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VOYAGER SPACECRAFT SYSTEM STUDY

(PHASE I-TITAN IIIC LAUNCH VEHICLE)

FINAL REPORT

VOLUME I

SUMMARY

7 AUGUST, 1964

Prepared Under Contract 950847

for

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
4800 OAK GROVE DRIVE
PASADENA, CALIFORNIA

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SPACECRAFT DEPARTMENT
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VOLUME I

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1. INTRODUCTION

Presented in this report is a summary of the results obtained from a conceptual design study for a Voyager spacecraft, to be launched by the Titan IIIC launch vehicle, to perform orbiting and landing missions to Mars during the opportunities from 1971 to 1977.

The objectives of the study were to:

1. Conduct a conceptual design of both a Bus/Lander and an orbiting spacecraft for Mars 1971.
2. Estimate spacecraft performance for Mars 1973, 1975 and 1977 and for Venus 1972.
3. Estimate the performance for a combined Orbiter/Lander system.
4. Estimate the cost and the development cycle for the Mars 1971 systems.

In conducting this study maximum utilization was made of the work performed during the Voyager Design Study (NASA Contract NASw-696) which assumed a Saturn 1B + SVI launch vehicle. This approach was taken so that the results of the two studies would be on the same basis, thus permitting a valid evaluation of the Titan IIIC and the Saturn 1B + SV launch vehicles for the Voyager mission.

The emphasis in this study was placed on the Bus/Lander and the Orbiter systems since the prior Voyager Design Study indicated the combined Orbiter/Lander system to be rather inefficient in the weight class (3600 pounds) associated with the Titan IIIC launch vehicle.

In the design of the Bus/Lander system the model atmospheres used were the ones characterized by a 11 to 30 Mb surface pressure.

2. MISSION AND SYSTEM ANALYSIS

The activities in this portion of the study consisted principally of adapting the results of the Saturn 1B Voyager study, specified in detail in the Final Report of Contract NASw-696, 15 October 1963, to the spacecraft systems and configurations considered for the Titan IIIC launch vehicle. The 1971 opportunity is the prime mission for the Titan IIIC systems, while 1969 was the prime opportunity in the prior study. Prior results were modified, revised, or ratioed, as required to suit the system capabilities and requirements of the Titan IIIC spacecraft concepts.

2.1 SUMMARY OF RECOMMENDED SYSTEMS

Separately boosted Bus/Lander and Orbiter systems are recommended for maximum attainable mission value when using the Titan IIIC launch vehicle. A performance summary for a Bus/Lander and all Orbiter mission for Mars in 1971 is tabulated in Table 2.1-1. A summary of the performance of the Orbiter/(2) Landers combination system for the Saturn 1B +SVI third stage for the 1969 Mars opportunity (data from the prior Voyager study) is also listed for comparison purposes.

2.1.1 BUS/LANDER

The Bus/Lander system is restricted by the diameter of the Titan III shroud. The standard diameter of 120 inches only allows a Lander entry weight of 1380 pounds based on a ballistic coefficient of 15 lb/ft^2 and an entry corridor of 20 degrees to 35 degrees. These entry parameters have been selected on the basis of expected 1965 state-of-the-art in retardation systems and prior estimates of Voyager guidance system accuracy.

The weight of this Lander (1380 pounds) and a transit Bus do not utilize the injection capability of the Titan IIIC launch vehicle in any of the opportunities under consideration in this study. Consequently larger shrouds accommodating larger lander diameters or some other means, such as extensible flaps, must be provided to increase available drag area, in order to more fully utilize the Titan IIIC energy.

Early in the study, a consideration of injection energy requirements and trip times for the 1971 through 1977 windows showed that a 2000 pound Lander could be launched in each opportunity. (See Table 2.1-2.) Because this was attractive for manufacturing, and development purposes, this Lander weight and size (134-inch base diameter) was selected for the prime Bus/Lander mission.

This requires the development of a 144-inch diameter shroud. Subsequent investigation of movable flaps that would enable the same 2000-pound and even heavier Landers to be launched within the standard shroud diameter revealed very reasonable weight penalties and other attractive program aspects. The development costs and problems of flapped Landers should be traded against costs and problems of the 144-inch shroud before making final decisions, but this work could not be accomplished in this study.

The capability of the Bus/Lander mission is summarized in Table 2.1-3.

TABLE 2.1-1. SYSTEM PERFORMANCE SUMMARY

	Titan IIIC			Saturn 1B SVI
	Bus-Lander	Orbiter	Orbiter/ Lander	Orbiter/ Lander
Injected Weight (lb)	2546	3600	3600	7030
Lander Weight (lb)	2042	—	1284	1450/ 1450
Lander Scientific Payload (lb)	387	—	110	211/ 211
Orbiter Weight (lb)	—	1815	1440	2059
Orbiter Scientific Payload (lb)	—	347	123	215
Orbit (n. mi.)	—	1000 x 2278	1000 x 19,000	1000 x 19,000

TABLE 2.1-2. ENTRY/LANDER WEIGHT RESTRICTIONS

- Injection Energy Requirements
- Reasonable Trip Times
- Shroud Diameter

	Trip Time (Max.) (Days)	Max. Lander Weight (Pounds)	Type Traj.
1971	225	2960	I
1973	195	2042	I
1975	420 336	2300 2042	II
1977	387 297	2570 2042	II

Shroud Diameter (Inches)	Max. Weight (Pounds)
120	1380
144	2042
170	2960

TABLE 2.1-3. BUS/LANDER CAPABILITY

Bus		455
Lander		2042
Entry Weight	1830	
Scientific Payload	387	
Fuel		49
Injected Weight		2546 Pounds

PAYLOAD

Biological	Geophysical-Geological	Atmospheric
Growth	Surface Penetrability	Temperature
Metabolic Activity	Soil Moisture	Pressure
Existence of Organic Molecules	Seismic Activity	Density
Existence of Photo-autotroph	Surface Gravity	Composition

TABLE 2.1-3. BUS/LANDER CAPABILITY (Cont'd)

PAYLOAD (Cont'd)

Biological	Geophysical-Geological	Atmospheric
Turbidity and PH Changes		Altitude
Microscopic Characteristics		Light Level
Organic Gases		Electron Density
Macroscopic Forms (TV)		
Surface Sounds		

+

Surface Roving Vehicle

2.1.2 ORBITER

The Titan IIIC launch vehicle can boost an All-Orbiter mission that can attain a favorable 1,000 n.mi. circular orbit with the same scientific weight (215 pounds) as the Orbiter in the prior study in the 1971 Mars opportunity. However, because the short orbit period caused problems in high data rate and power supply weights, the orbit was made slightly eccentric. The final orbit, 1000 x 2278 n.mi. was selected on the basis of a relationship of the orbit period to the Mars rotational period which minimizes the probability of repeating surface tracks in a synchronous fashion.

The weights and scientific mission of this Orbiter are summarized in Table 2.1-4. The payload allowance for this orbit is large enough to permit mounting a solid rocket motor to reduce the periapsis altitude after completion of the initial map, and high resolution optics on the Orbiter in order to obtain ~ 3 meter resolution pictures of a portion of the surface.

Variation of Orbiter performance in subsequent opportunities is tabulated in Table 2.1-5.

TABLE 2.1-4. ORBITER CAPABILITY

WEIGHT STATEMENT

Orbiting Weight		1815	
Payload	347		
Fuel (1000 x 2278 N.Mi.)		1634	
Injected Weight		3449	pounds
Adapter and Δ Shroud		151	pounds
		3600	pounds

TABLE 2.1-4. ORBITER CAPABILITY (Cont'd)

Scientific Capability		
Television	1 KM	Stereo Map
	140 M	Blue
		Red
		Green-Yellow
	3-20M	B&W
Upper Atmosphere Composition & Density		
Ionosphere Profile		
Particles & Fields		
UV & IR Radiation		

TABLE 2.1-5. VARIATION OF ORBITER PERFORMANCE WITH OPPORTUNITY

Year	1971	1973	1975	1977
Launch Weight (lb)	3600	2850	3100	3200
Injected Weight (lb)	3449	2699	2949	3049
Orbiting Weight (lb)	1815	1815	1815	1815
Orbit (n. mi.)	1000 x 2278	1000 x 20000	1000 x 11500	1000 x 3400
Trip Time (max.) (days)	225	202	385	332

2.2 SCIENTIFIC MISSION AND PAYLOADS

2.2.1 GENERAL

This portion of the study was mainly the adaptation of the results of the mission analysis developed in the prior Voyager study for the Saturn 1B with the S VI third stage, under contract No. NASw/696, to the mission capability and specific requirements of the all Orbiter, Bus/Lander and combined Orbiter/Lander spacecraft systems when boosted on a Titan IIIC launch vehicle in the 1971 Mars launch opportunity. No additional experiments or instruments were considered in the same detail as in the prior study. Some rearranging of instruments and priorities was done as was deemed appropriate for a particular system. Payload complements for Orbiters and Landers for the series of missions in the prior study were combined, duplications were eliminated, and priorities were arranged in descending order for instruments originally planned to fly in later years after 1969 in the prior study.

2.2.2 BUS/LANDER

Roving capability was incorporated in the large payloads of the Bus/Lander entry vehicle, utilizing weight estimates obtained in other GE/MSD study efforts. Roving surface vehicle weights were charged against scientific payload capacity. The sample sites on the surface that can be reached by a Rover were considered as adding an amount of mission value that declined with each successive site and were incorporated in the estimates of attainable mission value used for the purpose of comparing effectiveness of the Titan IIIC launched systems with those launched on the Saturn 1B/S VI in the prior study.

Otherwise, the payload of the 2042-pound Lander is essentially the same as for the 1969 Mars 1450-pound Lander in the prior study. Individual atmospheric constituent gas analyzers were substituted for the mass spectrometer in the interest of reliable discrete data. (See Table 2.2-1.)

TABLE 2.2-1. SCIENTIFIC PAYLOAD FOR 2042-POUND LANDER

Priority	Name	Inst. No.	Weight (Pounds)	Accum. Weight (Pounds)	Power (Watts)
1.	Temperature	I-24	0.3	0.3	0.07
2.	Sounds	I-34	0.5	0.8	1
3.	Pressure	I-17	0.3	1.1	0.10
4.	Density	I-20	1.5	2.6	2
5.	Multiple Chamber	I-54	4.0	6.6	2
6.	Surface Penetration Hardness	I-25	4.5	11.1	0.1
7.	Photoautotroph	I-62	3.0	14.1	1
8.	Light Intensity (Sun Sensor)	I-84	0.5	14.6	0.1
9.	Composition, H_2O	I-44	1.5	16.1	1
10.	Composition, O_2	I-45	1.5	17.6	1
11.	Turbidity & PH Growth Detector	I-53	4.0	21.6	1

TABLE 2.2-1. SCIENTIFIC PAYLOAD FOR 2042-POUND LANDER (Cont'd)

Priority	Name	Inst. No.	Weight (Pounds)	Accum. Weight (Pounds)	Power (Watts)
12.	Wind Speed and Direction	I-67	2.0	23.6	0.5
13.	Gas Chromatograph	I-8	7.0	30.6	4.5
14.	Composition, N_2	I-48	1.0	31.6	1
15.	Composition, CO_2	I-49	1.0	32.6	1
16.	Soil Moisture	I-70	2.0	34.6	25
17.	TV Camera, Panorama TV		20.0**	54.6	20
18.	Radioisotope Growth Detector	I-19	6.0	60.6	3
19.	Composition, O_3	I-46	1.5	62.1	1
20.	Composition, A	I-47	1.5	63.6	1
21.	Precipitation	I-36	1.0	64.6	1
22.	Electron Density (Langmuir Probe)	I-39	3.0	67.6	3
23.	Surface Gravity	I-72	3.0	70.6	3
24.	Radar Altimeter	I-5	15.0	85.6	25
25.	TV Microscope and Subsurface Group	I-71	75.0	160.6	200
26.	Seismic Activity	I-21	8.0	168.6	1

**Incl. 10 pounds TV Deployment

2.2.3 ORBITER

The instrument complement from the prior study for the 1969 opportunity was modified by the deletion of the nadir vidicon camera and by the addition of a "retro rocket and high resolution package." This provides a means of lowering the periapsis altitude of the orbit after acquisition on the initial map and an additional telephoto lens on the 20-meter image orthicon camera in order to obtain 3-to 7-meter resolution pictures of a small area of the planet. (See Table 2.2-2.)

TABLE 2.2-2. ALL-ORBITER PAYLOAD

No.	Name	Inst. No.	Weight (Pounds)	Accum. Weight (Pounds)	Power (Watts)	Accum. Watts	Origin. Planned to Fly in Yr.
1.	Magnetometer	I-23	5	5	5	5	1969, 1971, 1973
2.	IR Multi-channel Radiometer	I-2	3	8	3	8	1969, 1971
3.	Solar Multi-channel Radiometer	I-79	3	11	3	11	1969, 1971
4.	Television 4 IO 2 Vid.		115	126	(140)	(151)	1969, 1971
5.	Charged Particle Flux Geiger Tubes & Ion Chamber	I-12	55	132	1	12	1969
6.	Far UV Radiometer	I-96	6	138	3	15	
7.	Micrometeroid Flux	I-55	8	144	1	16	1969
8.	Bi-Static Radar (Ionospheric Profile)	I-85	13	159	2	18	1969
9.	Polarimeter-Skylight Analyzer	I-68	4.5	163	4.5	23	1969
10.	IR Spectro meter	I-1	29	192	7	30	1969
11.	Retrorocket & Hi-Resolution Package		146	338			
12.	Mass Spectrometer	I-43	6	344	6	36	1973
13.	Electron Probe (Langmuir Probe)	I-39	3	347	3	39 + (140 for TV) = 179w.	1973

2.2.4 ORBITER/LANDER COMBINATION

The instrument complement for this small Orbiter was set at an arbitrary 123 pounds which is almost one half of the 215 pounds provided on the Orbiter in the prior study. The 20-meter resolution image orthicon television camera of the prior study was eliminated and only the most essential particle and field detectors are carried. (See Table 2. 2-3.)

The Lander payload is based on the capability of the Lander size that could be launched together with the Orbiter carrying the payload shown in Table 2. 2-4 in an orbit of 1000 x 19,000 n. mi.

The Lander payload is shown in Table 2. 2-4.

TABLE 2. 2-3. SCIENTIFIC PAYLOAD FOR ORBITER OF ORBITER/ LANDER COMBINATION

Priority	Name	Inst. No.	Weight (Pounds)	Accum. Weight (Pounds)	Power (Watts)	Year Originally Planned to Fly
1	2 Vidicon Cameras	TV	83.0	83.0	25.0	1969
2	3 IO Cameras	TV				
3	IR Flux	I-2	3.0	86.0	3.0	1969
4	Visible Radio-meter	I-79	3.0	89.0	3.0	1969
5	Magnetometer	I-23	5.0	94.0	5.0	1969
6	Far UV Radio-meter	I-96	3.0	97.0	3.0	--
7	Micrometeroid Flux	I-55	3.0	100.0	0.5	1969
8	Charged Particle Flux		5.5	105.5	1.0	1969
9	Polarimeter	I-95	4.5	110.0	4.5	1969
10	Bistatic Radar	I-85	13.0	123.0	2.0	1969

TABLE 2.2-4. SCIENTIFIC PAYLOAD FOR LANDER OF LANDER/ ORBITER COMBINATION

Priority	Name	Inst. No.	Weight (Pounds)	Accum. Weight (Pounds)	Power (Watts)	Year Originally Planned to Fly
1.	Temperature	I-24	0.3	0.3	0.77	1969
2.	Sounds	I-34	0.5	0.8	1	1969
3.	Pressure	I-17	0.3	1.1	0.10	1969
4.	Density	I-20	1.5	2.6	2	1969
5.	Multiple Chamber	I-54	4.0	6.6	2	1969
6.	Surface Penetrability/Hardness	I-25	4.5	11.1	0.1	1969
7.	Photoautotroph Detector	I-62	3.0	14.1	1	1969
8.	Light Intensity (Sun Sensor)	I-84	.5	14.6	0.1	1969
9.	Composition, H_2O	I-44	1.5	16.1	1	--
10.	Composition, O_2	I-45	1.5	17.6	1	--
11.	Turbidity & PH Growth Detector	I-53	4.0	21.6	1	1969
12.	Wind Speed & Direction	I-67	2.0	23.6	0.5	1969
13.	Gas Chromatograph	I-8	7.0	30.6	4.5	1969
14.	Composition, N_2	I-48	1.0	31.6	1	--
15.	Composition, CO_2	I-49	1.0	32.6	1	--
16.	Soil Moisture	I-70	2.0	34.6	25	1969
17.	TV Camera, Panorama	-	20.0**	54.6	20	1969
18.	Radioisotope Growth Detector	I-19	6.0	60.6	3	1969
19.	Composition, O_3	I-46	1.5	62.1	1	--

**Inc. 10 Pounds TV Deployment

TABLE 2.2-4. SCIENTIFIC PAYLOAD FOR LANDER OF LANDER/ORBITER COMBINATION (Cont'd)

Priority	Name	Inst. No.	Weight (Pounds)	Accum. Weight (Pounds)	Power (Watts)	Year Originally Planned to Fly
20.	Composition, A	I-47	1.5	63.6	1	--
21.	Precipitation	I-36	1.0	64.6	1	1969
22.	Electron Density (Langmuir Probe)	I-39	3.0	67.6	3	1969
23.	Surface Gravity	I-72	3.0	70.6	3	1969
24.	Radar Altimeter	I-5	15.0	85.6	25	1969
25.	Seismic Activity	I-21	8.0	93.6	1	1969

2.3 SYSTEM CONSTRAINTS AND REQUIREMENTS

The performance of the Titan IIIC launch vehicle, including the standard Titan IIIC shroud is given by Figure 2.3-1. It was determined during the study that the standard Titan IIIC shroud shown in Figure 2.3-2 seriously restricted the spacecraft design and that a new shroud design would be necessary unless Landers with extensible flaps compatible with the standard 120-inch diameter shroud were utilized.

The choice of the transit trajectory for each of the spacecraft systems followed the method and analyses in the Voyager Design Study except for minor modifications to allow for All-Orbiter and All-Lander systems.

The guidance system is essentially the same as in the previous Voyager study. Approach guidance is required and obtained by viewing the planet against the star background with a TV camera and transmitting the picture to Earth for processing. With approach guidance, a 0.99 probability of meeting the required entry angle corridor of 20 degrees to 35 degrees is assured. With the elimination of a synchronized Orbiter, line-of-sight between the Earth and the Lander must be maintained during Lander entry for transmittal of entry data.

