material
16. Multiple Chamber Growth
    Detector
17. Photoautotroph Detector
18. (Insolation) Pyrheliometer
19. Surface Radioactivity
20. Meteor Trails
21. Ionospheric Profile:
    Bottomside Sounder
22. Sferics Storm detector
23. Eclipse by Phobos
24. Insect Attractor
25. Pulse Light
26. Drill
27. Pulverizer
28. Sample Handling Equipment

III. Orbiter

1. IR Multichannel Radiometer
2. Solar Multichannel Radiometer
3. Magnetic Field
4. Electron Spectra and Direction
5. Proton Spectra and Direction
6. Television (multi-color; Stereo)
7. Cosmic Dust
8. Ionospheric Profile: Radio
   Propagation Experiment
9. Altitude (Radar Altimeter)
10. UV Multichannel Radiometer
11. UV Solar Spectrometer
12. Sferics
13. Faraday Cup
14. X-ray Flux From Sun

TABLE 3.0-4. MARS 1973

I. Landers During Entry or Descent

1. Temperature
2. Pressure
3. Density
4. Gas Chromatograph
5. Altitude (Radar Altimeter)
6. UV Solar Spectrum
7. Electron Density (Langmuir Probe)

II. Landers on Surface

1. Temperature
2. Pressure
3. Density
4. Gas Chromatograph
5. Wind Speed & Direction
6. Television
7. Surface Sounds
8. Microscopic Analysis
    a. Atmospheric Aerosols
    b. Surface Material
    c. Biological Material
9. Multiple Chamber Growth Detector
10. Photoautotroph Detector
11. Meteor Ionization Trails
12. Ionospheric Profile Bottomside Sounder
13. Rocket Soundings of Atmosphere (8 rockets)
14. Seismic Properties
    a. Natural
    b. Induced
15. Polarimeter - Skylight Analyzer
16. Insect Attractor
17. Pulse Light
18. Sample Handling Equipment

III. Orbiter

1. Magnetic Field
2. Proton Telescope
3. Electron Telescope
4. Mass Spectrometer
5. Electron Probe (Langmuir Probe)
6. X-ray Flux from Sun
7. γ -ray Spectrometer
8. Faraday Cup

TABLE 3.0-5. MARS 1975

I. Landers During Entry or Descent

1. Temperature
2. Pressure
3. Density
4. Gas Chromatograph
5. Altitude (Radar Altimeter)
6. UV Solar Spectrometer
7. Aerosol Profile
8. Electron Density (Langmuir Probe)
9. Solar 3-channel Radiometer

II. Landers on Surface

1. Temperature
2. Pressure
3. Density
4. Gas Chromatograph
5. Wind Speed & Direction
6. Television - 2 Cameras
7. Surface Sounds
8. Polarimeter - Skylight Analyzer
9. Surface Penetrability
10. X-ray Diffractometer
11. Thermal Diffusivity of Ground
12. Electrical Conductivity of Ground
13. Microscopic Analysis
    a. Atmospheric Aerosols
    b. Surface Materials
    c. Biological Materials
14. Surface Radioactivity
15. Insolation - Pyrheliometer
16. Laser-induced Gaseous Emission Spectra
17. Laser Atmospheric Back-scatter Probe
18. Sample Handling Equipment

TABLE 3.0-6. VENUS 1970

I. Landers During Entry or Descent

1. Temperature
2. Pressure
3. Density
4. Cloud Properties
5. Altitude
   (Radar Altimeter)

II. Landers on Surface

1. Temperature
2. Pressure
3. Density
4. Surface Sounds
5. Television Panorama
6. Television Light Source
7. Atmospheric Composition - Gas Chromatograph
8. Wind Speed
9. Light Levels
10. Polarimeter

III. Orbiter

1. IR Multichannel Radiometers
2. Solar Multichannel Radiometer
3. Magnetic Field
4. Charged Particle Flux (Ionization Chamber and G-M Tube Assembly)
5. Television (Single color - not Stereo)
6. Cosmic Dust
7. Ionospheric Profile: Radio Propagation Experiment
8. Radar Map

TABLE 3.0-7. VENUS 1972

I. Landers During Entry or Descent

1. Temperature
2. Pressure
3. Density
4. Gas Chromatograph
5. Altitude (Radar Altimeter)
6. Cloud Properties
   a. Liquid Particles
   b. Solid Particles
      1) Filter
      2) Light Reflection
7. Television (See Lander on Surface)
8. UV Solar Spectrometer

II. Landers On Surface

1. Temperature
2. Pressure
3. Density
4. Gas Chromatograph
5. Wind Speed & Direction
6. Television - Panorama
7. Surface Sounds Modified
8. Skylight Analyzer
9. Surface Penetrability
10. Seismic Activity (Natural, one-axis) Seismograph
11. X-ray Diffractometer
12. α-Particle Scattering
13. Thermal Diffusivity of Ground
14. Electrical Conductivity of Ground
15. Microscopic Analysis
    a. Atmospheric Aerosols
    b. Surface Materials
    c. Biological Materials
16. Insolation - Pyrheliometer
17. Surface Radioactivity γ-ray Spectrometer
18. Meteor Trails
19. Ionospheric Profile - Bottomside Sounder
20. Sferics

III. Orbiter

1. Magnetic Field
2. Electron Spectra and Direction (Electron Spectrometer)
3. Proton Spectra and Direction (Proton Spectrometer)
4. Cosmic Dust
5. IR Multichannel Radiometer
6. Airglow Analyzer
7. Television (Color Filter and Zoom)
8. Ionospheric Profile: Radio Propagation Experiment

TABLE 3.0-8. MARS TITAN III-C SYSTEMS

	1969	1971	1973	1975	1977	1979
All-Lander System						
Weight Injected (lbs.)	3000	3000**	2750	2750*	3000*	3000
Lander Weight (lbs.)	2230	2230	2000	2000	2230	2230
Trip Time (days)	275	128	167	~325	(Less than 325)	~180
All-Orbiter System						
Weight Injected (lbs.)	3350	3600	2800	-	-	-
Scientific Payload in Orbiter (lbs.)	223	223	223			
Orbit (nm)	1000 X 19,000	1000 X 1000	1000 X 13,000			

* Type II trajectories, but higher than minimum energy trip.
** Higher than minimum energy trip to minimize changes in Lander Size.

SECTION 4.0 MARS EXPLORATION PROGRAM

In order to fully assess the scientific merit of the Voyager program it is informative to compare the scientific measurements on Mars which can be obtained from the Voyager vehicles with those which are likely to be obtained by Mariner vehicles and by various configurations of manned missions. Such a comparison requires that two types of value judgements be made. They are:

a. Judgment of the relative value of different types of measurements - biological, atmospheric, geophysical, and interplanetary environmental.

b. Judgment of the degree to which each type of system will provide the data required.

In making the first of these judgments, the objectives of the planetary exploration program must be reviewed. As discussed above, life detection and its characterization is of highest priority for missions to Mars, geophysical measurements are second, and atmospheric data are third. A fourth category, which is of low priority for planetary exploration, is an improved definition of the interplanetary environment. For purposes of the present analysis, the following relative values have been arbitrarily assigned to the four types of information:

Biological	45 points
Geophysical	30 points
Atmospheric	20 points
Environmental	5 points

The judgment of the extent to which the various systems will provide the required information is a difficult one, particularly since the system concepts for manned missions are still very primitive. Of course, the basic nature of a mission may automatically eliminate certain types of measurements for that mission. For instance, it is obviously impossible to make detailed morphological analyses of surface material from an all-Orbiter configuration. Conversely, a manned Lander is ideally suited to taxonomic studies of organisms, and a comprehensive determination of the magnetic fields around the planet is best done by an orbiting vehicle.

This type of analysis for the scientific exploration of Mars is summarized in the top portion of Figure 4.0-1 for five different types of programs - Mariner, Voyager, and three types of manned missions. The various scientific areas have been divided up into the individual types of information desired, and a judgement of the ability of each of the five systems to provide the data required is indicated by the length of the appropriate bar of the graph. The distance of the line representing 100 percent capability from the base line varies in accordance with the value assignments discussed above.

It is evident from the chart that the system most nearly fulfilling all requirements is the manned Orbiter combined with landing probes. The manned Lander is particularly adapted to the characterization of life on the surface of Mars, but falls short in measurements of the large scale geophysical properties unless an unmanned Orbiter of the Voyager type is also provided. The manned Orbiter without Landers has severe shortcomings in all areas. The Mariner has minimal type capabilities fairly well spread between atmospheric and geophysical measurements. Mariner also has some possibility in the detection of life, but relatively little in its characterization. For a satisfactory biological exploration of Mars, Voyager is a necessity.

Voyager is a very close second to the manned Orbiter combined with landing probes in its ability to conduct a scientific exploration of Mars. (A manned Lander with an unmanned Orbiter would represent a much greater capability.) The Orbiter-Lander

	INFORMATION DESIRED	INFORMATION ATTAINED				
		MARINER B	VOYAGER	MANNED SYSTEMS		
				ALL ORBITING	ORBITING & LANDING PROBES	ALL LANDING
SCIENTIFIC EXPLORATION	BIOLOGICAL	0% (100%)				
	1. DETECTION OF LIFE					
	2. BIO-CHEMICAL ANALYSIS OF ORGANISMS					
	3. MORPHOLOGICAL STUDIES OF ORGANISMS					
	4. TAXONOMIC STUDIES OF ORGANISMS					
	5. MACRO-DISTRIBUTION					
	6. ADAPTATION TO ENVIRONMENTS					
	7. ECOLOGICAL ANALYSES					
	8. INTERACTION OF EARTH & MARS ORGANISMS	VERBOTEN	VERBOTEN			
	ATMOSPHERIC					
	1. VERTICAL PROFILES OF STATE VARIABLES					
	2. ATMOSPHERIC CIRCULATIONS					
	3. CLIMATOLOGY					
	4. ELECTRON DENSITY PROFILE					
	5. AEROSOLS AND PARTICULATES					
	6. ATMOSPHERIC RADIATION					
	GEOPHYSICAL - GEOLOGICAL					
	1. PLANET - PHYSICAL PROPERTIES					
	2. GEOLOGICAL PROVINCES					
	3. TOPOGRAPHY					
	4. SURFACE COMPOSITION					
	5. BEARING STRENGTH					
	6. GEOLOGICAL STRATA					
	7. SEISMIC ACTIVITY					
	8. RADIOACTIVITY					
	9. MAGNETIC FIELDS					
	10. PLANETARY ALBEDO					
	ENVIRONMENTAL					
	1. MAGNETIC FIELDS					
	2. TRAPPED RADIATION					
	3. MICROMETEOROID POPULATION					
DIRECT SUPPORT OF MANNED LANDING PROGRAM	PLANETARY					
	1. RADIATION BELTS					
	ATMOSPHERE					
	1. PROFILE 0 - 100 NM					
	2. COMPOSITION					
	3. IONOSPHERIC PROFILE					
	IDENTIFICATION & CHARACTERIZATION OF POSSIBLE LANDING LOCATIONS					
	1. LOCATE POSSIBLE AREAS ON GROSS BASIS					
	2. BEARING STRENGTH & SIZE OF OBSTRUCTIONS					
	3. HABITABILITY (TEMPERATURE, SURFACE RADIATION, WINDS, BIOLOGICAL)					
	4. PRESENCES OF RESOURCES					
	5. SCIENTIFIC INTEREST					

Figure 4.0-1. MARS Exploration Program

combinations possible on Voyager provide both versatility and excellent capability in all of the areas. However, the geological provinces and topography can probably be determined somewhat better by a manned Orbiter than by the use of television alone. It should be kept in mind, however, that the cost of the manned Orbiter systems is higher than that of the Voyager system by at least an order of magnitude.

In fairness to the manned systems, it should be pointed out that these comparisons are based solely on the scientific measurements listed. There are undoubtedly other, more subjective, considerations which are important for manned missions but which do not show up in this type of tabulation.

A similar analysis was performed for comparing the different types of missions in furnishing the data necessary for developing manned landing systems. The results, as shown by the bottom section of Figure 4.0-1, are essentially the same as those for scientific exploration. The two most attractive systems, with little difference between them, are the manned Orbiter combined with landing probes, and the Voyager system. Reasonably good atmospheric measurements can be made by Mariner, but Mariner is weak otherwise. The chart shows that the manned Orbiter by itself is not well suited to supplying planetary information for the manned landing program.

In summary, it is evident that Voyager would provide a significant increase in capability over Mariner, especially in the areas of biological and geophysical-geological exploration of Mars, and provide the necessary data for the development of a manned landing system. It is the logical and necessary step between Mariner and a manned landing system.

SECTION 5.0 SYSTEM DESIGN DESCRIPTION

5.1 ORBITER

The Orbiter is designed so that it is capable of carrying a variety of different payloads and sizes of Entry/Landers with a minimum amount of modification. Structural support for the Landers is provided by four fixed points per Lander at a 91-inch diameter. These points are capable of supporting Landers varying in weight from 1,200 to 2,200 lbs. Any adaptation required to match these permanent pick-up points will be made in the Lander adapter. Single Landers are capable of being easily attached to the Orbiter. A single Lander requires only an adapter in order to maintain clearance between the Lander rocket engine and the upper surface of the Orbiter. The mounting location is above the spacecraft center of gravity and along the X-axis. Capability can be built into the Orbiter to mount single Landers weighing from 525 lbs. to 4,000 lbs.

The Orbiter structure consists of two main beams, of sandwich construction, which support the required fuel and oxidizer tankage and one-half the Lander load; four main machined fittings which support the remaining half of the Lander loads; sandwich panels which provide support for the fittings and mountings areas for the payload packages; and miscellaneous fittings and brackets. The Orbiter, without Landers, is shown in Figure 5.1-1.

5.1.1 CONFIGURATION

Mounted on the external surface of the Orbiter is the Planet Horizontal Package (PHP), the 10-foot diameter high gain antenna, the magnetometer and magnetometer boom, the VHF omniantenna, the DSIF omniantenna, Landers, the main engine, and solar cells.

The PHP is a self-contained structure capable of viewing the planet at all times during the orbit. The Mars 1969 PHP is designed at present to have two degrees of freedom. A further sophistication of the design is in progress in order to give three degrees of freedom if it is decided that the additional sophistication is necessary. In normal operation, the PHP is stowed at the side of the Orbiter and mounted to it during the boost phase. After injection into orbit about the planet, the PHP is deployed and, by means of sensors mounted within the PHP, pointed at the planet. The total PHP weight is 360 lbs. which consists of structure, thermal control and items such as TV cameras, IR sensors, an ultraviolet radiometer, an IR spectrometer, and other scientific instrumentation. Thermal control is provided for the instruments so that an operating environment of 30°F to 100°F will be maintained.

Mounted upon the PHP is a VHF yagi antenna which is used for communications with the Lander during the orbit lifetime of the Orbiter. This antenna is stowed during transit. After the PHP is released and extended, the communications is switched from the VHF body-mounted antenna to the VHF yagi antenna on the PHP.

Power for the Orbiter is provided by means of solar panels and batteries. The solar panels are mounted on the base of the Orbiter so that the structure serves the dual purpose of carrying Orbiter loads and of supplying a base on which to mount solar cells. The total area available for body-mounted solar cells is 82 sq. ft. Any additional wattage required above what can be provided by the body mounted cells will be provided by extended, fixed panels around the periphery of the Orbiter. These fixed panels are capable of supplying a maximum of 220 additional watts of power. For a requirement in excess of this value, the use of deployable panels or a radioisotope generator would have to be examined.

During launch and transit, the high gain antenna is located in a vertical position at the side of the Orbiter. It is supported by a hinge mechanism attached on the base of the Orbiter and by two brackets at the upper surface of the Orbiter; these help to reduce vibration amplitudes during the boost phase. After injection into transit trajectory, the high gain antenna will be released and deployed while close to the Earth in order that all systems may be checked out. After systems checkout, the antenna will be stored in the launch position for approximately 187 days, when it will be deployed and become the standard method of communications between the Earth and the spacecraft. At this time, the requirement for data rate precludes the use of the DSIF omniantennas. In addition, there is a potential problem of the antenna sensors viewing the Sun. These several factors combine to select the period of 187 days as the time of high gain antenna deployment.

Locating the antenna in the launch position allows the antenna to have a fixed feed. The antenna structure is built so that it may withstand maximum expected loads from the main engine, regardless of antenna position.

A three-axis magnetometer and a 13-ft boom are mounted on the sun side of the Orbiter. Provisions are made so that the boom may be deployed and erected in orbit. The system is built to rotate the boom 180 degrees per day. Time of rotation will be approximately 0.1 second. A 3 ft x 10 ft dipole antenna is mounted on the magnetometer boom; this will be used as the antenna for the Radio Propagation experiment. The magnetometer and boom will be stowed during transit and will be deployed only after the orbit is obtained.

Located on both the sun side and the shaded side of the Orbiter are DSIF omniantennas. These antennas are located so that communication is possible between Earth and the spacecraft regardless of orientation.

In addition to the DSIF omniantenna, a VHF antenna is mounted external to the Orbiter. This antenna will be deployed immediately after the Landers are ejected and will provide communications between the Landers and the spacecraft during entry and descent. While in orbit, the VHF yagi antenna on the PHP will be used.

The main engine, which is used for midcourse corrections and for orbit insertion, is located on the base of the Orbiter, opposite to the Landers. The engine is gimballed and provisions have been made for thrust vector control by means of hydraulic actuators. Two degrees of freedom of movement are provided, to the extent of $\pm 7^{\circ}$. The engine is located so that thrust vector control will be minimized. It is expected that the maximum static c.g. shift requiring thrust vector control will be in the order of ± 0.5 in, which is equivalent to $\pm 1^{\circ}$ of engine gimballing.

The Landers are mounted side-by-side on the shaded side of the Orbiter. They are attached by four explosive bolts per Lander. These bolts will be bonded to the biological barrier so that a sterile interface will be maintained between the Orbiter and Landers.

Other than the four structural attachments of the Landers to the Orbiter, there will be only an electrical connector which breaks the biological barrier. An in-flight disconnect is built into the Lander structure. This in-flight disconnect will be separated just prior to firing the four explosive bolts. Since the separation is within the sterilized area, there will be no contamination of the Landers.

5.1.2 LANDER SEPARATION

Upon command, the explosive bolts attaching the Lander to the Orbiter are fired, releasing the structural tie between the Lander and the Orbiter. Cold gas jets or hot gas rockets mounted on the Lander will be fired imparting a translation to the Lander. These rockets are located so that there will be no impingement of the gases on the Orbiter. After a translation of about 3 ft, spin rockets will be actuated on the Lander in order to spin it. After the first Lander is ejected, the Orbiter is reoriented to a pre-selected angle and the remaining Lander is ejected.

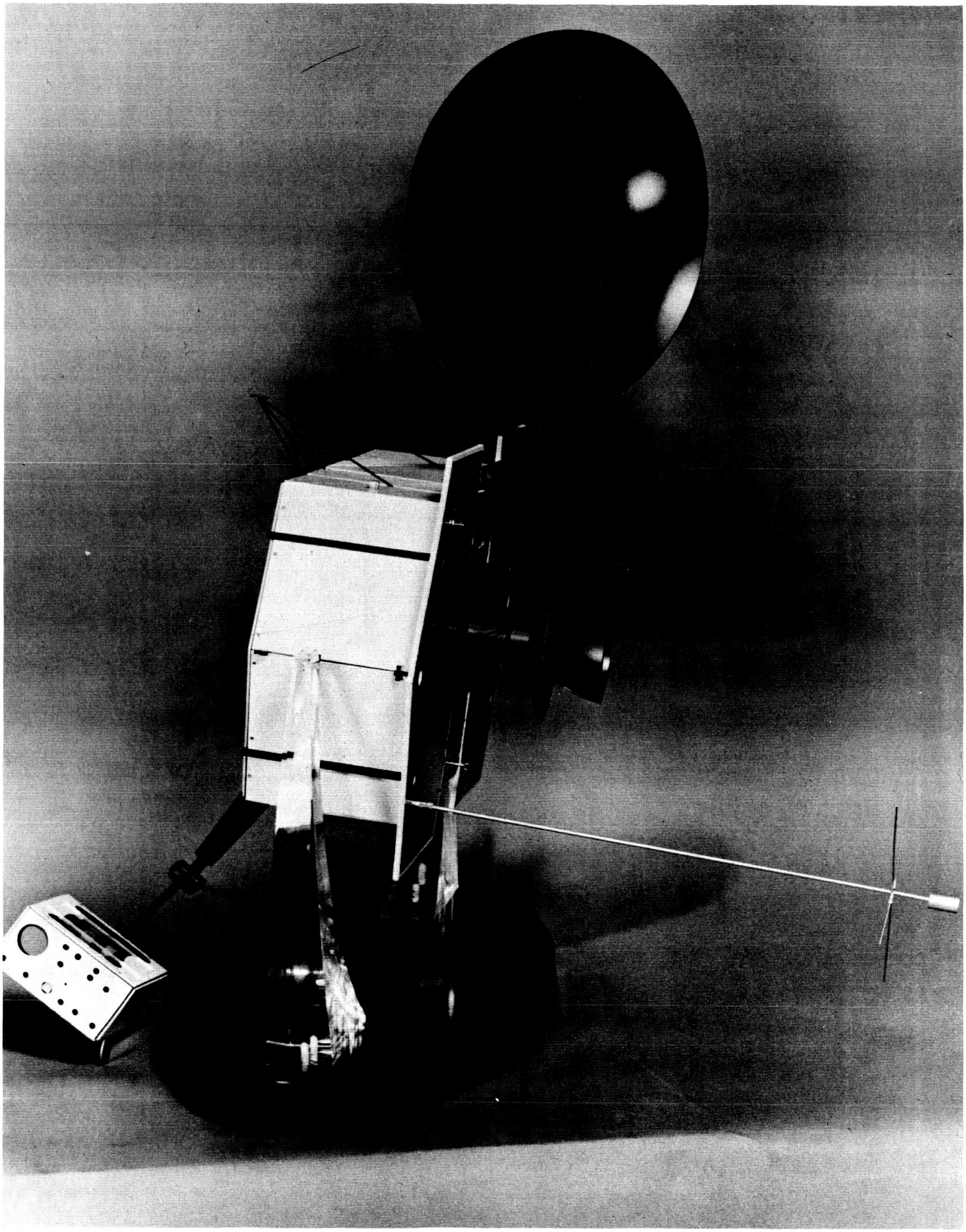

Figure 5.1-1. Mars 1969 Orbiter

5.1.3 PACKAGING

The overall packaging of the Orbiter is as follows. The main mass items, which are the fuel and oxidizer tanks, are mounted between two main sandwich beams extending from one side of the Orbiter to the other. This gives a stiff, low deflection structure and results in a resonant point at high frequencies. Allowance is provided for growth of the fuel tanks to a maximum of 40 in. diameter. Tanks are located so that they will give minimum c.g. shift about the yaw and pitch axis of the spacecraft. However, considered in the c.g. calculation is a potential 3 percent differential usage rate between fuel and oxidizer tanks.

Freon tanks for attitude control and helium tanks for pressurization purposes are mounted outboard of the fuel area with one trunnion attached to the same beam that supports the fuel tanks and the other trunnion attached to a separate structure provided for this support.

The main payload of the Orbiter, with the exception of the PHP, is mounted on the two ends of the Orbiter. These payloads for the various subsystems are mounted directly on a sandwich panel which views free space. Each panel on which payload packages are mounted is provided with quick-release structural fasteners, thus giving ready accessibility to any subsystem unit. Thermal control, both active and passive, is provided in order to maintain a transit temperature of $0^{o}F$ to $100^{o}F$ and an operating temperature of $30^{o}F$ to $100^{o}F$.

Also mounted within the Orbiter, is a separate Image Orthicon Camera provided for use in terminal guidance observation. This camera will take pictures of the target planet and star background and, from this information, final trajectory corrections will be made. A capability is built into the mounting provisions of the camera so that it may be repositioned to required coordinates at any time during the pre-launch period.

Sensors required on the spacecraft are grouped into general terms: fine and coarse sun sensors and star trackers on the Orbiter; earth sensors on the High Gain Antenna; planet sensors on the PHP; temperature sensors for the thermal control shutters; and diagnostic sensors as required.

Two Canopus trackers are installed on the Voyager Orbiter. One tracker is considered "prime" and is used for orientation of the spacecraft when the Southern Hemisphere of Mars is to be mapped. Assuming both launches during the window to be successful, it is desirable to be able to put the second Orbiter in an orbit so that the Northern Hemisphere of Mars can be mapped. By switching to the secondary Canopus tracker, the spacecraft will be oriented in the correct attitude to map the Northern Hemisphere.

5.1.4 MECHANISM ACTUATION

The main method of separation considered is by means of explosive actuators. Pyrotechnic devices will be used to separate the attachments on the high gain antenna, the Planet Horizontal Package, the Magnetometer boom and the Landers.

Actuation of the required components will be provided by means of both spring actuators and motor drives. The PHP and the high gain antenna will operate by means of motor drives and the magnetometer boom will be actuated by means of springs. In addition, the magnetometer boom will have an energy absorption device in order to precisely locate the boom at the point desired.

5.1.5 GROWTH CAPABILITY

Capability for other years and other missions has been one of the prime factors in the Orbiter design. The Orbiter is designed so that dual Landers in the range of about 1,200 to 2,200 lbs. each may be mounted on the Orbiter by means of an adapter. Single Landers can be mounted on the upper surface of the Orbiter above the center of gravity and along the X-axis. An adapter is required which will maintain clearance between the Lander rocket engine and the Orbiter. Single Landers varying in weight from 525 lbs. to 4,000 lbs. can be carried to the selected planet.

Revisions in payload are accommodated by changing packages in the payload mounting area for various missions. Changes in the PHP necessitates only revisions to the attachment fittings and a check of the repackaging in order to keep the c.g. shift to a minimum.

