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VOYAGER SPACECRAFT SYSTEM STUDY
(PHASE I-TITAN IIIC LAUNCH VEHICLE)
FINAL REPORT
VOLUME IIb

7 AUGUST, 1964

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4. SUBSYSTEM DESIGN STUDIES

4.1 COMMUNICATIONS

4.1.1 SUMMARY

A. GENERAL

Communication subsystems are analyzed and defined in this report for three vehicle configurations: the Bus/Lander, All-Orbiter, and Orbiter/Lander. Each subsystem comprises an S-Band Deep Space Transmission Subsystem for tracking and for 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 1-B 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b. All thermoplastic recorders (TPR's) have been replaced by magnetic tape recorders.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.

B. 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).

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).

C. 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.

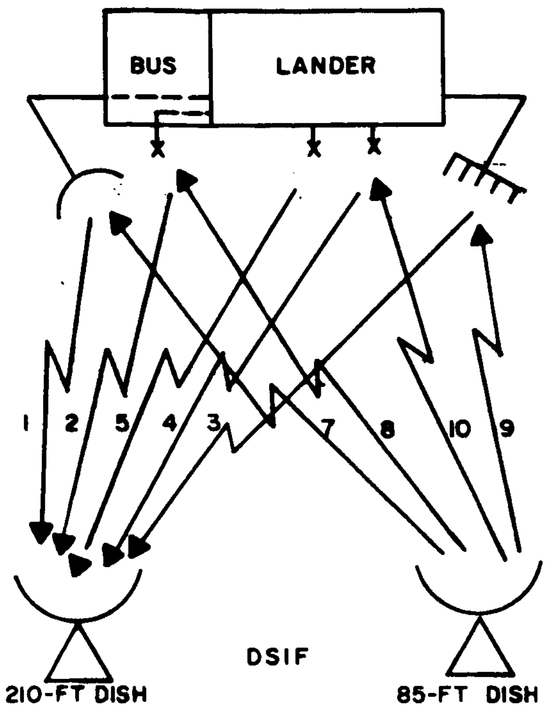


Figure 4.1-1. Bus/Lander Communication Links

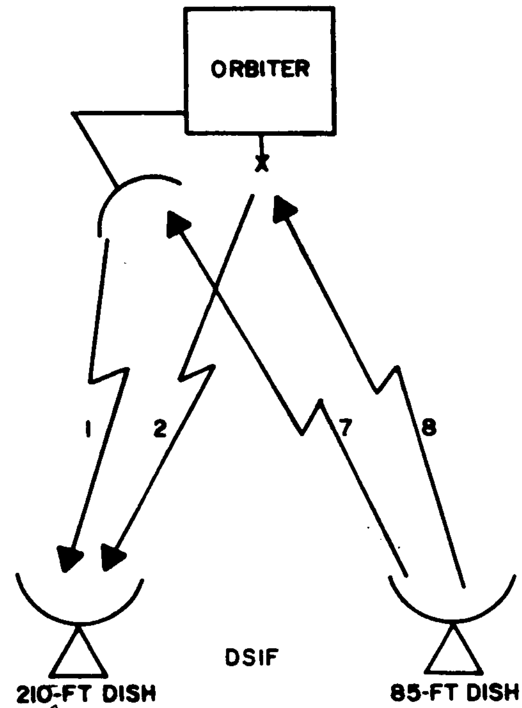


Figure 4.1-2. All Orbiter Communication Links

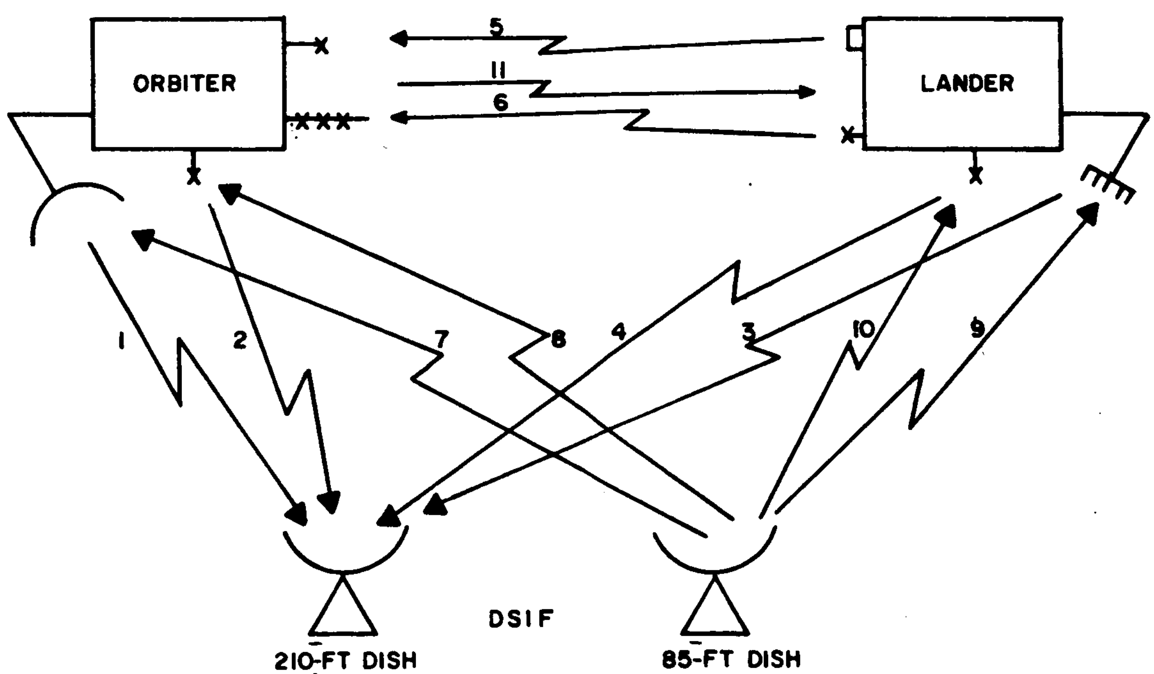


Figure 4.1-3. Orbiter/Lander Communication Links

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.

D. CRITICAL PROBLEM AREAS

1. Electrostatically Focused Klystron (ESFK)

Electrostatically Focused Klystrons have been selected for all applications requiring long life. Output power requirements range from 15 watts to 57 watts. Units in this power range have been built by Litton Industries for the Apollo program; therefore, the primary effort remaining is that of life-testing.

2. Raytheon Amplitron

The Raytheon Amplitron has been selected for applications requiring their unique feed-through feature and high efficiency. Amplitrons are presently available with power outputs up to 70 watts; however, the status of tubes in the 150-watt region is not known. Although development will be required for these high-power tubes, no extensive life tests are required since none are required to operate in any of the systems for more than a few hours.

3. Antenna Breakdown

The antenna configurations selected for operation in the Martian atmosphere are not expected to break down at the power levels required. However, experimental verification will be required early in the program. Of prime importance is the S-band atmospheric-descent antenna which radiates 100 watts (including losses). If power levels in this range cannot be obtained, a VHF relay link will probably be required between Lander and Bus during the descent phase.

4. Descent-Phase Direct Link Simulation

The proposed descent-phase direct link is characterized by a weak received signal, uncertainty of the received signal frequency after entry, and uncertainty of the variations of the received signal frequency during descent. Predetection recording and signal processing will be required to recover the transmitted data. Although this link appears feasible, it is yet to be verified under the expected operating conditions. A simulation of the link is therefore recommended. Included in the simulation should be an evaluation of other modulation, detection (including noncoherent), and synchronization techniques. This simulation should be carried out early in the program since the decision to use either a direct or relay link can affect the design of the Bus and its subsystems considerably.

5. Sterilizable Tape Recorders

Since tape recorders are recommended for the Landers, sterilization will be required. At least one company (Raymond Engineering Laboratory, Inc.) has investigated the resulting problems (JPL contract) and has indicated that solutions are available. H-film, which can withstand the temperatures required, is presently the prime candidate for tape backing. Continued effort is recommended in this area to prove feasibility early in the program.

4.1.2 COMMUNICATION SUBSYSTEM ANALYSIS

A. LINK CALCULATIONS

Link calculations for the three missions are given in Tables 4.1-2, 4.1-3 and 4.1-4. All parameter values utilized in the calculations are given in Tables 4.1-5, 4.1-6 and 4.1-7.

All S-band links are calculated for a transmission range of 1.0 AU, the descent-phase relay links at 3680 n.mi. The latter is the range to a Lander on the planet's horizon when the orbiter is at the altitude (2300 n.mi.) where the 3-db beam width of its antenna is just subtended by the planet.

The following ground rules should also be noted:

a. The product of the gain, pointing loss, and polarization loss of each "omni" antenna has been taken to be unity in the calculations. The actual values in each case will depend on the interacting effects of the radiating elements and the vehicle in addition to the orientation of the vehicle with respect to Earth. With proper design, however, this assumption is valid over most of the solid angle.

b. The gain, pointing loss, and polarization loss given for the Lander direct link descent phase and surface phase encapsulated antennas are based on reception with a linearly polarized antenna.

c. The APC noise bandwidths can be increased for faster acquisition in links where the margin allows and where the modulating sidebands are sufficiently removed from the vicinity of the carrier.

TABLE 4.1-2. LINK CALCULATIONS (BUS/LANDER)

	Link	1	2a	2b	3	4	5	6	7	8	9	10	11
No.	Parameter	B-E TLM (Prime)	B-E TLM (Early Transit)	B-E TLM (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)	-
1	Total Transmitter Power	43.8	41.7	51.7	43.8	51.7		51.7	70.0	80.0	70.0	80.0	
2	Transmitting Circuit Loss	- 2.0	- 2.0	- 2.0	- 2.0	- 2.0		- 2.0	- 0.5	- 0.5	- 0.5	- 0.5	
3	Transmitting Antenna Gain	24.1	0.0	0.0	26.7	2.0		2.0	51.0	51.0	51.0	51.0	
4	Transmitting Antenna Pointing Loss	- 0.2	0.0	0.0	- 1.2	- 2.0		- 2.0	-	-	-	-	
5	Space Loss	-263.2	-263.2	-263.2	-263.2	-263.2		-263.2	-262.5	-262.5	-262.5	-262.5	
6	Polarization Loss	-	0.0	0.0	-	0.0		0.0	-	0.0	-	0.0	
7	Receiving Antenna Gain	61.0	61.0	61.0	61.0	61.0		61.0	23.5	0.0	26.0	2.0	
8	Receiving Antenna Pointing Loss	-	-	-	-	-		-	- 0.2	0.0	- 1.0	- 2.0	
9	Receiving Circuit Loss	- 0.2	- 0.2	- 0.2	- 0.2	- 0.2		- 0.2	- 2.0	- 2.0	- 2.0	- 2.0	
10	Net Circuit Loss	-180.5	-204.4	-204.4	-178.9	-204.4		-204.4	-190.7	-214.0	-189.0	-214.0	
11	Total Received Power	-136.7	-162.7	-152.7	-135.1	-152.7		-152.7	-120.7	-134.0	-119.0	-134.0	
12	Receiver Noise Spectral Density (N/B)	-183.2	-183.2	-183.2	-183.2	-183.2		-183.2	-164.0	-169.0	-164.0	-164.0	
13	Carrier Modulation Loss	- 6.0	- 3.0	- 3.0	- 6.0	- 3.0		- 3.0	- 4.4	- 4.4	- 4.4	- 4.4	
14	Received Carrier Power	-142.6	-165.7	-155.7	-141.1	-155.7		-155.7	-125.1	-138.4	-123.4	-138.4	
15	Carrier APC Noise BW	10.8	10.8	10.8	10.8	10.8		10.8	13.0	10.0	13.0	10.0	
	CARRIER PERFORMANCE TRACKING (one-way)												
16	Threshold SNR in $2B_{LO}$	0.0	0.0	0.0	0.0	0.0		-	-	-	-	-	
17	Threshold Carrier Power	-172.4	-172.4	-172.4	-172.4	-172.4		-	-	-	-	-	
18	Performance Margin	29.8	6.7	16.7	31.3	16.7		-	-	-	-	-	
	CARRIER PERFORMANCE TRACKING (two-way)												
19	Threshold SNR in $2B_{LO}$	2.0	2.0	2.0	2.0	2.0		-	3.0	3.0	3.0	3.0	
20	Threshold Carrier Power	-170.4	-170.4	-170.4	-170.4	-170.4		-	-148.0	-156.0	-148.0	-151.0	
21	Performance Margin	27.8	4.7	14.7	29.3	14.7		-	22.9	17.6	24.6	12.6	

TABLE 4.1-2. LINK CALCULATIONS (BUS/LANDER) (Cont'd.)

No.	Link / Parameter	1 B-E TLM (Prime)	2a B-E TLM (Early Transit)	2b B-E TLM (Backup)	3 L-E TLM (Prime)	4 L-E TLM (Backup)	5 -	6 L-E TLM (Descent)	7 E-B Command (Prime)	8 E-B Command (Backup)	9 E-L Command (Prime)	10 E-L Command (Backup)	11
	<u>CARRIER PERFORMANCE — TELEMETRY AND COMMAND</u>												
22	Threshold SNR in $2B_{LO}$	6.0	6.0	6.0	6.0	6.0		6.0	6.0	6.0	6.0	6.0	
23	Threshold Carrier Power	-166.4	-166.4	-166.4	-166.4	-166.4		-166.4	-145.0	-153.0	-145.0	-148.0	
24	Performance Margin	23.8	0.7	10.7	25.3	10.7		10.7	19.9	14.6	21.6	9.6	
	<u>DATA CHANNEL</u>												
25	Modulation Loss	- 1.3	- 3.0	- 3.0	- 1.3	- 3.0		- 3.0	- 1.9	- 1.9	- 1.9	- 1.9	
26	Received Data Subcarrier Power	-138.0	-165.7	-155.7	-136.4	-155.7		-155.7	-122.6	-135.9	-120.9	-135.9	
27	Bit Rate (1/T)	26.0	6.0	6.0	29.0	6.0		6.0	10.0	0.0	6.0	0.0	
28	Required ST/N/B	6.0	8.0	8.0	6.0	8.0		8.0	18.5	18.5	18.5	18.5	
29	Threshold Subcarrier Power	-151.2	-169.2	-169.2	-148.2	-169.2		-169.2	-135.5	-150.5	-139.5	-145.5	
30	Performance Margin	13.2	3.5	13.5	11.8	13.5		13.5	12.9	14.6	18.6	9.6	
	<u>SYNC CHANNEL</u>												
31	Modulation Loss	- 1.3	- 3.0	- 3.0	- 1.3	- 3.0		- 3.0	- 1.9	- 1.9	- 1.9	- 1.9	
32	Receiver SYNC Subcarrier Power	-138.0	-165.7	-155.7	-136.4	-155.7		-155.7	-122.6	-135.9	-120.9	-135.9	
33	SYNC APC Noise BW	- 3.0	- 3.0	- 3.0	- 3.0	- 3.0		- 3.0	3.0	3.0	3.0	3.0	
34	Threshold SNR in $2B_{LO}$	10.0	10.0	10.0	10.0	10.0		10.0	15.5	15.5	15.5	15.5	
35	Threshold Subcarrier Power	-176.2	-176.2	-176.2	-176.2	-176.2		-176.2	-145.5	-150.5	-145.5	-145.5	
36	Performance Margin	38.2	10.5	20.5	39.8	20.5		20.5	22.9	14.6	24.6	9.6	

TABLE 4.1-3. LINK CALCULATIONS (ALL ORBITER)

No.	Link / Purpose / Parameter	1	2a	2b	3	4	5	6	7	8	9	10	11
		O-E TLM (Prime)	O-E TLM (Early Transit)	O-E TLM (Backup)					E-O Command (Prime)	E-O Command (Backup)			
1	Total Transmitter Power (dbm)	47.6	47.6	50.0					70.0	80.0			
2	Transmitting Circuit Loss	- 2.0	- 2.0	- 2.0					- 0.5	- 0.5			
3	Transmitting Antenna Gain	33.8	0.0	0.0					51.0	51.0			
4	Transmitting Antenna Pointing Loss	- 1.0	0.0	0.0					-	-			
5	Space Loss	-263.2	-263.2	-263.2					-262.5	-262.5			
6	Polarization Loss	-	0.0	0.0					-	0.0			
7	Receiving Antenna Gain	61.0	61.0	61.0					33.0	0.0			
8	Receiving Antenna Pointing Loss	-	-	-					- 0.9	- 0.0			
9	Receiving Circuit Loss	- 0.2	- 0.2	- 0.2					- 2.0	- 2.0			
10	Net Circuit Loss	-171.6	-204.4	-204.4					-181.9	-214.0			
11	Total Received Power (dbm)	-124.0	-156.8	-154.4					-111.9	-134.0			
12	Receiver Noise Spectral Density (N/B) (dbm)	-183.2	-183.2	-183.2					-164.0	-169.0			
13	Carrier Modulation Loss	- 6.0	- 3.0	- 3.0					- 4.4	- 4.4			
14	Received Carrier Power (dbm)	-130.0	-159.8	-157.4					-116.3	-138.4			
15	Carrier APC Noise BW	10.8	10.8	10.8					13.0	10.0			
	<u>CARRIER PERFORMANCE TRACKING (one-way)</u>												
16	Threshold SNR in $2B_{LO}$	0.0	0.0	0.0					-	-			
17	Threshold Carrier Power (dbm)	-172.4	-172.4	-172.4					-	-			
18	Performance Margin	42.4	12.6	15.0					-	-			

TABLE 4.1-3. LINK CALCULATIONS (ALL ORBITER) (Cont'd.)

No.	Link / Purpose / Parameter	1	2a	2b	3	4	5	6	7	8	9	10	11
		O-E TLM (Prime)	O-E TLM (Early Transit)	O-E TLM (Backup)					E-O Command (Prime)	E-O Command (Backup)			
	CARRIER PERFORMANCE TRACKING (two-way)												
19	Threshold SNR in $2B_{LO}$	2.0	2.0	2.0					3.0	3.0			
20	Threshold Carrier Power (dbm)	-170.4	-170.4	-170.4					-148.0	-156.0			
21	Performance Margin	40.4	10.6	13.0					31.7	17.6			
	CARRIER PERFORMANCE — TELEMETRY AND COMMAND												
22	Threshold SNR in $2B_{LO}$	6.0	6.0	6.0					6.0	6.0			
23	Threshold Carrier Power (dbm)	-166.4	-166.4	-166.4					-145.0	-153.0			
24	Performance Margin	36.4	6.6	9.0					28.7	14.6			
	DATA CHANNEL												
25	Modulation Loss	- 1.3	- 3.0	- 3.0					- 1.9	- 1.9			
26	Received Data Subcarrier Power (dbm)	-125.3	-159.8	-157.4					-113.8	-135.9			
27	Bit Rate (1/T)	40.8	6.0	6.0					13.0	0.0			
28	Required ST/N/B	6.0	8.0	8.0					18.5	18.5			
29	Threshold Subcarrier Power (dbm)	-136.4	-169.2	-169.2					-132.5	-150.5			
30	Performance Margin	11.1	9.4	11.8					18.7	14.6			
	SYNC CHANNEL												
31	Modulation Loss	- 1.3	- 3.0	- 3.0					- 1.9	- 1.9			
32	Receiver SYNC Subcarrier Power	-125.3	-159.8	-157.4					-113.8	-135.9			
33	SYNC APC Noise BW	- 3.0	- 3.0	- 3.0					3.0	3.0			
34	Threshold SNR in $2B_{LO}$	10.0	10.0	10.0					15.5	15.5			
35	Threshold Subcarrier Power	-176.2	-176.2	-176.2					-145.5	-150.5			
36	Performance Margin	50.9	16.4	18.8					31.7	14.6			

TABLE 4.1-4. LINK CALCULATIONS (ORBITER/LANDER)

No.	Link / Parameter \ Purpose	1 O-E TLM (Prime)	2a O-E TLM (Early Transit)	2b O-E TLM (Backup)	3 L-E TLM (Prime)	4 L-E TLM (Backup)	5 L-O TLM (Alt. Prime)	6 L-O TLM (Descent)	7 E-O Command (Prime)	8 E-O Command (Backup)	9 E-L Command (Prime)	10 E-L Command (Backup)	11 O-L Command (Alt. Prime)
1	Total Transmitter Power	46.3	46.3	50.0	43.8	51.7	44.0	37.0	70.0	80.0	70.0	80.0	37.0
2	Transmitting Circuit Loss	- 2.0	- 2.0	- 2.0	- 2.0	- 2.0	- 2.0	- 2.0	- 0.5	- 0.5	- 0.5	- 0.5	- 2.0
3	Transmitting Antenna Gain	32.8	0.0	0.0	26.7	2.0	2.0	2.0	51.0	51.0	51.0	51.0	10.0
4	Transmitting Antenna Pointing Loss	- 0.5	- 0.0	0.0	- 1.2	- 2.0	- 3.0	- 3.0	-	-	-	-	- 3.0
5	Space Loss	-263.2	-263.2	-263.2	-263.2	-263.2	-149.2	-152.0	-262.5	-262.5	-262.5	-262.5	-149.2
6	Polarization Loss	-	0.0	0.0	-	0.0	- 3.0	- 3.0	-	0.0	-	0.0	- 3.0
7	Receiving Antenna Gain	61.0	61.0	61.0	61.0	61.0	10.0	2.0	32.0	0.0	26.0	2.0	2.0
8	Receiving Antenna Pointing Loss	-	-	-	-	-	- 3.0	- 3.0	- 0.7	0.0	- 1.0	- 2.0	- 3.0
9	Receiving Circuit Loss	- 0.2	- 0.2	- 0.2	- 0.2	- 0.2	- 1.0	- 1.0	- 2.0	- 2.0	- 2.0	- 2.0	- 1.0
10	Net Circuit Loss	-172.1	-204.4	-204.4	-178.9	-204.2	-149.2	-160.0	-182.7	-214.0	-189.0	-214.0	-149.2
11	Total Received Power	-125.8	-158.1	-154.4	-135.1	-152.7	-105.2	-123.0	-112.7	-134.0	-119.0	-134.0	-112.2
12	Receiver Noise Spectal Density (N/B)	-183.2	-183.2	-183.2	-183.2	-183.2	-169.8	-167.0	-164.0	-169.0	-164.0	-164.0	-167.0
13	Carrier Modulation Loss	- 6.0	- 3.0	- 3.0	- 6.0	- 3.0	- 6.0	- 6.0	- 4.4	- 4.4	- 4.4	- 4.4	- 6.0
14	Received Carrier Power	-131.8	-161.1	-157.4	-141.1	-155.7	-111.2	-129.0	-117.1	-138.4	-123.4	-138.4	-118.2
15	Carrier APC Noise BW	10.8	10.8	10.8	10.8	10.8	19.3	19.3	13.0	10.0	13.0	10.0	19.3
	CARRIER PERFORMANCE TRACKING (one-way)												
16	Threshold SNR in $2B_{LO}$	0.0	0.0	0.0	0.0	0.0	-	-	-	-	-	-	-
17	Threshold Carrier Power	-172.4	-172.4	-172.4	-172.4	-172.4	-	-	-	-	-	-	-
18	Performance Margin	40.6	11.3	15.0	31.3	16.7	-	-	-	-	-	-	-
	CARRIER PERFORMANCE TRACKING (two-way)												
19	Threshold SNR in $2B_{LO}$	2.0	2.0	2.0	2.0	2.0	-	-	3.0	3.0	3.0	3.0	-
20	Threshold Carrier Power	-170.4	-170.4	-170.4	-170.4	-170.4	-	-	-148.0	-156.0	-148.0	-151.0	-
21	Performance Margin	38.6	9.3	13.0	29.3	14.7	-	-	30.9	17.6	24.6	12.6	-

TABLE 4.1-4. LINK CALCULATIONS (ORBITER/LANDER) (Cont'd)

No.	Link / Purpose / Parameter	1 O-E TLM (Prime)	2a O-E TLM (Early Transit)	2b O-E TLM (Backup)	3 L-E TLM (Prime)	4 L-E TLM (Backup)	5 L-O TLM (Alt. Prime)	6 L-O TLM (Descent)	7 E-O Command (Prime)	8 E-O Command (Backup)	9 E-L Command (Prime)	10 E-L Command (Backup)	11 O-L Command (Alt. Prime)
	CARRIER PERFORMANCE – TELEMETRY AND COMMAND												
22	Threshold SNR in $2B_{LO}$	6.0	6.0	6.0	6.0	6.0	6.0	6.0	6.0	6.0	6.0	6.0	6.0
23	Threshold Carrier Power	-166.4	-166.4	-166.4	-166.4	-166.4	-144.5	-141.7	-145.0	-153.0	-145.0	-148.0	-141.7
24	Performance Margin	34.6	5.3	9.0	25.3	10.7	33.3	12.7	27.9	14.6	21.6	9.6	23.5
	DATA CHANNEL												
25	Modulation Loss	- 1.3	- 3.0	- 3.0	- 1.3	- 3.0	- 1.3	- 1.3	- 1.9	- 1.9	- 1.9	- 1.9	- 1.3
26	Received Data Subcarrier Power	-127.1	-161.1	-157.4	-136.4	-155.7	-106.5	-124.3	-114.6	-135.9	-120.9	-135.9	-113.5
27	Bit Rate (1/T)	37.8	6.0	6.0	29.0	6.0	42.1	20.0	13.0	0.0	6.0	0.0	13.0
28	Required ST/N/B	6.0	8.0	8.0	6.0	8.0	10.8	10.8	18.5	18.5	18.5	18.5	13.7
29	Threshold Subcarrier Power	-139.4	-169.2	-169.2	-148.2	-168.2	-116.9	-136.2	-132.5	-150.5	-139.5	-145.5	-140.3
30	Performance Margin	12.3	8.1	11.8	11.8	13.5	10.4	11.9	17.9	14.6	18.6	9.6	26.8
	SYNC CHANNEL												
31	Modulation Loss	- 1.3	- 3.0	- 3.0	- 1.3	- 3.0	- 1.3	- 1.3	- 1.9	- 1.9	- 1.9	- 1.9	- 1.3
32	Receiver SYNC Subcarrier Power	-127.1	-161.1	-157.4	-136.4	-155.7	-106.5	-124.3	-114.6	-135.9	-120.9	-135.9	-113.5
33	SYNC APC Noise BW	- 3.0	- 3.0	- 3.0	- 3.0	- 3.0	3.0	3.0	3.0	3.0	3.0	3.0	3.0
34	Threshold SNR in $2B_{LO}$	10.0	10.0	10.0	10.0	10.0	10.0	10.0	15.5	15.5	15.5	15.5	10.0
35	Threshold Subcarrier Power	-176.2	-176.2	-176.2	-176.2	-176.2	-156.8	-154.0	-145.5	-150.5	-145.5	-145.5	-154.0
36	Performance Margin	49.1	15.1	18.8	39.8	20.5	50.3	29.7	30.9	14.6	24.6	9.6	40.5

TABLE 4.1-5. SUMMARY OF LINK PARAMETERS (LANDER/BUS)

Link No.	1	2	3	4	5	6	7	8	9	10	11
Purpose / Parameter	B-E TLM	B-E TLM (Backup)	L-E TLM	L-E TLM (Backup)	-	L-E TLM (Descent)	E-B Command	E-B Command (Backup)	E-L Command	E-L Command (Backup)	-
Frequency (mc)	2295	2295	2295	2295	-	2295	2115	2115	2115	2115	
Power Transmitted (watts)	24	15 (Early Transit) 150 (Backup)	24	150	-	150	10 kw	100 kw	10 kw	100 kw	-
Transmitting Antenna	3-ft dish	Dual Turnstile (Omni)	26.7 db Helix Array	Turnstile (Encaps)	-	Turnstile (Encaps)	85-ft dish	85-ft dish	85-ft dish	85-ft dish	-
Receiving Antenna	210 ft dish	210 ft dish	210 ft dish	210 ft dish	-	210 ft dish	3-ft dish	Dual Turnstile (Omni)	26.0 db Helix Array	Turnstile (Encaps)	-
Receiver Noise Figure	-	-	-	-	-	-	10 db	5 db	10 db	10 db	-
Receiving System Noise Temp. (^oK)	35	35	35	35	-	35	3200	1000	3200	1000	-
Probability of Bit Error	$1.4x10^{-3}$	$1.4x10^{-3}$	$1.4x10^{-3}$	$1.4x10^{-3}$	-	$1.4x10^{-3}$	10^{-5}	10^{-5}	10^{-5}	10^{-5}	-
Transmitting Ant. Pointing Error (Degrees)	1.0	-	2.5	-	-	75 deg	Neg	Neg	Neg	Neg	-
Receiving Ant. Pointing Error (Degrees)	Neg	Neg	Neg	Neg	-	Neg	1.0	-	2.5	-	-
Carrier APC Noise BW (2 B_{LO}) (CPS)	12	12	12	12	-	12	20	10	20	10	-
Sync APC Noise BW (2 B_{LO}) (CPS)	0.5	0.5	0.5	0.5	-	0.5	2.0	2.0	2.0	2.0	-
Phase Deviation (Degrees)	± 60	± 45	± 60	± 45	-	± 45	± 53	± 53	± 53	± 53	-

TABLE 4.1-6. SUMMARY OF LINK PARAMETERS (ALL-ORBITER)

Link No.	1	2	3	4	5	6	7	8	9	10	11
Purpose / Parameter	O-E TLM	O-E TLM (Backup)	-	-	-	-	E-O Command	E-O Command (Backup)	-	-	-
Frequency (mc)	2295	2295	-	-	-	-	2115	2115	-	-	-
Power Transmitted (watts)	57	57 (Early Transit) 100 (Backup)	-	-	-	-	10-kw	100-kw	-	-	-
Transmitting Antenna	9-ft dish	Dual Turnstile (Omni)	-	-	-	-	85-ft dish	85-ft dish	-	-	-
Receiving Antenna	210 ft dish	210 ft dish	-	-	-	-	9-ft dish	Dual Turnstile (Omni)	-	-	-
Receiver Noise Figure	-	-	-	-	-	-	10 db	5 db	-	-	-
Receiving System Noise Temp. (^oK)	35	35	-	-	-	-	3200	1000	-	-	-
Probability of Bit Error	1.4×10^{-3}	1.4×10^{-3}	-	-	-	-	10^{-5}	10^{-5}	-	-	-
Transmitting Ant. Pointing Error (Degrees)	1.0	-	-	-	-	-	Neg	Neg	-	-	-
Receiving Ant. Pointing Error (Degrees)	Neg	Neg	-	-	-	-	1.0	-	-	-	-
Carrier APC Noise BW ($2 B_{LO}$) (CPS)	12	12	-	-	-	-	20	10	-	-	-
Sync APC Noise BW ($2 B_{LO}$) (CPS)	0.5	0.5	-	-	-	-	2	2	-	-	-
Phase Deviation (Degrees)	± 60	± 45	-	-	-	-	± 53	± 53	-	-	-

TABLE 4.1-7. SUMMARY OF LINK PARAMETERS (ORBITER/LANDER)

Link No.	1	2	3	4	5	6	7	8	9	10	11
Purpose	O-E TLM	O-E TLM (Backup)	L-E TLM	L-E TLM (Backup)	L-O TLM (Surface)	L-O TLM (Descent)	E-O Command	E-O Command (Backup)	E-L Command	E-L Command (Backup)	O-L Command
Frequency (mc)	2295	2295	2295	2295	100	100	2115	2115	2115	2115	100
Power Transmitted (watts)	43	100	24	24	25	5	10 kw	100 kw	10 kw	100 kw	5
Transmitting Antenna	8-ft dish	Dual Turnstile (Omni)	26.7 db Helix Array	Turnstile (Encaps)	Turnstile (Hemi)	"Trans-mission Line"	85-ft dish	85-ft dish	85-ft dish	85-ft dish	10-db yagi
Receiving Antenna	210-ft dish	210-ft dish	210-ft dish	210-ft dish	10-db yagi	Turnstile (Hemi)	8-ft dish	Dual Turnstile (Omni)	26.0 db Helix Array	Turnstile (Encaps)	Turnstile (Hemi)
Receiver Noise Figure	-	-	-	-	4 db	4 db	10 db	5 db	10 db	10 db	4 db
Receiving System Noise Temp. (^{o}K)	35	35	35	35	755	1435	3200	1000	3200	3200	1435
Probability of Bit Error	1.4×10^{-3}	1.4×10^{-3}	1.4×10^{-3}	1.4×10^{-3}	10^{-3}	10^{-3}	10^{-5}	10^{-5}	10^{-5}	10^{-5}	10^{-5}
Transmitting Ant. Pointing Error (Degrees)	1.0	-	2.5	-	-	-	Neg	Neg	Neg	Neg	Variable with Altitude
Receiving Ant. Pointing Error (Degrees)	Neg	Neg	Neg	Neg	Variable with Altitude	-	1.0	-	2.5	-	-
Carrier APC Noise BW ($2 B_{LO}$) (CPS)	12	12	12	12	85	85	20	10	20	10	85
Sync APC Noise BW ($2 B_{LO}$) (CPS)	0.5	0.5	0.5	0.5	2.0	2.0	2.0	2.0	2.0	2.0	2.0
Phase Deviation (Degrees)	± 60	± 45	± 60	± 45	± 60	± 60	± 53	± 53	± 53	± 53	± 60

d. The values of $\frac{ST}{N/B}$ are derived as follows:

Telemetry links to earth ($P = 1.4 \times 10^{-3}$)

Theoretical = 6.5 db

Coding gain = 1.5 db

Detection losses = $\begin{cases} 1.0 \text{ db (strong carrier)} \\ 3.0 \text{ db (thresholding carrier)} \end{cases}$

Total = 6.0 db (strong carrier)
8.0 db (thresholding carrier)

Command links from earth ($P = 10^{-5}$)

Theoretical = 9.7 db

Detection loss = 8.8

Total = 18.5

The above total has been quoted by Motorola for their double-channel detector operating at a rate of one bit per second. This value is expected to be conservative for the single-channel detector, especially at higher bit rates.

Relay telemetry links ($P = 10^{-3}$)

Theoretical = 6.8 db

Detection losses = 4 db

Total = 10.8 db

Relay command links ($P = 10^{-5}$)

Theorectical = 9.7 db

Detection losses = 4 db

Total = 13.7 db

B. ANTENNA BREAKDOWN

Of prime concern in the Lander telemetry links was the reasonable assurance that antenna breakdown would not occur in the Martian atmosphere. Although a considerable amount of data has been accumulated in recent years concerning microwave discharges in air, none was available for a simulated Martian atmosphere. An experiment was therefore devised at GE-MSD to determine the extent that the breakdown characteristics might be modified by the Martian atmosphere. The principle constituents of the Martian atmosphere are argon, nitrogen, and carbon dioxide in unknown amounts. To obtain a conservative estimate, only argon was utilized in the simulated atmosphere. This represents a worst case as the addition of the other constituents mentioned can only raise the breakdown level, particularly in the high pressure region, because of the possibility of electronic attachment.

The experiment was conducted at 240 megacycles at pressures between approximately 0.1 and 4.0 millibars using a monopole over a ground plane. The monopole had a diameter of one-eighth inch and a length-to-wavelength ratio of 0.26. To check the validity of the data, the experiment was also performed in air and the results were compared with results available in the literature. Excellent correlation was obtained.

The conclusions made from the experiments were that the breakdown power levels in argon are roughly 50 percent lower than those in air for the same antenna. This also held true when the monopole was teflon capped. The minimum breakdown power level was at a pressure of approximately 0.4 millibars in both air and argon.

Correlating these results with known breakdown characteristics in air at other frequencies and with other antenna configurations, it was concluded that with reasonable care in design (assuming a minimum Martian surface pressure of approximately 11 millibars) the Lander antennas would not be subject to breakdown at the power levels prescribed by other constraints. These antennas and power levels are:

a. 100 mc turnstile on surface (25 watts)
b. 100 mc transmission-line antenna during descent (5 watts)
c. 2.3 kmc encapsulated turnstile during descent and on surface (150 watts)
d. 2.3 kmc helix array (12 helices) on surface (24 watts).

C. DATA STORAGE REQUIREMENTS

The magnetic tape recorders are characterized by their input data rates, output data rates, and data storage volume.

The highest input rate was selected on the basis of a compromise between low rates required for low recorder power and high reliability, and high rates required for short camera storage period. A lower input rate was selected for inputs from the buffer and Data Processing Unit.

The output rates were selected to include the nominal data transmission rate of the associated vehicle at worst-case encounter range. Rates higher and lower than the nominal rate account for changes in transmission capability due to variations of transmission range and subsystem performance.

Storage volume was selected on the basis of the desired playback period at the nominal transmission rate under the constraints of power, weight, and reliability.

The following results correspond to the requirements of the All-Orbiter and Landers of the Bus/Lander and Orbiter/Lander configurations. Although the results for the All Orbiter are also applied to the Orbiter of the Orbiter/Lander, the storage volume is inadequate for full-time transmission during the orbital period of the latter (23 hours). Since the weight constraint limits the number of recorders which can be used, further tradeoffs of storage capacity (including the addition of more tape in each recorder) and transmission capability should be made; however, this iteration was not completed for this study.

1. Orbiter

The nominal transmission rates for the All-Orbiter and Orbiter of the Orbiter/Lander are 12 and 6 kilobits per second, respectively. Output data rates selected were, therefore, 24, 12, 6, 3, and 1.5 kilobits per second.

The input rates selected were 48 and 12 kilobits per second. TV frame periods associated with the 48 kbps are 23 and 90 seconds for the vidicons and image orthicons, respectively.

Storage volume is 2×10^8 bits based on continuous transmission during a 4.3 hour orbit at 12 kilobits per second.

Two recorders are required in the record mode during the one-hour TV-mapping period to allow for time overlap of TV frames in the picture-taking sequence. A third recorder is required during this time for playback. This sets a requirement for three recorders for optimum performance; however, it should be noted that approximately 75 percent of the maximum data volume per orbit can be transmitted to Earth if only one of the three recorders is operating.

2. Lander

The nominal transmission rate for the Landers is 800 bits per second. Output data rates selected are therefore 3200, 1600, 800, and 400 bits per second.

The input rates selected were 12.8 kbps and 1.6 kbps resulting in a 32-second frame time for the TV. These input/output rates result in an overall speed change of 32 to 1 which can probably be accomplished by a single drive motor. Transmission rates lower than 400 bps can be implemented by reading the stored data into the plated-wire buffer storage at a high rate and then reading out of the buffer at the desired rate.

The storage volume selected is 10^7 bits. Although three times this amount is required for a 10-hour transmission period (line-of-sight with Earth) at 800 bits per second, little system performance is lost since data can be acquired from the panorama or microscope TV at any time. This relatively low storage requirement allows a compact, rugged, low-power recorder design. Two units are prescribed for added reliability.

D. ALTERNATE TECHNIQUES

1. Power Amplifier Configuration

The configuration shown in the block diagrams for obtaining two power levels through a single antenna utilize a low-power klystron, which can be powered continuously by the vehicle power supply, followed by a Raytheon Amplitron which can be actuated for a short period of time, along with the klystron, by secondary batteries during emergencies in which prime power is lost. This configuration takes advantage of the feed-through properties of an Amplitron. This eliminates the RF switching or double antenna required if two klystrons are used. Another technique, however, which might be used is that of operating a single klystron at the two required power levels by adding a second power supply. This could be applied in the orbiters where power levels of approximately 2 to 1 are required. It does not appear applicable, however, in the Landers where power ratios of 10 to 1 are required.

2. Early-Transit Antenna

The Earth sensor used to point the high-gain antennas during transit requires an Earth-Vehicle-Sun angle greater than 30 degrees. This constraint is satisfied only after the vehicle is farther than approximately 26 x 10^6 nautical miles from Earth. Although all communication functions can be accomplished through the omni-antennas to this range, the high-gain antenna can be used at close range, if required, by directing it by command from Earth. The pointing requirement of one degree can be relaxed during this period since greater pointing loss can be tolerated at close range.

3. Link Parameters

Performance better than that available with the listed parameters can be attained in various links as follows:

a. The APC noise bandwidths can be widened in the links presently having more than adequate margin in the given bandwidths. This will allow shorter acquisition periods.

b. Higher command rates can be used where required in links where the margin allows.

c. The 100-kw transmitter can be used in the Lander prime command mode to increase the rate given (2 bits per seconds).

d. The minimum carrier APC bandwidths can possibly be reduced in the backup telemetry links by programming the ground frequency reference to follow the expected doppler rate of change thereby reducing the static phase error in the loop.

4. Component Switching

The transmission subsystems contain identical or similar components which, as presently shown, are not switchable from one link or mode to another. If desired, however, switches could be incorporated to provide added redundancy. The extent of switching to be used in the final design must result from a thorough analysis, including that of the reliability of the switching functions and the radiated power reduction caused by insertion loss.

5. Lander Data Storage

If sterilizable tape recorders for the Landers are found to be beyond the state-of-the-art in the desired time period, a thin-film plated-wire storage unit could be used as a buffer between the TV cameras and the transmitters. Although system performance would be degraded because of the low volume of data which can be stored prior to transmission, the utilization of this technique rather than tape recorders would not invalidate the mission and instrumentation concepts recommended. Univac has estimated that a 10^6-bit buffer could be designed with a weight of 15 pounds, a volume of 450 cubic inches and a power requirement of 0.5 watt.

4.1.3 SUBSYSTEM DESIGNS

Each Communication Subsystem comprises a Deep Space Transmission Subsystem, Command and Computer Subsystem, and a Data Processing and Storage Subsystem. In addition, the Orbiter/Lander communications includes a Relay Transmission Subsystem. Because of the similarity from vehicle to vehicle, each of the above subsystem designs are described in individual sections with the differences noted for each vehicle. In subsequent sections, the functions, performance, and power, weight, and size estimates are given for the composite Communication Subsystem of each vehicle.

A. SUMMARY OF REQUIREMENTS

Table 4.1-8 summarizes the general functional requirements for the Communication Subsystem of each vehicle configuration.

B. COMMAND AND COMPUTER SUBSYSTEM

The Command and Computer Subsystems recommended for the Titan III-C Voyager vehicles are identical, or nearly identical, to those recommended in the Saturn 1-B Voyager study. Therefore, only the overall description will be given in this report.

1. Description

The Orbiter/Lander mission does not appear attractive for reasons other than communications; however, since the associated Orbiter Command and Computer Subsystem is required to perform functions in addition to those required in the other vehicles, the following description is for that subsystem. The All Orbiter subsystem differs in that none of the functions related to the relay link are required. Subsystems for the Landers differ primarily from those for the Orbiters in the number of commands they are to execute (1024 for Orbiter, 512 for Lander).

TABLE 4.1-8. SUMMARY OF COMMUNICATION SUBSYSTEM REQUIREMENTS

Configuration	All-Orbiter	Bus/Lander		Orbiter/Lander	
Requirement / Vehicle	Orbiter	Bus	Lander	Orbiter	Lander
Receive command data from Earth	X	X*	X*	X	X
Relay selected command data to Lander	-	-	-	X	-
Receive command data from Orbiter	-	-	-	-	X
Provide command data storage	X	-	X	X	X
Issue commands to all vehicle subsystems	X	-	X	X	X
Collect data from all vehicle science and eng. sensors	X	-	X	X	X
Receive data from Lander	-	-	-	X	-
Provide storage for all collected data	X	-	X	X	X
Transmit data to Earth	X	X*	X*	X	X
Transmit data to Orbiter	-	-	-	-	X
Provide doppler and angular tracking of vehicle from Earth	X	X*	X*	X	-
Provide ranging from Earth during early transit	X	X*	X*	X	-
Provide computer functions as required	X	-	X	X	X

*Bus provides antennas for Lander transmission subsystem during transit.

a. Functional

The Voyager Command and Computer Subsystem will be designed to perform the following functions:

1. Receive and verify command words, prepare the word format for use by the subsystem and determine if the command is to be stored or operated upon directly.
2. Store command words for later use by the vehicle and, in the case of the Orbiter Command and Computer Subsystem of the Orbiter/Lander configuration, to hold command words to be relayed to the Lander.
3. Generate and supply to the subsystems, timing pulses of required repetition rates and provide a present time clock for the timing and execution of stored commands.
4. Search the memory and retain the time tag and command for the next event to be executed.
5. Provide magnitude information to designated elements in the control system and provide decoded output with sufficient drive capability to operate command relays.
6. Compute a time increment, for certain data channels when selected, and add to the command's time tag before returning both to memory.
7. Provide power conversion for selected subsystem components.

b. Block Diagram

The elements and their interconnections are shown in Figure 4.1-4.

c. Command Word Format

The command word formats with which this system will be required to operate are as follows:

Type I Command Input; Requires two words

Word I

			8 bit prog.	20 bit time tag

to be used as command prefix

First three bits to indicate if command is to be stored or executed directly and if command is for relay to a Lander from the Orbiter at the Orbiter/Lander. Three bits can designate the Orbiter and up to two Landers; only one bit is required in the All Orbiter and each of the Lander Subsystems.

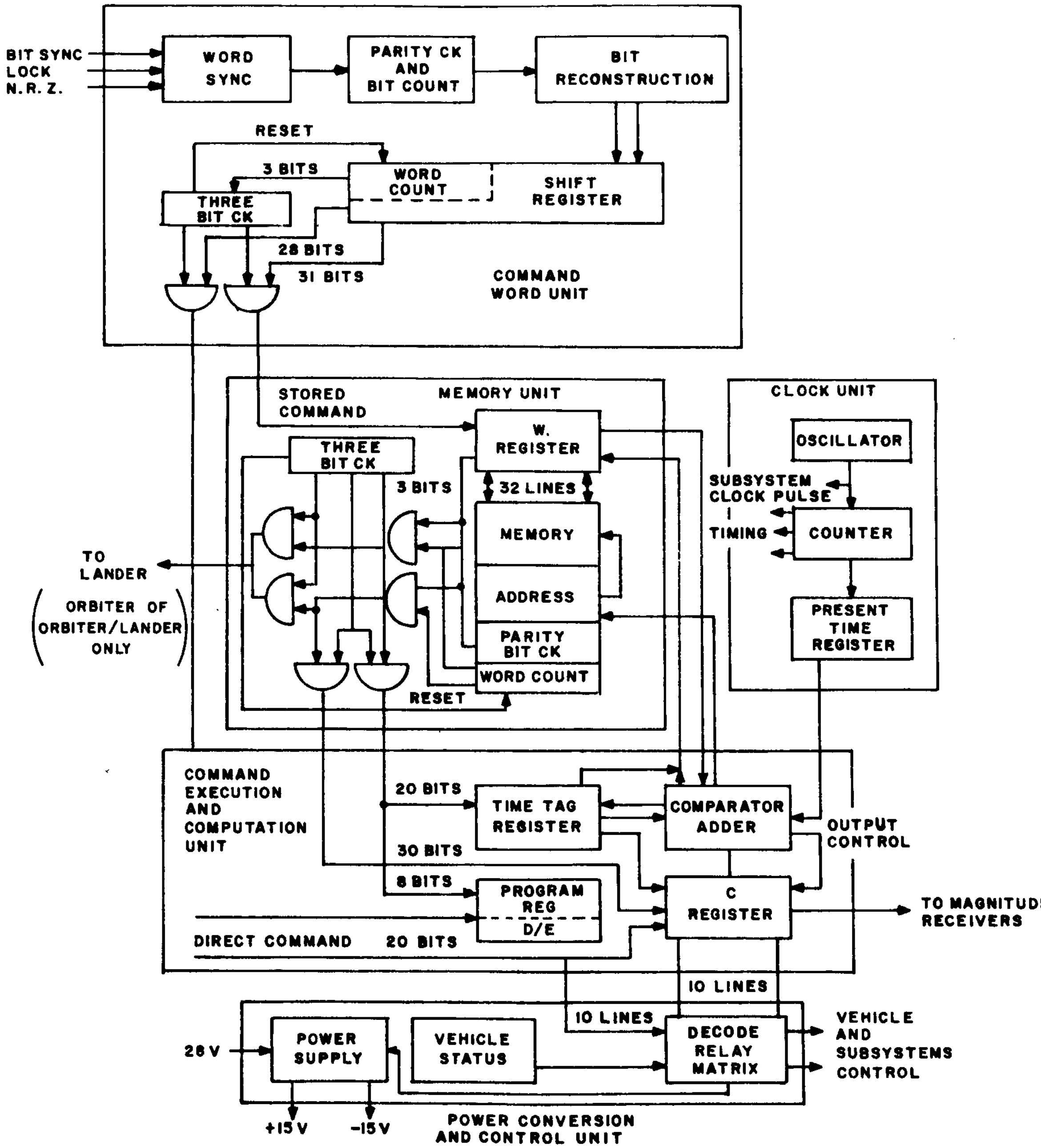


Figure 4.1-4. Block Diagram, Command and Computer Subsystem

Word II

Six 5-bit command suffix, or up to 30-bit magnitude word.

Type II Command Correction; Requires one word.

1			8 bit prog.	20 bit time tag

The code word indicates to the program encoder/decoder that the command stored with the same time tag is to be erased.

Type III Clear Commands stored; Requires 1 word.

1			8 bit prog.	20 bit time tag

The code word indicates that all commands with time tags greater than the time given are to be erased.

Type IV Computer Constant; Requires 1 word.

0	0	0	8 bit prog.	20 bit magnitude

Provides, through the program decoder/encoder, the address for storing the magnitude word.

Type V Immediate Commands; Requires 1 word.

1			8 bit prog.	No. 1	No. 2	No. 3	No. 4

Four, 5-bit command suffix words

Command prefix word

Indicated immediate action and along with the other two bits, indicates if it is for the Orbiter or for the Landers.

The first three bits, as indicated, are used to indicate the recipient of the following command. For the condition of one Orbiter and two Landers, the code assignment may be as follows:

000	Orbiter stored command
001	Lander No. 1 stored command
010	Lander No. 2 stored command
101	Lander No. 1 immediate command
110	Lander No. 2 immediate command
111	Orbiter immediate command

The next eight-bit program portion of the command word is used in conjunction with the Program Register Decoder/Encoder Unit. The eight-bit code may be partitioned into three groups as follows:

Group I:	From To	00000000 00011111	32 code words to represent first half of the command word.
Group II:	From To	00100000 01111111	96 to be reserved for use by the Command Subsystem; this is to include the computer operations.
Group III:	From To	10000000 11111111	128 code words to be used to select data blocks for acquisition.

The following twenty-bit time tag, at a four-second time increment, will allow for a 48-day unambiguous command period. Second-half, or suffix, command words are decoded, five bits at a time, and used in conjunction with the outputs generated by the eight-bit program word.

1. Modes of Operation

a. Command Acquisition

When a sub-carrier lock signal is received by the Command and Computer Subsystem, a search is initiated for a combination of incoming data in the form of 111000. This is the signal that the following data is to be interpreted as a command word. The Command and Computer Subsystem receives information from the Command Demodulator in a Manchester format at a command rate of ten bits per second. A command bit of data requires two Manchester bits of data, where certain combinations of bits in the Manchester code are recognized as errors. The system processes the Manchester data by bit counting, parity checking, and command-bit verification. It then converts the Manchester binary data to the conventional format and transfers the data to a shift register. Reject and accept signals are generated in the command acquisition phase.

With the command word in the shift register, the first three bits are checked and the appropriate output gate opened. If it is to be stored, the contents of the shift register are transferred to the "W" register in the Memory Unit. If the Command is to be operated on immediately, the eight-bit program word is sent to the Program Register Decoder/Encoder element and the twenty-bit suffix portion to the "C" register.

b. Command Storage

All command and control information is stored in the Memory Unit. Inputs are in serial form into the "W" registers and may be received from the following elements:

1. Shift register in the command word unit
2. Time tag register
3. Comparator/Adder
4. Data Storage Unit

The information is transferred into the memory storage cells and received back through the "W" register.

Since command words are stored in pairs, a first-second word control is exercised over the decoding of the output of the "W" register. On the first word the first three bits are checked for the destination of the command. If, in the case of the Orbiter, it is an Orbiter command, then the next eight bits are sent to the Program Register and the following twenty to the time tag register. The second word is sent to the "C" Register. If the command is intended for a Lander and transmission to the Lander is available as indicated by the Vehicle Status Elements, then the two words are shifted out for relay via the communication link.

Computer words are stored in a separate part of the memory and are selected by direct addressing. Command words are selected through a sequential search. The output of the "W" Register is checked for parity. This check insures that the information has been properly stored and retrieved from the memory. Only command words to be sent out of the Command Storage Unit will contain a parity check bit.

c. Subsystem Time Pulse Generation (Clock Unit)

The clock unit, as indicated in the block diagram, is composed of three elements - the Oscillator, the Counter, and the Present Time Register.

(1) Oscillator — The Oscillator will be crystal-controlled with a long-term stability of 50 parts per million. The frequency selected is 524.29 kc, since this frequency is needed in other subsystems, such as digital TV. It also provides the standard binary number system.

(2) Counter — The functional requirement for multiplication by powers of two is performed by the Counter. With an input pulse period of 1.91 microseconds, all binary values up to one pulse in four seconds are available. The 0.125 second pulse period will be used in timing data acquisition in the Data Processing and Storage Subsystem. The shorter pulse periods will be used in timing the duration of commands used in the guidance subsystem and in the communication equipment.

(3) Present Time Register — The four-second pulse period output of the Counter is accumulated in the Present Time Register. This provides a continuous indication of elapsed time for the comparison of command time tags and for their execution. After 48.5 days the register will restart from zero. If stored commands span the time period when the register reverts to zero time, then the command search program must take this into account.

d. Command Selection and Execution

After a command has been executed, the subsystem will start a routine search of the Memory Element. The objective will be to locate the next command by finding the lowest time tag that is greater than the time given by the Present Time Register. The double constraint, of (1) lowest of all stored values and (2) greater than the value of the present time, allows command time tags to span the period when the clock restarts from zero.

Only Type I command inputs, containing two words, are kept in storage. The command search starts by extracting the first command encountered in the memory and storing both of its words in the appropriate registers: Three bits to the Three-Bit Check Register; eight bits to the Program Register; twenty bits to the Time Tab Register; thirty bits to the "C" Register. The next command encountered is held in the "W" Register and its time tag compared with the one stored in the Time Tag Register. If it meets the conditions with respect to the Time Tag Register and the Present Time Register, then both its words are transferred out of the "W" Register, replacing the previous command which is then returned to the memory. If it does not meet the required conditions, it is transferred back to the memory, and the next command is similarly checked. Following this procedure, all of the stored commands are tested. After this search is concluded, the command to be executed next will be in the "C" Register, its program part in the Program Register, and the time of execution in the Time Tag Register.

With the search over, the comparator element determines when time for execution occurs. During this period, between search and execution, the only parts of the subsystem that are active are the Registers, the Comparator and Clock Unit. During this phase of the operation, the power expended is at a minimum, and increases only as a function of the command activity of the subsystem. At the time of execution, the program word in the Program Register Decoder/Encoder selects one out of 10 output lines. The second word in the "C" Register is decoded, five bits at a time, to select one out of another set of 10 lines. It is the combination of selected lines that, through the Decoder Relay Matrix, selects one out of 1024 output lines. After each selection, the word in the "C" Register is shifted, five bits at a time, so that each of the six parts may be decoded.

The vehicle status element is to be used to modify the operation of the Decoder Relay Matrix in accordance with the condition of the vehicle audit sensors. This allows for interlocking of commands with system performance. It will also allow recalling subsystem commands as may be required by the vehicle.

e. Computation

Data acquisition, as a function of time, forms the basis for an on-board computational requirement. The routine operations during the transit phase will be accomplished by selecting from among preset modes. Once in orbit about the target planet, however, it may be found necessary to change the mode and rate at which data is acquired so as to adapt to the conditions actually observed.

The computer capability is inherently available with the elements required to perform the command function. This includes all elements with the exception of the following:

1. Vehicle Status
2. Decode Relay Matrix
3. All elements in the Command Word Unit

The essential changes are the addition of ADD logic to the Comparator element and the addition of related transfer and control lines. The limited forms of computation required allow the functional organization to be most efficient for the intended programs. Specialized instructions and programs will be wrritten to take maximum advantage of the design.

The computational requirements are as follows:

1. Given selected sensor channels and assigned time increments, update the time tag of each selected channel by the assigned increment after the data has been obtained.
2. For orbital-sensitive channels, such as television pictures, radar mapping and infrared, the time increment for each succeeding operation will be computed by evaluating a polynomial with constants, related to the orbit obtained, to be supplied from earth. The time increment will be added to the time tag before it is returned to the memory.

The inclusion of the computer function in no way obstructs the Command Execution and Computation Unit to operate in either the stored pre-programmed mode or the direct command mode. It does, however, provide a degree of flexibility and adaptability that could only be achieved by expanding the storage capacity of the system and increasing the command load on the communication link.

C. DATA PROCESSING AND STORAGE SUBSYSTEM

1. Functional Requirements

The Data Processing and Storage Subsystems on the Voyager Landers and Orbiters are the gathering points for all data including diagnostic, scientific, and television information which is to be prepared for transmission to earth. In general, it is required to perform the following functions: It must sample selected groups of scientific and diagnostic data inputs from other subsystems, digitize those inputs which are analog voltages, organize the data into identified frames and route the resultant information into the storage unit, to the transmitter, or both, at the desired data rate. It must be capable of storing large quantities of digital television data in addition to the vehicle data. On command it must read out both TV and a relatively small quantity of vehicle data, pseudo noise and Manchester coded for transmission, to the transmitting equipment at the desired bit rate. It must be capable of generating error control code groups for all links transmitting directly to earth.

2. Description

The general block diagrams for the subsystem are shown in Figures 4.1-5 and 4.1-6 for the Orbiter and Lander. As in the previous section for the Command and Computer Subsystem, the functions described are those required for the Orbiter/Lander mission because of the additional requirements. Those for the All Orbiter and Bus/Lander are identical except that all functions related to the Relay Transmission Subsystem are omitted.