Three possible planet approach trajectories were considered as follows: 1) flyby trajectory with the Bus/Lander always on a miss trajectory with a velocity impulse applied to the Lander after separation; 2) impact trajectory with the Bus/Lander always on an impact trajectory, with a velocity impulse applied to the Bus after Lander

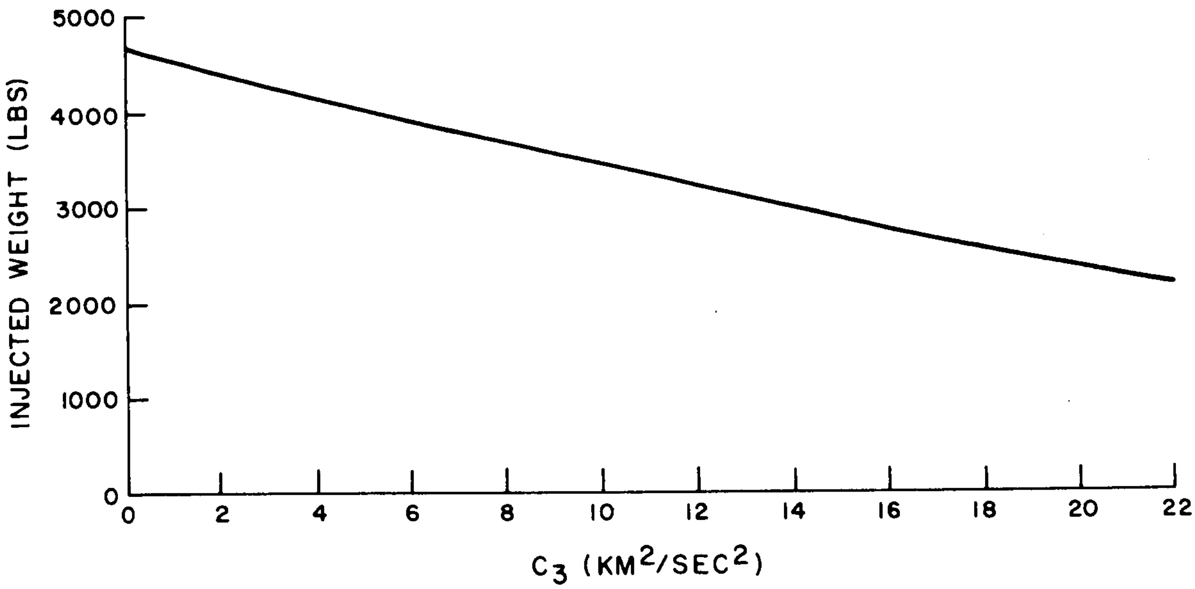

Figure 2.3-1. Titan IIIC Performance

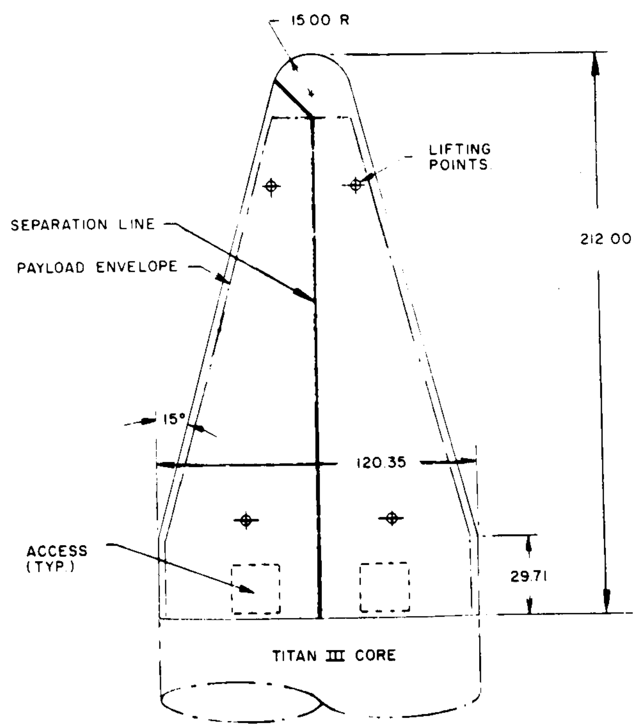

Figure 2.3-2. Standard Shroud

separation; and 3) flyby/impact trajectory with the Bus/Lander on a flyby trajectory until the approach correction maneuver and on an impact trajectory thereafter with a velocity impulse applied to the Bus after Lander separation. Since error analyses showed the capability of meeting the entry corridor and landing-site dispersion requirements with a flyby trajectory and reliability analyses showed a requirement for propulsion and communication redundancy, the flyby trajectory was selected as a basis for system design.

By the selection of the flyby trajectory, requirements for Bus usage after Lander separation were eliminated. This allows an almost fully integrated Bus/Lander with rational component allocation between Bus and Lander.

2.4 COMMUNICATION MODES AND DATA RATES

2.4.1 GENERAL

Data requirements for each mission phase of each system were established. Prime and "back-up" modes were selected in order to accommodate the respective data requirements. In the case of the Lander descent phase, there is only one mode designed to provide the most essential entry dynamics diagnostic and prime atmospheric parameter data prior to impact on the surface.

Back-up modes are intended for use when there is loss of attitude control or failure of a high-gain antenna pointing mechanism, or when maneuvering requirements preclude data transmission through the high gain antennas. Data rates are drastically reduced to accommodate only critical diagnostic and non-pictorial scientific data of highest interest and lowest bit requirements.

Back-up mode data rates are usually near marginal for the distance prevailing in the mission, since these modes use very broad beam "omnidirectional" antennas in order to have the communication links operate independently of vehicle attitude.

Prime data rates were established by balancing high data volume generating payload, such as television cameras, with reasonable power supply weights and antenna sizes that were compatible with the spacecraft system.

All communication from the spacecraft to the Earth is at S-Band frequency and requires the 210-foot DSIF antenna to obtain the data rates shown. The command communication from Earth to the spacecraft is also at S-Band frequency and requires the 85-foot DSIF antenna.

2.4.2 BUS/LANDER

When the Bus/Lander is near Earth, continuous tracking and telemetry data are transmitted at 15 watts radiated through an omni-antenna. Further out, a three-foot-diameter parabolic antenna is utilized. A 24-watt klystron, in the Lander, transmits through the three-foot antenna to provide 400 bits/second at maximum expected encounter ranges of 1.33 AU. The terminal guidance TV frame requires 45 minutes to transmit. Back-up telemetry radiates 150 watts through the omnidirectional antenna at 4 bits/second.

After the Lander is separated from the Bus, Lander diagnostic telemetry radiates 150 watts through a 150-degree beam omnidirectional antenna mounted in the center of the aft cone. This link is available after the solid fueled ΔV rocket motor used to place the Lander on its impact trajectory is jettisoned. The rate is 4 bits/second. After entry and blackout, critical entry diagnostic and atmospheric scientific information is transmitted direct to Earth through this same link during descent. Predetection recording is required for this link.

A continuous low power direct link is provided for telemetry during the surface phase of the Lander mission. The 24-watt klystron transmits through a 27-db gain helical array Earth pointing antenna at 800 bits/second. This provides 5000 Lander TV frames in the first 90 days of surface operation, and a total of 8350 frames for the complete 180 days of the surface mission.

2.4.3 ORBITER

Continuous tracking is provided during early transit by the 57-watt klystron though an omni-antenna up to a range of 0.3 AU. Farther out, the nine-foot diameter parabolic antenna is deployed. Terminal guidance information is transmitted at 6000 bits/second through this link, with the TV frame transmitted in three minutes.

Back-up telemetry provides 4 bits/second with 100 watts radiated through an omni-antenna.

The prime rate during the orbiting phase of the mission is 12,000 bits per second. This rate can accommodate six sets of four (4) Image Orthicon (3 color and 1 black and white) television frames and 76 vidicon stereo mapping frames per 4.3-hour orbit. Fifty-seven watts are radiated through the nine-foot diameter parabolic antenna to maintain this rate until the initial map is acquired, which occurs ten days after encounter for an Orbiter launched on the last day of the launch window, or at a distance of 1.405 AU.

2.4.4 ORBITER/LANDER COMBINATION

Transit telemetry is transmitted at 43 watts through an omni-antenna in near-Earth ranges and the eight-foot diameter parabolic antenna farther out and at encounter.

An additional 60 watts are provided for a back-up link, in transit, through an omni-antenna for a rate of 4 bits/second.

Terminal guidance television information is transmitted at 3000 bits/second by the 43-watt Klystron through the eight-foot diameter dish, one frame requiring six minutes at encounter range.

In the orbiting phase, the prime link transmits at 6000 bits/second and can continue until 28 days after the latest encounter, when the Earth/Mars range will have become 1.57 AU. This rate provides two Image Orthicon sets of three frames per each pair of stereo vidicon frames, for a total information rate of 6.16×10^8 bits per 27.6-hour orbit.

A VHF relay link is also provided in this orbiter. The VHF command transmitter sends five watts through a 10 db Yagi antenna mounted on the Planet Horizontal Package and thus is always pointing to the center of Mars while the Orbiter is in orbit. This relay link receives and retransmits 5000 Lander TV frames in 90 days.

Post separation and descent telemetry for the Lander are at 25 watts through an omni-antenna at the 100-mc relay link frequency. Sixteen-thousand bits/second can be transmitted to the 10 db Yagi on the Orbiter during descent when the Orbiter is available (in sight of the Lander at a range of less than 2000 n. mi.).

A direct, low power, continuous link to Earth is also provided in this Lander. Four-hundred bits per second are transmitted when the relay link is not being used, and when Earth is in view, radiating 12 watts through the Earth-pointing 27-db helical array.

A direct, descent link through an omni-antenna is not incorporated in this Lander.

2.5 ORBIT ANALYSIS AND SELECTION

Initial estimates of the All-Orbiter configuration showed that a 1000 x 1000 n. mi. circular orbit could be achieved with the original 215 pounds of orbiter payload from the prior study. However, this orbit required unreasonably high data rates from the Orbiter. Consequently, the orbit period was increased, to permit the inclusion of the larger power supply required to maintain telemetry during the entire orbit, to provide additional time for communication and to provide additional payload capability. A 4.3-hour, 1000 x 2278 n. mi. orbit was selected to provide the minimum possibility of the occurrence of a synchronous relationship between orbit and Mars rotational period. (See Figure 2.5-1.)

The orbit of the Orbiter/Lander combination system is constrained by weight limitations to a 1000 x 19,000 n. mi. orbit.

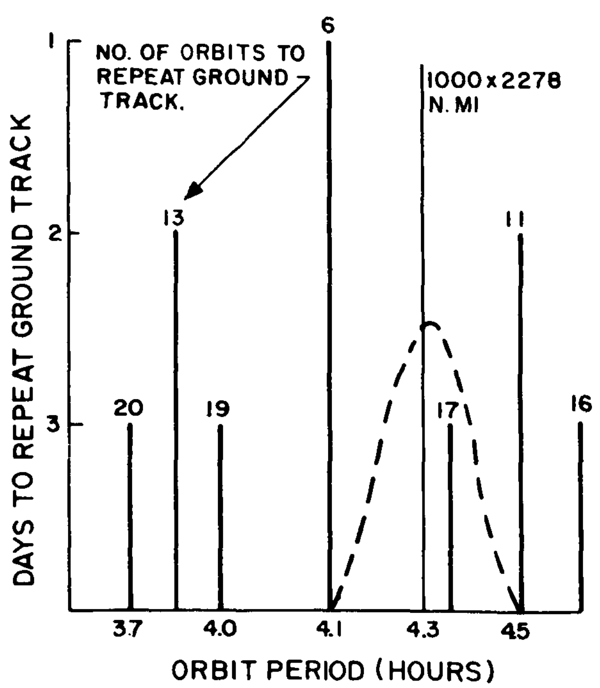

Figure 2.5-1. Orbit Selection

2.6 MISSION AND POWER PROFILES

The major events of the transit phase of the missions studied are injection, orientation, communication, midcourse correction, terminal guidance observation of target planet, approach correction, lander separation as applicable, and orbit injection.

The orbiting phase of the Titan IIIC All-Orbiter mission is altered from that of the prior study because of the change in orbit, discussed in Section 2.5, Orbit Analysis and Selection, and the elimination of the relay mode of communication. All other mission functions are the same as in the prior study.

The descent phase of the Landers is the same as for the Saturn 1B Landers except for the direct transmission of entry diagnostic and atmospheric scientific information, as discussed in Section 2.4.2. The surface phase of the Lander mission has the same objective as the Landers in the prior study, i.e., life detection, landscape television, geological and atmospheric determinations. Operation of the Lander TV system or the subsurface drill requires interruption of the telemetry or postponement of these operations until after Earth-set. This is discussed in Section 2.6.1. All mission phases of the Titan IIIC Orbiter/Lander combination system are the same as for the Saturn 1B Voyager system.

A transit phase, electrical power matrix and power profile are shown in Table 2.6-1 and Figure 2.6-1. These are typical of similar items produced for all systems discussed and for all mission phases.

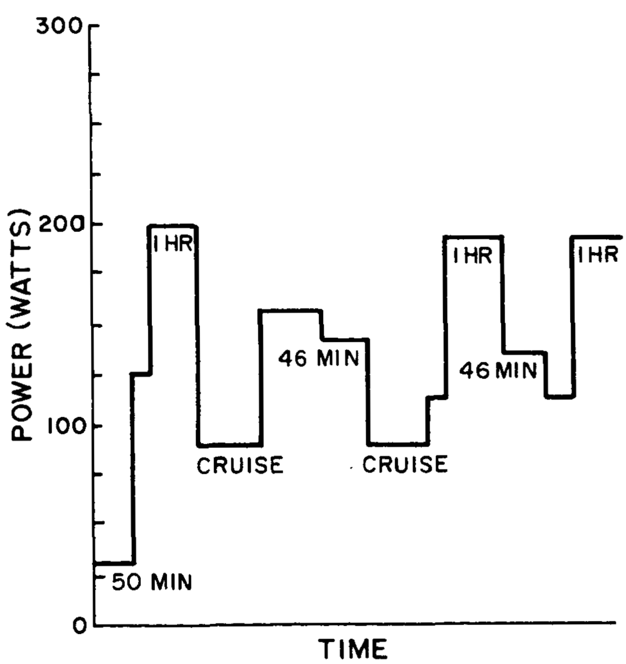

Figure 2.6-1. 2042-Pound Bus/Lander Only Transit Phase Power Profile

TABLE 2.6-1. 2042-POUND BUS/LANDER (TRANSIT PHASE) POWER MATRIX

	Launch and Injection	Initial Orientation	Initial Orientation Earth Link to Verify Canopus	Cruise	Cruise and Earth Link	Midcourse Manuevers and Reorientation	Cruise	Term. Guidance Obser.	Cruise and Earth Link Xmit Term. Guid. Info.	Final Trajectory Correction	Cruise and Term. Guidance Obser.	Cruise and Earth Link Xmit Term. Guid. Info.
I Communication												
Earth Link			75		75				120			120
Data Storage & Processing		4	4	4	4	4	4	4	4	4	4	4
Tape Recorder								5	2		5	2
Commands and Tracking		8	9	8	9	8	8	8	9	8	8	9
Power Conv. & Control		10	10	10	10	10	10	10	10	10	10	10
II Guidance and Control												
Attitude Control		65	65	28	28	65	28	28	28	65	28	28
Gyro and Electronics	16.5	17	17			17				17		
Hi Gain Antenna Point.				29 watts for 5 minutes once per day								
Body Mounted IO Camera and Electronics								25			25	
III Thermal Control												
Bus		5	5	15	15	15	15	15		15	15	
Landers	15	15	15	15	15	15	15	15		15	15	15
IV Engineering Data												
Bus	2	1	1	1	1	1	1	1	1	1	1	1
Landers		1	1	1	1	1	1	1	1	1	1	1
V Science												
Descent												
Surface												
Total Power Required	35.5	126	202	82	158	136	82	112	190	136	112	190
Duration	50 min	15 min	1 hr	Cruise	Cont.	46 min	Cruise	15 min	1 hr	46 min	15 min	1 hr

2.7 RELIABILITY AND VALUE ANALYSIS

The reliability analysis for the Titan IIIC Voyager System is summarized and compared with the corresponding values of the Saturn 1B Voyager System by Table 2.7.1.

TABLE 2.7-1. SYSTEM RELIABILITY -- SINGLE LAUNCH
(Launch Through 100 Hours After Arrival)

	Saturn 1B		Titan IIIC
Lander "Surface" Data			
Martian Terrain Suitability	90.0%		90.0%
Lander Reliability	84.7		*76.0
Lander Instrument Reliability	96.5		96.5
Orbiter Through Transit	76.8	Bus	91.5
Orbiter During 1st 100 Hours	98.6	Incl In	-
Booster	80.0	Lander	80.0
Subsystem	44.6		48.3
Lander "Entry" Data			
Lander Through Entry	88.3		79.2
Lander Instrument	99.5		99.5
Orbiter Into Orbit	76.8	Bus	91.5
Booster	80.0		80.0
Subsystem	54.0		57.7
Orbiter Data			
Orbiter Through 100 Hours	75.7		76.8
Orbiter Instrument	96.5		96.5
Booster	80.0		80.0
Subsystem	58.4		59.2
Effective Single System Reliability	49.7		52.5

e.g. $\frac{(44.6 \times 60\%\ V) + (54.0 \times 10\%\ V) + (58.4 \times 30\%\ V)}{100\%\ V}$

*Lander Through Transit = 79.2 $\therefore$ 79.2 x 96.0 = 76

The large single lander of the Titan IIIC study can be applied to the Saturn 1B - Voyager System and the increased mission value attainable by the use of this Lander system is presented in Figure 2.7-1. This figure also presents the mission value attainable by the use of this lander as a part of the Titan IIIC recommended system. Direct comparisons may be made between potential capabilities of Titan IIIC and Saturn 1B systems using Figure 2.7-1.

Since the extra payload capability of the Titan IIIC Lander and Orbiter may be applied completely to reliability improvement for systems otherwise identical, a curve providing this comparison is shown as Figure 2.7-2. This Figure also provides for reference a Titan IIIC system line in which no advantage is taken of the extra payload capability.

The effects upon the attainable mission value resulting from various reliability improvements have been studied. Wherever such an improvement affects a complete system, the resulting effects, even upon "multiple launch" attainable mission values, are directly proportional to the change in system reliability. Whenever such improvements affect only certain portions of the scientific payload or where the values considered applicable to second and subsequent missions may be altered as a function of the success or failure of corresponding portions of prior mission flights, the effects are not directly proportional and a more detailed analysis is required before valid comparisons can be drawn.