5.1.6 CONSTRUCTION

The Orbiter structure is of semi-monocoque construction, with loads introduced along sheet-stiffened longerons. The choice of semi-monocoque construction was dictated by the expected vibration environment, which by and large is the limiting condition. Additionally, this type of construction enhances thermal control and affords greater flexibility for packaging efficiency. The structure does not have the familiar appearance of the sheet-stiffener shell. This was because simplicity, and hence reliability, was stressed in all phases of design, manufacturing and performance. The goal of simplicity was achieved in design by using simple tension-compression members and avoiding complex load paths wherever possible. Also, elastic shear buckling and diagonal tension was eliminated as a failure mode. This latter condition was deemed necessary in view of the non-linearity effects which would result under dynamic loads. As a result, all primary shear webs are honeycomb panels designed to be shear stable.

Simplicity in manufacturing (where no significant weight savings can be obtained) led to the use of flat rather than curved panels and the use of familiar aircraft materials, fabricating techniques, and fasteners and fastening techniques. The structure appears as a box; however, it should be pointed out that structurally speaking the Orbiter structure is a network of deep beams capable of reacting both vertical and side loads plus overturning moments. The configuration was not designed for volumetric efficiency, but rather for a low silouette and minimum contour dimensions which would still allow two Landers to be supported side-by-side.

An area where manufacturing simplicity has been compromised for structural efficiency has been in the use of high strength alloys in the "foil size" gages. Usually, the only alloys available for thin face sheets (0.012 in. thick) are the softer aluminums with a maximum allowable strength of 20,000 psi. The present design calls for chemical milling of 0.012 in. thick 7075-T6 aluminum sheet (F_{cy} = 60,000 psi) to the thinner gages thereby resulting in honeycomb panels of three times the strength of the equivalent panels used in the past.

5.1.7 THERMAL CONTROL

The temperature control of the major Voyager Orbiter subsystems is achieved by simple, lightweight and reliable means which are feasible in view of present state-of-the-art. Future developments in thermal hardware and in analytical techniques capable of increasing the quality of the thermal design have not been overlooked. An integrated structure-thermal design has evolved which places a minimum of constraints on the spacecraft mission.

Four major subsystems have been considered in the thermal studies: a) Planetary Horizontal Package, b) the Orbiter payload components, c) the fuel tanks, and d) the solar cells.

The Voyager temperature control system utilizes a combined active and passive design concept for the first two sybsystems and an entirely passive one for the last two. The active control consists of thermally-actuated louvers which will be employed to vary the effective emittance of electronic component panels and, therefore, maintain adequate temperature limits under various load rejection levels. The passive control is composed of optical coatings to be applied to particular internal and external surfaces, of multiple reflective radiation shields to minimize heat gains and losses, and of heaters designed to compensate for temperature changes resulting from the continuous decrease in solar input and/or from variable power loads.

The Planetary Horizontal Package has been assumed to be in a dormant state from launch until orbit injection, and will, therefore, have a large portion of its periphery insulated to keep internal heater power requirements to a minimum. The lower temperature of the external PHP surfaces during transit has been established at 0^oF, and their temperatures range in orbit between 30^oF and 100^oF when the PHP is deployed and in operation. In order to meet this range, the non-insulated external surface will consist of louvers, completely closed in transit, but activated in orbit when the PHP components must dissipate energy. Since camera lenses and some scientific instruments are exposed to direct normal solar energy in transit, a fixed sun shield prevents the formation of local hot spots but allows some of the energy absorbed by the shield to be reradiated to the lenses which, in turn, reduces heater requirements.

Payload components will be thermally isolated from the effects of solar distance variation. In order to meet this goal, an insulation shielding will extend over all component surfaces exposed to internal sections of the vehicle; one blanket will cover all components located on one vehicle panel, allowing some radiation heat transfer between black boxes. Heat dissipating components will be mounted on the vehicle skin, and will, therefore, possess one surface facing a near black space environment. This surface will be covered with thermally controlled louvers through which internally generated energy will be discarded. When the equipment is non-operating, the louver will be fully closed, keeping component heat leaks to a minimum. Heaters will supply enough energy to maintain black box skins above 0^oF. When the equipment is operating, base plate temperature limits are 30^oF to 100^oF.

The various tanks which are mounted aboard the Voyager each have their own temperature limitations. To meet these requirements, desirable internal compartment sink temperatures during transit were obtained by proper choice of emittances for Orbiter external side surfaces. Since the heat input to the tanks emanates from the rear surface of the spacecraft's sun side, the changing solar constant in turn affects the compartment sink temperatures as the mission progresses. When a tank lower temperature limit is reached, heaters are turned on to prevent subcooling. Those tanks which eventually will demand heater power will be wrapped in a lightweight insulation blanket to make effective use of this power.

The Voyager Orbiter electrical power is collected by solar cells mounted on the sun side of the spacecraft. To optimize this power output, the cells are oriented normal to the sun's rays during the entire mission except for spacecraft reorientations of short duration. Since the solar cells have a fixed solar absorptance to emittance ratio, their equilibrium temperature will depend largely on the quantity of incident solar energy present at any particular position in space. This temperature level will always be below that resulting when cells are mounted on a panel with an adiabatic back face, as some heat is transferred to the internal compartments of the spacecraft as mentioned previously.

Spacecraft surfaces are coated with materials chosen to exhibit long term stability of their radiative characteristics, and to reduce thermal gradients in a particular subsystem. Coatings of low ratio of solar absorptance to infrared emittance are employed only when no alternative exists, due to their instability when exposed to ultraviolet radiation.

Scientific instruments and electrical components will perform satisfactorily if mounted on a heat sink maintained within the temperature limits of 30^oF to 100^oF. Certain components, such as the communication power amplifiers and sun sensors, have wider limits of permissible temperature; the battery, on the other hand, must be kept between 40^oF and 90^oF during operation.

The high gain antenna will experience a change in temperature as a function of solar distance and must endure large temperature gradients. The use of a low ratio of solar absorptance to emittance coating will tend to reduce the temperature gradients along the antenna honeycomb faces.

5.1.8 ORBITER WEIGHTS

The subsystem weights for the Mars 1969 Orbiter are tabulated on Table 5.1.8-1.

TABLE 5.1.8-1. VOYAGER WEIGHTS, MARS 1969

	(lbs.)	(lbs.)
STRUCTURE		419
Orbiter Structure	316	
Hardware	40	
PHP Structure	57	
Hardware	6	
HARNESSING - VEHICLE		107
POWER SUPPLY		217
Batteries	21	
Electronics	16	
Harness (Solar Array)	7	
Fixed Array	173	
GUIDANCE AND CONTROL		225
Electronics	149	
(FE 14) Tank & Gas	52	
Hardware	24	
COMMUNICATIONS		291
Electronics	259	
Antenna (10' Dish)	32	
DIAGNOSTIC INSTRUMENTATION		30
THERMAL CONTROL		87
PAYLOAD		215
Scientific	91	
TV	124	
PROPULSION		467
Fuel System	364	
Pressurization System	103	
ORBITING WEIGHT		2058

5.2 ENTRY/LANDER

The Entry/Landers for the various Voyager systems are designed to enter the Mars and Venus atmospheres ballistically from an interplanetary transfer trajectory. Although the system requirements specify the landing areas on Mars, the entry/landers for both Venus and Mars are designed to enter at any entry angle from the "skip" limit to a vertical entry. Entry at path angles less than 20 degrees are not desirable because of the very high integrated aerodynamic heating near the capture angle and because of the uncertainties involved in defining the capture angle in the unknown atmospheres.

Although commonality of the Landers for both Venus and Mars was a goal, the penalty proved to be too great. The Venus vehicle is characterized by high entry deceleration, high structural temperatures at impact, rudimentary retardation system, and a cooling type thermal control system. The Mars vehicle is characterized by relatively low entry deceleration, a multi-stage supersonic parachute system, and a heating type thermal control system.

A Photograph of the Mars 1969 Entry/Lander is given in Figure 5.2-1. A summary of the design characteristics for both the Venus and Mars Landers is given in Table 5.2-1.

Both the Mars and Venus systems are designed for minimum descent time consistent with reliability requirements. At other than the limiting conditions (thin Mars atmosphere and vertical entry), a radar altimeter will be used to prevent deployment of the parachute system on the Mars Lander above 30,000 ft. Deployment of the aft cover of the Venus vehicle to gain additional drag will be delayed in a similar fashion until the Lander is within 5000 feet of the surface of the planet. These minimum descent times are necessary to insure line-of-sight between the Lander and Orbiter at the time of impact of the Lander.

TABLE 5.2-1 LANDER DESIGN — CHARACTERISTICS

	Mars			Venus	
	1969	1971	1973 & 1975	1970	1972
Gross Weight, Lbs.	1450	2000	2000	525	2600
Entry Weight, Lbs.	1270	1768	1768	447	2358
Base Diameter, In.	92	108	108	58	123.5
Nose Radius, In.	16.1	16.1	16.1	10.0	21.6
Drag Coefficient	0.78	0.77	0.77	0.65	0.65
W/C_DA, Lbs/Ft^2	35	35.3	35.5	45	44
Length, In.	45.94	55.44	55.44	40.5	71.50
Semi-Cone Angle, Deg.	40	40	40	30	35
Retardation	3 stage parachute			Atmosphere & Aft Cover	

Although the subsonic descent time on the Mars vehicle may be very short, both the Venus and Mars Landers will perform atmospheric experiments after blackout and prior to impact.

All Mars vehicles have been designed using a radioisotope power supply. The radioisotope was chosen because the expected low temperature Mars environment requires vehicle payload heating, and because of the long surface life time required (6 months).

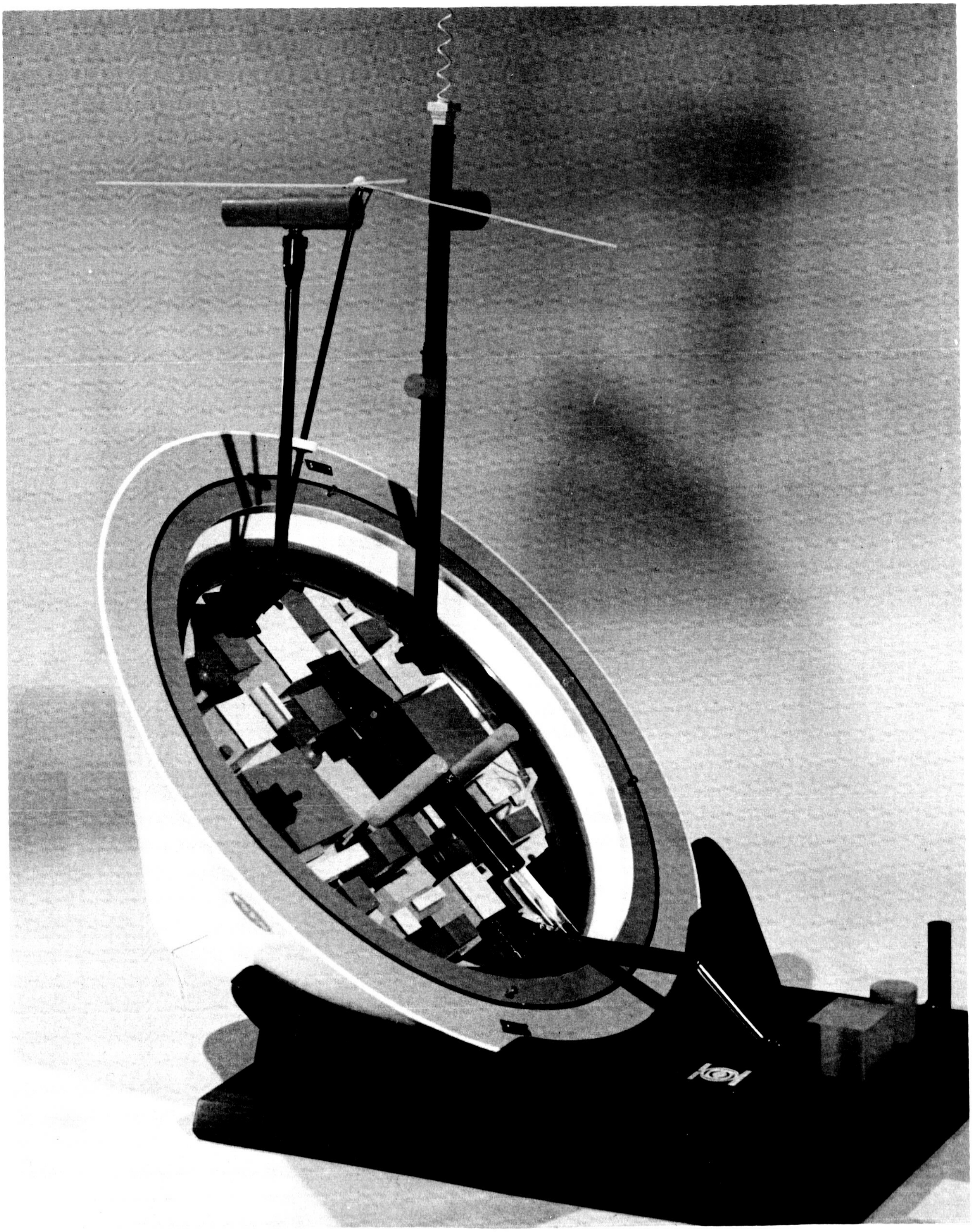

Figure 5.2-1. Mars Entry Lander Deployed

Venus vehicles must survive temperatures of 1050°F on the planet surface. Even after optimization, the weight of the Venus thermal control system is the largest single subsystem weight for a design which permits operation for 4 or 5 hours. Therefore, the 525 pound Entry/Lander for Venus 1970 has only the capability to survive to impact plus a 10-minute life on the surface; at this time the line-of-sight with the orbiter is lost. The larger vehicle in the Venus 1972 system is designed to survive until the orbiter regains communications (line-of-sight) on its first complete orbit (6 hours). All Venus Landers employ an Orbiter relay communication link.

The variation of the maximum deceleration rate with planetary atmosphere is shown for Mars on Figure 5.2-2, and for Venus on Figure 5.2-3. The maximum deceleration loads occurred on the Mars lower atmosphere and the Venus standard atmosphere. Maximum deceleration loads occurred in the model atmosphere, with the steepest density gradient in the 100,000 to 500,000-ft. range.

The maximum design deceleration load for Mars was taken as 125 g's, and for Venus as 325 g's. The vehicles are designed such that these decelerations will be experienced upon surface impact in the thinnest atmosphere; therefore, there is some conservatism in the design from the standpoint of the anticipated entry environment.

For the supersonic parachute system proposed, approximately 17,000 feet will be lost during deployment staging; therefore, to insure that the main stage parachute is fully deployed at 5,000 feet, a Mach number of 2.5 must occur at altitudes higher than 22,000 feet. Figure 5.2-4 shows that the vehicle must have a ballistic coefficient of 35 lbs/ft^2 or less to insure full deployment of the parachute system if the thinnest atmosphere (lower) is encountered.

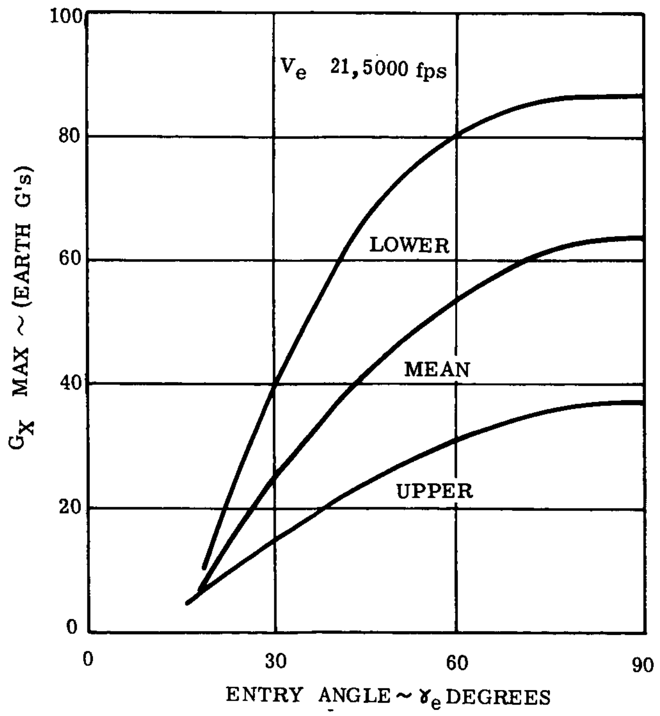

Figure 5.2-2. Peak Axial Deceleration versus Entry Angle for W/C_DA - 35 PSF

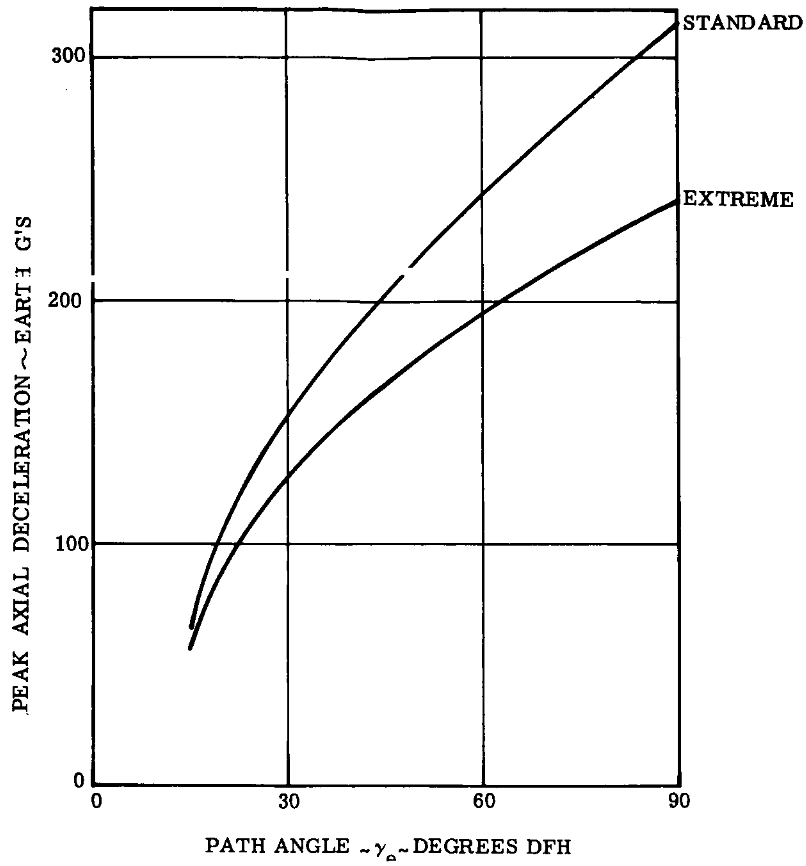

Figure 5.2-3. Peak Axial Deceleration for Two Venusian Atmospheres Where W/C_DA = 40 psf and V = 38,000 Ft/Sec.

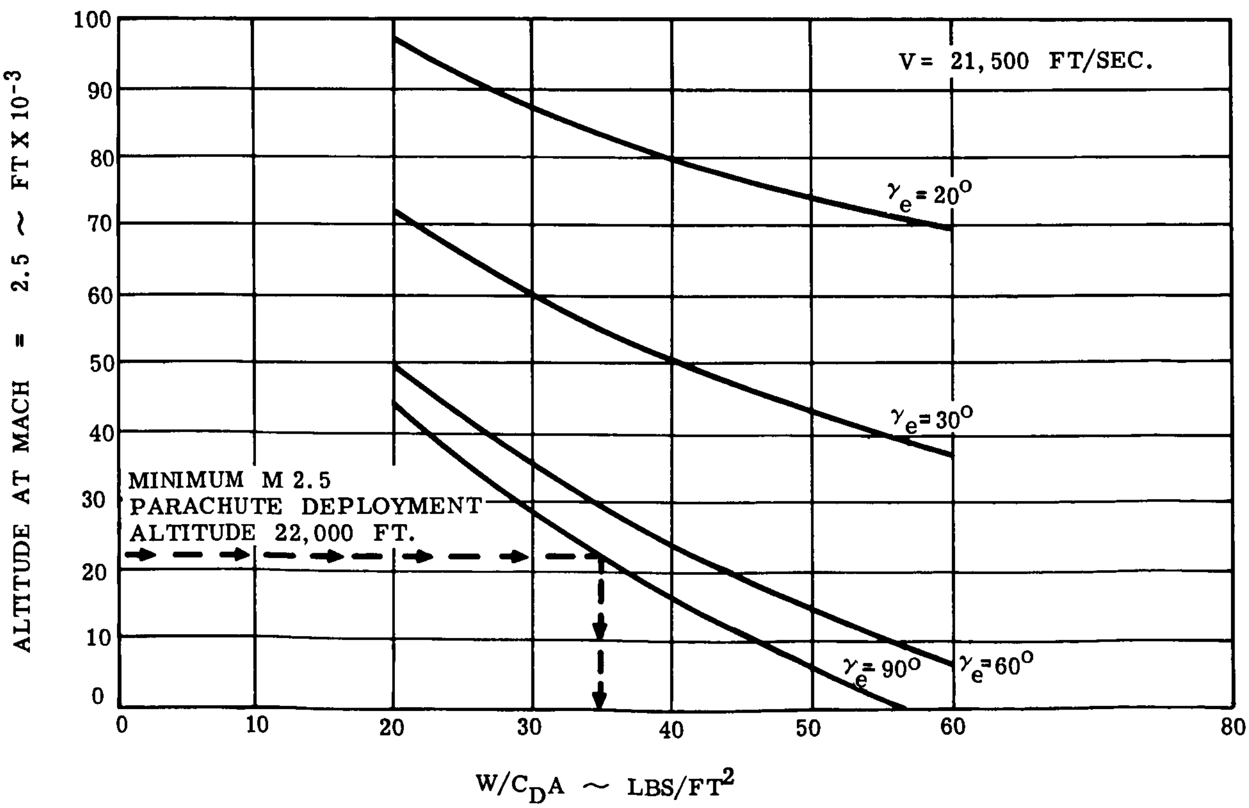

Figure 5.2-4. Altitude of Occurrence at M - 2.5 for Entry Into Mars Lower Atmosphere

5.2.1 VEHICLE CONFIGURATION

Moderately blunt sphere cones were selected for both the Venus and Mars Landers because they avoid the potential dynamic stability problems of very blunt (Apollo) shapes, and because the need for a high-drag shape prohibits the use of very sharp nosed vehicles. This does not represent a penalty in heat shield design since aerothermodynamic analysis has not shown the need for sharp nosed vehicles to avoid radiative heating problems on a steep Venus entry. Thus, the mid-range of sphere cones was selected for this study to fully utilize the wealth of flight experience obtained by General Electric on the RVX, Mark 2, Mark 6, Discoverers, and other programs. A ballistic coefficient of 35 lb/ft^2 was selected for the Mars Lander to insure successful deployment of multi-stage supersonic parachute systems.

A ballistic coefficient of 45 lb/ft^2 was selected for the Venus Lander to insure a fast descent to be consistent with a surface impact velocity of 60 feet per second, and to avoid higher heating rates over the body. The aft cover of the Venus Lander is deployed at 5000 ft. to achieve this impact velocity.

The bluntness ratio, cone angle, and base diameter were selected to achieve the desired weight vehicle, the ballistic coefficient required for retardation, and a reasonable packaging density. A packaging density of 20 lbs. per cubic foot was found to be representative of the scientific payload for Mars and Venus. Cone angles of approximately 40 degrees and bluntness ratios near 0.6 are required for reliable side orientation on the surface of the planet. The optimum bluntness ratio for a given vehicle depends on the location of the vehicle center of gravity, the vehicle cone angle, and the depth of the crushable material in the region of the nosecap. For the type of vehicles chosen, a bluntness ratio of 0.4 to 0.5 is near optimum. However, the vehicles selected have a conservative bluntness ratio of 0.35 which was chosen before the optimization and tradeoff studies were completed.

5.2.2 THERMAL PROTECTION

The thermal protection system for the Venus and Mars Landers must protect the Landers from the typical heating conditions tabulated below:

	Mars Mean Atmosphere		Venus Standard Atmosphere	
Entry Path Angle (degrees)	20	90	15	90
Heating Time (Seconds)	150	25	55	10
Peak Heating Rate Q (BTU/Sec/ft^2)	180	450	1190	2,750
Integrated Heating Q (BTU/Sec/ft^2)	9000	4200	2750	10,200

The shield selected for the Mars Lander is a GE developed Elastomeric Shield Material (ESM). This material was selected because of its high heat of ablation, tolerance to thermal gradients, high insulating properties, and because it appears it can be made transparent to radar. The ESM shield is particularly suited to the long, relatively low, heat pulse encountered on a Mars entry. The elastomeric nature of the material leads to other advantages such as resistance to handling damage and protection against micrometeorite damage during the long transit trajectory. The shield designs contain a 50 percent ablation margin and an insulation layer required to hold the shield bond temperature to 300°F — a limit established for the fiberglass crushable material. The shields are designed to enter at any path angle greater than 20 degrees.

Phenolic nylon was selected for the Venus thermal shield because of the flight-proven characteristics of the material at high-heat loads. Phenolic nylon is a member of an all-organic class of materials whose ablation performance improves with increasing heat rates. Phenolic graphite is selected as an alternate for phenolic nylon because of its potential performance improvement.

The basic construction of the Mars vehicle consists of a heat shield bonded to the fiberglass crush-up structure which is in turn bonded to the aluminum honeycomb sandwich primary structure. As mentioned above, the shield thickness includes insulation to maintain the shield/crush-up bond line temperature below 300°F. The fiberglass honeycomb crush-up structure was selected because of its high energy absorption characteristics, because it can be made radar transparent, and because of its insulating value on the planetary surface. The primary structure can be made from a low temperature lightweight material such as aluminum, magnesium or beryllium. Aluminum was selected over magnesium or beryllium because of cost and state-of-the-art considerations. The honeycomb construction technique was selected because of its weight advantage over other conventional construction techniques, because of the double seal protection offered by the two face sheets, and because the honeycomb is in itself a crushable material.

The Venus Lander is similar to the Mars Lander except that high temperature materials have been substituted for the crush-up structure and the primary structure. The high (1050°F) surface temperatures of Venus require the use of stainless steel or titanium honeycomb for both the crush-up material and the primary structure. Stainless steel was selected in preference to titanium because of state-of-the-art considerations.