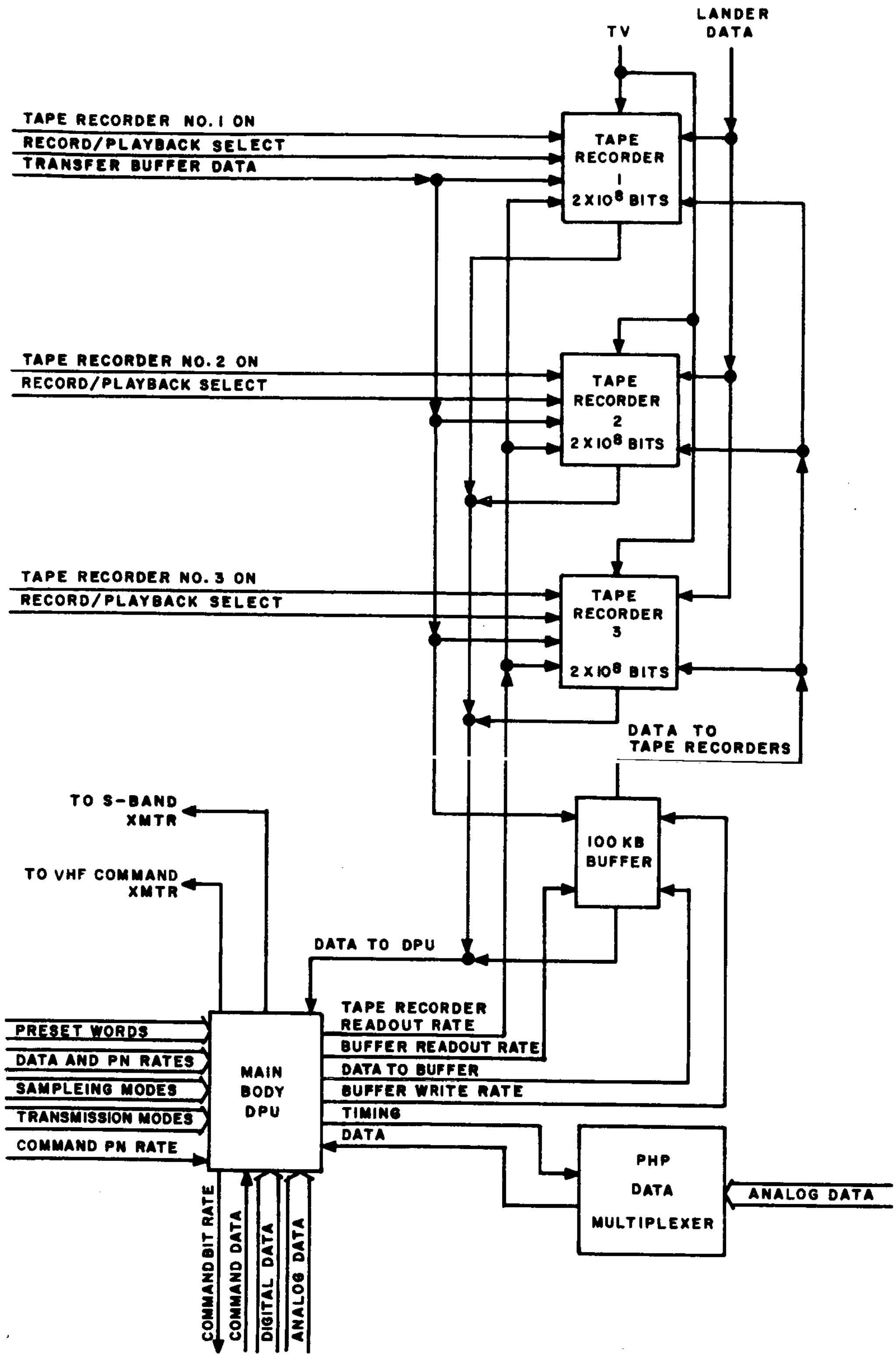


Figure 4.1-5. Data Processing and Storage Subsystem, Orbiter of Orbiter/Lander

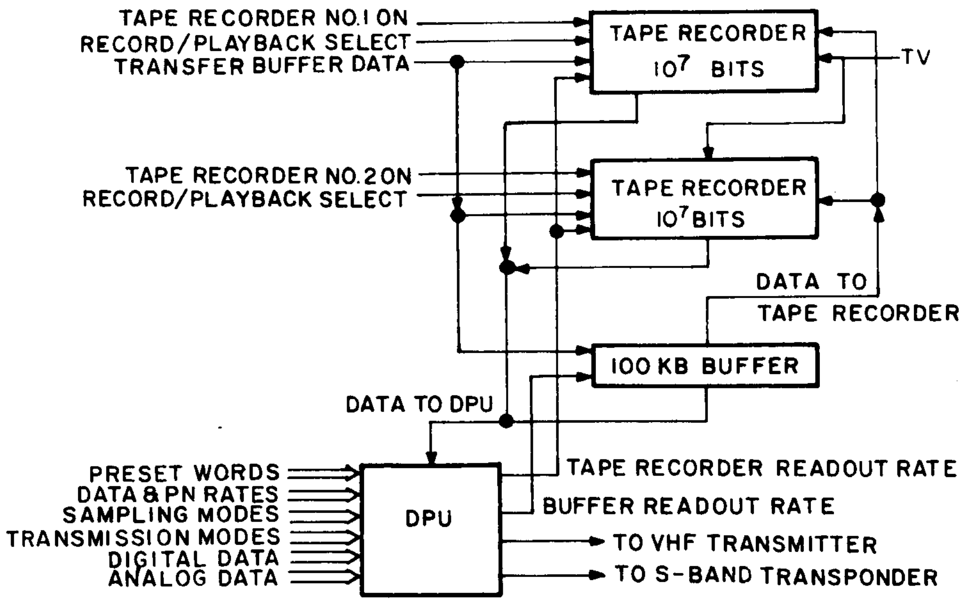


Figure 4.1-6. Data Processing and Storage Subsystem, Lander of Orbiter/Lander

Only a general description of the subsystem functions and implementation are given here; detailed block diagrams and descriptions are included in the previous Voyager Design Study.

a. Data Processing Unit

The data processing unit will perform the following functions:

1. It will accept:

   a. High-level analog inputs, 0-5 vdc

   b. Low-level analog inputs 0-50 mv dc

   c. Digital data

   d. Event occurrence pulses.

   The analog voltages will be converted in the data multiplexer to a six-bit digital equivalent, thereby providing a measurement resolution of 1.6%.

2. It will assemble the data into frames containing 1024 words (512 words for Lander), uniquely locating the data from each of the input lines within the frame.

3. It will generate any one of many programmed sampling modes upon receipt of a mode-select pulse and preset word from the Command and Computer Subsystem. A mode is defined as a particular combination of frame format and bit rate. The basic frame consists of 1024 words, each from a different

input line. The format may be changed by sampling sub-groups of the total 1024 more often than once per frame, e.g., 512 inputs sampled twice per frame, 32 inputs sample 32 times per frame, etc. The ability to create a variety of modes during a mission permits most efficient use of the Data Processor through format control, and flexibility of sampling rate through bit-rate control. This capability permits utilization of the same Data Processer on all Voyager missions.

4. It will provide either recycling sampling operation or one-cycle-per-command pulse operation, depending on the mode selected.

5. It will insert proper identification and synchronization data into each frame of 1024 words. These include:

   a. Barker code for frame synchronization of ground equipment (Other suitable codes are available; the Barker code is herein used as an example).

   b. Time label having a four-second resolution.

   c. Mode identification for locating specific data in a given frame.

   d. Data origin point identification.

   e. Four sub-frame identification words spaced 256 words apart for ease of data location.

6. It will route the output data train to the Data Storage Unit, the transmitter, or both, depending on the mode.

7. It will provide digital output buffering for simultaneous storage and real-time transmission at different bit rates.

8. It will generate any one of several transmission modes upon receipt of a transmission mode select pulse from the Command and Computer Subsystem. A transmission mode is defined as a particular combination of data source (tape recorder, DPU, buffer), data rate, and encoding scheme.

9. It will provide a Manchester-coded 511-bit PN code for encoding all data for transmission to Earth.

10. It will provide 28 bits of error control code for each 45 data bits if requested by mode select pulse. Then it will be PN and Manchester coded for transmission.

11. It will apply a Manchester-coded 31-bit PN code to command data being transmitted to the Landers from the Orbiter. It will apply a Manchester coded three-bit PN code to data being transmitted from the Landers to the Orbiter.

b. Data Storage Unit

Thermoplastic recorders (TPR's), as recommended in the previous GE Voyager Study, are not expected to be within the state-of-the-art in the required time period; therefore, they have been replaced in the presently recommended systems by magnetic tape recorders. The subsystem flexibility is reduced because of the limited number of record/playback rates practical in a tape recorder; however, the data transmission capability is not reduced significantly.

Reduction of transmitted data results from blank spaces in the tape produced during the stop/start periods between data frames during the record mode. These blanks can be eliminated to a certain extent by backing up after each stop so that the tape will attain synchronous speed and phase lock as the record head passes over the end of the previously recorded data block during the recording of the subsequent block. This technique, however, does not appear necessary with the data block and stop/start durations expected. TV data blocks dominate the recorded information. Since the shortest of these blocks are in the order of 20 seconds and the anticipated stop/start duration is two seconds total, the maximum data loss would be ten percent. A lesser value can be expected during normal operation, however, since many blocks will be recorded contiguously without stopping and since the longest single TV blocks are approximately 90-seconds long.

The general tape recorder characteristics for the Orbiters and Landers are as listed below:

ORBITER

Storage Capacity	- 2×10^8 bits
Input Data Rates	- 48 and 12 kilobits per second
Output Data Rates	- 24, 12, 6, 3, and 1.5 kilobits per second
Stop/Start Duration	- 1 second each
Data Format	- NRZ
Output Bit-Rate Variation	- Phase locked with spacecraft clock

LANDER

Storage Capacity	- 10^7 bits
Input Data Rates	- 12.8 and 1.6 kilobits per second
Output Data Rates	- 3.2, 1.6, 0.8, and 0.4 kilobits per second
Stop/Start Duration	- 1 second each
Data Format	- NRZ
Output Bit-Rate Variation	- Phase locked with spacecraft clock.

Investigations by RCA and Raymond Engineering Laboratory, Inc., indicated that the Orbiter requirements could be satisfied by a recorder having the following power, weight, and size requirements (including electronics):

Size	9 in. x 10.5 in. x 8 in. (760 in.^{3})	11.5 in. x 11.5 in. x 6 in. (800 in.^{3})
Weight	15 pounds	less than 20 pounds
Power	15 watts (max. record rate) 5 watts (min. play-back rate)	16 watts record 14 watts play-back

RCA recommended using the same recorder for the Lander to eliminate an additional development program. Raymond Engineering Laboratory recommended a new development to obtain the reduced size, weight, and power requirements given below:

Size	7.5 in. x 8.5 in. x 5 in. (320 in.^{3})
Weight	8 pounds
Power	5 watts (max. record speed) 1 watt (min. play-back speed)

D. DEEP SPACE TRANSMISSION SUBSYSTEM

1. Functional Requirements

The Deep Space Transmission Subsystems of both Orbiters and the Bus/Lander are required to:

1. Accept a serial digital waveform containing both data and bit-sync information from the Data Processing and Storage Subsystem.
2. Phase-modulate an RF carrier with the composite signal and transmit it to Earth.
3. Receive command data on a phase-modulated RF carrier from Earth.
4. Demodulate the command signals.
5. Provide the demodulated data along with bit-sync pulses and a bit-sync lock signal to the Command and Computer Subsystem.
6. Accept mode change commands from the Command and Computer Subsystem.
7. Coherently translate the frequency and phase of the received RF carrier by a ratio of 240/221 to obtain the transmitted frequency.
8. Provide an auxiliary stable frequency source which controls the transmitted frequency when no signals are being received from Earth.
9. Receive and transmit a ranging code.

All of the above functions except the last three are also required of the Lander of the Orbiter/Lander configuration; however, functions (7) and (8) will be included for possible doppler experiments with the Lander. Turn-around ranging cannot be accomplished at Mars distances; therefore, function (9) is omitted for this Lander.

2. Description

The Deep Space Transmission Subsystem of each vehicle comprises all S-band RF components and the associated command detectors. The block diagrams and descriptions in subsequent sections define the components and their general functions. Detailed block diagrams of the transponders, command detectors, and high-voltage power supplies are included in the previous GE Voyager Design Study Report. Additional outputs from the data detectors not shown in the block diagrams of the report are the bit-sync and bit-sync-lock signals provided to the Command and Computer Subsystem. The lock signal prevents command data from reaching the latter subsytem when the data is not being detected properly.

All transponders have 10-db noise figures; however, those in the Orbiter backup links are preceded by tunnel-diode preamplifiers having a noise figure of 5 db. No preamplifiers are utilized in the Landers because they cannot meet the sterilization requirements.

The carrier phase-lock-loop bandwidth ($2\ B_{LO}$) of the transponders in the prime command links is 20 cps (this could be increased in most prime links for faster acquisition since the margins are presently more than adequate). In the backup links, the loop bandwidths have been reduced to 10 cps to achieve greater command range.

Details of all antenna designs are given in the previous Voyager Design Study Report; however, the S-Band turnstile antenna described there has been encapsulated for the descent and surface phases of the Lander of the Bus/Lander configuration as shown in Figure 4.1-7.

E. RELAY TRANSMISSION SUBSYSTEM

The Relay Transmission Subsystem comprises all VHF components and the associated data detectors in the Orbiter/Lander configuration. The block diagrams and descriptions in subsequent sections define the components and the general functions. Detailed block diagrams of the transmitters, receivers, and data detectors are included in the previous GE Voyager Design Study Report.

F. BUS/LANDER

1. Functional Description

Figure 4.1-8 shows the functional block diagram of the Bus/Lander communication subsystem. This over-all subsystem comprises the Deep Space Transmission Subsystem, the Command and Computer Subsystem, and the Data Processing Subsystem.

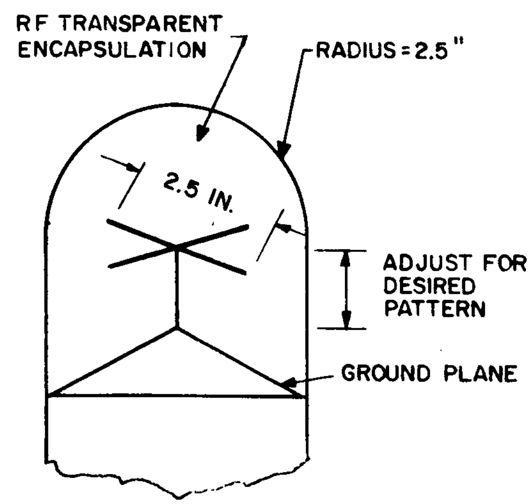


Figure 4.1-7. Descent- and Surface-Phase S-Band Antenna

The Deep Space Transmission Subsystem provides for transmission of all data from the spacecraft to Earth, reception of commands from Earth and cooperates with the DSIF in the tracking (doppler, angle, and turn-around ranging) of the spacecraft from Earth. All equipment is located in the Lander except for a high-gain and a low-gain antenna on the Bus. These are utilized until Lander separation, at which time they are disconnected and the Lander antennas are switched on for the remainder of the mission.

The low-gain Bus antenna comprises two turnstiles located on opposite sides of the vehicle. It gives nearly omnidirectional coverage except in the meridial plane between the two radiating elements and is used during early transit and as a back-up for the normal mode after early transit. The high-gain Bus antenna is a three-foot dish which is used in the normal mode after early transit. Although it provides transmission of scientific and engineering data during this phase, the relatively high data rate (400 bits per second) that it allows at encounter is used primarily for transmission of TV approach guidance date ($\approx$45 minutes per frame).

The 24-watt klystrons associated with the high-gain antenna cannot be powered continuously by the Lander RTG Unit, but rather are to be used eight hours per day

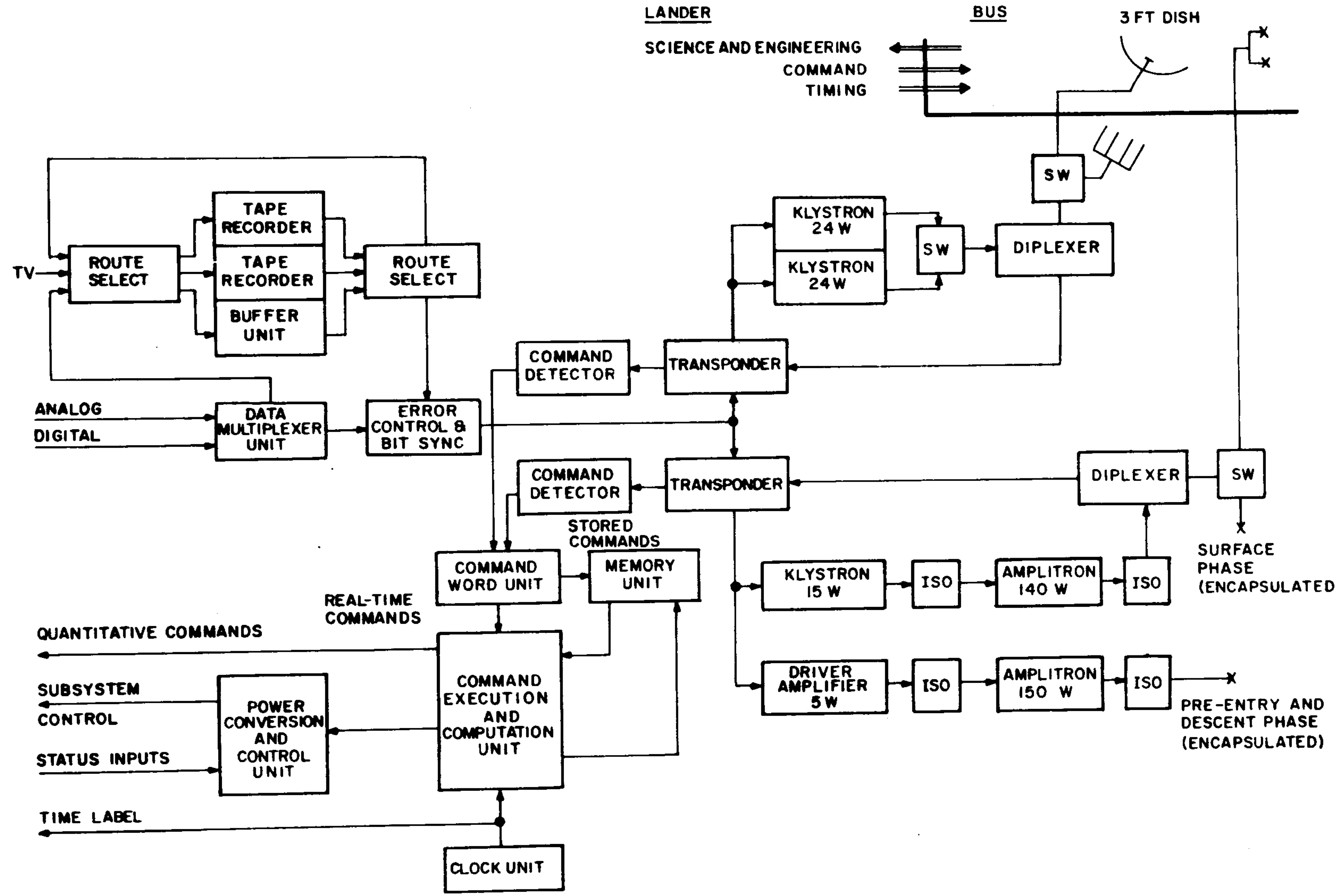


Figure 4.1-8. Bus/Lander Communications Subsystem Block Diagram

when on the planet surface using secondary batteries to allow the peak load. They are therefore also limited to this duty cycle during transit. Since continuous communication is required for tracking during portions of the early-transit phase, a 15-watt klystron, which can be powered continuously, is used in conjunction with the low-gain antenna. To allow the use of the same low-gain antenna at longer range as back-up, a 140-watt Amplitron is included in the amplifier chain. At short range the Amplitron is inactive and essentially acts as a waveguide. At long range it is powered by a secondary battery and is driven by the klystron.

After separation from the Bus and prior to surface impact, the Lander transmits through a separate 150-watt amplifier chain and antenna. Each transmission period is limited to a few minutes by the secondary power source; however, this time period ($\approx$ 10 minutes) is much greater than that required for the atmosphere descent phase. The antenna used for this phase is an encapsulated turnstile giving at least unity gain over a 150-degree angle. The Lander attitude and trajectory is constrained such that the Earth is always included in this portion of the pattern.

Antenna encapsulation precludes breakdown in the Martian atmosphere at the high radiated power level (approximately 100 watts including losses prior to the antenna). Approximately four bits per second can be transmitted through this link.

After Lander impact, a steerable helix array giving 26.7-db gain is erected for the prime link using the 24-watt amplifiers. A data rate of 800 bits per second is provided by this link at encounter. A second encapsulated turnstile antenna is used in the back-up link.

All transmitted data is digital and is combined with a pseudo-noise (PN) sequence on a square-wave subcarrier prior to transmission. This composite signal is used at the receiver to derive bit sync. In addition, it moves the sidebands of the transmitted signal away from the RF carrier so that an uncluttered carrier will be available for tracking and synchronous detection.

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

The command word unit accepts digital data and associated sync pulses from the command detector when a lock signal is received; otherwise, it will not accept or act on any data. The command word unit interprets the word-start symbols, determines its destination, verifies the validity of the received data and, if accepted, delivers real-time data to the Command Execution and Computation Unit and stored data to the Memory Unit.

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

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

The Data Processing and Storage Subsystem has four different functions:

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

The first function applies only to the narrow-band data sensors as used in most cases for both science and engineering data. Wide-band data such as TV is encoded by an A/D encoder within the TV subsystem and separate from that used for the narrow-band data. Multiplexing of TV data with narrow-band data is directed by the Command

Subsystem. The data format and rate are also determined by the Command Subsystem. The format determines which sensors are sampled in a particular frame. For instance, during maneuvers only selected diagnostic sensors will be sampled, while during orbit most of the data collected will be scientific. The data collection rate will be commanded from Earth, based on the anticipated rate of change of sensor outputs and will be constrained by the rate at which data can be sent to Earth over an available time period. The narrow-band data can be either stored or transmitted directly.

The storage devices utilized are a 100-kilobit plated-wire storage unit and two 10^7-bit magnetic tape recorders. The buffer storage unit is used for the storage of low-rate data and as a buffer between the tape recorders and the Error Correction Encoder. The encoder requires that a burst of 45 bits be read in at the transmitted "digit" rate and that no data be read in during the subsequent period in which 28 check bits are added (as defined here, the "digit" rate is 73/45 times the effective bit rate). Data is therefore accepted continuously by the buffer from the tape recorders at the effective transmitted bit rate and supplied to the encoder in bursts at the "digit" rate.

Each 10^7-bit magnetic tape recorder can record data at two rates and reproduce data at four rates. The high input rate is used for TV data and the low rate for data from the data processor or buffer. The four output rates include the nominal transmission rate and higher and lower rates compatible with possible range and subsystem parameter variations. All output rates are phase locked with the spacecraft clock.

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

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

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

2. Performance Characteristics

The Bus/Lander Communications Subsystem performance is characterized by the data rate capability of each link and the maximum transmission range possible with each link.

Figure 4.1-9 shows the data rate as a function of range for the Lander prime telemetry link as determined in Section 4.1.2(A) and including an 8-db margin. All other selected rates are given in Section 4.1.1(C).

The maximum range at which each of these links can operate is determined by the threshold constraints in the data and carrier channels. Since the data rate can be selected for a link such that threshold is not reached in the data channel before it is reached in the carrier channel, only the range at which carrier channel threshold occurs is considered here. This range is shown for each link in Table 4.1-9. under four conditions. These conditions are:

1. Reception with a 210-foot dish with an 8-db margin in the link
2. Reception with a 210-foot dish with no margin in the link
3. Reception with an 85-foot dish with an 8-db margin
4. Reception with an 85-foot dish with no margin.

The first conditions gives the worst case design range for each link while the range determined under the second condition indicates the maximum range which is possible if the parameters are at their nominal values. The latter is listed for all links but should be used only to indicate possible performance for an emergency or backup link. The third and fourth conditions are utilized to indicate the tracking range utilizing an 85-foot dish. A gain of 51.8 db was used for the 85-foot dish to determine each range.

Similarly, the range at which carrier threshold occurs is shown in Table 4.1-9 for each of the command links. The four conditions used in this case were:

1. Transmission of 10 kilowatts with 8-db margin in link
2. Transmission of 10 kilowatts with no margin in link
3. Transmission of 100 kilowatts with 8-db margin in link
4. Transmission of 100 kilowatts with no margin in link

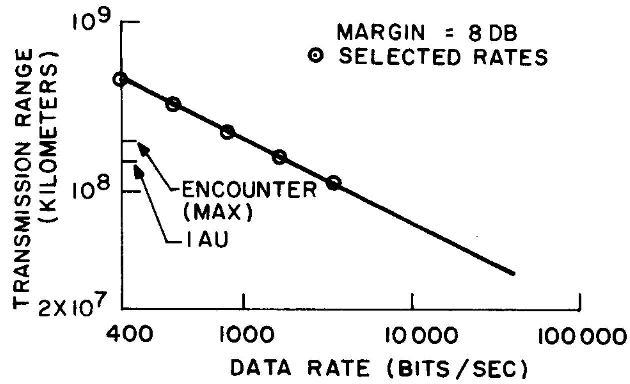


Figure 4.1-9. Lander Transmission Capability (Bus/Lander)

The first condition indicates the design range while the second condition indicates the maximum possible range if all parameter values are nominal. The latter indicates possible performances in an emergency mode if the 100-kw transmitter is not available. Conditions three and four indicate the design range and maximum range, respectively, for backup modes when the 100-kw transmitter is used.

The range at which threshold is reached in the data channel of each command link is not shown; however, the data rate has been selected in each link such that either the carrier and data channel thresholds are reached simultaneously (rate = 0.5 bits/sec) or the data channel threshold cannot be reached at the maximum earth-planet range under design conditions (8-db margin and 10 kilowatts transmitted).

3. Power, Weight, and Size

The component lists and estimates of power, weight, and size are given in Table 4.1-10 for the Lander and Bus Communication subsystems. It does not include the

TABLE 4.1-9. CARRIER-LOCK RANGE IN MILLIONS OF KILOMETERS (BUS/LANDER)

Link No.		1	2a	2b	3	4	6	7	8	9	10
Purpose		B-E TLM (Prime)	B-E TLM (Early Transit)	B-E TLM (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)
210 ft	8-db Margin	930	66	205	1100	205	205				
	No Margin	2200	165	515	2780	515	515				
85 ft	8-db Margin	320	244	72	380	72	72				
	No Margin	810	590	180	960	180	180				
10 KW	8-db Margin							590	100	720	57
	No Margin							1480	257	1800	145
100 KW	8-db Margin							1850	320	2300	180
	No Margin							4700	810	5700	450

TABLE 4.1-10. POWER, WEIGHT, AND SIZE ESTIMATES (BUS/LANDER)

Component	No.	Unit Size (in.)	Unit Weight (lb)	Unit Power (w)
		Lander		
Deep Space Transmission Subsystem				
1. Diplexer	2	6 x 3.25 x 2	1.0	-
2. Helix Array Antenna	1	33 x 33 x 9	10.0	-
3. Encapsulated Turnstile Antenna	2	5 D x 7.5	5.0	-
4. Transponder	2	184 $in.^3$	5.4	2.0
5. Power Amplifier (24W)	2	3.5 D x 5.5	3.0	96.0*
6. Power Supply (24W Amp)	2	4 x 4 x 6	6.0	120.0
7. Power Amplifier (15W)	1	3.5 D x 5.5	3.0	60.0*
8. Power Supply (15W Amp)	1	4 x 4 x 6	6.0	75.0
9. Power Amplifier (140W)	1	2.75 D x 4	4.0	284.0*
10. Power Supply (140W Amp)	1	5 x 5 x 6	8.0	355.0
11. Power Amplifier (150W)	1	2.75 D x 4	4.0	304.0*
12. Power Supply (150W Amp)	1	5 x 5 x 6	8.0	380.0
13. Driver Amplifier and Power Supply (5W)	1	4 x 4 x 6	4.0	25.0
14. Command Detector	2	4 x 4 x 5	3.0	1.75
15. RF Switch	3	2 x 2 x 2	1.0	-
16. Isolator and Load	4	10 $in.^3$	0.75	-
Data Processing and Storage Subsystem				
17. Data Processing Unit	1	5 x 5 x 10	16.0	3.5
18. Buffer Unit	1	140 $in.^3$	4.0	0.5
19. Tape Recorder	2	5 x 7.5 x 8.5	8.0	5.0 max.
Command and Computer Subsystem				
20. Command & Computer Equip.	1	4 x 5 x 10	14.0	1.8
21. Power Conversion & Control	1	3.5 x 5 x 11	7.0	10.0
		Bus		
Deep Space Transmission Subsystem				
1. High-Gain Antenna	1	3-ft dia.	7.0	-
2. Omni-Antenna	2	2.5 x 2.5 x 1.3	2.0	-

*Included in value for associated Power Supply.

cabling, harnessing, and payload compartment package structure listed in the vehicle weight section. See Sections 3.2.2(H) and 3.2.3(E).

G. ALL ORBITER

1. Functional Description

Figure 4.1-10 shows the functional block diagram of the All-Orbiter Communication Subsystem. It comprises the same subsystems as the Lander described in the previous section. Here, the Deep Space Transmission Subsystem radiates 57 watts through a nine-foot dish in the prime mode, and the same power through an "omni" during early transit. A 45-watt Amplitron is actuated in the latter link in the back-up mode resulting in a total radiated power of approximately 100 watts. Bit rates of 12,000 bits per second and 4 bits per second can be transmitted at encounter range in the prime and back-up modes, respectively.

Functionally, the Command and Computer Subsystem is identical to that described previously for the Lander except that commands are also transferred to the Planetary Horizontal package. To minimize the number of lines to the PHP, a separate decoder and power conversion and control unit are utilized. Only the data and control lines are therefore required. Also, for the Orbiter, the computer can be used to program the picture-taking sequence once the Orbiter orbital parameters are determined at Earth and the proper coefficients are transmitted for storage in the memory unit.

The Data Processing and Storage Subsystem is also functionally the same as that described previously; however, it includes an additional multiplexer on the PHP. Also, three magnetic tape recorders having capacities of 2×10^8 bits each are used for high-volume storage. They have two record rates, five playback rates and read out synchronously with the spacecraft clock in the same manner as described previously.

2. Performance Characteristics

Figure 4.1-11 shows the data rate of the prime telemetry link as a function of transmission range for the communication subsystem of the All-Orbiter as determined in Section 4.1.2(A). The data rates selected for all links are given in Section 4.1.1(C). Table 4.1-11 gives the maximum carrier-lock range for each link under the conditions stated in Section 4.1.3(F)(2).

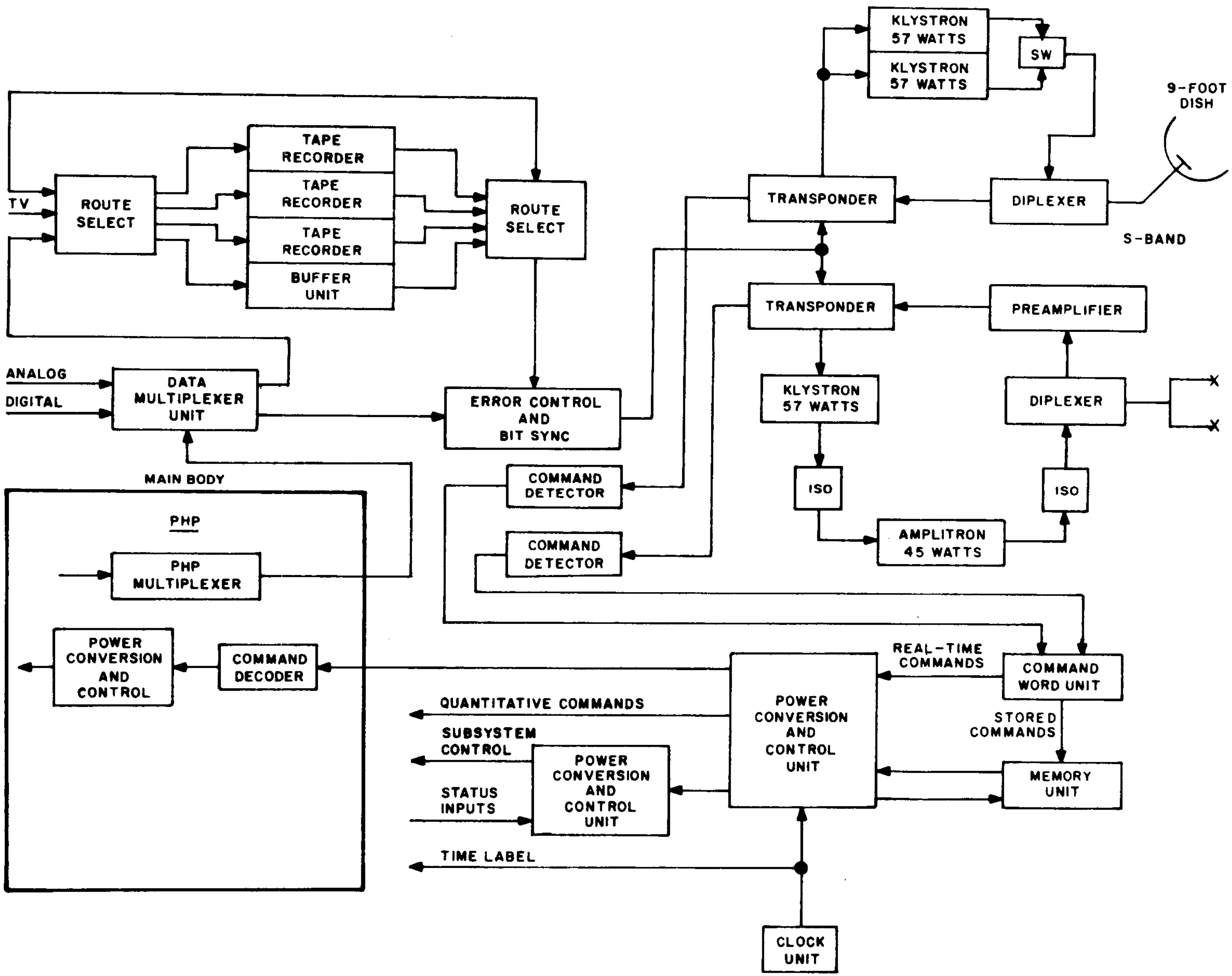


Figure 4.1-10. All-Orbiter Communication Subsystem Block Diagram

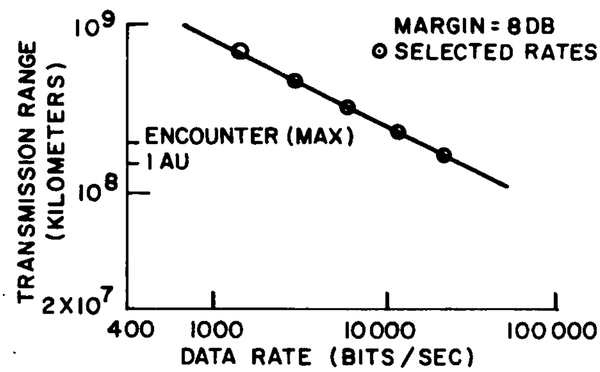


Figure 4.1-11. Orbiter Transmission Capability (All-Orbiter)

3. Power, Weight, and Size

The All-Orbiter Communication subsystem component list and estimates of power, weight and size are given in Table 4.1-12. It does not include the cabling, harnessing, and payload compartment package structure listed in the vehicle weight section.

H. ORBITER/LANDER

1. Functional Description

a. Orbiter

The functional block diagram for the Orbiter Communication Subsystem is shown in Figure 4.1-12. It is identical to the All-Orbiter described previously except that the power radiated and antenna diameter for the prime mode are reduced to 43 watts and eight feet, respectively, and a Relay Transmission Subsystem has been added.

TABLE 4.1-11. CARRIER-LOCK RANGE IN MILLIONS OF KILOMETERS (ALL ORBITER)

Link No.		1	2a	2b	3	4	6	7	8	9	10
Purpose		O-E TLM (Prime)	O-E TLM (Early Transit)	O-E TLM (Backup)				E-O Command (Prime)	E-O Command (Backup)		
210 ft Dish	8-db Margin	4000	128	168							
	No Margin	10000	320	420							
85 ft Dish	8-db Margin	1380	45	58							
	No Margin	3400	110	148							
10 KW	8-db Margin							650	100		
	No Margin							1600	257		
100 KW	8-db Margin							2040	320		
	No Margin							5100	810		

TABLE 4.1-12. POWER, WEIGHT, AND SIZE ESTIMATES (ALL ORBITER)

Component	No.	Unit Size (in.)	Unit Weight (lb)	Unit Power (w)
Deep Space Transmission Subsystem				
1. Preamplifier	1	2 x 2 x 4	2.0	1.0
2. High-Gain Antenna	1	9-ft dia.	28.0	-
3. Omni	2	2.5 x 2.5 x 1.3	2.0	-
4. Diplexer	2	6 x 3.25 x 2	1.0	-
5. Transponder	2	184 $in.^3$	5.4	2.0
6. Power Amplifier (57W)	3	3.5 D x 5.5	3.0	190.0*
7. Power Supply (57W Amp)	3	4 x 4 x 6	6.0	238.0
8. Power Amplifier (45W)	1	2.75 D x 3.0	2.5	94.0*
9. Power Supply (45W Amp)	2	4 x 4 x 6	4.5	117.0
10. Command Detector	2	4 x 4 x 5	3.0	1.75
11. RF Switch	1	2 x 2 x 2	1.0	-
12. Isolator and Load	2	10 $in.^3$	0.75	-
Data Processing and Storage Subsystem				
13. Data Processing Unit	1	3 x 6 x 10	12.25	2.1
14. Multiplexer (PHP)	1	2 x 7 x 10	10.0	1.0
15. Tape Recorder	3	8 x 9 x 10.5	15.0	15.0 max.
16. Buffer Unit	1	104 $in.^3$	4.0	0.5
Command and Computer Subsystem				
17. Command & Computer Equip. (Main Body)	1	3.5 x 10 x 10	20.0	6.4
18. Power Conversion & Control (Main Body)	1	3 x 10 x 10	12.0	5.0
19. Command Decoder (PHP)	1	2.5 x 2.5 x 10	4.0	0.3
20. Power Conversion & Control (PHP)	1	2 x 2.5 x 11	2.0	10.0

*Included in value for associated power supply.

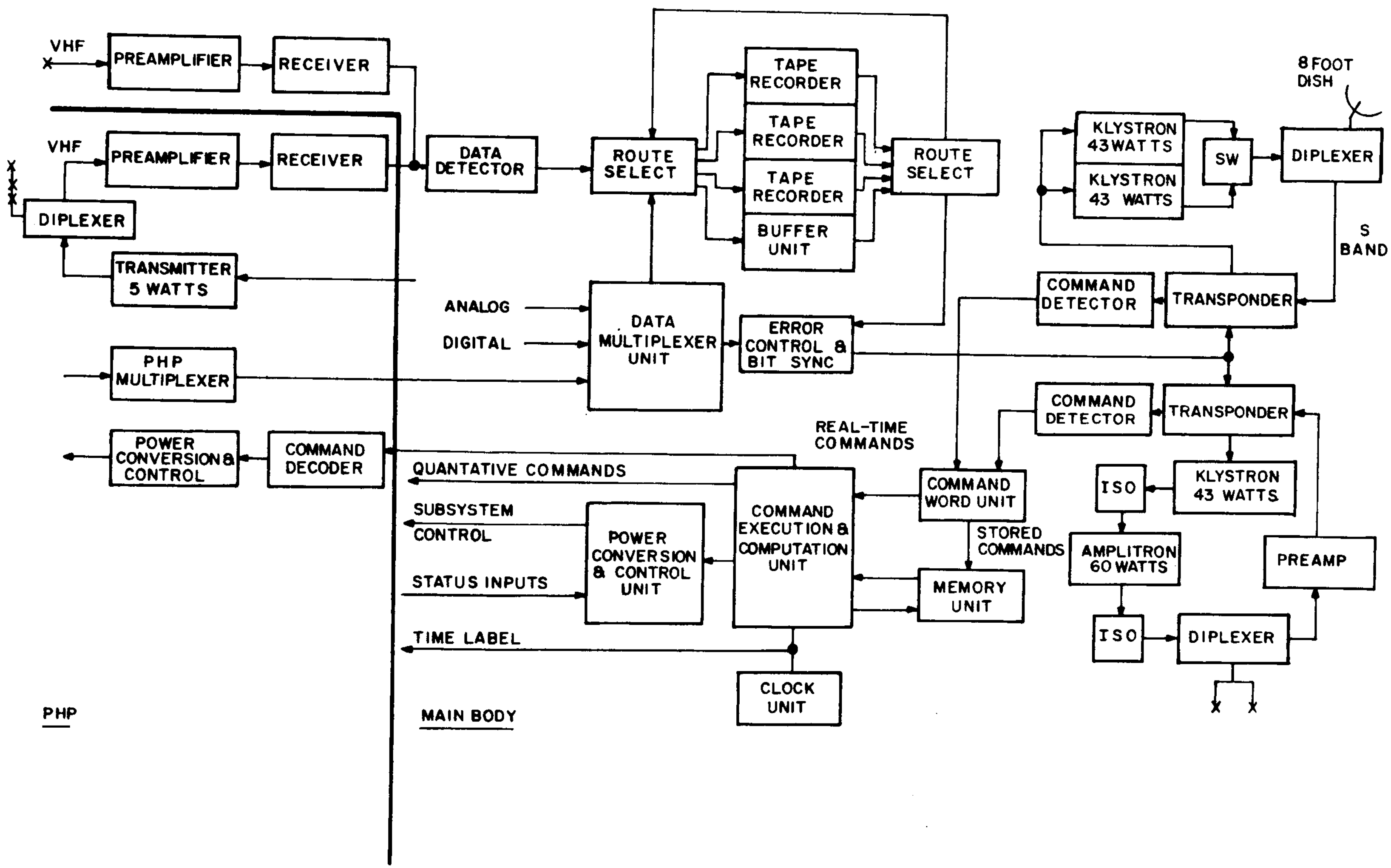


Figure 4.1-12. Orbiter Communication Subsystem Block Diagram (Orbiter/Lander Configuration)

The latter subsystem is divided between the Main Body and the PHP. The detector in the Main Body is used to detect all received data, since the two receivers are not utilized simultaneously. Both receivers have four-db noise figures. The receiver in the Main Body is fed by a turnstile antenna. It is used only during the post-separation and descent phases of the Landers. The receiver in the PHP cannot be used at that time, since the PHP is stowed until after retro firing for orbit insertion, which is not accomplished until the Lander has completed or nearly completed its descent phase. The turnstile antenna is located on the Main Body so as to give complete coverage of the planet when the Orbiter is in the retrofire attitude.

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

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

The bit-sync signal is used not only for detection, but is also recorded on the timing track of the tape recorder for synchronous playback to Earth. It is also possible to bypass the tape recorders, using the bit-sync signal to drive the Earth-link bit-sync and error correction encoder for direct transmission to Earth. The relay data rates selected for the surface phase are 12,000, 6,000, 3,000, and 1,500 bits per second, matching those of the Orbiter Earth link.

The five-watt command transmitter in the PHP is modulated by commands sent from the Command and Computer Subsystem. The Command and Computer Subsystem also initiates and controls the lock procedure which ensures both carrier and bit-sync lock in the relay links. Carrier-lock signals from the PHP receiver and sync-lock signals from the data detector are sent to the Command and Computer Subsystem as a part of this procedure.

b. Lander

The functional block diagram for the Lander Communication Subsystem is shown in Figure 4.1-13. The Command and Computer Subsystem and the Data Processing and Storage Subsystem are identical to those of the Bus/Lander; however, the S-band amplifier chain and antenna used during the atmospheric-descent phase have been deleted, and a Relay Transmission Subsystem has been added.

The VHF Relay Transmission Subsystem provides telemetry transmission to the associated Orbiter during the pre-entry and descent phase radiating five watts through a "transmission line" antenna. It also provides telemetry transmission to and command reception from the Orbiter during the surface phase. The latter links are alternates to the S-band direct links. Since the "transmission line" antenna can be damaged upon landing, a separate turnstile antenna is erected for surface-phase relay communications. A 25-watt, solid-state power amplifier is used for transmission.

Only the direct links to Earth utilize error correction encoding. When data is being transmitted to the Orbiter, the encoder is bypassed.

2. Performance Characteristics

The data rates of the Orbiter and Lander prime telemetry links are shown in Figure 4.1-14 as functions of transmission range based on the calculations of Section 4.1.2(A). Figures 4.1-15 and 4.1-16 show the relay data rates as functions of orbital altitude and range, respectively. Data rates selected for these links are indicated on the graph. Rates selected for all links are summarized in Section 4.1.1(C). Maximum carrier-lock range is given for each S-Band direct link in Table 4.1-13 under the conditions stated in Section 4.1.3(F)(2).

The relay performance of links 5 and 11 given in Figure 4.1-15 are based on the assumptions that the Lander is on the planet's horizon and the Orbiter antenna is pointed at the center of the planet. The antenna gain in the direction of the Lander is therefore varied with orbital altitude to account for the angle between the Lander and antenna boresight.

The relay capability during descent (link 6) given in Figure 4.1-16 is with respect to a turnstile antenna on the Orbiter; no variation of pointing loss is assumed.

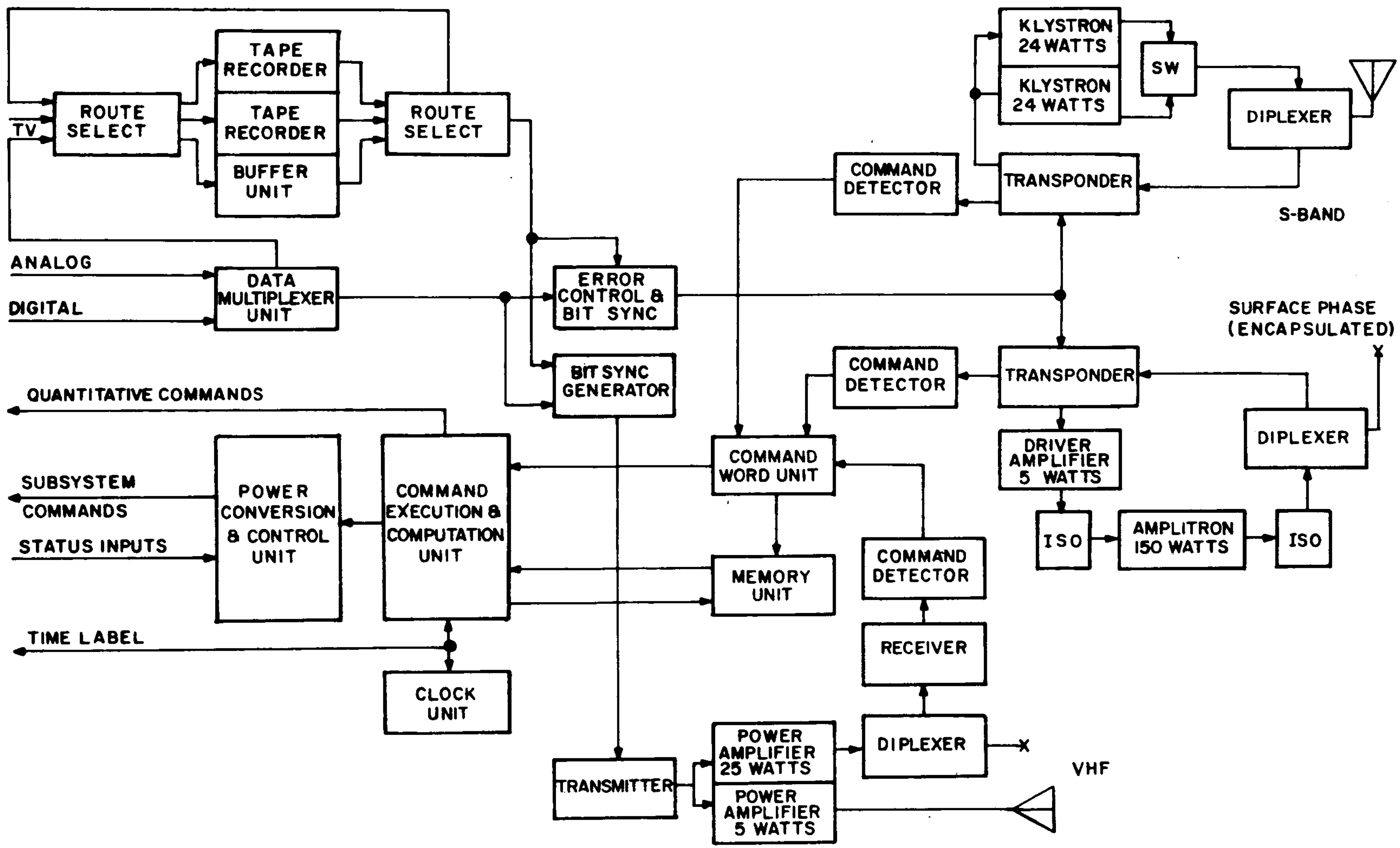


Figure 4.1-13. Lander Communication Subsystem Block Diagram (Orbiter/Lander Configuration)

TABLE 4.1-13. CARRIER-LOCK RANGE IN MILLIONS OF KILOMETERS (ORBITER/LANDER)

Link No.		1	2a	2b	3	4	6	7	8	9	10
Purpose		O-E TLM (Prime)	O-E TLM (Early Transit)	O-E TLM (Backup)	L-E TLM (Prime)	L-E TLM (Backup)		E-O Command (Prime)	E-O Command (Backup)	E-L Command (Prime)	E-L Command (Backup)
210 ft Dish	8-db Margin	3200	110	168	1100	205					
	No Margin	8100	278	420	2780	515					
85 ft Dish	8-db Margin	1100	38	58	380	72					
	No Margin	280	96	148	960	180					
10 KW	8-db Margin							1500	100	720	57
	No Margin							3700	257	1000	145
100 KW	8-db Margin							4700	320	2300	180
	No Margin							11800	810	5700	450

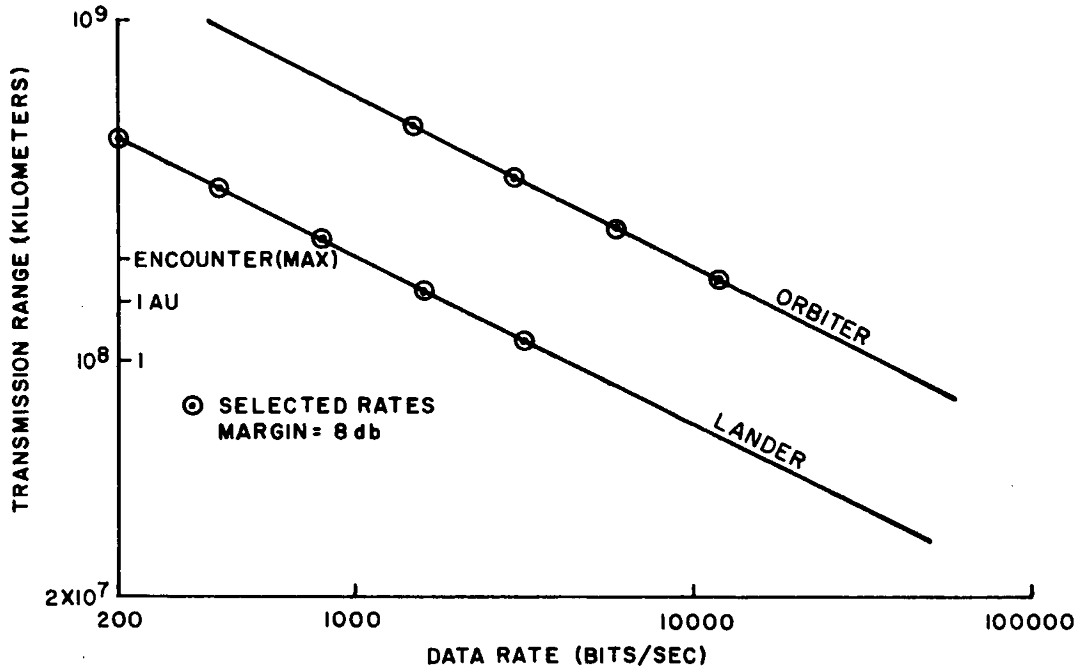


Figure 4.1-14. Orbiter and Lander Transmission Capability (Orbiter/Lander)

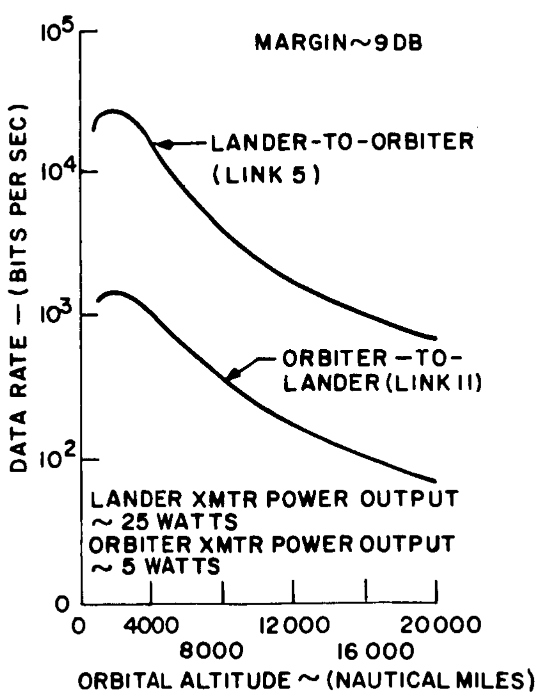


Figure 4.1-15. Data Rate in Orbit

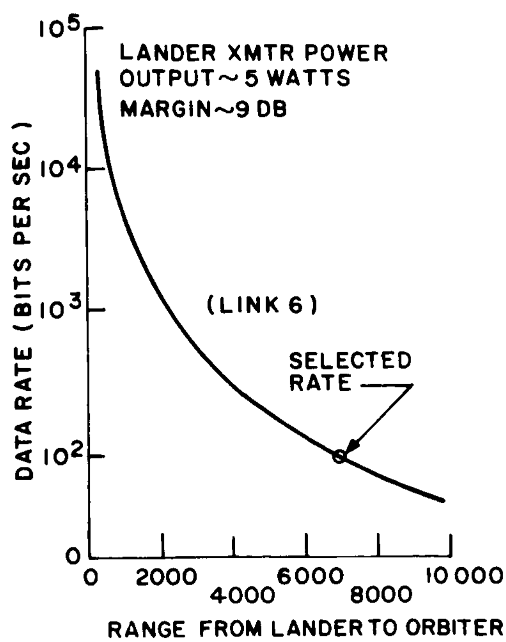


Figure 4.1-16. Lander-to-Orbiter Data Rate During Mars Lander Descent

3. Power, Weight, and Size

The component lists and estimates of power, weight, and size are given in Table 4.1-14 for the Orbiter and Lander Communication subsystems. They do not include cabling, harnessing, and payload compartment package structure listed in the vehicle weight sections (Section 3.4.2(H).

TABLE 4.1-14. POWER, WEIGHT, AND SIZE ESTIMATES (ORBITER/LANDER)

Orbiter

Component	No.	Unit Size (in.)	Unit Weight (lb)	Unit Power (w)
1. Preamplifier	1	2 x 2 x 4	2.0	1.0
2. High-Gain Antenna	1	8-ft dish	23.0	-
3. Omni	2	2.5 x 2.5 x 1.3	2.0	-
4. Diplexer	2	6 x 3.25 x 2	1.0	-
5. Transponder	2	184 $in.^3$	5.4	2.0
6. Power Amplifier (43W)	3	3.5 D x 5.5	3.0	143.3*
7. Power Supply (43W Amp)	3	4 x 4 x 6	6.0	180.0
8. Power Amplifier (60W)	1	2.75 x 3.0	2.5	125.0*
9. Power Supply (60W Amp)	1	4 x 4 x 6	4.5	156.0
10. Command Detector	2	4 x 4 x 5	3.0	1.75
11. RF Switch	1	2 x 2 x 2	1.0	-
12. Isolator and Load	2	10 $in.^3$	0.75	-
Data Processing and Storage				
13. Data Processing Unit	1	3 x 6 x 10	12.25	2.1
14. Multiplexer (PHP)	1	2 x 7 x 10	10.0	1.0
15. Tape Recorder	3	8 x 9 x 10.5	15.0	15.0 max.
16. Buffer Unit	1	140 $in.^3$	4.0	0.5

*Included in value for associated power supply.

TABLE 4.1-14. POWER, WEIGHT, AND SIZE ESTIMATES (ORBITER/LANDER) (Cont'd)

Component	No.	Unit Size (in.)	Unit Weight (lb)	Unit Power (w)
	Orbiter			
Command and Computer Subsystem				
17. Command & Computer Equip. (Main Body)	1	3.5 x 10 x 10	20.0	6.4
18. Power Conversion & Control (Main Body)	1	3 x 10 x 10	12.0	5.0
19. Command Decoder (PHP)	1	2.5 x 2.5 x 10	4.0	0.3
20. Power Conversion & Control (PHP)	1	2 x 2.5 x 11	2.0	10.0
Relay Transmission Subsystem				
21. VHF Antenna (Yagi)	1	13x2.1x2.1 (ft)	16.0	-
22. VHF Antenna (Turnstile)	1	4.2x4.2x2.5 (ft)	5.0	-
23. VHF Diplexer	1	2 x 2 x 2	1.0	-
24. VHF Transmitter (5W)	1	1.3 x 3 x 3	0.6	15.0
25. VHF Receiver	2	2 x 4 x 7	2.0	1.5
26. Data Demodulator	1	3 x 5 x 6	3.5	1.75
	Lander			
Deep Space Transmission Subsystem				
1. Diplexer	2	6 x 3.25 x 2	1.0	-
2. Helix Array Antenna	1	33 x 33 x 9	10.0	-
3. Encapsulated Turnstile Antenna	1	5 D x 7.5	5.0	-
4. Transponder	2	184 in.^{3}	5.4	2.0
5. Power Amplifier (24W)	2	3.5 D x 5.5	3.0	96.0*
6. Power Supply (24W Amp)	2	4 x 4 x 6	6.0	120.0
7. Power Amplifier (140W)	1	2.75 D x 4	4.0	284.0*
8. Power Supply (140W Amp)	1	5 x 5 x 6	8.0	355.0
9. Driver Amplifier & Power Supply (5W)	1	4 x 4 x 6	4.0	25.0

TABLE 4.1-14. POWER, WEIGHT, AND SIZE ESTIMATES (ORBITER/LANDER) (Cont'd)

Lander (Cont'd)

Component	No.	Unit Size (in.)	Unit Weight (lb)	Unit Power (w)
10. Command Detector	2	4 x 4 x 5	3.0	1.75
11. RF Switch	1	2 x 2 x 2	1.0	-
12. Isolator and Load	2	10 in.^{3}	0.75	-
Data Processing and Storage Subsystem				
13. Data Processing Unit	1	5 x 5 x 10	16.0	3.5
14. Buffer Unit	1	140 in.^{3}	4.0	0.5
15. Tape Recorder	2	5 x 7.5 x 8.5	8.0	5.0 max.
Command and Computer Subsystem				
16. Command and Computer Equip.	1	4 x 5 x 10	14.0	1.8
17. Power Conversion and Control	1	3.5 x 5 x 11	7.0	10.0
Relay Transmission Subsystem				
18. VHF Antenna System**	1	-	10.0	-
19. VHF Diplexer	1	2 x 2 x 2	1.0	-
20. VHF Transmitter (25W)	1	3.5 x 3 x 3	1.3	115.0
21. VHF Transmitter (5W)	1	1.3 x 3 x 3	0.6	15.0
22. VHF Receiver	1	2 x 4 x 7	2.0	2.0
23. Command Detector	1	4 x 4 x 5	3.0	1.75

**Includes "transmission line" and turnstile antennas.

4.2 GUIDANCE AND CONTROL

4.2.1 GUIDANCE

A. OBJECTIVE AND GROUND RULES

Studies of the Guidance Subsystem were undertaken only where the requirements of the Titan mission differed from the capabilities of the previous Voyager study. Investigation of alternatives to minimize communication requirements was not to be included unless it became necessary because of other limitations.

B. SUBSYSTEM DEFINITION

As in the previous study, the Guidance Subsystem consists of the DSIF transponder (included with the Communications equipment) plus a sensor for taking successive readings of the orientation of the line of sight to the planet. The latter is determined by reading the angles between the planet and two or more stars at intervals, beginning at a point approximately two million n.mi. from the planet. By instruction at the beginning of the study, the only means considered for accomplishing this was to take pictures of the planet and stars with an image orthicon TV camera and transmit the pictures to Earth for processing.

C. RESULTS OF STUDY

No requirement for further studies of this subsystem arose, inasmuch as the communications requirements were met on each of the vehicles. The maximum time for transmitting a TV frame is 45 minutes (for the Bus/Lander). Two items of information did develop during this period which are of interest to this study:

1. The accuracy with which the line of sight can be determined from a TV frame, 1 milliradian, is recognized as a 3σ value rather than 1σ as it was considered previously.
2. During a Company-funded investigation a very simple solution was found to the problem of the effective range of brightness between the planet and stars. By appropriate choice of the stars of interest, it is possible to provide separate optical paths for the star field and the planet. As a result, the planet's image may be filtered as heavily as desired without affecting the star images or obstructing the field of view. In this way the camera can be assured that the planet's image will not exceed a comfortable brightness level.