It should be noted that a final review of the alternatives considered and applied during the Titan IIIC study are directly applicable to the Saturn 1B system insofar as the effect on system reliability is concerned. Thus, the reliability differences between these systems are significant only as the greater payload capability per space vehicle available with the Titan IIIC system is applied to reliability improvement in the final design.

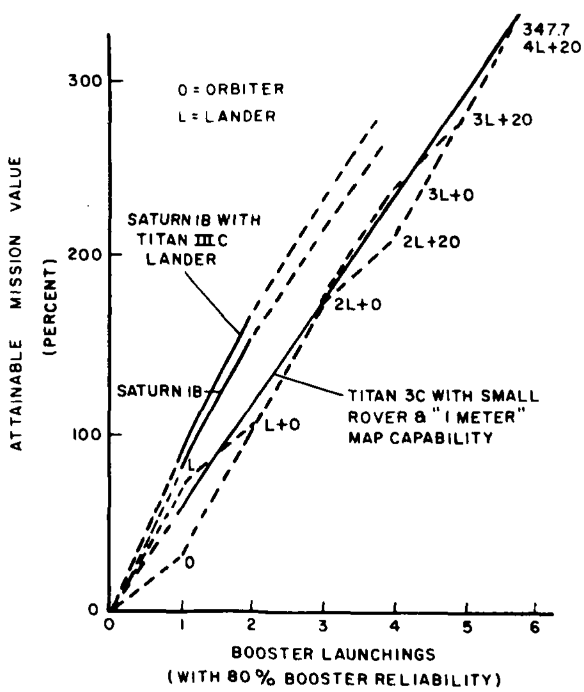

Figure 2.7-1. Attainable Mission Value

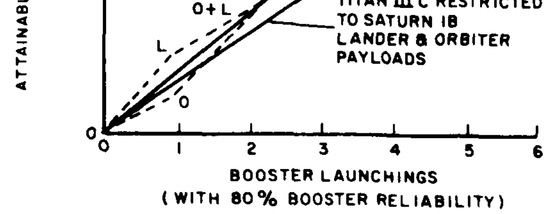

Figure 2.7-2. Attainable Mission Value (Identical Payload)

2.7 RELIABILITY AND VALUE ANALYSIS

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Booster	80.0		80.0
Subsystem	54.0		57.7
Orbiter Data			
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Effective Single System Reliability	49.7		52.5

e.g. $\frac{(44.6 \times 60\%\ V) + (54.0 \times 10\%\ V) + (58.4 \times 30\%\ V)}{100\%\ V}$

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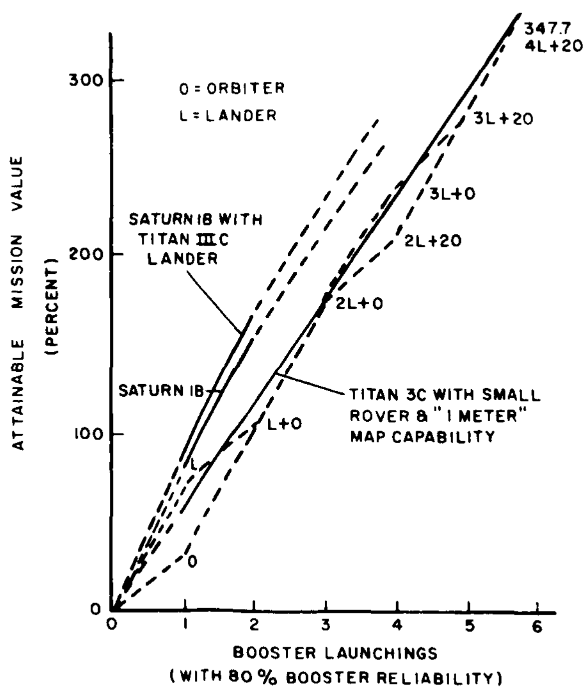

Figure 2.7-1. Attainable Mission Value

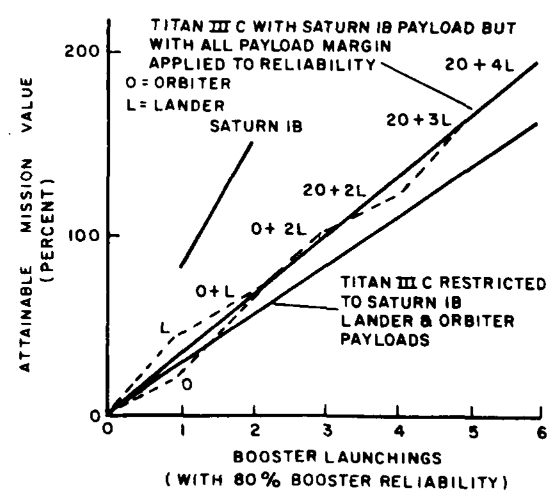

Figure 2.7-2. Attainable Mission Value (Identical Payload)

A summary of the reliability analyses of the three major spacecraft systems investigated during this study is provided below in Table 2.7-2.

TABLE 2.7-2. MARS 1971 VOYAGER SYSTEMS RELIABILITY SUMMARY

	System and Subsystem	Reliability: 100 Hrs. After Transit			Reliability: 3 Mo. After Transit		
		Bus/ Lander	Orbiter	Orbiter/ Lander	Bus/ Lander	Orbiter	Orbiter/ Lander
Bus or Orbiter	Communications	0.999	0.866	0.856	0.999	0.793	0.742
	Guidance & Control	0.920	0.912	0.912	0.920	0.831	0.831
	Power Supply	--	0.980	0.980	--	0.973	0.973
	Propulsion						
	Hot Gas	0.999	0.998	0.998	0.999	0.998	0.998
	Cold Gas	0.997	0.996	0.996	0.997	0.990	0.990
	Vehicle	0.915	0.768	0.758	0.915	0.633	0.587
Lander	Communications	0.863	--	0.989	0.817	--	0.952
	EP & D	0.970	--	0.970	0.959	--	0.959
	Prop. & Sep.	0.972	--	0.972	0.972	--	0.972
	Thermal Control	0.957	--	0.957	0.947	--	0.947
	Retardation	0.984	--	0.984	0.984	--	0.984
	Orientation	0.993	--	0.993	0.993	--	0.993
	Lander	0.760	--	0.872	0.704	--	0.822
Complete System		0.696	0.768	0.661	0.645	0.633	0.482

2.8 APPLICABILITY TO 1972 VENUS MISSION

An Orbiter designed for the 1971 Mars mission can be easily modified for the 1972 Venus mission. Guidance, attitude control, communication and propulsion subsystems are essentially the same. The solar array used for Mars is reduced in area, because of increased solar radiation and lower power requirements, by omitting array segments and by removing a portion of the body mounted cells to reduce power and thermal peaks on spacecraft components. The Mars PHP is removed and a mapping radar antenna with a small package of planet scanning instruments is substituted on the same mounting hardware. The Orbiter is still Sun and Canopus oriented in transit and in orbit. Data rates from the Mars 71 communication system are quite suitable for the mapping radar at Earth/Venus distances in the 1972 type II trajectory. The orbit is 1000 x 13,000 nautical miles inclined 67 degrees to the equator.

A Bus/Lander Titan IIIC mission can be flown to Venus in 1972 by modifying the bus with the addition of a solar cell array for electrical power during transit. The Lander subsystem would handle data and communication during transit just as in the Mars mission. The Lander would enter Venus atmosphere at 85/90 degrees at the sub-earth point on Venus surface so that the 20-degree beamwidth of the fixed antenna on a vertically oriented descending entry lander would cover Earth. Lander data for the 10-hour mission would be 30×10^6 bits. The Lander would necessarily be designed and developed for this mission and would have no relationship to the Mars Landers. Power supply during the surface mission would be primary batteries.

2.9 APPLICABILITY TO MARS 1969

Simplified versions of the Voyager Orbiter and Bus/Lander systems presented in this report can be considered very seriously for the Mars 1969 Mariner mission. This mission would not have the same scientific payload sophistication, would have reduced requirements for most of the subsystems and would be designed to accommodate wider overall system uncertainties. However, by properly anticipating the 1971 mission requirements, a great deal of the development for the Mariner equipment could be applied to the 1971 Voyager.

Table 2.9-1 shows the possibility of utilizing the same Orbiter design for both 1969 and 1971. Payload and subsystem simplifications could be effected without altering the basic similarity of the two systems.

The Lander could have many variations in size and payload. It is assumed that the Mariner 1969 mission would have a more conservative design which would permit unrestricted entry corridors into the 11 mb atmosphere. Wide entry corridors would reduce the required guidance accuracy and dependence upon sophisticated terminal guidance and approach correction techniques. Payload of such Landers would be minimal with the emphasis being upon the determination of atmospheric characteristics and of basic life detection experiments. Table 2.9-2 shows two possibilities of using variations of the basic Lander design for both 1969 and 1971. The first system uses the 1971 Lander design with the gross payload reduced to 367 pounds. This reduces the W/C_DA sufficiently to increase the acceptable entry corridor to a range of 20-60 degrees in the 11 mb atmosphere. The second system uses the 1971 Lander design with the gross payload reduced to 222 pounds and modification to the retardation and structural subsystems sufficient to reduce the entry weight to 1094 pounds. This would permit unrestricted entry into the 11 mb atmosphere.

The use of Landers with extensible flaps in 1971 would permit a wide variety of compatible 1969-1971 Landers to be designed.

Work on 1969 Mariner systems has been accomplished only to the extent of identifying the possibility of an orderly evolution of the Mariner 1969 into the heavier and more sophisticated Voyager 1971 design. Additional effort will be required to detail the systems.

TABLE 2.9-1. ADAPTABILITY TO MARS 1969

(ORBITER)

	1969	1971
Orbiting Weight	1701	1815
Payload	233	347
Fuel	1578	1634
Injected Weight	3279	3449
Adapter & Shroud	151	151
	3430 pounds	3600 pounds
Orbit (n. mi.)	1000 x 19,000	1000 x 2278

TABLE 2.9-2. ADAPTABILITY TO MARS 1969

(BUS/LANDER)

	144-Inch Shroud			120-Inch Shroud (Flaps)	
	1969		1971	1969	1971
Total Lander Weight, lb	1360	1094	2042	1455	2042
Entry Weight, lb	1170	975	1830	1370	1830
Gross Payload, lb	367	222	857	364	782
Ballistic Coefficient (W/C_DA), lb/ft^2	9.6	8	15	11.2	15
Entry Corridor (11 mb) (γ_e in degrees)	20-60	20-90	20-35	20-50	20-35

3. SPACECRAFT SYSTEM DESIGN STUDIES

3.1 GENERAL

Three separate and distinct types of spacecraft are considered:

1. An integrated Bus/Lander
2. An All-Orbiter
3. An Orbiter/Lander

The Bus in the Bus/Lander system is suitable for various sizes of Landers. The designs shown are for Mars 1971 with variations for other years and missions indicated. The Bus/Lander system is integrated, with maximum use being made of the Lander subsystems in the Bus itself.

The All-Orbiter system delineates missions and payloads required of an Orbiter. A design for the Mars 1971 Orbiter is shown along with all required subsystems. The Mars 1971 Orbiter is designed for TV mapping of the planet plus other scientific payload. The Orbiter design can be easily modified for other years and missions, so that variations in payload may be accommodated. The prime power is solar, and the spacecraft is Sun oriented. A Planet Horizontal Package (PHP) contains all of the experiments and instrumentation requiring direct viewing of the planet.

An Orbiter/Lander system was designed which permits an Orbiter payload of 123 pounds and a Lander payload of 110 pounds in 1971. Increased energy requirements for later years, however, severely curtails these payloads, and this system was not investigated in depth.

3.2 SPACECRAFT SYSTEM

3.2.1 SYSTEM CONFIGURATION STUDY AND ANALYSIS

The Bus/Lander system has been selected as the prime approach for a mission involving a landing on the planet. The prime system consists of an integrated Bus/ Lander which is suitable for launch in any window from 1971 through 1977. The maximum payload that could be launched in 1971 is not utilized on this system because of decreased capability in 1973. However, the Lander presented carries all of the presently identified payload with adequate margins.

The Lander vehicle design meets the requirements and ground rules noted in Section 3.2.2, Volume II with the maximum of reliability and payload. Table 3.2-1 identifies the prime vehicle subsystems and the reasons for selection.

The bus of the integrated Bus/Lander uses the maximum of Lander equipment during transit to Mars. All power supply and communication equipment, except the transit antennas, are located in the Lander. The Bus consists of guidance and control systems, mid-course propulsion, the antennas for use in transit and the necessary structure to support these components and attach the Lander to the Launch vehicle. After the Lander is separated, the Bus does not have electrical power or communication capability and becomes inoperative.

Estimated weights for the Bus/Lander for a Mars 1971 trip are shown in Table 3.2-2.

TABLE 3.2-1. LANDER SUBSYSTEM SUMMARY

Subsystem	Selected Approach	Reason for Selection	Past Work
Configuration	Sphere Cone $\theta_c = 51.5°$ $R_n/R_b = .47$ Base Dia. = 134 in.	High Drag Passive Dynamic Stability Ground & Flight Test Data	Mark II Vehicle Voyager Study Mariner B Study
Structure	Aluminum Honeycomb	Minimum Weight Reasonable Cost	Voyager Study Mariner B Study Mark VI Vehicle
Heat Shield	Elastomeric Shield Material (ESM)	Compatibility with Space Environment and Sterilization High Heat of Ablation Good Insulation Properties	Voyager Study Mariner B Study Mariner 66 Proposal
Retardation System	3-Stage Chute Terminal Retro Fiberglass Impact Attenuation Material	Minimum Weight Reliability	Voyager Study Discoverer Bios
Thermal Control	Liquid Loop Heat Exchanger Passive RTG Cooling on Surface	Minimum Weight and Maximum Reliability	Voyager Bios
Ground Orientation	Clamshell Opening Harpoon & Support Legs	Minimum Weight and Complexity Maximum Reliability	Mariner B Study Voyager

The selected retardation system uses one supersonic decelerator parachute, one main terminal parachute and terminal braking rockets. The crushup material is designed to absorb the residual energy rising from the uncertainties in the Mars atmosphere. Figure 3.2-1 shows the sequence of events of the Lander after separation from the Bus.

The mode of communications to Earth is a direct link S-band system which operates through an encapsulated turnstile atenna during planetary approach and entry. Surface communication is maintained by a steerable, helix array antenna. Backup is provided by an omnidirectional antenna. The Entry/Lander in the surface deployed configuration is shown in Figure 3.2-2.

Parametric analyses were performed in the areas of structural and impact attenuation material, weight of parachutes and terminal retrorockets, heat shield and thermal control systems. Prime attention has been given to the retardation system wherein four combinations of parachutes, retrorockets, sensors and impact attenuation were considered. Alternate analyses were conducted in the areas of:

1) Effect of variation of lateral wind
2) Design for a 90-degrees entry
3) Payload penalty in designing to a range of atmosphere
4) Effect of firm definition of the Martian atmosphere during a hardware program
5) Extensible Flare/Lander designed to permit packaging within a 120-inch diameter shroud.

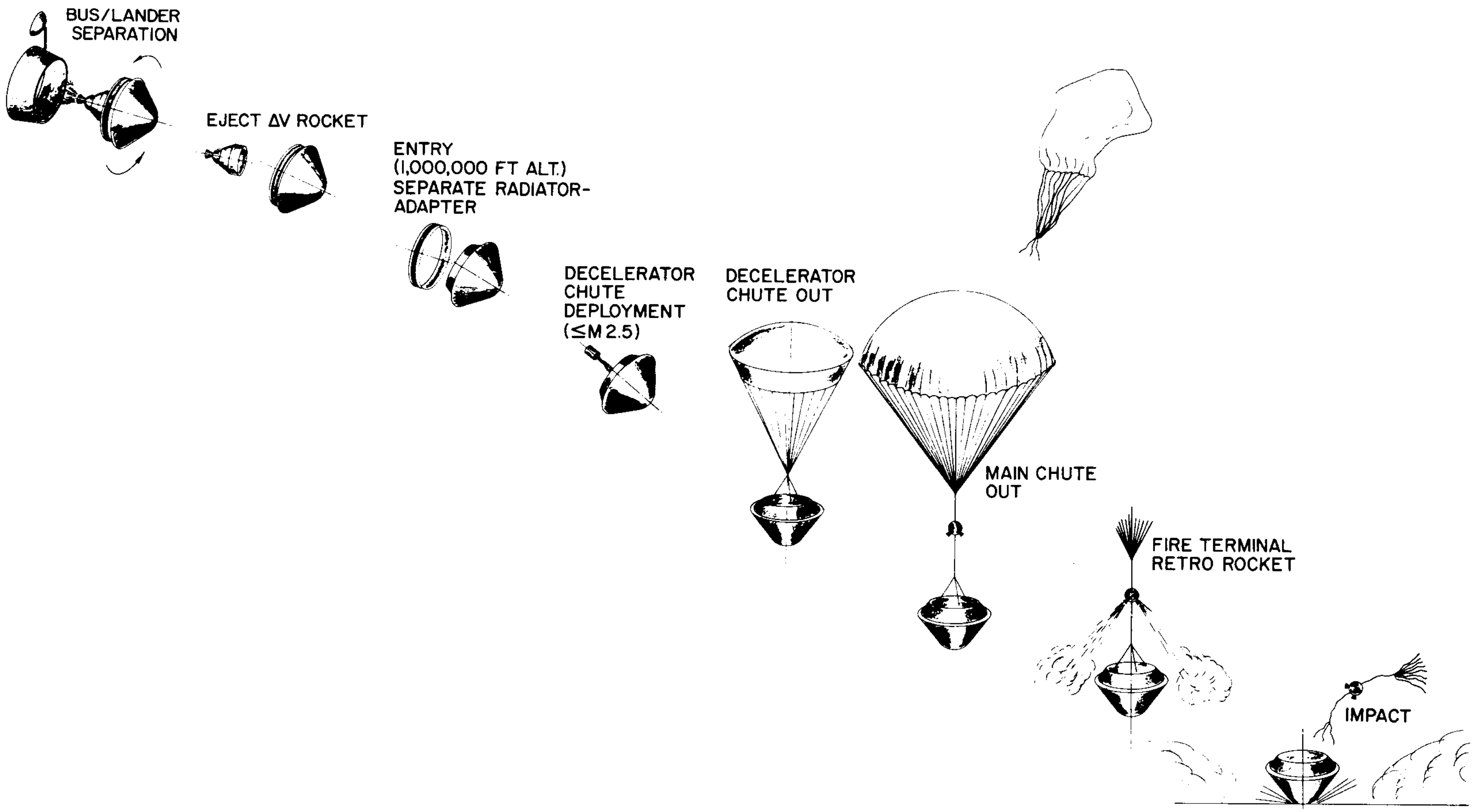

Figure 3.2-1. Entry/Lander Sequence Diagram

TABLE 3.2-2. MARS 1971 BUS/LANDER WEIGHTS

Guidance and Control	154	
Power	2	
Communications	22	
Diagnostic Instrumentation	12	
Propulsion	39	
Thermal Control	16	
Harnessing	21	
Structure	189	
Total Bus		455
Lander		2,042
Propellant		49
Δ Shroud Weight		86
Launch Weight		2,632

3.2.2 LANDER CONFIGURATION DESIGN

The Lander system selected for prime investigation in this study has a sphere-cone configuration with a half-cone angle (θ_c) of 51.5 degrees, a base diameter of 134 inches, and a bluntness ratio of 0.47. Primary structure is aluminum honeycomb sandwich. Fiberglass honeycomb shock attenuation material, used to limit impact loads, is bonded to the primary structure. The selected heat shield material is ESM-elastomeric shield material-an ablative material developed specifically for a low heat flux type of entry environment.