5.2.3 RETARDATION

The difference in the Mars and Venus atmospheres is graphically illustrated by the differences in the retardation system of the Venus and Mars entry Landers. The terminal descent velocities at the planet's surface for the Venus and Mars Landers are tabulated as follows for the vehicles without auxiliary drag devices.

	Maximum Terminal Impact Velocity	Minimum Terminal Impact Velocity
Venus (fps)	260	120
Mars (fps)	690	480

The retardation system must reduce the velocity of the vehicle at impact so that the shock attenuation system can absorb the remaining energy. The optimum weight combination of parachutes and shock absorption material occurs at impact velocities of about 70 feet per second. The depth of the crushable material at this impact velocity, however, is approximately 9 inches, which will pose practical problems in manufacturing and achieving an adequate payload volume and/or ballistic coefficient. Pending further study, it is felt that the optimum impact velocity will be in the range of 50 to 60 feet per second.

The parachute system proposed for the Mars vehicle consists of a Mach 2.5 supersonic parachute which is ejected from a mortar tube to provide the first stage. Drag from the parachute is then used to separate the aft cover, extract the main parachute and remove the deployment bag from the main parachute canopy. A second stage of deceleration is then accomplished by the reefed main canopy. Final deceleration is provided after the main parachute is disreefed and the canopy is fully inflated. The Mach number 2.5 was chosen as a conservative estimate of the state-of-the-art in supersonic parachutes.

The thick Venus atmosphere makes possible the use of very simple retardation devices to achieve impact velocities of 50 to 60 feet per second. The retardation of the Venus vehicle is accomplished by deployment of the vehicle aft cover as a drag plate.

The deployment for the Mars landing sequence is initiated by the retardation programmer. The programmer uses a series of "g" switches and timers to sense Mach 2.5 flight speed regardless of the entry angle or the atmosphere encountered. In order to prevent excessive descent times, the radar altimeter is used to prevent deployment of the parachute system at altitudes above 30,000 ft.

5.2.4 SURFACE ORIENTATION

Since the prime objective of the Voyager program is the collection and transmission of surface data from the planets, orientation of scientific equipment and communication antennas is vitally important.

Four basic orientation modes were evaluated for this study: nose-up, nose-down, side with vehicle orientation, and side with payload orientation. The nose-up and nose-down orientation systems were discarded because of the difficulty in bringing the Lander to the final position from any other position it might assume. By limiting the bluntness ratio as a function of the vehicle cone angle, the vehicle can be made unstable in the nose-down position, which makes the vehicle most likely to come to rest on its side. Once the vehicle has come to rest, it is preferable to orient the payload with respect to the ground rather than reorient the vehicle, since by definition the vehicle comes to rest in a minimum energy position.

The orientation sequence is as follows. A position sensor determines the position of the vehicle and then determines the next step in the orientation sequence. If the vehicle is nose down, rockets are used to tip the vehicle on its side; if the vehicle is on its base, the tip bars are extended to tip the vehicle on its side. Once on its side, the aft bulkhead of the vehicle is rotated to the proper position with respect to the ground. The tip bar is then fully deployed until it contacts the ground. Explosive anchors mounted to the tip bars are then fired to stake the vehicle in position.

5.2.5 PACKAGING AND DEPLOYMENT

Packaging studies were conducted on the scientific payloads, the power supplies, and the communication equipment recommended for the various opportunities. Based on past experience in packaging scientific payloads, a maximum permissible packaging density of 20 pounds per cubic foot was used in this study. As mentioned above, the vehicle was varied to maintain the maximum packaging density.

In the course of the study, several important packaging considerations were uncovered. Because of the radiation hazard caused by the cm 244 fuel used in the Radioisotope Thermoelectric Generator (RTG) to personnel, heavy shielding or insertion of the radioactive fuel as one of the last operations prior to launch is required. Insertion of the fuel just prior to launch was chosen since it can be easily accomplished on the launch pad by removal of a nose cap. Therefore, the RTG was located in the nose of the vehicle.

Components that cannot be thermally sterilized must be carefully located to permit post-sterilization installations by remote handling devices or glove-box techniques.

Because of the relative rigidity of the antenna coaxial cables, all communication equipment must be located on the rotating bulkhead to avoid twisting the cables.

The radar altimeter for the Mars vehicle is located in the crush-up material between the shield and structure since both the shield and crushable material can be made radar transparent. The radar altimeters must be deployed on the Venus vehicle because the shield and crush-up structures required for the Venus Lander are not radar transparent.

Equipment requiring contact with the surface of the planet will be deployed with the tip bars.

5.2.6 MARS THERMAL CONTROL SUBSYSTEM

Because of the Radiosiotope power supply, the Mars Lander vehicle requires a thermal control system to be operative during the prelaunch, launch, transit, entry, and surface modes of the mission. The system must serve the dual purpose of cooling the RTG unit and providing heat to the payload within the vehicle as required. Since the high temperature RTG is located inside the vehicle, a coolant loop is necessary to connect the RTG with an external radiator during the transit period. The existence of this loop provides the feasibility of a secondary loop for localized payload temperature control. A schematic of the proposed control system is shown in Figure 5.2.6-1.

In the prelaunch phase, the RTG will be cooled by an externally supplied coolant flowing through the RTG heat exchanger. During the powered flight, while the booster nose fairing covers the Voyager vehicle, RTG cooling will be accomplished by an evaporative heat exchanger using an on-board water supply as the cooling agent. In transit, the RTG coolant releases its heat either to the payload coolant or to the transit radiator which is mounted on the Orbiter-Lander adapter. The flow path of the RTG coolant is controlled to maintain payload temperatures.

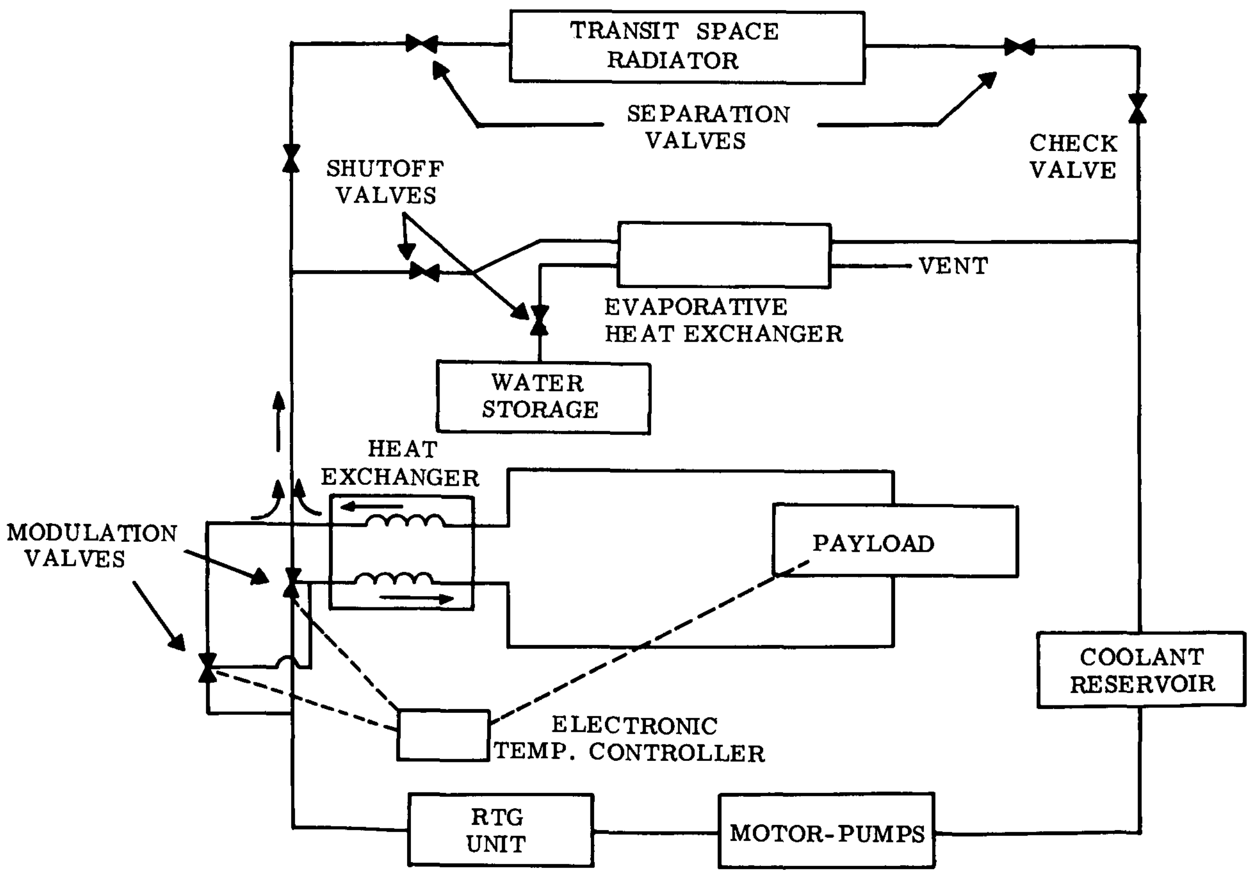

Figure 5.2.6-1. Thermal Control System for Mars Lander

During entry, after the transit radiator has been separated from the Lander, the RTG will again be cooled directly by a water boiler. This allows additional use of the heat exchanger which is used for RTG cooling during the launch phase.

After entry and vehicle orientation, the nose segments are separated from the vehicle to expose the RTG. Cooling is accomplished by radiation of heat to the surrounding environment. Payload thermal control is maintained, as before, through the use of a heat exchanger and closed loop coolant system.

Components (battery, biological experiments) with maximum allowable temperatures below the limits for the electronic and communication equipment are controlled by waxes which attain a two-phase (liquid-solid) state at these maximum temperatures.

Venus Thermal Control Subsystem

The Venus Lander will require transit, entry, and surface thermal control systems. Since system considerations require the Lander to be shade-oriented during transit, electrical heaters using power from the spacecraft solar panels will be used to maintain the internal temperatures between 50 and 100°F. An aluminized mylar insulation blanket over the heat shield will be used to minimize the power requirements for internal heaters and to minimize the thermal stresses that exist between the shield and structure.

On the surface of the planet the hot (1050° F) environment will require a cooling system to maintain the payload within acceptable bounds. Of the four cooling systems evaluated: thermoelectric, heat pump, vapor compression, and expendable phase change working fluid, the expendable system was found to be the lightest.

The choice of the working fluid depends on the design surface pressure. For atmospheric pressures greater than 10 atmospheres, an ice-water system was found best, while for pressures below 10 atmospheres a liquid ammonia proved to be best. In the model atmospheres specified for this study, the ammonia system is recommended. The liquid ammonia system is capable of removing approximately 500 BTU per pound of coolant expended. The Venus cooling system schematic drawing is shown in Figure 5.2.7-1.

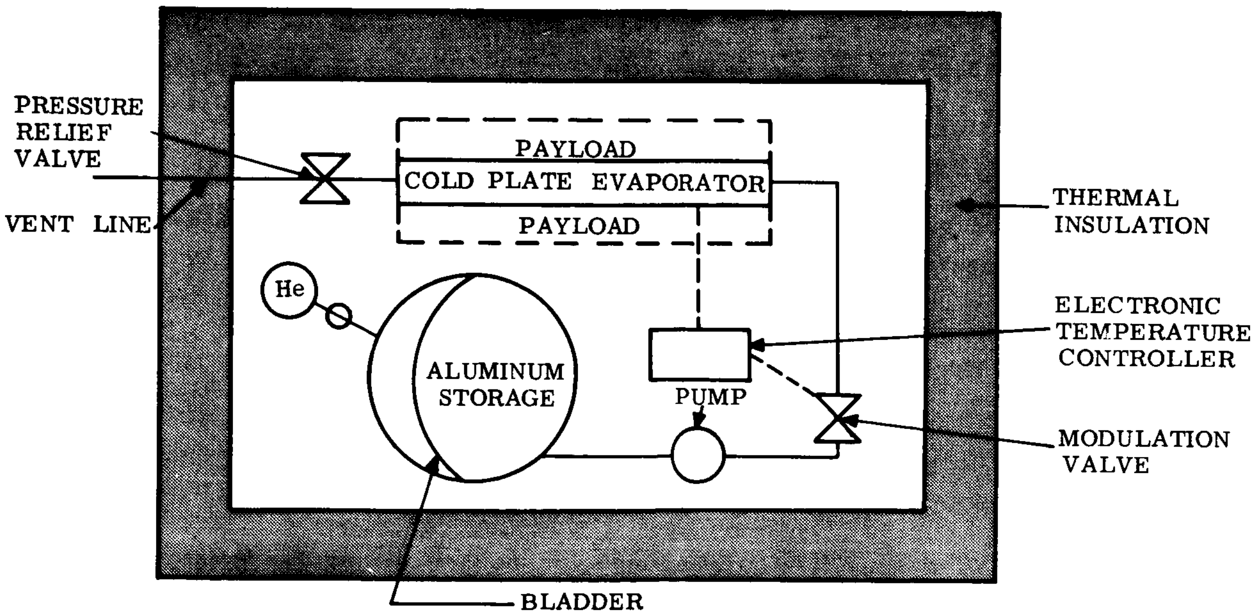

Figure 5.2.7-1. Thermal Control System for Venus Lander

5.3 COMMUNICATIONS SUBSYSTEM

The recommended Communications Subsystem for the Mars 1969 System is described in the following paragraphs. Although differences exist between subsystems recommended for the various missions, the goal has been to utilize the same techniques and components where possible in all subsystems with changes only in subsystem parameter values and component interconnections. These changes can be made in most cases with little or no equipment re-development. Most of the components are either readily available or are in the advanced stages of development.

Only two major components are recommended that are in the early stages of development, an electrostatically focused klystron and a thermoplastic recorder. Although they are described and discussed in detail in subsequent sections, it should be noted here that the versatility of both of these components throughout the various missions makes them extremely attractive. A single development program will suffice for all the mission requirements in each case.

The high gain klystron can be driven with a very low-power (and therefore highly reliable) transmitter. Its saturated power output can be changed across a wide range with little loss in efficiency so that a single tube can be developed for all the power levels anticipated in the Voyager missions.

Thermoplastic recorders appear to offer the ultimate in versatility in high-volume storage devices. Not only can a single recorder be read-in and read-out at bit rates covering an extremely broad range and synchronized with the spacecraft clock, but it can also provide random access as required.

Subsystem versatility is also enhanced by the exclusive utilization of digital techniques. Although the relative performance of analog and digital techniques was debatable in the case of wideband TV data the incorporation of the narrow-band digital data into a hybrid system was found to be cumbersome and relatively inflexible as compared to a completely digital subsystem.

A single modulation and detection technique is utilized in all transmission links. Both data and bit synchronization signals are placed on a single square-wave subcarrier. The two-level composite signal phase-modulates the transmitted carrier between two values, $\pm$ 60 degrees being utilized in most links.

Figure 5.3-1 illustrates the communication links recommended for the Mars 1969 System. Links (1) through (6) are utilized for telemetry and links (7) through (11) are utilized for command. Specifically, each link may be described as follows:

Link (1): Prime data link from Orbiter to Earth utilizing Orbiter high-gain antenna.

Link (2): Secondary data link from Orbiter to Earth utilizing Orbiter omni antenna. To be used during early transit, during maneuvers, and as back-up to link (1).

Link (3): Data link from Lander to Earth utilizing Lander high-gain antenna. To be used as secondary data link if link (5) exists or as prime data link to earth if (5) does not exist.

Link (4): Data link from Lander to Earth utilizing Lander omni antenna. To be used to assist in initial acquisition of link (3) and as a back-up to link (3).

Link (5): Data link from Lander to Orbiter to be used after Lander is on planet surface.

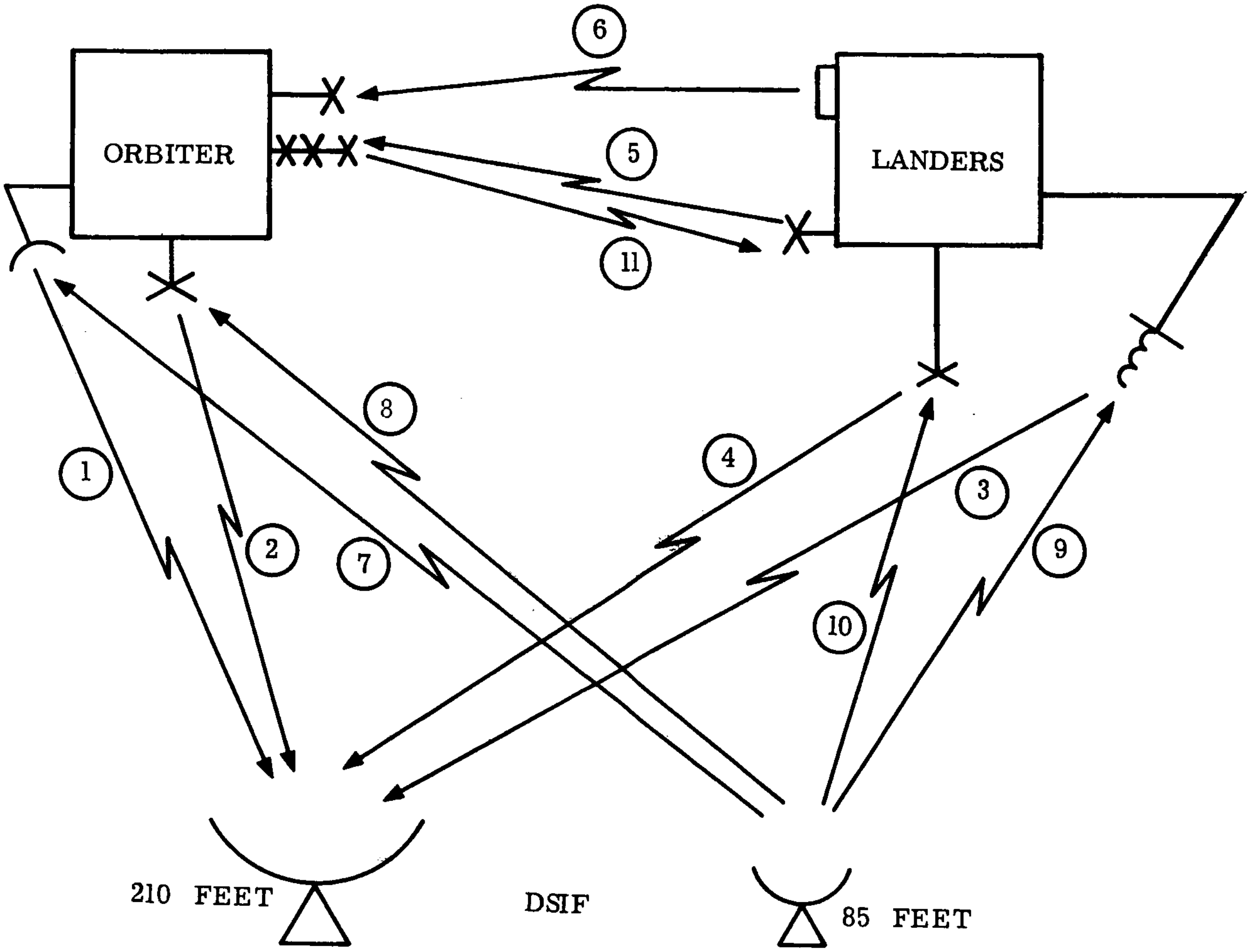

Figure 5.3-1. Mars 1969 and 1971 Communication Links

Link (6): Data link from Lander to Orbiter to be used after Lander separation from Orbiter until Lander impact.

Link (7): Prime command link from Earth to Orbiter utilizing Orbiter high-gain antenna.

Link (8): Secondary command link from Earth to Orbiter utilizing Orbiter omni antenna. To be used during early transit, during maneuvers, and as a back-up to link (7).

Link (9): Command link from Earth to Lander utilizing Lander high-gain antenna. To be used as secondary command link if link (11) exists or as prime command link if (11) does not exist.

Link (10): Command link from Earth to Lander utilizing Lander omni antenna. To be used for initial acquisition of link (3) and as a back-up to link (3).

Link (11): Prime command link from Orbiter to Lander to be used after Lander is on the surface.

The parameters of each of the links are summarized in Table 5.3-1.

5.3.1 DESCRIPTION OF COMMUNICATIONS SUBSYSTEM

A. General

The communications subsystem of each vehicle comprises the following subsystems:

1. Deep space transmission subsystem
2. Relay transmission subsystem
3. Command and computer subsystem
4. Data processing and storage subsystem

Their functions and general implementations are summarized for each mission in the subsequent sections. The early Mars missions are described in the most detail and only the differences are noted for the other missions.

B. Mars 1969 and 1971

(1) Orbiter

Figure 5.3-2 shows the functional block diagram of the Orbiter Communications Subsystem for the Mars 1969 and 1971 missions. The Deep Space Transmission Subsystem provides for transmission of all data from the Orbiter to Earth, reception of commands from Earth and cooperates with the DSIF in the tracking (dopper, angle, and turn-around ranging) of the Spacecraft from Earth. Independent equipments are utilized in the functions associated with the high-gain and low-gain antennas, each utilizing a separate 50-watt klystron for transmission of data to Earth. The high-gain antenna used in the normal mode after early transit is a ten-foot parabolic dish having a beamwidth of three degrees and capable of being pointed with an accuracy of $\pm$ one degree. The low-gain antenna gives nearly omnidirectional coverage except in the meridial plane between the two radiating elements and is used during early transit and as a back-up for the normal mode.

When in the normal mode, the Deep Space Transmission Subsystem is capable of transmitting 16 kilo-bits per second at encounter range. The digital data is combined with a pseudo-noise (PN) sequence on a square-wave subcarrier prior to transmission. This composite signal is used at the receiver to derive bit sync. In addition it moves the sidebands of the transmitted signal away from the RF carrier so that an uncluttered carrier will be available for tracking purposes.

TABLE 5.3-1

Link	1	2	3	4	5	6	7	8	9	10	11
* Purpose	O-E TLM	O-E TLM	L-E TLM	L-E TLM	L-O TLM	L-O TLM	E-O Command	E-O Command	E-L Command	E-L Command	O-L Command
Frequency (mc)	2295	2295	2295	2295	94 96	94 96	2115	2115	2115	2115	105
Power Transmitted (watts)	50	50	70	70	25	25	10,000	10,000 (100,000 backup)	10,000	10,000 (100,000 backup)	5
Transmitting Antenna	10-ft. Dish	Turnstile	(21-db Helix)** (26.7-db Helix Array)***	Turnstile	Turnstile	Transmission line	85-ft. Dish	85-ft. Dish	85-ft. Dish	85-ft. Dish	10-db Yagi
Receiving Antenna	210-ft. Dish	210-ft. Dish	210-ft. Dish	210-ft. Dish	10-db Yagi	Turnstile	10-ft. Dish	Turnstile	(20.3-db Helix)** (26-db Helix Array)***	Turnstile	Turnstile
Receiver Noise Figure	--	--	--	--	4-db	4-db	10-db	5-db	10-db	10-db	4-db
Receiving System Noise Temp. (^oK)	35	35	35	35	--	--	--	--	--	--	--
Probability of Bit Error	1.4×10^{-3}	1.4×10^{-3}	1.4×10^{-3}	1.4×10^{-3}	10^{-3}	10^{-3}	10^{-5}	10^{-5}	10^{-5}	10^{-5}	10^{-5}
Transmitting Ant. Pointing Error (Degrees)	1.0	--	3.0	--	--	--	Neg.	Neg.	Neg.	Neg.	Function of Orbital Altitude
Receiving Ant. Pointing Error (Degrees)	Neg.	Neg.	Neg.	Neg.	Function of Orbital Altitude	--	1.0	--	3.0	--	--

*O — Orbiter
L — Lander
E — Earth

**Mars 1969 only

***Mars 1971 only

In the normal command mode, commands are transmitted from the Earth using the 85-foot DSIF antenna and 10-kw transmitter, and reception is through the high-gain antenna after early transit. As a back-up mode, reception is through the omni, and the 100-kw transmitters must be used at the longer ranges.

The Orbiter is visible full-time from the Earth for the first 30 days after orbit insertion. Also, it is within line of sight of the sun during that period. Continuous transmission to and from the vehicle is therefore allowable.

The command word unit accepts digital data and associated sync pulses from the command detector when a lock signal is received; otherwise, it will not accept or act on any data. The command word unit interprets the word-start symbols, determines its destination, verifies the validity of the received data and, if accepted, delivers real-time data to the Command Execution and Computation Unit and stored data to the Memory Unit. All commands to the Lander are stored until another command, either real-time or stored, directs a read-out to the Lander through the Relay Transmission Subsystem.

The Command Execution and Computation Unit executes all real-time commands upon reception. It also selects the command in the Memory Unit to be executed next and holds it in a register until its time label coincides with that of the spacecraft clock. It then executes the command and selects the next command from the memory to be executed and holds it in the register unitl executed. This process is repeated until all commands in the memory have been executed. Such a technique minimizes the number of times the memory must be interrogated and therefore minimizes the probability of producing an error in the process. A parity check is also made before a stored command is executed, thereby further reducing the probability of initiating an incorrect command. Both quantitative and discrete (on-off) commands are initiated by the Command Execution and Computation Unit. This unit in conjunction with the Memory Unit forms a special-purpose computer which can be used to compute such things as mapping time sequences upon receipt of the appropriate coefficients from Earth.

The clock is the central time reference for the spacecraft. It provides a time label and timing pulses for all subsystems as required. The time label is used to determine the time at which a command is to be executed and also is inserted into each frame of data being taken by the Data Processing Unit.

To minimize the number of lines to the Planet Horizontal Package (PHP) a separate Decoder and Power Conversion and Control Unit are utilized on the PHP. Only the data and control lines are therefore required.