4.2.2 CONTROL

A. OBJECTIVE

The objective of this study was to identify the areas in which either a mission or the vehicle design would be significantly affected by a change from the Saturn S1B booster to the Titan IIIC. For certain areas where changes are not anticipated, "housekeeping" studies were required to provide updated values of parameters such as the required total impulse for attitude control for each of the new vehicle configurations.

B. GROUND RULES

Except for those cases where changes are indicated as a result of the Titan mission, the control subsystem is unchanged from that of the previous Voyager study. No additional studies were devoted to component analysis.

The two areas of change are the PHP drive and the Lander antenna drive.

C. SUBSYSTEM DEFINITION

The control subsystem for a Bus/Lander consists of the following:

1. Vehicle Control
   - Gyros (3)
   - Gyro control
   - Accelerometer (3-axis)
   - Autopilot amplifier
   - Sun sensors, fine and coarse
   - Canopus tracker
   - Logic, storage, and relay units
   - Power supply
2. Antenna Control
   - Antenna drive electronics
   - Earth sensor
   - Antenna actuators (2)

The Orbiters have, in addition, the PHP control which consists of:

Horizon Scanner (IR)
PHP drive electronics
PHP actuators (3)

As in the previous Voyager, Orbiters may also carry a second Canopus tracker to permit inverting the vehicle in case of a decision to change the orbit plane after launch.

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

D. RESULTS OF STUDY

1. Gas Consumption

Values were obtained for attitude control total impulse for each of the vehicles in the same manner as in the previous Voyager study. Calculations in each case are based on the dimensions and inertias of the respective vehicles.

Impulse for initial acquisition is based upon worst rates and attitude about each axis. Midcourse maneuvers are based on acceleration to maneuver rate, and deceleration, for each axis. Solar torque impulse calculations assume a high gain antenna to be continuously extended in the worst attitude to the sun for the duration of the trip.

The results may be summarized as follows:

Source	Impulse Required		
	All-Orbiter	Bus/Lander	Orbiter/Lander
Initial Acquisition	23	28	39
Midcourse Maneuvers (5)	21	35	39
Solar Torque in Transit	353	39	480
Solar Torque in Orbit	141	--	192
Reacquisitions (5)	110	134	186
Rocket Burning Roll Control	27	19	33
Gravity Gradient in Orbit	407	--	48
Total Impulse Required	1082 lb-sec	255 lb-sec	1017 lb-sec

These values will vary with trip time as shown in Figure 4.2-3.

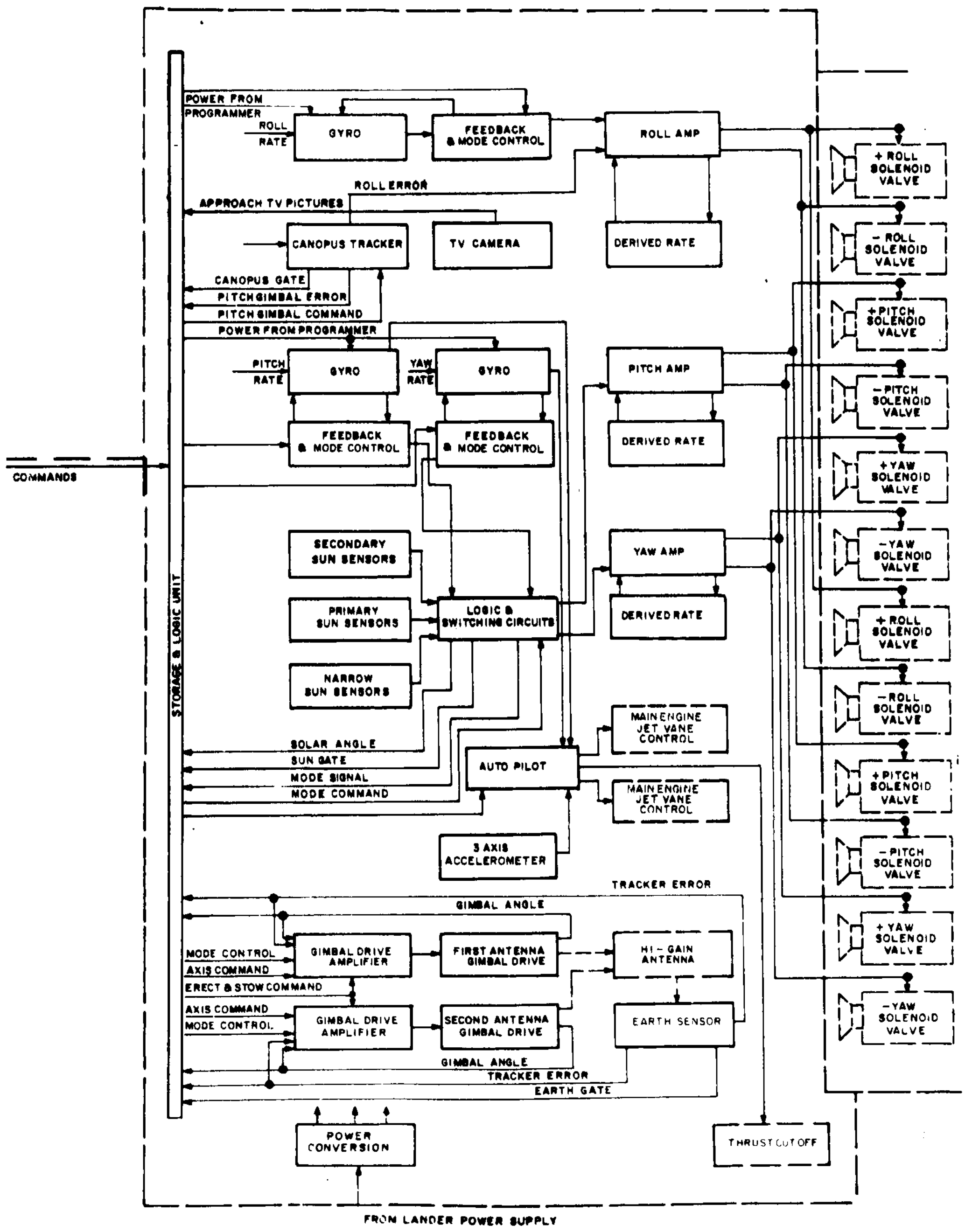


Figure 4.2-1. Guidance and Control Subsystem — Bus/Lander

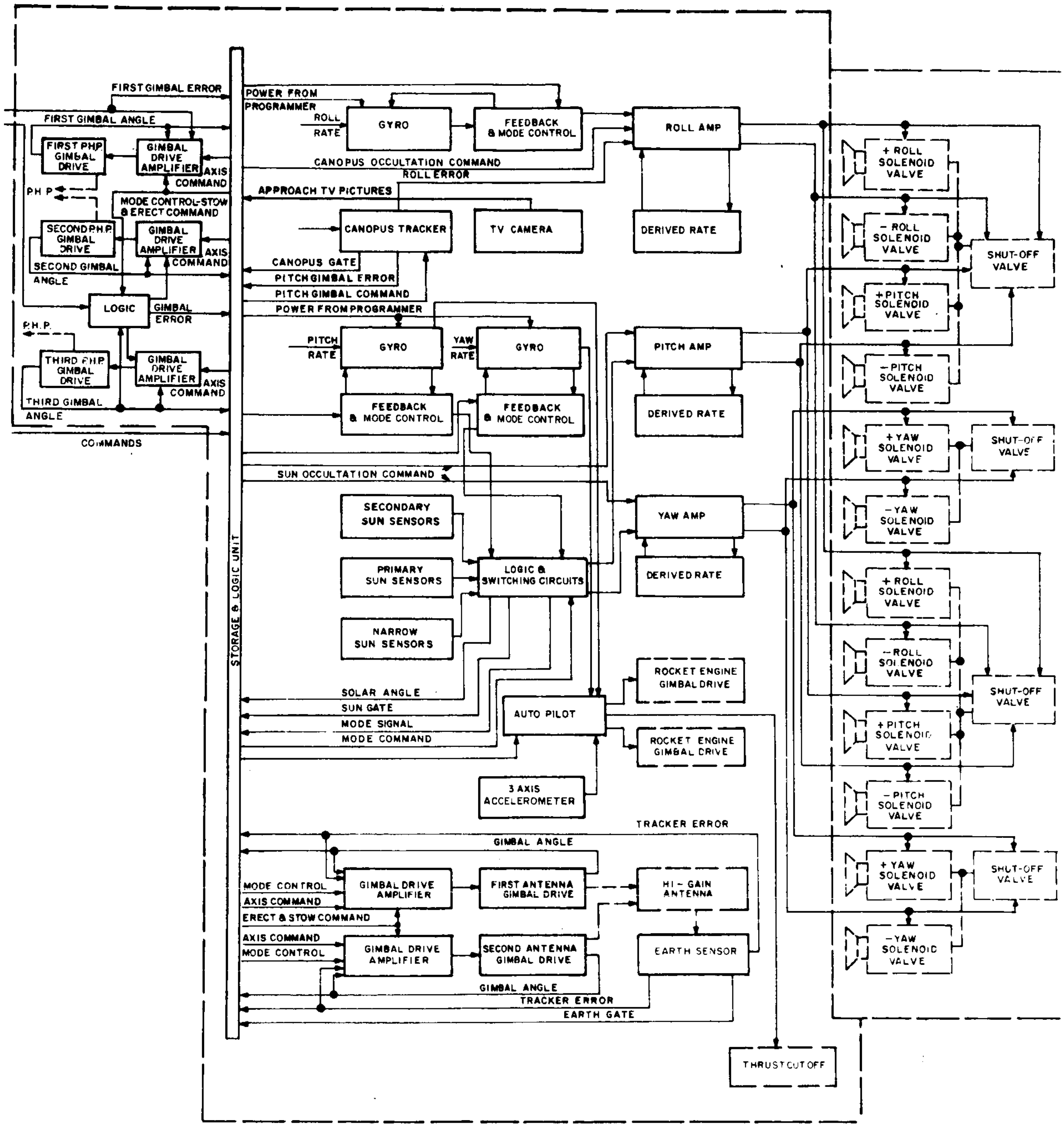


Figure 4. 2-2. Guidance and Control Subsystem — All-Orbiter and Orbiter/Lander

These results are based on the following parameters and assumptions;

1. Mars 1971 trajectory
2. Vehicle average acceleration 0.25 mr/sec^2
3. Transit time 225 days
4. Orbit time (All-Orbiter and Orbiter/Lander) 90 days
5. Jet minimum on-time 30 ms
6. Position deadbands ±4 mr
7. Initial rates 3°/sec
8. Maneuver rates 10 mr/sec
9. Engine roll torque scaled for 900-pound thrust

For the All-Orbiter case a 1000 n.mi. circular orbit was assumed, and for the Orbiter/Lander, a 1000 x 5000 n.mi. orbit.

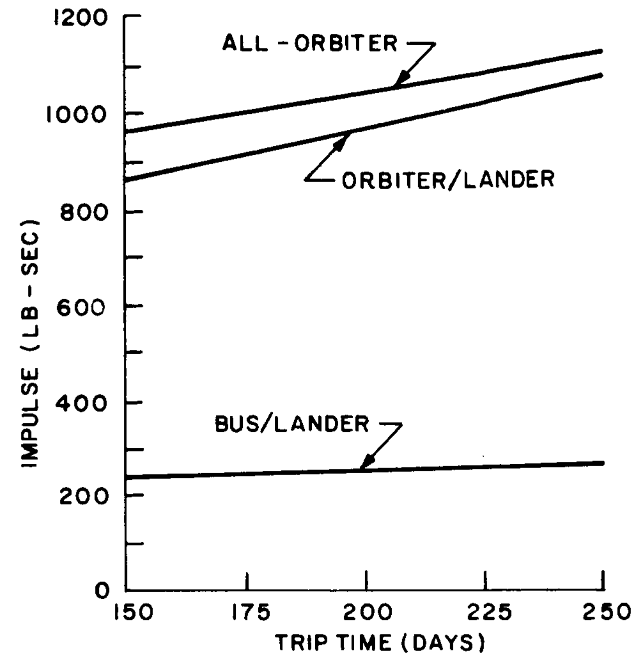


Figure 4.2-3. Variation of Control Impulse Requirements with Trip Time

No additional impulse is required for limit-cycle operation since the value of solar torque calculated is such as to ensure one-sided operation for the control system parameters selected. The values of impulse listed are as-calculated values and do not include any multiplier or safety factor.

2. Nozzle Location

It was concluded that no problem would result if the attitude control nozzles are located off the primary vehicle axes in order to avoid control gas impingement on the PHP or high-gain antenna. In this case the attitude sensors and the attitude of the vehicle would be unchanged; the nozzle locations would be accommodated by resolving the attitude sensor signals into the nozzle coordinates. Inertia products are not expected to cause difficulty.

3. PHP Drive

One area in which the 1971 Titan-launched vehicle differs from the 1969 Saturn-launched vehicle is in the configuration and drive requirements for the Planet Horizontal Package.

In the previous Voyager, a two-axis arrangement was adequate. The configuration selected there was equivalent to a conventional azimuth and elevation drive.

The azimuth axis in that case is fixed relative to the vehicle and lies nominally normal to the orbit plane. The elevation drive accommodates any deviation or change in the orbit plane so that the PHP is oriented in the plane of the orbit.

For the Titan-launched 1971 Mars mission, the selected orbit has a higher inclination than previously and it is also a more nearly circular orbit. As a result, the orbit nodal regression is much greater than in the previous Voyager study. Consequently, a three-axis configuration is required to enable the PHP to rotate in the orbit plane and achieve the necessary look angles with a reasonable configuration.

In the previous system, a two-axis horizon sensor provided the control signal for the PHP. One axis of the sensor controlled the "azimuth" or primary PHP drive (in-plane motion) and the other axis of the sensor controlled the "elevation" drive which accommodated the difference between the nominal and the actual orbit plane.

In the three-axis drive the first channel of the horizon sensor controls the PHP motions in the orbit plane as before. The second axis of the horizon sensor controls the remaining two drive axes so that the first axis tracks the normal to the orbit plane, enabling the PHP motion to follow the plane of the orbit. As the PHP rotates, the second channel of the IR sensor controls alternately the remaining two drives. At one point in the orbit it senses error in one drive; 90 degrees around the orbit it senses the other drive. Consequently, to change from a two-axis to a three-axis drive it is necessary only to provide the additional drive mechanism for the third axis, and to resolve or switch the second horizon sensor channel between the second and third axis drives according to the orientation of the primary drive.

4. Lander Antenna Drive

A specific study of the Lander Antenna Control subsystem was not included in the scope of this study. In general, the approach described in Section 4.1 was covered in the previous Voyager study. The system as now described includes an "equatorial" axis that is mechanically aligned to the Mars axis. Relative to this axis and the Mars-Sun line, the direction to the Earth at any time is known.

Once aligned, the antenna drive needs only to remove the planet's rotation, and update the Earth-Mars-Sun angle from time to time. The latter may be programmed or by command. The former can be accomplished either by a clock drive or time program of angles, or by actively tracking the sun.

Alignment of the equatorial axis can be carried out with a sun sensor as follows. The equatorial axis that carries the antenna is in turn carried on two axes. One of these can be aligned with respect to the sun at either sunrise or sunset; the other, at noon. In each case the axis will be approximately normal to the sun line; deviating according to the time of year of landing. Once aligned, the drives are turned off. They do not change with the season.

An alternative way of aligning the equatorial axis would be with gyrocompassing techniques.

5. Planet-Oriented Orbiter

If an RTG power supply is provided, it would be possible to orient the entire Orbiter to the planet during the orbiting period rather than orienting to the Sun and Canopus, and eliminate a PHP entirely. The payload instrumentation would then be mounted directly in the Orbiter and no particular orientation relative to the Sun would be required.

No particular problems would be expected in controlling the vehicle to do this. Earth orbiting satellites with two-axis horizon sensor control to the local vertical have already been developed. For these cases, yaw orientation (rotation about the local vertical) normally requires a gyro. The Voyager orbiting life is short enough that gyro life will not be a problem.

The primary control problem associated with a planet oriented orbiter is found in the high-gain Earth antenna control. The problem is analogous to giving an Earth-orbiting satellite a requirement to acquire and track Mars each orbit. In a vehicle referenced to the Sun and Canopus, the orientation of the antenna relative to the vehicle does not change from Earth-set to Earth-rise. The only antenna motions required are those resulting from the seasonal motion of the Earth and Mars around the sun. For this reason, antenna motions are very slow and it is a simple matter to program the antenna orientation relative to the vehicle as a backup mode of operation.

With a planet-oriented vehicle, the antenna orientation relative to the vehicle changes through a large angle between Earth-set and Earth-rise. Hence, programming the antenna orientation relative to the vehicle becomes a major problem.

In order to take stereo pictures it is necessary for the vehicle to maintain a specific axis in the orbit plane. For this reason it is necessary to give the antenna a two-axis drive, or more probably a three-axis drive because of the degree to which the orbit plane will precess. The problem here is similar to that of the PHP drive mentioned above.

It is true that while the motions between the antenna and the vehicle are complex, the antenna motion relative to inertial space remains very slow the same as if the vehicle were Sun/Canopus referenced. Consequently, it would be possible to mount gyros on the antenna and transfer control to them during the periods of Earth occultation. Another alternative would be to mount a star tracker on the antenna looking nominally away from the Earth. During the period of Earth visibility, this star tracker can lock onto any convenient star on its field of view. When Earth visibility is lost, the star tracker will then control the antenna to maintain the star in the same relative position. It is not necessary to select or even know which star is utilized. Either way the antenna orientation can be maintained so that a full Earth search sequence is avoided each orbit.

The two primary disadvantages are (1) the difficulty in programming the antenna orientation to acquire the Earth initially or in case of any malfunction or maneuver; (2) no use is made of the Sun as a reference during the orbiting period. Because of the overwhelming visibility and ease of identifying the Sun, and the simplicity of Sun sensors, the use of other references should be restricted to those cases where the Sun cannot fully meet the needs.

4.3 POWER SUPPLY

4.3.1 INTRODUCTION

In the Voyager - Saturn 1B study, a detailed investigation was made of the following potential power supplies for unmanned Mars Orbiters and Landers.

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_2-0_2 Fuel Cells
14. Secondary Nickel Cadmium Batteries
15. Secondary Silver Cadmium Batteries
16. Primary Silver Zinc Batteries

The criteria used in evaluating these as potential power supplies were:

1. Availability (Including Development Uncertainty)
2. Weight and Size
3. Complexity/Reliability
4. Cost
5. Degree of Uncertainty in Performance Estimates

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.

The primary reasons for rejecting the other power supplies are summarized in Tables 4.3-1 and 4.3-2.

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 radioisotope availability estimates on which both of these studies were based are given in Figure 4.3-1. 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.

MARS ORBITER

Solar Cells
Radioisotope Thermoelectrics
Secondary Nickel Cadmium Batteries

MARS BUS/LANDER

Radioisotope Thermoelectrics
Secondary Nickel Cadmium Batteries

Each of these recommended energy conversion and storage means have been used to supply space power. In fact solar cells and radioisotope thermoelectrics represent the only two energy conversion systems that have been used to date in space. Their demonstrated availability, performance and reliability were major factors behind their selection for consideration in this study.

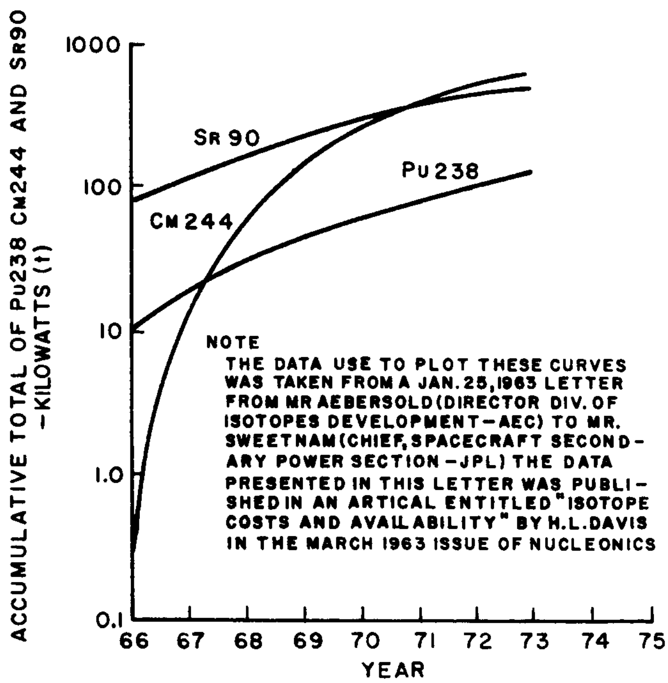


Figure 4.3-1. AEC Availability Estimates for CM 244, Pu 238 and SR 90

TABLE 4.3-1. POWER SYSTEMS CONSIDERED FOR ORBITERS

Power Supply	Major Reasons for Rejection	
	Mars 1969	Future Mars Missions
Nuclear Thermoelectric	Weight, Availability	Weight
Nuclear Thermionic	Weight, Availability	Weight
Nuclear Dynamic	Weight, Availability	Weight
Isotope Thermoelectric	Isotope Availability	Possible Alternate
Solar Thermionic	Availability, Environmental Uncertainty (Effects on Collector)	No Weight Saving Environmental Uncertainity (Effects on Collector)
Solar Thermoelectric	Size Environmental Uncertainty (Effects on Selective Coatings or Collectors)	Size Environmental Uncertainty (Effects on Selective Coatings or Collectors)
Solar Dynamic	Availability Environmental Uncertainty (Effects on Collectors)	Complexity Environmental Uncertainty (Effects on Collectors)
V-Ridge Photovoltaic	Environmental Uncertainty (Effects on Reflective Surfaces)	Environmental Uncertainty (Effects on Reflective Surfaces)
Unconcentrated Photovoltaic	Recommended System	Recommended System
Concentrating Photovoltaic	Weight Environmental Uncertainty (Effects on Collector)	Weight Environmental Uncertainty (Effects on Collector)

TABLE 4.3-2. POWER SYSTEMS CONSIDERED FOR LANDERS

Power Supply	Major Reasons For Rejection	
	Mars 1969	Future Mars Missions
Nuclear Thermoelectric	Weight, Availability	Weight
Nuclear Dynamic	Weight, Availability	Weight
Isotope Thermoelectric	Recommended System	Recommended System
Isotope Thermionic	Availability	Possible Alternate

TABLE 4.3-2. POWER SYSTEMS CONSIDERED FOR LANDERS (Cont'd)

Power Supply	Major Reasons For Rejection	
	Mars 1969	Future Mars Missions
Solar Systems	Deployment and Orientation Reliability Weight Effects of Clouds and Atmospheric Conditions on the Planet	Deployment and Orientation Reliability Weight Effects of Clouds and Atmospheric Conditions on the Planet
Primary Fuel Cells	Weight	Weight
Primary Batteries	Weight	Weight
Secondary Batteries	Recommended for Handling Peak Loads	Recommended for Handling Peak Loads

4.3.2 POWER SUPPLY SUBSYSTEM DESIGN

The recommended power supply subsystem designs for the Bus/Lander, Orbiter, and Lander/Orbiter systems are summarized in this section.

A. BUS/LANDER

1. Power Requirements

The Bus/Lander power requirements are summarized in Figures 4.3-2, 4.3-3, and 4.3-4. A detailed discussion of how these profiles were established is presented in Section 2.5.

2. Functional Description

The power for the Bus/Lander is supplied for both the transit and the surface portions of the mission by the Lander power supply. The prime Lander power supply is a radioisotope thermoelectric generator. It is supplemented during peak loads with rechargeable nickel cadmium batteries.

A schematic diagram of the Bus/Lander power supply is shown in Figure 4.3-5. The efficiencies assumed for the performance of each of the components are also shown on Figure 4.3-5. The RTG supplies power to the load and charges the batteries during off-peak periods.

The battery charge regulator controls the rate at which the batteries are charged. The battery provides coarse voltage regulation, approximately ± 15 percent at the bus. Each load provides its own precise voltage level and regulation.

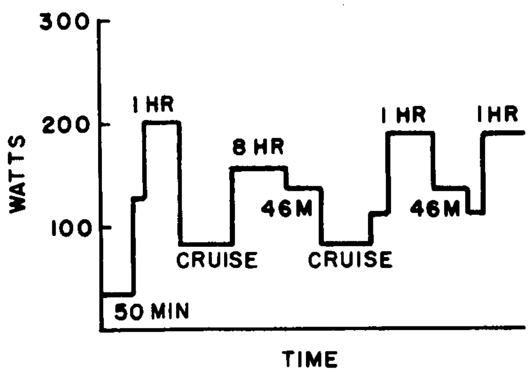


Figure 4.3-2. 2060-lb Bus/Lander Only Transit Phase Power Profile

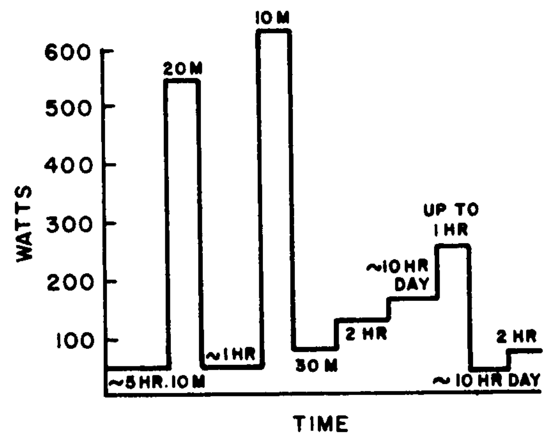


Figure 4.3-3. 2060-lb Lander/Separate Phase Power Profile

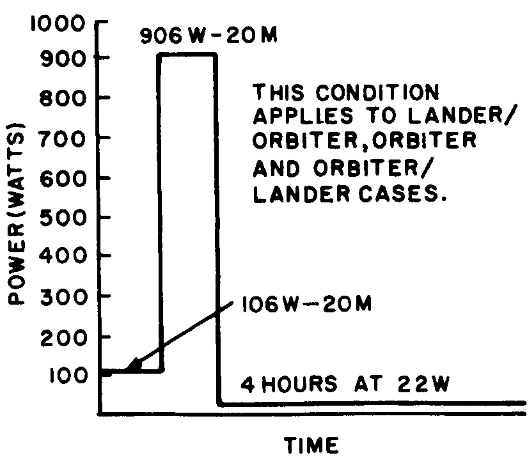


Figure 4.3-4. Emergency Power Requirements (No Solar Power)

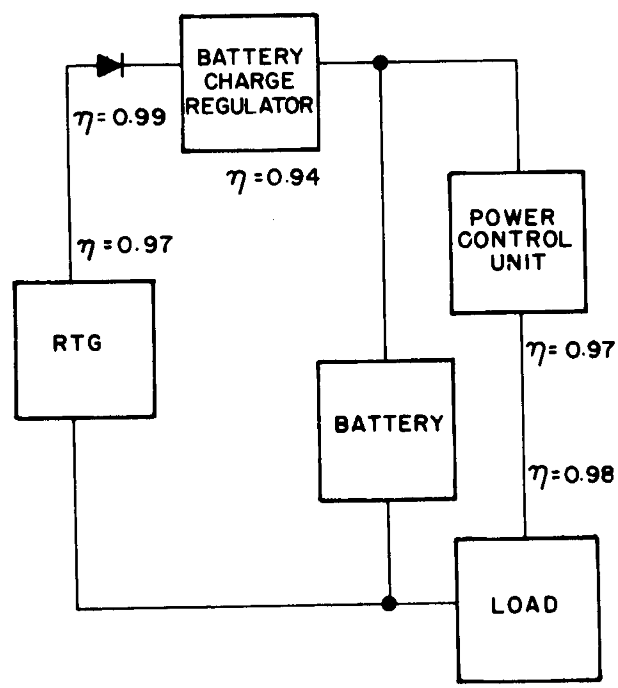


Figure 4.3-5. Power System Simplified Block Diagram

The power control unit provides for switching of the various component loads by command or preprogramming. It also can provide some circuit protection.

The RTG will be cooled by either convection or radiation depending on the mode of operation. Convection will be used until planetary impact, with the circulating coolant rejecting heat to a ground radiator prior to launch, a water evaporator during launch and entry and to a space radiator during transit. After planet impact, the heat will be primarily rejected by radiation.

3. Performance Characteristics

The radioisotope thermoelectric generator performance characteristics are presented in Table 4.3-3.

TABLE 4.3-3. ISOTOPE THERMOELECTRIC GENERATOR DESIGN

Power Output of Generator	198 watts
Power Available at the Load	170 watts
Output Voltage	28 volts
Thermoelectric Efficiency	4.7 %
Generator Efficiency	4.3 %
Thermoelectric Material	Ge Si
Number of Thermocouple Pairs	240
Number of Series Strings	2
Isotope	Cm 244
Initial Thermal Output	4780 watts
Thermal Output - 1 year	4600 watts
Hot Junction Temperature	1300^{o}F
Cold Junction Temperature	575^{o}F

4. Size and Weight

The size and weight of the two major components of the Bus/Lander power supply are given in Table 4.3-4. Auxiliary regulation and control equipment sizes and weight are presented in the detailed system weight breakdowns in Section 3. 2.

TABLE 4.3-4. BUS/LANDER POWER SUPPLY SIZE AND WEIGHT

Radioisotope Thermoelectric Generator	
Weight	89.2 pounds
Distance Across Flats	8.5 inches
Height	16.3 inches
Fin Length	8.1 inches
Rechargeable Nickel-Cadmium Batteries	
Weight	24.6 pounds
Volume	344 $in.^3$
Capacity	8 amp hours

B. ORBITER

1. Power Requirements

The power profile for the 1971 Mars mission is shown in Figures 4.3-6 and 4.3-7. Investigating the detailed power breakdown given in Section 2.6, it is noted that the peak load occurs during the period when the television is on and the orbiter is communicating to earth.

The maximum load on the battery is when the vehicle is in orbit and the occultation time is a maximum.

2. Functional Description

The Orbiter power supply schematic is identical to the Bus/Lander power supply schematic except that the RTG is replaced by a solar array. The array is composed of two portions, a body mounted section and a shelf. Approximately three-fourths of the power comes from the shelf mounted array.

3. Performance Characteristics

The performance of the Orbiter solar array power supply is itemized in Table 4.3-5.

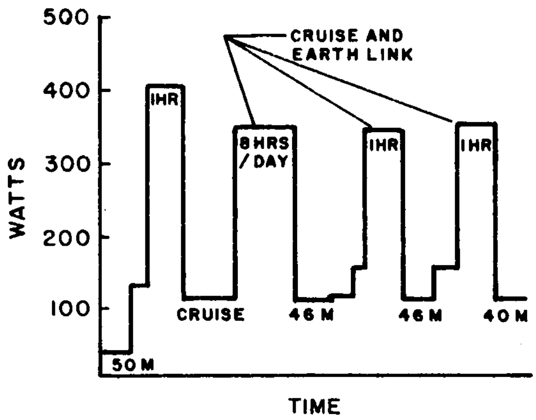


Figure 4.3-6. All-Orbiter Power Profile-Transit Phase

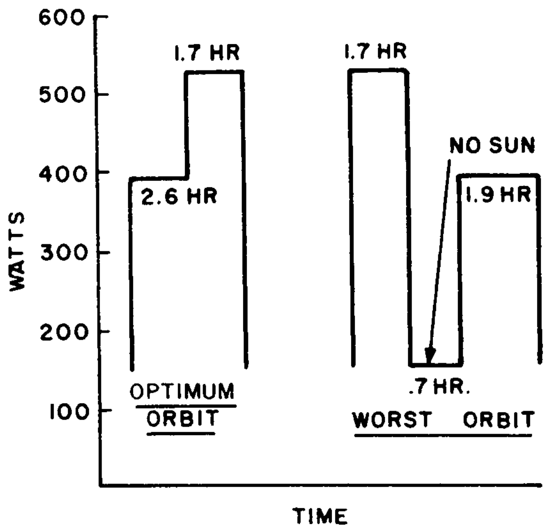


Figure 4.3-7. All-Orbiter Power Profile-Orbiting Phase

TABLE 4.3-5. SOLAR ARRAY PERFORMANCE

Solar Array	
P1/A - Body Mounted Array[1]	3.02 watt/ft^2
P1/A - Shelf Mounted Array[1]	3.21 watt/ft^2
Pa/A - Body Mounted Array[2]	3.52 watt/ft^2
Pa/A - Shelf Mounted Array[2]	3.75 watt/ft^2
Body Cell Temperature	79^{o}F
Shelf Cell Temperature	44^{o}F
Solar Flux	51.2 watt/ft^2
Distance from Sun	1.594 AU

NOTES:

1. Based on power to load and active cell area equal to 0.9 array area.
2. Based on array power output and active cell area equal to 0.9 array area.

The individual solar cell performance factors are given in Table 4.3-6.

TABLE 4.3-6. SOLAR CELL PERFORMANCE FACTORS

Type of cell	N/P
Efficiency of bare cell (Free space, 85°F)	11%
Manufacturing loss factor	0.97
Ultra-violet degradation factor	0.95
Meteorite loss factor	0.95
Packing factor	0.9
Temperature degradation	-0.26% per °F above 85 °F
Radiation degradation factor	0.78
Cover glass thickness - mils (Fused silica)	6
Solar absorptivity	0.938
Transmittance factor (relative to bare cell)	1.000

The Solar array thermal factors which were used are given in Table 4.3-7.

TABLE 4.3-7. SOLAR ARRAY THERMAL FACTORS

Emissivity	
Front of cell	0.83
Front of structure	0.80
Back of structure	0.90
Solar absorbitivity	
Front of structure	0.10
Mars albedo	0.15 sun
Mars effective radiating temperature, °F	-47

4. Size and Weight

The solar array has been designed to provide 600 watts of power to the load. This agrees within 1 percent of the detailed estimate of power required. The requirement is based on the assumption that the battery will be charged for 3.6 hours during the worst orbit condition.

The details of the solar array and battery size and weight are presented in Table 4.3-8.

TABLE 4.3-8. SOLAR ARRAY AND BATTERY SIZE AND WEIGHT

Solar Array		
Body mounted array area	52.26	ft^2
Shelf mounted array area	137.75	ft^2
Total weight	190	pounds
Battery		
Weight	26.3	pounds
Volume	368	$in.^3$
Capacity	8.5	amp hours

C. ORBITER/LANDER

1. Power Requirements

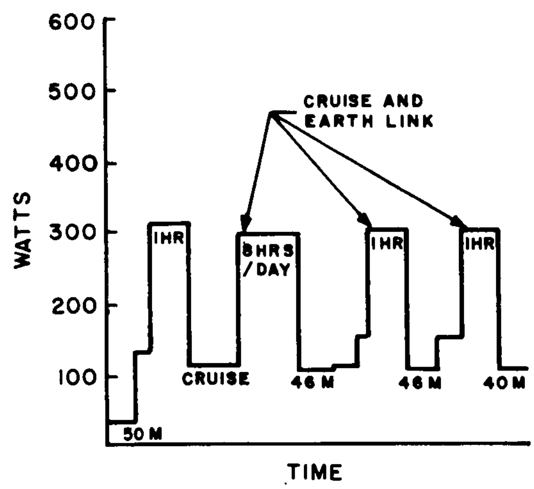
The power requirements for the Orbiter/Lander are outlined in Figures 4.3-8 and 4.3-9, and a detailed breakdown of how these profiles were obtained is given in Section 2.6.

2. Functional Description

The Lander power supply consist of a RTG supplemented with secondary nickel cadmium batteries. The Orbiter power supply is based on solar cells and secondary nickel cadmium batteries. The Lander and Orbiter power supplies operate as described in Sections 4.3.2.A and 4.3.2.B, respectively.

3. Performance Characteristics

The solar array performance characteristics are identical to those outlined in Tables 4.3-5, 4.3-6, and 4.3-7. The RTG performance characteristics are summarized in Table 4.3-9.



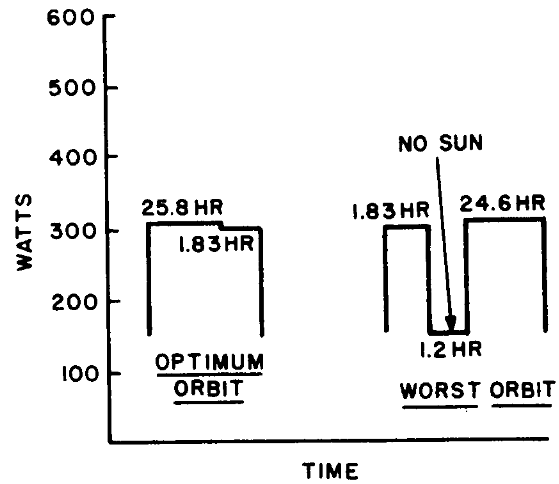


Figure 4.3-8. Orbiter/Lander Power Profile - Transit Phase

Figure 4.3-9. Orbiter/Lander Power Profile - Orbiting Phase

TABLE 4.3-9. ISOTOPE THERMOELECTRIC GENERATOR DESIGN

Power output of generator	128 watts
Power available at load	110 watts
Output voltage	28 volts
Thermoelectric efficiency	4.7 %
Generator efficiency	4.3 %
Thermoelectric material	Ge Si
Number of thermocouple pairs	240
Number of series strings	2
Isotope	Cm244
Initial thermal output	3090 watts
Thermal output - 1 year	2970 watts
Hot junction temperature	1300 ^{o}F
Cold junction temperature	575 ^{o}F

The orbiter battery in this configuration will be depth of discharge limited rather than charge rate limited, but the basic battery design will not be significantly different.

4. Size and Weight

The orbiter and lander power supply sizes and weights are presented in Tables 4.3-10 and 4.3-11, respectively.

TABLE 4.3-10. ORBITER POWER SUPPLY SIZE AND WEIGHT

Solar Array	
Body mounted array area	51.19 ft^2
Shelf mounted array area	54.0 ft^2
Weight of solar array	105.2 pounds
Battery	
Weight	34.2 pounds
Volume	480 $in.^3$
Capacity	11 amp hours

TABLE 4.3-11. LANDER POWER SUPPLY SIZE AND WEIGHT

Radioisotope Thermoelectric Generator	
Weight	73 pounds
Distance across flats	8.5 inches
Height	14 inches
Fin length	7.8 inches
Rechargeable Nickel-Cadmium Batteries	
Weight	7.7 pounds
Volume	109 $in.^3$
Capacity	2.5 amp hours

4.3.3 POWER SUPPLY SUBSYSTEM ANALYSIS

A. BUS/LANDER

1. Power Supplies Considered

As stated in the introduction, Section 4.3.1, only a radioisotope thermoelectric generator with secondary nickel cadmium batteries was considered for the Lander power supply. Only a radioisotope thermionic system was considered as a possible alternate. It does not appear that sufficient advances have been made in the state-of-the-art of radioisotope thermionic generators in the past year to warrant more serious consideration of them for the lander power supply. If the system development of radioisotope thermionic generators were to make a significant advancement, the Lander could be modified in a relatively simple manner to accept the thermionic system.

If it were assumed that the specific weight of an isotope thermionic generator would be in the range of 254 lb/kw as shown in the Voyager - Saturn 1B study, then the weight saving using a 198w thermionic generator would be approximately 39 pounds. It seems doubtful that this weight saving would be sufficient to warrant the development of an isotope thermionic system.

If sufficient incentive were apparent for using an isotope thermionic system, development would have to be started immediately since it was shown in the Voyager-Saturn 1B study that approximately 5.8 years would be required to develop the thermionic system.

2. RTG System Design

The radioisotope thermoelectric generator has been designed to provide sufficient power for the lander when in a direct communication mode. This agrees with the results obtained in the Voyager - Saturn 1B study which indicated that for charge times longer than 4.5 hours and discharge times greater than 3.0 hours that a lighter system resulted from using an RTG to handle the entire load rather than supplementing the RTG with batteries for the long discharge times. The RTG is supplemented with rechargeable nickel-cadmium batteries to handle the peak loads which occur:

1. During direct communication with the hi-power omni-antenna
2. During drilling

3. During transmission of terminal guidance information
4. During transmission while making initial orientation maneuver

If operations 1, 3, and 4 were the only requirements for the battery to supply power, a primary battery might have been considered. However, drilling will occur at least once each night and will require at least 180 recharge cycles of the battery. This dictated the choice of nickel-cadmium rechargeable batteries for the Bus/Lander.

A bonus also results from this sizing philosophy since the batteries can be used to power the high power omni directional antenna for short periods of time if the operation of the directional antenna becomes impaired.

a. Nickel-Cadmium Battery Characteristics

The following assumptions were made in estimating performance of sealed, rechargeable nickel cadmium batteries:

1. Battery capacity, including the case but not including thermal control, is assumed to be 9 watt-hour/lb for 100 percent depth of discharge.
2. Constant current charging is assumed throughout the charge and overcharge period.
3. The maximum allowable current during the overcharge condition is assumed to be that which will supply 100 percent ampere hour capacity in a period of six hours. (This value is based on past experience. Charging currents in excess of this are considered to run too high a risk of battery failure due to excessive generation of gas and build-up of internal pressure. There is also a heating problem.)
4. The maximum allowable depth of discharge for repeated cycling is assumed to be 60 percent. (For charging times less than 4.5 hours, the maximum allowable current during the overcharge condition as noted in 3 will determine battery size, and the depth of discharge will be less than 60 percent, varying linearly with charge time up to a charge time of 4.5 hours. For charge times greater than 4.5 hours, the charging rate is cut back from the 6-hour rate in order to hold depth of discharge at 60 percent.)
5. The excess ampere-hours of overcharge required to maintain continuous cycling is assumed to be 25 percent for a six-hour charging rate, increasing linearly with charging rate to a value of 100 percent for a 16-hour charging rate. It is further assumed that the charging current cannot be reduced to less than the 16-hour rate if continuous cycling is maintained. (The assumption of a linear variation of excess ampere hours with charging rate is arbitrary. The other assumptions are based on strong, but not necessarily conclusive, indications from past testing experience, principally on the Advent program.)

B. ORBITER

1. Power Supplies Considered

Following the guide provided by the Voyager-Saturn 1B study, photovoltaic arrays and radioistope thermoelectric generators were considered as the primary power supply. Both of these systems would be supplemented by rechargeable nickel-cadmium batteries if required.

For the purposes of initial comparisons both systems were assumed to supply 600 W of power to the load.

2. RTG System Design

As in the Bus/Lander system, Cm 244 was chosen as the radioisotope. It was considered prudent based on discussions with generator vendors to build two generators to supply the load. A configuration similar to that presented in Reference 1 was assumed. An outline drawing of the generator is shown in Figure 4.3-11. The specific characteristics of the RTG are shown in Table 4.3-12.

If it is assumed that five sets of two generators each would be required for the Titan III mission, they would then require 17.8 percent of the Curium 244 available through 1970.

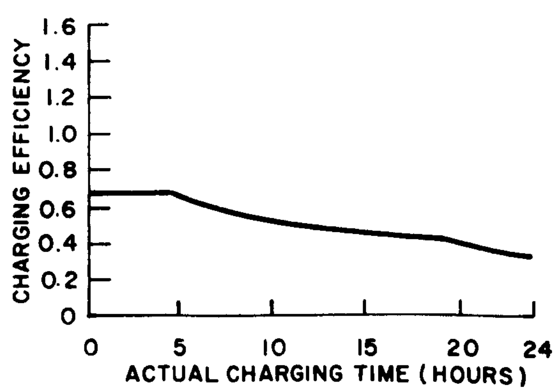


Figure 4.3-10. Estimated Charging Efficiency of Nickel Cadmium Battery

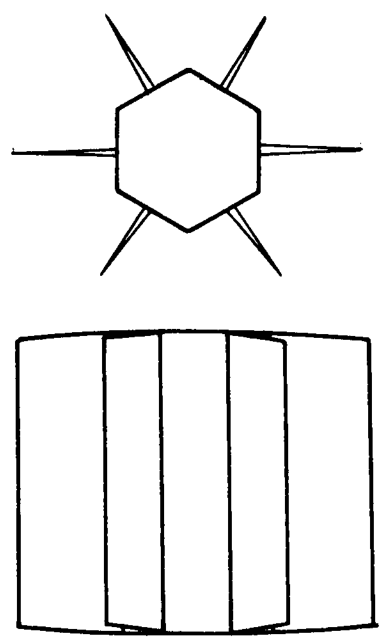


Figure 4.3-11. RTG Configuration

6. For purposes of calculating charging efficiency of the battery, defined as the ratio of the watt-hours delivered during discharge to the watt-hours put back into the battery during the charge plus overcharge periods, it is assumed that the average discharge and charge voltages are 1.2 and 1.43 volts per cell, respectively. This assumption, together with the assumptions of 5, results in a variation of charging efficiency with actual charging time as indicated in Figure 4.3-10. (Test data indicates that these voltage assumptions are reasonable.)

b. Shielding

The radioisotope thermoelectric generator has been located on the aft cover of the Lander to minimize the problem of shielding it from the sensitive electronic components. By placing the RTG at a distance of 28.6 inches from the nearest electronics during powered flight and then moving it on the aft cover to approximately 90 inches from this electronics package, it was determined that no shielding would be required other than the self-shielding provided by the generator. The total dose at the end of 10 days on the pad, 225 days in flight (a maximum value) and 180 days in the Martian surface yielded a dose of 10^{12} n/cm^2 to the most exposed electronics.

The calculations were based on the dose rates given for Cm 244 in Reference 2. With this configuration, gamma irradiation did not pose a problem since the most highly exposed electronic components received a dose of 4.36×10^3r which provides a margin of safety of approximately 2 (per Reference 3). The electronics which are closest to the RTG after landing on the Martian surface were found to have a dose of approximately one-half the allowable dose.

This technique of moving the RTG after landing provided a saving of from 40 to 50 pounds over shielding the RTG to reduce the dose rate to an acceptable level.

c. Selection of Cm 244

Curium 244 was selected as the radioisotope to be used in the Lander power supplies because it will be more available than Plutonium 238 by 1970, because it provides a lighter system and because it provides a smaller system. Only 8 percent of the predicted accumulative supply of Cm 244 by 1970 would be required to supply 5 generators for the Bus/Lander while 37 percent of the available Pu 238 would be required.

Preliminary estimates indicate that a Curium 244 system would provide a weight saving of approximately 56 pounds over a Plutonium 238 system of comparable design.

TABLE 4.3-12. RADIOISOTOPE THERMOELECTRIC GENERATOR USED IN ORBITER POWER SUPPLY COMPARISON

Power Output Required at Load	600 Watts
Number of Generators	2
Power Output of Generator	316 Watts
Power Available at Load	300 Watts
Weight	93.2 pounds
Distance Across Flats	9.1 inches
Height	24 inches
Fin Length	8 inches
Generator Efficiency	5 %
Isotope	Cm 244
Thermoelectric Material	Ge/Si
Void Volume	100%
Hot Junction Temperature	1500°F
Cold Junction Temperature	600°F

Using the dose rate values given in Reference 2 for Cm 244 at 50 Cm from the edge of the fuel slug, it was determined that no shielding would be required to protect the electronics on the Mars orbiter. The calculated dose rates and the allowable dose rates are shown in Table 4.3-13.

TABLE 4.3-13. DOSE RATES FOR ORBITER RTG

Radiation Type	Calculated Dose	Allowable Dose	Reference
Neutron	0.923×10^{12} n/cm^2	10^{12} n/cm^2	Transistorized circuits per Reference 2
Gamma	2.73×10^{3} r	10^{4} r	Surface effects threshold per Reference 3

3. Solar Cell System Design

The results of the solar cell system design used in the comparison are presented in Section 4.3.2 B. A description of the specific parameters used in sizing the array are presented below.

a. Efficiency

An air mass zero efficiency of 11 percent was assumed for a bare cell at 85°F. This is considered to be a reasonable assumption for the delivery period involved. MSD has measured the efficiencies of several N/P cells from one vendor at 10.5 percent, and these cells were mechanical rejects with no particular attention paid to trying to select high efficiency. The same vendor has submitted price and delivery estimates to MSD within the past few months covering a range of air mass zero efficiencies from 9 percent to 11 percent in quantities up to several hundred thousand. They indicate deliveries in the tens of thousands per month are obtainable for the 11 percent cells beginning about six months after receipt of an order. An indication of the yield of these higher efficiency cells is provided by the fact that the estimated price for an 11 percent cell is about 50 percent greater than that for a 9 percent cell. As an additional item, recent performance estimates of the Nimbus photovoltaic panel indicate they are based on a cell efficiency of about 11 percent.

b. Manufacturing Loss Factor and Ultraviolet Degradation Factors

Values of 0.97 and 0.95, respectively have been assumed. These are based on past experience by MSD. The former covers losses incurred in soldering, etc., during manufacture. The latter covers an observed decrease in output shortly after exposing the cell cover-glass combination to sunlight. This has been attributed to a decrease in the transmission properties of the filter due to exposure to ultraviolet. It has been found that the bulk of this effect occurs during the first 20 hours in sunlight, either in vacuum or in the atmosphere. No significant further deterioration is experienced after the first 20 hours. Testing has confirmed this conclusion for periods up to a simulated 4.3 years of sunlight exposure. Investigation into the detailed mechanism of this effect indicates there is a possibility of eliminating it by proper treatment of the filter. If this proves to be the case, this loss factor can be eliminated.

c. Meteorite Loss Factor

Measurements made at MSD indicate the maximum degradation from micrometeorites to be five percent. Solar cell-filter composites were prepared, their output measured, and then they were thoroughly sandblasted using a fine abrasive. Measured output after sandblasting showed a 4.5 percent reduction in short circuit current and a five percent reduction in current at the maximum power voltage.

d. Packing Factor

A ratio of active cell area (1.9 cm^2 for a 1 x 2 cm cell) to panel area of 0.9 has been assumed. This is reasonable for this type of design based on past experience.

e. Temperature Degradation

A degradation factor of -0.26 percent per degree F temperature rise above 85^oF has been assumed. This is based on measured data of the aforementioned 10.5 percent efficient N/P cells.

f. Radiation Degradation Factors

The radiation degradation factors assumed in this study are given in Figure 4.3-12. The primary source of damage is expected to be protons due to solar flares. The effect of the unknown trapped radiation environment, if any, in the vicinity of Mars or Venus is assumed to be negligible with respect to these solar protons. The damage due to the latter may be quite severe, inasmuch as the 1971 Mars mission occurs close to the time of the next expected peak of solar sunspot activity, these peaks occurring about every 11 years.

For purposes of estimating radiation damage for the 1969 through 1972 missions, the solar proton integral flux per year at Earth was assumed to be as indicated in Figure 4.3-12. This total dose and spectrum corresponds approximately to the occurrence during the vehicle lifetime of approximately two flares like that which occurred in May of 1959. This is the same environment that was specified by NASA Ames for use by those contractors who recently submitted studies of a 1967 Solar Probe. Since the solar sunspot maximum year is expected to be about 1968, it was felt that the difference in launch dates for the Solar Probe mission and the 1969-1972 Voyager missions would not seriously affect the basis for using this environment.

For purposes of comparison, four additional radiation environment curves are shown on Figure 4.3-12, based on data taken during 1956-61 (Reference 4). This period covers the last maximum in solar flare activity. These curves are as follows.

1. Average yearly dose rate over the six year period.

2. Total dose rate in each of the years 1959 and 1960, which had the greatest total doses during the period.

3. An assumed curve equal to ten times the average yearly doses rate.

For these last four curves, the straight line variation with proton energy is an assumption, but one which is believed to be conservative. Actually, data are only available for the 30 Mev and the 100 Mev values and are so indicated.

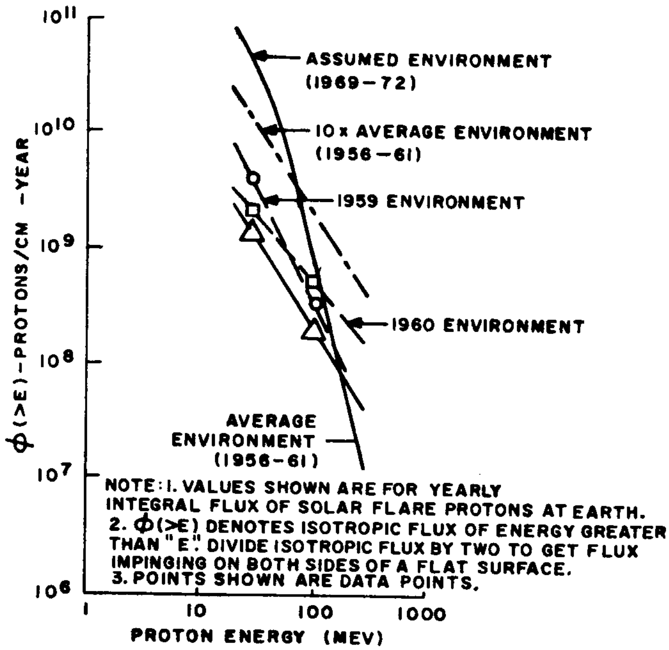


Figure 4.3-12. Solar Flare Proton Environment

In comparing the various curves of Figure 4.3-12, the following points are useful to keep in mind:

1. A cover glass thickness of 6 mils will stop all protons below about 4 Mev, so the portions of the curves less than this energy have little significance.

2. The damage to solar cells from protons decreases continuously as energy level increases. This, in combination with the reduced flux at the higher energy levels, makes the proton flux above a few hundred Mev, for the levels indicated on these curves, a minor factor in the damage.

3. There is some reason to believe that the solar flare activity during the next peak period will not be as great as during the last one. (See Reference 5.)

From the foregoing, it is believed that the protons environment assumed for this study is a fairly reasonable one and possibly may be somewhat conservative.

In order to convert the radiation environment at Earth to that expected for vehicles having varying distance from the sun, one procedure is to assume that proton flux varies inversely as the square of the distance from the Sun and to time-average this effect over the mission ignoring its discrete nature. This was done for the Mars 1969 mission, but it was found that the resulting degradation factors was sufficiently close to that obtained using the total flux for one year at Earth that the latter value was used for simplicity and is conservative.

Damage calculations were carried out using a computer program which has been developed for this purpose by this Company.

Values of the resulting degradation factors as a function of thickness of fused silica cover glass are shown in Figure 4.3-13. These are given for the assumed 1969-72 environment as well as for the average yearly dose rate during the 1956-61 period and ten times that average. Using the assumed environment, a weight optimization study indicated minimum solar array weight would occur with about 6 mils of cover glass, and this is the value used in this study, with a resulting degradation factor for the 1969-72 period of 0.78. However, for ease of handling, an actual design might well use a somewhat thicker glass, possibly 10 mils, with little weight penalty.

g. Filter Characteristics

Studies carried out in Reference 6 indicate that maximum array output for a solar paddle in the vicinity of Mars is obtained using no filter. Since early studies of the Mars orbiter assumed paddles, initial array output calculations were based on this assumption, with the resultant characteristics indicated in Table 4.3-6. As the design evolved to body mounted cells, these characteristics were not changed. However, a final design might very well use a blue or blue-red filter for two reasons. First, output for body mounted cells would be increased somewhat, perhaps as much as 8 percent. Second, at least a blue filter might be required to prevent deterioration of the glass-to-cell bond due to ultraviolet.

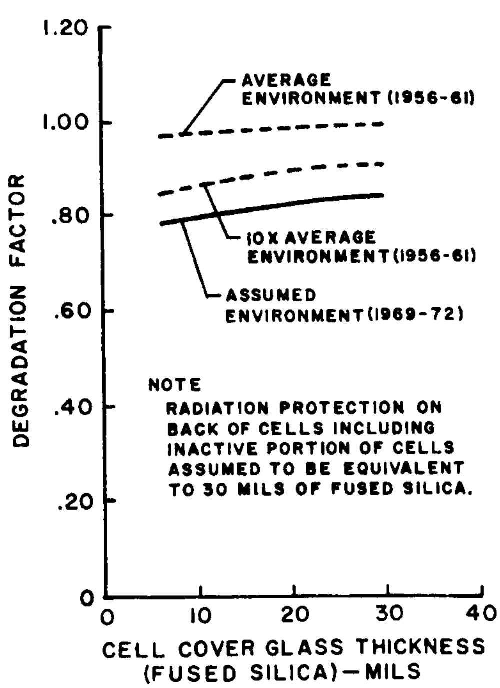


Figure 4.3-13. Solar Cell Radiation Factors Due to Solar Flare Proton

4. RTG Versus Solar Cell Trade-Off

The systems studies presented in this section and in Section 4.3.2.B indicate that either the RTG or the solar array-battery system could be used as the orbiter power supply. It is estimated that the RTG power supply would weight approximately 40 pounds less than the solar array system. However, the total weight differences for an Orbiter system must be considered before a selection can be made.

The reasons for selection of the solar array-battery system are presented in detail in Section 3.3.

C. LANDER/ORBITER

Based on the results of the Bus/Lander and the Orbiter systems studies, a radioisotope thermoelectric generator with a rechargeable nickel cadmium battery was chosen for the lander and a solar array-battery ststem was chosen for the orbiter. The resulting power supplies were presented in Section 4.3.2.C. The performance characteristics of the solar array and battery are presented in Sections 4.3.3.A and 4.3.3.B.

4.3.4 REFERENCES FOR SECTION 4.3

1. Radioisotope — Fueled Generator Compendium and Parametric Study, MND-2994, Martin Company Nuclear Division, June 1963 (Classified — Confidential, Restricted Data).
2. J. P. Nichols and E. D. Arnold, "Shielding Isotopic Power Sources for Space Missions", Nucleonics, Volume 22, Number 2, February 1964.
3. Personal communication with J. C. Peden, Consulting Engineer, Radiation Damage Effects.
4. W. R. Webber, "Solar Flare Proton Data", Nucleonics, Volume 21, pp 61 - 65, March 1963.
5. C. M. Minnus, "An Estimate of the Peak Sunspot Number in 1968", J. Atm. Terr. Phys., Vol. 20, pp 94 - 99, 1961.
6. J. K. Baker, "Temperature Control Technique for Solar Energy Converters", ASD-TR 61-689, Contract AF 33(616)-7889, General Electric Company, December 1961.

4.4 PROPULSION

Five separate propulsion systems are required for the Bus/Lander and Orbiter/Lander, and two for the Orbiter. A summary of parameters 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.

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 used; 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.

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	Monopropellant	Hydrazine	11,200	230	50	88.2	48.9
ΔV 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	.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	1,925(3)	1,634
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
ΔV 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

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 centered 45° from the support centerline; system specific impulse drops to 160 seconds.

4.4.1 BUS/LANDER

A. IN-TRANSIT ENGINE

A hydrazine monopropellant system is selected for this application. The system utilizes a catalytic start thrust chamber at a thrust level of 50 pounds. A butyl bladder is used. Nitrogen is used for pressurization. The Mariner jet vane system is used. All controls are redundant, and solenoid valves are used in on-off applications. The propellant weight given is sufficient to import a 100-foot per second ΔV to a maximum booster capability Lander. This represents 13 pounds of reserve fuel for the 1971 Bus/Lander.

1. Requirements

The primary requirement for the In-Transit Engine is to impart a total ΔV of 100 feet per second to the Bus/Lander over a number of firing cycles not to exceed six. Allowable maximum Bus/Lander weight, including propulsion, is about 3600 pounds. Thrust level is not critical; levels from 1000 pounds down to less than one pound would be acceptable. Repeatability of pulses should be such that the inaccuracy of ΔV imparted, due to propulsion, is less than one fps. Response time is not critical. Exhaust products should be compatible both with the spacecraft and the scientific mission. Power is not critical, since peaking is not a problem during periods of firing, and total on time is very short. Weight should be minimum consistent with high reliability and low development risk.