Components are mounted both on an aluminum honeycomb cruciform structure and on the aft cover. Upon landing, the aft cover is opened exposing the S-band antenna, TV camera, and deployable scientific instruments. Primary power is obtained from a Radioisotope Thermoelectric Generator (RTG) supplemented with a rechargeable nickel-cadmium battery.

An active thermal control system serves to cool the RTG power source and to maintain payload operating temperatures. On the Mars surface, the RTG is passively cooled by thermal radiation.

Two alternate designs were also prepared for use on the Bus/Lander system, the Extensible Flare/Lander and the Limited Rover Lander, and are presented with differences from the prime vehicle noted. The Extensible Flare/Lander has a folding flare section consisting of four flap type surfaces with support structure and linkage, all of which are contained within a 110-inch diameter for launch on a booster with a 120-inch diameter shroud. Immediately after the shroud is jettisoned on leaving the Earth's atmosphere, the flaps are extended and locked in place to become fixed structure for the remainder of the mission. The extensible flare section is jettisoned when the decelerator chute is deployed after entry to reduce chute and impact loads and to eliminate chute fouling problems. The Limited Rover Lander design was prepared as a conceptual approach to using a small wheeled vehicle to obtain additional mission value. Adequate payload capability is available to provide the mobile vehicle which carries the surface sampling instruments over an area limited by a trailing cable attached to the main vehicle for power supply and communication.

Summary weight statements of the prime system, the Extensible Flare Configuration and the Limited Rover Lander are presented in Table 3.2-3.

3.2.3 BUS CONFIGURATION DESIGN

System trade-offs indicated the desirability of an integrated Bus/Lander for the Voyager mission. The decision to use an integrated system made possible the elimination of various subsystems from the Bus. Listed below are the Bus subsystems.

1. Structure - required.
2. Communications - high-gain antenna only required; all other communications in Lander.
3. Power - nothing required; all power is from the Lander RTG.
4. Guidance and Control - required.
5. Propulsion - required.

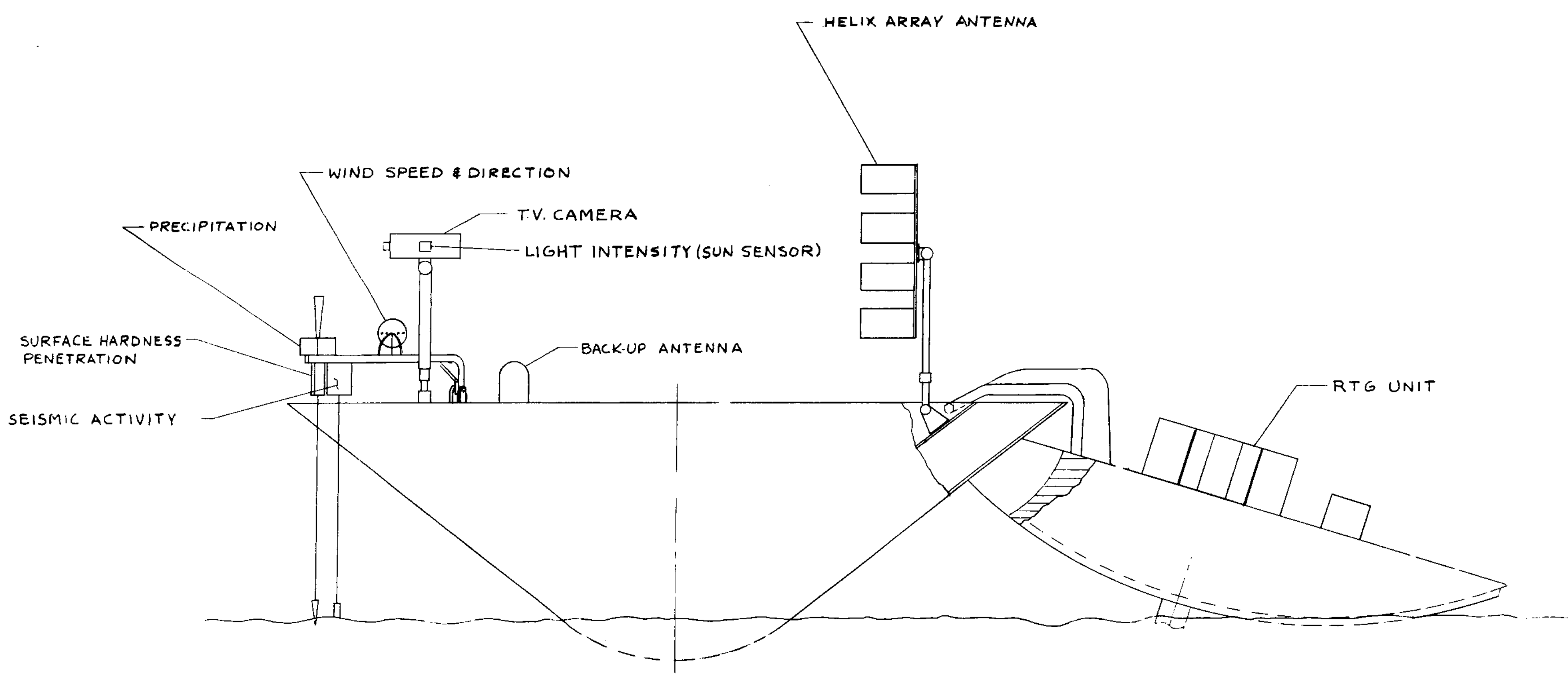

Figure 3.2-2. Entry/Lander Surface Deployed Position

TABLE 3.2-3. LANDER SUMMARY WEIGHT STATEMENT

Item / Lander Configuration	Prime D_B = 134 Inches		Ext. Flare D_B = 138 Inches		Limited Rover	
Structure		(367)		(210)		(382)
Heat Shield		(166)		(111)		(166)
Retardation		(414)		(360)		(414)
Chutes	93		77		93	
Retro	41		35		41	
Impact Att.	244		212		244	
Hardware & Housing	36		36		36	
Ground Orientation		(26)		(20)		(26)
Gross Payload		(857)		(782)		(842)
Experiments	231		231		146	
Communications	198		198		198	
Elect. System	143		143		143	
Thermal Control	129		129		129	
Rover & Experiments	--		--		138	
Unspecified	156		81		88	
Extensible Flare Including Radiator, Spin & Separation		--		(459)		--
Total Entry Weight		1830		1942		1830
Adapter		30		--		30
Radiator		31		--		31
Δ V Rocket		98		100		98
Spin & Separation		53		--		53
Entry/Lander Total		2042		2042		2042

The Bus is designed to support the Lander during boost and transit. The structure is composed of eight longerons, of which four carry Lander loads, and several sandwich panels for shear stability. The structure serves a dual purpose in that it provides environmental control for the Bus subsystems and also acts as an adapter between the Lander and the launch vehicle.

After separation from the launch vehicle a three-foot diameter high-gain antenna and two omni-antennas are used for earth communication.

Approach Guidance is provided in the Bus for the precise terminal trajectory determination necessary to eject the Lander into the narrow entry corridor. Attitude control is provided by means of Freon 14 gas with the jets mounted on booms which extend out beyond the base diameter of the Lander, thus eliminating probability of the gas impinging on the Lander. The jets may be operated even if the attitude control booms do not deploy.

Mid-course corrections are provided by a 50-pound thrust mono-propellant propulsion system. The nozzle is mounted on the outer surface of the lower ring of the structure. Maximum CG shift is in the order of 7 minutes of arc and engine capability for thrust vector misalignment is approximately ± 6 degrees.

Figure 3.2-3 shows the Bus/Lander configuration immediately after ejection of the Lander from the Bus and Figure 3.2-4 shows a more detailed arrangement of this Bus/Lander system.

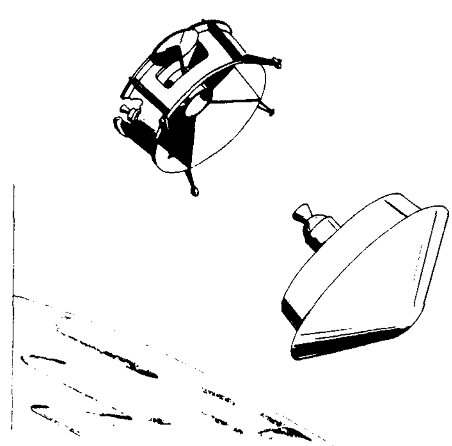

Figure 3.2-3. Lander Immediately After Separation from Bus

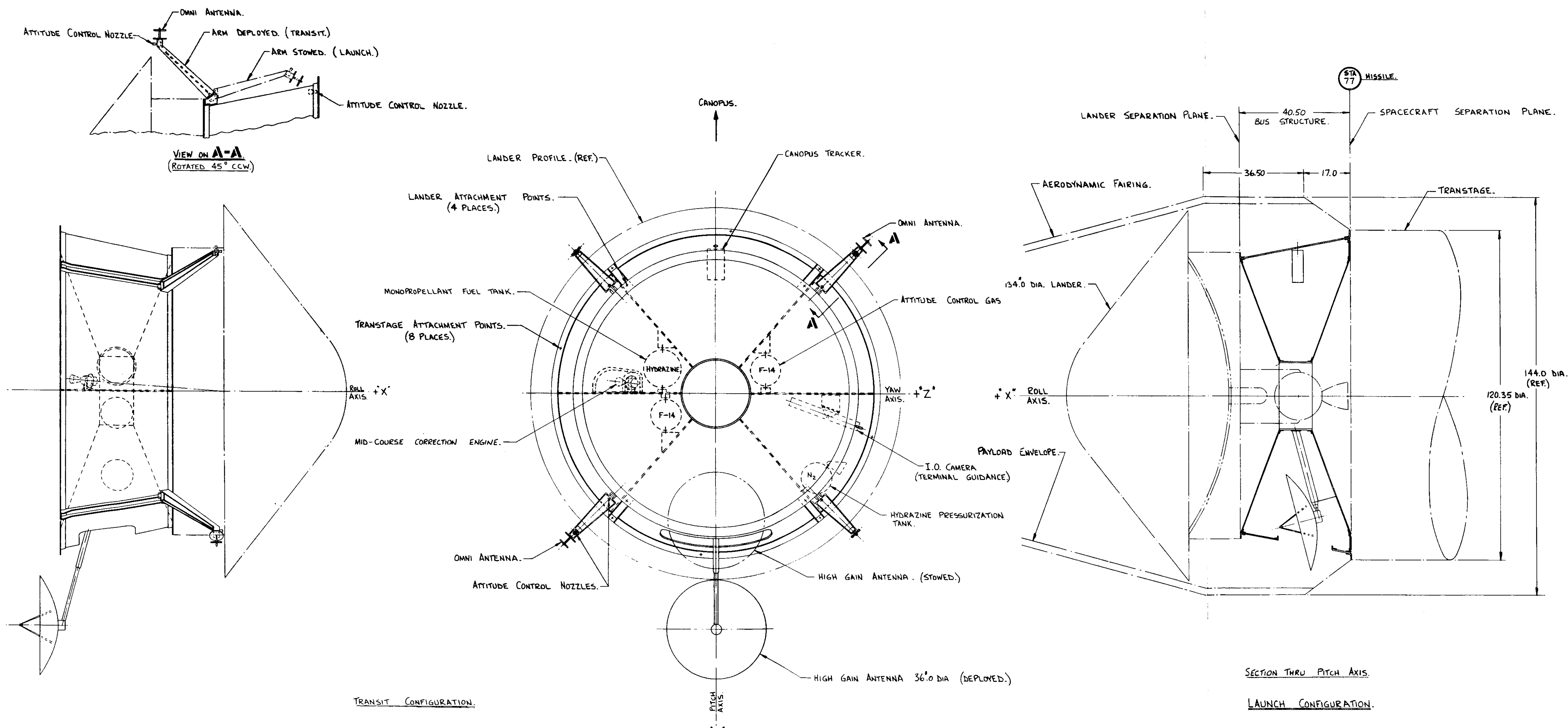

Figure 3.2-4. Bus/Lander Combination with 134-Inch Base Diameter Lander

3.3 ORBITER SPACECRAFT SYSTEM

3.3.1 SYSTEM CONFIGURATION STUDY AND ANALYSIS

The launch vehicle constraints on the Titan IIIC Voyager are:

1. Maximum launch weight = 3600 pounds for Mars 1971
2. Maximum spacecraft diameter of 110 inches
3. Booster attachments
4. Launch environment

The mission constraints are:

1. TV Mapping
2. Posigrade orbit at 1000 x 2278 n. mi.
3. 592 watts of power required (at load)
4. High data rate requiring the maximum diameter high-gain antenna.

The subsystem constraints are:

1. Sun-oriented spacecraft during transit and orbit
2. Easy component accessibility
3. Specified temperature limits
4. TV terminal guidance
5. Main engine to be used for both mid-course and orbit insertion

The Orbiter is attached to an adapter which transmits boost loads to appropriate locations on the Launch Vehicle. The separation plane is the interface between the adapter and the spacecraft.

Figure 3.3-1 shows the standard shroud for the Titan IIIC and the extension required for the Orbiter.

The Orbiter is designed to be adaptable for a variety of planetary missions. The configuration as shown in Figure 3.3-2 is modified to match the attachment points on the Titan IIIC. The structure is semi-monocoque with longerons to carry the point loads and sandwich panels to provide shear capability. Two main beams are provided to support the propellant tanks plus other bulkheads to provide support for various subsystem components. Figure 3.3-3 shows the transit configuration of the Orbiter with deployed equipment.

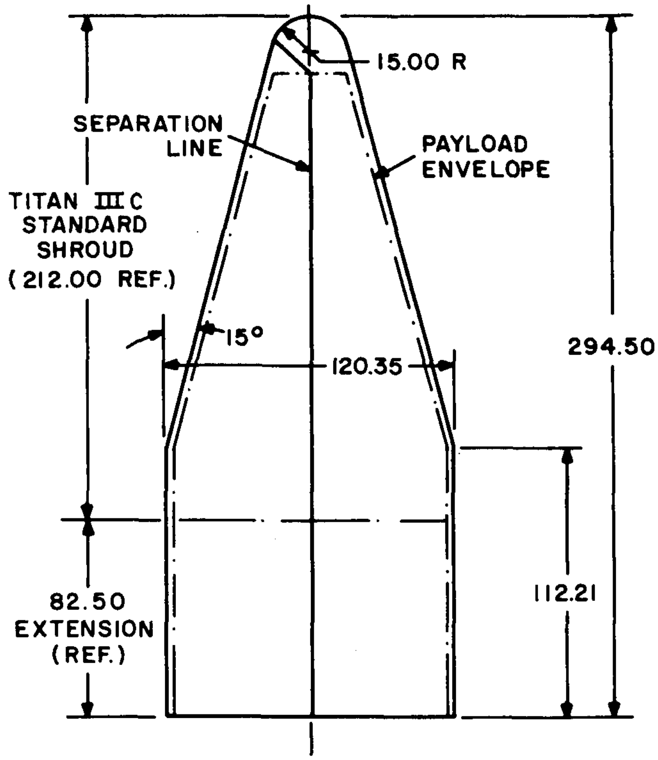

Figure 3.3-1. All-Orbiter Shroud

3.3.2 ORBITER CONFIGURATION DESIGN

Mounted on the Orbiter external surface is the Planet Horizontal Package (PHP), the nine-foot diameter high gain antenna, the magnetometer and magnetometer boom, the radio propagation experiment antenna, the DSIF omni-antennas, the main engine, and the solar cells.

The PHP contains all instruments requiring direct viewing of the planet. During launch, transit, and orbit insertion, it is stowed on top of Orbiter on the sunny side. After orbit insertion, the PHP is deployed and pointed at the planet. Three axes of control are provided in order to compensate for the angular changes required by the movement of the orbit plane about the polar axis (about 1.72 degrees/day).

The 592 watts of required power are provided by a combination of body mounted cells and deployable panel cells. During transit only 511 watts of power are available since the PHP shields a portion of the body mounted cells and one solar panel is not deployed until after orbit insertion. However, this is sufficient for the power requirements during transit.