The Data Processing and Storage Subsystem has four different functions:

1. Digitize and multiplex data
2. Store data
3. Encode data for error control
4. Generate bit sync signal

The first function applies only to the narrow-band data sensors as used in most cases for both science and engineering data. Wide-band data such as TV is encoded by an A/D encoder within the TV subsystem and separate from that used for the narrow-band data. Multiplexing of TV data with narrow-band data is directed by the Command Subsystem. The data format and rate are also determined by the Command Subsystem. The format determines which sensors are sampled in a particular frame. For instance, during maneuvers only selected diagnostic sensors will be sampled, while during orbit most of the data collected will be scientific. The data collection rate will be commanded from earth, based on the anticipated rate of change of sensor outputs and will be constrained by the rate at which data can be sent to earth over an available time period. The narrow-band

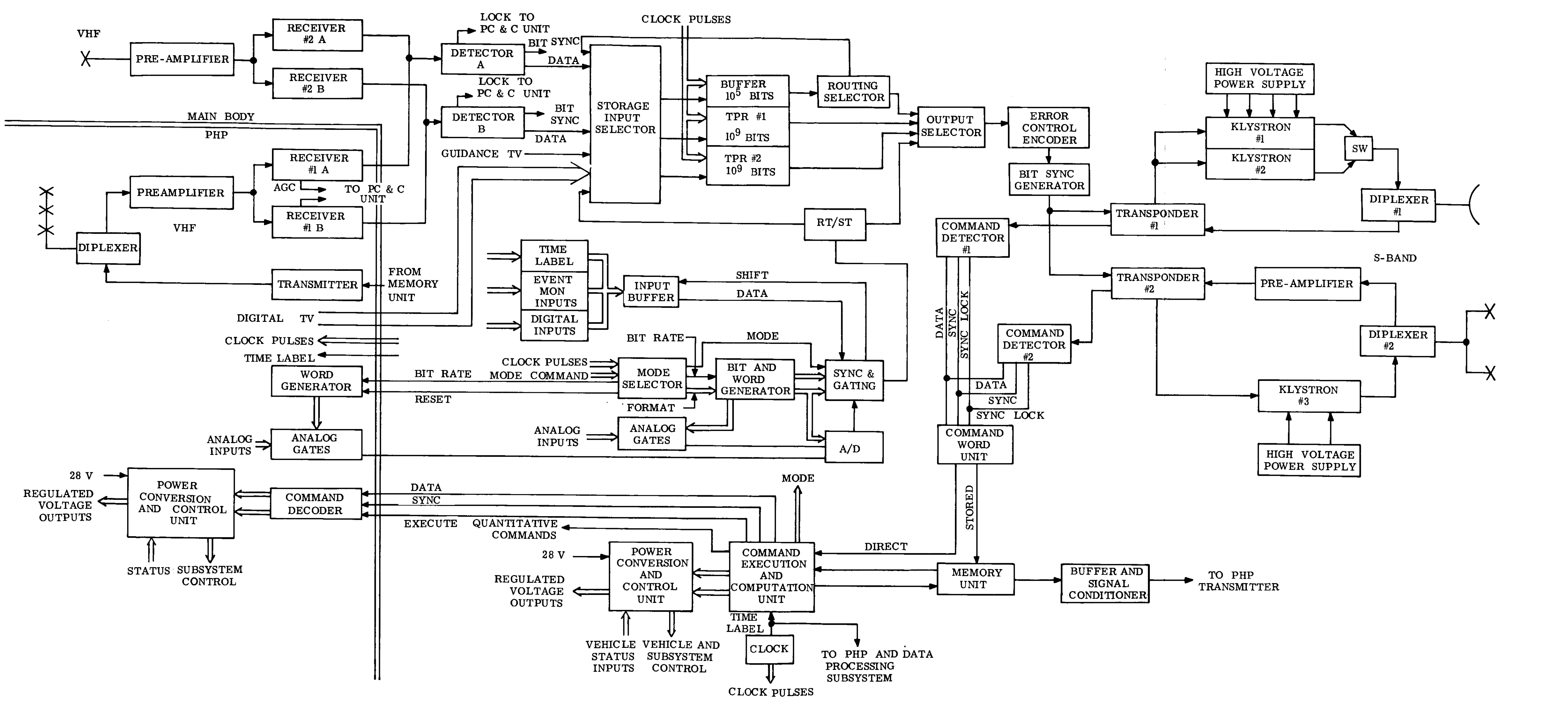

Figure 5.3-2. Mars 1969 and 1971 Orbiter Communications Subsystem

data can be either stored or transmitted directly. The storage devices utilized are a 100-kilo-bit plated-wire storage and two 10^9-bit thermoplastic plate recorders. Because of their ability to record over an extremely wide range of data rates, the TPR's could probably be utilized for all recording and the plated-wire storage eliminated; however, there are periods of time such as during maneuvers in which it is desirable to record low-rate data with a minimum of power expenditure. In this case the plated-wire storage can be utilized with a power requirement of milliwatts, as opposed to the TPR's which would require about 25 watts. The TPR's would then be activated only for short periods of time if required to accept data from the wire storage when the latter is filled.

One of the primary features of the TPR's is that data can be clocked in and out. This means that during read-in a constant number of bits are stored on each line of a TPR plate regardless of the read-in rate. If magnetic tape recorders were used, either the tape speed would have to be changed for each new rate, or a tape speed higher than that actually required would be necessary. The ability to read-in at various speeds in a tape recorder can result in considerable waste of storage volume. The disadvantage of start-stop time is also eliminated in the TPR's. This is especially important during the in-orbit mapping procedure, since the frame rate is not constant.

The ability to clock-out the data also results in ease of implementation by automatically satisfying the requirements imposed by the error control encoding and bit sync techniques. The Error Control Encoder requires that a burst of 45 bits be read into the encoder at the transmitted bit rate and that no data be read-in during the subsequent 28 bit periods. The TPR output can be presented in this fashion without the use of buffers, whereas a magnetic tape recorder output must first be fed into buffers at 45/73 times the transmitted bit rate and the buffers clocked out in bursts as described previously. This would present a problem in that the data rate from the tape must be extremely close to that required so that the buffers do not overflow or become empty. Running the tape at a higher speed into a high-volume buffer and stopping it when the buffer filled would alleviate this problem; however, this again would result in added complexity and reduced reliability.

If the Error Control Encoder were not in the system, or if it were by-passed during portions of the mission, the bit sync requirements still make clocked data desirable. The more precisely the PN sequence is combined in phase with the data and the more stable the bit rate, the better the detection capability at the receiver. Therefore, for tape recorders either the inferior detection capability must be accepted or a buffer must still be utilized to allow clocking out the data.

The TPR's also accept data directly from the TV Subsystem and the Lander Data Detector. The TV sampling and digital encoding processes as well as the TPR read-in are controlled by synchronous clock pulses from the Command Subsystem. Recording of data from the Lander Data Detectors is clocked by the bit sync pulses from the detectors.

The Error Control Encoder, a unit of the Data Processing and Storage Subsystem, accepts bursts of 45 bits as described previously and computes and appends 28 check bits in a cyclical register. Its output to the bit sync generator is then a serial string of 73 bits. Approximately 1.5 db reduction of required transmitter power is accomplished by the error control encoding.

The Bit Sync Generator combines a 511-bit PN sequence on a square-wave subcarrier with the 73 data bits. This allows seven PN bits per data bit. At the receiver, the subcarrier and PN sequence are cross-correlated with identical locally generated waveforms. When the two PN sequences are in phase or correlated, the PN generator in the receiver provides outputs indicating the beginning of each data bit period and the beginning of each group of 73 bits. The former output allows accurate detection of each bit in an integrate-and-dump circuit, while the latter resets the error control decoder each time a group of 73 bits is decoded.

The composite signal from the Bit Sync Generator is a two-level waveform. This signal is used to phase-modulate the carrier generated in the transmitter portion of the transponder. The carrier is shifted therefore between two values of phase (± 60 degrees utilized in this case) resulting in a spectrum with a discrete carrier frequency and sidebands containing the data and synchronization information. The sidebands are sufficiently removed in frequency from the carrier so that the spectrum is relatively uncluttered near the carrier as required for tracking.

The Relay Transmission Subsystem on the Orbiter is divided between the Main Body and the PHP. The detectors in the Main Body are used to detect all received data, since only one of the sets of receivers will be utilized at one time. The receivers in the Main Body are fed by a turnstile antenna. They are used only during the cruise and descent phases of the Landers. The receivers in the PHP cannot be used at that time, since the PHP is stowed until after retro firing for orbit insertion, which is not accomplished until the Lander has completed or nearly completed its descent phase. The turnstile antenna is located on the Main Body so as to give complete coverage of the planet when the Orbiter is in the retrofire attitude.

After orbit insertion the PHP is deployed, and a 10-db yagi on the PHP is extended toward the planet. Reception thereafter is through this antenna and the receivers on the PHP, although the omni can serve as back-up.

The modulation technique utilized in all relay links is PCM/PSK (± 60 degrees). Synchronous reception and matched-filter data detection are also used. Bit sync is similar to that described for the deep-space links; however, faster lock is attained by reducing the length of the PN sequence used for bit sync to three bits in the Lander-Orbiter telemetry links and 31 bits in the Orbiter-Lander command link. The PN sequence is repeated each data bit rather than for a group of 73 bits as described for the deep-space links. The latter is not a requirement in the relay links since error control encoding is not used.

Each Lander has its own associated receiver in the PHP and a receiver and detector in the Main Body of the Orbiter. The Landers transmit on separate frequencies, since there are periods when they must operate simultaneously. Transmission to the Landers is on a single frequency, and all commands are received by both vehicles if they are within line of sight of the Orbiter. Each command contains an address that indicates which Lander is to accept the command. The five-watt command transmitter in the PHP is modulated by commands sent from the Command & Computer Subsystem. The Command & Computer Subsystem also initiates and controls the lock procedure which ensures both carrier and bit sync lock in the relay links. Carrier-lock signals from the PHP receivers and sync-lock signals from the data detectors are sent to the Command, and Computer Subsystem as a part of this procedure.

(2) Lander

The functional block diagram of the Lander Communications Subsystem for the Mars 1969 and 1971 missions is shown in Figure 5.3-3. It comprises the same four subsystems (Deep Space Transmission, Command, and Computer Data Processing, and Relay Transmission). However, each subsystem differs to a certain extent from its counterpart in the Orbiter.

The Deep Space Transmission Subsystem has been relegated to a secondary role in the Lander and is a reduced version of that in the Orbiter. Only one 70-watt klystron, one transponder and one command detector have been used. Transmission and reception can be switched between a low-gain turnstile antenna and a high-gain antenna. The high-gain antenna is a 21-db helix for the 1969 mission and 26.7-db helix array for the 1971 mission. Each of these antennas is to be pointed to Earth with an accuracy of ± three degrees.

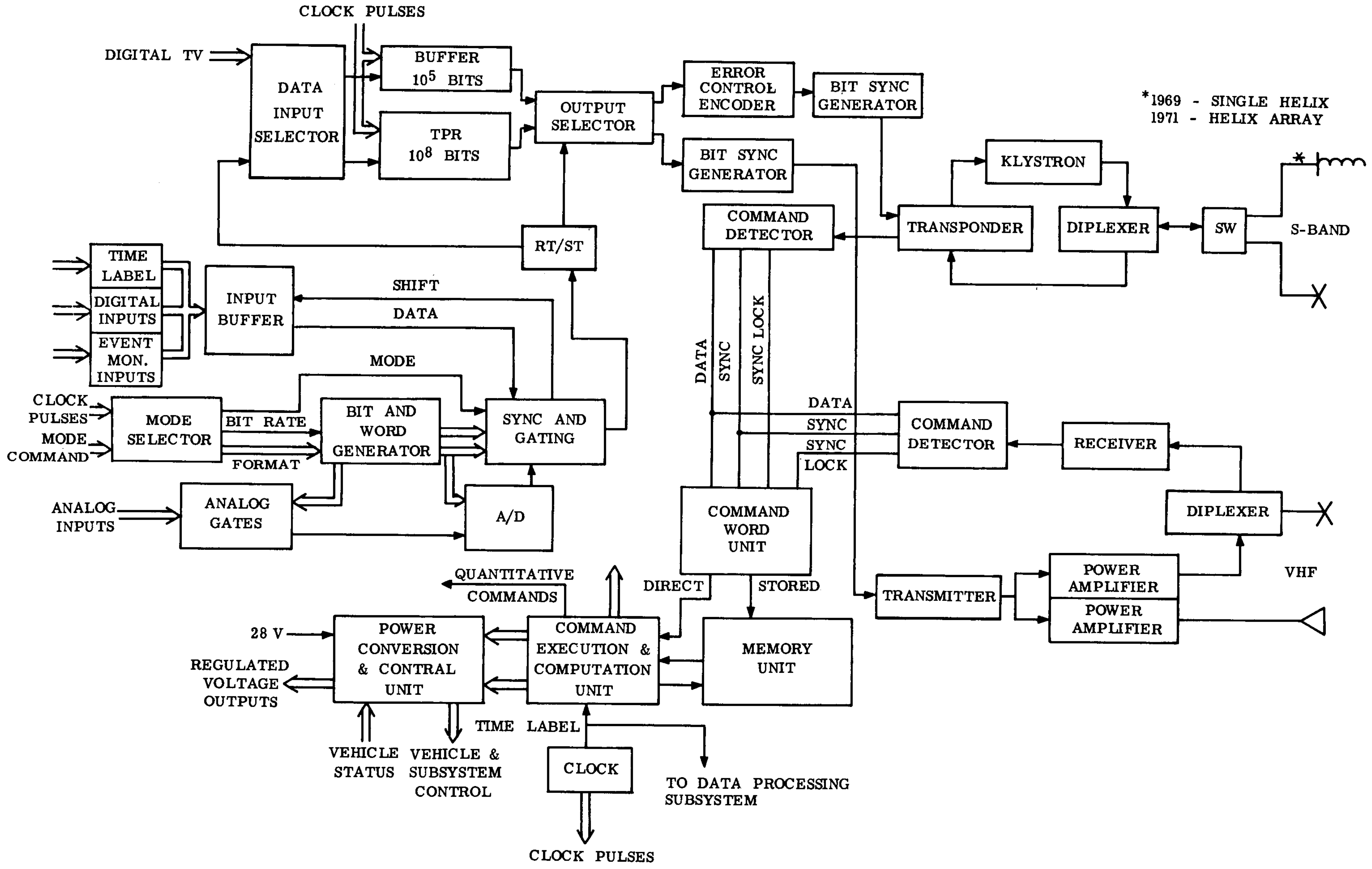

Figure 5.3-3. Mars 1969 and 1971 Lander Communications Subsystem

The prime data links to and from the Landers are provided by the Relay Transmission Subsystem. Link (6) as described previously is used during the Lander cruise and descent phases. A separate power amplifier and antenna are used in the Lander for this link. The "transmission line" antenna will survive entry but can be damaged upon landing. Therefore, a turnstile antenna is erected after the Lander is on the surface and the aft cover has been removed. Individual power amplifiers are utilized to eliminate RF switching.

The Lander Command and Computer Subsystem function is reduced from that in the Orbiter, in that the commands to the PHP and those relayed through the Orbiter to the Lander are not required in this case. The Memory Unit capacity and the number of command words are also reduced.

The number of inputs to the Data Processing and Storage Subsystem is also reduced, and, since transmission can occur for only one-half hour every six hours because of energy storage limitation, the data storage volume is reduced. A plated-wire storage and a thermoplastic recorder provide capacities of 10^5 and 10^8 bits, respectively. The small plated-wire storage is used primarily for narrow-band data and data accumulated during entry and descent. Actuation of the TPR is not required until after the Lander has reached the planet surface.

Only the direct links to Earth utilize error control encoding. When data is being transmitted via the Relay Transmission Subsystem to the Orbiter, the encoder is bypassed.

C. Mars 1973 and 1975

(1) Orbiter

The block diagram of the Orbiter Communication Subsystem for the 1973 and 1975 missions is identical to that described for the 1969 and 1971 missions with the elimination of some components and the reduction of radiated power. In particular, the PHP and its associated electronics have been removed, only one TPR is used, and the S-band pre-amplifier has been eliminated for the 1973 mission, because no pre-amplifier has been found capable of meeting the heat sterilization requirements. Each klystron here radiates only 35 watts, instead of 50.

The relay receivers are used only during the Lander cruise and descent phases to monitor the Lander before it reaches the surface. After landing, the Lander Deep Space Transmission Subsystem provides all communications. There is no command transmission from the Orbiter to the Lander at any time.

(2) Lander

Because the Deep Space Transmission Subsystem provides the prime link to and from earth, it is constructed in two independent sections to increase its reliability - one section is associated with the high-gain antenna and the other with the low-gain antenna. A back-up klystron, which can be switched into the high-gain section, is provided. The high-gain antenna is a helix array giving a gain of approximately 26.7 db. This antenna is to be pointed at the earth with an accuracy of ± three degrees.

The Relay Transmission Subsystem comprises only a VHF transmitter, power amplifier, and a "transmission line" antenna. Upon Lander impact, the latter is discarded, and the relay link can no longer be utilized.

D. Venus 1970

(1) Orbiter

The Deep Space Transmission and Data Processing and Storage Subsystems are identical to those described for the Mars 1969, and 1971 missions. The Command Subsystem differs only in that no commands are relayed to the Lander.

In the Relay Transmission Subsystem, a 10-db planet-tracking yagi antenna is used to receive data from the single Lander both before and after impact. Communication time after Lander impact is expected to be in the order of 10 to 30 minutes. The Lander will be dead on the second orbit; therefore, the yagi pointing need be programmed only for the first pass.

(2) Lander

This subsystem has no Deep Space Transmission Subsystem and therefore depends solely on the relay links to retrieve cruise, descent, and surface data. A "transmission line" antenna is used during the first two phases but is disconnected upon impact, and a whip antenna is used for the surface phase.

Only a 10^5-bit plated-wire storage is used in the Lander. This will allow storage of entry data. Because TV pictures must be taken and transmitted in a short period of time, the TV camera itself will be used for short-term storage, and the picture will be transmitted directly as it is scanned from the targe element of the camera tube.

Because there is no radio command capability to the Lander here, the Command Subsystem takes the form of a Sequence Timer. It is pre-programmed either on the earth or through the Orbiter Command Subsystem prior to separation. It also accepts signals such as that from a "g"-switch which initiates programs during and after entry.

E. Venus 1972

(1) Orbiter

It is identical to that described for the Venus 1970 mission, except that the klystrons radiate only 35 watts instead of 50 and it provides for command transmission to the single Lander. The 10-db yagi is body-mounted and is programmed to point in the direction of the Lander both on the first pass and after completion of the first orbit. The program can be updated in the latter case after the parameters of the attained orbit has been determined by the tracking station. The Lander will be designed for only a one-orbit lifetime, and the relay link will be de-activated after that time.

Commands will be sent to the Lander only when the Lander comes within line-of-sight at the completion of the first orbit. This allows time for transmission to earth and evaluation of the pictures taken on the first pass. The commands will, therefore, be based on information obtained from the first pictures.

(2) Lander

The Venus 1972 Lander Communication Subsystem is most nearly like that described for the Mars 1969 and 1971 Landers. The two subsystems differ in that the Venus 1972 Lander does not have a Deep Space Transmission Subsystem. The other subsystems are identical.

5.3.2 CRITICAL PROBLEM AREAS

A. Electrostatically Focused Klystron (ESFK)

The electrostatically focused klystron has been selected for the S-band power amplifier because of its favorable characteristics in the categories of weight, gain, efficiency, and predicted lifetime. To date only prototype units have been built and tested. A program for further development, qualification, and life-testing will be necessary. It is estimated that this will require about two years plus life-test.

B. Thermoplastic Recorder (TPR)

Thermoplastic recorders have been selected for both the Lander and the Orbiter because of their high data storage density, ease of operation and lack of rotating parts. It is also expected that these will prove to be more amenable to heat sterilization than magnetic tape.

A breadboard model of a thermoplastic recorder is presently being developed at the GE Advanced Technologies Laboratory under contract to JPL. Rather extensive development and testing will be required to qualify this for spacecraft use, but the time schedule will allow this. About three years plus life-test should suffice.

C. Sterilization of Magnetic Tape

In the event that unforeseen difficulties arise in the development of the thermoplastic recorder, it is recommended that a magnetic tape recorder development program be started. Heat sterilization of magnetic tape will cause material problems, thus requiring the development of a tape, perhaps metal backed. Several companies claim to be studying this problem already, and such programs should be continued. About a two-year development program will be required.

5.4 TELEVISION SUBSYSTEM

In general, the mission for the Voyager television is to obtain biological and geological information about Mars, and information about cloud movements and possibly geological features of Venus. The television mission for the various vehicles and mission opportunities is given in Tables 5.4-1 and 5.4-2; Table 5.4-3 lists the systems constraints imposed on the television subsystem in the form of orbital geometry and transmission bandwidths. The characteristics that are required of the television camera to perform these missions are listed in Table 5.4-4.

5.4.1 ORBITER TELEVISION

The daylight portions of the Martian surface and the Venus cloud cover are to be mapped by television cameras having various resolutions installed in an Orbiter. The Mars Orbiter television cameras are designed to provide optical resolutions of 1 km, 140 m (in color), and 20 m at the periapsis. The low resolution cameras provide a stero pair having a height resolution of 345 m. The Venus cloud cover will be mapped with an optical resolution of 2 km at periapsis.

5.4.2 LANDER TELEVISION

The Mars Landers are equipped with one television camera with steerable optics such that clouds, the horizon, and the terrain in the immediate vicinity of the landing site can be scanned through 360 degrees during daylight hours. A television camera attached to a microscope is also provided for examination of soil samples and for planned biological experiments. The panoramic camera will resolve 3 minutes of arc (in color); the microscope will resolve 1 μ, 5 μ, and 50 μ (in color). The Venus Lander television will provide infrared surface information during descent and limited panoramic surface coverage in the visible spectrum, using flash illumination. A microscope has been incorporated only in the longer-life Venus 1972 Lander.

5.4.3 RESOLUTION PARAMETERS

For optimum bandwidth utilization, four bits per sample has been chosen in the Orbiter digital television cameras while the tube raster contains the maximum number of resolvable lines (512 for a one-inch vidicon and 1,024 for a two-inch image orthicon). The four-bit quantization was selected after studies including study of photo-interpretation techniques and in consideration of the low resolution obtainable. The number of raster lines was made to maximize the field of view. In the Lander television, full tonal rendition (6 bits per sample) seems necessary, while 256 lines per raster provides a reasonable field of view (about 4 1/2 degrees).

5.4.4 CAMERAS

Since the vidicon is an inherently simple and rugged camera tube which has been used previously in space applications and can be built to withstand heat sterilization, it is used where practicable in the recommended subsystem. Although the image orthicon does not offer these features, it is recommended for the medium - and high-resolution Orbiter cameras, since its high sensitivity allows the use of much smaller lenses. The minimum signal-to-noise current ratio in the camera video signal has been set at 35. The slow-scan vidicon was analyzed and an appropriate derating factor was found to account for the long frame times necessary at Voyager bandwidths. The tube was considered noiseless. All noise was considered as originating in the pre-amplifier. The sensitivity at 3-second frame rates was calculated to be approximately 0.33 foot-candle-second.

The sensitivity of an image orthicon at a signal-to-noise current ratio of 35 was found to be approximately 3.3×10^{-4} foot-candle-second. The noise originating at the photocathode, the target, the first dynode, and in the beam was considered to be the major noise contribution in the system.

The dependence of the signal-to-noise ratio on the scan velocity indicates that the dwell time of the beam on each picture element should be minimized while the frame time remains long. A digital scan, therefore, is recommended. In this type of scan, the beam remains only a short time on the element to be sensed and then returns to a dormant part of the target.

Special automatic control circuits are needed to operate the cameras without adjustments over a long period of time. Automatic vidicon cameras have already been developed. Self-adjusting image orthicon cameras are now being designed by the Hazeltine Corporation and the General Electric Advanced Electronics Center. Highlight determination, using the camera tube as a sensor, and protection of the tube face from direct sunlight will also be accomplished. A computing circuit designed for Project Mariner is selected for highlight determination. A separate sun sensor will be incorporated for sunlight protection.

5.4.5 OPTICS

Optical systems have been calculated for the various vehicles and missions. A simple telescopic lens was found sufficient for the low resolution Orbiter stereo cameras. Maksutov folded optics are selected for the medium-and-high-resolution Orbiter cameras. A double Gaussian type lens is selected for the Lander panoramic television. The microscope and the infrared descent optics are also state-of-the-art design.

5.4.6 STEREO

The height resolution of the stereo cameras was calculated using empirical factors obtained from the experimental data of photo-interpretation experience. The 1-km resolution cameras will resolve 345 meters at a canting angle of 20 degrees to the local vertical. This height resolution is to be interpreted as the ability of the television system to deliver stereoscopic pictures on which spot height differences of 345 meters can be recognized with 95 percent confidence while lesser heights cannot be determined. It is expected that a general physiographic map of the planet can be assembled from the information obtained.

5.4.7 ARTIFICIAL ILLUMINATION FOR VENUS

Artificial illumination is considered necessary to obtain television pictures from the Venus Lander in the visible spectrum. Electronic flash equipment and chemical flares were investigated for this mission; both appear feasible. The electronic flash equipment is recommended, however, on the basis of repeatability, reliability, and the uncertain atmospheric information available for the development of usable chemical flares.