2. Analysis and Design

Of the various systems that could be considered for this mission, four could be considered to be state-of-the-art in 1965. These are cold gas, monopropellant peroxide, monopropellant hydrazine, and one of the present earth storable bi-propellant systems. Solids cannot be seriously considered, since the required total impulse per firing cannot be determined prior to launch. The four systems can be easily compared on a weight basis, and such a comparison is given in Table 4.4-2. The specific impulse of 300

TABLE 4.4-2. TRANSIT ENGINE WEIGHT COMPARISON

	Cold Freon-14 Gas	Peroxide Monopropellant	Catalytic-Start Hydrazine Monopropellant	Bi-Propellant (Radiation Cooled)
	Weight in Pounds			
Thrust Chamber	0.2	2.5	2.5	2.0
Propellant	233.0	70.0	48.9	37.5
Nitrogen	---	3.0	2.1	1.6
Propellant Tank and Bladder	70.0	6.2	4.4	4.5
Pressurization Tank	---	5.8	4.0	3.1
Controls and Piping	7.2	17.3	17.3	22.9
TOTAL	310.4	104.8	79.2	71.6

seconds chosen for the bi-propellant is still moderately high, even for 1965. An ablative bi-propellant system would be even higher than the radiation chamber shown.

From the above, it can be seen that the cold gas system must be eliminated from a weight standpoint. The other three systems can be considered further.

From a development standpoint, only the peroxide system can be considered fully developed. A number have flown on Mercury and Scout, and capability of long term space storage has been demonstrated by Syncom. Hydrazine systems have flown successfully on Ranger/Mariner and others, but these utilized an oxidizer slug start instead of the catalytic-start chamber. Considerable work is being done on the catalyst, however, and this system can probably be considered to be state-of-the-art by 1965. A number of bi-propellant systems in this thrust range are in development but, at the present time, none are achieving 300 seconds reliably. It is believed, however, that by the end of 1965, 300 second engines will be operating.

The monopropellant hydrazine system has a very definite advantage from the standpoint of propellant stability. Hydrazine has been stored for years in sealed containers without appreciable pressure buildup. Ranger/Mariner flights were conducted without relief valves. Butyl bladders can be used. With an Earth storable bi-propellant system, the oxidizer, whether N_2O_4 or MON, presents a bladder compatibility problem. Presently available elastomers cannot be used, and the use of compatible Teflon bladders results in excessive permeation, and low resistance to mechanical failure. Metal bladders or diaphragms, or bellows are not comparable, from either a weight or performance standpoint, with a Butyl bladder, for propellants which are compatible with Butyl. Peroxide can be stored safely if proper cleanliness precautions are taken, although venting provisions are required. Flexible bladders can be used with peroxide.

A bi-propellant system is inherently more complex than a catalytic-start monopropellant system. One additional explusion system is required as well as an additional set of propellant valving and means to separate the pressurant in the two propellant tanks.

A bi-propellant system is more susceptible to system ΔP changes caused by temperature changes, filter clogging, valve malfunction, injector heating, etc. This ΔP change in a bi-propellant system causes premature exhaustion of one of the propellants with accompanying loss of performance due to the unburned propellant. Such ΔP change has little effect on a monopropellant system.

A major advantage of a peroxide or hyrazine monopropellant system is the relative ease of thrust vector control. With these systems, the temperature is sufficiently low such that jet vanes can be used. Earth-storable bi-propellants operating at a specific impulse of 300 seconds require a more complex means of thrust vector control, such as secondary injection, gimballing, or auxiliary jets, with attendant complexity and weight penalties.

The greatest single advantage of a peroxide or hydrazine monopropellant system is the lack of susceptibility to chamber burnout. Current earth storable bi-propellant systems operating at 300 seconds specific impulse are extremely sensitive to hot spots caused by off-design operation of the injector. These bi-propellant systems normally have a cool film of gas at the wall, and breakthrough of this film by the hot core gases, which can be caused by a number of things, can easily cause wall burn-through. The temperature of the gases in a hydrogen peroxide or hydrazine system, however, are sufficiently low such that a homogeneous gas can be contained with no probability of burn-through.

Based on the advantages noted above which are summarized in Table 4.4-3, it is felt that a slight weight penalty should be taken in order that a monopropellant can be used. Although the hydrazine system represents a slightly higher development risk than the peroxide system, it is chosen since the risk is still small and a weight advantage of over 25 pounds can be realized.

In the event the catalytic start system is not developed in time, the N_2O_4 slug-start system could be incorporated. Six slugs with accompanying valves would weigh 5.4 pounds. This would be reduced if the number of starts required decreases, which is probable.

A thrust level of 50 pounds is chosen in order to take advantage of Ranger/Mariner experience, and to use the Ranger/Mariner jet vane assembly. The controls are changed considerably to provide multistart capability and to provide additional redundancy. The controls are shown in Figure 4.4-1, and are patterned after the Orbiter control system. Only a double failure or a structural failure will cause failure of the system.

Nitrogen is chosen for the pressurant gas based on its use on Mariner. This represents an increase in weight of about two pounds over a helium system.

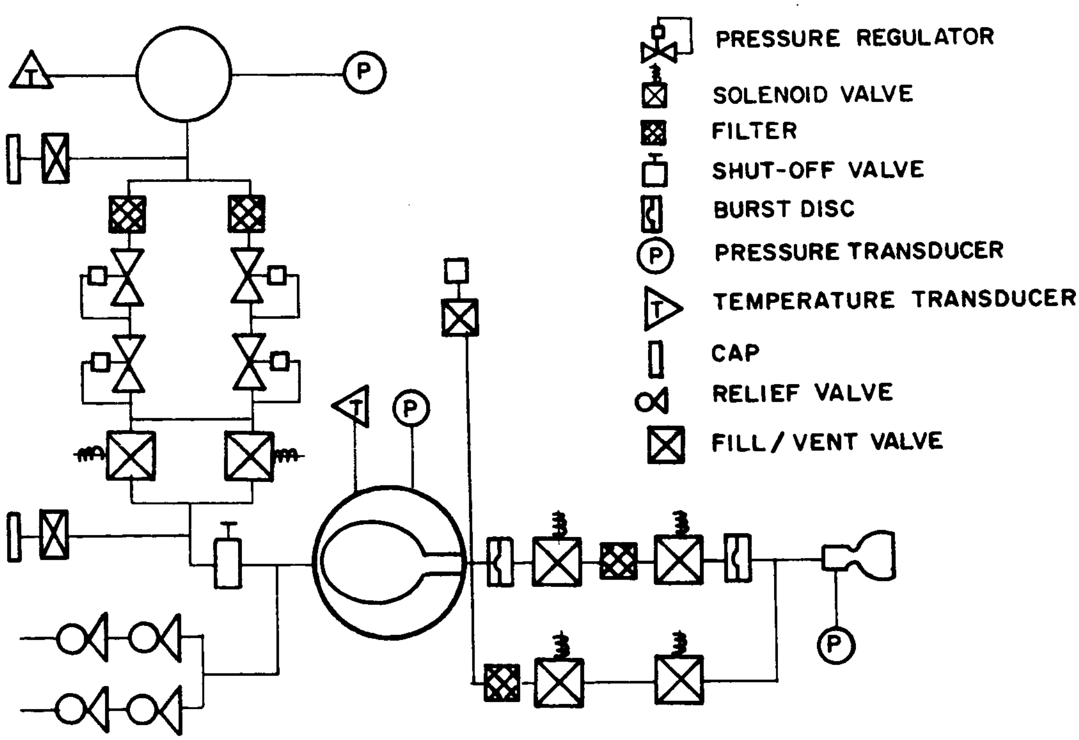


Figure 4.4-1. Bus/Lander In-Transit Propulsion System Schematic

TABLE 4.4.-3. COMPARISON OF CANDIDATE IN-TRANSIT PROPULSION SYSTEMS

	Cold Freon Gas	Peroxide Monopropellant	Hydrazine Monopropellant Catalytic Start	Bi-Propellant (Radiation Cooled)
Weight of System	310.4	104.8	76.7	71.6
Development Risk - Index		Best	Good	Fair
Expulsion Device - Index		Good	Best	Poor
Complexity		Least Complex		Most Complex
Susceptibility to ΔP Change		Little Effect		Performance Loss
Thrust Vector Control		Simplest		Most Complex
Chamber Burn-Through Susceptibility		Not Susceptible		Most Susceptible

Another system which was considered, but was rejected based on lack of development experience is the hydrazine blow-down system. Such a system represents a potential reliability improvement, and should be considered if development work is undertaken, and results are positive. The main disadvantage of a blow-down system when bi-propellants are used is the change in combustion efficiency as injector ΔV drops. With a monopropellant hydrazine system, this will not be a problem, although ammonia decomposition in the catalyst bed might degrade performance slightly. One propulsion company has indicated that a chamber pressure of 50 psia is feasible, and that a system ΔP of 120 psia is also feasible. Using these pressures with the present system configuration, would result in a system weight drop from 75.5 pounds to 72.6 pounds. If, however, a blow-down system were utilized, and the lower pressures shown were the final pressures, a number of components could be eliminated, and weight would drop from 75.5 pounds to 66.3 pounds if performance were not affected. The decreased system complexity can be seen by comparing Figures 4.4-1 and 4.4-2.

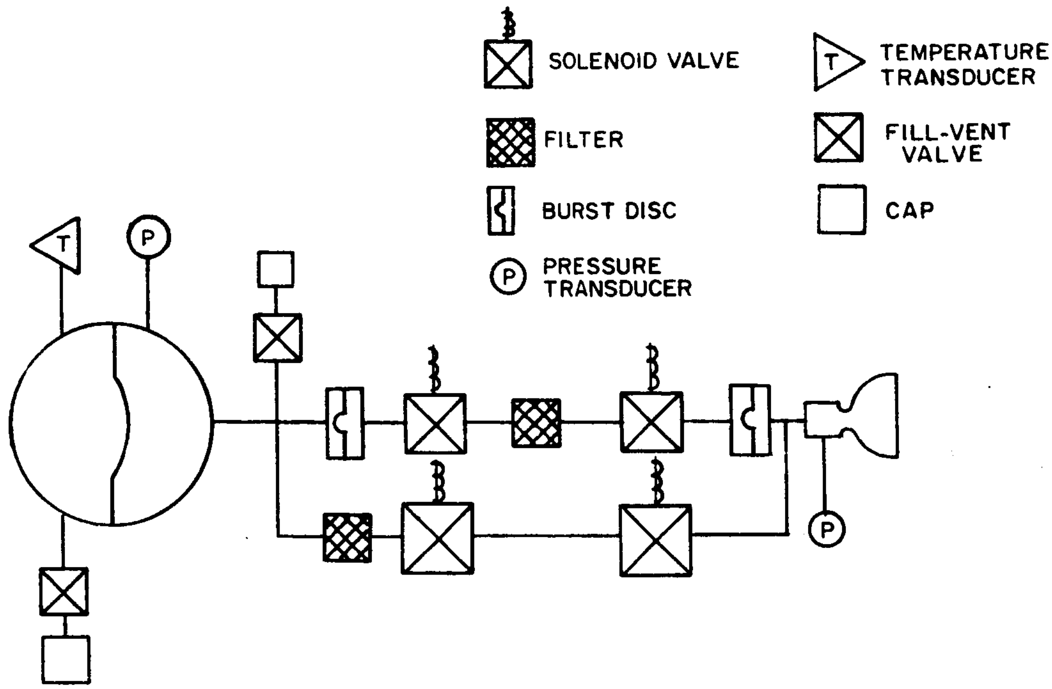


Figure 4.4-2. Bus/Lander In-Transit Blow Down System Schematic

3. Power, Weight, Size

Since this engine will be fired for only about 230 seconds, and peak power is not a problem at this time, relatively high power solenoids can be used. A total of 50 watts is assumed during the firing cycle.

Weight of the system, including residual propellant, jet vane system, brackets and harness is given in Table 4.4-4.

Hydrazine tank outside diameter is 14.2 inches. Gas tank diameter is 8.1 inches. Thrust chamber length is 10-1/4 inches; diameter is 2-1/2 inches. Total volume of other components is less than 0.2 cubic foot.

B. <u>ΔV MOTOR</u>

The ΔV motor is a spherical motor with steel case, and is heat sterilizable. No specifice propellant is selected. A propellant specific impulse of 230 seconds is assumed.

TABLE 4.4-4. IN-TRANSIT PROPULSION SYSTEM WEIGHTS

N_2H_4 Tank	3.4
Thrust Chamber	2.5
Residual propellant	4.0
Insulation	.1
Fill Valve	.2
Propellant Valves (4)	2.0
N_2 Pressure Transducers	.1
N_2H_4 Pressure Transducers	.3
N_2 Sensors	.1
Harness	1.3
Lines, Fittings and Manifold	3.2
Brackets	4.0
N_2H_4 Temp. Sensors	.1
Jet Vane System	2.3
Chamber Pressure Transducer	.3
Bladder	1.0
Burst Discs (2)	.2
Filters (2)	.4
N_2 Relief Valve (4)	1.0
N_2 Hand Valve	.3
N_2 Solenoid Valve (2)	1.0
N_2 Regulators (4)	4.8
N_2 Filters (2)	.2
N_2 Fill Valves (2)	.4
N_2 Tank	4.0
Gas (N_2)	2.1
Usable propellant	48.9
Total Weight	88.2

Expansion ratio is 33, chamber pressure is 1000 psia, thrust is 1900 pounds, and dual pyrogens are used.

1. Requirements

After separation from the bus, a ΔV of 300 feet per second must be imparted to the lander. Lander weight at this time is 2042 pounds. The propulsion system to accomplish this must be heat sterilizable, and should have a thrust vector sufficiently defined such that spinning will be adequate for stabilization.

2. Analysis and Design

With the low ΔV and relatively small amount of total impulse required, both solid and packaged liquid engines could be utilized within a reasonable weight allowance, neither solid nor liquid heat sterilizable systems have been developed, however, and an extensive development program would be required to bring either system to flight readiness. For a given development expenditure, it is felt that the solid motor would be more reliable. For this reason, a solid motor is selected for this application.

Work is continuing on heat-sterilizable propellants within the industry, but at a very low level of effort. Propellants which have been considered were discussed in the classified portion of the Saturn 1 report, and will not be mentioned here. The biggest problem in a sterilizable motor is the interface between the propellant and case, rather than in the propellant itself. Propellants which have been suggested range between 230 seconds and 290 seconds in specific impulse. These two values represent a difference of only 14 pounds in motor weight. A specific impulse value of 230 seconds is assumed, so that the greatest number of candidate propellants can be considered. No specific propellant is selected at this time.

The motor case is made of steel; the very small weight advantage of fiberglass or titanium does not justify the additional development required to use these materials. Expansion ratio is 33. Chamber pressure is 1000 psia. Dual sealed pyrogens will be used. Thrust is 1900 pounds; burning time is 10 seconds.

The effect of long term space storage on solid propellant motors cannot be adequately assessed at this time. Based on this uncertainty, a sealed system with frangible throat closure is utilized.

3. Power, Weight, Size

The only power required will be a 100-millisecond pulse to fire the pyrogen squibs, unless some additional thermal control is required.

The weight of the motor without mounting hardware is 94 pounds; motor diameter is 14.2 inches. Motor diameter and weights are shown in Figure 4.4.-3.

C. ATTITUDE CONTROL PROPULSION

Freon-14 cold gas is used as the propellant. Two independent systems are used, each supplying gas to a couple half. Three times as much gas as would be required for a normal mission is provided, so that no single failure will cause a mission failure. The propulsion system is sterilized internally, prior to installation in the spacecraft, and the propellant is sterilized prior to filling.

1. Requirements

With the exception of the amount of total impulse, and the thrust levels, the requirements given in paragraph 4.4.2.B are applicable also for the Bus/Lander. Total impulse required is 255 pound-seconds. Since the In-Transit Engine will utilize jet vanes, no provisions are necessary to offset induced roll; therefore, using a higher thrust level for the roll nozzles is not required. All thrust levels will be 0.01 pound.

2. Analysis and Design

With the very small amount of gas required, the hardware becomes a significant percentage of the total weight. This makes the use of a different type of redundancy, which is used on Ranger and Mariner vehicles, more attractive. With this arrangement, two separate systems are used, but they are not interconnected. Redundant components within each of the two systems are not used; i.e., shutoff valves and additional regulators used in the Orbiter Attitude Control Propulsion System are not required. This is illustrated in Figure 4.4-4, and can be compared with the Orbiter system in Figure 4.4-3. The amount of gas carried is three times the amount required for a standard mission, carried in two tanks. The reason for this can be illustrated by assuming that a (+) pitch nozzle valve fails in the open position. As the gas in the tank is expelled, both (-) pitch nozzles operate to maintain a stable position. If it is

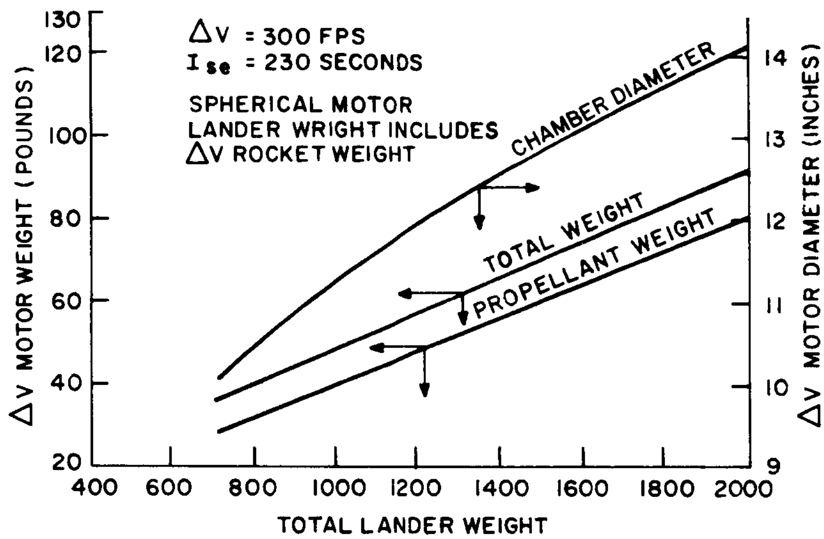


Figure 4.4-3. Total Lander Weight vs ΔV Motor Weights and Diameter

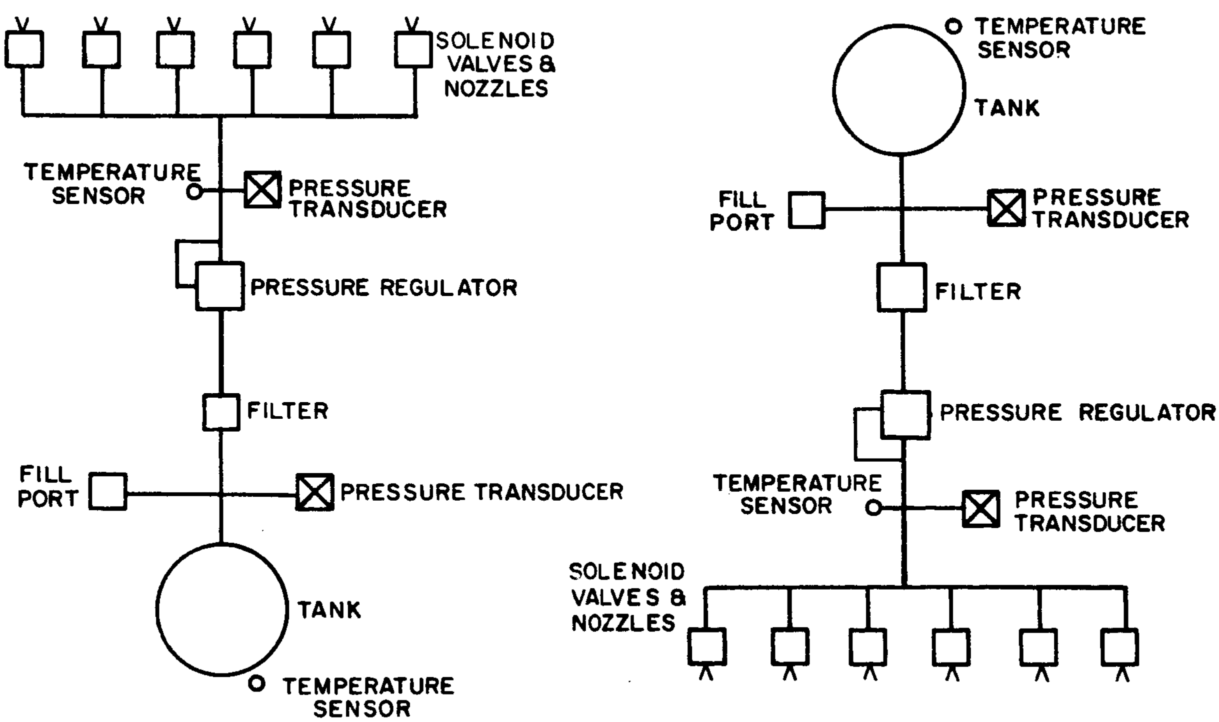


Figure 4.4-4. Bus/Lander Attitude Control System Schematic

assumed that the efficiency of all nozzles is the same, then the amount of gas from each (-) nozzle is equal to 1/2 of the amount of gas from the leaking nozzle. The total gas lost, therefore, is all the gas from the leaky system, 2/3 through the (+) pitch nozzle, and 1/3 through the (-) pitch nozzle, and 1/3 of the gas from the non-leaking system, through the (-) pitch nozzle. This leaves 2/3 of the gas in a single tank, which is 1/3 of the total carried. This is, of course, the amount required for a normal mission. If the valve were not stuck open, but were only leaking, the pressure at the nozzle would be less, which would result in lower efficiency and a lower specific impulse. The amount of gas necessary to overcome the torque would be less, so the amount of loss from the non-leaking system would be even less than 1/3. A comparison of weights of this system, and the system type used for the Orbiter Attitude Control is given in Table 4.4-5. Also shown, for purposes of comparison, is a nitrogen system.

TABLE 4.4-5. BUS/LANDER ATTITUDE CONTROL SYSTEM WEIGHTS

	Mariner-Type System Freon-14	Orbiter-Type System Freon-14	Mariner-Type System Nitrogen
	Weight in Pounds		
Gas	16.5	5.5	11.4
Tanks (2)	6.4	3.1	17.0
Fill Valve	-	.1	-
Fill Valves (2)	.2	-	.2
Filters (2)	.8	.8	.8
Pressure Regulators (2 dual)	-	6.2	-
Pressure Regulators (2 single)	5.8	-	5.8
Shut-Off Valves (4)	-	5.0	-
Solenoid Valves	5.2	5.2	5.2
Nozzles (12)	1.2	1.2	1.2
Tubing	2.8	2.8	2.8
Pressure Transducers (4)	1.0	1.0	1.0
Temperature Sensors (4)	1.0	1.0	1.0
Latch Valve	-	1.8	-
Total	40.9	33.7	46.4

As can be seen in Table 4.4-5, the Mariner-type system, carrying three times the amount of gas as the Orbiter-type system, is only 7.2 pounds heavier, since the latch valve, and shut-off valves are not required, and single rather than dual regulators can be used. A savings of 5.5 pounds can be realized by using Freon-14 instead of Nitrogen.

The system and gas will be sterilized the same as with the Orbiter Attitude Control Propulsion System. The probability of contamination with this spacecraft is, of course, much less than with an Orbiter, but since a study to determine probability of contamination with an unsterilized system has not been done, it will be assumed that sterilization is required.

The system for later opportunities is not expected to differ from this system.

3. Power, Weight, Size

Assuming a power requirement of 6 watts per valve, and an on-time/off-time ratio of 0.0003, the average power consumption during the mission is 0.002 watt.

Total weight is 40.9 pounds. This is shown in detail in the preceding section.

The diameter of each tank is 9.0 inches. Total volume of other components is less than 0.1 cubic feet.

No changes in power, weight, or size would be required for the later opportunities.

D. SPIN SYSTEM

The spin system selected utilizes cold nitrogen gas as the propellant. Hermetically sealed dual tanks are used, each dumping into a common spin manifold through its own squib valve. Proper operation of both systems will result in a spin rate of 60 rpm, which is optimum, although failure of one system will result in a spin rate of 30 rpm which will give a high probability of successful entry. Tanks are designed to give a safety factor of 2.0 while at sterilization temperature.

The single lander configuration has more weight capbility than is needed, so the use of a cold gas system with a 2.0 safety factor is clearly indicated.

The weight figures given in this study are based on a moment of inertia of 675 slug-ft^2. A more precise calculation after final design showed the actual moment of inertia to be

367 slug-ft^2, but an additional iteration reflecting these new numbers was not accomplished. Therefore, the weights given herein could be reduced in proportion to the moment of inertia reduction.

1. Requirements

The Lander spin system is required to spin stabilize the lander immediately after separation from the bus in order to negate potential velocity vector errors which would be caused by angular tip off rates initiated at separation and by the ΔV rocket thrust vector misalignment. A spin rate of 60 rpm is required.

2. Analysis and Design

a. Solid Spin Rockets

A minimum installation for solid spin rockets would consist of 2 solid rockets mounted on the outer periphery of the lander, and the thermal control equipment necessary to keep the rocket temperature within design limits. The environmental control might be passive but would probably have to be active. The necessity for power switching would decrease the reliability somewhat, but probably not appreciably. Whether passive or active, some additional weight would be required for the thermal control. Overall weight of a candidate system is estimated to be 12.5 pounds. This assumes a propellant specific impulse of 230 seconds, which is consistent with the specific impulses assumed for other Lander engines. For the 989 pound-seconds impulse required to accelerate the Lander, 2.2 pounds of propellant in each of two motors would be required. A propellant mass factor of 0.4 is assumed. This is slightly higher than for the units of the same general size in use on present space missions. Assuming a value of 1-1/2 pounds for the thermal control system results in a total weight for this candidate system of 12.5 pounds.

One serious disadvantage of this system is that the sterilizable motors would have to be developed, and at considerable expense. Although a sterilizable propellant must be developed regardless, for use on the retardation motor and the Lander ΔV motor, the biggest development effort for a heat sterilizable motor is probably not in the propellant itself, but in the grain case interface. Therefore, a separate development is required for each different rocket motor design.

Another serious disadvantage to this system is the effect of non-simultaneous firing of the two solid rockets. A plane through the two motors, perpendicular to the spin axis, is considerably aft of the C. G. If one motor developed an ignition delay, the other motor could cause a considerable angular displacement of the lander, which is precisely what the spin is meant to prevent. If one motor burned appreciably faster than the other, the same error could result. The probability of such an occurrence would be much higher than with non-sterilizable, propellant, since the experience with the sterilized propellant would be much less than with the conventional propellants.

b. Solid Gas Generator

The primary disadvantage of the solid spin motors can be eliminated by using a centralized solid gas generator connected to two nozzles. This automatically eliminates the problems caused by ignition delay or mis-matched thrust. Thermal control becomes easier, since the solid grain can be centrally located inside the bus. Weight is estimated to be 18.8 pounds, with dual igniter systems, which is somewhat higher than the spin motors. As with the solid spin motors, development costs for a sterilizable unit would be quite high.

c. Cold Gas Spin System

A cold gas spin system has only one serious disadvantage, that of weight. This weight problem is compounded in a heat-sterilizable system, since the tanks must take the increased stress caused by the increased pressure during sterilization. The candidate system selected weighs 47.9 pounds. There are many redeeming factors, however. Development of a heat-sterilizable gas system is a much lesser problem. Only the pyrotechnic actuation device must be developed, and since a number of such devices are required in the system, it is probable that one of the others can be used in this application. No stringent thermal control is required. Only the actuation device is temperature sensitive, and this to a much lesser degree than a solid propellant. Also, this device is located within the lander and not outboard, making it doubtful that additional thermal control would be required.

Problems due to ignition delay are non-existent. Two nozzles are used and piped by identical piping to the squib valves. This assures that the thrust will be the same for each of the two nozzles.

The potential problem of gas leakage can be resolved by using systems which are hermetically sealed. A metal diaphragm is welded to the mouth of the bottle, and is pierced by a pyrotechnic-driven knife to actuate spin. The system can be weighed after vibration and thermal testing, and after an elapsed time of several months, to assure leak tightness of welds.

d. System Selected

Table 4.4-6 summarizes the advantages and disadvantages of the various systems. For a 1971 mission, weight is not at a premium for equipment which is not attached at entry. The choice of systems is, therefore, obviously cold gas.

Freon-14 and Nitrogen are the top contenders for the gas to be used. In normal operation when used in an attitude control system where temperature is near 70^{o}, and where stored in a tank with a safety factor of 2.0 at $70^{o}F$, a nitrogen system has a weight penalty of 24 percent over a Freon-14 system. For this application, however, the penalty is only 10 percent. The reason for this change is that the system must be designed for the sterilization temperature of $297^{o}F$, and the compressibility factors at this temperature are considerably different. This 10 percent is based on a safety factor of 1.5 during sterilization, which would necessitate some protection for personnel during the sterilization cycle. Raising this safety factor to 2.0 during sterilization, consistent with safety practices with personnel in the area would raise this penalty to 21 percent. Actual systems weights are shown in Table 4.4-6. As noted in the Table 4.4-6, some weight saving is possible by sterilizing an empty tank, then filling with a sterilized gas. Two major disadvantages exist; probability of contamination is much higher, and the probability of leakage is much higher since a hermetically sealed system cannot be used. It should be noted that all these systems have a factor of safety in excess of 2.0 when at ambient temperature.

From the foregoing, it would appear that, for a safety factor of 2.0, the Freon-14 system represents a weight advantage of eight pounds. However, these figures do not take into consideration the possible effects of a very rapid blow-down. Under such conditions, it is possible that as the blow-down progresses some of the Freon-14 may liquify, resulting in a performance loss. Because of this unknown, nitrogen is selected for this application. Since for the 1971 mission, there is no weight problem for systems which do not enter, a safety factor of 2.0 is selected. To increase reliability, two

TABLE 4.4-6. COMPARISON OF CANDIDATE SPIN SYSTEMS

	Solid Spin Motors	Solid Gas Generator	Cold Gas System Freon-14 - 2 Tanks Sterilization Safety Factor 2.0	Cold Gas System Nitrogen - 1 Tank Sterilization Safety Factor 2.0	Cold Gas System Nitrogen - 2 Tanks Yield at Sterilization; Safety Factor 1.2	Cold Gas System Nitrogen - 2 Tanks Sterilization Safety Factor 1.5	Cold Gas System Nitrogen - 2 Tanks Sterilization Safety Factor 2.0	Cold Gas System Nitrogen - 2 Tanks Filled with Sterile Gas After Sterilization
Weight	12.5 Pounds	18.8 Pounds	39.9 Pounds	45.9 Pounds	37.3 Pounds	40.5 Pounds	47.9 Pounds	37.3 Pounds
Development Cost	Very High	Very High	Very Low	Very Low	Very Low	Very Low	Very Low	Very Low
Thermal Control	Required	May Be Required	Probably Not Required	Probably Not Required	Probably Not Required	Probably Not Required	Probably Not Required	Probably Not Required
Ignition Delay	Could Cause Mission Failure	Could Not Cause Problem	Could Not Cause Problem	Could Not Cause Problem	Could Not Cause Problem	Could Not Cause Problem	Could Not Cause Problem	Could Not Cause Problem
Flight Reliability	Many Potential Failure Modes	High	Unknown - See Text	High	Very High	Very High	Very High	Less Than Other Cold Gas Systems
Personnel Protection Required During Sterilization	Yes	Yes	No	No	Yes	Yes	No	No
Probability of Successful Sterilization	High	High	High	High	High	High	High	Less Than Other Systems

tanks are used, each with a pyrotechnic valve. Both empty into a common manifold as shown below.

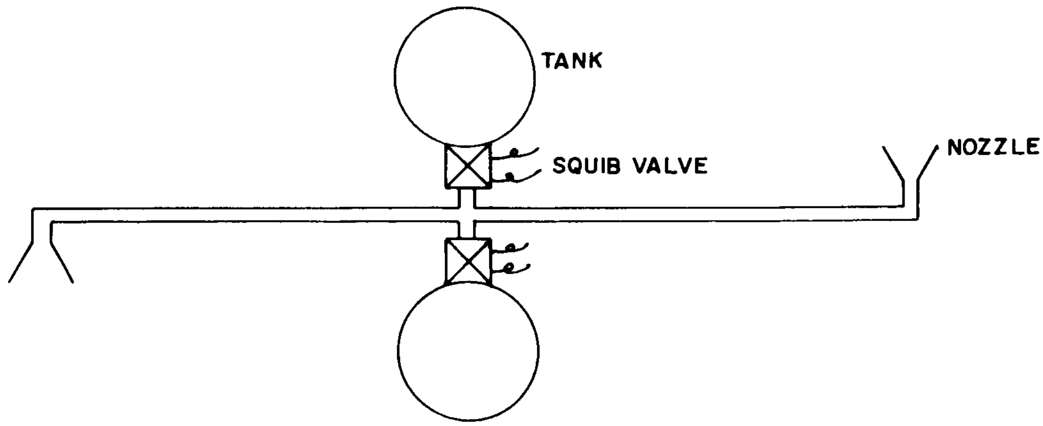


Figure 4.4-5. Cold Gas Spin System

If one valve fails to fire, a spin rate of 30 rpm will be attained instead of the 60 rpm desired. Even this spin rate, however, will give a high probability of successful entry. The thrust level will have no effect on the final spin rate, so no value is selected for this study.

Storage tank pressure selected is 2000 psi. This gives a pressure of 4700 psi during sterilization, which is within the pressure range of state-of-the-art titanium 6A14V pressure tanks.

3. Power, Weight, Size

Although doubtful, a small amount of power may be required for thermal control. The only additional power will be that necessary for about 100 milliseconds for firing the squibs. Outside diameter of each spherical tank is 12.0 inches. System weights are given in Table 4.4-7.

TABLE 4.4-7. COLD GAS SPIN SYSTEM WEIGHTS

Item	Weight (Pounds)
Tanks (2)	30.2
Nitrogen	13.4
Squib Valves (2)	1.5
Tubing	2.0
Nozzles and Fittings	.8
Total	47.9

E. RETARDATION MOTOR

The retardation motor is a heat sterilizable spherical solid motor with two nozzles canted at 45° from the support centerline. Propellant specific impulse is 230 seconds; motor specific impulse is therefore 160 seconds. Thrust level is about 2000 pounds; burn time is 3 seconds.

1. Requirements

The retardation propulsion system must impart a ΔV of 80 feet per second to the Lander immediately prior to impact. Lander weight at this time, including retardation system, is 1700 pounds. The propulsion system must be heat sterilizable, and hot gases must not impinge upon the lander suspended 75 feet below, nor upon the cable from which the Lander is suspended, in an eleven millibar atmosphere.

2. Analysis and Design

A solid motor, with a propellant specific impulse of 230 seconds is selected for the same reasons it was chosen for the ΔV motor, as discussed in Section 4.4.1.B.

Since Mars gravity is acting upon the lander at this time, the lander will be accelerated at the same time it is being decelerated. If the parachute were not exerting a drag, then the acceleration would be constant during the burning, as shown on the "zero drag parachute" line in Figure 4.4-6. Actually, the parachute will continue to exert drag in a decreasing amount during the burn, and this is shown on the "estimated drag parachute" line in the Figure 4.4-6.

In order to avoid pluming of the hot gases upon the lander or support cable, dual nozzles, canted 45° from the support centerline, must be used. Area ratio is 40:1. With the canted nozzles, effective specific impulse of the motor is 160 seconds.

Motor weights for terminal velocities from 30 feet per second to 500 feet per second are given in Figure 4.4-7. This takes into consideration gravity loss.

Burn time is approximately 3 seconds. Thrust is approximately 2000 pounds. Dual pyrogens are used, and, as with the ΔV rocket, a frangible throat closure is used.

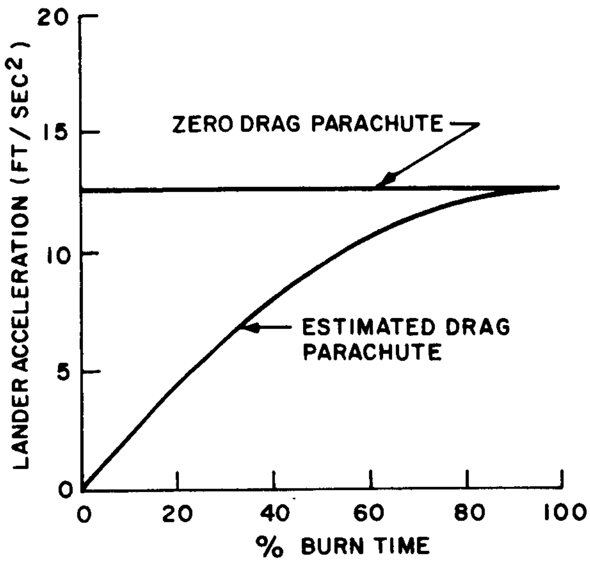


Figure 4.4-6. Lander Acceleration vs Retardation Motor Burn Time

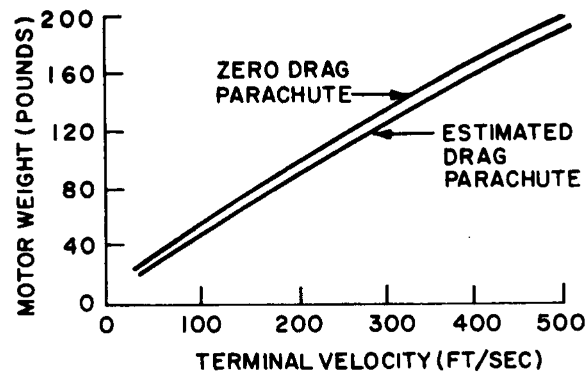


Figure 4.4-7. Retardation Motor Weight vs Terminal Velocity

3. Power, Weight, Size

A 100 millisecond pulse will be required for the pyrogen squibs; no additional power is required unless needed for thermal control.

The weight of the motor is 41 pounds. Motor diameter is 11 inches.

4.4.2 ORBITER

A. MAIN PROPULSION SYSTEM

The Orbiter Main Propulsion System is a bi-propellant system utilizing N_2O_4 and 50% N_2H_4/50% UDMH at a mixture ratio of 1.65. The system is pressure fed, utilizing helium regulated from a stored pressure of 3000 psia to 200 psia in the spherical partial-diaphragm propellant tanks. The ablative chamber with radiation skirt has a service life of 600 seconds, and produces 900 pounds thrust with a chamber pressure of 100 psia and an area ratio of 100:1. The thrust chamber specific impulse is 308 seconds. Redundancy is used extensively, and only a double failure or a structural failure will cause propulsion system failure.

1. Requirements

The Main Propulsion System must provide a ΔV of approximately 6400 feet per second to a 3413-pound total weight Orbiter for injection into orbit, and a ΔV of 100 feet per second for in-transit adjustments. Cutoff accuracy and minimum ΔV should be less than five feet per second. Provisions must be made to remove any disturbances to the spacecraft caused by the propulsion system. Vehicle acceleration must not exceed 2 g's. The development risk should be minimum and reliability maximum consistent with keeping weight to a value such that the overall mission can be accomplished.

2. Analysis and Design

In the classified volume of the Saturn 1 report, the selection of propellants was discussed. The "high energy" propellants mentioned therein would increase non-propulsive payload considerably less than 10 percent for the ΔV required for this mission. The reasons for the selection of N_2O_4/50-50 for the propellants are valid also for the Titan III, and these propellants are, therefore, selected.

Considerable work has been done on thrust termination of solid systems, but the many other problems, especially the high thrust required in order to obtain acceptable burn time, remain as discussed in the previous report.

a. <u>Pressurization System</u>

In the Saturn 1 Study, a number of pressurization systems were considered. These included pumped system, propellant injection system (direct tank injection), stored liquid system, solid cartridge pressurization, and heated and unheated stored gas systems. The system chosen was stored gas heated prior to orbit injection. The requirements for the Titan III Orbiter propulsion system are changed little from the Saturn 1 system, except for the amount of total impulse required. No significant changes, or predicted changes, in the area of pressurization systems have been noted since the Saturn 1 study. Therefore, the Saturn 1 pressurization system analysis is valid for the Titan III, and the same system chosen will be used on the Titan III, except as modified in size to reflect the smaller propellant supply.

The technique of heating the tank prior to orbit injection provides a considerable savings in tank weight with only a slight weight increase from added insulation and heater circuitry. On the launching pad, the tank safety factor is 2.0 at 70°F. As the vehicle travels from Earth to Mars, the vehicle temperature, and, therefore, gas temperature, drops due to the increasing sun distance. Heating the gas back up to its original takeoff temperature would permit less gas to be carried. A greater savings could be realized by heating the gas to an even higher temperature, provided valve seals were not affected and a reasonable unmanned safety factor was observed. Further advantage is that additional heat is available in the tank material to be transferred to the gas as the temperature drops during propellant explusion at orbit injection. Tests conducted in a thermal-vacuum chamber at Valley Forge Space Technology chamber since the Saturn 1 study indicate that a very considerable amount of heat transfer can be expected to take place during a 10-minute firing cycle. These tests were of course not entirely valid since the gravity field during actual firing would be only about 0.25, and convective heat transfer might be different; strategic placement and geometry of outlet fittings could probably be of even greater value in inducing heat transfer, however. For this design, an initial gas temperature of 170°F is selected as being compatible with valving. Heat is supplied gradually during low power demand times, over a period of hours or days. The tank is of course insulated to reduce heat loss during the heating cycle.

As with the Saturn 1, an initial pressure of 3000 psi, decaying to 300 psi, is chosen. For the Titan III Mars 71 tankage, a helium weight of 5.6 pounds is required. The weight of the titanium tank to contain it is 53.2 pounds. Outside diameter of the spherical tank is 20.0 inches.

b. Thrust Level Selection

In the Saturn 1 study, a common chamber design was utilized for both the Mars 69 and the Venus 70 missions, and since the Venus 70 total impulse requirement was much larger than for the Mars 69, a large weight penalty was taken on the Mars 69 thrust chamber. In this study, the chamber is optimized for the 1971 Mars opportunity.

In the Saturn 1 study, the various factors affecting selection of a thrust level were discussed. These are reviewed briefly below, except in cases where changed requirements affect the optimization, in which cases, these factors are treated more thoroughly.

(1) Effect of Thrust Level on Thrust Chamber Weight — With an ablative chamber, for a given total impulse, with all other parameters the same, a low thrust chamber is considerably lighter, as indicated in Figure 4.4-8. From the standpoint of reliability and development risk, however, there is a lower limit. There is little ablative experience at present, or projected for the immediate future, for chambers in the 500-to-10,000-pound thrust class operating in excess of ten minutes. For the Mars 71 mission, a total impulse of about 500,000 pound-seconds is required. For a 10-minute burn time, this corresponds to a thrust level between 800 and 900 pounds.

(2) Effect of Thrust Level on Gravity Loss — Although no additional computer runs were conducted on gravity loss during this study, previous runs indicate that the gravity loss for a 900-pound thrust chamber would be only about five pounds if a gravity turn or constant pitch rate control mode were utilized.

(3) Effect of Thrust Level on Specific Impulse — In the Saturn 1 study, it was noted that the thrust level could possibly have an effect on specific impulse due to the effect of gas stay time on kinetic loss. Empirical data obtained since this study indicates that these losses are more a function of chamber geometry than thrust level, for the thrust ranges of interest here.

(4) Effect of Thrust Level on Vehicle Minimum ΔV and Cutoff Accuracy — With a 900-pound thrust, a cutoff repeatability of less than 10 pound-seconds can be obtained. This is equivalent to about 0.1 feet per second. Minimum impulse could easily be held to 200 pound seconds, which is equivalent to about three feet per second. Both of the values are well within the system requirements.

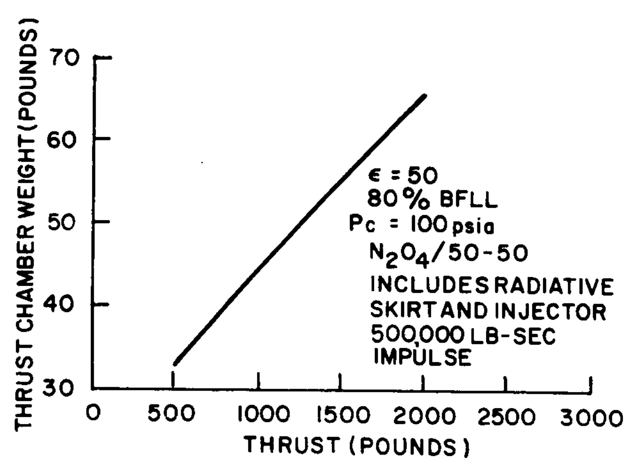


Figure 4.4-8. Ablative/Radiative Skirt Thrust Chamber Weight vs Thrust Level

(5) Effect of Thrust Level on Pressurant System Weight — Maximum firing time is desirable from the standpoint of maximizing heat transfer to the pressurizing gas. The effect on weight of the gas and tank is difficult to predict, but it would probably be less than five pounds, and could be less than one pound.

(6) Effect of Thrust Level on Valving Weight — This effect is small, and would be only about 15 pounds between thrusts of 500 pounds and 3500 pounds.

(7) Effect of Thrust Level on Heat Flux to Vehicle — Higher thrust levels on chambers with radiation skirts will result in higher vehicle heat fluxes; however, the radiation shield weight difference for thrusts between 900 and 2200 pounds is expected to be very small.

(8) Effect of Thrust Level on System Reliability — As noted in the Saturn 1 study, thrust level appears to have only a small effect on system reliability, except for the thrust chamber reliability. As noted before, thrust chambers with total firing times of more than 600 seconds for the thrust levels under consideration could not be considered reliable at the present time.

(9) Effect of Thrust Level on Structure Weight — Within the thrust ranges under consideration, the stresses imparted to the spacecraft are not severe, and are not the limiting factor in the design. Of considerable importance, however, is the effect of thrust chamber length on interstage structure weight, which weighs about 2.8 pounds per inch. From this standpoint, a low thrust level is desirable.

(10) Thrust Level Selection — Table 4.4-8 summarizes the relative advantages of low and high thrust for ablative chambers with radiative skirts. It can be seen that there are no significant advantages for running at a high thrust, but many disadvantages. For this reason, a thrust level of 900 pounds is selected as being the minimum value acceptable, consistent with firing time limitations.

TABLE 4.4-8. THRUST LEVEL COMPARISON, ABLATIVE CHAMBER WITH RADIATIVE SKIRT

Factor	Low Thrust Advantage	High Thrust Advantage
Firing Duration		Higher thrust allows shorter firing time with cooler walls and more predictable ablative process.
Chamber Weight	Significantly lighter	
Valving Weight	Lighter	
Vehicle Heat Flux	Minimum heat flux	
Reliability	Insignificant difference, except for thrust chamber as noted above	
Interstage Structure Weight	Significantly lighter	
Gravity Loss	Negligible above 900 pounds thrust.	
Specific Impulse	Data inconclusive to date	
Minimum ΔV and Cutoff Accuracy	No effect within range considered	
Pressurant System Weight	Minimum weight; effect small	

c. Chamber Type

The selection of the type of thrust chamber to be used was treated at length in the Voyager Saturn 1 study. The propulsion requirements for the Titan III are not sufficiently different from the Saturn 1 to require a change in the chamber type. An ablative chamber is, therefore, selected. The experience within the industry since the Voyager Saturn 1 report should, however, be noted. Considerable work has been done on radiative chambers, but the disadvantages noted in the previous study continue to exist. An additional problem has been recognized; that of extremely high peak pressures upon start, especially at low temperatures. Under certain conditions which have not been wholly ascertained, molybdenum chambers are fractured by the high pressures. Certain propulsion contractors claim that some of their development designs are not susceptible to this fracturing, however, until a very considerable amount of additional development is done this problem must be considered an additional deterrent to the use of radiative chambers.

Since the Saturn 1 report, ablative chambers have also had their share of problems. The most serious problem has been that of continuing ablation due to soak-back after shut-down. This was discussed briefly in the Saturn 1 report. This problem is most serious with attitude control engines, or other engines which are required to operate a number of times. For the Voyager Titan III application, the in-transit adjustment firings, which will number a maximum of six, will total only 12 seconds, and it is felt that the soak-back effect for these firings will not be large. Ninety-eight percent of the firing time will be at the final firing, and the soak-back effect of this firing can best be overcome by designing for a high outer wall temperature, and applying insulation between the ablative liner and the structural outer wall.

Another problem which has not been investigated sufficiently is the possible effects of long-term space environment on the subsequent performance of the thrust chamber. The Orbiter is designed so that the thrust chamber is in the vehicle shadow except during maneuvers, which should minimize the problem of chamber outgassing. Even with this precaution, however, a considerable unknown exists which cannot be resolved without extensive simulated space testing.

As noted in the previous report, it is strongly felt that a homogeneous combustion gas is not optimum for an ablative chamber, but that the injector must be designed for a relatively cool outer barrier gas, and a hot core.

Considerable development effort within the industry is being expended on combination types of chambers, especially those using regeneration cooling in addition to ablation or radiation. These combination types hold considerable promise, but even if their development is successful, they will not be state-of-the-art by 1965.

d. Chamber Pressure

Chamber pressure selected for the Titan III-C Voyager, as for the Saturn 1 Voyager, is 100 psia, and for the same reasons. The two primary reasons are (1) to reduce inner wall temperatures and (2) to take advantage of present development work, which is predominantly in the 100 psia category. A computer run was made, and showed that the overall propulsion system weight for a 100-psia system was three pounds more than for a 150 psia system. Interstage structure, which was not in the computer run, would add an additional 18 pounds, for a total savings of 21 pounds. Even with this weight penalty, however, the 100 psia pressure is selected in order to reduce development risk.

e. Area Ratio

The advantage of high performance with a high area ratio is partially offset by the additional weight required for the increased chamber length, the additional heat radiated to the spacecraft with the increased length, the increased moment of inertia which must be considered in the gimballing equipment, the additional cost, the increase in handling complexity of the thrust chamber and spacecraft, and the increase in interstage structure weight due to the increase in length. In the Saturn 1 study, one of the ground rules was that the interstage structure weight should not be considered. This resulted in an optimum area ratio, from a weight standpoint, of more than 100:1. For the Titan III study, the interstage structure weight is considered, and the area ratio optimizes at slightly less than 100:1 from a weight standpoint alone, as noted in Figure 4.4-9. It should be noted that the payload increment between area ratios of 60:1 and 120:1 is less than three pounds, so that small errors in the weight estimates for interstage structure and skirt could change very considerably the optimum area ratio. The difference in length between an 80:1 and a 100:1 chamber is about four inches. With many ablative chambers, especially those which are of long duration, the throat area increases during the firing, thus, gradually decreasing the area ratio. For purposes of this study, therefore, the area ratio of 100:1 selected for the Saturn 1 study, is used here also.

The optimum area ratio for later Mars opportunities will change somewhat due to the differences in propellant weight; for example, the 1977 opportunity, with the least propellant being utilized, optimizes at about 60:1 from a weight standpoint. However, the use of the same 100:1 thrust chamber would result in less than a 10-pound weight penalty; the same is true for the other opportunities.

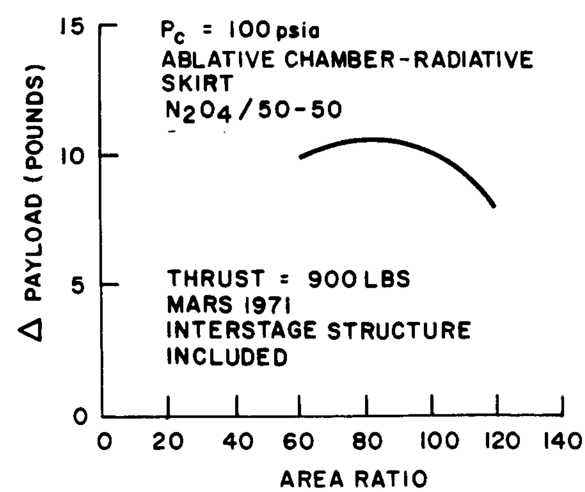


Figure 4.4-9. Payload Increment vs Area Ratio

f. Nozzle Contour

The weight of a 60-percent bell nozzle system has been compared to a 80-percent system, based on present performance data, and found to be lighter by six pounds. Taken into consideration was the increased chamber and interstage structure weight of the 80-percent bell chamber, and the performance increase of the 80-percent chamber. It should be noted, however, that a considerable amount of development work is being done to determine the effect of nozzle contour on kinetic losses at higher expansion ratios. Subject to new data which may come from this development work, a 80 percent bell chamber is selected.

g. Mixture Ratio

There is still considerable disagreement within the industry insofar as the optimum mixture ratio for maximum performance is concerned, although opinion is not as diverse as it was during the Saturn 1 study. Recognition of the magnitude of kinetic losses has caused estimated optimum mixture ratio to drop. Unless other factors dictate, a value slightly on the low side of the optimum point should be used in orbit to reduce gas temperature. On this basis a value of 1.65 is selected for this study.

h. Propellant Supply

System requirements for the propellant supply are about the same as for the Saturn 1. Propellant volumes are in the same range as for Mars 69. The total number of starts required cannot be determined for a certainty, but the six starts assumed for the Saturn 1 should be a conservative number. As before, provisions must be made to assure that the propellants are available at the propellant valves prior to the start of each engine firing, and to minimize C.G. shift due to propellant migration between firings or propellant sloshing during firing. Control of this C.G. shift is especially critical in the Voyager configurations, since the axial distance between vehicle C.G. and gimbal axis is relatively short. This C.G. shift would have maximum effect when starting a firing with partially full tankage.

Another requirement pertains to venting of the propellant tanks. Temperature in the tank compartment drops about 70°F during transit. This means that the tank temperature during the early portions of the flight will be in excess of 100°F. If the temperature during launch is relatively low, then expansion of the propellant will cause a rise in tank pressure. Calculations show that it is not economical from a weight standpoint to provide ullage space sufficient to prevent overpressure. A means for venting that will assure that propellant is not lost is therefore required.

Theoretically, another alternative exists. If the tanks were pressurized to only one atmosphere prior to launch, or were pressurized only with propellant vapors, then pressure would not rise above safe levels, and venting would not be required. However, an important criterion in the propulsion design is that a single valve malfunction not cause mission failure, and minor leakage of a pressurization valve prior to tank temperature rise would cause pressure buildup. A malfunctioning thermal control system could also cause overpressure. From a practical standpoint, therefore, tank venting is a requirement.

There is one requirement which has changed, and this can have a significant effect on the propellant supply design. On the Mars 69 mission, it was necessary to remove propellant on the launcher at various times during the launch countdown if launch delays were encountered. This requirement may not exist on a Mars 71 launch, and if it does, the effect will be small. Since only 2% of the propellant is used prior to the final firing, a large C. G. shift at this time, or prior to this time, is not possible.

In the Saturn 1 study report, the propellant supply system discussed were surface effects, vehicle acceleration, bladders, diaphragms, and bellows. These are still the contenders, but their applicability may be changed somewhat.

(1) Surface Effects — Considerable work has been done in this area since the last Voyager report. Further, this system becomes considerably more attractive where maintaining C. G. prior to firing is not a problem. The problem of providing gas instead of liquid, at the vent, however, still exists; this area has received relatively little attention, in the industry, although some contractual work, in addition to NASA in-house research is being done. Fluid positioning at the tank outlet and sloshing control during firing could probably be done with dual-purpose baffles or tubes.

This system has a relatively low weight, almost unity expulsion and volumetric efficiency, and has no moving parts. The relatively early state of development, especially in the venting area, makes it unsuitable for consideration at this time.

(2) Vehicle Acceleration — Accelerating the propellants by spinning the vehicle is a very simple and effective way of assuring not only that propellant is at the tank outlet, but also that gas is at the vent. Due to many vehicle considerations, it is not practical to spin the vehicle.

An axial acceleration, applied just prior to firing, is also effective. This could be done on the Titan III-C Voyager by using small solids or by using attitude control gas. This system has the same deficiencies as the preceding one insofar as venting is concerned.

(3) Bladders — It can be seen from the foregoing that a physical barrier between gas and propellant is required to facilitate venting. The system with the most development and flight experience which meets this requirement is the bladder. This device is considered to be somewhat unreliable, due to susceptibility to leaks and permeation. Sloshing is dampened considerable, but possibly not sufficiently for the Titan III-C Voyager.

(4) Diaphragms — The difference between a bladder and a diaphragm is not universally agreed upon. Normally, diaphragms are generally considered to be devices attached at the tank equator. Disadvantages are low volumetric and expulsion efficiencies, high ΔP, low development experience, and difficulty in acceptance testing. Advantages are low permeation and probably good liquid damping characteristics.

(5) Bellows — Bellows have the disadvantages of low volumetric efficiency, low expulsion efficiency, weight, and high weight of tank shell. Advantages are low ΔP (at low expulsion efficiencies), ease of test, low permeation, and relatively high reliability.

(6) Propellant Supply System Chosen — The systems considered, together with characteristics pertinent to the Titan III-C Voyager, are given in Table 4.4-9. The technique which was used on the Saturn I Voyager, that of using a partial bellows, and shifting pressurization directly into the propellant after the start of the final firing, could be applied to the Titan III-C also. Since a much smaller volume is involved, however, it appears that a diaphragm which does not require gross changes in shape during cycling could be utilized. Such a diaphragm would not possess the

TABLE 4.4-9. COMPARISON OF CANDIDATE PROPELLANT SUPPLY SYSTEMS

	Surface Effects	Vehicle Acceleration	Plastic Bladders	Metal Bladders	Metal Diaphragm	Metal Bellows	Partial Bellows	Partial Diaphragm
							--With Slosh Plates--	
Permeation - Index	Not Applicable	Not Applicable	Poor	Excellent	Excellent	Excellent	Excellent	Excellent
Expulsion Efficiency With Low ΔP - Index	Excellent	Excellent	Good	Fair	Poor	Very Poor	Excellent	Excellent
Expulsion Efficiency With High ΔP - Index	Not Applicable	Not Applicable	Good	Good	Fair	Poor	Not Applicable	Not Applicable
Susceptibility to Tear or Puncture - Index	Not Applicable	Not Applicable	Poor	Poor	Poor	Good	Excellent	Good
Susceptibility to Binding or Hang-Up - Index	Not Applicable	Not Applicable	Fair	Fair	Good	Fair	Fair	Excellent
Weight - Index	Excellent	Excellent	Good	Good	Fair	Poor	Good	Excellent
Complexity - Index	Excellent	Good	Good	Good	Fair	Poor	Poor	Poor
Slosh Control - Index	Very Poor	Very Poor	Poor	Poor	Fair	Excellent	Good	Good
Venting Control - Index	Unknown	Very Poor	Excellent	Excellent	Excellent	Excellent	Excellent	Excellent
Volumetric Efficiency - Index	Excellent	Not Applicable	Good	Good	Poor	Poor	Excellent	Excellent
Reliability - Index	Unknown	Excellent	Poor	Poor	Fair	Fair	Good	Good

disadvantages of high ΔV and uncontrolled folding inherent in a full diaphragm. Advantages over a partial bellows would be greater resistance to vibration, ease of fabrication, and better predictability of expulsion cycles. This device is shown in Figure 4.4-10. The anti-slosh provision is shown conceptually in Figure 4.4-10 as a perforated cylindrical sleeve. Actual configuration could only be obtained by detailed analysis of the tankage system and vehicle control loop.

In the event that a larger void volume becomes necessary, due to a requirement for off-loading, or other reasons, the partial bellows used in the Saturn I study should be used. This is shown in Figure 4.4-11.

i. Control System

The control system is identical to the one used on the Saturn I study, and is shown schematically in Figure 4.4-12. Redundancy is provided such that only a structural failure or a double failure will result in failure of the system. Redundant components are of different designs and are from different manufacturers to minimize double failures.

3. Power, Weight, Size

Main pressurizing valves and secondary pressuring valves are latch-type valves, and required only about 150 milliseconds to open or close. The squib valves require only a momentary 100-millisecond pulse. Orbit adjust valves require five watts.

The solenoid control valves and isolation valves required 20 watts each, resulting in a total of 60 watts during propulsion system operation.