The high gain antenna is mounted above the PHP during launch. Immediately after separation from the booster, the antenna is deployed and used to verify orientation of

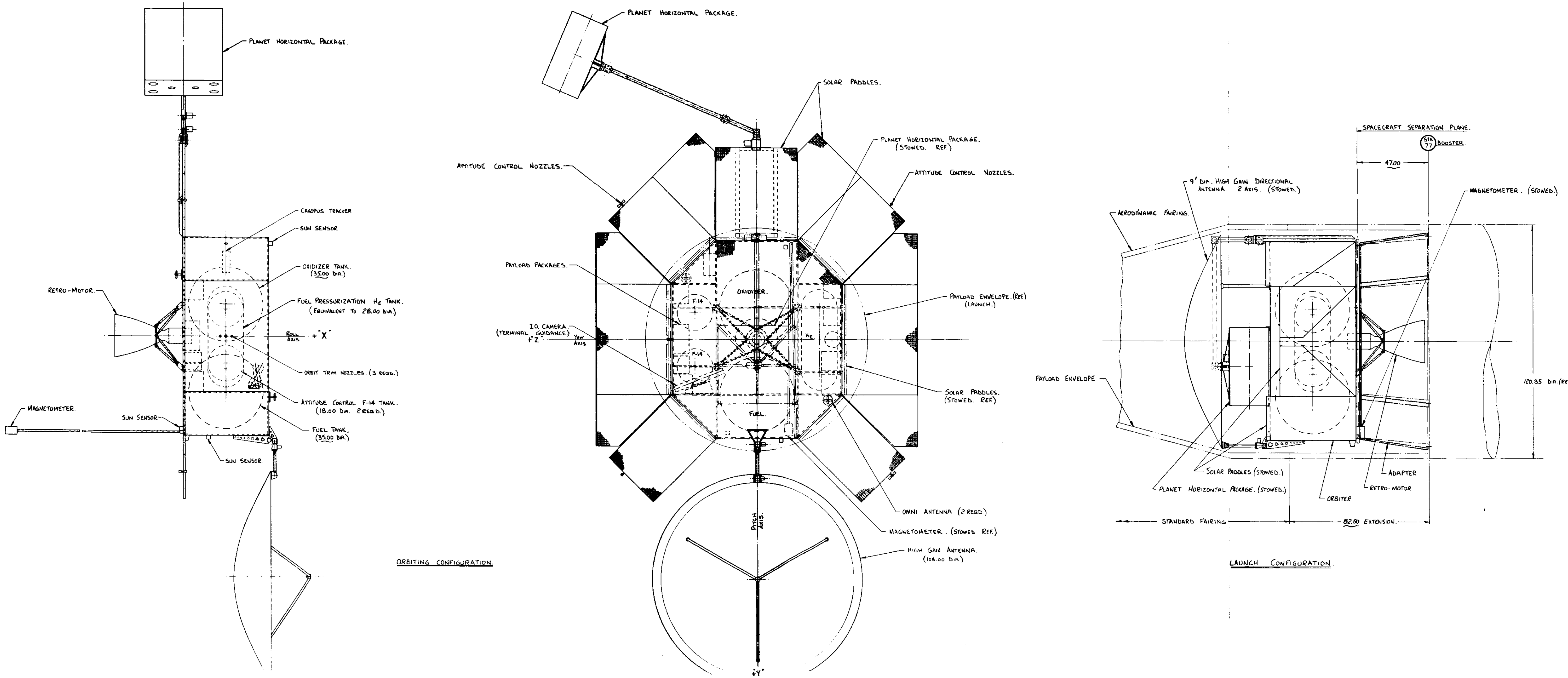

Figure 3.3-2. Orbiter-Solar Powered

the spacecraft. After verification, the antenna is stored in a transit configuration for about 120 days at which time the antenna will be deployed and become the standard means of Earth-spacecraft communication.

Mounted on the sun side of the Orbiter is a three-axis magnetometer and a 16-foot magnetomer boom. Provisions are made such that the boom may be deployed and erected in orbit. Capabilities are built into the system to rotate the boom 180 degrees per day. Time of rotation will be approximately 0.1 second. Mounted on the magnetometer boom is a 3-foot x 10-foot dipole antenna which 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.

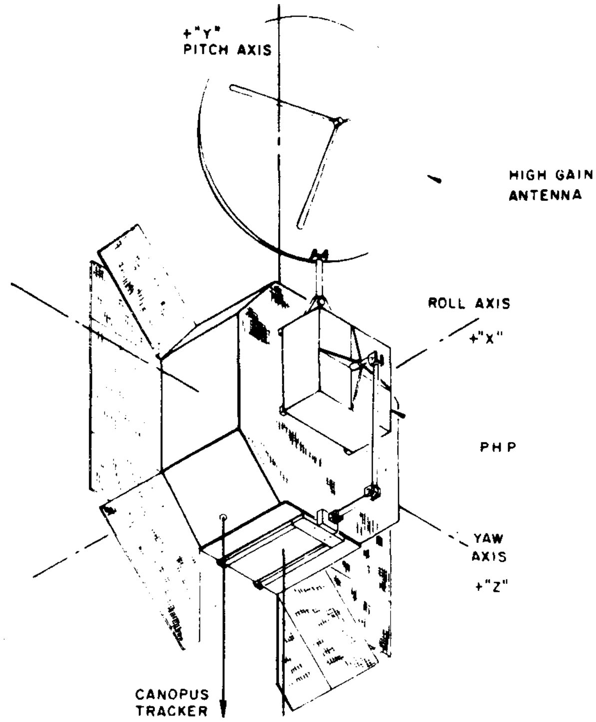

Figure 3.3-3. Transit Configuration of Orbiter with Deployed Equipment

Omni-antennas are located on both the sun side and the shaded side of the Orbiter such that communication may be maintained between Earth and the spacecraft regardless of orientation.

The main engine, which is used for mid-course corrections and for orbit insertion, is located on the base of the Orbiter. The engine is gimballed and provisions are made for thrust vector control by means of hydraulic actuators. Two degrees of freedom (± 6 degrees) of movement are provided. It is expected that the maximum static CG shift requiring thrust vector control will be in the order of ±0.1 inch, which is equivalent to ±10 minutes of arc.

The main electronic packages are mounted internally on the base of the Orbiter. With the orientation of the spacecraft to the sun, the base of the orbiter views free space thus providing for efficient thermal control. Ready accessibility is provided by means of quick release structural panels. Both active and passive thermal control are provided in order to maintain a transit temperature of 0°F to 100°F and an orbiting temperature of 30°F to 100°F.

A separate image orthicon camera for terminal guidance observation is also mounted within the Orbiter. This camera will take pictures of the target planet and star background and, final trajectory corrections will be computed from this information. The camera can be positioned "on the pad" to required coordinates at any time during the prelaunch period.

Sensors required on the spacecraft are grouped as follows: 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 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 Mars 1971 window to be successful, it is desirable to put the second Orbiter in an orbit such that the Northern Hemisphere of Mars can be mapped. By switching to the secondary Canopus tracker, the second Orbiter will be oriented in the correct attitude to map the Northern Hemisphere.

A. SEPARATION AND ACTUATION

Pyrotechnic devices will be used to separate the attachments on the high gain antenna, the Planet Horizontal Package and the magnetometer boom.

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

B. STRUCTURE

The Orbiter structure is of semi-monocoque construction, with loads introduced along sheet stiffened longerons. The choice of semi-monocoque construction was dictated by expected vibration environment. This type of construction also enhances thermal control and affords greater flexibility for packaging efficiency.

C. THERMAL CONTROL

The Voyager temperature control system utilizes a combined active and passive design concept for the PHP and Orbiter payload and an entirely passive one for the propellant tanks and the solar cells. The active control consists of thermally actuated louvers employed to vary the effective emittance of electronic component support panels and maintain adequate temperature limits under varying load rejection levels. The passive control is composed of: 1) optical coatings to be applied to particular internal and external surfaces, 2) multiple reflective radiation shields to minimize heat gains and losses, and 3) 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 will have a portion of its periphery insulated. The lower temperature of the PHP surfaces during transit has been established at 0°F, with a temperature range in orbit between 30° and 100°F 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.

D. SPACECRAFT WEIGHTS

The subsystem and system weights for the Mars 1971 mission are shown in Table 3.3-1.

TABLE 3.3-1 SYSTEM AND SUBSYSTEM SPACECRAFT WEIGHTS FOR MARS 1971

Structure	257
Harnessing	106
Power Supply	246
Guidance and Control	212
Communications	227
Diagnostic Instrumentation	30
Thermal Control	49
Propulsion	341
Payload	347

TABLE 3.3-1. SYSTEM AND SUBSYSTEM SPACECRAFT WEIGHTS FOR MARS 1971 (Cont'd)

Orbiting Weight	1815
Propellant	1598
Mid-Course Correction Propellant	36
Adapter and Δ Shroud Weight	151
Total Injected Equivalent Weight	3600 pounds

3.4 ORBITER/LANDER SPACECRAFT SYSTEM

3.4.1 SYSTEM CONFIGURATION STUDY AND ANALYSIS

The Orbiter/Lander system is planned to provide TV mapping of Mars plus entry data during Lander descent and biological data from the surface. A relay system of communications is provided with the Orbiter for transmitting Lander data to Earth.

The orbit is to be 1,000 X 19,000 n.mi. Most of the systems in the Lander are the same as or scaled down from those used on the Lander of the Bus/Lander System. Significant differences are discussed in the following section.

System weights for the Orbiter/Lander are shown in Table 3.4-1.

TABLE 3.4-1. ORBITER/LANDER SYSTEM WEIGHTS

Guidance and Control	210
Power Supply	164
Communications	254
Diagnostic Instrumentation	30
Payload	123
Propulsion	233
Thermal Control	55
Harnessing	50
Structure	321
Total Orbiter	1,440
Orbit Insertion & Mid-course Propellant	720
Lander	1,284
Adapter and Δ Shroud	156
Total Launch Weight	3,600 pounds

It is evident that 1971 is a good year since an Orbiter and a 1284-pound Lander can be launched. However, later years do not maintain this weight capacity. Table 3.4-2 shows system changes in 1973, 1975 and 1977.

TABLE 3.4-2. ORBITER/LANDER COMPARISON

	1971	1973	1975	1977
Total Orbiter	1,440	1,440	1,440	1,440
Orbit Insertion Fuel	684	596	900	550
Lander	1,284	622	568	1,018
Mid-Course Fuel	36	36	36	36
Adapter and Shroud	156	156	156	156
Total Launch Weight, lb	3,600	2,850	3,100	3,200

3.4.2 LANDER CONFIGURATION DESIGN

The Entry/Lander system which has been designed for use with an Orbiter is identical aeromechanically with the Entry/Lander used with the Bus/Lander configuration. Since the ballistic parameter is the same (15 psf), trajectory characteristics will be the same. The Lander, however, has a base diameter of 106 inches and an entry weight of 1137 pounds.

There are significant differences in the vehicle subsystems. This Lander is equipped with both a relay and direct link. The relay link is a VHF, 100-mc system using a transmission line antenna during descent and a five-foot turnstile antenna for surface operations.

Power requirements are reduced so a smaller RTG is utilized. The delta velocity rocket, spin and separation system, retardation system, and thermal control as well as heat shield and structure are smaller versions of those used on the Lander of the Bus/Lander System.

A summary weight statement of the Lander of the Orbiter/Lander System is presented in Table 3.4-3.

TABLE 3.4-3. SUMMARY WEIGHT STATEMENT ORBITER/LANDER-ENTRY LANDER

Item		Weight (Pounds)
Structure		(197)
Heat Shield		(103)
Retardation		(312)
Chutes	59	
Retro	27	
Impact	190	
Hardware and Housing	36	
Ground Orientation		(20)
Gross Payload		(505)
Experiments	110	
Communications	184	
Electrical System	101	
Thermal Control	110	
Total Entry Weight		1137
Adapter		27
Radiator		22
ΔV Rocket		66
Spin and Separation		32
Entry Lander Total		1284

3.4.3 ORBITER CONFIGURATION DESIGN

A. CONFIGURATION STUDY AND SOLUTION

The Orbiter/Lander System is designed to be packaged within an extended standard shroud with a maximum spacecraft dimension of 110 inches in any direction except the roll axis (launch vehicle thrust axis). Because of this constraint, the spacecraft in the launch condition has the high-gain antenna packaged below the main engine and the PHP packaged within the Lander to Orbiter adapters. Figure 3.4-1 shows in more detail the packaging arrangement.

Power is provided by means of solar panels hinged at the Orbiter base and attached at the upper end during launch. In addition, solar cells are attached to the fixed structure which forms a bulkhead at the upper surface of the Orbiter. These cells will not generate power until after orbit insertion when the Lander is ejected and the PHP has been deployed.

Attitude control jets are provided on the outboard ends of four of the solar panels. Freon 14 is used as the attitude control gas and flexible joints are provided in the lines at the base of solar panels.

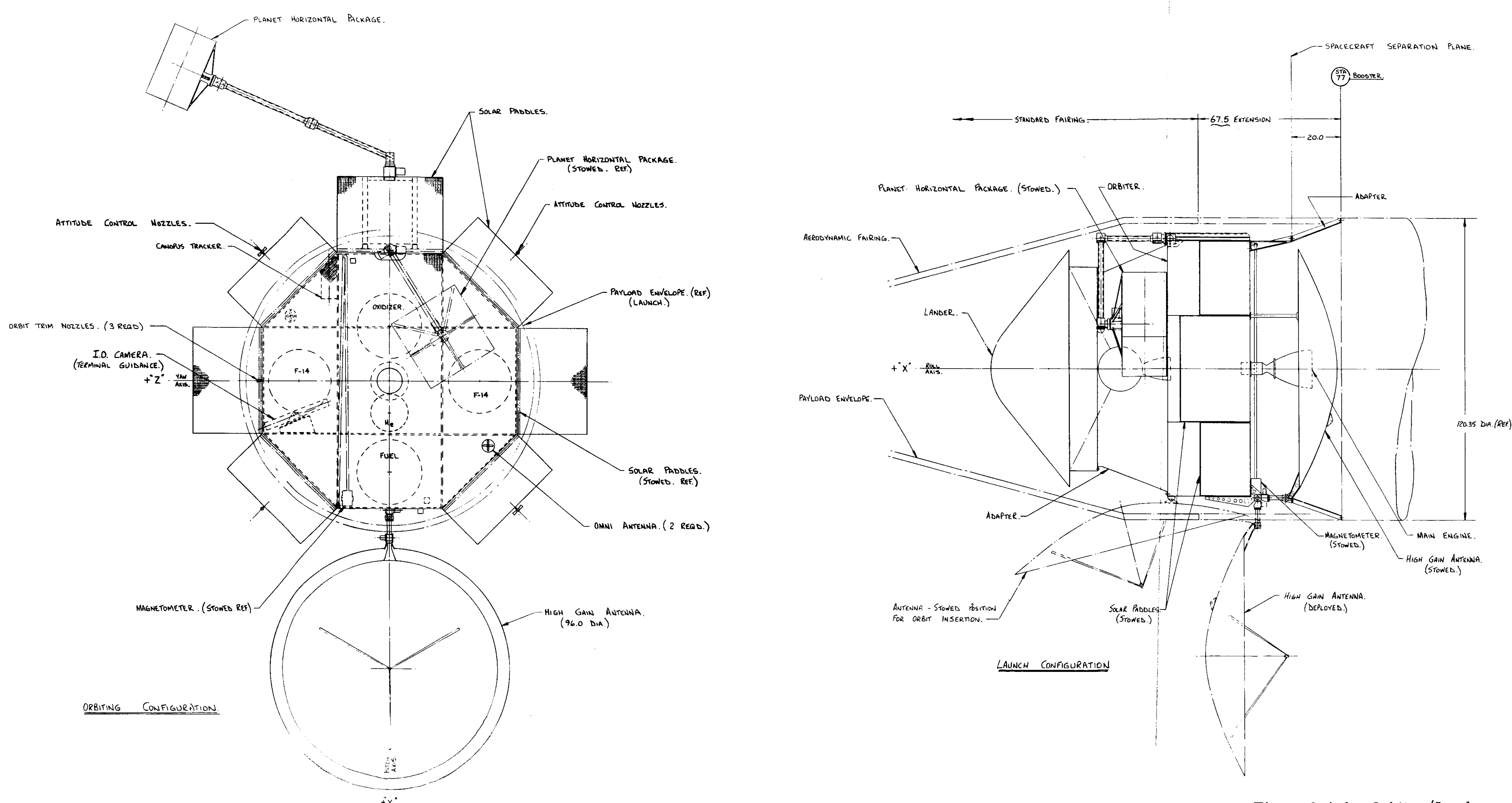

Figure 3.4-1. Orbiter/Lander Configuration

4. SUBSYSTEM DESIGN STUDIES

4.1 COMMUNICATIONS

4.1.1 GENERAL

Communication subsystems are analyzed and defined in this report for three spacecraft configurations: The Bus/Lander, All-Orbiter, and Orbiter/Lander. Each subsystem comprises an S-Band Deep Space Transmission Subsystem for tracking and communications with Earth; a Command and Computer Subsystem for control of all vehicle subsystems; and a Data Processing and Storage Subsystem for collection of data from all sensors. In addition, the Orbiter/Lander Communications includes a VHF Relay Transmission Subsystem for the relay of Lander telemetry and command data to and from the Earth via the Orbiter.

Most techniques and component types are the same as those recommended in the previous GE-Voyager Design Study for the Saturn 1B launch vehicle; however,

a. No relay capability is included in the All-Orbiter or the Lander of the Bus/ Lander configuration

b. All thermoplastic recorders (TPR's) have been replaced by magnetic tape recorders.

The relay capability is not included in the above vehicles to eliminate the dependence of the Lander on the separately launched Orbiter.

Magnetic tape recorders are used because TPR's as defined in the previous report are not expected to be within the state-of-the-art in the required time period. Although subsystem flexibility is reduced by these changes, performance degradation in the Titan IIIC systems is not significant.

4.1.2 LINK DESCRIPTIONS

All communication links provided for each mission are shown in Figures 4.1-1, 4.1-2, and 4.1-3. The numbering system used to designate the various links is identical for all missions. 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 or Bus to Earth through high-gain antenna.

Link (2) Secondary data link from Orbiter or Bus to Earth through omni-antenna. To be used during early transit, during emergencies, and as a backup to link (1).

Link (3) Prime data link from Lander to Earth through high-gain antenna.

Link (4) Secondary data link from Lander to Earth through omni-antenna. To be used to assist in initial acquisition of link (3) and as a backup to link (3).

Link (5) Relay data link from Lander to Orbiter. To be used during Lander surface phase as an alternate to link (3).