TABLE 5.4-1. MARS TELEVISION MISSION

	Mars 1969 & 1971 Orbiters*				Mars 1969, 1971, 1973 & 1975 Landers	
Function	Map	Map	Map	Map	Panorama	Microscope
Number of Cameras	2	1	3	1	1	1
Optical Resolution	1 Km	1 Km	140 m	20 m	3 Min of Arc	1 μ, 5 μ, 50 μ
Stereo	345 m	no	no	no	no	no
Color	no	no	yes	no	3 filters	3 filters

*The Mars 1973 ORBITER and 1975 fly-by do not carry TV

TABLE 5.4-2. VENUS TELEVISION MISSION

	Venus 1970 & 1972 Orbiters	Venus 1970 Lander		Venus 1972 Lander		
Function	Map Clouds	Descent	Panorama	Descent	Panorama	Microscope
Number of Cameras	1	1	1	1	1	1
Optical Resolution	2 Km	1 Min of Arc Max	3 Min of Arc	1 Min of Arc Max	2 Min of Arc	1 μ, 5 μ, 50 μ
Stereo	no	no	no	no	no	no
Color	no	no	3 Filters	no	3 Filters	3 Filters

TABLE 5.4-3. VOYAGER SYSTEMS CONSTRAINTS

	Mars 1969	Mars 1971	Venus 1970	Venus 1972
Orbit (nm)	1000 x 19,000	1000 x 19,000	1000 x 4300	1000 x 7300
Inclination	55°S	45°N	68°	90°
Number of Recorders	2	2	2	1
Recorder Input	300 kbps	300 kbps	100 kbps	100 kbps
Transmission Bandwidth	2-16 kbps	2-16 kbps	8-64 kbps	2-16 kbps

TABLE 5.4-4. TELEVISION CAMERA CHARACTERISTICS

Module	Camera	No. of Lines	No. of Bits/Sample	No. of Bits/Frame
Orbiters	Low Resolution (Vidicon)	512	4	1,048,576 + Sync and Ident $\approx 1.1 \times 10^6$
	Medium Resolution (Image Orthicon)	1024	4	4,194,304 + Sync and Ident $\approx 4.3 \times 10^6$
	High Resolution (Image Orthicon)	1024	6	6,514,584 + Sync and Ident $\approx 6.6 \times 10^6$
Landers	Descent (Vidicon)	512	6	1,572,864 + Sync and Ident $\approx 1.7 \times 10^6$
	Panorama & Microscope (Vidicon)	256	6	393,216 + Sync and Ident $\approx 4.2 \times 10^5$

5.5 RADAR

The presence of extensive cloud cover about the planet Venus makes it expedient to obtain information about the planet by the use of frequencies outside the optical portion of the spectrum. In particular, the microwave and UHF frequencies are attractive.

The following information can be obtained from sensors on an Orbiter about Venus:

a. A topographic map* of the terrain can be generated for all or part of the planet's surface.

b. Surface characteristics can be determined for particular areas on the planet.

c. The surface temperature can be determined by radiometry for particular areas.

d. The height of the ionosphere and the electron density can be measured.

e. The altitude of the Orbiter above the surface of the planet can be measured to aid in establishing the actual ephemeris of the Orbiter.

The Mars and Venus Landers will use radar altimeters to give height above terrain to be used for the correlation of scientific measurements made during descent. The Venus Lander will also carry doppler radar to measure the effect of winds. These equipments are the same as those used for Surveyor with minor modifications.

5.5.1 TERRAIN MAPPING

The methods of terrain mapping which were considered would generate narrow strip maps which could be combined into a continuous terrain map of the planet. The planetary map will provide a basic reference against which all subsequent atmospheric and surface measurements may be evaluated. It will show areas of differing radar reflectivity, and will show radar shadows. The total effect is similar to that obtained by a black and white aerial photograph with the sun casting distinct shadows. The terrain mapping radar is considered to be the primary sensor for the Venus 1970 Orbiter.

Two synthetic aperture radars were compared for this application. A synthetic aperture radar obtains by means of special signal processing, a finer resolution than that obtained by virtue of the antenna beamwidth in a conventional radar. The SAHARA (Synthetic Aperture High Altitude Radar) radar designed by the Light Military Electronics Department of General Electric Co., is compared with a coherent pulse doppler radar designed by the Conductron Corporation. The pulse doppler radar extracts the radar map information from a series of short pulses, whereas SAHARA obtains this data from a single pulse of long duration.

The pulse doppler radar obtains finer resolution and requires less communications bandwidth than SAHARA, because processing is feasible within the Orbiter. However, due to the pulsed nature of the pulse doppler system, there are potential range and azimuth ambiguities which require a 14-foot diameter antenna operating at a wavelength of 3.2 cm in order to obtain directivity to suppress them adequately. SAHARA can use a 10-foot diameter antenna operating at a wavelength of 13 cm. The 14-foot antenna at the 3.2 cm wavelength requires a significant increase in antenna weight because of its increased size and smaller tolerance due to the shorter wavelength.

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* Map in this report does not mean a survey or chart, but a representation of the surface from which approximate sizes, separations, locations, and altitudes of features can be obtained.

It is recommended that the SAHARA system be used for the Venus 1970 Orbiter. The recommendation is based primarily on the significantly lower antenna weight which is estimated to be between 60 and 90 pounds.

The SAHARA radar has the following characteristics:

frequency	2295 Mc (S-band)
antenna diameter	10 feet
power requirement	440 watts
weight	109 pounds
volume	2.0 ft^3 (excluding antenna)
polarization	circular
peak transmitter power	302 watts
final map resolution	1.0 nm x 1.0 nm
required data transmission rate	63,000 bits/sec

The video signals at the output of the radar must be transmitted to the earth for the processing required to obtain a map.

The 1 nm x 1 nm resolution are values of azimuth and range resolution obtainable from an altitude of 1000-miles. The range resolution can be made constant with altitude but the azimuth resolution degrades with altitude, (e.g., it is 1.6 miles for a 2500 mile altitude). Map quality is a function of signal-to-noise ratio which, in turn, is a function of altitude. A good quality map can be obtained with average terrain reflectivity up to an altitude of 1200 miles and a fair quality map up to an altitude of 1600 miles. A calculation was made of the radar system weight if the transmitter power is increased to give constant map quality over the altitudes of interest. These weights are large and cannot be accommodated in the Venus 1970 Orbiter. With an elliptical orbit, mapping will be limited to the lower altitude regions.

5.5.2 RADAR SURFACE SOUNDER

The surface roughness of the planet may be determined by using multi-frequency sounding radar in the microwave frequency range. The variations in polarization and amplitude of the radar return are evaluated as a function of the aspect angle to determine the dimensions of the surface roughness. The terrain mapping antenna is used for this system which requires a scan of ±50 degrees about the nadir.

5.5.3 RADIOMETER

Radiometer experiments will be conducted from the Orbiter. These will extend the information obtained from Mariner II and, especially, make measurements with finer spatial resolution. Data will be obtained which will give surface temperature, roughness, and perhaps composition. The radiometer will use the radar antenna, part of the radar surface sounder equipment, and five pounds of additional equipment.

5.5.4 IONOSPHERIC SOUNDING

Ionospheric sounding measurements of the height, thickness, and electron density of the ionosphere of Venus may be obtained from the Orbiter. The ionospheric sounding equipment would direct RF pulses towards the ionosphere at various frequencies. The amplitude, polarization, and time delay of the return would provide the basic data from which ionospheric measurements can be deduced. Discrete frequencies between 5 and 15 megacycles are transmitted using a long extensible rod or whip antenna.

5.5.5 ORBITER RADAR ALTIMETER

The mapping radar for the Venus 1970 Orbiter may also be used in a radar altimeter mode. The nominal orbit for the Venus 1970 mission is an ellipse with a minimum altitude of 1,000 nm and maximum altitude of 4,300 nm. The eccentricity of the orbit actually attained may be obtained from measurements of height-above-terrain with the use of radar altimeter. If such information is obtained for a few orbits the effects of uneven terrain may be smoothed. An accuracy of ±0.66 nm is estimated for the altitude measurements.

5.5.6 LANDER RADARS

A radar altimeter will provide a data base for atmospheric measurements made during the sub-sonic portion of the descent of the Mars-Venus Landers. Present Lander design studies indicate that the vehicles will not attain sub-sonic speeds until below an altitude of 200,000 feet. Altitude requirements for the atmospheric experiments for Mars and Venus entry vehicles are as follows:

a. Accuracy of ±2 percent of altitude, or ±100 feet, whichever is larger.

b. Measurements to be made every 1,000 feet after sub-sonic velocity is attained.

The altitude measurements for the Mars Landers can be made with a pulsed radar altimeter. The altimeter will be mounted behind a radome constructed of non-charring ESM which has been found to have acceptable electrical characteristics. The altimeter is a modification of the Surveyor Altitude Marking Radar. The modifications consist of a new antenna and circuitry to allow altitude to be measured continuously.

For the Venus Landers there is an additional requirement to measure the horizontal displacement of the capsule during descent. Measurements of horizontal displacement rate may be used to deduce horizontal wind velocities. The Surveyor Radar Altimeter and Doppler Velocity Sensor which integrates the velocity and altitude functions, is recommended for use in this application. It measures altitude to an accuracy of ±2 percent and horizontal velocity to an accuracy of ±1 percent, which is adequate for the Voyager mission. The antennas which would be deployed when subsonic velocity is attained will be re-designed for mounting in the Venus Lander. A solid-state transmitter will be used, which will result in a lower input power requirement and will be able to withstand the entry deceleration.

5.6 GUIDANCE AND CONTROL

In the interplay between subsystems and mission requirements, the capabilities of the Guidance and Control subsystems, and the costs of achieving the desired capabilities, become constraints on the design of the mission itself. This applies to system versatility as well as accuracy. Almost inevitably, key subsystems become more complex as the vehicle (and the missions it can perform) becomes more flexible.

Guidance and control considerations also play an important part in defining the vehicle configuration. The choice of reference attitude sensors influences the number and placement of hinges articulating the vehicle, and the appropriate locations of significant amounts of equipment. High data rate requirements for guidance information help define the periods of deploying the high gain antenna. Decoupling problems and loop interactions argue against the use of certain physical configurations.

The decision to orient portions of the vehicle to the planet vertical, rather than the entire body of the vehicle, determines the range of motions required and the boom length in order to see past portions of the vehicle. On the other hand, rotation of the orbiting vehicle around the vertical to the planet would cause difficulty in obtaining stero pairs of TV frames. As another example, the attitude control torquing capability of the Orbiter determines the attitude disturbances that can be tolerated during separation of the Landers.

Bearing in mind that it is desired to orient continuously to the Earth, Sun and planet, a reasonable compromise has been achieved between vehicle complexity and mission flexibility. The same basic vehicle is capable of executing both Mars and Venus missions and it is possible to accommodate a change of orbit plane up to or even after the launch.

5.6.1 SUBSYSTEM DESCRIPTION

The guidance and control requirements of the Voyager mission can be met using primarily existing techniques and types of equipment. In a few cases, i.e., the electrostatic image orthicon tube and camera and the Lunar and Planetary Horizon Sensor, current development programs are presumed to be successfully concluded by the time the equipments are needed.

An exception to this may be found in the autopilot. There is no serious question of ability to accomplish the required performance; it is a matter of the degree of control system sophistication required to do it. The most obvious problem is the result of the short distance between the c.g. and the point at which the thrust vector pivots. This results in a requirement for relatively large motions of the thrust vector, and in extreme cases could require faster response than that readily obtainable from servos driving a gimballed engine. The result could be a requirement for a larger than customary hydraulic servo, or secondary injection for thrust vector control. Preliminary studies of the frequency response requirements for the thrust vector control actuation indicate that they can be met within the capabilities of existing equipments; however, further studies of the disturbance from fuel motions in a vehicle with this configuration will be needed to demonstrate this conclusively.

Voyager could represent the first vehicle to practice interplanetary celestial navigation. In performing this function as also in other cases, operational simplicity as well as component simplicity have been considered. Hence, a TV picture of the planet against the star background, with data reduction on Earth, is favored over planet/star tracker combinations that would require search and acquisition sequences, and possibly star identification techniques on board. For a Venus mission, special techniques will be utilized to obtain clear images of both planet and stars, due to the large apparent brightness range. Several techniques are known, each of which appears capable of accommodating the brightness range. Further tests are expected to verify this. For a Mars mission, no special techniques are required to see 4th and probably 6th magnitude stars. Fourth magnitude stars are numerous enough to meet the need. Use of the high-gain antenna permits the TV frame to be transmitted to Earth without requiring complex computation or data processing on board. However, bit reduction techniques have been defined which would permit the use of a medium gain fixed antenna for any vehicle, such as a separate Lander bus, which has no other need for high-gain antenna.

With the exception of the readings of line-of-sight to the planet during the approach phase, both the Guidance and Control Subsystem bear a strong resemblance to the Mariner systems. As the block diagram, Figures 5.6-1 and 5.6-2 show, the vehicle is referenced to the Sun and Canopus, with the Earth sensor that controls the antenna capable of acting as a backup for Canopus. In general, accuracy of $\pm 1^{\circ}$ for the Attitude Control and the antenna and PHP drives is adequate. However, vehicle attitude errors of less than $\pm 1/4^{\circ}$ are readily achievable and have been called out. Greater accuracy is also possible for the other drives, without undue difficulty if they should become appropriate.

Relatively rapid vehicle motions minimize the time during which the vehicle is oriented away from the Sun. A maximum of six minutes is required for any maneuver about any one axis.

There are numerous opportunities for completely redundant control information, e.g., either a canopus tracker or an Earth sensor can be used as a roll reference. There is a choice between these sensors and gyros for vehicle attitude reference during all periods of calculation.

The antenna and PHP are articulated to the vehicle proper using stepping servo drives. Momentum cancelling techniques are simple to implement and may be desirable in view of the driven inertia, to prevent vehicle disturbances during slewing or stepping.

Attitude control of the spacecraft, outside of the rocket firing period, represents a comparatively mature technology. The theoretical minimum of required impulse can be closely approached, with confidence, using derived rate techniques. The resulting vehicle rates are low enough to avoid customary limit cycling. As a result a single mode of operation is satisfactory; no advantage would be realized by providing coarse and fine control modes. The low vehicle rates also avoid the need for flywheels or other momentum storage devices unless cyclic maneuvers or greater smoothness of camera pointing are required. Hence an all cold-gas system is recommended. Cyclic torques experienced in the orbits studied are not large enough to justify the weight of flywheels.

5.6.2 PREDICTED ACCURACY

With the atmospheres initially studied, and neglecting surface winds, a desired landing site can be obtained within less than $\pm 4^{\circ}$ (125 nm) where the landing site is in the plane of the approach trajectory. For one studied landing site, 32° out of the trajectory plane, the dispersion increases to $\pm 8.0^{\circ}$ or 250 nm. If the larger rocket, required to execute this out-of-plane landing, is used for the first (in-plane) case also, the errors for that case increase to 5.5° (173 nm). Lander dispersion is highly insensitive to the range from the planet at which separation occurs. The figures quoted are for separation at 150,000 nm and include both separation and navigation errors, with Lander total orientation errors of 3°.

With conservative assumptions, the desired orbit perifocal altitude of 1000 $\pm$ 100 nm is possible. One approach correction after taking the first line-of-sight data (at 2 x 10^6 nm) is probable. Additional sightings thereafter will give the necessary inputs to adjust the injection ΔV in order to minimize the resulting orbital period error. Orbit injection is relatively insensitive to injection parameters other than ΔV. Holding a constant attitude during rocket firing, or delay of ignition until perifocus, causes 1% gravity loss.

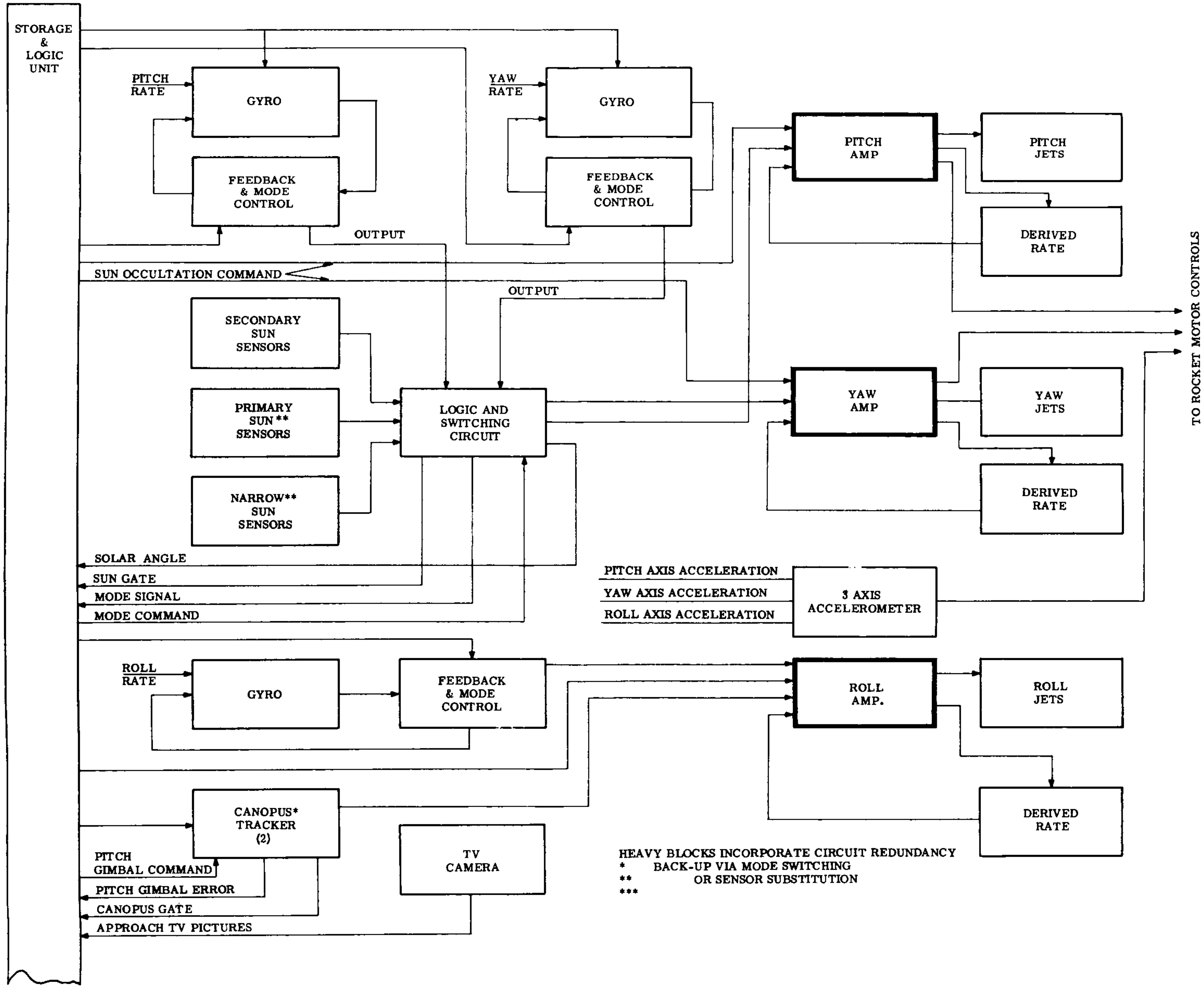


Figure 5.6-1. Guidance and Control Subsystem

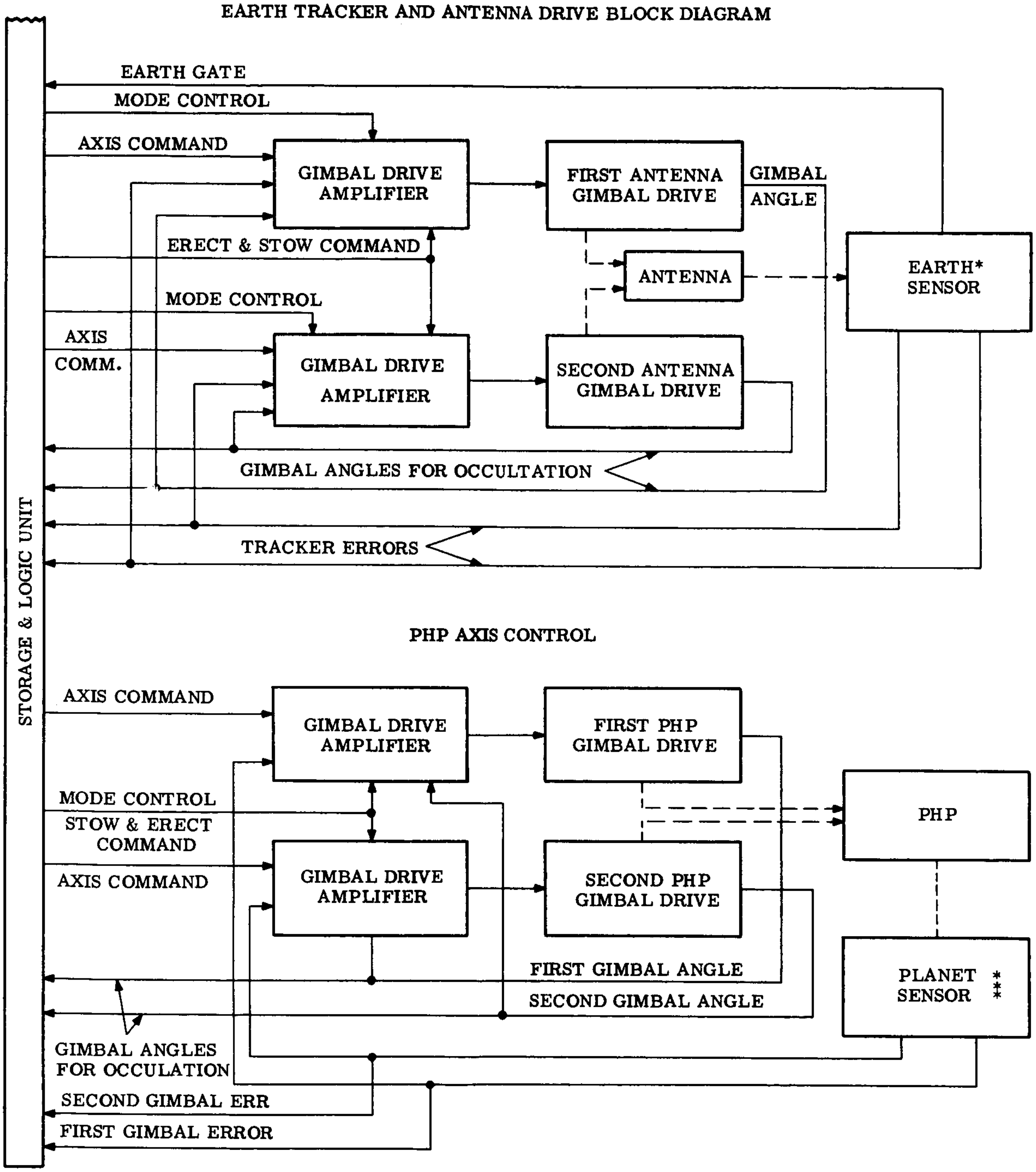


Figure 5.6-2. Guidance and Control Subsystem

5.6.3 ORBITER-LANDER LINE-OF-SIGHT

Line-of-sight is maintained between Orbiter and Lander until the Lander reaches the surface of Mars for all cases studied, if the parachute is not deployed before reaching 30,000-foot altitude. To accomplish this it is not necessary to retard the Orbiter, or accelerate the Landers to a greater degree than the separation sequence itself would produce with the rockets that have been specified for the Landers.

A requirement for Orbiter retardation after separating the Mars Landers, as an alternative to delaying parachute deployment or to permit injection burn before perifocus (but after the Landers land), would entail a moderate propulsion penalty. More important, it would be a major source of error in perifocus altitude and it would add another sequence of maneuvers to a list which one would prefer to shorten. On the other hand, Orbiter retardation may be appropriate for Venus.

Conservative assumptions are favored in analysis of guidance accuracy. The adequacy of a vehicle and mission being defined now should not be contingent on a future successful Mars fly-by. In addition, it is possible that gas leaks and unbalanced couples for the attitude control as well as uncertainties in solar radiation pressure can cause trajectory perturbations large enough to cause difficulty in determining the trajectory to an accuracy consistent with the accuracy of the DSIF tracking input.

Accordingly, there is a non-trivial probability that readings of line-of-sight to the planet during approach will be required in order to achieve both the desired altitude from the planet at perifocus and landing site accuracy.

5.6.4 LANDER EARTH ANTENNA TRACKING

A simple means has been defined for orienting the high-gain antenna on each Lander to the Earth. With a simple initial orientation plus occasional updating, which can be accomplished by command, the antenna will be directed to the Earth for a minimum of 1/2 hour per day, which is the maximum time per day for which there is power to utilize this antenna. This system requires minimal sensors to operate on the surface of Mars. At the absolute minimum, all that is needed is initial functioning of a sun sensor and detection of the presence of an RF signal from Earth when the antenna scan is directed toward Earth in a simple slow control scan.

5.6.5 GUIDANCE AND CONTROL – SPACECRAFT DESIGN

As might be expected, there are instances where vehicle mechanical simplification results in considerable complication in the control system. A reasonable tradeoff has been reached in such instances. The result is a vehicle with great flexibility in accommodating different missions or orbits, without causing unduly complex control loops. (For a specific mission it may be possible to simplify the spacecraft with no reduction in capability.)

The performance of the Guidance and Control subsystem has been defined at levels which appear to be fully adequate to meet the requirements, using simple and proven techniques to the fullest extent. Higher performance in most cases can be readily obtained whenever it is desirable. Where the cost (particularly in reliability) of such upgrading is inconsequential, it should be provided so as to provide the maximum tolerance of degradation either there or elsewhere. Otherwise, upgrading should be confined to the instances where the need justifies it if there is even a minor attrition of reliability.

As mission capabilities increase, highly accurate low orbits or narrow entry corridors may become a requirement. To accomplish these, planet tracking may be required beyond the point where information can be sent to Earth for computation and commands sent back. This situation would present the first definite requirement for on-board computation.

When the Electrostatic Gyro becomes available it will make possible significant operational advantages. Even before the ultimate anticipated accuracy is attained, the elimination of gyro torquing, and the absence of stops that limit the allowable transient motions of the vehicle would simplify the control system design. To some degree the same sort of benefits would accrue to the vibrating string gyro.

5.6.6 CRITICAL PROBLEM AREA

The systems described in some respects do not represent sharp optima. Alternative approaches may have equal merit under slightly different assumptions. Although analysis shows that the desired performance is obtainable with the techniques described, advance development and test is highly desirable in certain areas such as the TV camera/transmission/data extraction sequence for approach line-of-sight information.

Several areas have been identified which warrant further study:

a. possible Orbiter disturbances associated with Lander separation (at present considered to be minimal).

b. autopilot operation during operation of the main rocket, including path angle control, in the presence of fuel motions.

c. the possibility of designing the spacecraft so that it can utilize passive means of stabilization during the major portion of the transit period where the requirement for accurate vehicle attitude can be relaxed.