System weight, including gimbal system, is 1925 pounds, of which 1634 pounds is propellant. Detailed weights are given in the weight section of this report.

B. ATTITUDE CONTROL PROPULSION

Freon-14 cold gas is used as the propellant. The propulsion system is sterilized internally, prior to installation in the spacecraft, and the propellant is sterilized prior

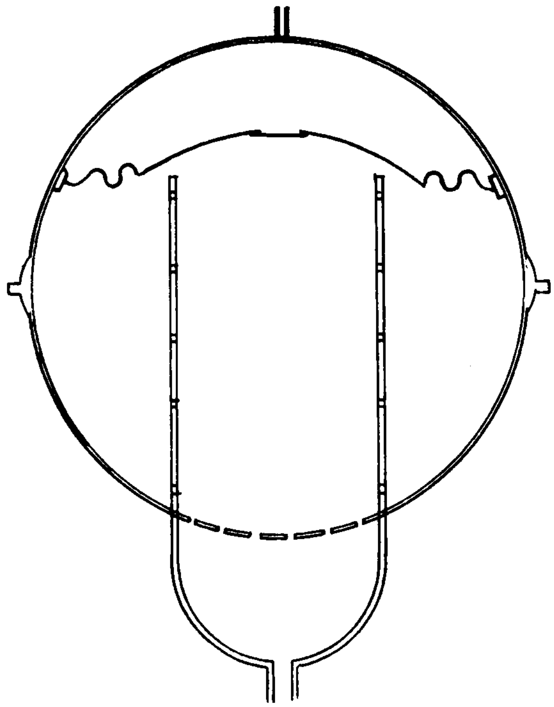


Figure 4.4-10. Partial Diaphragm Propellant Tank

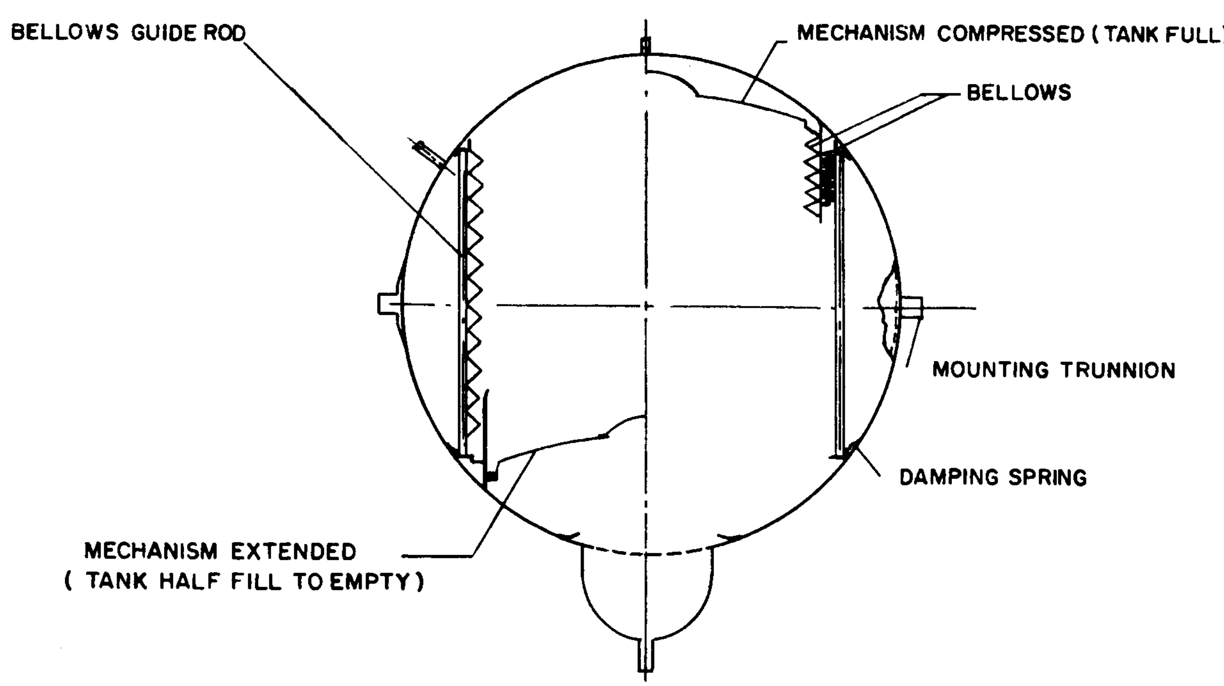


Figure 4.4-11. Partial Bellows Propellant Tank

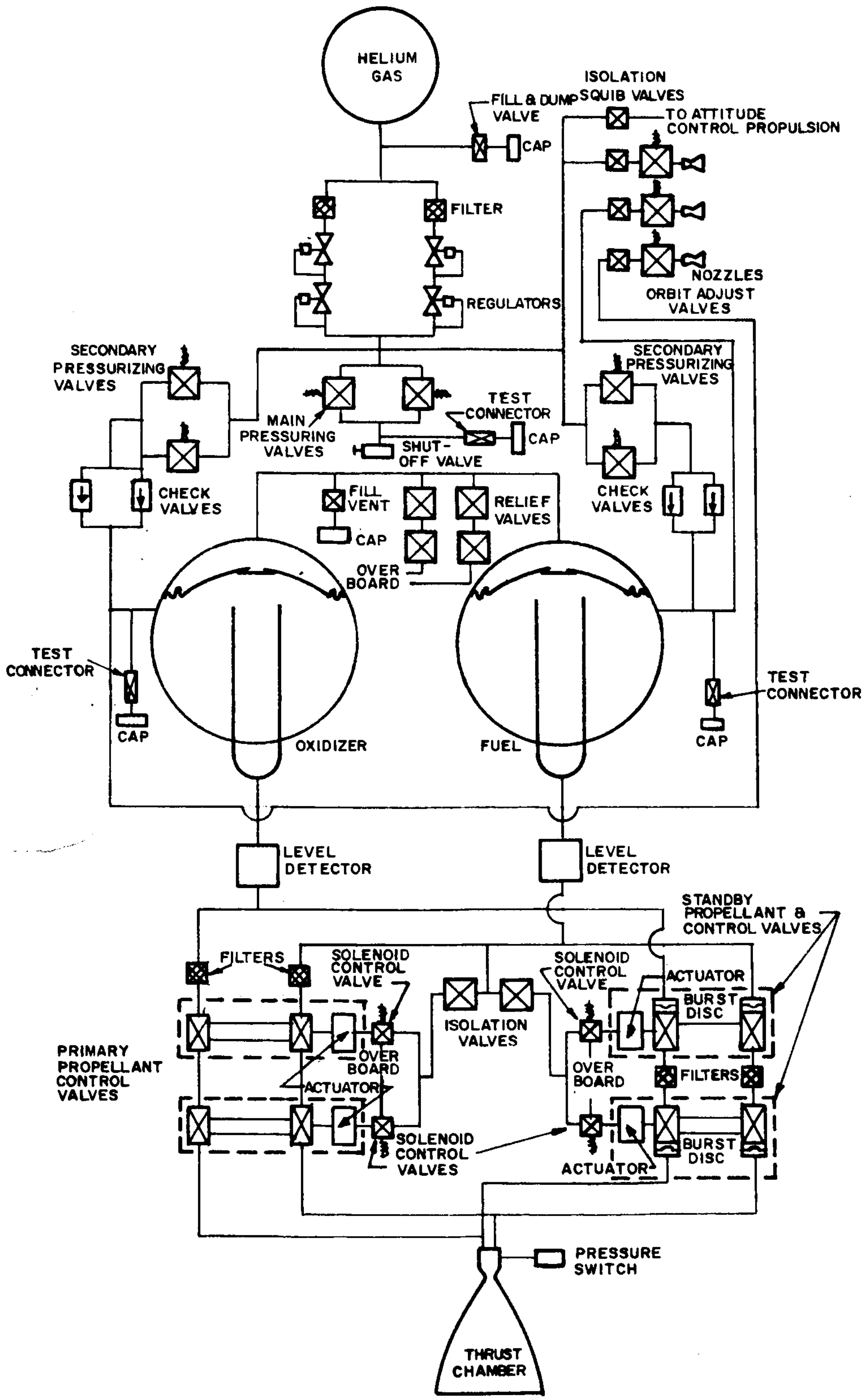


Figure 4.4-12. Orbiter Propulsion System Schematic

to filling. Redundancy is provided such that only a structural failure, or a double failure, will cause failure of the mission.

1. Requirements

Total required impulse is 1082 pound-seconds, which includes all attitude control requirements including the torque necessary to offset the roll torque induced by the main engine. Control in either direction about all the axes is required, and the inducement of lateral motion by the attitude control system is undesirable, but not prohibited. Minimum pulse length is 30 milliseconds. Thrust of approximately 0.01 pound is required in all axes, except that roll thrust should be between 0.1 and 0.2 pound to offset main engine induced roll. Exhaust products must be compatible with the rest of the spacecraft, and must not have a detrimental effect on the scientific mission. Weight should be kept at a minimum consistent with maintaining a high reliability and a low development risk. Power is not critical, since the total on time is only a small portion of one percent of the mission time, and there are no peaking problems.

2. Analysis and Design

The Saturn 1 study considered the use of other than cold gas systems for attitude control, and concluded that a gaseous-stored cold gas system should be used. This study reaches the same conclusion. The requirements have changed little, and no great advances have been made on the other systems considered. The Curtiss-Wright "Cap Pistol" is scheduled for flight test, but for the very low total impulse bits required here, weights are still not competitive. Rocket Research Corporation is continuing work on their "subliming solid", but the thermal control requirement noted in the previous study remains a problem.

Freon-14 is again chosen over the other candidate gases, based on the weight savings over nitrogen of nine pounds. It should be noted that some concern has been expressed on the possibility of radiation causing breakdown of the Freon-14 molecules with subsequent attack on the titanium tank walls. This possibility appears to be very remote.

Specific impulse used is 45.3 seconds. The assumption is made that 70 percent of the gas is used in the pulsing mode. If it were 100-percent pulsing, this would drop to 44.0 seconds. This is equivalent to 63.8 seconds for nitrogen, which is considerably higher than the 35 seconds used in JPL studies. This discrepancy is discussed in the Saturn I study.

Total amount of gas used is 23.4 pounds; a 5-percent leakage factor is included in the total impulse. A factor of safety of 2.0 at $70^{o}F$ is assumed for ground safety, and a factor of safety of 1.5 during flight. Since the tank may reach $170^{o}F$ in flight, the pressure in flight is the determining factor.

The configuration of the system is the same as with the Saturn 1 system, and is shown in Figure 4.4-13. The two systems are completely separate, except for the latch valve which connects the two. This valve is used in the event of a fail-to-open situation with either a regulator or solenoid, and the remaining gas is allowed to flow into the other tank. Subsequent operation is in a degraded mode, i.e., only one half of the couple will be operating, and will result in some translation, but is not believed to be serious from an overall mission standpoint. Series shutoff valves are provided for the solenoid valves, so that only a structural failure, or a double failure, will cause mission failure. Shutoff valves are located immediately upstream of the nozzle valves to minimize gas loss in the event of leakage of the nozzle valves. All joints are welded or double-sealed.

There appears to be a very high probability that amounts of the attitude control gas will impact on the planet, especially after being released during the orbit phase. The possibility of carrying viable organisms appears to be very real. For this reason, the system is internally sterilized prior to use, either by the use of ethylene oxide, or by heat sterilization as a system prior to installation. The gas is sterilized prior to filling.

The system for later opportunities is not expected to differ from the system outlined above.

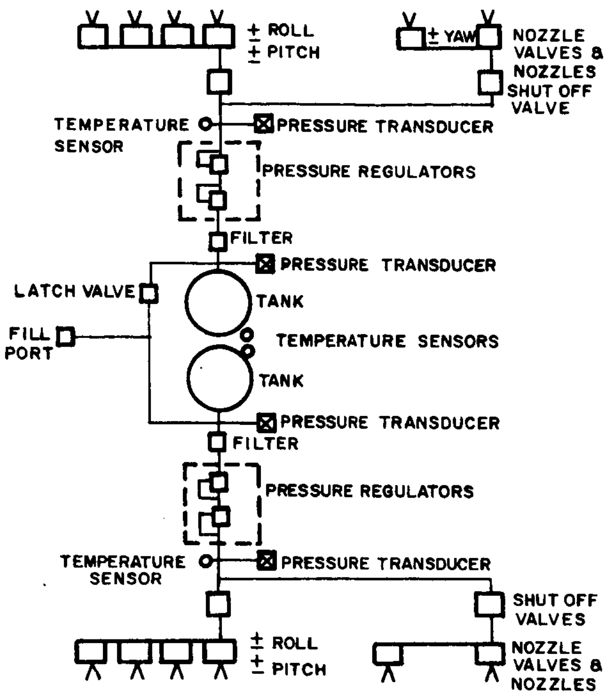


Figure 4.4-13. Orbiter Attitude Control System Schematic

3. Power, Weight, Size

Nozzle valves and shutoff valves are expected to require about 6 watts each. Total on time is approximately three hours. Considering the proportion of on time to off time, the average power is 0.004 watt, excluding any power which may be required for thermal control. The weights are given in Table 4.4-10.

Two tanks are used, each with a diameter of 10.1 inches. Other components are relatively small; total volume is less than 0.2 cubic foot.

Power, weight, and sizes for later opportunities are not expected to differ from those given above.

TABLE 4.4-10. ORBITER ATTITUDE CONTROL SYSTEM WEIGHTS

Gas	23.4
Tanks (2)	7.6
Check Valve	.1
Filters (2)	.8
Pressure Regulators (2 dual)	6.2
Shut-Off Valves (4)	5.0
Solenoid Valves (12)	5.2
Nozzles (12)	1.2
Tubing	2.8
Pressure Transducers (4)	1.0
Temperature Sensors (4)	1.0
Latch Valve	1.8
Total	56.1

4.4.3 ORBITER/LANDER

A. MAIN PROPULSION SYSTEM

This system is identical to the Orbiter Main Propulsion System except for weight, size, and thrust level. Thrust level is 400 pounds; duration is 530 seconds.

1. Requirements

Requirements are identical to the Orbiter Main Propulsion System requirements, except that total impulse required, with a 308 second specific impulse, is 210,000 pound-seconds.

2. Analysis and Design

The analysis shown in paragraph 4.4.2.A for the Orbiter Main Propulsion System is applicable also for the Orbiter/Lander, except as weights, sizes, and thrust levels effect the analysis.

The limitation of 600 seconds for total burn time was applied to the total impulse required, and a thrust level between 300 and 400 pounds was obtained. Thrust chamber weights are given in Figure 4.4-14. A 400-pound thrust level was chosen in order to stay within the limitation. No computer runs on gravity loss were made, but losses for this vehicle at this thrust level and burn time are not expected to be excessive. No optimization calculations were made on area ratio; the 100:1 selected for the orbiter is used here also.

3. Power, Weight, Size

Power requirements are identical to the Orbiter Main Propulsion System. Overall system weight is 903 pounds, of which 720 pounds is propellant. Detailed weights are given in the weight section of this study.

B. ΔV MOTOR

This ΔV motor is identical to the Bus/Lander motor except for size and weight, and thrust level.

1. Requirements

Requirements are the same as for the Bus/Lander, except that lander weight is 1284 pounds, and exhaust products cannot be allowed to damage the Orbiter.

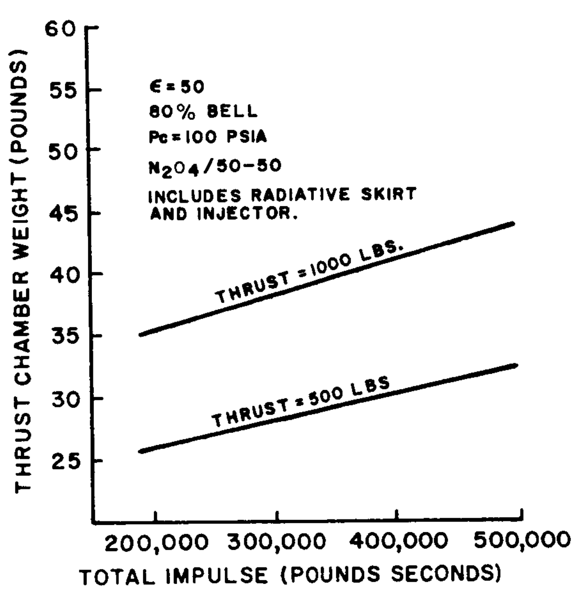


Figure 4.4-14. Ablative/Radiative Skirt Thrust Chamber Weight vs Total Impulse

2. Analysis and Design

The analysis given in Section 4.4.1.B.(2) is applicable for the Orbiter/Lander also. Thrust level is 1200 pounds. The additional problem of Lander motor gas impingement on the orbiter does not lend itself readily to analytical solution. Possible problems can be minimized by allowing a maximum amount of time to elapse between the separation and firing of the solid motor, and by minimizing solids content of the motor exhaust. Development to determine and, if necessary, minimize, the effects of motor exhaust on spacecraft structure will be necessary.

3. Power, Weight, Size

Except for the possibility of thermal control, the only power required is a 100-millisecond pulse to fire the pyrogen squibs. Weight of the motor without mounting hardware is 62 pounds; motor diameter is 12.2 inches.

C. ATTITUDE CONTROL PROPULSION

This system is identical to the Orbiter system except that less gas and a smaller tank are required to provide the smaller total impulse required.

1. Requirements

Except for total impulse, the requirements given in paragraph 4.4.2.B are applicable also to the Orbiter/Lander. Total impulse required is 1017 pound-seconds.

2. Analysis and Design

Using the same specific impulse as used for the orbiter, the total amount of gas required is 21.8 pounds. The possibility of using the same system used for the Bus/Lander Attitude Control System can be considered, and a weight comparison, showing also a cold nitrogen system, is given in Table 4.4-11. The controls and piping weights are the same as for the orbiter.

Because of the weight penalty with the Mariner-type system, the Orbiter-type system is selected. A nine-pound weight advantage over the nitrogen system is realized.

TABLE 4.4-11. ORBITER/LANDER ATTITUDE CONTROL SYSTEM COMPARISON

	Orbiter-Type System Freon-14	Bus/Lander (Mariner) Type System Freon-14	Orbiter-Type System Nitrogen
	Weight in Pounds		
Gas	21.8	65.4	15.2
Tankage	7.1	21.3	22.7
Controls & Piping	25.1	18.0	25.1
Total	54.0	104.7	63.0

3. Power, Weight, Size

Power consumption is the same as for the Orbiter, an average of 0.004 watt for the mission. Total weight is 54.0 pounds including 21.8 pounds of gas and 7.1 pounds of tank. Outside diameter of the tank is 9.8 inches. Volume of the other components is less than 0.2 cubic foot.

No changes are anticipated to be required for later opportunities.

D. SPIN SYSTEM

The spin system is identical, except for size and weight, to the one utilized on the Bus/Lander, and discussed in paragraph 4.4.1.D.

1. Requirements

The spin system serves the same purpose on the Orbiter/Lander as on the Bus/Lander. Requirements are the same, except for the lower moment of inertia associated with the smaller Lander.

2. Analysis and Design

The analysis given in paragraph 4.4.1.D for the Bus/Lander is applicable to the Orbiter/Lander. A cold nitrogen system, with a safety factor of 2.0 during sterilization, is utilized. If additional payload weight allocation becomes necessary, a solid gas generator could be used with little reliability penalty, the tankage safety factor could be reduced, or gas filling could be accomplished after sterilization.

3. Power, Weight, Size

Some power may be required for thermal control, but this is not probable. No other power is required except for about 100 milliseconds when squibs are fired. Outside diameter of the tank is 10 inches. Systems weights are given in Table 4.4-12.

TABLE 4.4-12. SPIN SYSTEM WEIGHTS

Item	Weight (Pounds)
Tanks (2)	16.9
Nitrogen	7.5
Squib Valves (2)	1.5
Tubing	1.6
Nozzles and Fillings	.8
Total	28.3

E. RETARDATION MOTOR

The Orbiter/Lander Retardation Motor is essentially the same as the Bus/Lander Retardation Motor, except that weight and size is decreased and thrust level is 1300 pounds.

1. Requirements

Requirements are the same as for the Bus/Lander except the weight of the Lander is 1078 pounds.

2. Analysis and Design

The analysis given in paragraph 4.4.1.E.(2) is applicable for the Orbiter/Lander also. Burn time is three seconds; thrust is approximately 1300 pounds.

3. Power, Weight, Size

Except for the possibility of thermal control, the only power required is a 100-millisecond pulse to fire the pyrogen squibs. Weight of the motor is 27 pounds; diameter is 10 inches.

4.5 TELEVISION SUBSYSTEM

The TV Subsystem recommended for the Titan IIIC Voyager missions are identical to those recommended for the Saturn Mars '69 Voyager in the previous Voyager Design Study except for the omission of the nadir vidicon in the Orbiter. Tables 4.5-1 and 4.5-2 give the recommended TV missions and camera characteristics.

Following is a general summary of the subsystems; a detailed analysis and description is included in the previous Voyager Design Study report.

4.5.1 OVERALL DESCRIPTION

A. ORBITER TELEVISION

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

B. LANDER TELEVISION

The Mars Landers are equipped with one television camera with steerable optics such that clouds, the horizon, and the terrain in the immediate vicinity of the landing site can be scanned through 360 degrees during daylight hours. A television camera attached to a microscope is also provided for examination of soil samples and for planned biological experiments. The panoramic camera will resolve three minutes of arc (in color); the microscope will resolve 1μ, 5μ, and 50μ (in color).

C. RESOLUTION PARAMETERS

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

TABLE 4.5-1. MARS TELEVISION MISSION

	Orbiters			Landers	
Function	Map	Map	Map	Panorama	Microscope
Number of Cameras	2	3	1	1	1
Optical Resolution	1 Km	140 m	20 m	3 Min. of Arc	$1\,\mu$, $5\,\mu$, $50\,\mu$
Stereo	345 m	no	no	no	no
Color	no	yes	no	3 filters	3 filters

TABLE 4.5-2. TELEVISION CAMERA CHARACTERISTICS

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

D. CAMERAS

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

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

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

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

E. OPTICS

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

F. STEREO

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

4.5.2 CAMERA DESCRIPTION

Figure 4.5-1 shows the general block diagram of the television cameras. A detailed list of components for each camera is given in Table 4.5-3 along with power, weight, and size estimates.

4.5.3 CRITICAL PROBLEM AREAS

In the previous study, three critical problem areas were found to exist in the TV Subsystem: vidicon sterilization, image orthicon tube development, and image orthicon camera development.

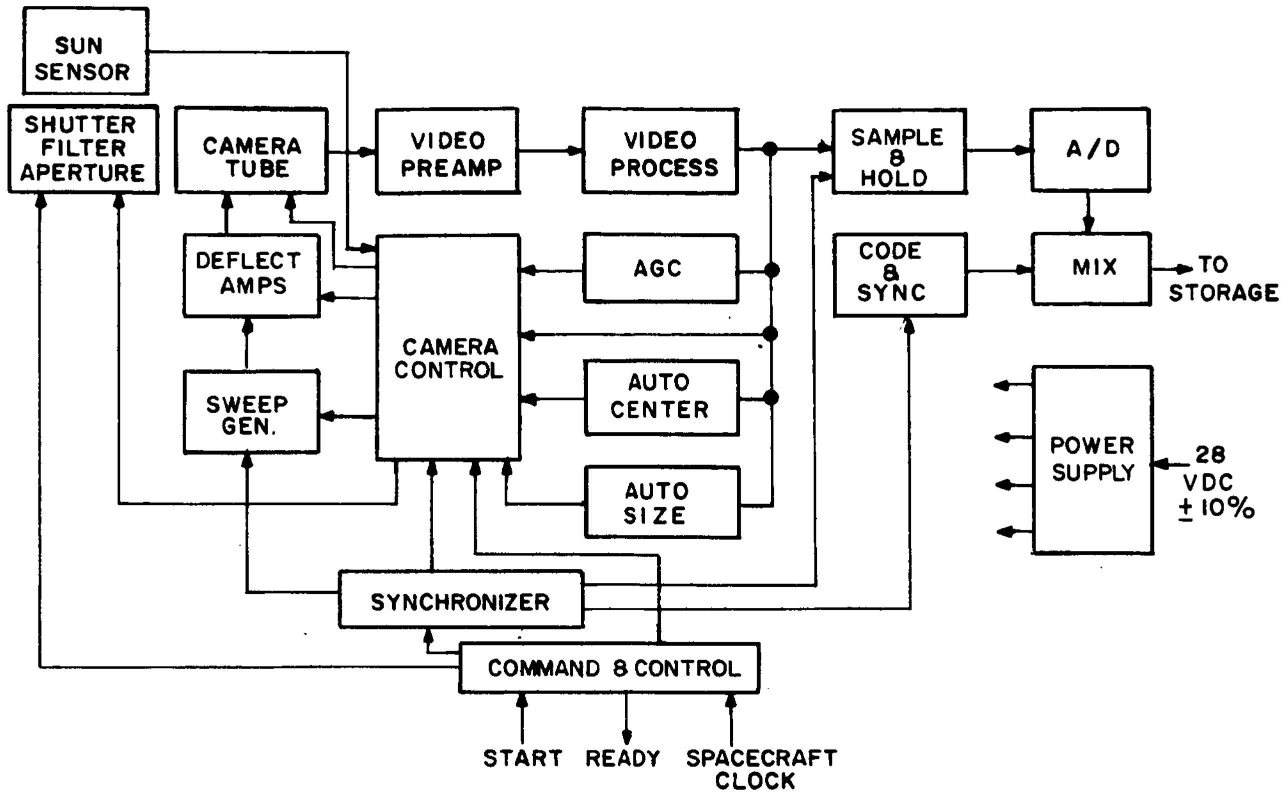


Figure 4.5-1. Television Subsystem Block Diagram

A. VIDICON STERILIZATION

Although semiconductors are basically able to withstand high storage temperatures, several problems are apt to occur during sterilization. General Electrodynamics Corp. (GEC) lists them as follows:

1. Modifications in the semiconductors
2. Interdiffusion of successive layers
3. Shifting of spectral response
4. Changes in secondary emission characteristics
5. Structural changes
6. Changes in dark conductivity affecting sensitivity and storage characteristics

Beyond these GEC lists vacuum tube problems which could arise due to sterilization:

1. Outgassing of components
2. Deterioration of the thermionic cathode
3. Leakage in the faceplate seal

TABLE 4.5-3. TELEVISION SUBSYSTEM POWER, WEIGHT, AND SIZE ESTIMATES

Subsystem Components	Camera Head, Vidicon	Camera Elect., Vidicon	Camera Head, I.O.	Camera Electronics, I.O.	Exposure Control (Iris and Shutter)	Light Meter	Control Electronics	Color Wheel	Color Wheel Actuator	Boom Erect. (Panoramic)	Boom	Azimuth Mechanism	Elevation Mechanism	Focus Control	Electronic Flash	Ablation Mat'l Remover	Mirror	Lens Weight (lbs)	Total Weight (lbs)	Total Power (watts)	
Component Characteristics																					
Power (Watts) Size (Inches) Weight (lbs)	2 8x2-1/2D 2.5	16 9x6x4 5.0	3 17.5x3D 4.0	20 9.2x8x5 13.5	2.6x2.6x1	2(total) 3x0.5D 2.0(total)	5x3x3	0 0.5x5D 0.5	1 2x2x2 0.5	1 4x1D 0.3	0 16x1D 2.0	1 2x2x3 2.0	1 2x2x3 2.0	0.5 2x2x3 2.0	2.5 12x4x4 6.0	1 4x1D 0.3	0 5x8x0.25 2.0				
FUNCTION																					
Orbiter, Vidicon, Stero, 345 meter height	x	x			x	x	x											1	20	40	(2/Vehicle)
Orbiter, I.O., Nadir, 140 meter, color			x	x	x	x	x											1	57	75	(3/Vehicle)
Orbiter, I.O., Nadir, 20 meter			x	x	x	x	x											18.8	37.8	25	
Lander, Vidicon, 3 minute, color	x	x			x	x	x	x	x	x	x	x	x	x			x	3	23.3	24.5	
Lander, Microscope, color	x	x						x	x						x			2	16	21.5	
																			154.1	186.0	Totals/vehicle

The faceplate seal leakage was pointed out by RCA as the most serious sterilization problem.

However, information received from General Electrodynamics Corp., indicates that a sterilizable, ruggedized vidicon having high sensitivity is indeed feasible.

B. IMAGE ORTHICON TUBE DEVELOPMENT

The electrostatic image orthicon is at this time being developed at the GE Power Tube Department, Syracuse, N.Y. Electrical tests have not shown completely satisfactory performance, especially concerning resolution. No environmental tests as severe as those required for Voyager have been performed on the tube. GE Power Tube Department, however, expects to have a ruggedized, high resolution tube developed within the next year.

C. IMAGE ORTHICON CAMERA DEVELOPMENT

Employment of image orthicons for the Voyager missions also depends on successful development of automatic control circuitry for long periods of unattended camera operation. A NASA contract has been awarded to Hazeltine Corp. for development of a space-qualified image orthicon camera. The GE Advanced Electronic Center, Ithaca, N. J. is also doing independent development work on a ruggedized, automatic image orthicon camera. It is, therefore, reasonable to assume availability of this equipment at the time of a Voyager design contract if the current developments are successful.

4.6 RADAR SUBSYSTEM

4.6.1 REQUIREMENTS

Two relatively different general sets of radar requirements are posed by each Lander. They are:

1. Altitude measurements to provide a data base for atmospheric measurements during the parachute descent.
2. Altitude measurement for the actuation of the braking rockets.

They differ in that the first requirement is for a relatively low-accuracy altitude measurement at high altitude while the second requirement is for high-accuracy measurements near the planet surface.

4.6.2 RECOMMENDED SYSTEM

A combination of the following two radars appears to satisfy the above requirements:

1. A pulsed radar altimeter which can be a modification of the Altitude Marking Radar made by Hughes Aircraft Company for the Surveyor.
2. An FM/FM radar recently proposed by the Light Military Electronics Department of G.E. for satellite rendezvous.

The essential characteristics of the two radars are as given in Table 4.6-1.

TABLE 4.6-1. CHARACTERISTICS OF ALTITUDE MARKING RADAR AND FM/FM RADAR

Modified Hughes Altimeter Marking Radar	
Altitude Range	1,000 to 200,000 feet
Accuracy	±100 feet or 2 percent, whichever is greater
Velocity	200 ft/sec maximum
Data Rate	One reading per second minimum
Modulation	Pulsed
Radar Reflection Coefficient	Similar to extremes of Earth terrain
Antenna Beamwidth	20 degrees
Transmitting Frequency	X-Band
Antenna Diameter	5 inches
Volume (including antenna)	300 cubic inches
Power Required	10 to 15 watts average
Weight	5.5 pounds

TABLE 4.6-1. CHARACTERISTICS OF ALTITUDE MARKING RADAR AND FM/FM RADAR (Cont'd)

GE-LMED Satellite Rendezvous Radar	
Maximum Range	500 feet for 10 ft^2 target
Minimum Range	2 feet
Range Accuracy	±1 foot or 3 percent, whichever is greater
Maximum Range Rate	500 ft/sec
Minimum Range Rate	0 ft/sec
Range Rate Accuracy	±1 ft/sec or 10 percent, whichever is greater
Modulation	FM/FM
Antenna Beamwidth	13 degrees
Transmitting Frequency	X-Band
Antenna Diameter	8 inches
Volume (including antenna)	400 cubic inches
Power Required	23 watts at 28 Vdc
Weight	11 pounds

Some reduction of overall power, weight, and size might be possible by integrating some of the functions of the two radars. A more extensive analysis is required to determine the feasibility and extent of such a reduction; however, most of the comparable functions appear to differ considerably.

5. RELIABILITY AND VALUE ANALYSIS

5.1 RELIABILITY EVALUATION OF CONFIGURATIONS STUDIED

5.1.1 GENERAL

During the course of this study, the principal efforts have been directed toward the optimization of system concepts and to the identification and evaluation of alternative subsystems, components and operational plans to establish a quantitative basis for those optimizations and provide a reasonably accurate indication of the attainable system reliability.

Reliability analyses were made of the following configurations or systems in varying degrees of refinement as deemed necessary for the proper evaluation of the various 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 analyses of systems 4 (Bus/Lander), 5 (Orbiter) and 6 (Orbiter/Lander) are described in Section 2. 6. 3. Also, reliability analyses of systems 4 and 5 are described in greater detail in Section 5. 2 under the classification of the recommended system. Therefore, only systems 1, 2 and 3 will be described in Section 5. 1.

5.1.2 RELIABILITY EVALUATION OF INTEGRATED OR SEPARATE BUS AND IMPACTING OR FLY-BY TRAJECTORY

The reliability analysis of the preliminary design concept of these configurations yielded the reliability estimates given in Table 5. 1-1 based on a 6960 hour transit period.

TABLE 5.1-1. RELIABILITY ESTIMATES FOR INTEGRATED AND SEPARATE BUS, AND IMPACTING AND FLY-BY TRAJECTORY

Subsystem	RELIABILITY							
	100-Hours Mission				3-Months Mission			
	Integrated		Separate		Integrated		Separate	
	Fly-by	Impact	Fly-by	Impact	Fly-by	Impact	Fly-by	Impact
Communications	0.904	0.904	0.872	0.872	0.823	0.823	0.794	0.794
Power Supply	0.969	0.950	0.950	0.950	0.949	0.930	0.950	0.930
Propulsion			(Same as Voyager Saturn I-B)					
G & C			(Same as Voyager Saturn I-B)					
Communications (with redundant RF)			0.962	0.962				

Several features of the different configurations obviously assume dominant positions in the reliability analysis. Some of these features are discussed in the following paragraphs.

Many communications components in a separate Bus duplicate components in the Lander but are not redundant. As presently designed, they cannot be used alternatively by programming or command and they simply serve to reduce the prior operating time of the Lander components. All Bus items essential to arrival at point of separation determine the probability of Bus success during transit. Immediately after separation, the reliability of the Lander items only will determine the reliability of the Lander system. Since the opportunity to use duplicated Lander subsystem components in redundancy for Bus carried components is not present, the reliability of the system is lower with separate Bus than with an integrated Bus.

In the analysis of an impact trajectory, consideration must be given to the Bus sterilization requirement. Bus sterilization cannot be depended upon until suitable sterilization methods can be defined and verified for all components (including their insides) of all subsystems in the Bus. Since such a sterilization is not yet assured for the inner composition of the image orthicon, the ethylene oxide sterilization applied to the Bus cannot be depended upon to assure that it is fully sterilized according to requirements. It can only assure that it is 99 percent sterile with no assurance that the other 1 percent is sterile even to 10^{-2}. Thus, the 10^{-4} sterilization requirement can only be met by assuring that the Bus is ejected into a fly-by trajectory after separation with a reliability exceeding 0.999.

The reliability of the 50-pound thrust mono-propellant engine has been compared with the reliability of the solid rocket alternative for course correction. The 50-pound thrust engine reliability for this operation is dependent upon the trajectory (impact or fly-by) selected as the objective of the midcourse corrections up to the final course correction which necessarily would place the combined Bus and Lander on an impacting trajectory. Considering these factors leads to the conclusion that the 50-pound thrust mono-propellant engine can be considered as approximately equal in reliability to the solid propellant engine but that neither by themselves are able to provide the 0.9999 reliability requirement and that at least two fully redundant propulsion systems must be provided, each having greater than 0.99 reliability, if there is to be assurance of meeting the 10^{-4} sterilization requirement while using an impacting trajectory prior to separation.

5.1.3 SOLAR POWER ORBITER VERSUS RTG POWER ORBITER

The reliability analysis of the designs of these two orbiter systems yielded the reliability estimate given in Table 5.1-2 based on some gross estimates of G & C component requirement for the RTG power Orbiter and also on a 6960 hour transit period.

TABLE 5.1-2. SOLAR POWER ORBITER VERSUS RTG POWER ORBITER RELIABILITY ESTIMATES

Subsystem	Solar Power		RTG Power	
	100 Hours	3 Months	100 Hours	3 Months
Communications	0.876	0.798	0.876	0.798
G & C	0.897	0.828	0.891	0.771
Power Supply	0.971	0.962	0.968	0.959
Hot Gas Prop.	0.999	0.999	0.999	0.999
Cold Gas Prop.	0.99	0.99	0.99	0.99
Solar Array Deploy.	0.999	0.999	-	-
Orbiter Vehicle Rel.	0.753	0.628	0.747	0.583

The communications subsystem is essentially the same for both solar and RTG power Orbiter. The propulsion subsystems (Hot Gas and Cold Gas) are exactly the same for the solar and RTG power Orbiter designs, as well as for the Voyager Saturn I-B Orbiter design specified in GE Document 63SD801 Volume II Section 4.

The G & C subsystem for the solar power Orbiter utilizes sun sensors, star tracker and a three-axis PHP, while the G & C subsystem for the RTG power Orbiter has no PHP as a separate guidance feature but follows the Earth with the Hi-Gain Antenna while in orbit around Mars.

5.2 RELIABILITY ANALYSIS OF THE RECOMMENDED SYSTEM

5.2.1 GENERAL

This section presents a detailed analysis of the recommended system which is composed of a combination of a Bus/Lander and an Orbiter. This analysis is the result of many analyses performed during the study as design deficiencies and critical problem areas which would seriously influence the required performance were investigated and corrected during the many design iterations. One of the objectives of all these design improvements and modifications was an increase in the inherent reliability of the proposed system.

This analysis followed the same reliability philosophy developed for the Voyager Saturn I-B study where the best available part and component information was utilized and the use of High Reliability parts (e.g., Minuteman, Advent, etc.) was specified wherever such parts could be considered applicable.

5.2.2 BUS/LANDER SYSTEM

A. SYSTEM DEFINITION

The Voyager Bus/Lander system is required to have the capability of transporting a Lander vehicle to Mars and placing it on the surface of Mars for the scientific investigation of the planetary surface and atmosphere.

Additional system definition and reliability analysis of the Bus/Lander system is given in section 2.6.3(A).

The mathematical model for this system is

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

Entering the computed reliability values in this mathematical model gives

$$(100 \text{ Hours}) \; R \text{ (System)} = (0.915)(0.760) = 0.696$$

$$(3 \text{ Months}) \; R \text{ (System)} = (0.915)(0.704) = 0.645$$

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 Transit	Subsystem	Reliability 100 Hours	Reliability 3 Months
Communications	0.999	Communications	0.863	0.815
Guidance & Control	0.920	EP & D	0.970	0.959
Hot Gas Propulsion	0.999	Prop. & 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
Bus Vehicle Reliability	0.915	Lander Vehicle Reliability	0.760	0.704

B. BUS VEHICLE

The Bus Vehicle has multiple functions in the mission. During the transit phase, it is the Earth-vehicle communications link, performs maneuvers, and transmits diagnostic data. At separation from the Lander, it is projected into a fly-by projectory to miss the planet Mars and to become inoperative.

A mathematical model is shown for the Bus operation from launch to point of Lander separation.

$$R_{(Bus)} = R_{(Communications)} \cdot R_{(G\&C)} \cdot R_{(Hot\ Gas)} \cdot R_{(Cold\ Gas)}$$

Substituting computed reliability values in the above equation gives

$$R_{(Bus)} = (0.999)(0.920)(0.999)(0.997) = 0.915$$

1. Communications Subsystem

Practically all of the components of the Bus/Lander communications subsystem are contained in the Lander vehicle and are analyzed in the Lander reliability section (see Section 5.2.2(b)(1)). Only one omni antenna and the Hi-Gain three-foot antenna dish are physically located on the Bus.

2. Guidance and Control Subsystem

The Guidance and Control Subsystem is designed to perform:

1. Transit orientation
2. Inertial reference
3. Antenna pointing
4. Trajectory correction.

Its two functional areas are: (See Block Diagrams, Figures 5.2-1 and 5.2-2.)

1. Attitude control
2. Earth tracker and antenna drive.

Attitude Control furnishes fine attitude correction to the vehicle by the magnitude of the error signals received from attitude sensors in the pitch, yaw and roll axes. Attitude Control is furnished by firing coupled cold gas jets. The firing time is dependent on the magnitude of the error signals received from the attitude sensors.

The Earth tracker and antenna drive keep the hi-gain communications antenna pointed to the Earth.

a. Reliability Analysis

Because attitude corrections will be necessary throughout the entire mission, the high usage equipments required for this function are in total redundancy or an alternate mode of operation is provided, given that a failure occurs in the primary mode.

All amplifiers, pitch, yaw and roll, are in redundancy and the earth tracker can be used as a back-up to the star tracker during some parts of the mission.

In the transit phase, the major sensing elements are in continuous operation, whereas the gyros and the other components only have periodic operation for monitoring purposes or reorientation.

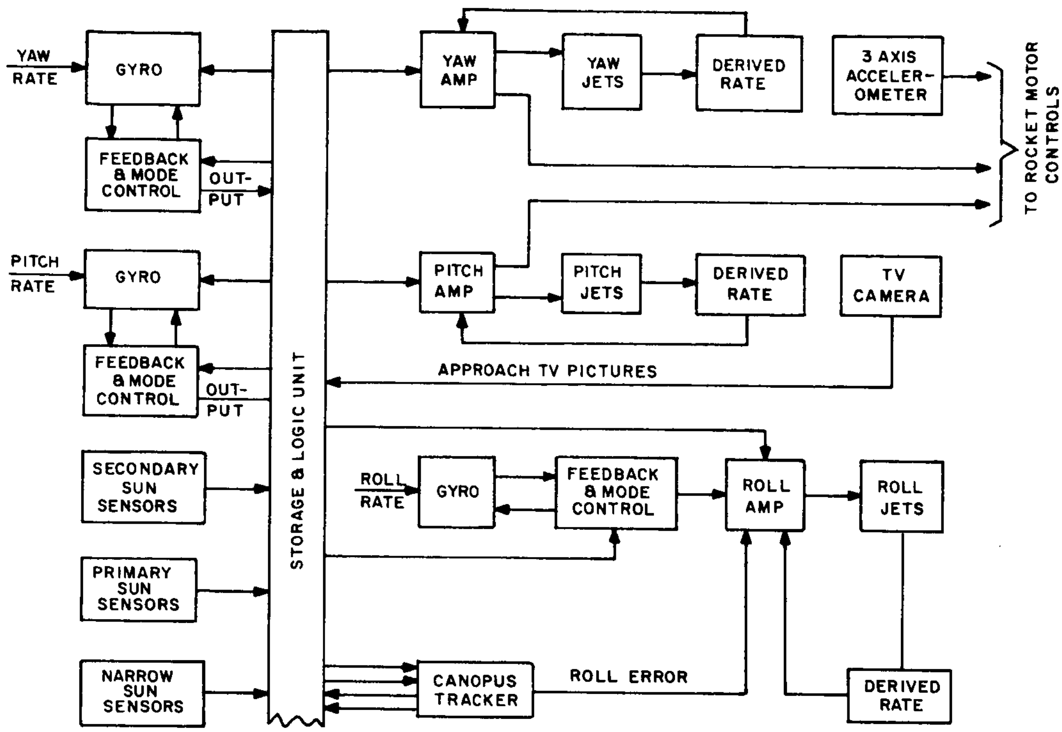


Figure 5. 2-1. Simplified Block Diagram - Bus Guidance and Control Subsystem

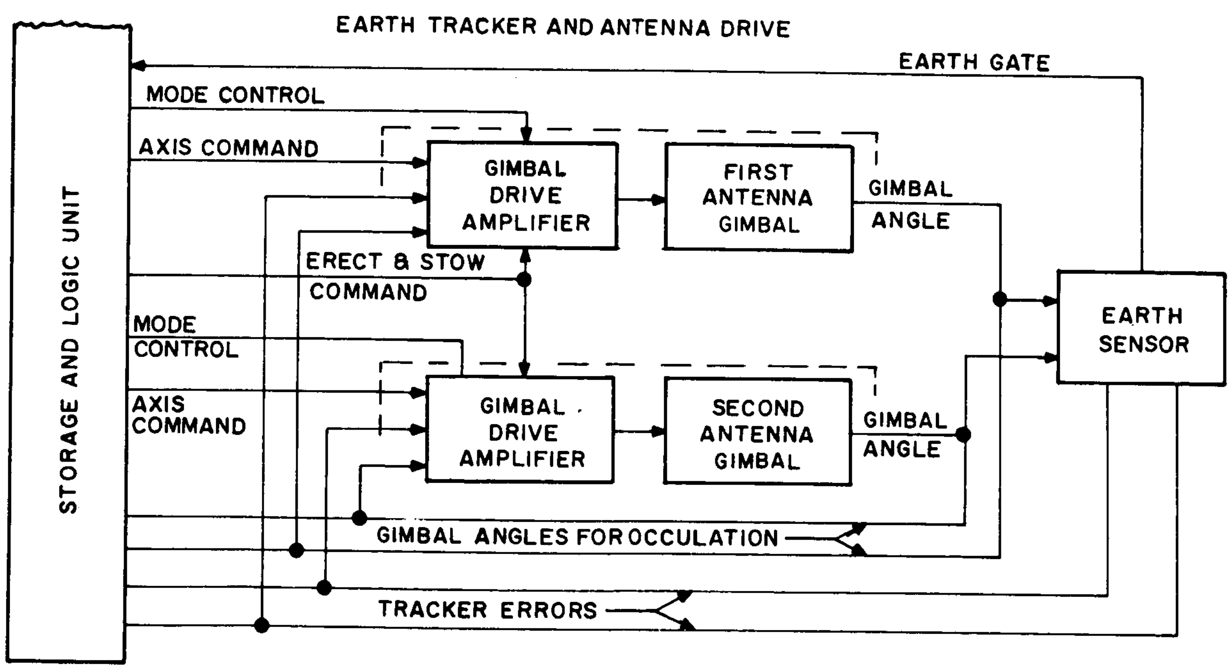


Figure 5. 2-2. Simplified Block Diagram - Bus Guidance and Control Subsystem

The failure of the narrow sun sensor degrades orientation function, but the vehicle then relies on the primary sun sensor for orientation.

The storage and logic unit has internal circuit redundancy and the majority of the circuits will only "see" a 60 percent duty cycle in the mission.

All gyros have a lifetime requirement of 8000 hours, whereas the estimated use time in the Mars '71 Mission is approximately 200 hours.

The thrust vector control and accelerometer are expected to have an operational life of only 20 hours, since they will only be energized prior to and during any hot gas firing.

b. Mathematical Model and Reliability Computations

The mathematical model for the Guidance and Control subsystem shows the components that are required to operate throughout the entire mission, and the back-up modes available in case of a functional failure of the primary mode of operation.

$$
\begin{aligned}
R_{(G\&C)} = {} & R_{(\text{star tracker})} \cdot R_{(\text{narrow sun sensor})} \cdot R_{(\text{primary sun sensor})} \\
& \cdot R_{(\text{secondary sun sensor})} \cdot \left[1-(1-R_{\text{amplifier}})^2\right]^3 \\
& \cdot R_{(\text{storage \& logic unit})} \cdot R^3_{(\text{gyros})} \cdot R_{(\text{earth sensor})} \\
& \cdot R^2_{(\text{antenna servos})} \cdot R^3_{(\text{feedback \& mode control amplifiers})} \\
& \cdot R^2_{(\text{thrust vector control})} \cdot R_{(\text{accelerometer})}
\end{aligned}
$$

Entering the proper component reliability values tabulated in Table 5.2-2 gives the estimated reliability of the G&C subsystem. Where redundancy exists within a component, it has been considered in calculating the "R" value for that component.

$$
\begin{aligned}
R_{(\text{transit})} & = (0.986)\,(0.999)\,(0.996)\,(0.999)\left[1-(1-0.995)^2\right]^3 \\
& \quad (0.986)\,(0.999)^3\,(0.980)\,(0.988)^2\,(0.999)^2\,(0.999)^3 \\
& \quad (0.999)^2\,(0.999) \\
& = 0.920
\end{aligned}
$$

Alternate back-up mode - Earth tracker in standby redundancy to star tracker:

$$R_{(\text{star tracker})} = \left[R_{(\text{star tracker})} + \frac{\lambda(\text{star tracker})}{\lambda(\text{earth tracker}) - \lambda(\text{star tracker})} \left(R_{(\text{star tracker})} - R_{(\text{earth tracker})} \right) \right]$$

TABLE 5.2-2. BUS GUIDANCE AND CONTROL SUBSYSTEM RELIABILITY DATA

Comp. No.	Component	Failure Rate (%/1000 Hours)	5400 Hours Transit Effective Time (Hours)	5400 Hours Transit Rel.
1	Gyro (yaw)	0.500	125	0.999
2	Gyro (pitch)	0.500	125	0.999
3	Gyro (roll)	0.500	125	0.999
4	Feedback & Mode Cont. (yaw)	1.200	125	0.999
5	Feedback & Mode Cont. (pitch)	1.200	125	0.999
6	Feedback & Mode Cont. (roll)	1.200	125	0.999
7	Power Amplifier (yaw)	0.093	5410	0.995
8	Power Amplifier (pitch)	0.093	5410	0.995
9	Power Amplifier (roll)	0.093	5410	0.995
10	Storage & Logic Unit	0.440	3250	0.986
11	Secondary Sun Sensors	0.080	136	0.999
12	Primary Sun Sensors	0.080	5410	0.996
13	Narrow Sun Sensors	0.010	5410	0.999
14	Star Tracker	0.256	5410	0.986
15	Accelerometer	0.178	64	0.999
16	Thrust Vector Control	0.228	64	0.999
17	Thrust Vector Control	0.228	64	0.999
18	Antenna Servo (first)	0.468	2530	0.988
19	Antenna Servo (second)	0.468	2530	0.988
20	Earth Sensor	0.797	2530	0.980

3. Propulsion Subsystem

The propulsion subsystem selected for the Bus is the same subsystem designed for the orbiter in the Voyager Saturn I-B study. For information on the reliability of this subsystem, see GE Document No. 63SD801, Volume II, pages 4-67 through 4-72 inclusive.

C. LANDER

The function of the Lander is to monitor Martian atmospheric and surface conditions and to perform specified scientific experiments during the entry, descent and surface phases of the Lander mission. In addition, the acquired data must be recorded and periodically communicated to Earth.

The Lander vehicle design has been subdivided into six functional subsystems.

The mathematical model used to obtain the estimated reliability of the Lander system is

$$R_{(\text{Lander})} = R_{(\text{Communications})} \cdot R_{(\text{EP\&D})} \cdot R_{(\text{Propulsion \& Separation})} \cdot R_{(\text{Thermal Control})} \cdot R_{(\text{Retardation})} \cdot R_{(\text{Orientation})}$$

Substituting the computed reliability values tabulated in Table 5.2-3, gives

$$R_{(\text{Lander } 100 \text{ Hrs})} = (0.863)(0.970)(0.972)(0.957)(0.984)(0.993) = 0.760$$

$$R_{(\text{Lander } 3 \text{ Months})} = (0.815)(0.959)(0.972)(0.947)(0.984)(0.993) = 0.704$$

TABLE 5.2-3. SUMMARY OF RELIABILITY VALUES FOR LANDER SUBSYSTEMS

Lander Vehicle Subsystems	Reliability	
	100 Hours	3 Months
Communications	0.863	0.815
Electrical Power & Distribution	0.970	0.959
Propulsion & Separation	0.972	0.972
Thermal Control	0.957	0.947
Retardation	0.984	0.984
Orientation	0.993	0.993
Lander Vehicle Reliability (π R)	0.760	0.704

1. Communications Subsystem (See Figure 5.2-3)

a. Reliability Analyses

The four sequentially operated communication links are designed to fill the broad spectrum of requirements necessary for this Bus/Lander system.

Certain design features are incorporated in the subsystem to increase reliability, such as:

1. The duty cycle of components are kept to a minimum by turn-on-off programming or switching techniques
2. Majority logic will be used in the logic circuitry
3. Only the receiver circuits of the transponders will be energized during the transit phase
4. Standby redundancy is used in the Hi-Gain loop with dual klystrons as back-up
5. The omni VHF loop is only in operation during the pre-entry and descent phase of the mission.

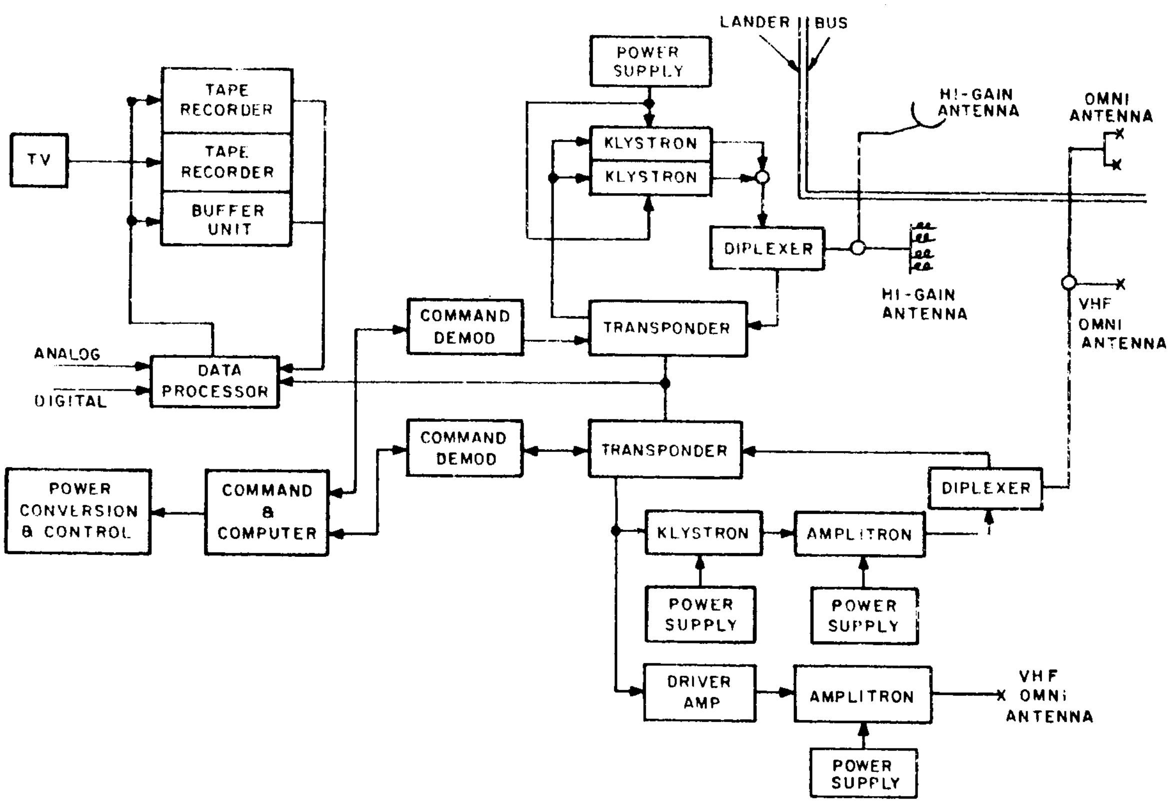


Figure 5.2-3. Simplified Block Diagram - Bus/Lander Communications Subsystem

b. Mathematical Model and Reliability Computations

The mathematical models define the components in each functional loop, the back-up capability and the mathematical interaction of the components.

$$R_{(\text{Lander Communications})} = R_{(\text{Omni Loop})} \cdot R_{(\text{Hi-Gain Loop})} \cdot R_{(\text{VHF Omni Loop})} \cdot R_{(\text{TV})} \cdot R_{(\text{Data Conversion})}$$

Substituting the computed reliability values from Table 5.2-4 into the mathematical model gives

$$R_{(\text{Communications})\,(100\text{ Hrs})} = (0.952)\,(0.920)\,(0.997)\,(0.989)\,(0.998) = 0.863$$

$$R_{(\text{Communications})\,(3\text{ Mos})} = (0.952)\,(0.876)\,(0.997)\,(0.985)\,(0.998) = 0.815$$

Where

$$R_{(\text{Omni Loop})} = R_1\,R_2\,R_3\,R_4\,R_5\,R_6$$

$$R_{(\text{Hi-Gain Loop})} = R_7\,R_8\,R_9\,(1 + \lambda t)\,R_{11}\,R_{12}$$

$$R_{(\text{VHF Omni Loop})} = R_{20}\,R_{21}\,R_{22}$$

$$R_{(\text{Data Conversion})} = R_{13}\,R_{14}\,R_{15}\,R_{16}\,R_{17}\,(1 + \lambda t)$$

$$R_{(\text{TV})} = R_{19}$$

The subscripts to each of the "R" factors refer to the identification numbers assigned to each of the components listed in Table 5.2-4.

TABLE 5.2-4. LANDER COMMUNICATIONS SUBSYSTEM RELIABILITY DATA

Comp. No.	Component		Failure Rate (%/1000 Hrs)	100-Hours Mission Effective Time (Hrs)	100-Hours Mission Rel.	3-Months Mission Effective Time (Hrs)	3-Months Mission Rel.
1	Transponder	(Omni)	1.060	2900	0.970	2900	0.970
2	Amplitron	(Omni)	1.536	396	0.994	396	0.994
3	Power Supply	(Omni)	0.249	396	0.999	396	0.999
4	Klystron	(Omni)	1.000	396	0.999	396	0.996
5	Power Supply	(Omni)	0.249	396	0.999	396	0.999
6	Command Demodulator	(Omni)	0.254	2900	0.993	2900	0.993

TABLE 5.2-4. LANDER COMMUNICATIONS SUBSYSTEM RELIABILITY DATA (Continued)

Comp. No.	Component	Failure Rate (%/1000 Hrs)	100-Hours Mission		3-Months Mission	
			Effective Time (Hrs)	Rel.	Effective Time (Hrs)	Rel.
7	Antenna & Diplexer (Hi-Gain)	1.820	2625	0.953	3675	0.935
8	Transponder (Hi-Gain)	1.060	2675	0.972	4775	0.951
9	Klystron (Hi-Gain)	1.000	149	0.999	1287	0.987
10	Klystron (Hi-Gain)	1.000	149	0.999	1287	0.987
11	Power Supply (Hi-Gain)	0.249	149	0.999	1287	0.999
12	Command Demodulator (Hi-Gain)	0.254	2675	0.993	4775	0.988
13	Command & Computer Equip.	0.340	1770	0.994	2820	0.991
14	Buffer Unit	3.500	101	0.999	311	0.999
15	Data Processor	0.698	101	0.999	311	0.998
16	Power Conversion & Control	0.002	Mission	0.998	Mission	0.998
17	Tape Recorder	3.180	141	0.995	1191	0.963
18	Tape Recorder	3.180	141	0.995	1191	0.963
19	Image Orthicon	1.256	191	0.998	191	0.998
20	Amplitron	1.536	92	0.999	92	0.999
21	Power Supply	0.249	92	0.999	92	0.999
22	Driver Amplifier	0.089	92	0.999	92	0.999
23	Hi-Gain Antenna	1.800	91	0.998	91	0.998

c. Stand-By Redundancy (Back-Up) in Communications S/S (Alternate Modes)

1. Start of transit phase to 2880 hours where omni loop is primary means of communication with hi-gain in standby redundancy.

   Mathematical Model:

$$R_{\substack{\text{(Communications)}\\ \text{up to 2880 hours}}} = R_{\text{(Omni Loop)}} + \frac{\lambda\text{ (omni loop)}}{\lambda\text{ (hi gain)} - \lambda\text{ (omni loop)}} \left[R_{\text{(omni loop)}} - R_{\text{(hi-gain loop)}} \right]$$

2. 2880 hours until mission completion where hi-gain loop is primary, with omni-loop in stand-by redundancy.