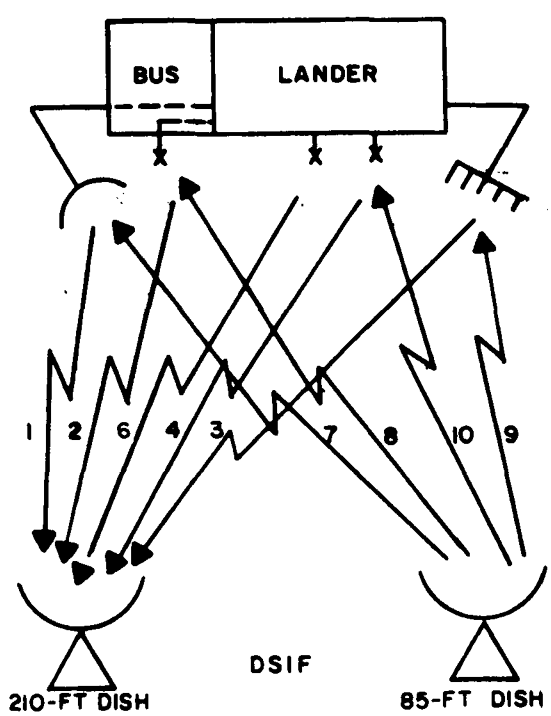

Figure 4.1-1. Bus/Lander Communication Links

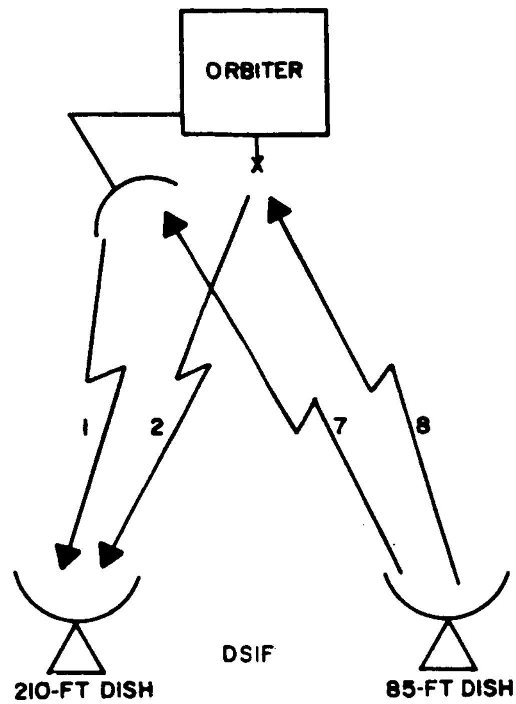

Figure 4.1-2. All-Orbiter Communication Links

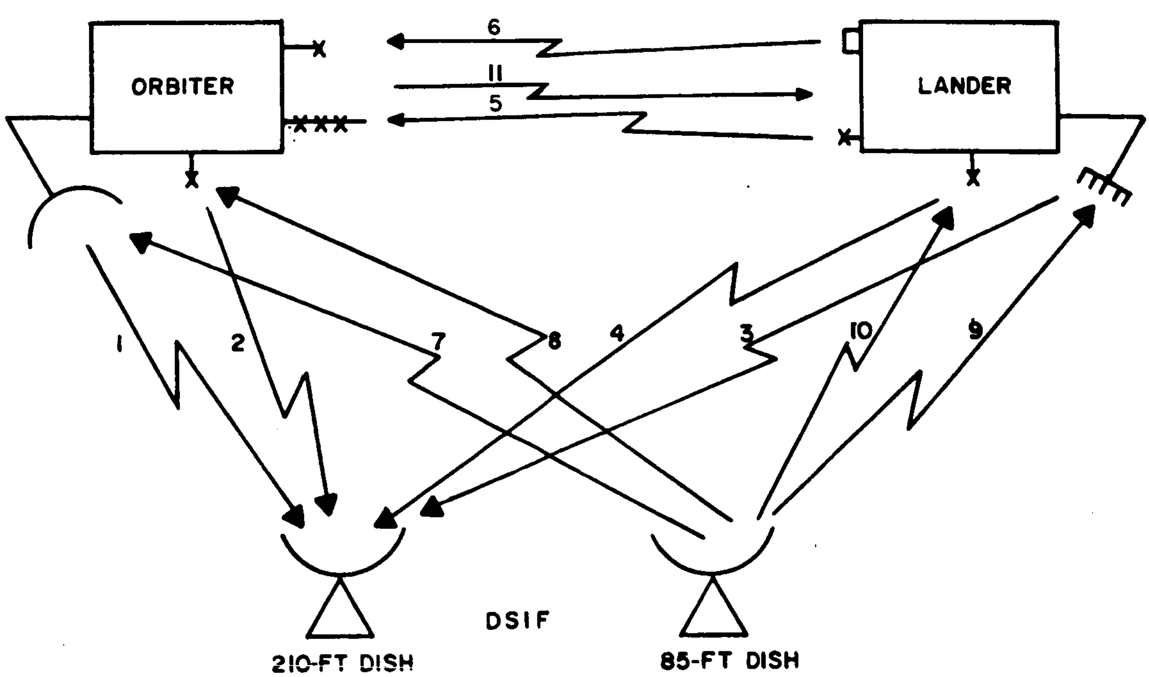

Figure 4.1-3. Orbiter/Lander Communication Links

TABLE 4.1-1. SELECTED DATA RATES (BITS PER SECOND)

Mission \ Link No.	1	2	3	4	5	6	7	8	9	10	11
Bus/Lander	B-E* TLM (Prime)	B-E TLM (Early Transit and Backup)	L-E TLM (Prime)	L-E TLM (Backup)	---	L-E TLM (Descent)	E-B Command (Prime)	E-B Command (Backup)	E-L Command (Prime)	E-L Command (Backup)	---
	400	4	3200 1600 800** 400 200	4	---	4	5	0.5	2	0.5	---
All-Orbiter	O-E TLM (Prime)	O-E TLM (Early transit and Backup)	---	---	---	---	E-O Command (Prime)	E-O Command (Backup)	---	---	---
	24000 12000** 6000 3000 1500	4	---	---	---	---	10	0.5	---	---	---
Orbiter/Lander	O-E TLM (Prime)	O-E TLM (Early transit and Backup)	L-E TLM (Prime)	L-E TLM (Backup)	L-O TLM (Alt. Prime)	L-O TLM (Descent)	E-O Command (Prime)	E-O Command (Backup)	E-L Command (Prime)	E-L Command (Backup)	O-L Command (Alt. Prime)
	12000 6000** 3000 1500	4	3200 1600 800** 400 200	4	12000 6000 3000 1500	100	10	0.5	2.0	0.5	10

*B - Bus
O - Orbiter
L - Lander
E - Earth

**Nominal rate at encounter.

Link (6) Data link for the transmission of pre-entry and atmosphere-descent data from Lander. Direct link to Earth from Lander of Bus/Lander and relay link to Orbiter from Lander of Orbiter/Lander.

Link (7) Prime command link from Earth to Orbiter or Bus through high-gain antenna.

Link (8) Secondary command link from Earth to Orbiter or Bus through omni-antenna. To be used during early transit and as a backup to link (7).

Link (9) Prime command link from Earth to Lander through high-gain antenna.

Link (10) Secondary command link from Earth to Lander through omni-antenna. To be used to assist in the initial acquisition of link (9) and as a backup to link (9).

Link (11) Relay command link from Orbiter to Lander during surface phase. To be used as alternate to link (9).

4.1.3 PERFORMANCE CHARACTERISTICS

The performance of the subsystem for each vehicle is characterized primarily by the data transmission capability of each of its links. A summary of the data rates selected for each link of each mission are given in Table 4.1-1. In general, at least an eight-db margin has been included in each prime data and command link at maximum operating range. The weakest backup links have approximately an eight-db margin at encounter.

4.2 GUIDANCE AND CONTROL

4.2.1 GUIDANCE SUBSYSTEM

Additional detailed analysis of the Guidance subsystem was not an objective of this study. By definition the subsystem is the same as in the previous Voyager study, and consists of the DSIF transponder plus a TV camera to transmit pictures of the planet and stars during the approach phase. From these the time profile of line-of-sight to the planet is obtained.

During the study period, however, independent company-funded work was carried on which produced a very simple passive means of accommodating the very large range of effective brightness between a planet and stars to 5th or 6th magnitude.

Camera accuracy has been re-evaluated, with the conclusion that the previously quoted figure, ± 1 milliradian, is a $3\,\sigma$ value.

Navigation accuracy has improved significantly over the results of the Saturn I-B Voyager study. Some of the reasons are:

1. Initial Navigation uncertainties at the beginning of the approach phase are based on JPL's present estimates of DSIF-based trajectory determination.
2. The 1 milliradian error assumed for line-of-sight determinations is now considered to be a 3 σ value.
3. Accuracy in 1971 is improved over 1969 because of differences in the trajectories.

Although navigation uncertainties, based on DSIF inputs alone, are considerably less than indicated in the Saturn 1B Voyager study, they are still too great to assure hitting an entry corridor between 20 degrees and 35 degrees with the lander or to achieve the desired orbit. Approach guidance inputs are therefore still required. With the use of approach guidance, the navigation errors at 140,000 n.mi. from the planet are as follows:

Errors in Impact Parameter Plane	Standard Deviation (Nautical Miles)
In-plane	26.2
Out-of-plane	24.4

The time of arrival uncertainty is 59 seconds.

If projected ahead to the point of closest approach the errors become:

Errors in Impact Parameter Plane	Standard Deviation (Nautical Miles)
In-plane (radial)	24.3
Out-of-plane	16.4
In direction of velocity	136

Lander guidance studies covered analysis of the problem of achieving entry within an entry angle corridor ranging from 20 degrees to 35 degrees, plus the associated surface dispersion. For a nominal 30-degree entry angle a 3 σ value of entry angle dispersion of 2.43 degrees was obtained. Most of the error is due to navigation errors.

This differs from the previous study where the rocket that put the Lander on its impact trajectory was oversized so that a landing appreciably out of the approach trajectory plane would be possible. Excess rocket impulse increases errors rapidly.

Surface dispersions are as follows (3σ):

	Down Range (Degrees)	Cross Range (Degrees)
Navigation errors	3.1	1.2
Execution errors	0.58	1.3
Total	3.2	1.8

4.2.2 CONTROL SUBSYSTEM

The control subsystem is essentially the same as in the previous Voyager study, and consists of the following:

A. ATTITUDE CONTROL

1. Gyros (3)
2. Gyro control
3. Accelerometer (3 axis)
4. Autopilot amplifier
5. Sun sensors, fine and coarse
6. Canopus tracker
7. Logic, storage, and relay units
8. Power supply

B. ANTENNA CONTROL

1. Antenna drive electronics
2. Earth sensor
3. Antenna actuators (2)

C. PHP CONTROL (ORBITERS ONLY)

1. Horizon scanner (IR)
2. PHP drive electronics
3. PHP actuators (3)

The block diagrams of these control subsystems are shown in Figures 4.2-1 and 4.2-2.

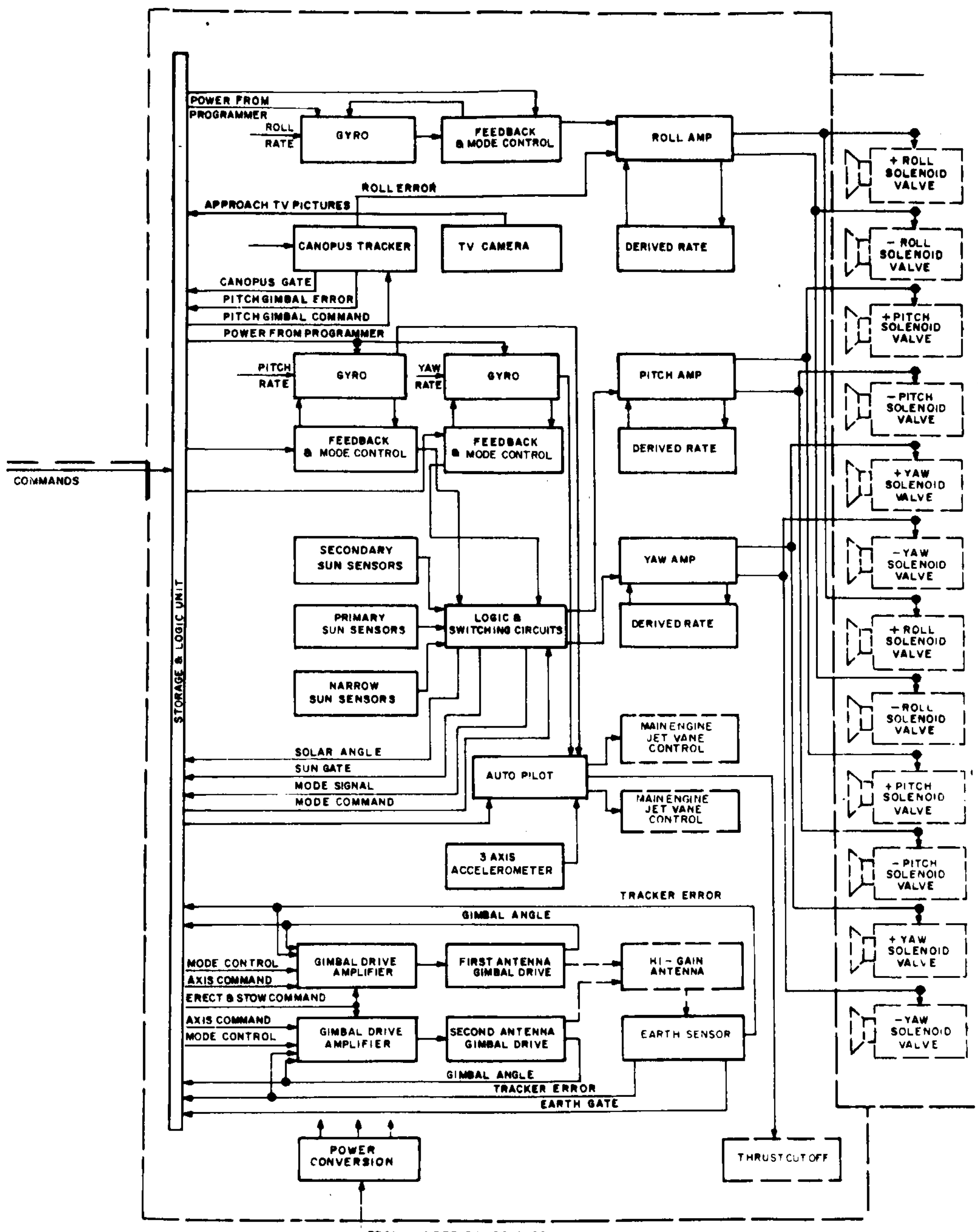

Figure 4.2-1. Guidance and Control Subsystem - Bus/Lander

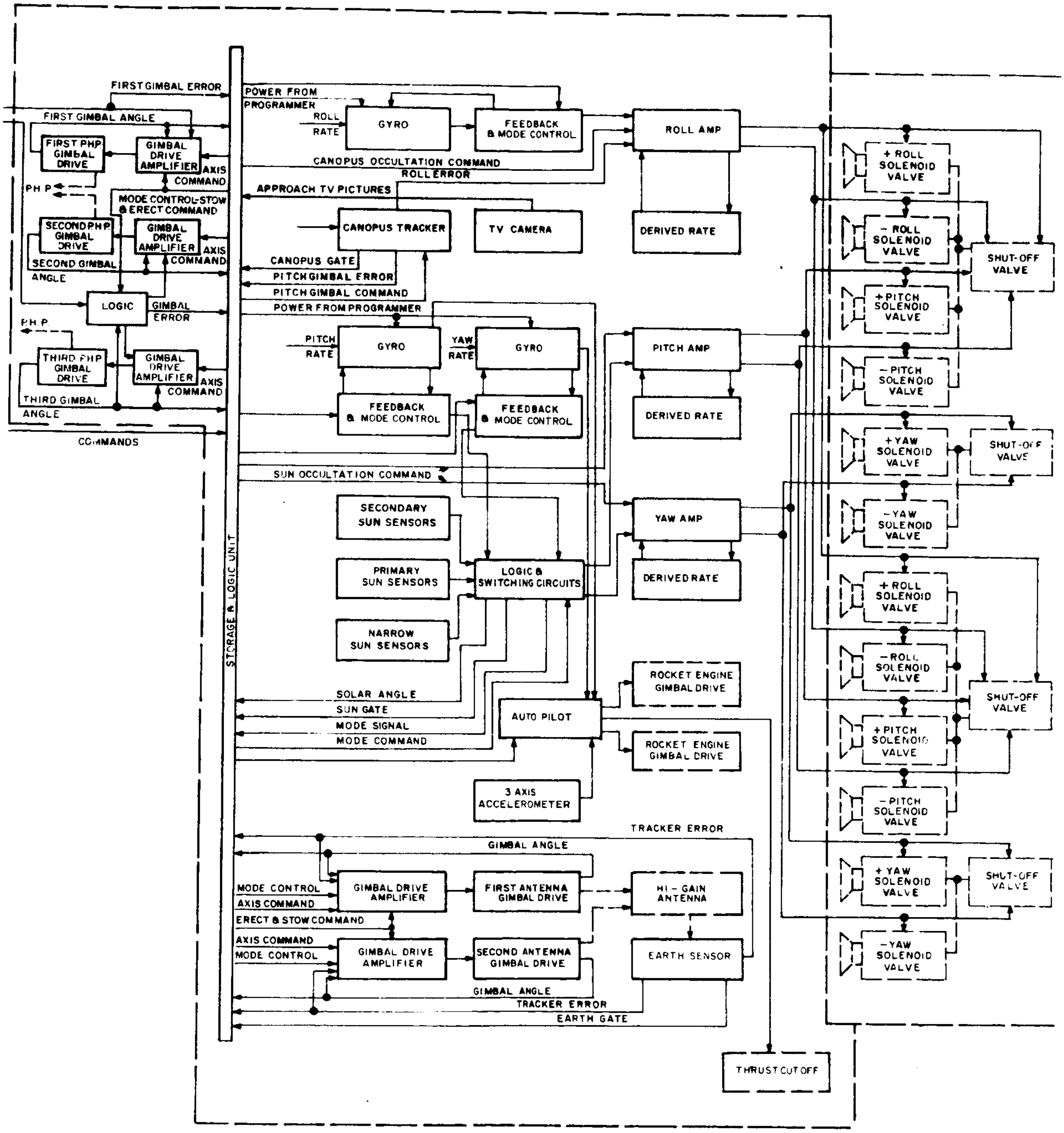

Figure 4.2-2. Guidance and Control Subsystem - All-Orbiter and Orbiter/Lander

Control Subsystem studies consisted of recalculating the impulse requirements for attitude control of the vehicles identified in this study, plus modifying the PHP drive and the Lander antenna drive.

For the Bus/Lander the PHP is deleted. Both classes of Orbiter have 3-axis drives for the PHP. Two of the drives share one channel of the IR planet horizon sensor, according to position in the orbit.

A simplified mode of erecting an equatorial axis was defined for the Lander antenna, to facilitate continuous tracking of Earth.