5.7 PROPULSION

The propulsion systems required for the Voyager mission are the Lander propulsion, the Orbiter propulsion, and the attitude control propulsion. The study analyzed the requirements for the three systems and selected basic system designs to meet these requirements. The Lander propulsion system selected is suitable for all missions with the exception that the case must be off-loaded for the smaller Landers. The Orbiter propulsion system is sized for the maximum mission with only tankage and propellant quantity changes required between missions. The missions which require a small total impulse are subject to a heavy propulsion system dry weight in order that a single design may be used. If subsequent requirements dictate a heavier payload for these missions, considerable weight can be eliminated by developing a second system with a smaller thrust level. The attitude control propulsion system can be used for all missions except for a change in tankage capacity.

5.7.1 ORBITER PROPULSION

Propulsion for in-transit adjustments and orbit insertion is provided by a single bipropellant pressure-fed engine utilizing N_2O_4 as one propellant and 50 percent UDMH-50 percent N_2H_4 as the other propellant. The thrust chamber is ablative with a radiative skirt and produces 2200 pounds thrust at a chamber pressure of 100 psia and an area ratio of 100. It is sized to produce sufficient impulse for the Venus 1970 mission, which is the maximum impulse required for all missions. For this mission, the total burn time is 11 minutes. Propellants are contained within spherical tanks, one for each propellant, and are pressurized to a regulated 200 psia with stored helium. Propellant expulsion and propellant c. g. shift are controlled through the in-transit adjustments and through a large portion of the orbit injection burning by means of laterally constrained partial bellows in the propellant tanks. When these bellows are nearly extended, the pressurization is stopped and reinstated directly on the liquid. Controls are redundant so that no single failure except a structural failure or thrust chamber burnout will cause the mission to fail. For the Mars 1969 and 1971 missions, an orbit adjustment is possible by exhausting residual helium gas through special nozzles.

Solids, hybrids, monopropellants, and bipropellants were considered for the Voyager missions. Monopropellants were rejected on the basis of low impulse for developed propellants. Hybrids were rejected on the basis of a low development level with attendant high risk. Solids appeared competitive from an overall payload capability standpoint and superior from a reliability standpoint, but the complexity of on-pad off-loading (required to maximize payload) and the low development status of restartable solids eliminated solids from further consideration.

A number of bipropellant combinations including high energy cryogenic and non-cryogenic combinations were considered. Cryogenics considered included combinations utilizing fluorine and OF_2. These high energy combinations showed non-propulsive payload increases up to 25 percent for the Venus 1970 mission and correspondingly lesser advantages for the lower ΔV missions. These high energy systems were rejected, since it was determined that the contemplated missions could be achieved with highly developed current Earth storables. In the event higher payloads become a requirement, a very serious development risk in going to higher energy propellants would be involved.

Of the current earth-storable oxidizers and fuels, N_2O_4, MON, N_2H_4, MMH, UDMH, and 50 percent UDMH-50 percent N_2H_4 were considered. Thermal analysis of the vehicle showed that it was possible to adjust temperature within the liquidus range of all these propellants. The N_2O_4/50-50 combination was chosen on the basis of a high level of present and contemplated near-future experience, since other considerations appeared to be about equal to the earth-storables.

Of the many thrust chamber types, the radiative, ablative, and regenerative were seriously considered. The regenerative chamber has an advantage of considerable development experience, but in order to get sufficient cooling from the limited propellant flow, a high chamber pressure (small chamber area) thrust chamber would be required. This would necessitate a pumped system. Although other disadvantages exist, this complexity was the prime reason for rejection of the regenerative system. The radiative chamber appeared very competitive with the ablative chamber, since for this application, a lower thrust could be utilized with a radiative chamber. This would give an added weight advantage over the ablative chamber of lighter control components. Three disadvantages of the radiative chamber were considered of major importance. First, although burial of the chamber was not a requirement, a radiation shield would be required to protect the solar cells. This shield would probably block some solar cells. Second, the radiative chamber is especially susceptible to off-design injector operation and attendant "hot spots". Although the ablative chamber is also susceptible, it appears to be so to a lesser degree, since circumferential and axial heat flow, due to the thick ablative material, will help transfer heat to adjacent areas. Thus, partial plugging of an injector would probably be catastrophic to a radiative chamber before it would be to an ablative chamber. Third, although the development status of ablative chambers is low, it is more advanced than is the radiative chamber, and projections indicate it will continue to have more emphasis placed upon it.

Primarily for these stated reasons, the ablative chamber was chosen over the radiative chamber at a considerable weight disadvantage.

Thrust level from a weight standpoint was optimized at a fairly low level. For the ablative chamber, this was approximately 750-1000 pounds. From a chamber weight, moment of inertia, thermal radiation to vehicle, controls weight, and pressurant system weight standpoint low thrust is desirable; but below about 750-1000 pounds, gravity losses become excessive. Past experience and projected near-future experience for ablative thrust chambers within this general range give little firing history above 600-700 seconds. To keep development risk to a minimum, a thrust level of 2200 pounds, corresponding to a run time of about 650 seconds, was chosen. Considerable work has been done by Aerojet-General at this thrust level, although a chamber cannot be considered fully developed. The nearest projected fully developed chamber is the LEM Descent chamber, projected at 3500 pounds thrust, burning the same propellants as projected for Voyager and almost exactly coinciding with the total impulse required for the maximum impulse mission. Use of this chamber, however, would entail an increase in total systems weight of about 40 pounds.

On a weight basis, there was little difference in propulsion system weight for chamber pressures between 100 and 150 psia. The increase in tankage and gas weight necessary to raise chamber pressure from 100 psia to 150 psia was offset by a decrease in thrust chamber weight. The lower chamber pressure of 100 psia was selected primarily on the basis of industry experience. Further, lower heat transfer to the chamber would result, thus less overstressing would occur in the event of off-design injector operation.

The mixture ratio chosen was 1.65. This selection was somewhat arbitrary, since the actual maximum impulse ratio is dependent upon experimental factors. Although the 1.65 allows a common tank size, experimental data may show that an increase in mixture ratio is desirable from an overall weight standpoint, and equal tanks cannot be used.

Thrust chamber area ratio is optimized in excess of 100 from an overall vehicle payload capability standpoint. However, at this ratio, increasing the ratio gives little additional payload advantage. Although in accordance with criteria established for the program, there is no package limitation, handling and separation complexity might be increased with a large motor. Also, heat radiation to the vehicle will increase with a possible blockage of solar cells with increased radiation shielding. For a gimballed engine the additional moment of inertia of the motor could increase weight of the actuation system. Therefore, expansion ratio could change when detailed hardware designs have been completed and some development tests have been run. This change could exceed 40 points.

Propellant pressurization systems which were considered include the pumped system, the solid propellant charge pressurization, stored liquid systems, and stored gas systems. Stored gas system was chosen based on reliability, simplicity, and development experience, but the solid propellant charge system had a considerable weight advantage. Consideration was given for heating of the helium pressurant; weight is saved and the risk of freezing the propellant is eliminated. The possibility of freezing occurring in the Voyager vehicle is enhanced by the expulsion mechanism arrangement used. Weight saved by heating would be 10 to 15 pounds on the Mars 1969 mission and approximately twice this on the Venus 1970 mission. Disadvantages of the heated system are the added complexity to the system, difficulty in maintaining acceptable temperatures, and the addition of contaminants to the system if heating is accomplished by a helium tank-enclosed, solid propellant grain. The unheated system was selected, except that tank temperature is raised back up to the highest transit value prior to firing for orbit injection. It is recognized that detailed heat transfer studies on the final design and development testing may show the necessity for adding additional heat to prevent propellant freezing, although this is not considered likely. In this event heat would be added by close proximity of helium lines to the thrust chamber.

A large number of devices were considered to provide positive expulsion of the propellant and to prevent excessive c.g. shift due to propellant sloshing. Although a diversity of opinion existed among propulsion companies as to the suitability of various mechanisms for controlling c.g. shift, it was felt that the laterally constrained partial bellows provided the most conservative approach and would provide sufficient propellant control and damping. The volume of the partial bellows in the extended position is approximately half of the propellant tank volume. The bellows is cylindrical in shape, and has a diameter slightly larger than the radius of the tank. It is constrained from moving laterally by four rods placed in the tank external to and parallel with the bellows. At the point in time shortly before the bellows reaches its extended length, pressurization into the bellows is stopped, and pressurant is fed directly on top of the propellants. This precludes higher stresses being placed on the bellows and assures 100 percent mechanism expulsion efficiency.

All controls are redundant so that no single malfunction other than a structural failure or thrust chamber burnout will cause a systems failure. In general, redundant circuits are of different designs, so that failure mechanisms will not be identical and manufacturing or handling errors affecting all pieces of a lot will not cause a double failure in identical parts. One parallel set of propellant controls will be hermetically sealed off (dry) from the rest of the system and will be operated only in the event of failure of the primary controls, as indicated by chamber pressure. Propellant controls are linked to prevent loss of propellant in the event of failure of a valve to open.

The selection of a thrust vector control system is an open item, although for consistency and weight purpose, it is shown as a gimballed system. The gimballed system is much preferred from the standpoint of propulsion system weight and development risk, but its slow response time may not be compatible with system requirements. If secondary injection were required for fast response time, additional development risk would be incurred.

5.7.2 LANDER PROPULSION

The Lander engine is a spherical, case-bonded, solid propellant motor. The titanium case is sized to provide sufficient impulse for the largest Landers. Smaller Landers will use the same case off-loaded to provide the lesser total impulse. All components will be heat sterilizable.

The chamber has been sized based on a vacuum specific impulse of 230 seconds. This conservative figure was based on the possibility of the necessity for using the lowest performing propellant of the four candidate propellants considered. It is probable that a higher impulse propellant can be used. A detailed discussion of the various propellants considered is presented in the classified appendix.

5.7.3 ATTITUDE CONTROL PROPULSION

The Voyager attitude control propulsion subsystem is a cold gas, gaseous-stored system. The propellant is Freon-14 stored at 1700 psia and regulated to 50 psia. Thrust is provided by couples to impart pure rotational motion and to provide redundancy in a slightly degraded mode. The nozzles, using an area ratio of 100, produce a thrust of 0.047, 0.043, and 0.066 pounds for motion about the roll, pitch and yaw axes. Redundancy is achieved by using two separate systems to provide the couple halves and by using solenoid valves and regulators in series.

The other systems that were considered are cold gas, liquid-stored; cold gas, solid-stored, hot gas; cap pistol; and electric propulsion.

From a weight standpoint the two systems that are decidedly better than the chosen system are the cold gas, solid-stored system, and the cold gas, liquid-stored system. The former has a weight advantage of 25 pounds, but the thermal control problems readily outweigh this advantage. In addition, little work has been done on such a system, and development risk would be high.

A cold gas, liquid-stored system was not chosen, since it could present an even greater development risk, although it does represent a weight saving of 36 pounds. This system also has the temperature control problem plus a phase separation problem. These problems would reduce the reliability of the system to below that of the gaseous stored system.

Hot gas systems, either liquid or gaseous stored, were rejected on the basis that due to the minimum impulse bit requirement the specific impulse would be exceptionally low. This, in turn, would increase the system weight to where it had no advantage over the chosen system. In addition, reliability is considerably decreased due to the necessity for more control components and for a high temperature combustion chamber.

A solid encapsulated pulse engine, the "Cap Pistol" under development by Curtiss Wright, was considered. This engine is not in a sufficient state of development to be chosen for the Voyager mission without prohibitively high risk.

Electric propulsion systems have been rejected, since they present too great a development risk for the early opportunities.

5.8 POWER SUPPLY

During the Voyager Design Study, the power supplies for each of the Mars and Venus missions received detailed attention. The power requirements for each mission were analyzed and a number of alternative ways for meeting these requirements were investigated and evaluated. Selection of specific power supplies were made consistent with the conservative system criteria established for the study. Itemized below are the major conclusions reached during the study.

The following power supply systems are recommended for the Voyager missions.

Orbiters

(1) Silicon solar cells and nickel cadmium batteries for all Mars and Venus missions through 1973.

(2) Silicon solar cells and a silver zinc battery for the Mars 1975 fly-by mission. (Nickel cadmium batteries might also be used, however, with only a small weight penalty, about 5 - 10 pounds.)

Landers

(1) A radioisotope thermoelectric generator (fueled with Cm 244) and nickel cadmium batteries for all Mars Landers.

(2) Silver zinc batteries for both Venus Landers.

Tables 5.8-1 and 5.8-2 summarize key parameters of these recommended systems.

Isotope thermoelectric generators, fueled with Cm 244, look promising from the standpoint of weight for all Orbiter missions. However, they are not recommended for the 1969 Mars mission because of uncertainty in isotope availability. They are considered to be possible alternate power supplies for Orbiter missions after Mars 1969.

Isotope thermionic systems are also possible alternates for Orbiter and Lander missions after 1969. Such systems are not expected to be available for the Mars 1969 opportunity.

Isotope availability may be a serious problem for the Mars 1969 Lander unless steps are taken immediately to assure production of the necessary quantities by the time required.

Uncertainty in earth safety and planet contamination ground rules has a major effect on performance estimates of isotope generator systems and should be resolved as soon as possible.

5.8.1 ORBITER POWER SUPPLY

A schematic diagram of the Mars 1969 Orbiter power supply, which is typical of the power supply for all of the Orbiter missions, is shown below. Efficiencies assumed for the various power supply components, including harness losses, are indicated on the figure. The solar array, using N/P silicon solar cells, supplies power to the load and charging power to the batteries when the vehicle is sun oriented. The rechargeable nickel cadmium batteries supply power requirements when the vehicle is not sun oriented.

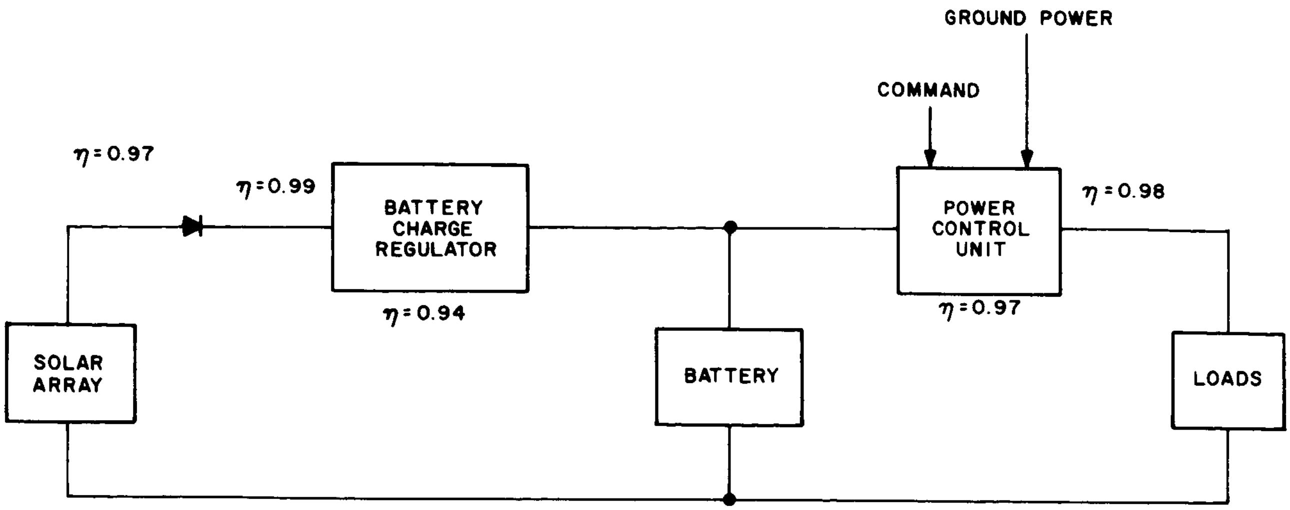
The battery charge regulator controls the rate at which the batteries are charged. It is the switching type unit which maintains the required average battery charging current and operates at a very high efficiency, as indicated. With this arrangement, the battery also provides coarse voltage regulation on the main bus (the bus voltage varying between battery charge and discharge values) with an expected variation of approximately ± 15 percent. Each load provides its own voltage level and regulation.

TABLE 5.8-1. ORBITER POWER SUPPLY SUMMARY

Planet		Mars						Venus	
Year		1969	1971	1973	1975	1973 Back-up	1975 Back-up	1970	1972
Array Power at Load	(watts)	440.0	446.0	155	130	446	155	589	300
Peak Load Power	(watts)	440.0	446.0	334	334	446	334	609	365
Array Area	($ft.^2$)	125.5	143.8	48.4	40.3	134.8	48.0	99.0	50.4
Solar Array	(lb.)	173.1	206.3	70.6	65.9	190.6	70.4	132.6	91.6
Battery	(lb.)	21.3	21.3	22.7	8.6	21.3	22.7	46.8	25.0
Battery Charge Control	(lb.)	7.1	7.2	2.5	2.1	7.2	2.5	9.5	4.8
Power Control Unit	(lb.)	3.5	3.6	2.7	2.7	3.6	2.7	4.1	2.9
Solar Array Harness	(lb.)	7.2	8.2	2.8	2.3	7.7	2.7	5.7	2.9
Inflight Disconnects	(lb.)	4.5	4.5	4.5	4.5	4.5	4.5	3.0	3.0
Diodes (Solar Array)	(lb.)	1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.0
Total Weight	(lb.)	217.7	252.1	106.8	87.1	235.9	106.5	202.7	131.2

TABLE 5.8-2. LANDER POWER SUPPLY SUMMARY

Planet		Mars						Venus	
Year		1969	1971	1973	1975	1973 Back-up	1975 Back-up	1970	1972
Generator Power at Load	(watts)	70	70	70	70	70	70	--	--
Peak Load Power	(watts)	362	360	362	362	356	362	207	232
Load Energy Required	(watt-hrs.)	--	--	--	--	--	--	219	570
Surface Lifetime		Months						30 min.	6.4 hr.
RTG Unit	(lb.)	54						--	--
Battery	(lb.)	28						8.5	18.5
Battery Charge Regulator	(lb.)	3						--	--
Power Control Unit	(lb.)	9						5	5
Power Supply Harness	(lb.)	5						3	3
Inflight Disconnect	(lb.)	1.5						1.5	1.5
Hardware and Supports	(lb.)	6						2	2
Total Weight	(lb.)	106.5						20	30



The power control unit provides for switching of various components according to command and/or programmer inputs, contains input connections from ground power, and may also provide some circuit protection.

More detailed performance parameters of the solar array for each of the Orbiter missions are given in Table 5.8-3.

5.8.2 MARS LANDER POWER SUPPLY

A schematic diagram of this system is the same as that shown for the Orbiters except that the isotope thermoelectric generator (RTG) replaces the solar array. For this mission, the batteries handle peak loads only.

Figure 5.8-1 is a drawing of the resulting generator design. The unit is cooled either by convection or radiation or by a combination of the two, depending upon the operating mode. The circulating coolant is used exclusively until planet impact, rejecting heat during the operating modes as indicated below:

Mode	Circulating Coolant Rejects Heat To
Pre-Launch	Ground radiator
Launch	Water evaporator
Transit	Space radiator
Re-entry and descent	Water evaporator

After planet impact, the nose portion of the Lander is removed, exposing the fins of the generator to the environment. Heat is, thereafter, largely rejected by the fins while the circulating coolant is used for vehicle thermal control.

The design concept indicated here is aimed toward satisfying the following ground rules:

- Isotope must be contained under all conditions for ten half-lives
- Sufficient void volume must be provided to contain the theoretical quantity of helium gas generated in the isotope decay
- Isotope temperatures must be below the melting point under normal operating conditions

TABLE 5.8-3. DETAILED SOLAR ARRAY PERFORMANCE

Planet		Mars						Venus	
Year		1969	1971	1973	1975	1973 Back-up	1975 Back-up	1970	1972
Body Mounted Array Area[1]	(ft.^{2})	81.8	81.8	48.4	40.3	81.8	48.0	81.8	50.4
Shelf Mounted Array Area[1]	(ft.^{2})	43.7	62.0	-0-	-0-	53.0	-0-	17.2	-0-
P_1/A-Body Mounted Array[2]	(watt/ft.^{2})	3.40	3.02	3.20	3.23	3.20	3.23	5.95	5.95
P_1/A-Shelf Mounted Array[2]	(watt/ft.^{2})	3.71	3.21	--	--	3.47	--	-- 4)	-- 4)
P_a/A-Body Mounted Array[3]	(watt/ft.^{2})	3.97	3.52	3.73	3.77	3.73	3.77	6.95	6.95
P_a/A-Shelf Mounted Array[3]	(watt/ft.^{2})	4.33	3.75	--	--	4.05	--	-- 4)	-- 4)
Body Cell Temperature	(^{o}F)	102	79	68	71	68	71	282	282
Shelf Cell Temperature	(^{o}F)	65	44	33	--	33	--	-- 4)	-- 4)
Solar Flux	(watt/ft.^{2})	65.8	51.2	48.0	48.7	48.0	48.7	248	248
Distance from Sun	(AU)	1.4	1.594	1.655	1.635	1.655	1.635	0.723	0.723

Notes:

1) Active cell area = 0.9 (Array area)

2) Based on power to load

3) Based on array power output

4) Not calculated. Assumed same as body cells for power output purposes.

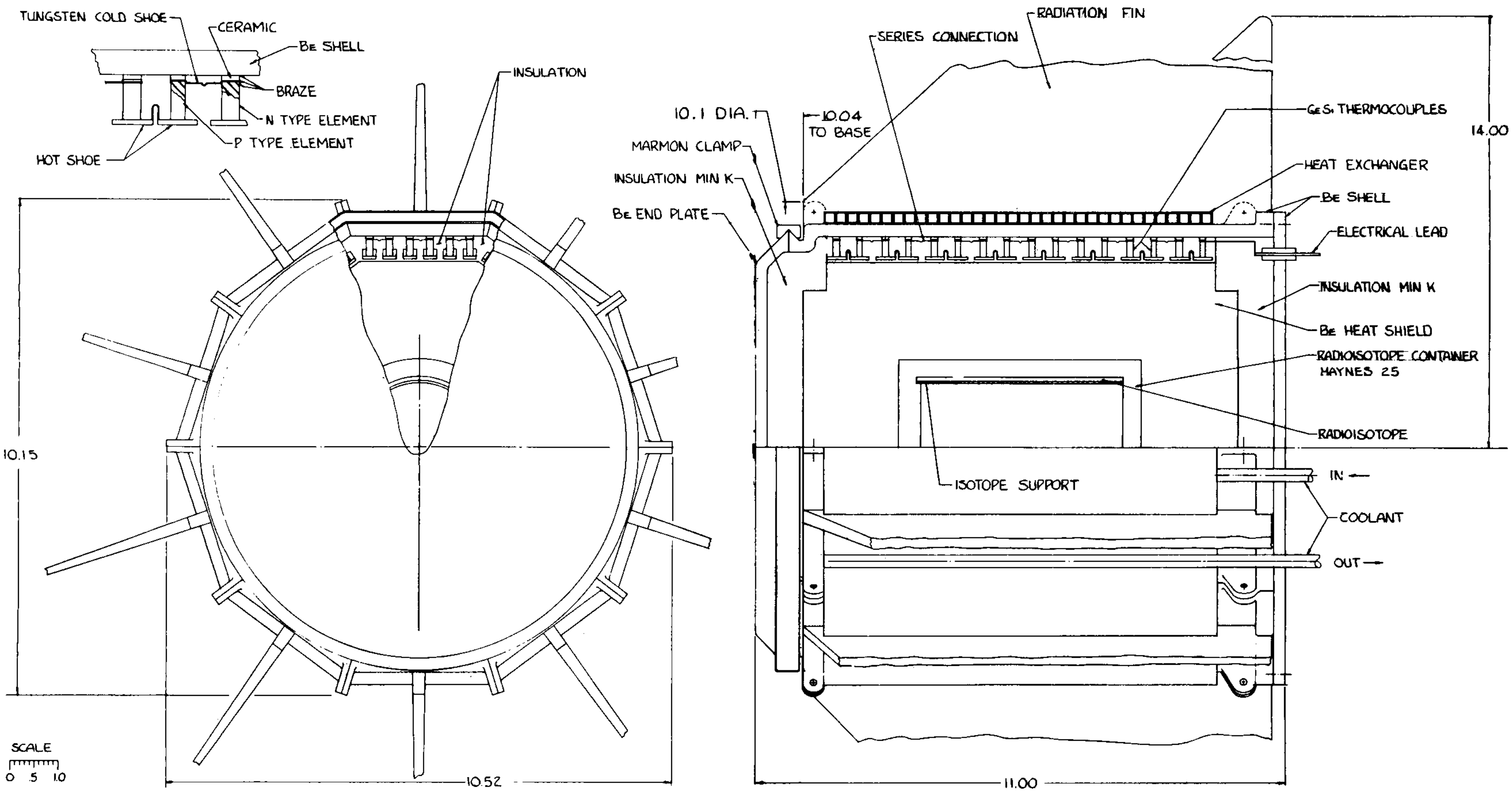


Figure 5.8-1. Radiosotope Thermoelectric Generator - Mars 1969 Lander

The first two ground rules are based on an interpretation of Reference 1, and discussions with various NASA personnel. The last rule occurs because of the nature of the design, wherein the hottest portion of the isotope is adjacent to a porous inner metallic container. It is not considered prudent to operate in the molten state under such conditions. At the present time, it is apparently not clear as to what the proper ground rules should really be as far as earth safety and planet contamination are concerned. However, in the absence of firm information, this study has emphasized the above approach.

Consequently, the design shown in Figure 5.8-1 has a large void volume, a thick isotope capsule wall, and a thick shield of beryllium for re-entry protection. Although the design shown is believed to be a reasonable approach towards satisfying the ground rules, it has not been possible to analyze all failure modes in the detail that would be desirable, and the design can only be considered to be approximate.

Table 5.8-4 provides a summary of pertinent performance characteristics of this design.