Mathematical Model:

$$R_{\substack{(\text{Communications})\\ 2880 + \text{hours}}} = R_{(\text{hi-gain loop})} + \frac{\lambda\ (\text{hi-gain loop})}{\lambda\ (\text{omni loop}) - \lambda\ (\text{hi-gain loop})} \left[R_{(\text{hi-gain loop})} - R_{(\text{omni loop})} \right]$$

2. Electrical Power and Distribution Subsystem (See Figure 5. 2-4)

a. Reliability Analysis

Generation of electrical power for the Bus/Lander System is provided by means of the Radioisotopic Thermoelectric Generator supplemented by rechargeable nickel-cadmium batteries during peak power periods. An additional function of the RTG is to provide a source of heat used for Lander thermal control. Power control is accomplished by switching functions initiated by the command portion of the communications system. Distribution will be provided by cabling harnesses to individual subsystems and components.

b. Mathematical Model and Reliability Computation

The mathematical model for the EP&D subsystem shows the components which are required to operate throughout the entire mission since this subsystem provides power to the Bus during transit as well as power to the Lander during Lander operation on Mars.

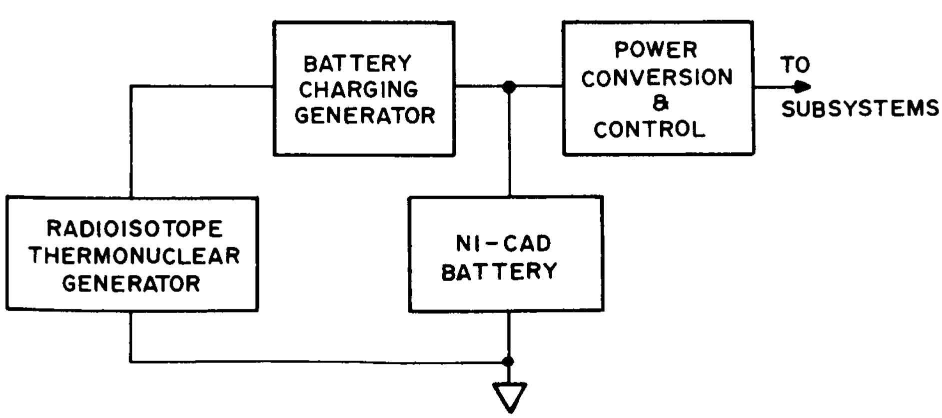


Figure 5. 2-4. Simplified Block Diagram - Lander Electrical Power and Distribution Subsystem

$$R_{(EP\&D)} = R_{(RTG)} \cdot R_{(Regulator)} \cdot R_{(Battery)} \cdot R_{(Cables\ \&\ Conn.)} \cdot R_{(PC\&C)}$$

Entering the proper component reliability values tabulated in Table 5.2-5 gives the estimated reliability of the Electrical Power and Distribution Subsystem.

$$(100\ \text{Hrs})\ R_{(EP\&D)} = (0.998)\ (0.988)\ (0.997)\ (0.994)\ (0.993) = 0.970$$

$$(3\ \text{Mos})\ R_{(EP\&D)} = (0.997)\ (0.983)\ (0.996)\ (0.992)\ (0.990) = 0.959$$

TABLE 5.2-5. LANDER ELECTRICAL POWER AND DISTRIBUTION SUBSYSTEM RELIABILITY DATA

Comp. No.	Component	Failure Rate (%/1000 Hrs)	100 Hrs Mission		3 Mos Mission	
			Effective Time (Hrs)	Rel.	Effective Time (Hrs)	Rel.
1	Radioisotope-Thermoelectric Generator	0.028	5540	0.998	7640	0.997
2	Regulator	0.211	5540	0.988	7640	0.983
3	Battery	0.050	5540	0.997	7640	0.996
4	Harness, Cabling, Connectors	0.100	5540	0.994	7640	0.992
5	Power Conversion & Control	0.175	5540	0.993	7640	0.990

3. Propulsion and Separation Subsystem (See Figure 5.2-5)

a. Reliability Analysis

This subsystem provides separation from the Bus, spin stabilization, and transfer into the planetary entry trajectory. Initial mechanical and electrical separation will be effected by explosive bolts and in-flight disconnects (each with redundant squibs). Subsequent separation and spin stabilization will be performed by a cold gas system and trajectory insertion by means of a solid rocket motor. All commands will be pre-programmed into the lander programmer and power will be supplied by the peaking batteries.

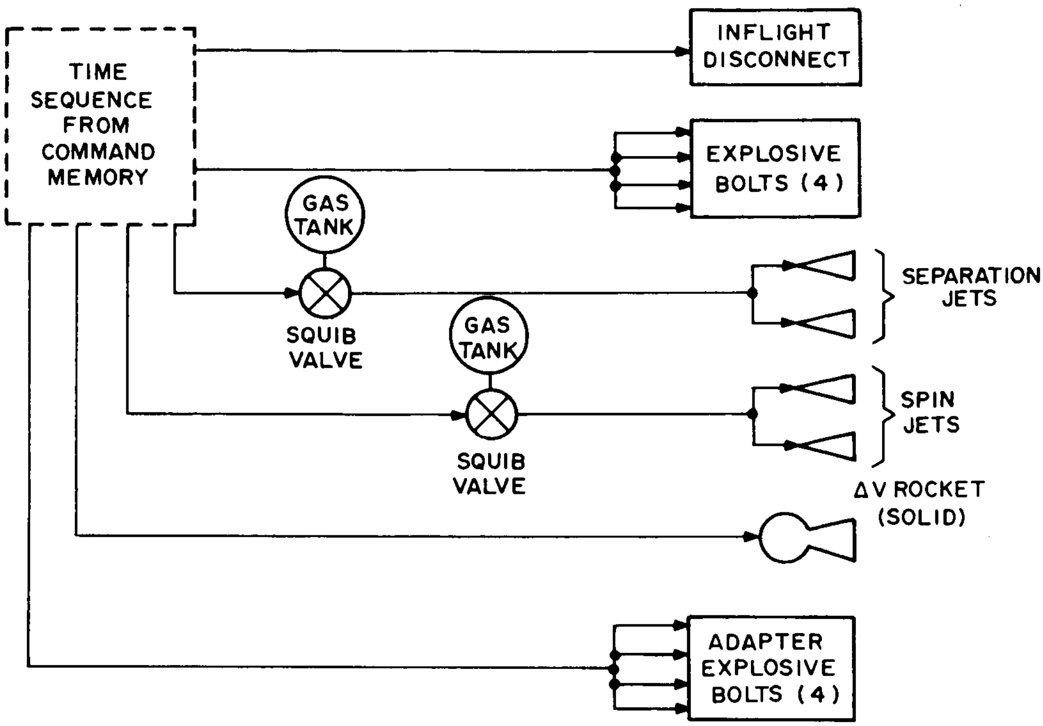


Figure 5.2-5. Simplified Block Diagram-Lander Propulsion and Separation Subsystem

b. Mathematical Model and Reliability Computation

The mathematical model for the Propulsion and Separation subsystem is

$$R_{(\text{Prop. \& Sep.})} = R_1 \cdot R_2 \cdot R_3 \cdot R_4 \cdot R_5 \cdot R_6 \cdot R_7 \cdot R_8 \cdot R_9 \cdot R_{10}$$

where the subscripts to each of the "R" factors refer to the identification numbers assigned to each of the subsystem components listed in Table 5.2-6. Where redundancy exists within a component, it has been considered in calculating the "R" value for that component.

Entering the proper component reliability values tabulated in Table 5.2-6, gives the estimated reliability of the Propulsion and Separation subsystem. The reliability of this subsystem is not affected by the duration of the surface mission on Mars.

$$R_{(\text{Prop. \& Sep.})} = (0.999)\ (0.997)\ (0.999)\ (0.992)\ (0.999)\ (0.999)$$

$$(0.992)\ (0.999)\ (0.999)\ (0.997)$$

$$= 0.972$$

TABLE 5.2-6. LANDER PROPULSION AND SEPARATION SUBSYSTEM RELIABILITY DATA

Comp. No.	Components	Qty.	Leakage F.R. $\lambda 10^{-5}$ Failures/Hr	Operation F.R. $\lambda 10^{-3}$ Failures/Operation	Transit Hours	Reliability	Remarks
1	Inflight Disconnect, Orbiter	1	-	1	5400	0.999	Red. Squibs
2	Orbiter Explosive Bolts	4	-	1	5400	0.997	Red. Squibs
3	Gas Tank (Separation)	1	0.008	-	5400	0.999	
4	Squib Valve (Separation)	1	0.113	1	5400	0.992	
5	Jets and Plumbing (Separation)	2	0.010	-	5400	0.999	
6	Gas Tank (Spin)	1	0.008	-	5400	0.999	
7	Squib Valve (Spin)	1	0.113	1	5400	0.992	
8	Jets and Plumbing (Spin)	2	0.010	-	5400	0.999	
9	Delta-V Solid Rocket	1	-	1	5400	0.999	Red. Squibs
10	Adapter Explosive Bolts	4	-	1	5400	0.997	Red. Squibs

4. Thermal Control Subsystem (See Figure 5.2-6)

a. Reliability Analysis

This subsystem provides active thermal control for the lander. The prime purpose of the subsystem is to dissipate excess heat generated by the RTG. This is accomplished by convection and thermal radiation during the in-transit and surface phases, and by liquid evaporation during boost and entry. A portion of the excess heat is utilized to maintain the temperature of internal components within specified design limits.

Working and standby redundancy are used extensively to reduce the probability of failure of the subsystem.

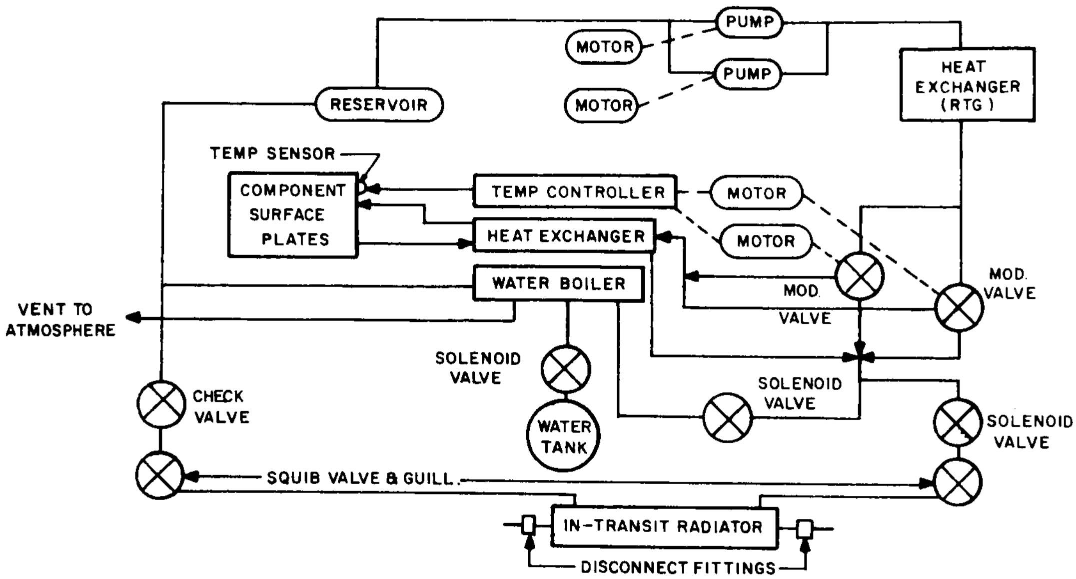


Figure 5. 2-6. Block Diagram-Thermal Control Subsystem

b. Mathematical Model and Reliability Computation

The mathematical model for the Thermal Control Subsystem is

$$R_{(\text{Thermal Control})} = R_1 \cdot R_2 \cdot R_3 \cdot R_4 \cdot R_5 \cdot R_6 \cdot R_7 \cdot \left[1 + (\lambda_6 + \lambda_7)t\right] \cdot \left[1 - (1-R_9)(1-R_{10})\right] \cdot \left[1 - (1-R_{10})(1-R_{11})\right] \cdot R_{12} \cdot R_{13} \cdot R_{14} \cdot R_{15} \cdot \left[1 + (\lambda_{14} + \lambda_{15})t\right] R_{16} \cdot R_{17} \cdot R_{18}$$

where the subscripts to each of the "R" factors refer to identification numbers assigned to each of the subsystem components listed in Table 5. 2-7.

Entering the proper component reliability values tabulated in Table 5. 2-7 gives the estimated reliability of the Thermal Control subsystem.

$$R_{\substack{\text{(Thermal Control)}\\ \text{(100 Hours)}}} = (0.998)(0.994)(0.994)(0.994)(0.994)(0.994)$$

$$(0.994)\left[1 + (0.224 \times 10^{-5})(5500)\right]\left[1 - (1 - 0.999)^2\right]$$

$$\left[1 - (1 - 0.999)^2\right](0.994)(0.998)(0.992)$$

$$\left[1 + (0.257 \times 10^{-5})(5500)\right](0.999)(0.997)(0.994)$$

$$= 0.957$$

$$R_{\substack{\text{(Thermal Control)}\\ \text{(3 Months)}}} = (0.998)(0.994)(0.994)(0.992)(0.992)(0.991)$$

$$(0.991)\left[1 + (0.224 \times 10^{-5})(7600)\right]\left[1 - (1 - 0.998)^2\right]$$

$$\left[1 - (1 - 0.998)(1 - 0.999)\right](0.994)(0.997)(0.989)$$

$$(0.999)\left[1 + (0.257 \times 10^{-5})(7600)\right](0.999)(0.996)(0.992)$$

$$= 0.947$$

TABLE 5.2-7. LANDER THERMAL CONTROL SUBSYSTEM RELIABILITY DATA

Comp. No.	Component	Qty.	Failure Rate (%/1000 Hrs)	100 Hrs Mission Effective Time (Hrs)	100 Hrs Mission Rel.	3 Mos Mission Effective Time (Hrs)	3 Mos Mission Rel.
1	Water Tank	1	0.035	5400	0.998	5400	0.998
2	Water Boiler	1	0.110	5400	0.994	5400	0.994
3	Solenoid Valve	1	0.113	5400	0.994	5400	0.994
4	RTG Heat Exchanger	1	0.110	5500	0.994	7600	0.992
5	Liquid to Liquid Heat Exchanger	1	0.110	5500	0.994	7600	0.992
6	Pumps	2	0.112	5500	0.994	7600	0.991
7	DC Motors	2	0.112	5500	0.994	7600	0.991
8	Solenoid Valve	1	0.113	-	-	-	-
9	Solenoid Valve	1	0.113	100	0.999	2200	0.998
10	Squid Valve and Guillotine	2 1	0.113	100	0.999	2200	0.998
11	Check Valve	1	0.011	100	0.999	2200	0.999
12	In-Transit Radiator	1	0.110	5400	0.994	5400	0.994

TABLE 5.2-7. LANDER THERMAL CONTROL SUBSYSTEM RELIABILITY DATA (Cont'd)

Comp. No.	Component	Qty.	Failure Rate (%/1000 Hrs)	100 Hrs Mission Effective Time (Hrs)	100 Hrs Mission Rel.	3 Mos Mission Effective Time (Hrs)	3 Mos Mission Rel.
13	Accumulator	1	0.035	5500	0.998	7600	0.997
14	Modulation Valves	2	0.145	5500	0.992	7600	0.989
15	DC Motors	2	0.112	64	0.999	274	0.999
16	Temperature Sensor	1	0.015	5500	0.999	7600	0.999
17	Temperature Controller	1	0.047	5500	0.997	7600	0.996
18	Plumbing, Fittings	-	0.110	5500	0.994	7600	0.992

5. Retardation Subsystem (See Figure 5.2-7)

a. <u>Reliability Analysis</u>

This subsystem will retard the Lander vehicle during atmospheric entry to provide time for experimentation during descent and to minimize landing impact. Retardation will be performed by means of a deceleration parachute, a main parachute, and

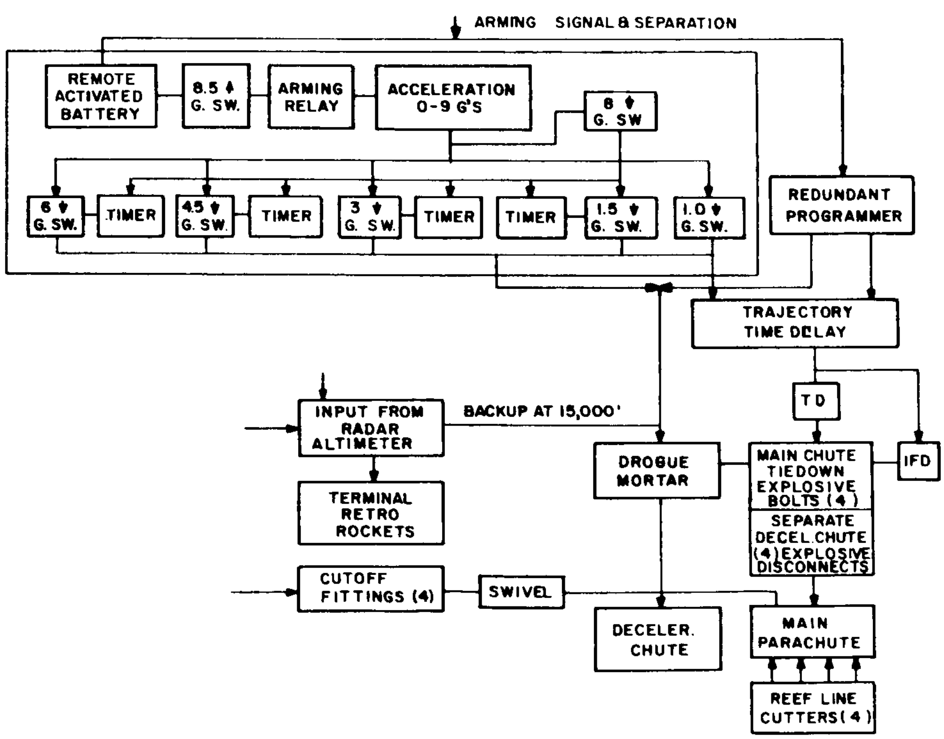


Figure 5.2-7. Block Diagram-Lander Retardation Subsystem

retro-rockets. Landing impact will be absorbed by the structural honeycomb crush-up material. As in the orientation subsystem, the retardation design must accommodate a wide range of environmental conditions due to trajectory uncertainty at entry and the unknown Mars atmosphere. Redundant programming and trajectory sensing, as well as redundant initiation of pyrotechnics, will be used. This subsystem, by necessity, will be completely independent of other subsystems with respect to programming and power requirements in order to assure successful entry and landing.

b. Mathematical Model and Reliability Computation

The mathematical model for the Retardation System is

$$R_{(\text{Retardation})} = \left[1 - (1-R_1 R_2 R_3^{\,7} R_4^{\,4})^2\right] \cdot R_5 \cdot R_6 \cdot R_7 \cdot R_8 \cdot R_9 \cdot R_{10}^{\,4} \cdot R_{11}^{\,4} \cdot R_{12} \cdot R_{13} \cdot \left[1 - (1 - R_{14})^4\right] \cdot R_{15} \cdot R_{16}$$

where the subscripts to each of the "R" factors refer to the identification numbers assigned to each of the subsystem components listed in Table 5.2-8. Where redundancy exists within a component, it has been included in the computation of the "R" value for that component.

The reliability of the Lander Retardation subsystem is not affected by the duration of the Lander surface mission. Therefore, entering the "R" values given in Table 5.2-8 into the mathematical model gives

$$R_{(\text{Retardation})} = 0.984$$

6. Orientation Subsystem (See Figure 5.2-8)

a. Reliability Analysis

Orientation of the Lander vehicle on the surface of Mars, including the deployment of experiments, is performed by the orientation subsystem. The selection of the final design configuration of side orientation was based on the minimum number of functions and operations required to orient. The major problem in the subsystem design was to accommodate the range of surface terrain conditions which could be expected and initial lander orientation after impact. The sequences for orientation is pre-programmed in the command programmer and will repeat until orientation is achieved, barring extreme circumstances.

TABLE 5.2-8. LANDER RETARDATION SUBSYSTEM RELIABILITY DATA

Comp. No.	Components	Qty.	Operation F.R. $\lambda 10^{-3}$ Failures/ Mission	Relia- bility	Remarks
1	Remote Activated Batteries	2	2	0.998	(2 Redundant Squibs)
2	Arming Relay	2	<.1	>0.9999	
3	G Switches	14	2	0.998	Redundant Programmers
4	Timers	8	1	0.999	
5	Time Delay, Trajectory	1	<.1	>0.9999	
6	Drogue Mortar	1	1	0.999	(2 Red. Squibs)
7	Decel. Chute	1	1	0.999	
8	Inflight Disconnect	1	1	0.999	(2 Red.Squibs)
9	Time Delay	1	<.1	>0.9999	
10	Tie-Down Explosive Bolts	4	1	0.999	(2 Red. Squibs)
11	Decel. Chute Explosive Disconnects	4	1	0.999	(2 Red. Squibs)
12	Main Parachute	1	1	0.999	
13	Swivel	1	<.1	>0.9999	
14	Reef Line Cutters	4	<.1	>0.9999	One of four required
15	Cutoff Fittings	4	<.1	>0.9999	(2 Red. Squibs)
16	Retro-Rockets	2	1	0.998	(2 Red. Squibs)

b. Mathematical Model and Reliability Computation

The mathematical model for the Orientation subsystem is

$$R_{(\text{Orientation})} = R_1 \cdot R_2 \cdot R_3 \cdot R_4 \cdot R_5 \cdot R_6 \cdot R_7 \cdot R_8 \cdot R_9$$

where the subscripts to each of the "R" factors refer to the identification numbers assigned to each of the subsystem components listed in Table 5.2-9. Each component "R" value given in Table 5.2-9 has been calculated for the total required quantity of that component.

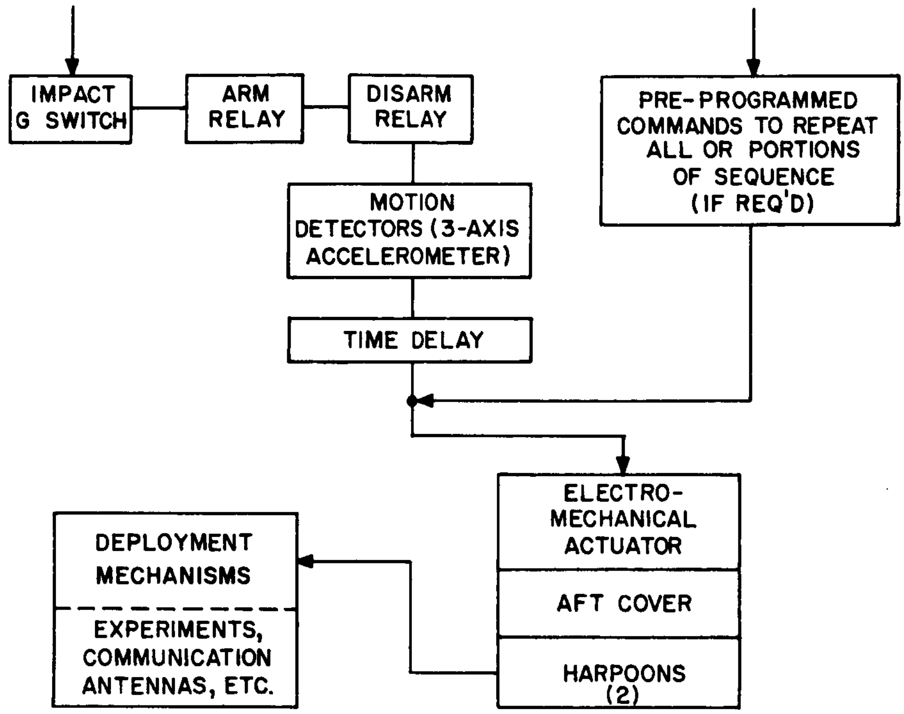


Figure 5. 2-8. Simplified Block Diagram-Lander Orientation Subsystem

TABLE 5. 2-9. LANDER ORIENTATION SUBSYSTEM RELIABILITY DATA

Comp. No.	Component	Qty.	Operation F. R. $\lambda 10^{-3}$ Failures/ Mission	$\Sigma N \lambda M$	Relia-bility	Remarks
1	G Switch, Impact	1	1	0. 001	0. 999	
2	Arm Relay	1	<0. 1	<0. 0001	>0. 9999	
3	Disarm Relay	1	<0. 1	<0. 0001	>0. 9999	
4	Mercury Switches	3	<0. 1	<0. 0001	>0. 9999	
5	Time Delay	1	<0. 1	<0. 0001	>0. 9999	
6	Deployment Mechanisms	≈10	0. 1	0. 001	0. 999	Initiation by Red. Squibs
7	Electro-Mechanical Actuator	1	1	0. 001	0. 999	
8	Tilt Bar	1	<0. 1	<0. 0001	>0. 9999	
9	Harpoons	2	1	0. 002	0. 998	Initiation by Red. Squibs

The reliability of the Lander Orientation subsystem is not affected by the duration of the Lander surface mission. Entering the "R" values given in Table 5.2-9 into the mathematical model gives

$$R_{(orientation)} = 0.993$$

5.2.3 ORBITER SYSTEM

A. SYSTEM DEFINITION

The Voyager Orbiter System is composed of a single vehicle with the capability of orbiting Mars for a six-month time period during which it will acquire scientific information about the Martian atmosphere and the space environment.

Additional system definition and reliability analysis of the Orbiter system is given in Section 2.6.3(B).

B. ORBITER VEHICLE

The Orbiter vehicle has multiple functions in the mission. During the transit phase, it is a communications link with Earth, performs maneuvers and transmits diagnostic data. In the orbiting phase, it acquires and transmits scientific information to Earth.

The Orbiter vehicle contains five major functional subsystems. The mathematical model of the Orbiter is

$$\begin{aligned} R(\text{Orbiter}) = {} & R(\text{Communications}) \cdot R(\text{G \& C}) \\ & \cdot R(\text{Power Supply}) \cdot R(\text{Hot Gas Prop.}) \\ & \cdot R(\text{Cold Gas Prop.}) \end{aligned}$$

Substituting computed reliability values in the above mathematical model gives

(100-Hour Orbit) $$R(\text{Orbiter}) = (0.866)(0.912)(0.980)(0.998)(0.996) = 0.768$$

(3-Month Orbit) $$R(\text{Orbiter}) = (0.793)(0.831)(0.973)(0.998)(0.990) = 0.633$$

This is summarized in Table 5.2-10.

TABLE 5.2-10. SUMMARY OF RELIABILITY VALUES FOR ORBITER SUBSYSTEMS

Orbiter Vehicle Subsystems	Reliability	
	100-Hour Orbit	3-Month Orbit
Communications	0.866	0.793
G & C	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 Vehicle	0.768	0.633

1. Communications Subsystem (See Figure 5.2-9)

a. Reliability Analysis

The Communications subsystem of the Orbiter is similar to the communications subsystem of the Bus/Lander with the exception that the VHF omni link for pre-entry and descent is omitted and also that three tape recorders are used instead of the two in the Bus/Lander. The design features mentioned in the Bus/Lander Communications analysis (Section 5.2.2(c)(1)) also apply to the Orbiter communications.

b. Mathematical Model and Reliability Computations

The mathematical model given below defines the components in each functional loop, standby redundancy and the mathematical interaction of the components.

$$R(\text{Communications}) = R(\text{Omni Loop}) \cdot R(\text{Hi-Gain Loop}) \cdot R(\text{TV}) \cdot R(\text{Data Conversion})$$

where

$$R\ (\text{Omni Loop}) = R_1 \cdot R_2 \cdot R_3 \cdot R_4 \cdot R_5 \cdot R_6 \cdot R_{19}$$

$$R\ (\text{Hi-Gain Loop}) = R_7 \cdot R_8 \cdot R_9 \cdot (1 + \lambda t)\, R_{11} \cdot R_{12}$$

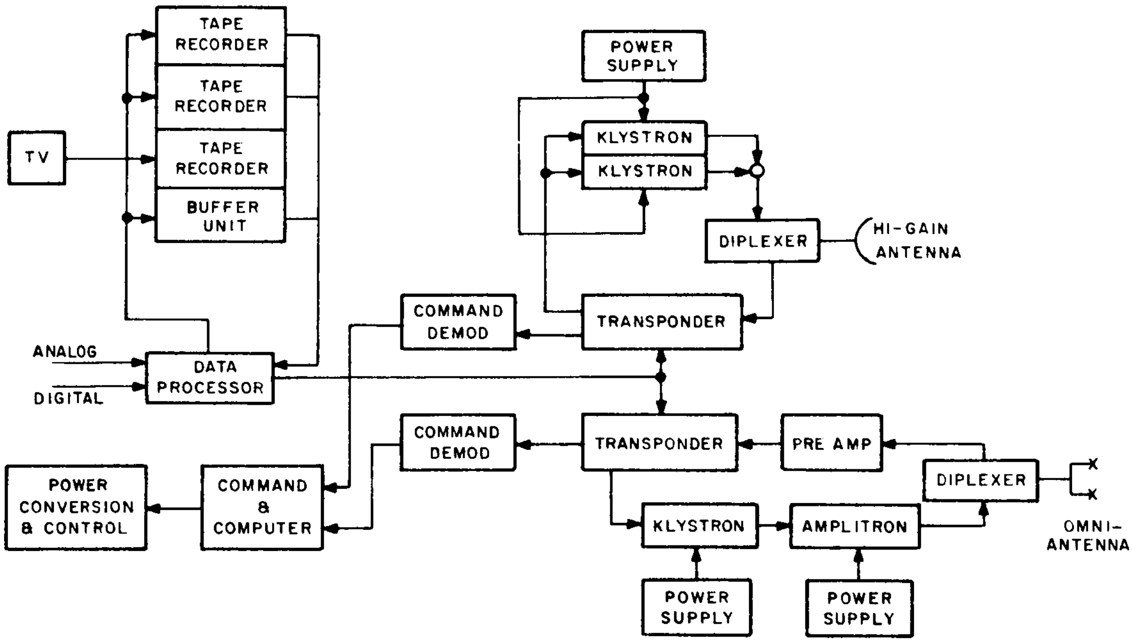


Figure 5. 2-9. Simplified Block Diagram-Orbiter Communications Subsystem

$$R(TV) = R_{18}$$

$$R(\text{Data Conv}) = R_{13} \cdot R_{14} \cdot R_{15} \cdot R_{16} \cdot R_{17}$$

The subscripts to each of the "R" factors refer to the identification numbers assigned to each of the subsystem components listed in Table 5. 2-11.

Substituting computed reliability values into the mathematical model gives

$$(100 \text{ Hrs}) \quad R(\text{Communications}) = (0.952)\,(0.919)\,(0.998)\,(0.990) = 0.864$$

$$(3 \text{ Mo}) \quad R(\text{Communications}) = (0.952)\,(0.855)\,(0.998)\,(0.981) = 0.793$$

TABLE 5.2-11. ORBITER COMMUNICATIONS SUBSYSTEM RELIABILITY DATA

Comp. No.	Component	Failure Rate (%/1000 Hrs)	100 Hours Orbit Effective Time (Hrs)	100 Hours Orbit Rel.	3 Months Orbit Effective Time (Hrs)	3 Months Orbit Rel.
1	Transponder (Omni)	1.060	2890	0.970	2890	0.970
2	Amplitron	1.536	396	0.994	396	0.994
3	Power Supply	0.249	396	0.999	396	0.999
4	Klystron	1.000	396	0.996	396	0.996
5	Power Supply	0.249	396	0.999	396	0.999
6	Command Demodulator	0.254	2890	0.993	2890	0.993
7	Hi-Gain Antenna & Diplexer	1.820	2675	0.953	4775	0.917
8	Transponder (Hi-Gain)	1.060	2675	0.972	4775	0.951
9	Klystron	1.000	195	0.999	2295	0.978
10	Klystron	1.000	195	0.999	2295	0.978
11	Power Supply	0.249	195	0.999	2295	0.995
12	Command Demodulator	0.254	2675	0.993	4775	0.988
13	Command & Computer Equip.	0.340	1820	0.994	3920	0.987
14	Buffer Unit	3.500	101	0.999	321	0.999
15	Data Processor	0.698	101	0.999	321	0.998
16	Power Conversion & Control	0.002	Mission	0.998	Mission	0.998
17	Tape Recorders	3.180(ea)	155	0.999	1497	0.999
18	Image Orthicon	1.256	191	0.998	191	0.998
19	PreAmp (Omni)	0.012	2890	0.999	2890	0.999

NOTES 1. All antennas and diplexers not listed in above table are considered to have a reliability of approximately 1.0 due to extremely low failure rates.

2. The tape recorders, Comp. No. 17, are three recorders in parallel with only two out of the three required for 100% operation.

2. Guidance & Control Subsystem (See Figures 5. 2-10 and 5. 2-11)

a. Reliability Analysis

The Guidance and Control subsystem of the Orbiter is similar to that utilized on the Bus in the Bus/Lander system with the exception that the Orbiter G & C contains a three-axis PHP. Thus the Orbiter G & C has three functional areas which are

1. Attitude Control
2. Earth Tracker and Antenna Drive
3. PHP Axes Control.

See Section 5. 2. 2(B)(2) for further information about the Guidance and Control subsystem.

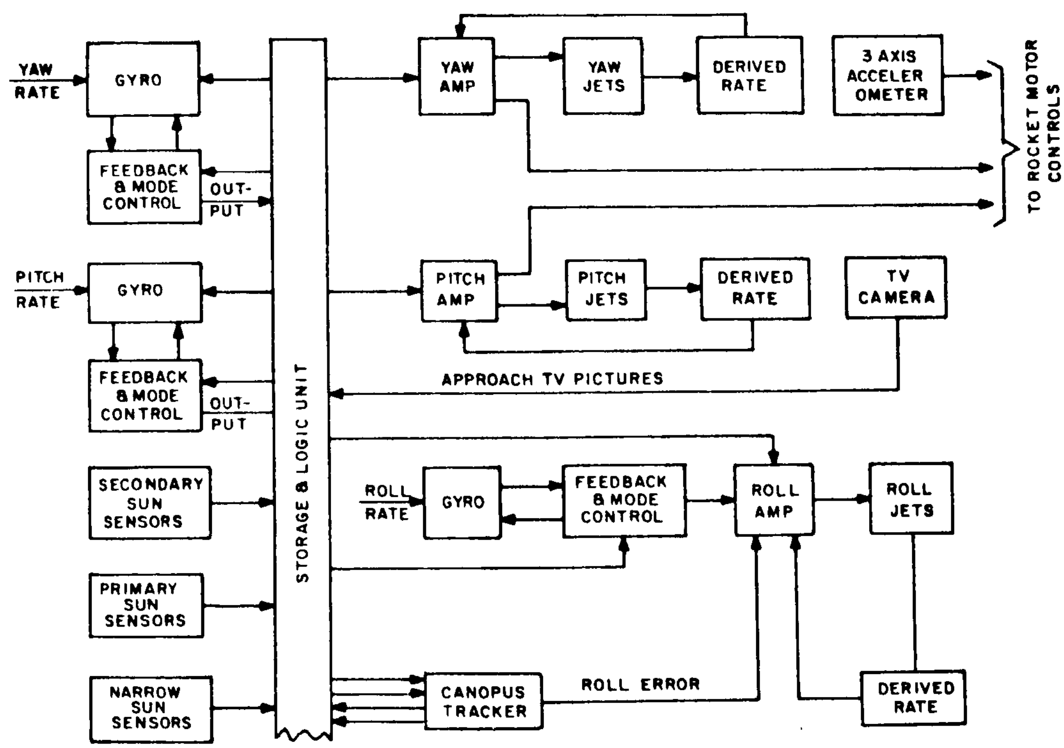


Figure 5. 2-10. Simplified Block Diagram-Orbiter Guidance and Control Subsystem

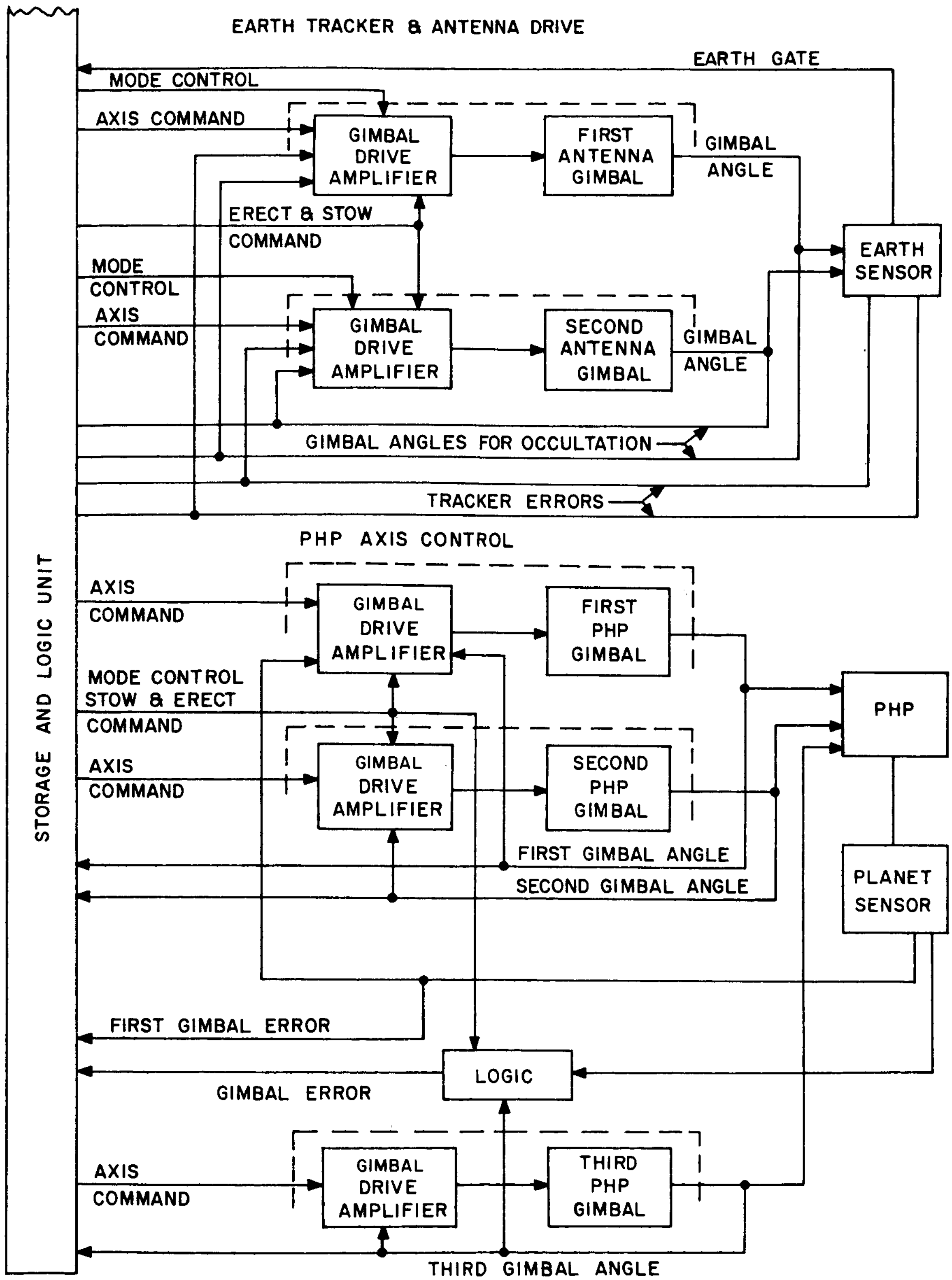


Figure 5. 2-11. Simplified Block Diagram-Orbiter Guidance and Control Subsystem

b. Mathematical Model and Reliability Computations

The following mathematical model for the Guidance and Control subsystem defines the mathematical interaction of all the components that are required to operate throughout the mission:

$$R(G\&C) = R(\text{Star Tracker}) \cdot R_{(\text{Narrow Sun Sensor})} \cdot R_{(\text{Primary Sun Sensor})}$$

$$\cdot R_{(\text{Secondary Sun Sensor})} \cdot \left[1 - (1 - R_{\text{amplifier}})^2\right]^3$$

$$\cdot R_{(\text{Storage \& Logic Unit})} \cdot R^3_{(\text{Gyro})} \cdot R_{(\text{Earth Sensor})}$$

$$\cdot R^2_{(\text{Antenna Servo})} \cdot R^3_{(\text{PHP Servo})}$$

$$\cdot R^3_{(\text{Feedback \& Mode Control Amplifier})} \cdot R^2_{(\text{Thrust Vector Control})}$$

$$\cdot R_{(\text{Accelerometer})} \cdot R_{(\text{Planet Sensor})} \cdot R_{(\text{PHP Logic})}$$

Entering the proper component reliability values tabulated in Table 5.2-12 gives the estimated reliability of the G & C subsystem. Where redundancy exists within a component, it has been taken into consideration when calculating the "R" value for that component.

$$(100\ \text{Hrs})\ R_{(G\ \&\ C)} = (0.986)\ (0.999)\ (0.996)\ (0.999)\ \left[1 - (1-0.995)^2\right]^3$$

$$(0.986)\ (0.999)^3\ (0.979(\ (0.988)^2\ (0.999)^3$$

$$(0.998)^3\ (0.999)^2\ (0.999)\ (0.999)\ (0.999)$$

$$= 0.912$$

$$(3\ \text{Mos})\ R_{(G\ \&\ C)} = (0.981)\ (0.999)\ (0.994)\ (0.999)\ \left[1-(1-0.993)^2\right]^3$$

$$(0.980)\ (0.998)^3\ (0.964)\ (0.978)^2\ (0.989)^3$$

$$(0.996)^3\ (0.999)^2\ (0.999)\ (0.996)\ (0.998)$$

$$= 0.831$$

TABLE 5.2-12. ORBITER GUIDANCE & CONTROL SUBSYSTEM RELIABILITY DATA

Comp. No.	Component	Failure Rate (%/1000 Hrs)	100 Hours Orbit		3 Months Orbit	
			Effective Time (Hrs)	Rel.	Effective Time (Hrs)	Rel.
1	Gyro (yaw)	0.500	155	0.999	335	0.998
2	Gyro (pitch)	0.500	155	0.999	335	0.998
3	Gyro (roll)	0.500	155	0.999	335	0.998
4	Feedback & Mode Cont. (yaw)	1.200	155	0.998	335	0.996
5	Feedback & Mode Cont. (pitch)	1.200	155	0.998	335	0.996
6	Feedback & Mode Cont. (roll)	1.200	155	0.998	335	0.996
7	Power Amplifier (yaw)	0.093	5510	0.995	7610	0.993
8	Power Amplifier (pitch)	0.093	5510	0.995	7610	0.993
9	Power Amplifier (roll)	0.093	5510	0.995	7610	0.993
10	Storage & Logic Unit	0.440	3310	0.986	4570	0.980
11	Secondary Sun Sensors	0.080	136	0.999	136	0.999
12	Primary Sun Sensors	0.080	5510	0.996	7610	0.994
13	Narrow Sun Sensors	0.010	5510	0.999	7610	0.999
14	Star Tracker	0.256	5510	0.986	7610	0.981
15	Accelerometer	0.178	66	0.999	96	0.999
16	Thrust Vector Control	0.228	66	0.999	96	0.999
17	Thrust Vector Control	0.228	66	0.999	96	0.999
18	Antenna Servo (first)	0.468	2630	0.988	4830	0.978
19	Antenna Servo (second)	0.468	2630	0.988	4830	0.978
20	Earth Sensor	0.797	2630	0.979	4830	0.964
21	PHP Servo (first)	0.468	191	0.999	2291	0.989
22	PHP Servo (second)	0.468	191	0.999	2291	0.989
23	PHP Servo (third)	0.468	191	0.999	2291	0.989
24	Planet Sensor	0.176	191	0.999	2291	0.996
25	PHP Logic	0.094	191	0.999	2291	0.998

3. Power Supply Subsystem (See Figure 5.2-12)

a. Reliability Analysis

The Power Supply subsystem uses silicon solar cells as the primary power source, with a nickel-cadmium battery as a back-up for peak power loads. A regulator limits the average battery charging current and the maximum voltage imposed on the battery to prescribed nominal values. The regulator will also serve as a battery over-voltage control in the event that chemical degradation of the battery allows an over-voltage to exist.

All components within the subsystem, except the battery, are in continuous usage during the mission. The battery is trickle charged from the solar array, and is estimated to be in use for only the high rates of acquisition (TV observation) during the orbiting phase and for midcourse maneuvering and orbit injection during the transit phase.

b. Mathematical Model & Reliability Computations

The mathematical model for the Power Supply subsystem is:

$$R_{(\text{power supply})} = R_{(\text{solar array})} \cdot R_{(\text{battery})} \cdot R_{(\text{Regulator})} \cdot R_{(\text{power control unit})}$$

Entering the proper component reliability values tabulated in Table 5.2-13 gives the estimated reliability of the Power Supply subsystem.

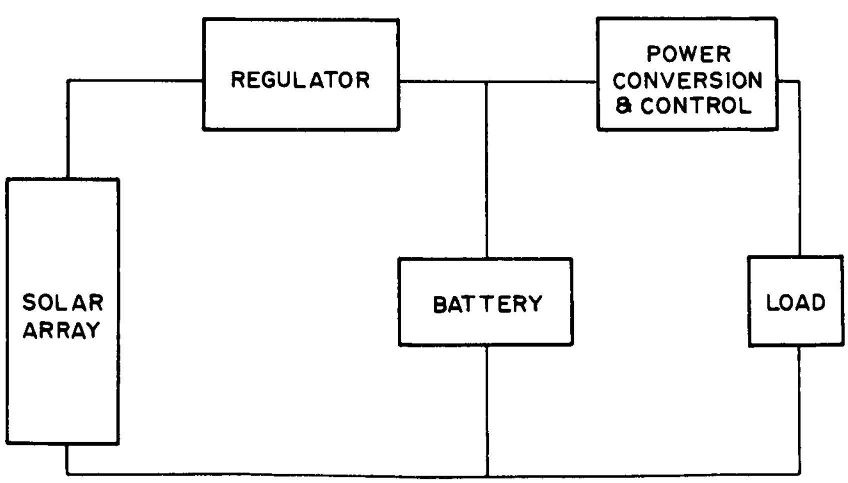


Figure 5.2-12. Simplified Block Diagram-Orbiter Power Subsystem

$$(100\ \text{Hrs})\ R_{(\text{power supply})} = (\sim 1.0)\ (0.999)\ (0.988)\ (0.992)$$

$$= 0.980$$

$$(3\ \text{Mos})\ R_{(\text{power supply})} = (\sim 1.0)\ (0.999)\ (0.984)\ (0.989)$$

$$= 0.973$$

TABLE 5.2-13. ORBITER POWER SUPPLY SUBSYSTEM RELIABILITY DATA

Comp. No.	Component	Failure Rate (%/1000 Hrs)	100 Hours Orbit Effective Time (Hrs)	100 Hours Orbit Rel.	3 Month Orbit Effective Time (Hrs)	3 Month Orbit Rel.
1	Solar Array	0.0001	5510	~1.0	7610	~1.0
2	Regulator	0.211	5510	0.988	7610	0.984
3	Battery	0.050	100	0.999	275	0.999
4	Power Control Unit	0.150	5510	0.992	7610	0.989

4. Propulsion Subsystem

The propulsion subsystem selected for the Orbiter is the same subsystem designed for the Orbiter in the Voyager Saturn I-B study. For information on the reliability of this subsystem, see GE Document No. 63SD801, Volume II, Pages 4-67 through 4-72 inclusive.

5.3 ANALYSIS OF THE TITAN IIIC VS. SATURN IB BASED UPON ATTAINABLE MISSION VALUES

5.3.1 MISSION VALUE ANALYSIS

The mission values of each instrument have been determined relative to the value of a completely successful mission by:

1. Establishing a point value for each scientific instrument
2. Tabulating these for all the instruments proposed and reviewing them with each of the scientists available to the study
3. Iterating this procedure until a reasonably firm mutual concurrence in these point values was obtained
4. Converting these point values into percent of a total available mission value in which one completely successful lander mission carrying all the lander instruments selected plus one completely successful orbiter mission carrying all the orbiter instruments selected was considered as the basic 100% available mission value to be used for subsequent components and comparisons
5. Dividing each of these evaluations by the weight which that instrument would add to the payload weight (including the weight required for any auxiliary mechanisms, brackets, wire, etc., which were unique to that instrument) to obtain mission value available per pound.

In this connection, it was evaluated (by the same joint scientific opinion) that the relative values contributed by the orbiter instrumentation represented 30 percent, the entry (atmospheric, etc.) data obtained by the lander prior to impact on the planet surface represented 10 percent and the values obtained from the surface of the planet at the location at which the lander first came to rest represented 60 percent of the mission value available. These judgements were the result of many iterations and in any future mission or study would require complete reanalysis and reappraisal. Scientific instrument weight was adjusted for an allowance for the weight of the hardware items, cables, connectors, etc., common to the scientific instruments as a whole. The resulting comparisons are based upon the "Net Scientific Payload Weight."

Each instrument's function was then individually considered by the same scientists with respect to the time in the sequence of mission events at which its scientific data would become available to be transmitted to the earth. In general, each instrument's value increased as subsequent readings were obtained. As these data became repetitions of prior readings, rather than new unique bits of information, the rate of accumulating mission value decreased becoming asymptotic at 100 percent of that instruments available point value. Iterations of this were also conducted until a reasonably firm agreement was concurred in by both scientists and the individual engineers directly involved.

The scientific values used were based upon the assumption that while some prior indications of atmospheric data might be available, the initial scientific value of the knowledge of chemical composition of the atmosphere, sand, dust, winds, radiation levels, etc., were still to be obtained. As successive planetary missions succeed, the composition and values of the scientific payload would, of course, be altered. New instruments for the classification of "life" data would replace and supplement life detection instruments. The same evolution of mission objectives and values will apply to all categories of instruments.

5.3.2 SCIENTIFIC VALUE ASSIGNMENT

In this Titan IIIC-Voyager Study the instruments selected and evaluated during the Saturn IB-Voyager Study were reviewed by the scientists and some instruments were added to those previously identified. However, since this area is one specifically excluded from any resolution by either of these studies it must be kept in mind that these evaluations are indicative only. Despite this limitation, this itemized and cumulative method of approach to the subject of relative mission value is so essential to the determination of the spacecraft requirements and to the selection of the more valuable mission and system configurations that it has been used in both studies and will be used on such future studies as involve these variables. The rates at which mission values accrue after the time of arrival at the planet is provided by Table 5.3-1 (compare with 4.5.1-3(A) & (B) from page 4-33, Vol. II, 63SD801).

Which of these mission objectives will be attained during any given launching cannot be determined with any certainty prior to their actual success. The probability of success (i.e., reliability) of each individual scientific instruments operation after it has been subjected to humidity, dust, handling, sterilization, etc. prior to launch, as well as after the launch, transit, impact and guidance and control functions and environments have been completed is directly involved. The reliability of the instrument is of equally great importance to that of the spacecraft subsystems themselves

TABLE 5.3-1a. AVAILABLE MISSION VALUES - ORBITER

ORBITER	Instrument Identifying Number	Total Orbiter Scientific Value Available (in percent)	ORBITER VALUE AVAILABLE VS TIME AFTER ARRIVAL (%)							
Name of Instrument			12 Hours	24 Hours	100 Hours	15 Days	30 Days	45 Days	60 Days	LBS
1. Magnetic Field	I-23	2	75	85	95	97	98	98.5	99+	
2. IR Flux	I-2	1	25	45	65	77	85	90	95	
3. Radiameter UV to IR	I-79	1	75	85	95	97	98	98.5	99+	
4. TV (Multicolor)	TV	20	20	40	60	74	80	87	98+	
5. Charged Particle Flux	I-12	1	75	85	95	97	98	98.5	99+	
6. Far UV — Radiameter	I-96	1	75	85	95	97	98	98.5	99+	
7. Micrometeoroids	I-55	1	60	75	90	96	98	98.5	99+	
8. Ionospheric Profile	I-85	1.5	75	85	95	97	98	98.5	99+	
9. Polarimeter	I-68	.5	75	85	95	97	98	98.5	99+	
10. IR Spectrum	I-1	1	60	75	90	96	98	98.5	99+	
Saturn IB Values		30	10.7	15.9	21.1	23.3	25.7	27.2	30.	215
4. TV (Vidicon Sterio Map)			50	60	70	100				
11. Orbit Decay - Upper Atmos.		4					80	100		+80
12. TV ("1 METER" High Resolution Package)		20					80	100		+50
Titan IIIC Values		54	16.7	19.9	23.1	28.5	48.9	53.7	54	345

NOTE: Eleven additional instruments have been identified as alternatives to using the extra Titan IIIC orbiter payload capability to obtain upper atmosphere and high resolution "1 meter" TV mapping. Since it was considered that their additional mission value was considerably less (i.e., < 13%) than could be obtained by a controlled orbit with a perigee at 100 N.M., the data for comparison is as noted above. The use of the additional instruments rather than the low orbit and "1 meter" mapping has been considered as an Alternate "A".

TABLE 5.3-1b. AVAILABLE MISSION VALUES - LANDER

LANDER: Name of Instrument	In Rover	Instrument Identifying Number	Total Entry Value Available (in percent)	Total Surface Value Available (in percent)	SURFACE VALUE AVAILABLE TIME AFTER ARRIVAL (%): 12 Hours	24 Hours	96 Hours	30 Days	60 Days
1. Temperature		I-24	1	3	50	60	90	95	98
2. Sounds		I-34	-	3	75	85	90	95	99
3. Pressure		I-17	1/2	1/2	90	92	97	99	99+
4. Density		I-20	2	4	50	60	90	95	98
5. Multiple Chamber	*	I-54	-	10	75	90	95	99	99+
6. Surface Penetration Hardness	*	I-25	-	2	95	95	95	98	99
7. Photoautotroph	*	I-62	-	3	75	90	95	99	99+
8. Light Intensity (Sun Sensor)		I-84	-	1/2	50	75	90	98	99
9. Composition, H_2O		I-44	1/2	1/2	90	92	97	99	99+
10. Composition, O_2		I-45	1/2	1/2	90	92	97	99	99+
11. Turpidity & PH	*	I-53	-	3	75	90	95	99	99+
12. Wind Speed & Direction		I-67	-	2	40	60	80	90	95
13. Gas Chromatograph		I-8	2	2	90	92	97	99	99+
14. Composition, N_2		I-48	1/4	1/4	90	92	97	99	99+
15. Composition, CO_2		I-49	1/4	1/4	90	92	97	99	99+
16. Soil Moisture	*	I-70	-	1	95	96	97	98	99
17. TV Camera, Panorama		TV	-	10	90	91	92	93	95
18. Radioisotope	*	I-19	-	3	75	90	95	99	99+
19. Composition, O_3		I-46	1/4	1/4	90	92	97	99	99+
20. Composition, A		I-47	1/4	1/4	90	92	97	99	99+
21. Precipitation		I-36	-	1/2	25	50	60	80	90
22. Electron Density (Langmuir Probe)		I-39	1/2	-	-	-	-	-	-
23. Surface Gravity		I-72	-	1/2	99	99+	99+	99+	99+
24. Surface Roughness Altimeter (Pulse Radar)		I-5	2	-	75	85	90	95	99
25. Microscope, including TV Camera, Drill, Handling Pulverizer, Sample	*	I-71	-	9-1/2	80	90	95	99	99+
26. Seismic Activity		I-21	-	1/2	50	70	80	95	99+
Lander Subtotals			10	60	45.32	50.82	56.3		

NOTE: The Rover is a mobile mount for the asterisked instruments which have an available value of 31-1/2% and a weight of 108-1/2 pounds. It is wire controlled and rechargeable battery powered (or equivalent). Structural weight including power supply, control wires and reel and mechanisms is estimated at under 125 pounds. Since the instruments are not duplicated, no additional weight need be allowed for them. Thus, a considerable weight margin remains available in the single Titan IIIC Lander.

if mission success is to be obtained. Since this area was excluded from the study, an estimated complexity and configuration of one of the more complex instruments was prepared and evaluated during the earlier Voyager studies. This value of Instrument Reliability was used during that study as being directly applicable to each instrument individually. Its effect is superposed upon the reliability of the other spacecraft subsystems and functions to provide a best estimate of the overall probability of success for a system in which this instrument was considered as the complete payload. This product of overall probability of success and available mission value represents the most likely value of the attainable mission value of that experiment.

5.3.3 MISSION EFFECTIVENESS

A weighted priority for mission-system tradeoff purposes is thus made directly available by dividing the attainable mission value by the costs uniquely related to that instrument (for cost effectiveness) or by the weight of that instrument (for payload weight effectiveness). Since the instrument cost information (including development, investment and other costs which would properly be included) was not available and estimated instrument weights were available and also since the booster, launch complex, mission support and spacecraft costs when prorated to the net scientific payload weight are expected to far outweigh the effect of actual instrument costs in establishing the overall cost effectiveness for a mission, the Attainable Mission Value per pound of net scientific payload weight was considered to be the best criteria available to this study as a measure of mission effectiveness.

The scientific instruments were, therefore, ranked in accordance with their attainable mission value per pound of instrument weight as shown in Table 5.3-2. For comparison purposes, this table is the same as that listed as 4.5.1-4a, b, c in the Saturn IB-Voyager Study, (Document 63SD801).

A few additional instruments have been identified for use on a Titan IIIC-Voyager Lander and/or Orbiter to take advantage of its increased payload capability when compared with the dual lander and orbiter system of the Saturn IB-Voyager systems. Further instrument additions or redundancies (for increased scientific instrument reliability) have also been considered. As can quickly be seen from Table 5.3-2 the individual instruments reliability is a minor part of the overall system reliability associated with the successful return to earth of the data obtained. Also as shown in Table 5.3-1, (23.1 percent Orbiter + 10 percent Entry + 56.3 percent Surface) approximately 90 percent of an orbiter + lander mission value is available during the first 100 hours after arrival. During this period, each scientific instrument is considered to have a reliability equal or greater than 96.5 percent.

5.3.4 SUBSYSTEM RELIABILITIES

Since each instrument contributes its mission value in parallel with and essentially independent of that of the others, they are essentially redundant to each other and thus the probability of obtaining successful operation of a large majority of these instruments is very high. As indicated in Table 5.3-3 (same as 4.2.1-2 of 63SD801 Vol. II) if each instrument were only 90 percent reliable and there were only 7, rather than over 96.5 percent the 38 or 40 actually redundantly involved, the probability of at least 6 of the 7 performing is 99.9 percent. For a reliability of 94 percent (each) and 30 instruments or more it becomes of value to provide individual instrument redundancy only where the Attainable Mission Value per pound of instrument weight will exceed that provided by an additional instrument of another type or by a comparable use of "weight" applied to redundancy within other subsystems. A considerable effort has already been expended in bringing the reliabilities of the Titan IIIC - Voyager System into an optimal reliability condition as represented by the data for the final system selection in Table 5.3-4. This includes redundancies of components and of subsystems elements in all those instances in which the reliability analysis, weight and cost data were sufficiently known to permit their rational consideration. As further improvement is made, the increase in reliability at system level per pound (or cost unit) by means of redundancy involves necessarily the simultaneous evaluation of an ever increasing number of alternatives as each of the subsystems and sub-subsystems are considered. Such refinement is felt to be justified only when more complete engineering definition can be given to the feasibility and practicability of these alternatives during preliminary design and design periods.