Impulse requirements for attitude control were calculated as follows:

All-Orbiter	1082 pound-seconds
Bus/Lander	255 pound-seconds
Orbiter/Lander	1017 pound-seconds

4.2.3 PLANET-ORIENTED ORBITER

The principal difficulty in control of a planet-oriented orbiter is found in control of the high-gain antenna to the Earth. Orbit-to-orbit reacquisition of Earth can be accomplished by giving the antenna its own celestial reference, equivalent to the references used by the other Orbiters. The problem is in initial Earth acquisition after injection into orbit, or in case a reacquisition should later become necessary. The fact that the vehicle is rotating at orbital rate adds considerable difficulty to any programmed search sequence; it would possibly result in an appreciable constraint on the antenna drive configuration (probably 3 axis). This appears to be an important enough problem to warrant special attention before making a decision to orient to the planet.

4.3 POWER SUPPLY

In the Voyager-Saturn 1B Study, a detailed investigation was made of the following potential power supplies for unmanned Mars Missions.

1. Nuclear Reactor Thermoelectric
2. Nuclear Reactor Turboelectric
3. Nuclear Reactor Thermionic
4. Radioisotope Thermoelectric

5. Radioisotope Thermionic
6. Solar Thermoelectric
7. Solar Thermionic
8. Solar Dynamic (Rankine)
9. Solar Dynamic (Stirling)
10. Solar Photovoltaic
11. V-Ridge Solar Photovoltaic
12. Concentrated Solar Photovoltaic
13. Primary H_2-O_2 Fuel Cells
14. Secondary Nickel Cadmium Batteries
15. Secondary Silver Cadmium Batteries
16. Primary Silver Zinc

As a result of this study the following recommendations were made for a 1969 Voyager-Saturn 1B Mission.

Lander - Radioisotope Thermoelectric Generator with Secondary Nickel Cadmium Batteries for handling peak loads.

Orbiter - Solar cells with secondary Nickel Cadmium Batteries for handling the energy storage requirements.

With one exception, the conclusions drawn for the 1969 Voyager-Saturn 1B Study are valid for a 1971 Voyager-Titan III Mission. In the Voyager-Saturn 1B Study radioisotope thermoelectric power supplies appeared very attractive for the Mars Orbiter, but were rejected for Radioisotope availability reasons. The availability of the desired Radioisotopes, Plutonium 238 and Curium 244 improves significantly between 1969 and 1971 so that Radioisotope availability is no longer an obvious reason for ruling out Radioisotope Thermoelectrics for the Mars Orbiter. For this reason the Voyager-Titan III Study concentrated on the following as potential power supplies.

- For Mars Orbiter and Transit Bus

  1. Solar Cells
  2. Radioisotope Thermoelectrics
  3. Secondary Nickel Cadmium Batteries

- For Mars Lander

  1. Radioisotope Thermoelectrics
  2. Secondary Nickel Cadmium Batteries

The resulting power supply selection for the various types of missions considered are summarized in Table 4.3-1.

TABLE 4.3-1. RECOMMENDED POWER SYSTEMS

Mission	Energy Conversion Device	Power Output of Energy Conversion Device at Load-Watts	Energy Storage Device	Capacity of Energy Storage Device Amp-Hrs
Bus/ Lander	RTG	170	Secondary Nickel Cadmium Batteries	8
Orbiter	Solar Cells	592	"	8.5
Orbiter/ Lander	Solar Cells/ RTG	328/110	"	2.5

The details supporting the selection of these power systems and the predicted performance is covered in Volume II.

4.4 PROPULSION

Five separate propulsion systems are required for the Bus/Lander and Orbiter/Lander and two for the Orbiter. A summary of paramaters for these systems are given in Table 4.4-1.

For the main propulsion systems, solids and high performance bi-propellants were considered but the increase in potential performance was very slight over the N_2O_4/ 50-50 which was selected. Ablative and radiative chambers were considered; the ablative chamber was selected. A stored-gas unheated pressurization system was selected based on maximum reliability. Thrust level, expansion ratio, and chamber contour were optimized on a weight basis taking into consideration the entire structure weight. A number of expansion systems were considered; a unique partial-diaphragm system was selected. Provisions are made to expel pressurant gas from the system after orbit injection in order to change the orbit slightly. Redundancy is used such that no single malfunction except a structural failure or thrust-chamber failure will cause propulsion system failure.

TABLE 4.4-1. TITAN III VOYAGER PROPULSION SYSTEMS

Propulsion	Use	Type System	Propellant	Total Impulse (Pound-Seconds)	Specific Impulse (Seconds)	Thrust Level (Pounds)	System Weight (Pounds)	Propellant Weight (Pounds)
Bus/Lander								
In-Transit Propulsion	In-Transit Adjustments	Mono-propellant	Hydrazine	11,200	230	50	88.2	48.9
Δ V Motor	Lander Velocity Vector Adjustment	Solid	(1)	19,000	230	1,900	94.0	82.0
Attitude Control Propulsion	Attitude Control	Cold Gas	Freon-14	255	45.3	0.01	40.9	16.5
Spin System	Lander Spin-Up	Cold Gas	Nitrogen	989	70	(1)	47.9	13.4
Retardation Motor	Lander Retardation At Planet Surface	Solid	(1)	6,000	160	2,000	41.0	37.0
Orbiter								
Main Propulsion System	In-Transit Adjustments & Injection into Orbit	Bi-Propellant	N_2O_4/ 50-50	500k	308	900	1925(3)	1634
Attitude Control Propulsion	Attitude Control	Cold Gas	Freon-14	1,082	45.3	0.01(2)	56.1	23.4
Orbiter/Lander								
Main Propulsion System	In-Transit Adjustments & Injection into Orbit	Bi-Propellant	N_2O_4/ 50-50	210k	308	400	903(3)	720
Δ V Motor	Lander Velocity Vector Adjustment	Solid	(1)	12,000	230	1,200	62.0	52.0
Attitude Control Propulsion	Attitude Control	Cold Gas	Freon-14	1,017	45.3	0.01	54.0	21.8
Spin System	Lander Spin-Up	Cold Gas	Nitrogen	560	70	(1)	28.3	7.5
Retardation Motor	Lander Retardation At Planet Surface	Solid	(1)	3,850	160	1,300	27.0	24.0

Notes: (1) Not Selected (2) 0.1 Pound for Roll (3) Does not include gimballing

For the in-transit propulsion system, a pressurized, catalytic-start, hydrazine system was selected. Peroxide, bi-propellant, and hydrazine blow-down systems were considered, but were rejected on the basis of weight, reliability and development risk, respectively. The system utilizes the jet vane system used on Mariner. The use of redundancy assures that only a structural failure or double failure will cause system failure.

For the attitude control systems, Freon-14 was chosen on the basis of minimum weight. Redundancy is used to assure that only a double or structural failure will cause mission failure. For the Bus/Lander system, three times the normally required amount of gas is available; a structural failure will not cause mission failure in this case. The systems are sterilized internally prior to assembly into the spacecraft, and the propellant is sterilized prior to filling.

The spin systems utilize nitrogen gas. Freon-14, solid motors, and a solid gas generator were considered. Nitrogen gas was selected since weight was not a serious problem, and it represented maximum reliability. A solid gas generator was recommended earlier but the inert weight became a critical factor. Tanks were designed to give a factor of safety of 2.0 during heat sterilization.

The ΔV and retardation motors were designed for a sterilizable propellant with a specific impulse of 230 seconds, although no specific propellant was selected. The retardation motor uses two nozzles canted 45 degrees from the support centerline; with this configuration, the system specific impulse is 160 seconds.

5. RELIABILITY AND VALUE ANALYSIS

5.1 RELIABILITY EVALUATION OF CONFIGURATION STUDIED

Reliability analyses were made of several alternates of the following configurations or systems in an effort to obtain a reasonably accurate indication of the attainable system reliability directed toward the optimization of system concepts:

1. Impacting Bus versus fly-by Bus
2. Integrated Bus/Lander versus separate Bus
3. Solar power Orbiter versus RTG power Orbiter
4. Bus/Lander system
5. All-Orbiter system
6. Orbiter/Lander system

The reliability evaluation of the integrated or Separate Bus and the impacting or fly-by trajectory indicates that the integrated fly-by Bus concept is the most acceptable for several reasons.

5.2 RELIABILITY ANALYSIS OF THE RECOMMENDED SYSTEM

5.2.1 BUS/LANDER SYSTEM

The mathematical model for the Bus/Lander system is

$$R_{(\text{Bus/Lander System})} = R_{(\text{Bus})} \cdot R_{(\text{Lander})}$$

Entering the computed reliability values in this mathematical model gives (For 100 Hours Mission)

$$R_{(\text{Bus/Lander System})} = (0.915)\ (0.760)$$
$$= 0.696$$

(For 3 Months Mission)

$$R_{(\text{Bus/Lander System})} = (0.915)\ (0.704)$$
$$= 0.645$$

For a summary of the Bus/Lander System reliability estimates, see Table 5.2-1.

TABLE 5.2-1. RELIABILITY SUMMARY FOR BUS/LANDER SYSTEM

Bus		Lander		
Subsystem	Reliability	Subsystem	Reliability	
	Transit		100 Hour	3 Months
Communications	0.999	Communications	0.863	0.815
Guidance and Control	0.920	Power Supply	0.970	0.959
Hot Gas Propulsion	0.999	Propulsion and Separation	0.972	0.972
Cold Gas Propulsion	0.997	Thermal Control	0.957	0.947
		Retardation	0.984	0.984
		Orientation	0.993	0.993

5.2.2 ORBITER SYSTEM

The mathematical model of the Orbiter is

$$R_{(Orbiter)} = R_{(Communications)} \cdot R_{(G\&C)} \cdot R_{(Power\ Supply)} \cdot R_{(Hot\ Gas\ Prop.)} \cdot R_{(Cold\ Gas\ Prop.)}$$

Entering the computed reliability values in this mathematical model gives

(For 100 Hours Orbit)

$$R_{(Orbiter)} = (0.866)\ (0.912)\ (0.980)\ (0.998)\ (0.996) = 0.768$$

(For 3 Months Orbit)

$$R_{(Orbiter)} = (0.793)\ (0.831)\ (0.973)\ (0.998)\ (0.990) = 0.633$$

For a summary of the Orbiter System reliability estimates, see Table 5.2-2.

TABLE 5.2-2. RELIABILITY SUMMARY FOR ORBITER SYSTEM

Subsystem	Reliability	
	100 Hours	3 Months
Communications	0.866	0.793
Guidance and Control	0.912	0.831
Power Supply	0.980	0.973
Hot Gas Propulsion	0.998	0.998
Cold Gas Propulsion	0.996	0.990
Orbiter System	0.768	0.633

5.3 ANALYSIS OF RELIABILITY AND MISSION VALUE

In order to make a comparison between the Titan IIIC and Saturn 1B systems capabilities, the value of one completely successful Saturn 1B Orbiter plus the value of one completely successful Saturn 1B Lander, in which each module carries the same complement of instruments as in the October 15, 1963 Voyager Report (63SD801 Vol. II), was considered as a basic unit mission value.

The reliability of each system has been established by detailed analyses as a best estimate of the probability of success of the system as applied to the specified mission.

The product of the mission values available from a particular Lander or Orbiter complement of scientific instruments times the probability of its successful completion of the mission is a measure of the mission value most likely to be attained. This value for a single launch is, of course, less than 100 percent of the basic unit mission value defined above. Where more than one launch is involved, and thus the possibility of more than one successful Orbiter and more than one successful Lander with different orbits and different landing sites is involved, the values attainable exceed those available from a single launch and thus, in multiple launches more than 100 percent of a single basic unit mission value is attainable. And, the attainable mission values in Figure 5.3-1, and in the various other figures and tables, correspondingly show figures of greater than 100 percent where more than one system (or more than one Lander) is launched.

Figure 5.3-1 illustrates the mission values attainable using the Titan IIIC-Voyager system recommended by this study for the 1971 opportunity. This system configuration includes a small, controlled, roving instrument carrier in the large Lander and the use of high resolution, three-meter resolution mapping capability as well as an upper atmospheric sampling capability in a sterilized Orbiter as the most valuable use of most of their extra payload carrying capabilities. A significant portion of the extra payload remains available for some further improvement in reliability. The mission values obtainable using the Saturn 1B and the same single large Lander are also provided for comparison.

Figure 5.3-2 illustrates a similar system but one in which sterilization of the orbiter is not required and in which the high resolution mapping and upper atmospheric data values are not obtained.

Many configurations and mission value combinations are possible and all are strongly dependent upon the relative point values considered applicable to each particular instrument or experiment in the light of prior available data (and confidence) and of the principal objectives of the missions under consideration.

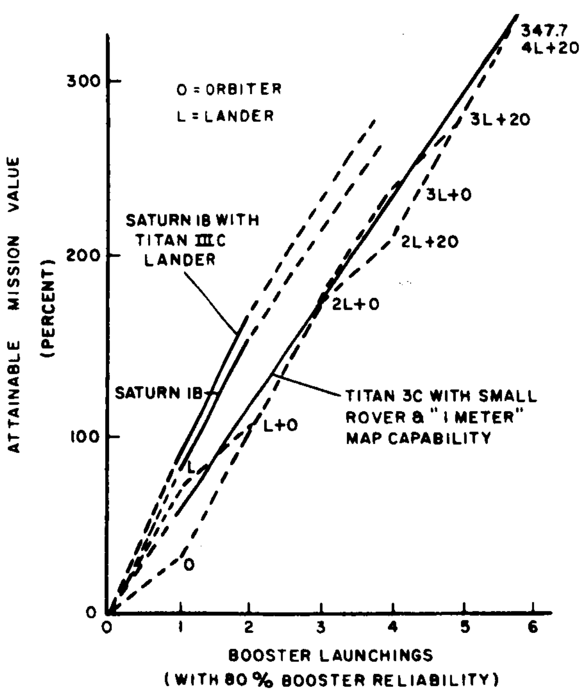

Figure 5.3-1. Attainable Mission Value Recommended for 1971 Opportunity

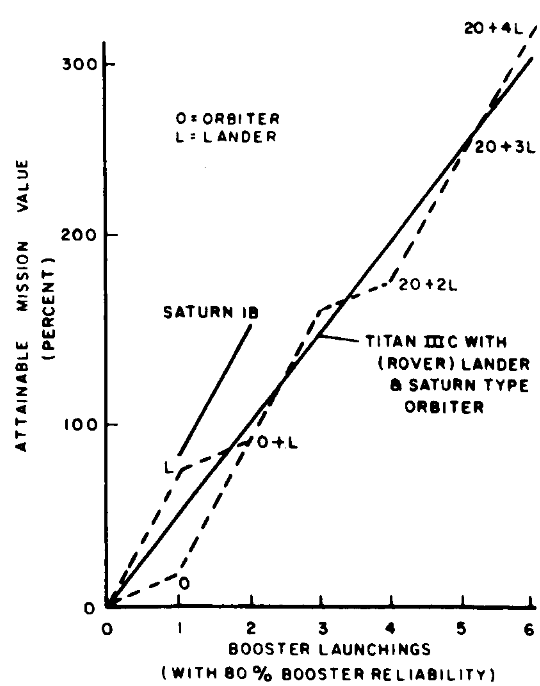

Figure 5.3-2. Attainable Mission Value Sterilization Not Required

The summary data presented in this volume is considered representative and illustrative both of the best estimates and of the range of values involved in comparing Titan IIIC-Voyager capabilities with those of the Saturn 1B-Voyager study. Additional detail is provided in Volume II.

6. PROGRAM PLANS AND COMPARISONS

6.1 PROGRAM PLAN

6.1.1 SUMMARY

The Titan IIIC Voyager Program has been planned for the design, qualification, manufacture and test of spacecraft for a 1971 Mars mission. This mission is comparable in objectives and attainable mission value to the Saturn 1B Voyager mission defined during the previous Voyager Design Study.

The spacecraft required to implement this equivalent program for cost estimating and scheduling purposes have been assumed as follows. This assumption is based on a conservative interpretation of mission value analyses and is discussed in Section 6.2.2, Definition of Equivalent Systems and Programs.

3 - Orbiters - (2 flight units, 1 backup unit plus replaceable spare components)
5 - Landers - (3 flight units, 1 backup unit and 1 sterile spare unit)
4 - Buses - (3 flight units, 1 backup unit plus replaceable spare components).

The program cost estimates, schedules and development problems summarized in Figure 6.1-1 relate to the design, qualification, manufacture and test of the above spacecraft. Costs of scientific payload, TV, RTG units, launch vehicles and post-launch activities are not included.

The above program involves simultaneous development and manufacture of the Orbiter and Lander spacecraft, which was necessary in the Saturn 1B Voyager program since the Orbiter served as a Bus and communications relay for the Landers. However, use of the Titan IIIC launch vehicle and the concepts developed during this study permit the separation of Orbiter and Bus/Lander programs and missions, if desired.

The costs for such separate programs, the combined program and the Saturn 1B Voyager program are shown in Figure 6.1-2.