TABLE 5.8-4. ISOTOPE THERMOELECTRIC GENERATOR DESIGN

Power Output of Generator	watts	82
Power Available at Load	watts	70
Output Voltage	volts	28
Weight	lb.	54
Diameter (Excluding fins)	inches	10
Length	inches	11
Fin Length	inches	9
Thermoelectric Efficiency	%	4.7
Generator Efficiency	%	4.3
Thermoelectric Material		GeSi
Number of Thermocouple Pairs		480
Number of Series Strings		2
Isotope		Cm 244
Initial Thermal Output	watts	1970
Thermal Output - 1 year	watts	1900
Isotope Melting Temperature	^{o}F	4062
Isotope Temperature	^{o}F	3269
Capsule Temperature - (Inner)	^{o}F	1709
Hot Junction Temperature	^{o}F	1300
Cold Junction Temperature	^{o}F	575
Fin Base Temperature	^{o}F	543
Void Volume	%	465

Helium Buildup	1 year	50 years	100 years
Capsule Temperature - ^{o}F	1700	740	150
Helium Pressure - psia	730	6180	3270

5.8.3 ISOTOPE AVAILABILITY

Consideration has been given to isotope availability for the Mars 1969 Lander power supply in relation to requirements based on the following ground rules.

a. A full size isotope capsule, with real or dummy generator, is needed by 1 July 1967 for checkout of ground handling equipment. This capsule need not be flight qualified.

b. All capsules and generators must be available at the launch site by 1 December 1968.

c. A total of seven flight units is required, consisting of two flight Voyager systems, a back-up system, and a spare unit. (There are two Landers and hence two generators per Voyager system.)

d. Two units must pass through qualification.

e. The two isotope capsules which go through qualification tests can be used later for flight units. Generators which go through qualification tests cannot be used later for flight units.

With the above ground rules, a development schedule was prepared which endeavored to delay isotope delivery as long as possible. This schedule assumed a two months delay from the time the isotope was available until it was encapsulated and delivered to the point of required use. With this schedule, the following key isotope availability dates resulted:

For checkout of ground support equipment (1 unit)	- 1 May 1967
For 1st qualification unit	- 1 September 1967
For 2nd qualification unit	- 1 October 1967

For flight units - 1 per month starting 1 June 1968, with a total of 5.

A similar procedure was followed for later missions, except that it was assumed that no units would be required for checkout of ground support equipment or for qualification tests.

The resulting isotope requirements are shown on Figure 5.8-2 and compared with the availability estimates for Cm 244 from Reference 2. Also shown on Figure 5.8-2 are the effects of a two-year delay in isotope availability. There are recent indications, for example, from discussions with the Isotopic Power Branch of the AEC, that there might be such a delay in availability of Cm 244 and perhaps a similar delay in Pu 238. Figure 5.8-2 shows that, if such a delay indeed occurs, there will not be adequate Cm 244 available for the Mars 1969 Lander mission, although there will be enough for later missions.

In addition to the ground rules listed at the beginning of this section, it would be highly desirable, and perhaps necessary, to obtain delivery of the RTG units, without isotope capsules, beginning 1 July 1967 at the rate of two a month. In order to meet this requirement without excessive overlap of development and production and consequent undue risk of unsatisfactory flight units, it might be necessary to shift the required availability dates of the qualification unit isotope capsules ahead by several months. This would further aggravate the isotope availability picture.

As a matter of interest, similar schedules were prepared assuming use of isotope thermoelectric units for Orbiter power supplies. The required total isotope availability for Landers and Orbiters is also shown on Figure 5.8-2. It indicates that the isotope requirements are dangerously close to the availability estimates, assuming no slippage. This situation is believed to constitute too great a risk of missing the launch window flight date and forms a major basis for not recommending isotope thermoelectric generators as the Orbiter power supply for the Mars 1969 mission.

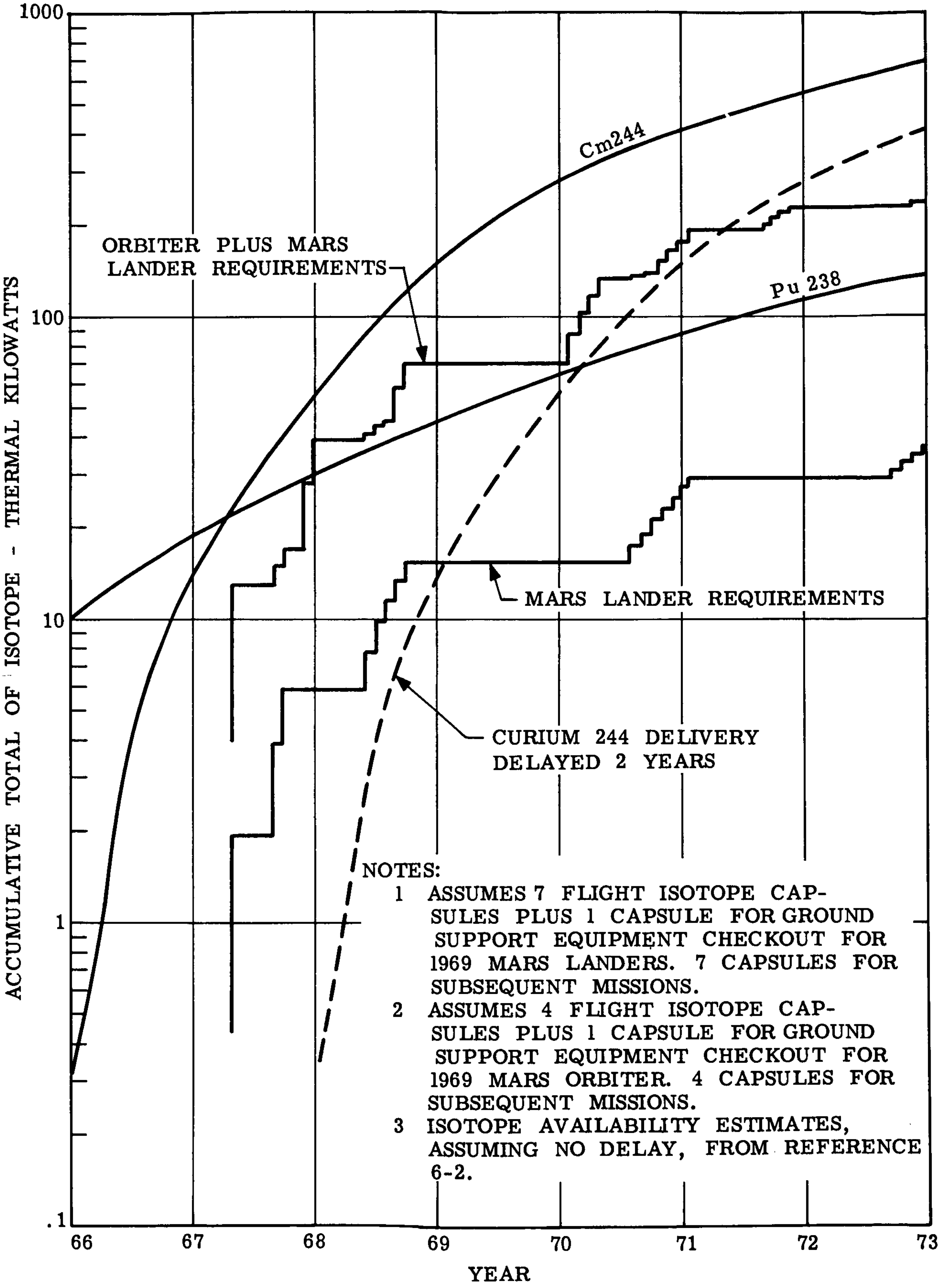


Figure 5.8-2. Comparison of Isotope Requirements and Availability Estimates

It is apparent from the foregoing that isotope availability in relation to requirements may be a serious problem and should receive immediate attention if a long lived Mars 1969 Lander is to be achieved. For, there is no presently conceivable way of obtaining mission lifetimes on the Martian surface longer than a few hours or days unless a radioisotope thermoelectric generator is used. Included in such considerations, of course, would be the possibility that availability of Pu 238 might be such that it should be used for the Mars 1969 Lander rather than Cm 244, even though a weight increase would result.

5.8.4 PROBLEM AREAS

The following problem areas require further resolution:

A. Solar Cells

(1) Radiation Environment

There is a high degree of uncertainty in the radiation environment to be expected; primarily that of solar protons due to solar flares, so that degradation estimates for solar cells due to this factor are open to serious question. Further effort should be made to resolve this uncertainty.

(2) High Temperature Degradation

There is some experimental evidence to indicate that solar cell performance will degrade significantly if cells are exposed to temperatures in the vicinity of 250°F or higher for long periods of time. Since solar cell operating temperatures of the present design for Venus Orbiters are in this range, this effect must be quantitatively evaluated early in the development program.

B. Batteries

This study has assumed that it will be possible to sterilize silver zinc batteries in time for use on the Venus Landers. Such a technique has not been demonstrated as yet. If it cannot be done, use of nickel cadmium batteries will probably be required, with a resultant increase in battery weight by approximately 17 and 37 pounds respectively for the 1970 and 1972 missions.

C. Isotope Generators

(1) Earth Safety and Planet Contamination Ground Rules

As previously mentioned, uncertainty in these ground rules has a major effect on performance estimates. This uncertainty should be resolved as soon as possible.

(2) Isotope Availability

There are indications that isotope availability may not be as great as once estimated. Significant slippage would make it impossible to use the isotope thermoelectric generator for the Mars 1969 Lander. Every effort should be made to establish firm isotope requirements as soon as possible and assure production of required quantities by the time they are needed.

(3) Helium Build-Up in Alpha Emitting Fuels

The amount of gaseous helium released during the decay of alpha emitting isotopes is uncertain. This amount has a significant effect on isotope generator design. Investigations now being made of this phenomenon, should therefore, be continued and perhaps accelerated.

REFERENCES

1. P. N. Haurlan, "Safety Requirements for Radiosotope Power Sources", Telegram to C. Taylor, 8 July, 1963.

2. Letter from Paul C Aebersold, Director, Division of Isotopics Development, Atomic Energy Commission, to G. E. Sweetnam, J. P. L. 25 January, 1963.

SECTION 6.0 EFFECT OF CHANGES IN ASSUMPTIONS ON SYSTEM PERFORMANCE

During the latter portion of the study, information was received from NASA that, as a result of some recent observations of Mars, the surface pressure was now believed to be much lower than earlier estimates have indicated and that the surface daytime winds may reach 200 ft/sec. Therefore, a brief study was conducted to determine the effect which the new model atmospheres would have on the design and capability of the Mars 1969 Lander.

In addition, an estimate was made as to the effect that a 10% reduction in Saturn C-1B payload capability would have on the overall system design and performance.

6.1 Effect of Low Pressure Martian Atmospheres

As indicated in Table 1.0-1, the new model atmospheres for Mars represent a reduction in surface pressure by almost a factor of four. Since retardation and parachute deployment were major problems even in the original atmospheres, both are severely aggravated by the new atmospheres. A reduction in the ballistic coefficient is required to partially alleviate them.

To achieve the lowest ballistic coefficient, the maximum drag shape known to be dynamically stable without active control devices was selected. This shape is a spherically blunted cone with a 52 degree half cone angle and a bluntness ratio of .47 (nose radius divided by base radius) which has been used successfully on the GE designed Mark II Ballistic Re-entry Vehicle. Having defined this shape and assuming the same payload density or total weight, a new ballistic coefficient of 16 lbs/ft^2 was obtained. With this value for the ballistic coefficient, the entry angle must be limited to values between 20° and 35° achieve the proper altitude for parachute deployment. (The results of the guidance analysis indicate that a 3-sigma accuracy of $\pm 4^\circ$ will be achieved).

Use of conventional parachutes and impact attenuation equipment in the low density atmosphere results in unreasonably high retardation system weights and impractical depths of crushable material. The use of retro rockets in conjunction with the parachutes and impact attenuation structure was found to result in a net weight saving of approximately 100 pounds. The reliability of the system using retro rockets is inherently lower because of the additional functions introduced in the retardation sequence and the sensing required to fire the retro rockets.

The shock attenuation system required for the 200 feet per second surface wind velocity will be 3 to 4 times the weight of the shock attenuation equipment required in the Mars 1969 Lander which was designed for 40 miles per hour. The design for the high wind velocities must be omnidirectional and will result in a data capsule concept. A vehicle design to withstand impact in such high surface winds is not considered within the present state-of-the-art and would require extensive development. Since it has been postulated that the high winds occur only during the daylight hours, landing during the hours of darkness are recommended to avoid the severe design penalties.

A weight statement comparing the Mars 1969 Lander and the low pressure vehicle design is given in Table 6.1-1.

The conclusions reached as a result of this study are:

1. An Entry/Lander can be designed for the low pressure model atmosphere without a significant sacrifice in payload capability.
2. In order to achieve the proper altitude for satisfactory parachute deployment without any serious reduction in payload, the entry angle must be limited to the range of 20° to 35°.

3. The proposed Lander system is somewhat less reliable than the Mars 1969 system because of the corridor restrictions and the more complex retardation system.

4. A design to accommodate a 200 ft/sec horizontal wind is not within the state-of-the-art and hence will require extensive development.

5. Because of the low entry angle, the line-of-sight problem between the Lander and the Orbiter will be serious unless the landing sites are in or close to the plane of the approach asymptote. This will restrict the choice of landing sites or the inclination of the orbit plane.

TABLE 6.1-1. WEIGHT STATEMENT

	Vehicle for Low Pressure Atmosphere		Voyager Mars 1969 Vehicle	
Shield		100		84
Structure		274		234
Aft Cover		66		57
Retardation & Crushup		280		288
Crushup Structure	126		133	
Parachutes	83		127	
Parachute Housings	11		12	
Programmers Batteries, etc.	12		8	
Harness	10		4	
Hardware	5		4	
Retro-rockets	33		-	
Ground Orientation		40		67
Separation		6		6
Payload		504		534
Thermal Control	72		72	
Power Supply	100		106	
Communications	145		141	
Scientific Payload	187		215	
TOTAL ENTRY WEIGHT		1270		1270
Adapter		71		71
ΔV Rocket		97		97
Spin System		12		12
Gross Entry Lander Weight		1450		1450

6.2 EFFECT OF 10% REDUCTION IN SATURN C-1B PERFORMANCE

At the request of NASA an estimate was made of the effect of a 10% reduction in the Saturn C-1B payload capability on the Mars 1969 system design. The problem was approached by removing items in the payload that made low contributions to the mission value in relation to their weight and re-sizing the vehicle or component for the lower weight and volume.

In the Lander, the subsurface drill, pulverizer and sampling handling equipment together weigh 50 pounds. This group of equipment obtains samples of subsurface material and presents them for observation to the petrographic microscope and the life detection group of instruments. Since surface debris can be scraped up and presented to the same instruments by much lighter equipment, eliminating these items would only reduce the number and range of available samples of Martian material, without either decreasing the number of experiments or reducing the breadth of determinations to be made in the Martian environment. With the elimination of these components, the Lander weight becomes 1300 pounds since there is also a savings in other components such as structure, heat shield, etc., due to a reduction in Lander diameter.

In the Orbiter, the high and medium resolution television and optical systems contribute pictorial information that refines and expands the photographic definition of a small portion of the surface area that is already mapped by the low resolution television mapping system. The 20 meter high resolution optics, and three color television cameras with associated electronics were removed from the payload. In their place a 140 Meter black and white television camera was added. The revised scientific payload package weighs 143 pounds and still provides a high percentage of the mission value obtainable from the original 215 pound package.

Additional weight savings in the Orbiter are realized by:

- Reducing the data rate by 40%
- Decreasing the PHP structure weight because of removal of part of the television subsystem
- Jettisoning the four end panels of the Orbiter structure prior to orbit insertion
- Reducing the propulsion system weight because of the smaller orbiter.

The resulting Voyager System Weight is given in Table 6.2-1.

TABLE 6.2-1. RESULTING VOYAGER SYSTEM WEIGHT

	10% Reduction in Total Weight	Original Mars 1969 System
Subsystem	Weight	Weight
Structure (less jettisonable panels)	336	419
Harness	106	106
Power Supply	190	217
Guidance and Control	226	226
Communication	287	291
Diagnostic Instrumentation	30	30
Thermal Control	87	87
Payload	143	215
Propulsion	414	467
Net Orbiting Weight	1819	2058
Jettisonable Panels	73	-
Gross Orbiting Weight	1892	2058
Orbit Insertion Fuel	1645	1962
Landers (Total)	2600	2900
Midcourse Fuel	190	210
	6327 lbs	7030 lbs

SECTION 7.0 STERILIZATION

7.1 REQUIREMENTS

It is required that the chance of releasing a viable micro-organism on the target planet shall be 10^{-4}. Only the Lander shall be considered; the probability of the Lander striking the target is assumed to be unity. The trajectory of the Orbiter will be such as to present less than 10^{-4} probability of planetary impact.

Sterilization shall be accomplished by thermal means. The use of Ethylene-oxide, as well as other gaseous and liquid sterilants, shall be restricted to organism "load" reduction or other secondary uses.

Inasmuch as sterile assembly is considered to present more than a 10^{-4} probability of organism survival, equipment, parts, components and structures shall be heated as a terminal measure for at least 24 hours to 135°C. Sterile assembly shall be limited to parts, components and structures that inherently cannot be heated to 145°C for three cycles of 36 hours. This latter temperature requirement is considered to be a qualification requirement. For qualification, parts, components and structure must show the required functional reliability (to be established for each mission) after a prescribed dormant period following thermal cycling.

All components, parts and structures, from the time of preliminary design, shall be classified as Class I, Class II or Class III.

Class I items can withstand three 145°C, 36 hour cycles at least once. They may, but not necessarily will, withstand further temperature soakings. These items may present problems, however, related to the organism-load factors.

Class II items will withstand only one thermal treatment; 135°C for 24 hours. An item may be Class II due to reliability questions.

Class III items will not withstand thermal cycling at prescribed sterilization temperatures.

An objective of the design program will be to eliminate all Class III and as many Class II items as possible from the flight hardware.

The three classes may be further subdivided. The basic philosophy of maximum practical sterilization treatment requires that from manufacture on, each part, structure and component will be subjected to the maximum sterilization environment without regard to the general sorting and classification of individual parts.

A list of known high-reliability parts has been carefully examined for thermal sensitivity. These parts and materials constitute the basic list for design engineers. The parts, assemblies and sub-assemblies chosen for flight hardware shall be qualified to both reliability and sterility standards. Where functions exist that cannot be performed by a space-qualified component, such functions may be omitted from the flight. Where mission success depends on a non-thermally qualifiable component, the item will be assembled by sterile techniques that will insure no more than 10^{-2} probability of contamination. The 24-hour final heat soak will continue to be a requirement for Class II items to assure the 10^{-4} standard.

Class III items are not expected to be a part of the final flight hardware. The appearance of a Class III item in the final design will constitute a serious problem to be solved only by detailed examination of that specific part and function.

7.2 PROCEDURE, PROCESSES AND DESIGN APPROVALS

Clean room requirements for this program are Class 10,000 or less. This manufacturing requirement is regarded as absolute.

During testing of the assembled vehicle, this requirement may be met by conducting the entire testing cycle within a pliable plastic container. It is not anticipated that Clean Rooms or containers will be sterile, nor will sterility be sought. During the testing cycles, the sealed portions of parts and components will not be compromised, and thus sterility problems will be limited to outer-surface areas. The final thermal treatment before sealing the protective flight container will sterilize these surfaces.

Class II and Class III spares and components, destined for assembly, require protection of a special nature. Soil contamination must be prevented as well as loss of sterility. Containers, designed to protect the items functionally, must include provision for maintenance of sterility and cleanliness.

A Sterility Contamination Control Group should review each design item. Each final machine assembly drawing will be approved and signed off by a member of this group. Design items will be checked for sterility interfaces, material compatibility, manufacturing processes and final packaging. During the assembly phase, as well as the manufacturing phase, critical steps and processes will be signed-off by a trained sterility inspector.

7.3 VERIFICATION

It is not practical to assay flight equipment. The adoption of a satisfactory method for sterilization, followed by a strict adherence to the system from manufacture through launch, is required to achieve 10^{-4} probability of a sterile vehicle.

At the time of equipment selection and qualification, selected components will be "seeded" during the manufacturing cycle. The "seeding" will utilize an organism resistant to thermal and gaseous sterilization. Microbiological assay will be conducted on these components and parts after trial assembly runs and sterilization tests to verify the efficacy of the prescribed sterilization procedure.

7.4 PRODUCTION

The sterility program requires a minimum of Clean Room assembly for all components. Class II and III parts and components require sterile assembly to insure not more than 10^{-2} probability of contamination. Each step of the manufacturing and assembly process will be monitored. Random samples of component parts will be selected throughout the manufacturing and production phase, and subjected to biological assay techniques. A rigorous personnel training and motivation program will be required.

Each step of the manufacturing process will be analyzed for sterility breaks. Packaging, transportation and storage represent areas where detailed procedures and minute attention to detail are the best assurance of a sterile end product.

7.5 FUNCTIONAL FLOW CONCEPT

Figure 7.5-1, the flow diagram of the sterilization sequence (factory-to-launch), shows the major operational steps of the sterilization plan.

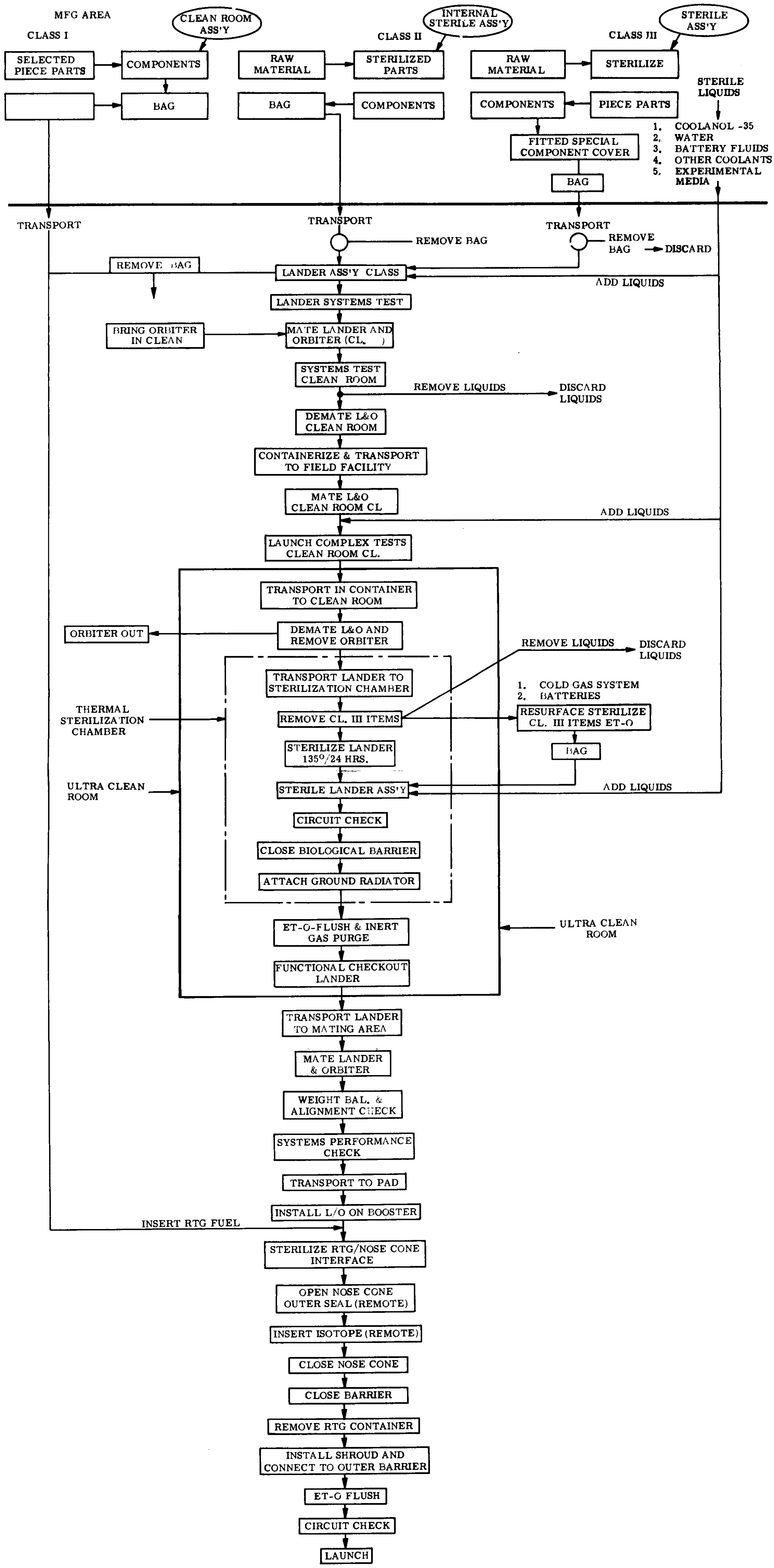


Figure 7.5-1. Flow Diagram of The Sterilization Sequence

SECTION 8.0 RELIABILITY

Throughout the Voyager Design Study careful attention was given to the reliability of the system and subsystem designs. Many system iterations were made to allow incorporation of redundancy, alternate operating modes to alleviate system effects of partial failures, and simplification of the design to provide higher reliability. Reliability apportionments were made and designs were modified whenever the predicted reliability was lower than apportioned. System design trade-offs were made by the reliability engineer, e.g., the trade-off between the longer trip time required by Type II trajectories and the redundancy made possible by the larger payloads permitted by this type trajectory. Reliability was considered in conjunction with the scientific mission analysis, and the Mission Scientific Value was maximized. Demonstration tests, parts selection, standards, quality control plans, sterilization techniques, etc., were all a part of the reliability analysis task.

8.1 PROGRAM IMPLICATIONS OF RELIABILITY ANALYSES

The results of reliability analyses with the incorporation of redundancy in the various subsystems as definitized by this study report clearly indicated the following:

<u>Practicability</u>

A Voyager System undertaken along these lines is practicable and is expected to be well within the attainable "state-of-the-art" applicable to the program plan as provided in Volume VI of this report.

<u>Mission Success Probability</u>

Mission success will be attainable in three out of every four flight opportunities. Mission success is defined as the successful return to earth of at least 75 percent of the scientific value of the many scientific instruments carried. A flight opportunity presumes the successful operation of the launch vehicle.