5.3.5 INCREASED MISSION VALUES VIA ROVER AND MAPPING

A very significant increase in Mission Value occurs as soon as the restriction of the landers remaining at its first landing location is removed. Gathering data with respect to an additional site location sufficiently well removed from the original site to provide

TABLE 5.3-2a. MISSION VALUE ANALYSIS SHEET

Priority Per Column 9	MISSION VALUE ANALYSIS SHEET FOR SINGLE LANDER AT 24 HOURS AFTER ARRIVAL WITH: TERRAIN, T, @ 90% LANDER, R, @ 84.5% *** INSTRUMENTS, R, @ 99.5%	1 Instrument Identifying Number	2 Individual Instrument Weight	3 ENTRY: Mission Value - % Available	4 ENTRY: R. R. V. Value - % Attainable	5 SURFACE: Mission Value - % Available @ 24 Hours	6 R. R. V. T. in % Mission Value Attainable in 24 Hours	7 Cumulative Mission Value Attainable in 24 Hours	8 Cumulative Instrument Weight	9 % Attainable Value Per Pound at 24 Hours
	Name of Instrument		lbs	%	%	%	%	%	lbs	%
1	Temperature	I-24	.3	1.00	.86	1.80	1.34	2.20	.3	7.10
2	Sounds	I-34	.5	-	-	2.55	1.90	4.10	.8	3.80
3	Pressure	I-17	.3	.50	.43	.46	.34	4.87	1.1	2.50
4	Density	I-20	1.5	2.00	1.72	2.40	1.80	8.39	2.6	2.35
5	Multiple Chamber	I-54	4.0	-	-	9.00	6.70	15.09	6.6	1.67
6	Surface Penetration Hardness	I-25	4.5	-	-	1.90	1.42	16.51	11.1	.95**
7	Photoautotroph	I-62	3.0	-	-	2.70	2.01	18.52	14.1	.67
8	Light Intensity (Sun Sensor)	I-84	.5	-	-	.38	.28	18.80	14.6	.58
9	Composition, H_2O	I-44	1.5	.50	.43	.46	.34	19.69	16.1	.51
10	Composition, O_2	I-45	1.5	.50	.43	.46	.34	20.58	17.6	.51
11	Turpidity PH	I-53	4.0	-	-	2.70	2.00	22.58	21.6	.50
12	Wind Speed & Direction	I-67	2.0	-	-	1.20	.90	23.48	23.6	.45
13	Gas Chromatograph	I-8	7.0	2.00	1.72	1.84	1.37	26.57	30.6	.44
14	Composition, N_2	I-48	1.0	.25	.22	.23	.17	26.96	31.6	.39
15	Composition, CO_2	I-49	1.0	.25	.22	.23	.17	27.35	32.6	.39
16	Soil Moisture	I-70	2.0	-	-	.96	.72	28.07	34.6	.36
17	TV Camera, Panorama	TV	20.0*	-	-	9.10	6.80	34.87	54.6	.34
18	Radioisotope	I-19	6.0	-	-	2.70	2.01	36.88	60.6	.33
19	Composition, O_3	I-46	1.5	.25	.22	.23	.17	37.27	62.1	.26
20	Composition, A	I-47	1.5	.25	.22	.23	.17	37.66	63.6	.26
21	Precipitation	I-36	1.0	-	-	.25	.18	37.84	64.6	.18
22	Electron Density (Langmuir Probe)	I-39	3.0	.50	.43	-	-	38.27	67.6	.14
23	Surface Gravity	I-72	3.0	-	-	.50	.37	38.64	70.6	.12
24	Surface Roughness & Altimeter (Pulse Radar)	I-5	15.0	2.00	1.72	-	-	40.36	85.6	.11
25	Microscope, Including TV Camera, Drill, Handling Pulverizer, Sample	I-71	75.0	-	-	8.55	6.37	46.73	160.6	.08
26	Seismic Activity	I-21	8.0	-	-	.35	.26	46.99	168.6	.03
	Lander Subtotals			10.00	8.62	51.18		46.99	168.6*	.15
	Orbiter: 10 Instruments In Order: I-23, 2, 79, TV, 12, 96, 55, 85, 95, 1 Subtotals					15.74		11.44	204.0	.40 to .03
	SYSTEM TOTALS					76.82		58.43	372.6	

* Incl. 10 lbs T V Deployment
** Less 3 lbs deployment
*** Not yet revised to include latest analysis per 4.5.3 A (2)

TABLE 5.3-2b. MISSION VALUE ANALYSIS SHEET

Priority Per Column 9 Table 4.5.1-4a	MISSION VALUE ANALYSIS SHEET FOR SINGLE LANDER AT 96 HOURS AFTER ARRIVAL WITH: TERRAIN, T, @90% LANDER, R, @84% INSTRUMENTS, R, @96.5%	1 Instrument Identifying Number	2 Individual Instrument Weights	3 ENTRY: Mission Value -% Available	4 ENTRY: R. R. V. Value -% Attainable	10 SURFACE: Mission Value -% Available @ 24 Hours to 96 Hours	12 R. R. V. T. in % = Mission Value Attainable @ 24 Hours to 96 Hours	13 Cumulative Mission Value Attainable in 96 Hours	8 Cumulative Instrument Weight	14 % Attainable Value Per Pound at 96 Hours
	Name of Instrument		lbs	%	%	%	%	%	lbs	%
1	Temperature	I-24	.3	1.00	.86	.90	.67	2.87	.3	9.60
2	Sounds	I-34	.5	-	-	.15	.11	4.88	.8	4.02
3	Pressure	I-17	.3	.50	.43	.02	.02	5.67	1.1	2.64
4	Density	I-20	1.5	2.00	1.72	.60	.88	10.07	2.6	2.93
5	Multiple Chamber	I-54	4.0	-	-	.50	.37	17.14	6.6	1.76
6	Surface Penetration Hardness	I-25	4.5	-	-	-	-	18.56	11.1	.95
7	Photoautotroph	I-62	3.0	-	-	.15	.11	20.68	14.1	.71
8	Light Intensity (Sun Sensor)	I-84	.5	-	-	.07	.06	21.02	14.6	.68
9	Composition, H_20	I-44	1.5	.50	.43	.02	.02	22.93	16.1	.53
10	Composition, 0_2	I-45	1.5	.50	.43	.02	.02	23.84	17.6	.53
11	Turpidity & PH	I-53	4.0	-	-	.15	.11	25.95	21.6	.53
12	Wind Speed & Direction	I-67	2.0	-	-	.40	.29	27.14	23.6	.56
13	Gas Chromatograph	I-8	7.0	2.00	1.72	.10	.07	30.30	30.6	.45
14	Composition, N_2	I-48	1.0	.25	.22	.01	.01	30.70	31.6	.40
15	Composition, $C0_2$	I-49	1.0	.25	.22	.01	.01	31.10	32.6	.40
16	Soil Moisture	I-70	2.0	-	-	.01	.01	31.83	34.6	.37
17	TV Camera, Panorama	TV	20.0	-	-	.10	.06	38.69	54.6	.34
18	Radioisotope	I-19	6.0	-	-	.15	.11	40.81	60.6	.35
19	Composition, 0_3	I-46	1.5	.25	.22	.01	.01	41.21	62.1	.27
20	Composition, A	I-47	1.5	.25	.22	.01	.01	41.61	63.6	.27
21	Precipitation	I-36	1.0	-	-	.05	.04	41.83	64.6	.22
22	Electron Density (Langmuir Probe)	I-39	3.0	.50	.43	-	-	42.26	67.6	.14
23	Surface Gravity	I-72	3.0	-	-	-	-	42.63	70.6	.12
24	Surface Roughness & Altimeter (Pulse Radar)	I-5	15.0	2.00	1.72	-	-	44.35	85.6	.11
25	Microscope, Including TV Camera, Drill, Handling Pulverizer, Sample	I-71	75.0	-	-	.47	.35	51.07	160.6	.09
26	Seismic Activity	I-21	8.0	-	-	.05	.04	51.37	168.6	.04
	Lander Subtotals			10.00	8.62	4.40		51.37	168.6	
	Orbiter: 10 Instruments In Order: I-23, 2, 79, TV, 12, 96, 55, 85, 95, 1									
	Subtotals					5.20		15.20	204.	
	SYSTEM TOTALS					86.42		66.57	372.6	

TABLE 5.3-2c. MISSION VALUE ANALYSIS SHEET

Priority Per Column 9 Table 4.5.1-4a	MISSION VALUE ANALYSIS SHEET FOR SINGLE LANDER SYSTEM AT 1 MONTH AFTER ARRIVAL WITH: TERRAIN, T 90%; LANDERS, R 76%; INSTRUMENTS, R 87.5%	1 Instrument Identifying Number	2 Individual Instrument Weights	3 ENTRY: Mission Value - % Available	4 ENTRY: R.R.V. Value - % Attainable	14 SURFACE: Mission Value - % Available ≥ 96 Hours to 1 Month	15 R.R.V.T. in % Mission Value Attainable in ≥ 96 Hours to 1 Month	16 Cumulative Mission Value Attainable in 1 Month	8 Cumulative Instrument Weight	17 % Attainable Value Per Pound at 1 Month	18 Cumulative Mission Value Attainable in 3 Months
	Name of Instrument		lbs	%	%	%	%	%	lbs	%	%
1	Temperature	I-24	.3	1.00	.86	.15	.09	2.96	.3	9.87	
2	Sounds	I-34	.5	-	-	.15	.12	5.09	.8	4.26	
3	Pressure	I-17	.3	.50	.43	.01	.01	5.89	1.1	2.67	
4	Density	I-20	1.5	2.00	1.72	..20	.12	10.41	2.6	3.02	
5	Multiple Chamber	I-54	4.0	-	-	.40	.24	17.72	6.6	1.82	
6	Surface Penetration Hardness	I-25	4.5	-	-	.06	.04	19.18	11.1	.97	
7	Photoautotroph	I-62	3.0	-	-	.12	.07	21.37	14.1	.73	
8	Light Intensity (Sun Sensor)	I-84	.5	-	-	.04	.03	21.74	14.6	.74	
9	Composition, H_2O	I-44	1.5	.50	.43	.01	.01	22.66	16.1	.53	
10	Composition, O_2	I-45	1.5	.50	.43	.01	.01	23.58	17.6	.53	
11	Turpidity & PH	I-53	4.0	-	-	.12	.07	25.76	21.6	.54	
12	Wind Speed & Direction	I-67	2.0	-	-	.20	.12	27.07	23.6	.66	
13	Gas Chromatograph	I-8	7.0	2.00	1.72	.04	.02	30.25	30.6	.46	
14	Composition, N_2	I-48	1.0	.25	.22	.01	.01	30.66	31.6	.41	
15	Composition, CO_2	I-49	1.0	.25	.22	-	-	31.06	32.6	.40	
16	Soil Moisture	I-70	2.0	-	-	.01	.01	31.80	34.6	.37	
17	TV Camera, Panorama	TV	20.0	-	-	.10	.06	38.72	54.6	.35	
18	Radioisotope	I-19	6.0	-	-	.12	.07	40.91	60.6	.36	
19	Composition, O_3	I-46	1.5	.25	.22	.01	.01	41.32	62.1	.27	
20	Composition, A	I-47	1.5	.25	.22	-	-	41.72	63.6	.27	
21	Precipitation	I-36	1.0	-	-	.10	.06	42.00	64.6	.28	
22	Electron Density (Langmuir Probe)	I-39	3.0	.50	.43	-	-	42.43	67.6	.14	
23	Surface Gravity	I-72	3.0	-	-	-	-	42.80	70.6	.12	
24	Surface Roughness & Altimeter (Pulse Radar)	I-5	15.0	2.00	1.72	-	-	44.52	85.6	.11	
25	Microscope, Including TV Camera, Drill, Handling Pulverizer, Sample	I-71	75.0	-	-	.38	.23	51.47	160.6	.09	
26	Seismic Activity	I-21	8.0	-	-	.08	.05	51.82	168.6	.04	
	Lander Subtotals			10.00	8.62	2.32	1.42	51.82	168.6		51.69
	Orbiter: 10 Instruments In Order: I-23, 2, 79, TV, 12, 96, 55, 85, 95, 1										
	Subtotals					4.57		18.04	204.		19.80
	SYSTEM TOTALS					93.31		69.86	372.6		71.49

2-1

TABLE 5.3-3. PROBABILITY OF SUCCESS OF AT LEAST "S" VOYAGER SYSTEMS FROM A NUMBER OF "n" LAUNCHINGS OF BOOSTER + VEHICLE RELIABILITY "R"

n	R = / s	90%	80%	70%	60%	50%
2	1	.990	.960	.910	.840	.750
	2	.810	.640	.490	.360	.250
3	1	.999	.992	.973	.936	.875
	2	.972	.896	.784	.648	.500
	3	.729	.512	.343	.216	.125
4	1	.999	.998	.991	.974	.937
	2	.996	.916	.916	.820	.687
	3	.947	.651	.651	.475	.312
	4	.656	.240	.240	.129	.062
5	1	.999	.996	.997	.989	.968
	2	.999	.993	.969	.912	.812
	3	.991	.942	.836	.682	.500
	4	.918	.737	.528	.336	.187
	5	.590	.327	.168	.077	.031
6	1	.999	.999	.999	.995	.984
	2	.999	.998	.989	.959	.890
	3	.998	.983	.929	.820	.656
	4	.984	.901	.774	.544	.343
	5	.885	.655	.420	.233	.109
	6	.531	.262	.117	.046	.015
7	1	1.0	1.0	.999	.998	.992
	2	1.0	1.0	.996	.981	.937
	3	.999	.995	.971	.904	.773
	4	.997	.966	.874	.710	.500
	5	.974	.852	.647	.419	.226
	6	.850	.577	.329	.159	.062
	7	.478	.210	.082	.028	.008

TABLE 5.3-4. RELIABILITY SUMMARY TABLE FOR THE SPACECRAFT SUBSYSTEMS OF THE TITAN IIIC VOYAGER SUBSYSTEM

	Systems Summary	I-B = 6960 Hrs III-C = 5400 Hrs Launch + Transit				From Separation 100-Hrs Mission				(2200 Hrs) 3 Mos Mission			
		I-B	III-C			I-B	III-C			I-B	III-C		
		Orbiter Lander	Orbiter Lander	Bus Lander	All Orbiter	Orbiter Lander	Orbiter Lander	Bus Lander	All Orbiter	Orbiter Lander	Orbiter Lander	Bus Lander	All Orbiter
ORBITER	Comm.	0.851	0.861	-	0.869	0.994	0.993	-	0.996	0.895	0.854	-	0.913
	G & C	0.909	0.918	-	0.918	0.995	0.993	-	0.993	0.917	0.905	-	0.905
	Power Supply	0.985	0.981	-	0.981	0.999	0.999	-	0.999	0.996	0.992	-	0.992
	Hot Gas	0.999	0.999	-	0.999	0.999	0.999	-	0.999	0.999	0.999	-	0.999
	Cold Gas	0.997	0.997	-	0.997	0.999	0.999	-	0.999	0.993	0.993	-	0.993
	Orbiter	0.768	0.771	-	0.779	0.986	0.983	-	0.986	0.810	0.761	-	0.812
	(L + T + M)					0.757	0.758		0.768	0.622	0.587		0.633
LANDER	Comm.	0.990	0.992	0.866	-	0.998	0.997	0.997	-	0.962	0.960	0.943	-
	EP & D	0.963	0.971	0.971	-	0.999	0.999	0.999	-	0.988	0.988	0.988	-
	Prop. & Sep.	0.982	0.986	0.986	-	0.986	0.986	0.986	-	0.986	0.986	0.986	-
	Thermal Control	0.946	0.958	0.958	-	0.999	0.999	0.999	-	0.988	0.988	0.988	-
	Retardation	0.999	0.999	0.999	-	0.987	0.985	0.985	-	0.987	0.985	0.985	-
	Orientation	0.999	0.999	0.999	-	0.990	0.994	0.994	-	0.990	0.994	0.994	-
	G & C	-	-	-	-	-	-	-	-	-	-	-	-
	Lander	0.883	0.908	0.792	-	0.959	0.960	0.960	-	0.905	0.905	0.889	-
	(L + T + M)					0.847	0.872	0.760		0.799	0.822	0.704	
BUS	Comm.	-	-	0.999	-								
	G & C	-	-	0.920	-								
	Hot Gas	-	-	0.999	-								
	Cold Gas	-	-	0.997	-								
	Bus	-	-	0.915	-								
	Redundant Landers	0.986	-	-	-	0.998	-	-	-	0.991	-	-	-
	(L + T + M)					0.977				0.960			
Complete System		0.757	0.700	0.725	0.779	0.984	0.944	0.960	0.986	0.803	0.689	0.889	0.812
Launch + Transmit + Mission						0.740	0.661	0.696	0.768	0.597	0.482	0.645	0.633

unique or separately valuable information may be considered equivalent to providing a second lander at that site (if the lander as a whole is made mobile) or equivalent to the mission value of the scientific instruments carried if a smaller Rover is provided as is proposed and evaluated by this study. Notes at the bottom of Table 5.3-1 provide the "values" information considered.

Similarly a very significant increase in Mission Value occurs as soon as the restriction of the Orbiter's remaining at least 1000 miles above the surface of the planet is removed. A sufficient payload weight capability exists to permit the inclusion of solid rockets (in numerous increments individually fired upon command) in the orbiter which would permit the orbiter to be brought closer and closer in toward the planet at the apogee of an orbit with the perigee remaining at about 1000 nautical miles. This would permit the selection of certain areas of the planet for detailed, close-up, TV mapping and thus providing, in a limited sense, a roving Orbiter at command.

Any or all of these avenues of improvement could be considered, if desired, as well as to establish more accurately the weights involved in each of the various subsystem components. In this latter respect, it is essential (if prior experience is applied) that allowance be made for increased weight beyond that of any preliminary systems weight analysis. Thus, it may be that to actually produce equipment with the reliability already determined by this study will require the use of all the weight margin now indicated as available.

As covered under the Lander subsystem portion of this report, a roving capability is considered practicable to make the surface instruments available sequentially at different (e.g., 100 yards or more separated) sites with a Rover equipment reliability of at least 95 percent, a terrain suitability of approximately 75 percent per move and with each site representing unique or valuable additional knowledge valued at approximately 100 percent of that for the same instruments at the prior site. The attainable mission value of such a lander would be 250 percent of that available for these same instruments at a single site location.

5.3.6 SYSTEM RELIABILITIES FOR SINGLE LAUNCHES

In the Saturn IB system, communication was by relay link from the lander to the orbiter (with a direct link backup at reduced capability — not sufficient for TV transmission and, therefore, not applicable for higher attainable mission value modes of system operation). The greater dependence of the Saturn IB lander upon the reliability of the orbiter is clearly shown in comparing this with the Titan IIIC Bus-Lander configuration. (See Table 5.3-5.)

TABLE 5.3-5. SYSTEM RELIABILITY AT 100 HOURS FOR SINGLE LAUNCHES

	SATURN IB	TITAN IIIC
Lander Surface Data		
Martian Terrain Allowance	90 %	90 %
Lander Thru 100 Hours	84.7	76.0
Lander Instrument 100 Hrs	96.5	96.5
Orbiter Thru Transit	76.8	(Bus) 91.5
Orbiter During First 100 Hrs	98.6	Incl. in Lander
Reliability (Product)	55.7%	61.0%
Booster Reliability	90	90
System	50.1%	55 %
Lander Entry Data		
Lander Thru Transit	88.3%	79.2%
Lander Instrument	99.5	99.5
Orbiter Thru Transit	76.8	(Bus) 91.5
Booster Reliability	90	90
System	60.7%	64.9%
Orbiter		
Orbiter Thru 100 Hrs	75.7%	76.8%
Orbiter Instrument	96.5	96.5
Booster Reliability	90	90
System	65.7%	66.7%

5.3.7 ATTAINABLE MISSION EFFECTIVENESS FOR SINGLE LAUNCHES WITH IDENTICAL "INSTRUMENTATION" AND "WINDOWS"

In comparing the attainable mission value of Saturn 1B and Titan IIIC systems, it will be assumed at this point that the same net scientific payload is carried by each of the dual Landers of the Saturn 1B system and each of the single Landers of the Titan IIIC system. Also, that the Orbiters of the two systems are identical in scientific instrumentation. Thus, for the moment, no advantage is being taken of the increased payload capability of the Titan IIIC booster.

Information received to date has indicated that in multiple launch combinations for any given launch window opportunity, two Saturn 1B and/or three Titan IIIC launching pads would be available with a single launch control network controlling all launches. With such a complex, the average interval between launches has been considered to be 20 days and 4 days respectively. Also, identical launch window opportunities of 30 days are considered available. Thus, for the moment no advantage is being taken of the longer launch window which is made possible with Titan IIIC by the separation of Orbiter and Lander launches.

The dependency of Orbiters and Landers has been previously discussed. This dependency is incorporated in the reliabilities involved. Thus, the (a) Lander-surface data, (b) Lander-entry data, and (c) Orbiter are to be treated as independent systems in determining the Attainable Mission Value (AMV) for various combinational opportunities.

In multiple Lander combinations, surface data is considered unique and of full value at each site, entry data is considered to be 50 percent less valuable for each successive entry. In multiple Orbiter combinations, each Orbiter was similarly considered to be 50 percent less valuable unless the orbits had significantly different inclination to Mars thus obtaining different coverage in which case full value was to be given the second Orbiter. In the evaluation for this study report, it has been assumed that the second Orbiter would be placed in or near a polar orbit and thus would be 100 percent valuable.

Applying the overall booster plus spacecraft system reliability figures corresponding to the time and values of Figure 5.3-1 for the instruments carried on a Saturn 1B single Orbiter and single Lander and using only this same instrumentation for the Titan IIIC as a limiting case, we have Table 5.3-6.

TABLE 5.3-6. PERCENT ATTAINABLE MISSION VALUES
(FOR SATURN IB INSTRUMENTATION)

Single System Lander Surface Data	Saturn IB 12 Hrs	Saturn IB 24 Hrs	Saturn IB 100 Hrs	Saturn IB 90 Days	Titan IIIC 12 Hrs	Titan IIIC 24 Hrs	Titan IIIC 100 Hrs	Titan IIIC 90 Days
Lander System								
Reliability	53.3	52.2	50.1	34.9	59	58	55	46
Value Increment	45.3	5.5	5.5	3.7	45.3	5.5	5.5	3.7
AMV Increment	24.1	2.9	2.7	1.3	26.7	3.2	3.0	1.7
Entry Data								
Sat IB 60.7 X 10%	6.1	6.1	6.1	6.1				
Titan 64.9 X 10%					6.5	6.5	6.5	6.5
AMV Increment	30.2	9.0	8.8	7.4	33.2	9.7	9.5	8.2
AMV Total - Lander	30.2	39.2	48.0	55.4	33.2	42.9	52.4	60.6
Orbiter								
Reliability	69.3	69.1	68.1	56.0	70.3	70.1	69.1	57.0
Value Increment	10.7	5.2	5.2	8.9	10.7	5.2	5.2	8.9
	7.4	3.6	3.5	5.0	7.5	3.6	3.6	5.1
AMV Total - Orbiter	7.4	11.0	14.5	19.5	7.5	11.1	14.7	19.8
AMV Orbiter + Lander	37.8	50.2	62.5	74.9	40.7	54.0	67.1	80.4

Thus, a single Saturn 1B with a single lander is capable of an Attainable Mission Value of 74.9 percent. Correspondingly two Titan IIIC's are required to provide an Attainable Mission Value of 80.4 percent for identical (Saturn 1B type) single lander systems.

5.3.8 ATTAINABLE MISSION EFFECTIVENESS FOR MULTIPLE LAUNCH OPPORTUNITIES

The Attainable Mission Value where "n" launches are made equals the sum of the probability of at least one success times the value of the first successful mission plus ---

the probability of at least a second success times the value of a second successful mission plus ---

plus the probability of at least "n" successes times the value available from the success of an "n"th successful mission.

A. INITIAL COMPARISON - SINGLE SATURN IB LAUNCH
(Using Saturn IB type Orbiter and Lander Instrumentation and Booster Reliability = 90 percent)

A single Saturn IB launch with dual landers and a single orbiter : using reliability and value increments of Table 5.3-6 and probabilities of a least S, successes from a set of "n" launches of identical systems of reliability "R" as illustrated by Table 5.3-3 — we have effectively

2 landers for surface data,
2 landers for entry data, and
1 orbiter for orbiter data.

To compare with this, three Titan launches must be made: one orbiter and two lander launches. Such a comparison, including a consideration of the changes in value and reliability as a function of time after arrival is provided in Table 5.3-7.

If this calculation above had been simplified by using the reliabilities applicable at 100 hours times the available values for the full three-month period, we would have

	(1) Saturn IB	(3) Titan IIIC
Lander "Surface" Reliability	50%	55%
(1) of (2)	75	79.8
(2) of (2)	25	30.2
Probabilities Sum	100	110.0
Value Increment	x 60%	x 60%
Surface AMV Increment	60%	66%
Entry AMV (as before)	10.2	10.8
Orbiter AMV (as before)	62.5	67.1
	132.7%	143.9%

By comparison with Table 5.3-7, it will be seen that this simplification does not seriously affect the decision and does not favor the Titan IIIC. This method has been used in subsequent comparisons.

TABLE 5.3-7. ONE SATURN IB VS THREE TITAN IIIC
PERCENT ATTAINABLE MISSION VALUES

Surface Data	Saturn IB 12 Hrs	Saturn IB 24 Hrs	Saturn IB 100 Hrs	Saturn IB 90 Days	Titan IIIC 12 Hrs	Titan IIIC 24 Hrs	Titan IIIC 100 Hrs	Titan IIIC 90 Days
Lander								
"Surface" Reliability	53.3	52.2	50.	34.9	59.	58.	55.	46.
Probability of at least (1) of (2)	78.1	77.1	75.	27.0	83.1	82.3	79.8	70.7
Probability of at least (2) of (2)	28.5	27.7	25.	12.0	35.	34.5	30.2	21.0
Probability Sum	106.6	104.8	100	37.0	118.1	116.8	110.0	91.7
Value Increment	45.3	5.5	5.5	3.7	45.3	5.5	5.5	3.7
AMV Increment	48.2	5.8	5.5	1.4	53.5	6.4	6.0	3.4
Surface AMV	48.2	54.0	59.5	60.9	53.5	59.9	65.9	69.3
"Entry" Reliability	60.7				64.9			
Probability of at least (1) of (2)	84.5				87.7			
Probability of at least (2) of (2)		36.9				42.2		
Probability Sum	NA				NA			
Value Increment	10.0	5.0			10.0	5.0		
AMV Increment	8.4	1.8			8.7	2.1		
Entry AMV			10.2				10.8	
Orbiter								
As in Table 5.3-6			62.5				67.1	

	Saturn IB	Titan IIIC
Total ATTAINABLE MISSION VALUE		
at 100 Hours	132.2%	143.8%
at 3 Months	133.6%	147.2%
Required	20 days 1 Launch	12 days 3 Launches

B. FINAL COMPARISONS — SATURN IB VERSUS TITAN IIIC

In making final comparisons of the Titan III system with the Saturn IB System, the following considerations have been included:

1. Booster reliability has been more conservatively taken as = 80 percent. This value has been used for both Titan and Saturn boosters in preparing the Systems Reliabilities Tables 5.3-8 and 5.3-9.

2. The increased payload capability of the Titan IIIC, both in the orbiter and in the single larger lander, has been applied using the data shown by Table 5.3-1.

3. The same number of orbiters have been launched for directly comparable systems and the number of landers has been varied.

4. The dependence of the Saturn IB landers upon communication via the orbiter has been included in its overall reliability calculations since TV transmission requires the higher data rates available via the orbiter.

The Reliability calculations for Saturn IB launches are shown in Table 5.3-8. The corresponding reliability figures for Titan IIIC are shown in Table 5.3-9. In Table 5.3-10, the data from the above tables is combined to provide Attainable Mission Values for comparing the Saturn IB and Titan IIIC systems. A sample calculation is immediately below each summary figure in the table. This detail is included to provide the "step-by-step" information as to the Attainable Mission Value method requested since the Saturn IB report was issued and also to facilitate the preparation of comparisons where different Mission Value estimates or other variations need to be considered which are not covered in this report.

Figures 5.3-1 through 5.3-4 have been prepared to show the results of these comparisons. They clearly demonstrate the relative values involved.

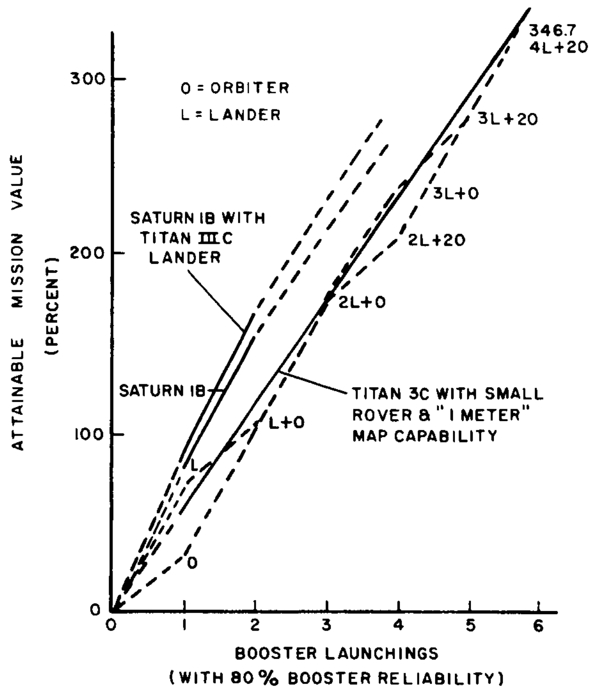


Figure 5.3-1. Attainable Mission Value

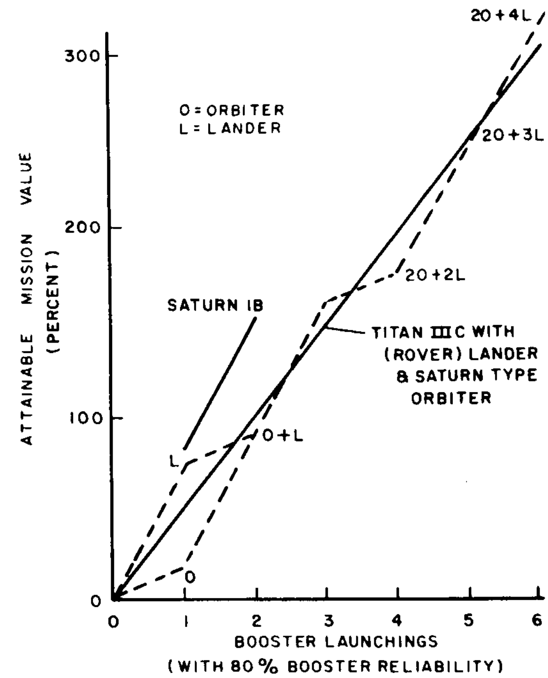


Figure 5.3-2. Attainable Mission Value

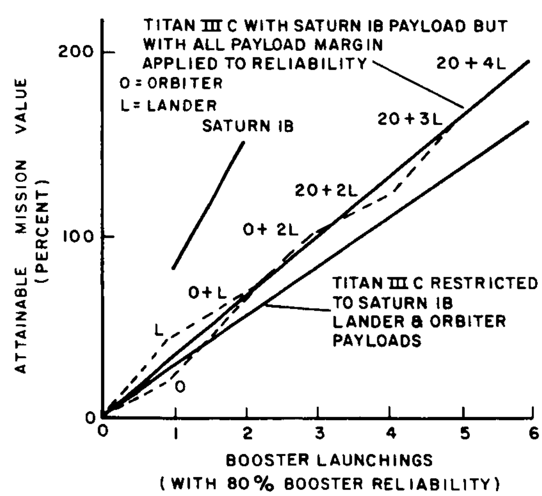


Figure 5.3-3. Attainable Mission Value

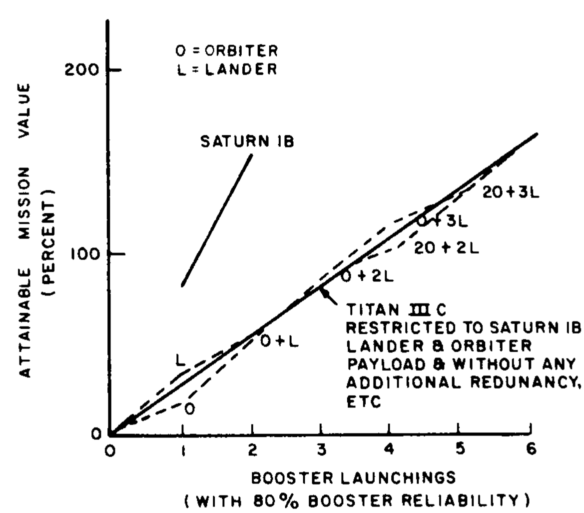


Figure 5.3-4. Attainable Mission Value

TABLE 5.3-8. SYSTEM RELIABILITY FOR TWO SATURN 1B LAUNCHES

LAUNCHES	SATURN IB (A) (Surface) %	(B) (Lander) %	(C) (Orbiter) %
Orbiter Reliability:			
Instrument Reliability			96.5
Orbiter thru 100 hours	75.7	76.8	75.7
X	80.0	80.0	80.0
Subtotal	60.5	61.5	58.4
Probability of Success for			
at least (1) of (2)	84.3	85.2	82.7
at least (2) of (2)	36.2	37.6	33.8
Probability Sum	120.5	122.8	115.5
Lander "Surface" Reliability			
Martian Terrain	90.0		
X Lander thru 100 hours	84.7		
X Lander Instrument	96.5		
Subtotal	73.6		
Probability of Success (per Launch) for			
at least (1) of (2)	93.1		
at least (2) of (2)	54.0		
Probability Sum	147.1		
Lander "Entry Reliability			
Lander Instrument		99.5	
X Lander thru Transit		88.3	
Subtotal		87.8	
Probability of Success (per launch) for			
at least (1) of (2)		98.5	
at least (2) of (2)		77.4	
Probability Sum		175.9	

Prob. of Launch + Orbiter + Landers success for at least

(1) of (4) --- 84.3% X 93.1% = 78.4%
(2) of (4) --- 84.3% X 54.0% = 45.6%
(3) of (4) --- 36.2% X 93.1% = 33.7%
(4) of (4) --- 36.2% X 54.0% = 19.6%

Probability Sum 177.3%

Prob. of Launch + Orbiter + Lander success for at least

(1) of (4) --- 85.2% X 98.4% = 83.8%
(2) of (4) --- 85.2% X 77.4% = 65.9%
(3) of (4) --- 37.6% X 98.4% = 37.0%
(4) of (4) --- 37.6% X 77.4% = 29.1%

NOTE:

If single Titan IIIC type landers were used in place of the dual landers on a Saturn IB launch, the lander "surface" reliability at 100 hours (including orbiter and booster) would be 60.5 X 73.6 = 44.5%. The probability of success for at least (1) of (2) = 69.0
at least (2) of (2) = 19.7
Probability Sum = 88.7

Similarly, for "entry", = 61.5 X 87.8% = 58%. The probability of success for at least (1) of (2) = 78.8
at least (2) of (2) = 29.0
Probability Sum = 107.8

TABLE 5.3-9. SYSTEMS RELIABILITY FOR MULTIPLE TITAN IIIC LAUNCHES

Orbiter Reliability:			
Instrument thru 100 hours	96.5%	Probability of success for	
X Booster	80	at least (1) of (2) ---	83.5
X Orbiter thru 100 hours	76.8	(2) of (2) ---	35.
System	59.3	Probability Sum -----	118.5

Lander "Surface" Reliability		Probability of success for	
Instrument thru 100 hours	96.5	at least (1) of (2) ---	73.3
X Martian terrain	90.0	(2) of (2) ---	23.2
X Lander thru 100 hours	76.0	Probability Sum -----	96.5
X Bus thru transit	91.5	or of at least (1) of (3) ---	86.2
Lander	60.5	(2) of (3) ---	47.5
X Booster	80.0	(3) of (3) ---	11.3
System	48.3	Probability Sum -----	145.0
		or of at least (1) of (4) ---	92.9
		(2) of (4) ---	76.2
		(3) of (4) ---	28.5
		(4) of (4) ---	5.5
		Probability Sum -----	203.1

Lander "Entry" Reliability		Probability of success for	
Lander Instrument	99.5	at least (1) of (2) ---	82.2
X Lander thru entry	79.2	(2) of (2) ---	33.1
X Bus thru transit	91.5	Probability Sum -----	115.3
Lander	72.1	or of at least (1) of (3) ---	92.4
X Booster	80	(2) of (3) ---	61.6
System	57.7	(3) of (3) ---	19.2
		Probability Sum -----	173.2
		or of at least (1) of (4) ---	96.7
		(2) of (4) ---	79.2
		(3) of (4) ---	43.5
		(4) of (4) ---	11.0
		Probability Sum -----	230.4

NOTE:

Rover "equipment" reliability is estimated at greater than 95%. The probability that the terrain will be such as to permit each successive relocation movement to be accomplished by the rover has been estimated at 75%. Thus, the relocation potential is estimated as $(.95 \times .75) + (.95 \times .75)^2 + (\ )^3 + (\ )^4$ -- $(\ )^{25} = 249\%$

TABLE 5.3-10. ATTAINABLE MISSION VALUES (With Titan IIIC "1 Meter" Orbiter and Lander-Rover)

SATURN IB

1 Launch ---- 80.8% AMV

Orbiter ------------------	17.5%
58.4% R X 30% V	
(2) Lander(s) "Entry" ---------	8.9%
(1) 61.5 R X 98.5 R X 10% V	
(2) 61.5 R X 77.4 R X 5% V	
(2) Lander(s) "Surface" -------	54.4%
60.5 R X 147.1 X 60% V	

NOTE: Were the large lander and rover used on the 1969 Saturn IB ---- AMV + 84.6%.

80.8 - 54.4 + (60.5% R X 73.6% R X 138.5% V. "surface")

80.8 - 8.9 + (61.5% R X 87.8% R X 10.0% V. "entry")

If the "1 Meter" orbiter were not possible (i.e. sterilization) the (2) launch Titan IIIC would have ---- AMV = 98.3%

104.8% - (59.3% R X 24% V)
+ (59.3% R X 13% V)

2 Launches ---- 154.1% AMV

(2) Orbiters -----------------	34.7
115.5% R X 30% V	
(4) Landers "Entry" ----------	13.0
(1) 83.8% R X 10% V	
(2) 65.9% R X 5% V	
(3) 37.0% R X 2.5% V	
(4) 29.1% R X 1.3% V	
(4) Landers "Surface" ---------	106.4%
(4) 177.3% R X 60% V	

NOTE: With the Titan IIIC large lander and rover applied on two Saturn IB Launches --- the resultant AMV = 166.4%.

154.1 - 106.4 + (44.6% R
for (1) of (2) = 68.9
+ (2) of (2) = 19.5
88.4 X 138.5%)
Surface

154.1 - 13.0 + (54.0% R
for (1) of (2) = 78.8 X 10% V
+ (2) of (2) = 29.0 X 5% V)
Entry

Similarly, if additional instruments are used on the orbiter rather than "1 Meter" mapping, the 3 Launch Titan IIIC would have AMV = 169.1%

175.6 = 59.3% R (24 - 13% V)

TITAN IIIC

2 Launches ---- 104.8% AMV

Orbiter ----------------	32.1
59.3% R X 54% V	
Lander "Entry" --------	5.8
57.7% R X 10% V	
Lander "Surface" -------	66.9
48.3% R X 60% V	
+ 48.3% R (249% X 31.5%V)	

3 Launches ---- 175.6% AMV

Orbiter ----------------	32.1
(2) Landers "Entry" --------	9.9
(1) 82.2 R X 10% V	
+(2) 33.1 R X 5% V	
(2) Landers "Surface" ------	133.6
(1) 73.3 R (60% V + 249% X 31.5% V)	
(2) 23.2 R (138.5% V)	

4 Launches ---- 207.5% AMV

(2) Orbiters ---------------	64.0
118.5 R X 54% V	
(2) Landers "Entry" -------	9.9
(2) Landers "Surface" -----	133.6

5 Launches ---- 277.8% AMV

(2) Orbiters --------------	64.0
(3) Landers "Entry" -------	12.8
92.4% R X 10% V	
61.6% R X 5% V	
19.2% R X 2.5% V	
(3) Landers "Surface" -----	201.0
145% R X 138.5% V	

6 Launches ---- 360.1% AMV

(2) Orbiters --------------	64.0
(4) Landers "Entry" -------	14.6
96.7% R X 10% V	
79.2% R X 5% V	
43.5% R X 2.5% V	
11.0% R X 1.25% V	
(4) Landers "Surface"------	281.5
203.1% R X 138.5% V	

C. RELIABILITY IMPROVEMENT ONLY — ATTAINABLE MISSION VALUES

The following is considered to be a "most optimistic" Titan IIIC Bus/Lander "Surface" calculation since it affects the least reliable subsystems and requires more than the payload margin in applying all available payload weight (191 pounds) to increase the spacecraft reliability. As noted in Table 5.3-4. the most critical subsystems are:

198 lbs. Communication	86.2%	Fully Redundant		97.5%
55 lbs. Thermal Control	95.7%	Fully Redundant		99.8%
146 lbs. Guidance and Control	92.0%	Fully Redundant		99.4%
All Other Subsystems	63.6%			63.6%
Bus Lander Titan IIIC ------	48.3%	Max. Improved Rel.	=	61.6%

With a basic reliability for "surface" data per complete lander system of 61.6%, the probabilities of success for multiple launches becomes for at least:

1 of 2	85.2%	1 of 3	94.4%	1 of 4	97.8%
2 of 2	37.7%	2 of 3	67.2%	2 of 4	83.9%
Sum	122.9%	3 of 3	23.6%	3 of 3	50.4%
		Sum	185.2%	4 of 4	14.6%
				Sum	246.7%

Correspondingly, the reliabilities for the Titan IIIC Bus/Lander "Entry" would be:

Communication	86.5%	Fully Redundant		97.6%
Thermal Control	95.8%	Fully Redundant		99.8%
Guidance and Control	92.0%	Fully Redundant		99.4%
All Other Subsystems	74.4%			74.4%
	57.7%	Max. Improved Rel.	=	72.0%

The probability of success for multiple launches using 72%, would then become:

1 of 2	92.2%	1 of 3	97.7%	1 of 4	99.4%
2 of 2	51.7%	2 of 3	80.8%	2 of 4	93.0%
Sum	143.9%	3 of 3	37.2%	3 of 4	68.8%
		Sum	215.7%	4 of 4	26.8%
				Sum	288.0%

And, for the Titan IIIC Orbiter, the affected reliabilities are:

Communication	86.5%	Fully Redundant	97.6%
Guidance and Control	91.5%	Fully Redundant	99.3%
All Other	74.9%	All Other	74.9%
System	59.3%		72.9%

The corresponding probabilities of success for at least:

1 of 2	92.7%
2 of 2	53.0%
Sum	145.7%

These improved reliabilities (if all these could be made redundant) when applied to a Titan IIIC which is restricted to the Saturn IB instrumentation payload result in Attainable Mission Values of ---- AMV for

1 Orbiter	72.9% x 30% V	= 21.9%
2 Orbiters	145.7% x 30% V	= 43.7%
1 Lander (Surface)	61.6% x 60% V	= 37.0%
2 Landers (Surface)	122.9% x 60% V	= 73.7%
3 Landers (Surface)	185.2% x 60% V	=111.1%
4 Landers (Surface)	246.7% x 60% V	=148.0%

1 Lander (Entry)	72% x 10% V	= 7.2%
2 Landers (Entry)	92% x 10% V	= 11.8%
	+51.7% x 5% V	
3 Landers (Entry)	97.7% x 10% V	= 14.7%
	+80.8% x 5% V	
	+37.2% x 2.5% V	
4 Landers (Entry	99.4% x 10% V	= 16.6%
	+93.0% x 5% V	
	+68.8% x 2.5% V	
	+26.8% x 1.25% V	

These values are shown in Figure 5.3-3 for comparison with the Attainable Mission Values through other combinations. The corresponding data is:

1 Orbiter				21.9% AMV
2 Orbiter				43.7%
1 Lander	37.0%	+	7.2%	44.2%
2 Landers	73.7%	+	11.8%	85.5%
3 Landers	111.1%	+	14.7%	125.8%
4 Landers	148.0%	+	16.6%	164.6%

As indicated by Figure 5.3-3, the improvement in Attainable Mission Value obtainable by a most optimistic estimate of improved spacecraft reliabilith through reduncancy (i.e., without any new component technology since this would be applicable to all system launch combinations) is approximately 20 percent.

The payload margin of the Titan IIIC Lander over that of one of the dual Landers of the Saturn IB is approximately 260 pounds. If smaller improvements in reliability and AMV are assumed to be feasible at at least this ratio, the unused Lander payload margin of approximately 130 pounds remaining with the proposed Titan IIIC Lander Rover could make a corresponding improvement in AMV.

Also, the combination has been analyzed in which the proposed Titan IIIC Lander/ Rover (without additional redundancy) is used with an Orbiter which has no capability for "1 meter" resolution mapping or upper atmospheric density determinations. This would result in Figure 5.3-2 in which

1 Orbiter	59.3% R x 30% V	=	17.8% AMV
2 Orbiters	118.5% R x 30% V	=	35.5%
1 Lander	(Table 5.3-9)	=	72.7%
2 Landers		=	143.5%
3 Landers		=	213.8%
4 Landers		=	281.5%

Applying the additional payload capabilities to key sub-subsystems within the three principal spacecraft subsystems to reliability improvement in combination with the Titan IIIC Lander/Rover and "1 Meter" Orbiter, an optimum Attainable Mission Value will result. Areas for such effort would include:

	Guidance and Control		
6.5 lbs	Earth Sensor	98.0% Rel.	If Made Fully Redundant 99.7% Reliability
5.5 lbs	Canopus Tracker	98.6% Rel.	
14.2 lbs	Storage and Logic	98.6% Rel.	
11.5 lbs	Antenna Servos	98.8% Rel.	
37.7 lbs		94.1% Rel.	
	All Other G&C	97.8% Rel.	97.8% Reliability
		92.0% Rel.	97.5% Reliability
55 lbs	Thermal Control	95.7% Rel.	99.8% Reliability
	Communication:		
28 lbs	High Gain Antenna, diplexer and earth sensor	95.3	
10.8 lbs	(2) Transponders	94.2	99.4
	All other Communications	96.0	96.0
131.5 lbs	Overall effect	86.2	95.4

5.4 RELIABILITY IMPROVEMENT

5.4.1 RELIABILITY GROWTH DURING SYSTEM TEST

In analyzing system and subsystem data, it appears that the value of test time in the growth of system reliability through the detection and elimination of defects of materials, processes, parts and of design, manufacture and testing in spacecraft programs which implement a high reliability demonstration and test requirement of the type recommended in this report may be indicated as in Figure 5.4-1. On log-log paper this is shown as a line lying along a slope of approximately 0.75 and passing through the point of 100 hours "time since last failure" at 1000 hours of testing for prototype equipments. For mature system equipment (i.e., composed of components for which 1000 hours Mean Time Between Failures (MTBF) has been demonstrated by prior "like" equipments) a line of the same slope but passing through the point of 300 hours "time since last failure" at 1000 hours of testing. As an evidence of the applicability of such lines, data from a recently completed set of spacecraft system tests has been plotted on Figure 5.4-1. The hours of "Time Since Last Failure are shown for the last three

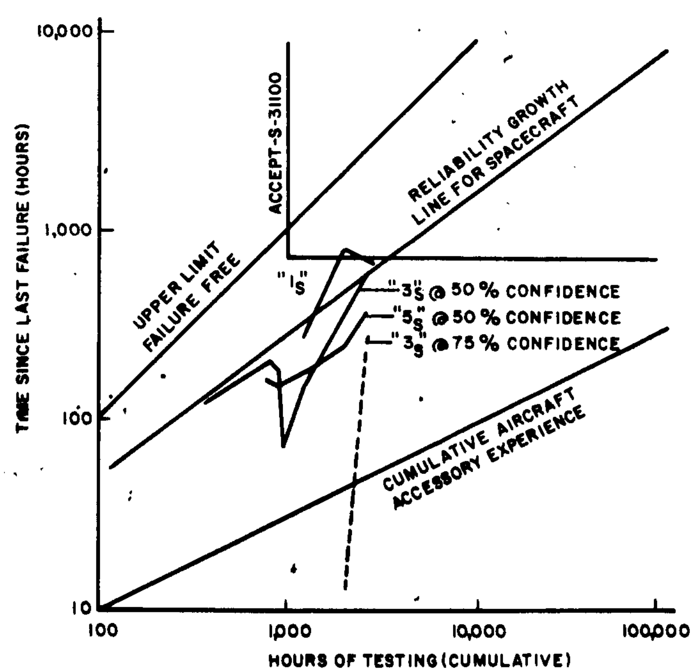


Figure 5.4-1. Reliability Growth Demonstration By Subsystem and System Testing

systems failures of record for a given flight system. These points are joined by the line identified as "1s" (i.e. single failure intervals).

These last three failure intervals were also averaged to obtain an indication of the MTBF which is representative of the design status and condition of the equipment at the end of the systems test period. By progressively eliminating the last interval and adding the next interval, successive groups of three failure intervals were averaged over the entire systems test period. These points are joined by the line identified as "3s". Similarly, successive groups of five failure intervals were averaged. These points are joined by the line identified as "5s".

Single sided confidence limits (i.e. the confidence with which it can be stated that the Time Between Failures will be greater than the value shown) were evaluated for the "5s" and for the "3s". Less variation is present, of course, in the "5s" data but it necessarily is somewhat less representative of the final condition of the system than are the "3s" points. The 75 percent confidence level established by the "3s" data is also shown in Figure 5.4-1.

The Reliability growth line shown in Figure 5.4-1 was drawn through the last point of "3s" and at a slope of 0.75 which was considered to be most representative of the data available for this study.

5.4.2 MINIMUM ACCEPTANCE REQUIREMENTS

A criteria for minimum acceptance testing was developed for the Voyager as specified by S-31100 (Draft) a copy of which is provided with this study report. This calls for all components (e.g., transmitters, amplifiers, etc. packaged as separate units) to be qualified and accepted by test plans which include a minimum of 150 hours of thermal vacuum testing and that this be extended as necessary to demonstrate a failure-free terminal period of not less than 100 hours. Also, following component qualification and acceptance, this calls for all systems to be qualified and accepted by test plans which (including the thermal vacuum testing at component levels) includes a minimum of 1000 hours of thermal vacuum testing and that this be extended as necessary to demonstrate a failure-free terminal period of not less than 700 hours. Complete fulfillment of all other environmental testing requirements is to preceed the demonstration of the component and system failure-free terminal periods noted above.

Based upon the analysis made during this study of the variations in "time since last failure" experienced during systems and subsystems testing, it is recommended that the requirement that "the average of the last three such times since last failure also be required to exceed 700 hours" be added to S-31100 as a requirement prior to the shipment of flight hardware.

5.4.3 SEPARATE CONSIDERATION OF DYNAMIC AND STATIC PORTIONS OF THE MISSION PROFILE

To be added to this specification S-31100 "Reliability Requirements for Subcontracted Components, Subsystems and Systems" is the separation of Reliability demonstration testing into two distinct categories termed (a) Equivalent Dynamic Mission Tests and (b) Equivalent Static Mission Tests.

Only the "Transients" as illustrated in Figure 5.4-2 are to be included in the "Dynamic" category of testing. All tests at component, subsystem and system levels are to be of this category.

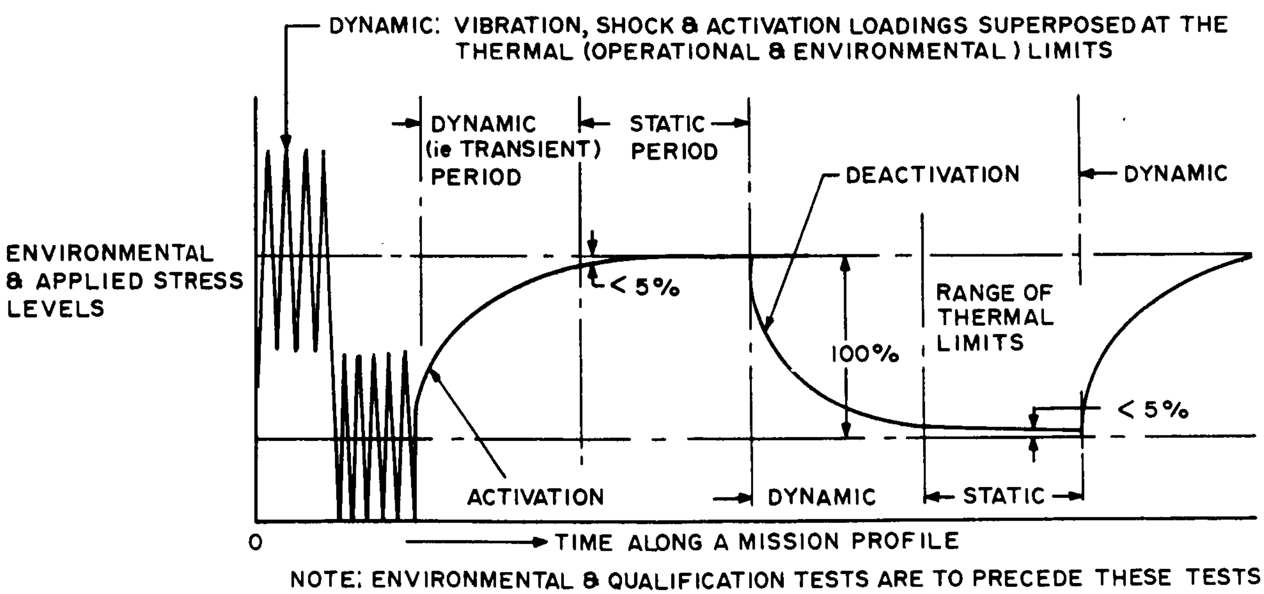


Figure 5.4-2. Dynamic and Static Mission Profile Elements

All "Static" mission capabilities are to be demonstrated at parts, materials and process units and subassemblies levels in which "long time" storage and long time active life reliability characteristics can be effectively and economically demonstrated.

The test plan for the Voyager Program should include the demonstration by actual environmental tests either at "as used" stress levels and in real time (or under "accelerated" test plans previously submitted to and approved by the customer's Project Manager) for a suitable number of "specimen mission times" at constant stress levels to statistically demonstrate that the design margins applied in the application of each part, material and process as used in the design and manufacture of each Voyager "Flight" system are adequate to assure the required Voyager System reliability during the static (e.g., passive or unactivated) periods of the Voyager mission.

Each Qualification and Acceptance test (as noted in S-31100) is to include as many "dynamic" cycles as possible within the test times called for in its applicable Test Specification. The requirements of S-31100 have been recommended as requirements with which the individual test specifications are to conform.

A preliminary analysis of the number of component and system actuations and of the number of changes in environments (i.e., transients in Figure 5.4-2) per Voyager Mission indicates that the test plan for the four developmental systems called for in the Voyager-Saturn IB Study Report of October 15, 1963 together with the acceptance tests of "flight" systems and components (including spares) will be able to provide a statistically sound demonstration of the System Reliability requirements for all actuations and environmental changes involved in the Mission.

By separating the Reliability Demonstration tests into those two separate categories and by accepting materials and parts test data as "a priori" test evidence as applicable to the fulfillment of the "Equivalent Static Mission Tests" to demonstrate the inherent reliability and performance capability of each of the basic elements of which the system is composed, a statistically sound and economical approach to the demonstration of the fulfillment of System Reliability requirements for all "steady state", non-transient portions of the Mission is possible. Trend analysis may be used together with suitable design margins to truncate these test periods to the degree required by the customer.

It is the recommendation of this study, however, that a full demonstration be required as a part of the Voyager Program for all items for which there is considered to be a greater than a 50 percent likelihood that they will be used in future spacecraft systems.

A study has been made of the time required to complete a test for which the specification contains a requirement to return to the beginning of the test whenever a failure occurs. (Ref. 64GL93). Figure 5.4-3 illustrates a simple system of the type analysed. If we consider the sum of the individual test time of the principal components or subsystems (e.g., 8 + 14 + 20 = 42 hours) as Equivalent Test Time this curve indicates that for such a system there is a 50 percent probability that the requirement will be completed in one Equivalent Test Time of 42 hours. Also, there is about a 77 percent probability that less than two Equivalent Test Times, 84 hours, would be required, and that there is a 90 percent assurance that no more than three (126 hours) would be necessary.

Upon the basis of similar experience and reasoning, it was considered in the Voyager Study, October 15, 1963, that the 100-hour failure-free requirement for the Equivalent Dynamic Mission Tests noted above would be able to be completed for components

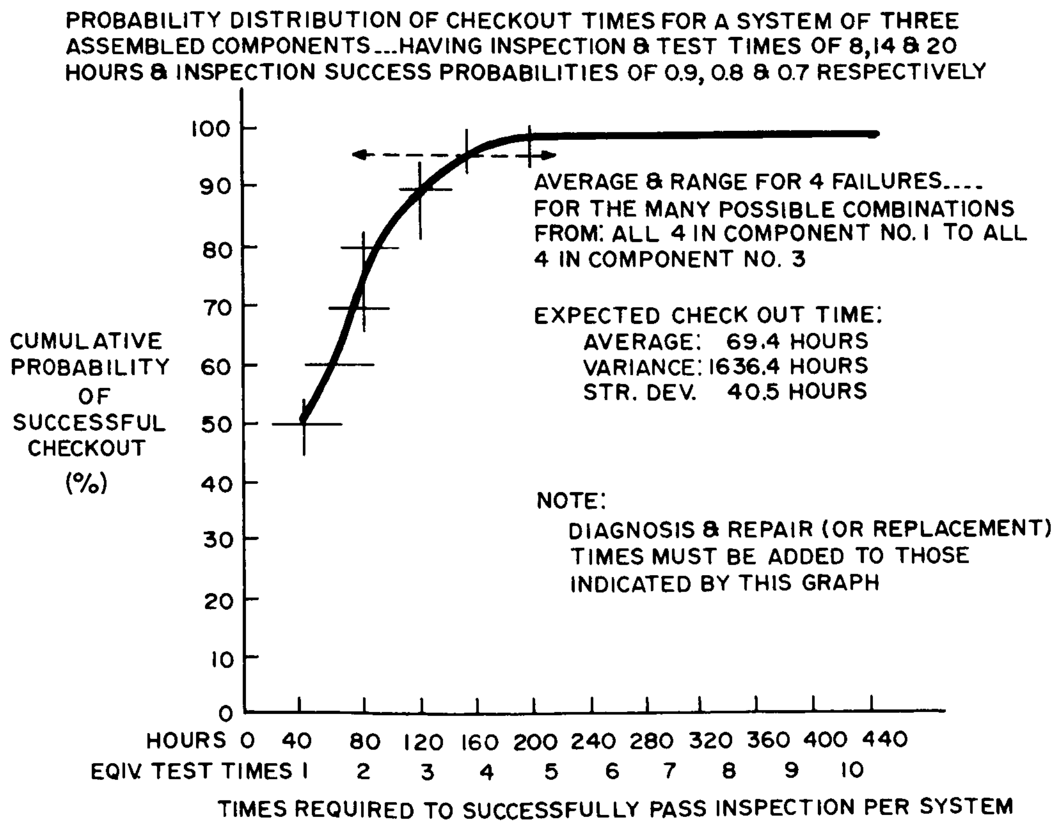


Figure 5.4-3. Checkout Time

within a programmed period of 300 hours. And, that the corresponding 700 failure-free hours requirement for systems tests would be able to be completed (including the 300 hours allowed for component tests) in an overall period of 2000 test hours. It may be of interest to note that results of the recently completed tests of the Nimbus Control System indicated that the production prototype No. 102 fulfilled a somewhat comparable 700 hour requirement in 5000 overall test hours and the first flight system No. 103 in 2,700 overall test hours.

A recommendation from the Reliability portion of this study would be that the division of reliability and performance capability testing requirements into separate Dynamic and Static requirements and the definitization of the ranges and cycles of environmental changes per System Test Equivalent Mission in each category be completed as early as possible in the Voyager program and that written test plans and documents be prepared and provided to the component, subsystem and system engineers to advance the date of design qualification and assure ample opportunity for a reasonable demonstration of design and manufacturing reliability and of the systems mission capability prior to the delivery pressures of the flight date.