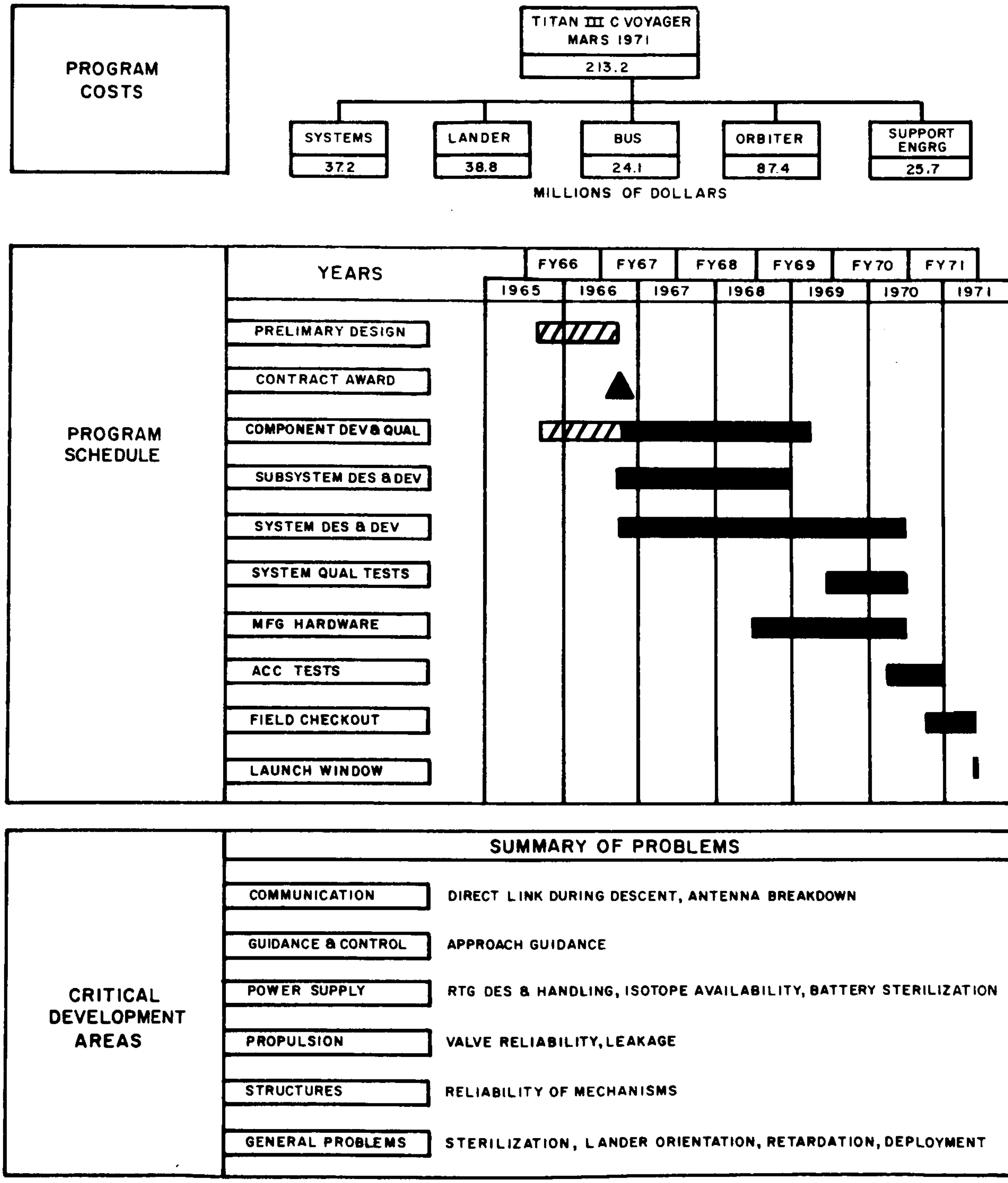

Figure 6.1-1. Program Plan Summary

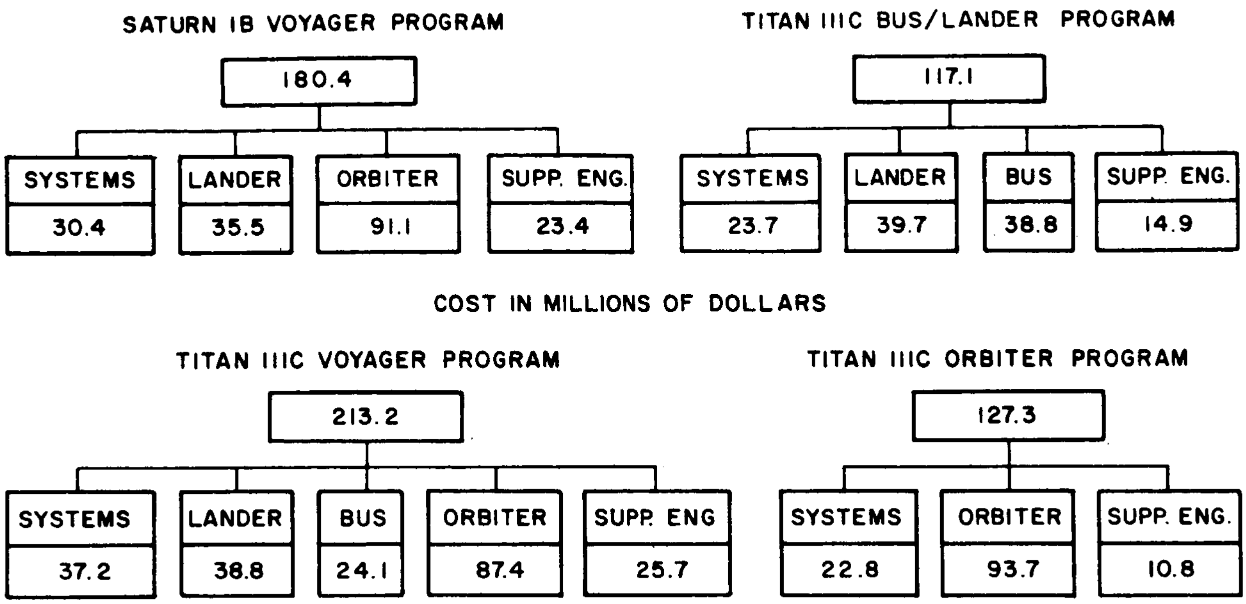

Figure 6.1-2. Program Cost Summary

6.2 PROGRAM COST AND SCHEDULE COMPARISONS

6.2.1 PROGRAM COSTS AND SCHEDULES

The comparisons of Titan IIIC and Saturn 1B Voyager program costs and schedules are summarized in Figure 6.2-1.

It will be noted that the major factor in increasing Titan IIIC program costs is the requirement for a Bus vehicle, which is not a part of the Saturn 1B Voyager. The cost comparison shown is for two Orbiter and three Bus/Lander flights requiring a total of five Titan IIIC launch vehicles against two Orbiters and four Landers using two Saturn 1B launch vehicles. Comparable back-up and spare units were assumed in both cases.

The schedule for performance of the Titan IIIC program has been increased five months in duration between contract award and launch to permit development and qualification of the increased number of types of spacecraft. This additional time has been allocated to that portion of the program where system integration and development are taking place.

PROGRAM COST COMPARISON		SATURN I B	TITAN III C	
SYSTEMS	PROG. MG'MT, RELIABILITY, SYS. TESTS	30, 456	37,980	
ORBITER	ALL OTHER COSTS FOR: 3 ORBITERS	91, 083	87,399	3 ORBITERS
LANDER	ALL OTHER COSTS FOR: 7 LANDERS	35, 467	38,771	5 LANDERS
BUS	ALL OTHER COSTS FOR:	NONE	24,113	4 BUSES
GSE	MFG, INSTALLATION, CHECKOUT	23, 423	25,724	
TOTAL COST		180, 429	213,187	

COSTS IN THOUSANDS OF DOLLARS

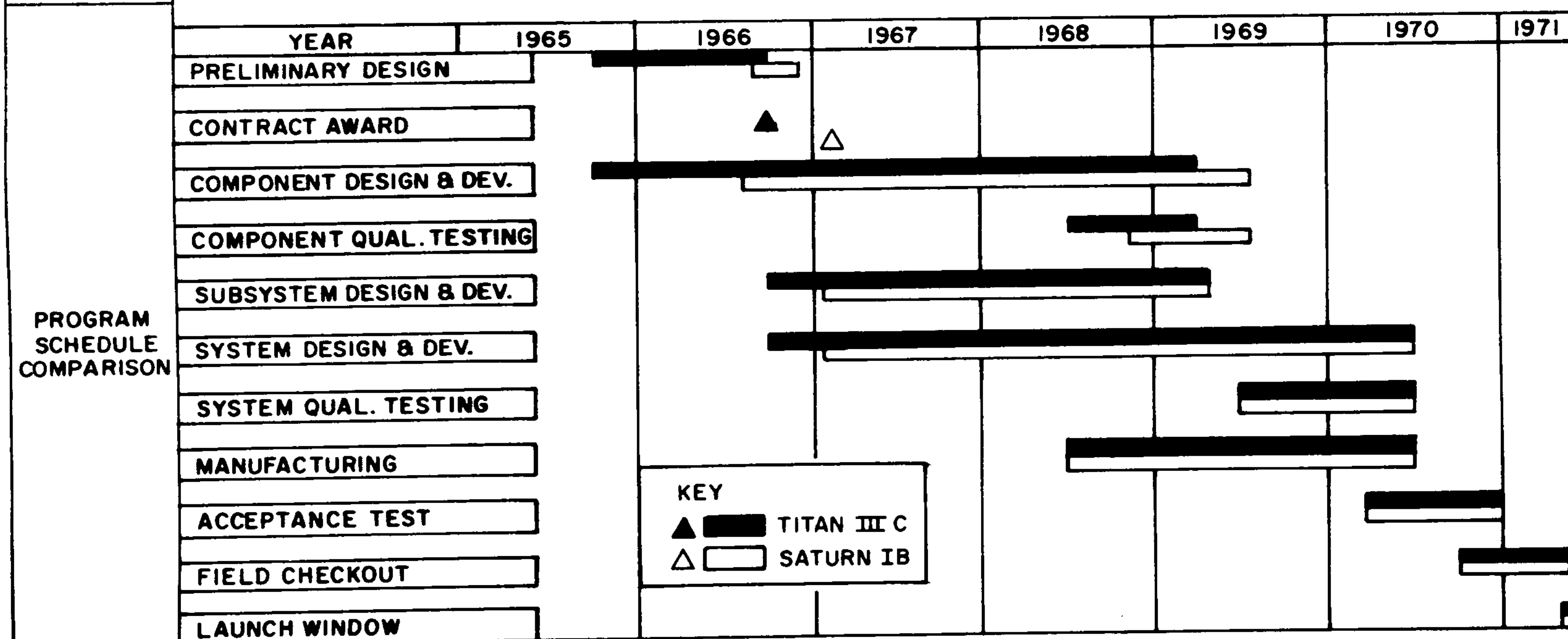

Figure 6.2-1. Summary of Program Costs and Schedule Comparisons

The one year preliminary design period to permit preliminary design, NASA evaluations and critical component development is considered to be more realistic than the four-month period indicated on the Saturn 1B schedule. Costs for this period are not included in this study.

6.2.2 DEFINITION OF EQUIVALENT SYSTEMS AND PROGRAMS

The comparisons of costs and schedule are made between the Saturn 1B Voyager System and Titan IIIC Voyager System for missions estimated to be capable of yielding similar attainable mission values.

Reliability and mission value analyses have been performed as a part of this study (refer to Volume II, Section 5). They indicate that mission values attained by a Titan IIIC Voyager system, consisting of two Orbiter and three Lander/Bus launches, may vary over a range from 106 percent to 180 percent of the values attained by the Saturn 1B Voyager system, consisting of two Orbiters and four Landers (two Saturn 1B launches), depending on the payload complement and reliability estimates employed.

The most conservative value, 106 percent, is based on the same scientific payload for Titan IIIC spacecraft as for Saturn 1B spacecraft, with the additional payload weight capability of the Titan IIIC spacecraft being utilized to increase reliability.

The more optimistic estimate, 180 percent, is based on the inclusion of a "rover" payload in each Titan IIIC Lander with a resulting value increase due to multiple site capability.

Since the concepts and analyses for such a "rover" were not included in this study and its applicability to Titan IIIC versus Saturn 1B has not been evaluated, the more conservative approach to definition of an equivalent system for estimating Titan IIIC spacecraft costs has been taken. The outcome of future "rover" and scientific payload studies could appreciably alter the composition of equivalent Titan IIIC and Saturn 1B Voyager systems.

The following equivalent systems were defined for spacecraft cost and schedule comparison purposes:

Saturn 1B	Titan IIIC Equivalent
2 Orbiters	2 Orbiters
4 Landers	3 Landers (with Buses)

6.2.3 COST-VALUE RELATIONSHIPS

The uncertainties in estimates of attainable mission values and launch vehicle costs make a parametric plot of their relationships a useful tool in understanding their effects on total program costs.

The following curves are plots of Titan IIIC versus Saturn 1B launch vehicle costs for Titan IIIC and Saturn 1B programs having various cost-value ratios, V_r, where program cost includes launch vehicle and spacecraft costs.

For comparing the cost of a Titan program including five launches (two Orbiters and three Lander/Buses) with a Saturn program of two launches (two Orbiters and four Landers), the following equation is applicable:

$$\frac{5\ T + 213}{2\ S + 180} = V_r$$

where:

T = Titan IIIC launch vehicle cost per launch ($ millions)

and Titan spacecraft program cost = $213 millions

S = Saturn 1B (& SVI) launch vehicle cost per launch ($ millions)

and Saturn spacecraft program cost = $180 million

let:

V_r = 1.0 for programs of equal attainable mission value

V_r = 1.8 where Titan IIIC program yields 180 percent of Saturn 1B program attainable mission value.

V_r = 1.06 where Titan IIIC program yields 106 percent of Saturn 1B program attainable mission value.

Using the cost-value ratios of 1.8 and 1.06, corresponding to the mission value relationships of 180 percent and 106 percent discussed in Section 6.2.2, the above equation has been plotted in Figure 6.2-2, which follows. Assuming launch vehicle costs for Saturn 1B and Titan IIIC of $25 million and $13 million respectively, the L/V cost point shown has been plotted to illustrate use of the curves. Where this point falls below the particular value line of interest, use of Titan is favored; where it falls above the line, use of Saturn is favored. In the example shown, if the Titan program will yield 180 percent of Saturn program attainable mission values, use of the Titan is favored from an overall cost viewpoint. If only 106 percent is obtainable, use of Saturn is favored.

Other values of launch vehicle cost may be substituted for those used in the illustration, and a new determination of the most favorable launch vehicle readily made.

Plots similar to those in the illustration but for an increased range of values are shown in Figure 6.2-3.

Figures 6.2-4 and 6.2-5 present similar data for Titan IIIC programs employing four and three launches, respectively.

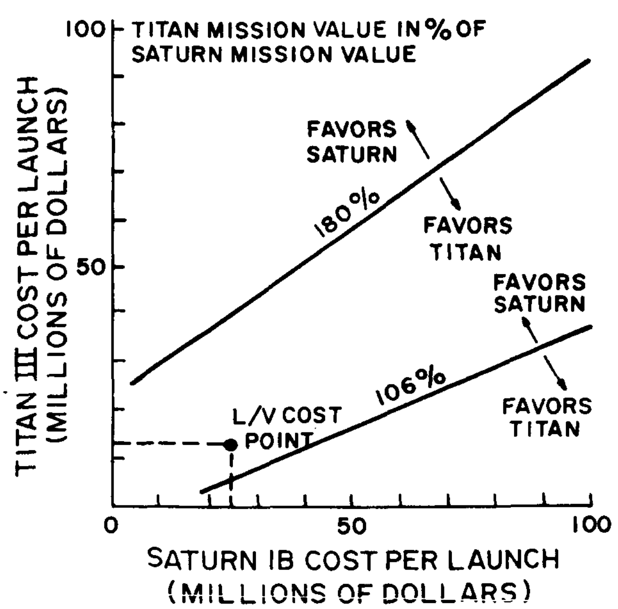

Figure 6.2-2. Titan Mission Value in Percent of Saturn Mission Value

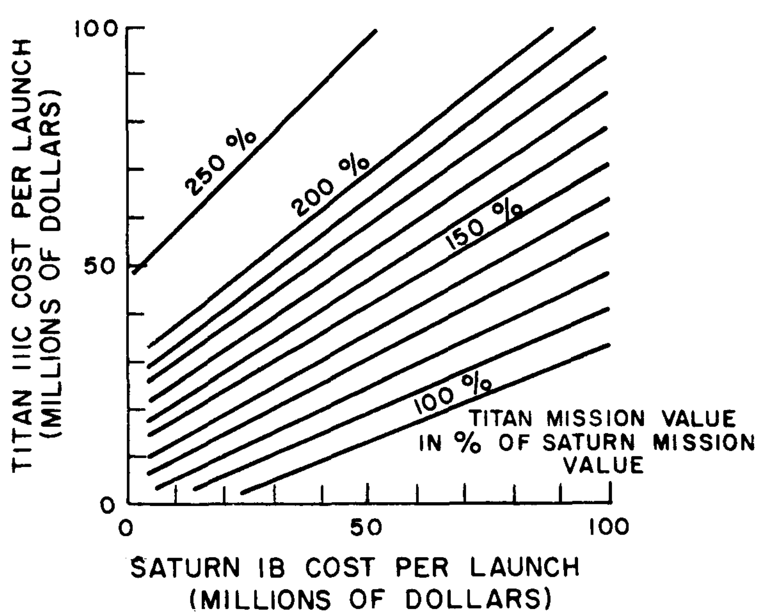

Figure 6.2-3. Titan Mission Value in Percent of Saturn Mission Value

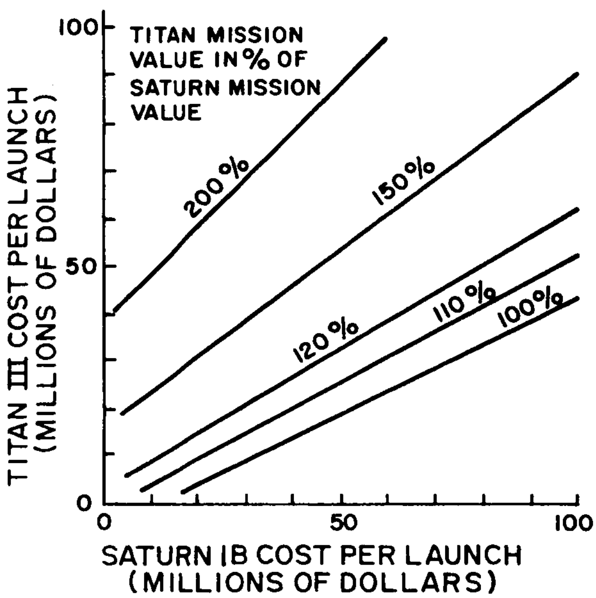

Figure 6.2-4. Titan Mission Value in Percent of Saturn Mission Value for Four Titan Launches

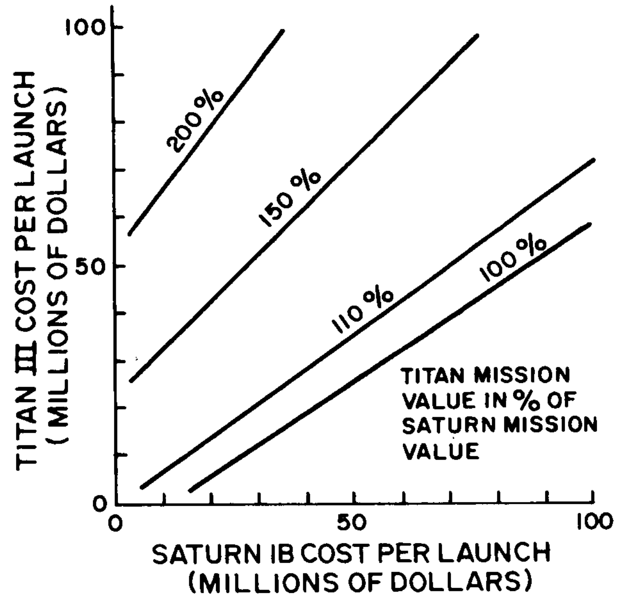

Figure 6.2-5. Titan Mission Value in Percent of Saturn Mission Value for Three Titan Launches