<u>Meticulous Engineering Efforts are Required</u>

The attainment of these successes will require the thorough and consistent application of (a) design standards, (b) selected materials, parts and processes, (c) safety factors, margins and allowances, (d) extensive testing, screening and evaluations for design development and quality assurance, (e) meticulous attention to detail in all portions of the program, (f) adequate time in the schedule for incorporation of the results of tests and evaluations into the design and production of the actual flight hardware, and (g) the reliability-life testing of final flight hardware components, subsystems and systems to demonstrate and verify that the designs and flight hardware actually launched are of the quality required.

<u>Sterilization Effects on Reliability</u>

It is considered that sterilization requirements can be satisfactorily achieved and sterilization (and resterilization) of all Lander components can be performed immediately prior to launch without adversely affecting the attainment of the system reliability and mission success noted above.

(a) Such attainment in practice will be best assured by the immediate or early redesign and development of these components, modules and material formulations and processes which this study identified as not being presently adequate for sterilization by "dry heat" procedures.

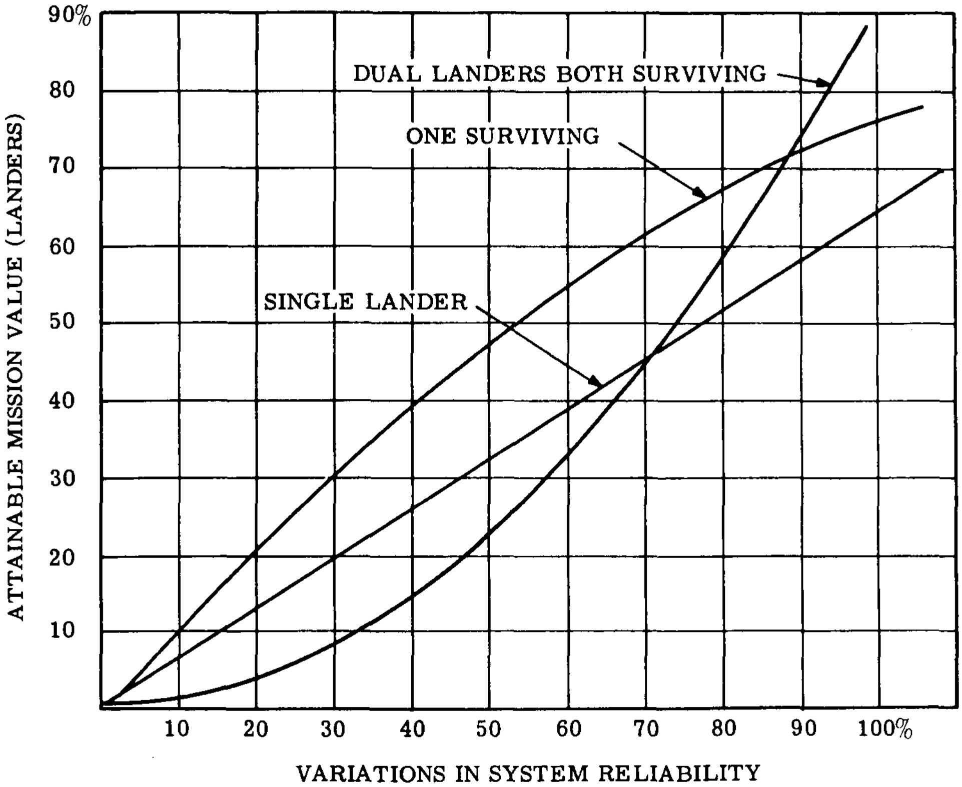
(b) It is significant to note the high percentage of the total complement of parts required for a Voyager System which are now fully sterilizable by "dry heat" procedures. Also, that in every instance in which redesign and development effort has been authorized or in which a detailed feasibility study carried out, a suitable solution or method of approach to redesign has been found.

(c) The verification of the performance capability and reliability of all sterilized components and component parts, materials and processes by reliability-life tests is considered essential to the success of this program.

Provision for Design Uncertainties

A review of available information both published and unpublished relative to the nature of the surface environments, atmosphere, etc., of Venus and of Mars has been made in the course of this study. With regard to survival of surface impact and subsequent operation by the Lander(s) on the surface of Mars, the terrain is indicated as being 75 percent desert. The degree to which rugged, rock filled canyons or other terrain may exist which is of such a nature as to present a high probability of serious, permanent damage or entrapment of the Lander(s) is not determinable.

Figure 8.1-1 provides graphic illustration of the effect of the super-position of any such variation in overall system reliability, and provides a direct comparison of the most probable results obtainable with each of the system configurations shown. The mission value of the complement of scientific instruments in each of the Landers was both complete and identical. The term "Attainable Mission Value" represents the product of the appli-



NOTE: MARTIAN TERRAIN RELIABILITY ... "T" ... WAS INCLUDED IN ATTAINABLE MISSION VALUE ANALYSES AS ... 90%

Figure 8.1-1. Attainable Mission Value (One Month) versus Variation in System Reliability cable System Reliability and the Value (in percent) of the Lander's scientific instruments as a portion of the total Value (100 %) available with one completely successful Lander, together with one completely successful Orbiter. The vertical line at 90 percent Terrain Reliability (this was considered to be a more than adequate contingency since no more than 10% of the Terrain should be of such adverse nature as to seriously damage a Lander System of the design proposed by this study for the Voyager System) thus provides a 10 percent contingency for the unknowns of Martian terrain. The effects of any other contingency which further design, development, experiment or analysis might make pertinent to Voyager could be considered directly on this chart.

It would appear that the results obtainable from the high reliability for at least one Lander surviving from a dual Lander system exceeds the value of a single Lander system for all superposed reliability considerations.

It is also of interest to note that the mission value attainable, because of the second set of instruments and unique location and environments of the second Lander of a dual Lander System, is sufficient to compensate for the higher risks involved (i. e., lower probability) in having both Landers of the dual system survive impact and provide fully satisfactory performance during the first month after arrival on the planet. At a system reliability (including terrain effects) greater than 75 percent of those which have been calculated as best estimates (i. e., most likely values) for the proposed Voyager dual Lander configuration, this compensation is sufficient to make it of greater value than the single Lander configuration. Also, at a system reliability greater than 88 percent of that applicable to the proposed design, the attainable value with dual Landers - both surviving exceeds that for dual Landers - one surviving.

System Reliability Analyses Comparing One Lander vs Two Landers Systems Configuration

To establish quantitatively the Attainable Mission Value which each configuration can achieve, each of the scientific instruments was reviewed by the responsible scientists, systems and reliability engineers, and a portion of the total value of the mission's full complement of instruments was apportioned to each instrument. These instrument "available" values and their accrued value at given times after arrival together with detailed methods of analysis are documented and described in Volume II, Section 4. 5. 1.

Thirty percent of the "Available" mission value for each system was assigned to data from the Orbiter, 10 percent to the Entry data from the first Lander and 60 percent to the Surface data from the first Lander. The surface data from a second Lander with identical instrumentation was considered to be of equal value, namely 60 percent. However, because the entry data from a second Lander would largely represent a duplicative set of readings (not being "geographically" unique by reason of its location), this value from a second Lander was reduced from 10 percent to 5 percent. The total "available" values are shown in Figure 8. 1-2.

When each instrument value (or value increment over a given time) has been multiplied by the system reliability at the same time ranked in order of greatest attainable value per pound, and applied in that order to the net payload capability of a Lander (and correspondingly for each of two Landers), an attainable mission value vs. Entry/Lander weight is obtained as shown in Figure 8. 1-3. From this type of analysis, including as it does the cumulative effects of all the earlier performance, weight, scientific and reliability tradeoffs, a decision point (e. g., 1840 pounds total weight available for one or two Landers) is clearly shown. Below this point a one Lander system is advantageous. Where available Lander(s) weight is greater, the marked advantages of dual Landers are shown.

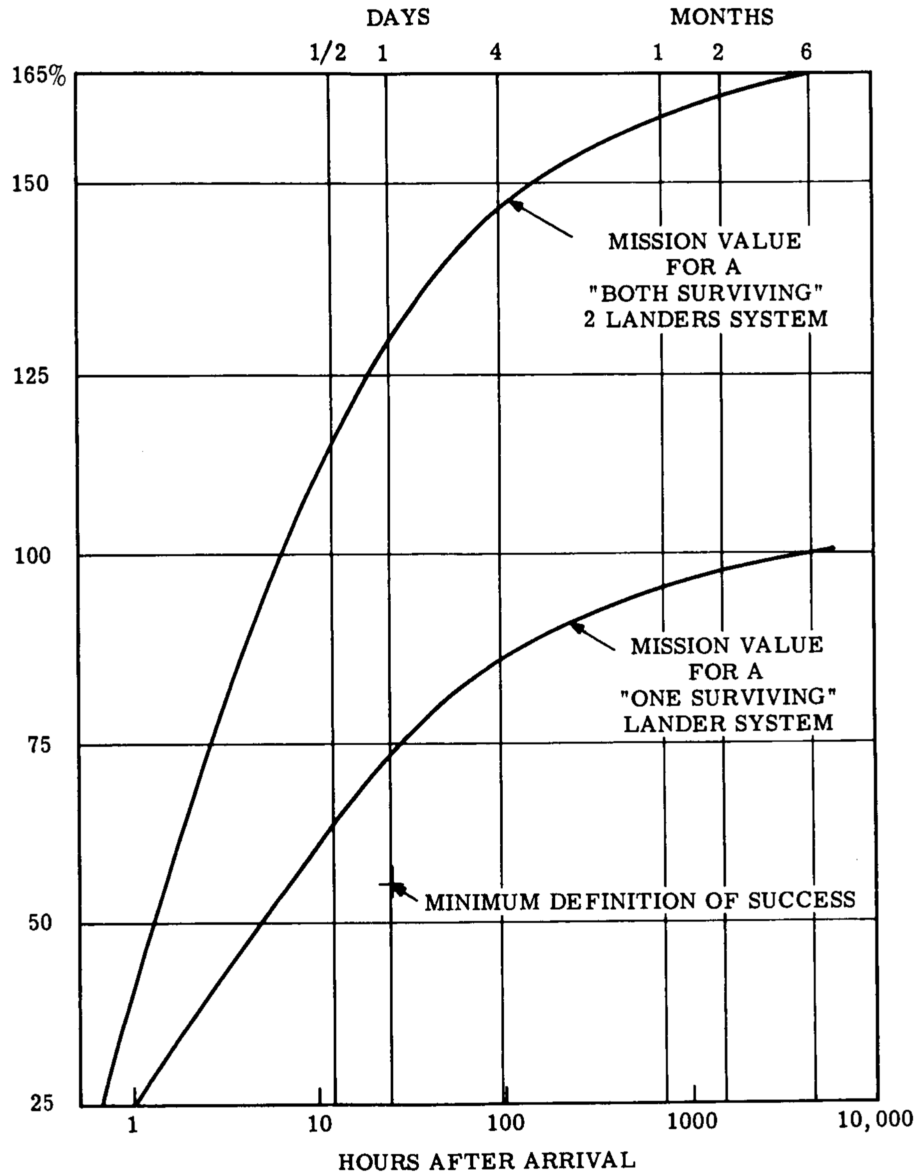


Figure 8.1-2. Mission Scientific Value

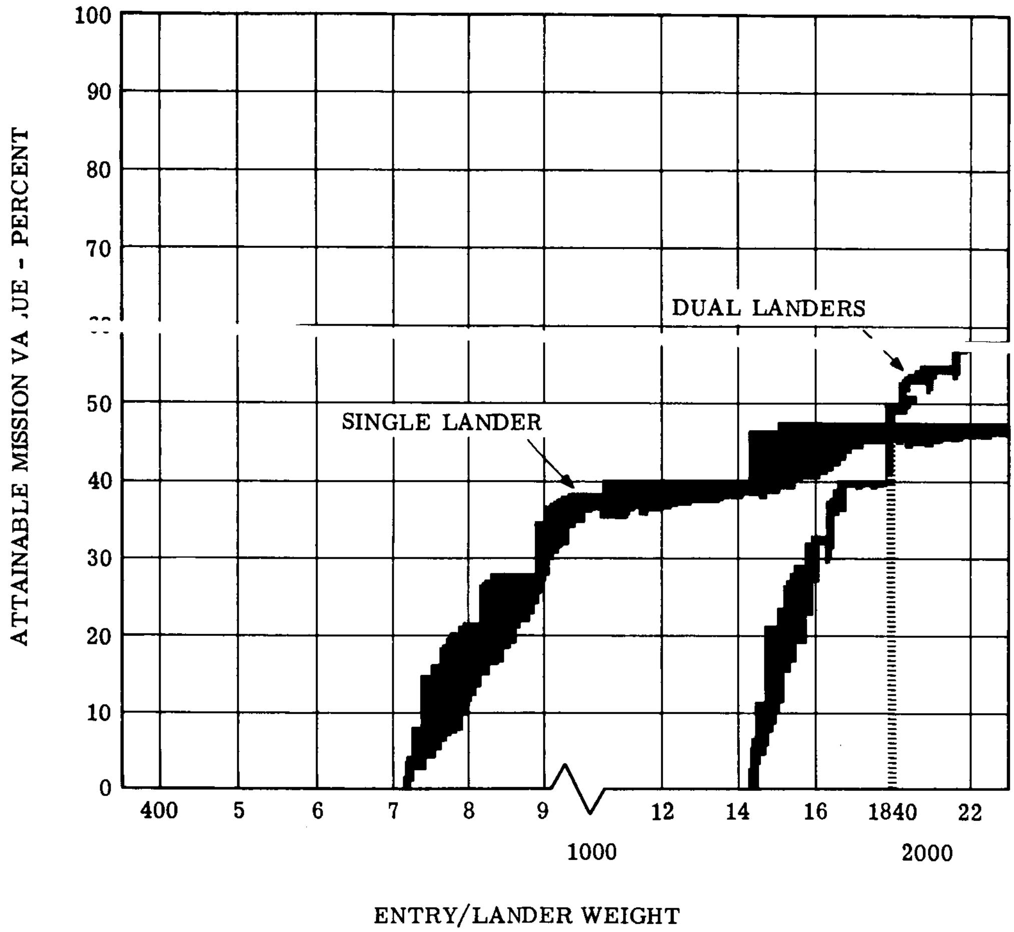


Figure 8.1-3. Attainable Mission Value at 24 Hours

One of the many alternatives studied covered a reduced Lander communications rate capability in which the power supply plus communications weight was reduced by 100 pounds. The effect of this was to reduce the decision point weight from 1840 to 1200 pounds.

As total Lander weight is available in the decision point, the relative value of the data from the second Lander may be established at less than that of the first Lander, with dual Landers still being preferred.

At later opportunities, should surface roving vehicles or other key scientific capabilities be considered to be a major portion of the total mission objective, a single Lander providing the required payload weight capability may be of first importance. Also, if low atmospheric densities (11 millibars) should greatly reduce the net payload capability of a dual Landar System, a single Lander System may be preferred. However, for the Voyager Systems design proposed by this final report, and for those studied in arriving at the finally recommended design, two Landers would appear to be definitely advantageous over a single Lander System.

Reliability Requirements and Apportionments

From the initial analyses made under this study, it was concluded that the Voyager System should be required to demonstrate a reliability and confidence prior to launch which would assure that mission success would be achieved in at least 3 out of 4 launches as noted above. This corresponds to a reliability of 65 percent, based upon exponential tables and methods of analysis with a lower limit confidence of 50 percent. The system proposed with two Landers is capable of meeting this requirement.

In the columns of the Reliability Management Matrix provided in the Reliability Program Plan of Volume VI, a detailed reliability estimate to subsystem and component level is provided. Since these calculated values are based upon the exclusive use of high reliability parts, materials, processes, etc., which have been qualified and controlled by the best known techniques, it is felt that they are representative of the best demonstrable levels.

8.2 MISSION EFFECTS OF OTHER RELIABILITY FACTORS

The success of the entire Voyager program is dependent upon many elements not included within the scope of the Voyager Spacecraft System for which the detailed reliability analysis and study has been prepared and documented in this report. A very significant, and perhaps the dominant factor of such elements, is the performance capability and reliability of the launch vehicles themselves.

Probability of Success

The probability of success for any launching for a Voyager is the product of the reliabilities of the Voyager System and those of the launch vehicle, etc. of which the data in Figure 8.2-1 are considered to be representative.

(a) If the 81 percent (or greater) successful launchings are to be considered applicable to each of the Mars 1969 opportunities, a period of operational testing of the components of the 1969 launch vehicle comparable to that which has been true of the components used in the systems from which Figure 8.2-1 was plotted must be provided. Without such opportunity to assure this probability of launch success, the probability of mission success of the Voyager System cannot be verified.

(b) It is equally important that a comparable opportunity to demonstrate the performance capabilities and reliability of the Voyager Vehicle System be provided prior to launch if mission success is to be assured.

Launch Opportunities

The launch opportunities periods are of limited duration and occur at widely separated intervals. Thus, any lack of readiness to launch upon demand could well consume major portions, if not all, of any given opportunity period.

The two year slippage of the program opportunity which would result from such an interruption or delay could be very costly, including as it would not only the cost elements included in the Voyager Spacecraft cost, but also large elements of the costs of the booster system, RTG and scientific payload, etc. involved in the launch preparations. It is expected that the administrative and financial significance of such slippage costs will make mandatory the launch of the best Voyager System operable, provided that such criteria as immediate performance and sterilization are satisfied.

Time is the most critical factor in the reliability area. Since the demonstration and verification of the reliability of the system design can only be begun when representative development hardware is available for that purpose, the most effective action which can be undertaken to increase the assurance of success is to advance the rate at which definitive design and development work is undertaken.

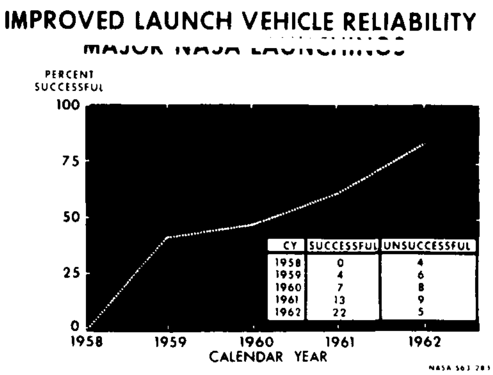


Figure 8.2-1. Improved Launch Vehicle Reliability - Major NASA Launchings

8.3 GENERAL

Titan IIIC Orbiter With Separately Launched Titan IIIC Dual Landers

Section II of this volume indicates that an Orbiter payload capability of 223 pounds is available for an Orbiter (with no Landers) launched by a Titan IIIC. This provides an Orbiter with complete communications and mapping capability. Its available mission value is 30 percent.

The same results also indicates a gross Lander weight of 2230 pounds in the Bus/Lander system for Titan IIIC. Since this is considerably greater than the 1840 pound point in Figure 8.1-3, above which it is of clear advantage to apply dual landers, the payload capability of each of two landers having a gross weight of 1115 pounds has been determined as 85 pounds. This is sufficient to include Instrument No. 24. For such a dual lander combination and using the same reliability values for the Titan as for the Saturn dual lander system, as noted above, the Attainable Mission Value is 63.5%.

Figure 8.3-1 graphically presents a comparison of the relative probability for success of the Saturn and Titan IIIC launched Voyagers.

Other combinations of scientific instruments in given orbiters or landers can be made to obtain specific information but which will lower the Titan IIIC curve somewhat from its maximum position shown in Figure 8.3-1. Many alternatives with notable flexibility and adaptability are available. On an Attainable Mission Value basis, there appear to be very significant advantages to the use of the Titan IIIC.

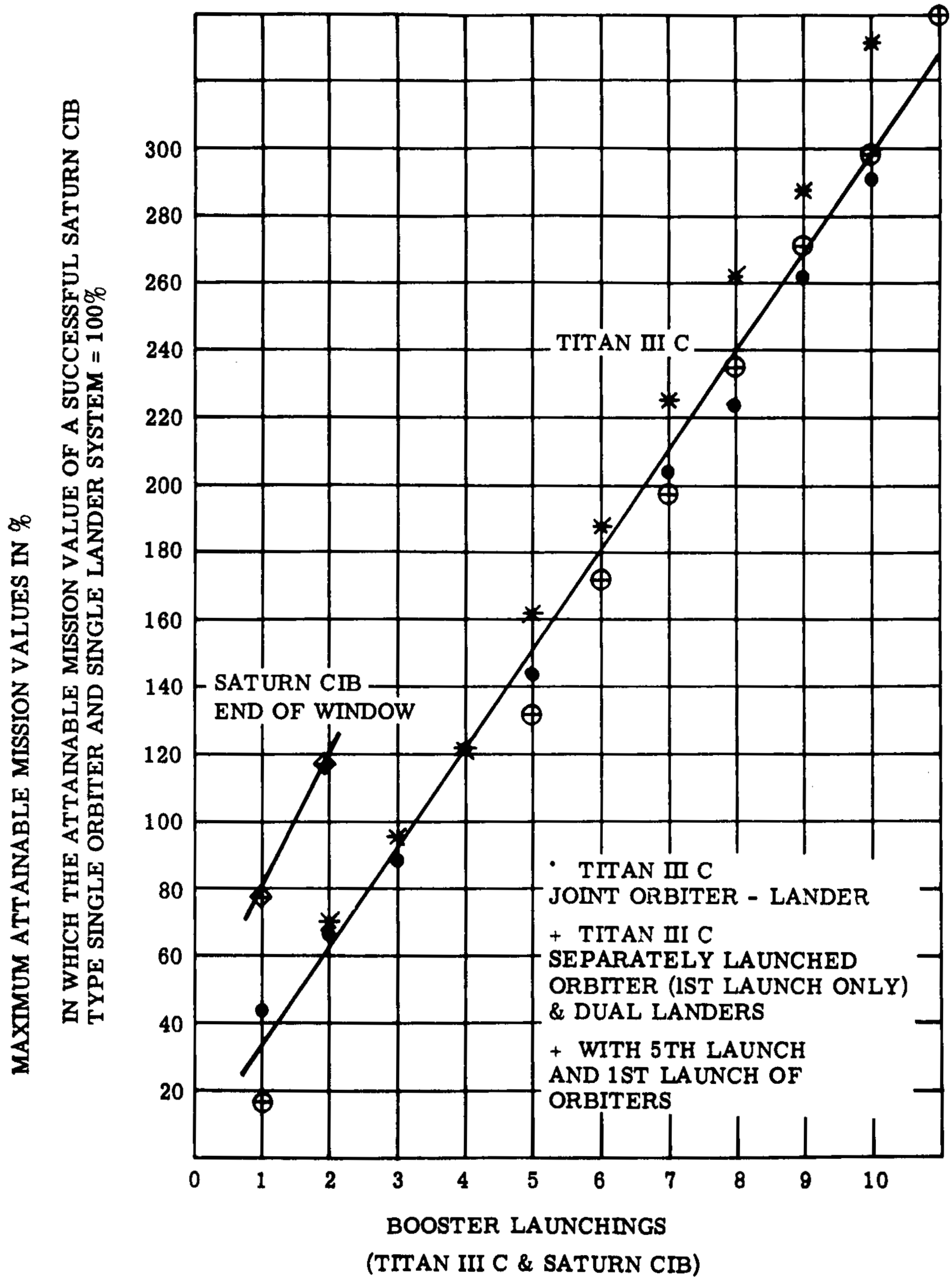


Figure 8.3-1. Attainable Mission Values for Multiple Launchings of Saturn versus Titan Booster Voyager Systems

8.4 CONCLUSIONS

During the Study Contract, the Reliability & Quality Assurance requirements for the Voyager System and Program have been considered in detail. The technological data, methods and approaches, and the organizational and management alternatives and procedures necessary for the successful attainment of the Voyager Program's objectives have been examined and evaluated. The conclusions and recommendations are:

1. That quantitative reliability requirements for the Voyager System be established, and that the capability of each component and subsystem to fulfill these requirements be demonstrated by tests prior to the scheduled launch of a Voyager System.

2. That these reliability requirements be so defined as to assure that an average of three (3) successes out of four (4) opportunities may be expected to be attained by the Voyager System.

3. That the demonstration of these reliability capabilities over the extensive time periods required for interplanetary flight and planetary operations be provided only at basic part, material and process levels.

4. That reliability demonstration tests for all ranges of variation and extremes of environment which may be anticipated during an actual mission be conducted at "system level".

5. That the suitability of each component to enter a systems test be demonstrated prior to its incorporation into any system intended for qualification, acceptance, or reliability demonstration tests. This demonstration should include satisfactory operation under thermal-vacuum conditions for not less than 150 hours of active operational testing, of which the last 100 hours are required to be FAILURE FREE as a condition for the satisfactory conclusion of the demonstration.

   And that the suitability of each system be required to be demonstrated by test prior to its acceptance for shipment and use. This demonstration should include the satisfactory operation of each component of the system under thermal-vacuum conditions for not less than 1000 hours, of which the last 700 hours are required to be FAILURE FREE as a condition for the satisfactory conclusion of the demonstration.

6. That the final design of the Voyager System be optimized, to provide a maximum "system effectiveness" per launch.

7. That the apportionment of reliability requirements to each Voyager System component and subsystem be based upon detailed reliability analyses of all significant system elements, and that the effects of the following items on system reliability be considered: (a) design margins, (b) "worst case" design limits, (c) alternative or "back-up" modes of operation, and (d) redundancy at piece part and component levels.

8. That, this apportionment be optimized during preliminary and final design to attain maximum reliability per pound of weight of the system.

9. That Approved Parts and Materials Lists and Design Standards be established for the Voyager Program, and that conformance of the Voyager System design to the Approved Parts, Materials and Standards be incorporated as requirements.

10. That each contractor and subcontractor be required to establish suitable facilities and organization to assure the accomplishment of the reliability requirements of the Voyager Program, and that this be documented as a portion of that contractor's Reliability Program Plan in accordance with NASA document NPC 250-1.

11. That for the Mars 1969 opportunity, two (2) Landers be used in each Voyager System.

12. That for the Mars 1969 opportunity, direct communication to earth be provided on each Lander in addition to the communications provided via the Orbiter.

13. That a cost effectiveness study be conducted, prior to preliminary design, which would compare total mission reliability, scientific value, and costs for programs utilizing the Titan IIIC and the Saturn C1-B launch vehicles.

14. That the Voyager Lander Parts Sterilization Compatibility Program be initiated as a part of the preliminary design phase of the Voyager program.