Based upon preliminary analysis of the Voyager mission profile, the four development systems during their 1000 to 2000 hour test period are expected to provide 98 percent Reliability at 50 percent Confidence (e.g., 93 percent at 90 percent Confidence) for the "dynamic" portion(s) of the Mars mission.

For example, during the transit period (ref. Voyager Table 4.5.1.6, Oct. 15, 1963) the cumulative "on" hours = 120 for most components. The "transient" content of this is estimated at or below 50% of this time or a total "dynamic" equivalent of 60 hours. The "dynamic" equivalent of the "continuously on" items (e.g. transponder) is also of the same order of magnitude during transit. During the first 100 hours after arrival at Mars, (i.e., 90 percent Available Mission Value life point) the "on" time is approximately 12 hours per day = 50 hours of "on" time which at this same "dynamic" content equals a maximum of 25 "dynamic" hours. The total dynamic hours for such a mission is then 85 hours. There will then be approximately 25 Dynamic Equivalent Missions in the programmed 2000 hours of testing per system. The four development systems would then provide 100 Dynamic Equivalent Test Missions of which the four 700 hour failure free periods would provide (2800/83) 33 failure free Dynamic Equivalent Missions. This would provide a statistical reliability and confidence in this aspect of

Mission performance of 98 percent at 50 percent Confidence (i.e., or 93 percent at 90 percent Confidence) using exponential reliability tables (R62SD135).

It is of interest to note that Figure 5.4-1 "Reliability Growth Demonstration", necessarily indicates an improvement in "time to failure" (i.e., reduced system failure rate as a function of testing time). This is comparable to using a Weibull distribution with a β value less than 1 in place of the exponential. Were such to be demonstrated as a β of 1/2 for example, the above test data would indicate a reliability of 0.998 at 50 percent Confidence (or 0.993 at 90 percent Confidence) using Weibull reliability tables (62SD172).

These long, steady-state test times must terminate in a dynamic test to assure complete start up and operation capability. This portion of the test plan will require particular attention in mechanical and electromechanical components and assemblies to assure that outgassing, physical and chemical changes, adhesion, etc. have not occurred to any adverse degree.

Confidence levels for these long time tests must be inferred from the design margin analysis and from the sample size, the trend data and the duration of test times available at time of launch decision. This area necessarily involves conclusions based upon tests of items not actually used in the specific flight hardware. However, a large amount of data is available for analysis and methods are available for making statistical correlation. Design margins are much more practicable for these steady state stresses and high confidence is considered practicable in every instance in which the test program is implemented. It would be a recommendation of the Reliability portion of this study that the Voyager program plan include a major effort in data compilation and reduction to Reliability Design Data form so as to make the extensive amount of parts and materials information for which tests have already been conducted more readily available for design use.

Reliability Improvement as a function of a planned program of testing during which there is time available for design and manufacturing error correction is easily demonstrated. Figure 5.4-1 graphically illustrates such "growth" in terms of steadily increasing "Time Since Last Failure". The fluctuation of individual failure times in the original data plots centered around the Reliability Growth Line For Spacecraft. In the case of the Nimco flight system No. 103, two sequential intervals above the "accept" line of 700 failure-free hours were obtained prior to shipment.

A cumulative MTBF line has been added to Figure 5.4-1 for reference. This is a more conservative line to draw since it retains in the data as failures all those failures for which corrective action has been taken to assure that such a failure cannot recur (i.e., on which the probability of recurrence has been greatly reduced). In Aircraft Accessories, which are considered as representative of mechanical and electromechanical items receiving improvement effort as any for which considerable amounts of data are available, the cumulative MTBF, has been remarkably consistent with the line shown.

It should be noted, however, that a major effect on such Reliability Growth lines is the opportunity to incorporate corrective actions and verify them by subsequent testing. The vested interests and costs involved once a design is in production and especially when it is in operational use are almost prohibitive of such opportunity. It is essential that spacecraft program planning schedule and meet advanced component and system testing dates and provide funding and opportunity to correct or rebuild deficient or high risk subsystems and test them to comparable maturity if highly reliable systems are to be attained. It is a reliability recommendation of this study that the Voyager program plans and schedules include this provision.

6. APPLICABILITY OF 1971 MARS SPACECRAFT DESIGNS TO 1972 VENUS MISSIONS

6.1 SUMMARY

6.1.1 ORBITER

An Orbiter designed for this 1971 Mars mission can be easily modified for the 1972 Venus mission (See Figure 6.1-1.). Guidance, attitude control, communication, and propulsion subsystems are essentially the same. The solar array used for Mars is reduced in area to suit Venus solar radiation and power requirements by omitting solar array segments and by removing a portion of the body mounted solar 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 1971 communication system are quite suitable for the mapping radar at Earth/Venus distances in the 1972 type II trajectory. Orbit is 1,000 x 13,000 nautical miles, inclined 67 degrees to equator. The All-Orbiter weight summary is shown in Table 6.2-1.

6.1.2 LANDER

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 80-90 degrees at the sub-earth point on Venus surface so that a 20-degree beamwidth fixed antenna or a vertically oriented descending entry Lander would intersect Earth. Data for the 10 hour mission would be 30×10^6 bits. The Lander would be designed and developed strictly for this mission and would have no relationship to the Mars Landers. Power supply during the surface mission would be primary batteries.

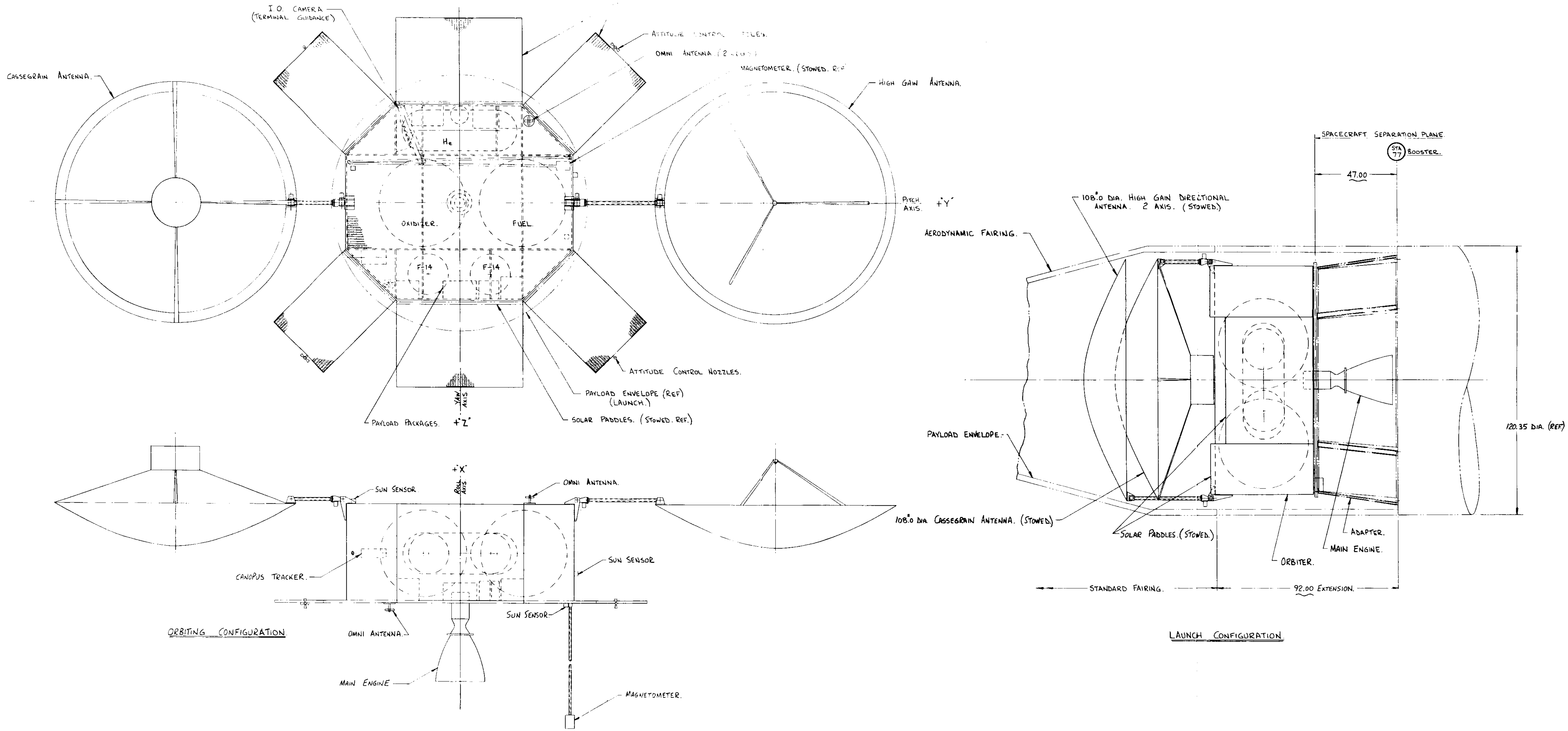


Figure 6.1-1. Mars Orbiter Adapted to 1972 Venus Mission

TABLE 6.1-1. ALL-ORBITER WEIGHT SUMMARY

Guidance and Control	212 pounds
Power Supply	181
Communications	227
Diagnostic Instrumentation	30
Science Payload	137
Propulsion	373
Thermal Control	49
Vehicle Harness	80
Structure	250
Total Orbiter	1538
Propellant	1982
Arrival Weight	3520
Mid-Course Fuel	36
Adapter and Δ V Shroud Weight	154
Total Injected Weight	3710 pounds - Orbit 1,000 x 13,000 nautical miles

6.2 1972 VENUS ORBITER

6.2.1 MISSION ANALYSIS

The power and communication data rate requirements for the radar, which is the prime experiment on a Venus Orbiter, were dominant in establishing the solar cell array area and the capability of the prime data link from Orbiter to Earth.

A cursory look at the Titan IIIC launch vehicle performance showed that 3710 pounds can be injected into transit for the 1972 Venus opportunity. The All Orbiter spacecraft, designed for the Mars 1971 mission, can be easily adapted to the Venus mission if a rather eccentric orbit, is utilized. This would allow the use of the mass propulsion system and propellant tanks, with minor modification for the Venus mission.

The mapping radar antenna from the 1970 Venus payload of the Saturn 1B Voyager study is substituted for the PHP on the Mars 1971 All Orbiter, and the small cloud-scanning vidicon TV and instrument package is again mounted on the radar antenna feed structure. This time, however, the package is aimed parallel to the base site of the antenna.

The mapping radar is the same SAHARA concept recommended in the report of the Saturn 1B Voyager study; this is a synthetic-aperture, side-scanning system. The resultant weight of the modified Orbiter permits an orbit of 1,000 x 13,000 n. mi.

This orbit is inclined at 67 degrees from the Venus equator, defined here as the intersection of Venus orbital plane with the surface of the planet. This inclination and required side looking angle of the radar system permits radar observation to be made of the "North" pole of Venus but not the "South" pole. The altitude is too high at the southern portion of the 1,000 x 13,000 orbit. Since the surface on either side of the orbit path will not have been observed by the radar system at the beginning of the mission, the antenna can be aimed at either side of the orbiting spacecraft. However, the orbital period of 8.14 hours and the swath width of the radar of 107 nautical-mile at periapsis, which occurs at a north latitude of $\sim$10 degrees, presents new area to be scanned every 3.4 orbits. Consequently, the operating sequence of the radar will be to scan one side for a complete available swath, (about 150 degrees of the Venus circumference), shift the antenna and instrument package to look straight down on the next pass and shift again to the other side for the third orbit, and continue the sequence throughout the mission. The vidicon TV and other planet pointing instruments will be operated mainly during the pass with the antenna pointing straight down. The radar mode in this pass is altitude and reflectivity measurement.

During the radar mapping portion of the orbit, which is 1.056 hours, 2.74×10^8 bits of information are stored in two tape recorders running simultaneously in order to obtain the necessary average rate of 72,000 bits/second. However, the instantaneous data rate of the radar is approximately 220,000 bits/second. The information is accepted alternately at this rate by one of two 100,000-bit plated wire buffer storage devices which supply each tape recorder at 48,000 bits/second.

The remainder of the orbit will have Earth sight and the 57 watt klystron and nine-foot dish planned for the 1971 Mars All-Orbiter mission which can provide 12,000 bits/second up to a distance of 1.405 AU is entirely adequate for this mission. The required data rate to completely empty the recorders during the remainder of the orbit period is 10,600 bits/second, leaving about 1400 bits/second for additional information. The altitude mode generates much less information and can be used to transmit TV pictures of the Venus cloud cover.

The communication distance is 0.92 AU at encounter and 1.47 AU 90 days after encounter. After the 1.4 AU distance is exceeded, the data rate is reduced by one half and the radar sequence is altered so that only the "leading" side of the progressing orbit plane is scanned once per three orbits.

Since the seasonal progression rate is 1.6 degrees/day, the initial scans on both sides of the orbit plane cover new ground for 28 days. After that period dual scanning produces repeat scans from the opposite side of the surface originally scanned by the leading side scan.

The orbital geometry was determined graphically and the resulting shadow times and power requirements determined above were used to establish the power profile. (See Figure 6.2-1.) The transit portions of the power profile are the same as for the 1971 Mars all orbiting mission. Batteries and solar array were sized on the basis of the maximum shadow orbit.

6.2.2 CHANGES REQUIRED IN THE MARS 1971 ALL ORBITER IN ORDER TO FLY TO VENUS IN 1972

The power required for the Venus 1972 trip is the same as that for the Mars 1971 Orbiter-600 watts. Therefore, in order to make as few changes as possible in the design, all solar panels except the six rectangular areas attached to the side of the Orbiter have been removed. In addition, 32 percent of the body-mounted solar cells from the top of the Orbiter have been removed. This gives 600 watts for Venus 1972 and effectively provides the same thermal control design within the Orbiter.

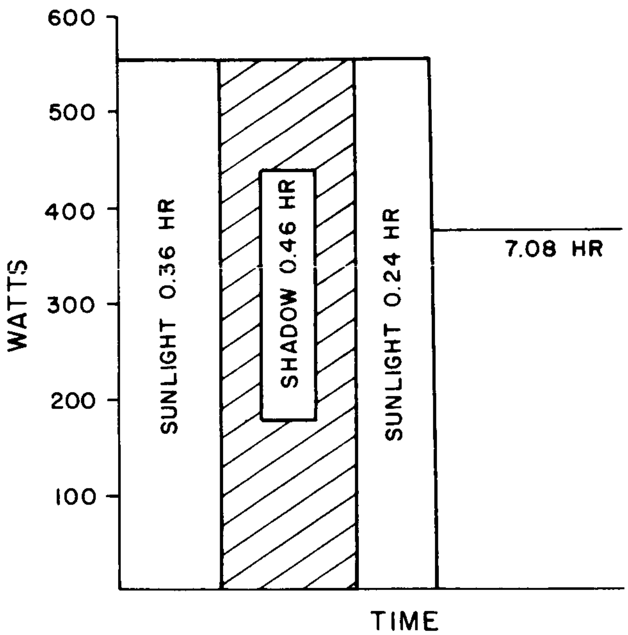


Figure 6.2-1. Venus 1972 All-Orbiter Power Profile - Orbiting Phase

Because of the change in instrumentation and experiments required for Venus 1972, a nine-foot diameter radar mapper antenna is required. Therefore, the PHP has been removed and a nine-foot Cassegrain antenna has been located in the PHP position. Mounted on the forward surface of the antenna are the items of instrumentation which required planet pointing capability.

The overall weight distribution of the Venus 1972 Orbiter has been obtained in the following manner:

1. Obtain instrumentation weight
2. Obtain total subsystem weight necessary to provide system capability
3. Subtract these weights from the injected weight to obtain fuel capability. The fuel capability then determines allowable orbit.

A. <u>SUBSYSTEM REVISIONS</u>

Communications	- none
Power	- reduction in solar cell area

Structure	- none
Thermal Control	- as required, except weight will be approximately the same
Guidance and Control	- none
Propulsion	- changes in propellant quantity and tank diameters

6.3 1972 VENUS LANDER

A Bus/Lander designed for Mars cannot be directly applied to a Venus mission primarily because of the hot Venus environment. Lander surface operation time is severely limited by the requirement of expending stored refrigerant during the surface mission to control payload temperatures. This shortens the surface mission from six months on Mars to a few hours on Venus and eliminates the use of a Radioisotope thermoelectric Generator as a power supply for the Lander. Primary batteries are far lighter than RTG's for such short durations. With no RTG in the Lander, the Bus must be equipped with a solar cell array for power during transit. This array can be sized for the requirements of transmitting the terminal guidance TV observations at the encounter distance of 0.92 AU.

A Bus/Lander mission precludes an accompanying Orbiter except one established in a coordinated all Orbiter mission launched by another Titan IIIC booster. The combined Orbiter/Lander approach is not competitive because of the small Lander weight possible with a useable Orbiter on a Titan IIIC launch vehicle.

If a relay link is the sole method of transmitting data from the Lander to Earth for a Bus/Lander configuration, then the operating time for the Lander would be limited to duration of line-of-sight between a Bus equipped with a repeater or other relay equipment and the descending Lander. Maximum surface time would be about 10 minutes and the descent would have to be a hurried one without taking advantage of the dense atmosphere to make a slow descent with TV observation above, in and below the cloud layer. The information rate for such a relay link can be as high as 50,000 bits/second or six TV frames/minute, with the total data transmitted from the Lander to the Bus on a variable rate basis (using preprogrammed tape recorder drive speeds) of $\sim 40 \times 10^6$ bits. (The information would be stored in the Bus and transmitted to Earth at the slower rate of the Bus communication system.)

The severely restricted surface durations of a relay type Bus/Lander mission leads to the consideration of direct communication from the surface of Venus to Earth. But the probability of erecting and orienting a steerable high gain antenna appears to be slight in view of the currently expected high temperature of the surface environment. However, a moderate gain antenna, 20-degree beamwidth, fixed to the center of the aft cover in the same location as the omnidirectional broad beam antenna used for the direct link for post separation and descent communication on the Mars Landers, described in Section 2.4.2 would be used on Venus. This would provide a data rate of ~ 1000 bits/second with the same 100-watt transmitter used on the Mars Landers. A 10-hour descent and surface mission could transmit $\sim 30 \times 10^6$ bits direct to earth.

The fixed antenna would require the landing site to be near the sub-Earth point on the surface of Venus. This would mean, for the Venus 72 type II trajectory selected for its low injection energy requirements, that the entry angle would be 80 to 90-degrees. The Venus Landers in the Saturn 1B Voyager study were designed for a 90 degree entry angle with peak 325 g axial deceleration.

This steep entry angle eliminates the effect of uncertainty of atmospheric determination on the variation in down-range dispersion of the landing site and thus insures the accurate placement of the Lander. When the Lander is oriented to local Venus gravity, its fixed antenna beam would contain Earth. Descent of the Lander would be slowed by a high temperature parachute in order to perform as many experiments as possible during descent while the orientation of the Lander hanging on its chute is assured (depending on unknown atmospheric turbulence, of course).

The Earth/Venus encounter geometry for a 1972 type II trajectory shows that the sub-Earth point will be in sunlight about 15 degrees from the terminator. This should be sufficient illumination for descent TV.

Power supply for the entire post separation phase of the Bus/Lander mission is by primary battery.

Thermal control for the expected hot atmosphere is by stored refrigerant. Ammonia was recommended in the Saturn Voyager Study report.

The relatively large weight of this Venus Lander, 3,000 pounds, should permit the sustaining of surface operation for 8 to 10 hours. Since communication is continuous and direct, battery, refrigerant weight, descent rates and payload can be balanced to provide maximum mission value without regard to a particular orbital period for the Orbiter relay link that was selected in the Saturn 1B Voyager study.

The weight capacity and expected surface survival duration also indicate the application of command capability. It was not utilized for either Lander size described in the Saturn 1B Voyager study.

7. PROGRAM PLAN, AND PROGRAM COST AND SCHEDULE COMPARISONS

7.1 PROGRAM PLAN

7.1.1 SUMMARY

The Titan III-C 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 are:

a. Three Orbiters - (2 flight units, 1 backup unit plus replaceable spare components)

b. Five Landers - (3 flight units, 1 backup unit and 1 sterile spare unit)

c. Four Buses - (3 flight units, 1 backup unit plus replaceable spare components).

The program cost estimates, schedules and development problems summarized in Figure 7.1-1, Program Plan Summary, 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 Lander/Bus programs and missions, if desired.

The costs for such separate programs, the combined program and the Saturn 1B Voyager program are shown in Figure 7.1-2; Program Cost Summary.

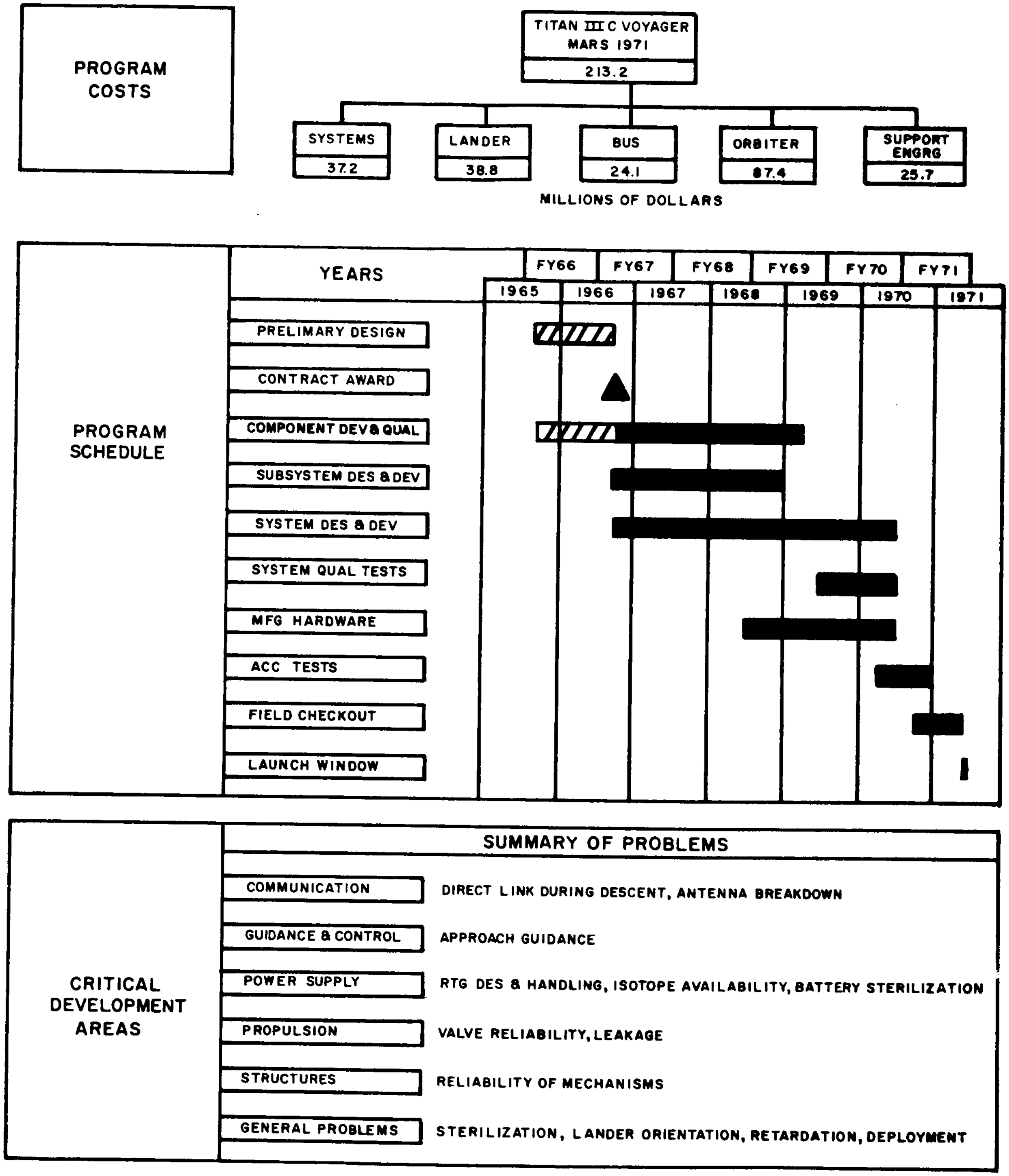


Figure 7.1-1. Program Plan Summary

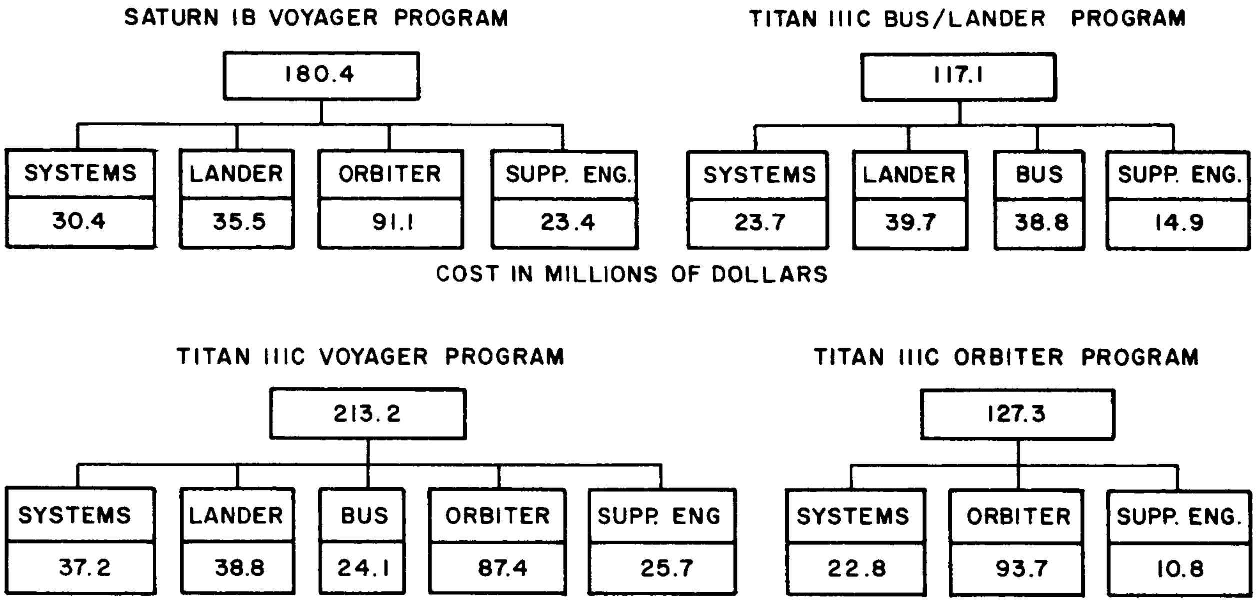


Figure 7.1-2. Program Cost Summary

7.1.2 TITAN IIIC VOYAGER PROGRAM COSTS - MARS 1971

A. COSTING GROUND RULES

The following rules have been observed in preparation of cost estimates:

1. Titan IIIC Voyager program equivalent to Saturn 1B Voyager program requires following delivery of hardware:

    2 - Orbiters - Flight Units

    1 - Orbiter - Backup Unit

    1 set - replaceable Orbiter components

    3 - Landers - Flight Units

    1 - Lander - Backup (to be mated to spare Bus - ready for launch).

    1 - Lander - Spare - (sterilized - ready for mating to Bus)

    2 sets - Ground Support Equipment for Handling, Servicing and Checkout of spacecraft in the field.

2. Costs are based on Saturn 1B Voyager estimates with incremental adjustments as dictated by Titan IIIC Voyager design.

3. The following costs are excluded:

   a. Scientific Payload and TV.

   b. RTG units and radioisotopes.

   c. Launch Vehicle costs.

   d. Post-launch costs.

4. Costs for Titan IIIC Voyager Program assume simultaneous development and manufacture of Orbiter, Lander and Bus spacecraft.

5. Costs for Titan IIIC Orbiter Program assume a separate Orbiter program.

6. Costs for Titan IIIC Lander/Bus Program assume a separate Lander/Bus program.

7. Unit cost is defined as the additional cost for manufacture and test of one additional unit.

B. PROGRAM COSTS

Figure 7.1-3, Titan IIIC Complete Program Costs, shows program costs down to the subsystem level.

The cost elements for these estimates are defined in Table 7.1-1 through 7.1-5.

Table 7.1-6, Program Costs by quarters, shows the complete program costs by fiscal quarters, and Figure 7.1-4 indicates the Program Expenditure Rate.

C. ORBITER PROGRAM COSTS

The Titan IIIC Orbiter program costs shown in Figure 7.1-5 have been derived by extracting appropriate elements from the complete program costs. They have been adjusted to reflect changes caused by reductions in the number of spacecraft types and quantities required and the elimination of interrelated or common development tasks for Orbiter and Lander/Bus which existed in the complete program. The quantity of Orbiters, spares and support equipment are equivalent to those delivered in the Orbiter portion of the complete program.

They represent the cost of a complete Orbiter spacecraft program for the Titan IIIC launch vehicle, subject to the exclusions listed in 7.1.2(A.), Ground Rules.

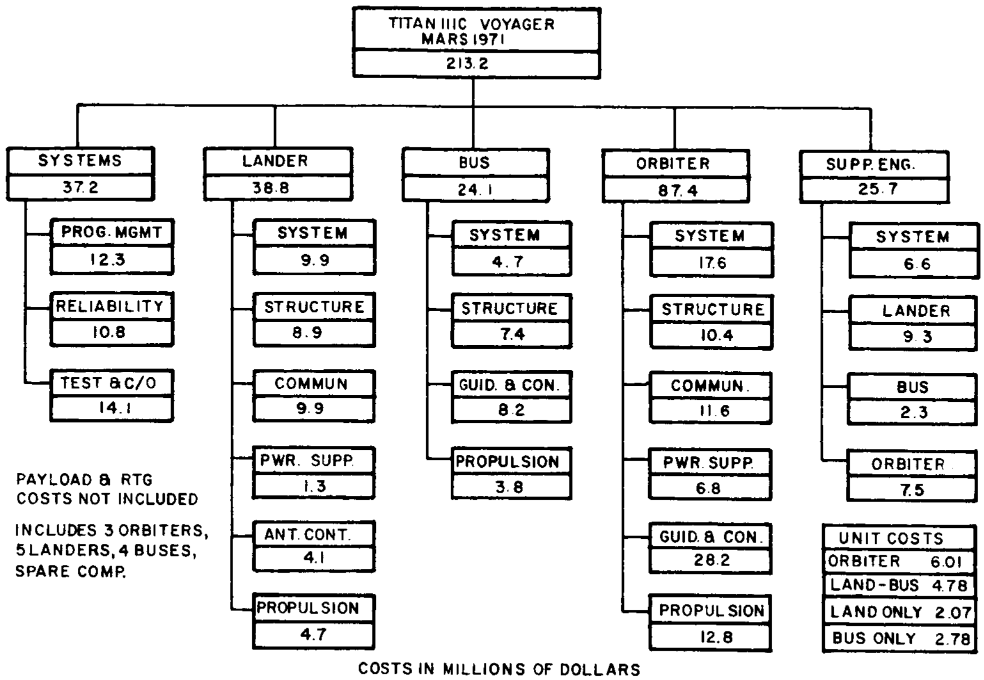
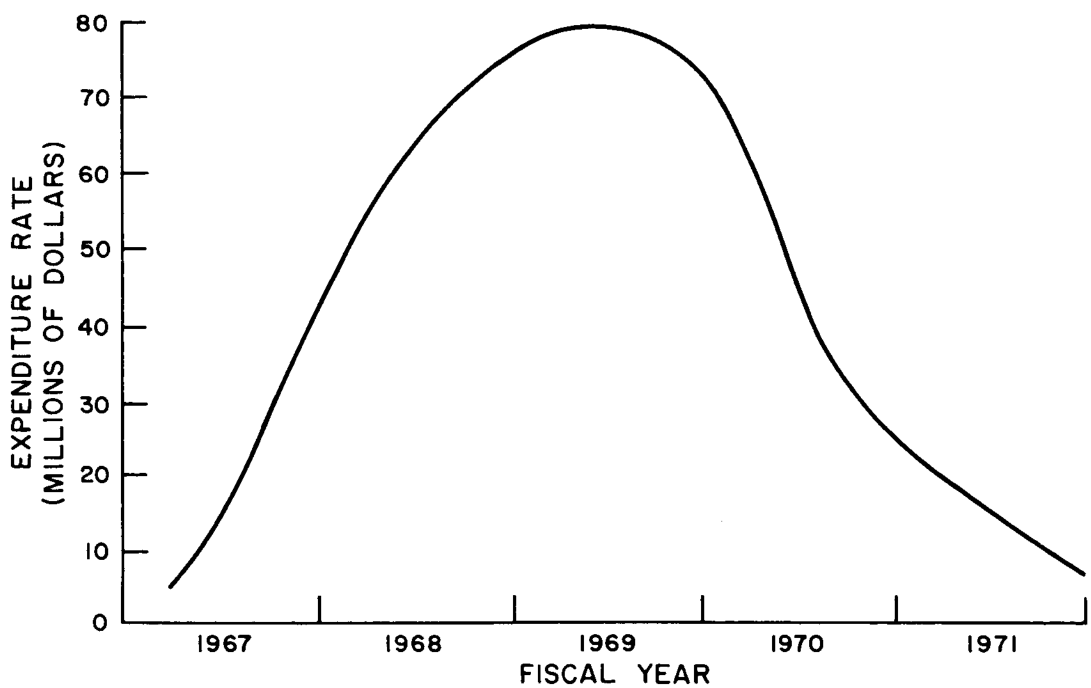


Figure 7.1-3. Titan IIIC Program Costs



14	62	78	45	14

TOTAL COSTS BY FISCAL YEAR (MILLIONS OF DOLLARS)

Figure 7.1-4. Titan IIIC Program Expenditure Rate

TABLE 7.1-1. TASK DEFINITIONS FOR MARS 1971 COST ELEMENTS - VOYAGER SYSTEMS

SYSTEMS

$37.2 x 10^6

Sub-Tasks	Activities	Hardware Required	Cost Descriptions
Program Management	Plans and Schedules	None	Includes all manpower required to perform these activities throughout the MARS 1971 Program.
	Program Control & Measurements	None	
	Reports & Documents	None	
	System Engineering & Integration	None	
	Contract Administration	None	
	Finance	None	
	Sterilization Management & Control	None	
Reliability	Parts Evaluation	Piece parts	
	Sterilization Effects	Piece parts	
	Parts Acceptance Test	All production piece parts	
	Analysis & Apportionment	None	
Systems Test & Checkout	System Development Testing	None costed - use Lander, Bus and Orbiter system development hardware	Includes all test manpower, special equipment & facility costs associated with Voyager system testing.
Systems Test & Checkout (Cont'd)	System Qualification Testing	None costed - use Lander, Bus & Orbiter system qualification hardware	Includes all test manpower, special equipment & facility costs associated with Voyager system testing.
	System Acceptance Testing	Costed under Lander, Bus and Orbiter system	Includes all test manpower, special equipment & facility costs associated with Voyager system testing.
	Field Test & Checkout	None	Field equipment costs part of Support Equipment task.

TABLE 7.1-2. TASK DEFINITIONS FOR MARS 1971 COST ELEMENTS - LANDER SYSTEM

LANDER

$\$38.8 \times 10^6$

Sub-Tasks	Activities	Hardware Required	Cost Descriptions
System	Design & Development	Prototype hardware for lander system development test	Includes fabrication, assembly and test costs.
	Qualification	2 lander system qualification units	Includes all materials, fabrication assembly and acceptance test of both lander units. Also includes the qualification testing unique to the lander system on one unit.
	Production-Final Assembly	4 Flight Units and 1 spare Lander	Includes all costs for final assembly of flight units including tooling.
	Production-QC&T	4 Flight Units and 1 spare Lander	Includes all costs for inspection and acceptance test of flight units during and after final assembly.
Structure-Sub-System Consists of: Shield & Structure Environmental Control Retardation Orientation Deployment Adaptor	Design & Development	Development hardware	Includes all the design and analysis associated with this subsystem and the materials, fabrication, assembly and test required for the development hardware used.
	Qualification	2 Sets of qualification hardware (not including structure)	Includes all Mfg & QC costs associated with the hardware to be qualified and the costs of manpower and equipment for qualification testing.
	Production-Mfg	4 Sets of Structure Subsystem hardware and 1 set of spares (not including structure)	Includes all manufacturing costs associated with this hardware up to but not including final assembly of system.
	Production-QC&T	4 Sets of Structure Subsystem hardware and 1 set of spares (not including structure)	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of system.
Communications-Sub-System Consists of: Earth Link Data Storage & Processing Command Power Conversion	Design & Development	Development hardware	Includes all the design and analysis associated with this subsystem and the materials, fabrication, assembly and test required for the development hardware used.
	Qualification	2 Sets of qualification hardware	Includes all Mfg & QC costs associated with the hardware to be qualified and the costs of manpower and equipment for qualification testing.
	Production-Mfg	4 Sets of Communication Subsystem flight hardware and 1 set of spares	Includes all manufacturing costs associated with this hardware up to but not including final assembly of system.
	Production-QC&T	4 Sets of Communication Subsystem flight hardware and 1 set of spares	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of system.
Power Supply-Sub-System Consists of: Batteries Regulation, control and distribution	Design & Development	Development hardware	Includes all the design and analysis associated with this subsystem and the materials, fabrication, assembly, and test required for the development hardware used.
	Qualification	2 Sets of qualification hardware or equivalent for batteries	Includes all Mfg & QC costs associated with the hardware to be qualified and the costs of manpower and equipment for qualification testing.
	Production-Mfg	4 Sets of Power Supply Subsystem hardware and 1 set of spares	Includes all manufacturing costs associated with this hardware up to but not including final assembly of system.
	Production-QC&T	4 Sets of Power Supply Subsystem hardware and 1 set of spares	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of system.
Antenna Control-Sub-System	Design & Development	Development hardware	Includes all the design and analysis associated with this subsystem and the materials, fabrication, assembly, and test required for the development hardware used.
	Qualification	2 Sets of qualification hardware	Includes all Mfg & QC costs associated with the hardware to be qualified and the costs of manpower and equipment for qualification testing.
	Production-Mfg	4 Sets of Antenna Control Subsystem flight hardware and 1 set of spares	Includes all manufacturing costs associated with this hardware up to but not including final assembly of system.
	Production-QC&T	4 Sets of Antenna Control Subsystem flight hardware and 1 set of spares	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of system.
Propulsion-Sub-System Consists of: Rocket Spin-de-Spin	Design & Development	Development hardware	Includes all the design and analysis associated with this subsystem and the materials, fabrication, assembly and test required for the development hardware used.
	Qualification	2 or equivalent sets of qualification hardware	Includes all Mfg & QC costs associated with the hardware to be qualified and the costs of manpower and equipment for qualification testing.
	Production-Mfg	4 Sets of Propulsion Subsystem flight hardware and 1 set of spares	Includes all manufacturing costs associated with this hardware up to but not including final assembly of system.
	Production-QC&T	4 Sets of Propulsion Subsystem flight hardware and 1 set of spares	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of system.

TABLE 7.1-3. TASK DEFINITIONS FOR MARS 1971 COST ELEMENTS - BUS SYSTEM

BUS
$24.1 x 10^6

Sub-Tasks	Activities	Hardware Required	Cost Descriptions
System	Design & Development	3 System prototypes-electrical, structural, thermal	Includues all bus system design and analysis and the fabrication, assembly and test costs of three bus system prototypes (less landers)
	Qualification	1 System qualification unit	Includes all manufacturing and quality control costs to procure, fabricate, assembly and acceptance test this unit. Also includes test costs for qualifying bus less lander.
	Production-Final Assembly	4 Flight Units	Includes all costs for final assembly of flight units including tooling.
	Production-Quality Control & Test (QC&T)	4 Flight Units	Includes all costs for inspection and acceptance test of flight units during and after final assembly.
Structure-Sub-System Consists of: Spacecraft Structure Antenna Structure Antenna Drive & Deployment Separation Mechanisms	Design & Development	Development hardware	Includes all the design and analysis associated with this subsystem and the materials, fabrication, assembly and test required for the development hardware used.
	Qualification	2 Sets of qualification hardware (not including structure) or equivalent quantities for pyrotechnics	Includes all Mfg & QC costs associated with the hardware to be qualified and the costs of manpower and special equipment for qualification testing.
	Production-Mfg	4 Sets of Structure flight hardware and 1 set of spares (not including basic structures)	Includes all manufacturing costs associated with this hardware up to but not including final assembly of system.
	Production-QC&T	4 Sets of Structure flight hardware and 1 set of spares (not including basic structures)	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of system.
Guidance & Control Sub-System Consists of: Attitude Control Antenna Control Thrust Vector Control Logic and Storage Power Conversion	Design & Development	Development hardware	Includes all the design and analysis associated with this subsystem and the materials, fabrication, assembly and test required for the development hardware used.
	Qualification	2 Sets of qualification hardware	Includes all Mfg & QC costs associated with the hardware to be qualified and the costs of manpower and equipment for qualification testing.
	Production-Mfg	4 Sets of Guidance & Control Subsystem flight hardware and 1 set of spares	Includes all manufacturing costs associated with this hardware up to but not including final assembly of system.
	Production-QC&T	4 Sets of Guidance & Control Subsystem flight hardware and 1 set of spares	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of system.
Propulsion-Sub-System Consists of: Main engine Attitude control	Design & Development	Development hardware	Includes all the design and analysis associated with this subsystem and the materials, fabrication, assembly and test required for the development hardware used.
	Qualification	2 Sets of qualification hardware or equivalent quantities for rocket qualification	Includes all Mfg & QC costs associated with the hardware to be qualified and the qualification testing costs of manpower and equipment.
	Production-Mfg	4 Sets of Propulsion Subsystem flight hardware and 1 set of spares	Includes all manufacturing costs associated with this hardware up to but not including final assembly of system.
	Production-QC&T	4 Sets of Propulsion Subsystem flight hardware and 1 set of spares	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of system.

TABLE 7.1-4. TASK DEFINITIONS FOR MARS 1971 COST ELEMENTS - ORBITER SYSTEM

ORBITER

$87.4 x 10^6

Sub-Tasks	Activities	Hardware Required	Cost Descriptions
System	Design & Development	3 System prototypes-electrical, structural, thermal	Includes all orbiter system design and analysis and the fabrication, assembly and test costs of three orbiter system prototypes (less landers).
	Qualification	1 System qualification unit	Includes all manufacturing and quality control costs to procure, fabricate, assemble and acceptance test this unit. Also includes test costs for qualifying orbiter less lander.
	Production-Final Assembly	3 Flight Units	Includes all costs for final assembly of flight units including tooling.
	Production-Quality Control & Test (QC&T)	3 Flight Units	Includes all costs for inspection and acceptance test of flight units during and after final assembly.
Structure-Sub-System Consists of: Spacecraft Structure PHP Structure Antenna Structure Antenna & PHP Drives & Deployment Separation Mechanisms	Design & Development	Development hardware	Includes all the design and analysis associated with this subsystem and the materials, fabrication, assembly and test required for the development hardware used.
	Qualification	2 Sets of qualification hardware (not including structure) or equivalent quantities for pyrotechnics	Includes all Mfg & QC costs associated with the hardware to be qualified and the costs of manpower and special equipment for qualification testing.
	Production-Mfg	3 Sets of structure flight hardware and one set of spares (not including basic structures)	Includes all manufacturing costs associated with this hardware up to but not including final assembly of system.
	Production-QC&T	3 Sets of structure flight hardware and one set of spares (not including basic structures)	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of system.
Communication-Sub-System Consists of: Earth Link Data Storage & Processing Command Power Conversion	Design & Development	Development Hardware	Includes all the design and analysis associated with this subsystem and the materials, fabrication, assembly and test required for the development hardware used.
	Qualification	2 Sets of qualification hardware	Includes all Mfg & QC costs associated with the hardware to be qualified and the costs of manpower and equipment for qualification testing.
	Production-Mfg	3 Sets of Communication Subsystem flight hardware and one set of spares	Includes all manufacturing costs associated with this hardware up to but not including final assembly of system.
	Production-QC&T	3 Sets of Communication Subsystem flight hardware and one set of spares	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of orbiter.
Power Supply-Sub-System Consists of: Solar array Batteries Regulation Control & Distribution	Design & Development	Development hardware	Includes all the design and analysis associated with this subsystem and the materials, fabrication, assembly and test required for the development hardware used.
	Qualification	2 Sets of qualification hardware	Includes all Mfg & QC costs associated with the hardware to be qualified and the costs of manpower and equipment for qualification testing.
	Production-Mfg	3 Sets of Power Supply Subsystem flight hardware and one set of spares	Includes all manufacturing costs associated with this hardware up to but not including final assembly of system.
	Production-QC&T	3 Sets of Power Supply Subsystem flight hardware and one set of spares	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of orbiter system.
Guidance & Control Sub-System Consists of: Attitude Control Antenna Control PHP Control Thrust Vector Control Logic and Storage Power Conversion	Design & Development	Development hardware	Includes all the design and analysis associated with this subsystem and the materials, fabrication, assembly and test required for the development hardware used.
	Qualification	2 Sets of qualification hardware	Includes all Mfg & QC costs associated with the hardware to be qualified and the costs of manpower and equipment for qualification testing.
	Production-Mfg.	3 Sets of Guidance & Control Subsystem flight hardware and one set of spares	Includes all manufacturing costs associated with this hardware up to but not including final assembly of system.
	Production-QC&T	3 Sets of Guidance & Control Subsystem flight hardware and one set of spares	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of system.
Propulsion-Sub-System Consists of: Main engine Attitude control	Design & Development	Development hardware	Includes all the design and analysis associated with this subsystem and the materials, fabrication, assembly and test required for the development hardware used.
	Qualification	2 Sets of qualification hardware or equivalent quantities for rocket qualification	Includes all Mfg & QC costs associated with the hardware to be qualified and the qualification testing costs of manpower and equipment.
	Production-Mfg	3 Sets of Propulsion Subsystem flight hardware and one set of spares	Includes all manufacturing costs associated with this hardware up to but not including final assembly of system.
	Production-QC&T	3 Sets of Propulsion Subsystem flight hardware and one set of spares	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of system.

TABLE 7.1-5. TASK DEFINITIONS FOR MARS 1971 COST ELEMENTS - SUPPORT EQUIPMENT SYSTEMS

SUPP. ENG.

$\$25.7 \times 10^6$

Sub-Task	Activities	Hardware Required	Cost Descriptions
Spacecraft	Design, Development & Manufacture	Hardware unique to support of Voyager System	Includes all the costs associated with the design development and manufacture of this equipment.
Orbiter	Design, Development & Manufacture	Hardware unique to support of Orbiter System	Includes all the costs associated with the design, development and manufacture of this equipment.
Lander	Design, Development & Manufacture	Hardware unique to support of Lander System	Includes all the costs associated with the design, development and manufacture of this equipment.
Bus	Design, Development & Manufacture	Hardware unique to support of Bus System	Includes all the costs associated with the design, development and manufacture of this equipment.

TABLE 7.1-6. MARS 1971 TITAN IIIC
VOYAGER COST SUMMARY

	66	1967				1968					1969					1970					1971				
	FY 67				FY 68					FY 69					FY 70					FY 71					Program Totals
	2	3	4	Tot.	1	2	3	4	Tot.	1	2	3	4	Tot.	1	2	3	4	Tot.	1	2	3	4	Tot.	
VOYAGER SYSTEM																									
Prog. Management	290	311	428	1,029	568	755	758	777	2,858	774	768	761	755	3,058	754	752	720	702	2,928	658	616	616	507	2,397	12,270
Reliability	194	533	858	1,585	956	968	901	773	3,598	765	761	760	705	2,991	501	460	408	399	1,774	320	264	178	90	852	10,800
System Test & C/O						42	192	562	796	696	1,003	666	580	2,945	696	1,124	1,132	1,412	4,364	2,185	2,906	1,334	580	6,005	14,110
System Total	484	844	1,286	2,614	1,524	1,765	1,851	2,112	7,252	2,235	2,532	2,187	2,040	8,994	1,957	2,336	2,260	2,513	9,066	3,163	2,786	2,128	1,177	9,254	37,180
ORBITER SYSTEM																									
Design & Dev.	740	1,429	3,691	5,860	5,231	5,354	5,914	6,607	23,106	5,141	3,598	3,114	2,630	14,483	2,144	1,516	1,405	1,026	6,091	383	376			759	50,299
Qual. Testing		184	210	394	506	643	850	1,044	3,043	2,181	2,647	2,147	1,600	8,575	647	609	154	140	1,550						13,562
Mfg. Hardware		41	46	87	82	123	159	311	675	1,675	2,090	2,150	2,718	10,133	1,864	2,296	1,174	547	5,881	395	202			597	15,873
Prod. Testing			49	49	135	256	353	529	1,273	692	850	820	813	3,175	785	656	787	444	2,672	334	162			496	1,665
Orbiter Total	740	1,654	3,996	6,390	5,954	6,376	7,276	8,4[illegible]	28,097	9,689	9,185	8,231	7,761	34,866	5,440	5,077	3,520	2,157	16,194	1,112	740			1,852	87,399
LANDER SYSTEM																									
Design & Dev.	449	1,040	1,660	3,149	2,010	2,433	2,365	2,075	8,883	1,500	916	709	468	3,593	313	181	14	9	517						16,142
Qual. Testing			20	20	46	109	289	783	1,227	1,366	1,688	1,308	1,088	5,450	863	225	225	127	1,440						8,137
Mfg. Hardware							62	160	222	504	1,230	2,208	2,022	5,964	909	770	276	202	2,157	46	9			55	8,398
Prod. Testing					11	69	256	664	1,000	938	776	355	370	2,439	663	573	493	420	2,149	304	202			506	6,094
Lander Total	449	1,040	1,680	3,169	2,067	2,611	2,972	3,682	11,332	4,308	4,610	4,580	3,948	17,446	2,748	1,749	1,008	758	6,263	350	211			561	38,771
BUS SYSTEM																									
Design & Dev.	199	385	986	1,570	1,410	1,445	1,592	1,940	6,387	1,385	970	850	706	3,911	576	409	379	276	1,640	103	103			206	12,140
Qual. Testing		47	57	104	136	173	229	282	820	589	985	444	297	2,315	174	164	42	38	418						2,656
Mfg. Hardware		10	13	23	22	33	43	84	182	451	699	715	868	2,733	503	615	317	147	1,582	107	54			161	5,099
Prod. Testing	1[illegible]		13	13	37	69	96	142	344	185	226	221	219	851	212	177	212	119	720	90	43			133	4,218
Bus Total	199	442	1,069	1,710	1,605	1,720	1,960	2,448	7,733	2,610	2,880	2,230	2,090	9,810	1,465	1,365	950	580	4,360	300	200			500	24,113
SUPPORT ENGINEERING																									
Design & Dev.		105	320	425	941	1,960	1,622	3,022	7,545	1,819	1,749	1,735	1,583	6,886	1,412	1,015	812	678	3,917	522				522	19,295
Mfg. Hardware															1,194	2,390	1,194	607	5,385	530				530	5,915
OP.,I & C/O																					296	144	74	514	514
Supp. Eng. Total		105	320	425	941	1,960	1,622	3,022	7,545	1,819	1,749	1,735	1,583	6,886	2,606	3,405	2,006	1,285	9,302	1,052	296	144	74	1,566	25,724
Total Program	1,872	4,085	8,351	14,308	12,091	14,432	15,681	19,755	61,959	20,661	20,956	18,963	17,422	78,002	14,216	13,932	9,744	7,293	45,185	5,977	4,233	2,272	1,251	13,733	213,187

Costs in Thousands of Dollars

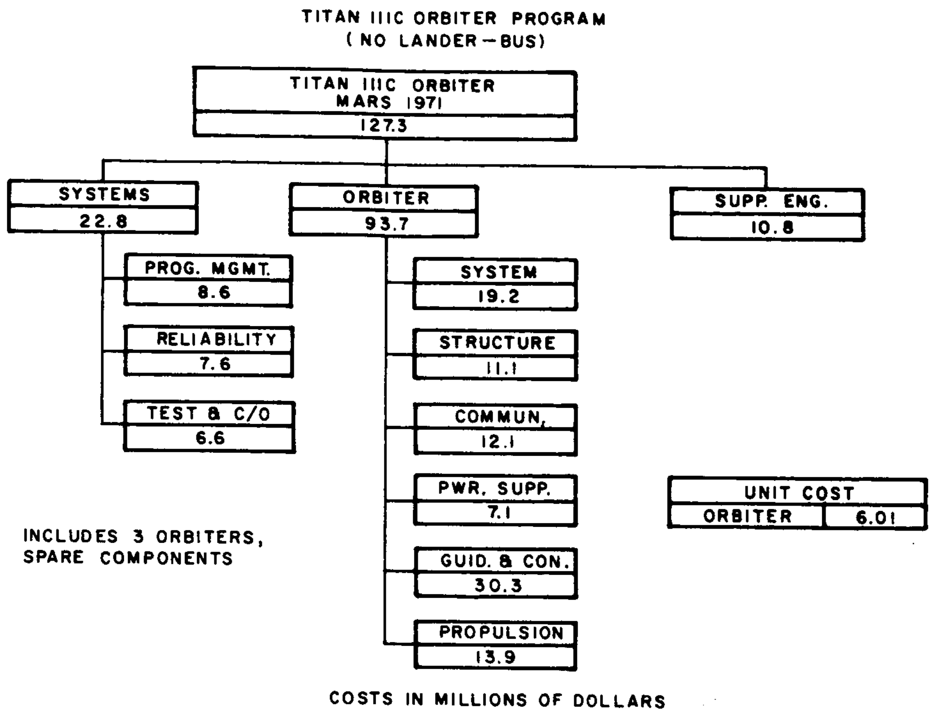


Figure 7.1-5. Orbiter Program Costs

D. LANDER/BUS PROGRAM COSTS

The Titan IIIC Lander/Bus program costs shown in Figure 7.1-6 have also been extracted from the complete program costs and adjusted to reflect elimination of contributions from an Orbiter development program.

These costs represent the costs of a complete Lander/Bus program for the Titan IIIC launch vehicle, subject to the exclusions of the Ground Rules of Section 7.1.2(A). Quantities delivered are identical to the Lander/Bus portion of the complete program.

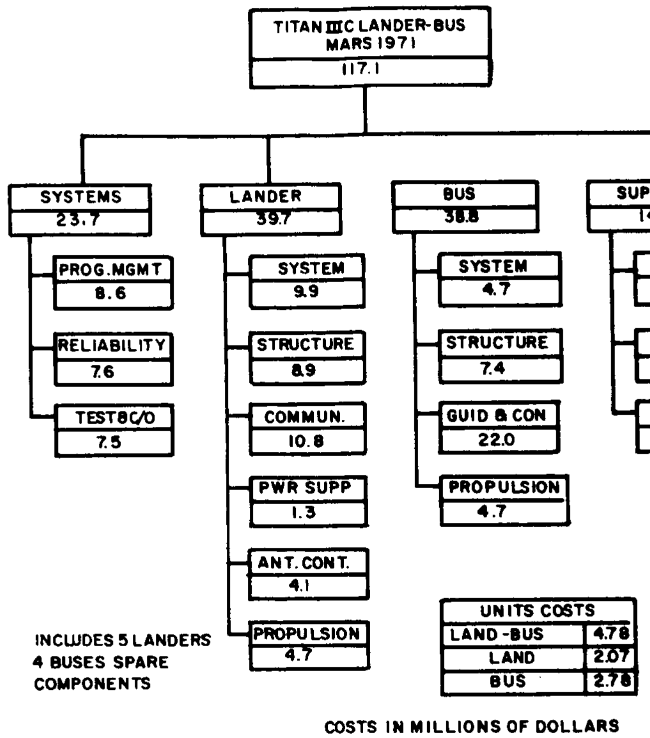


Figure 7.1-6. Bus/Lander Program Costs

7.1.3 SCHEDULE FOR TITAN IIIC VOYAGER PROGRAM - MARS 1971

A. SCHEDULING GROUND RULES

The following rules have been observed in preparation of the schedule:

1. 1965 State-of-Art, as applied during Saturn 1B Voyager Design Study.
2. Schedule back from 1971 Mars launch window.
3. One-year period prior to hardware contract for preliminary designs and their evaluation by NASA.
4. Critical component development started during preliminary design phase.
5. Test program includes qualification of components and systems, as outlined in Saturn 1B Voyager Integrated Test Plans.
6. Flight units to AMR four months prior to launch.

B. PROGRAM SCHEDULE

This schedule, shown in Figure 7.1-7, has been prepared using the Saturn 1B Voyager schedule as a base to work from. Changes have been made which reflect the increase

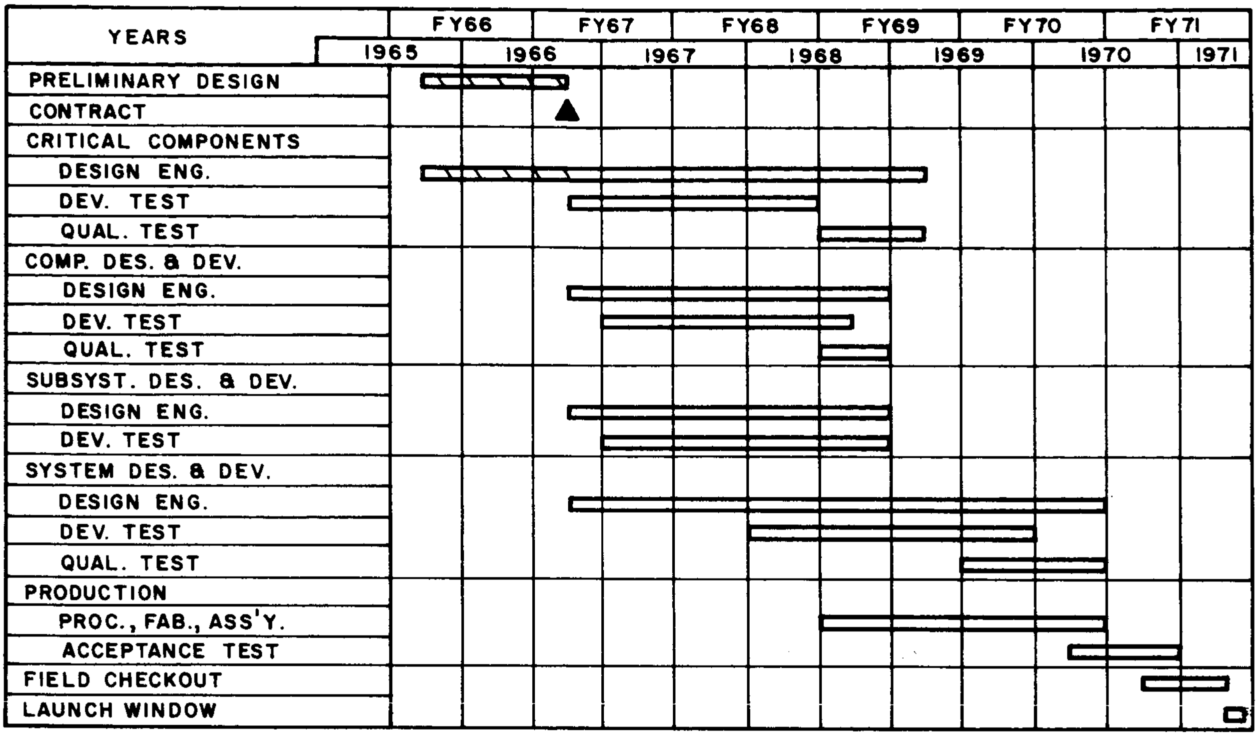


Figure 7.1-7. Program Schedule (Titan IIIC Voyager)

in the number of types and quantity of hardware to be developed, manufactured and tested, principally due to the addition of the Bus spacecraft.

The preliminary design effort and development of critical components during a one-year period prior to the hardware contract are considered essential to conduct the program on the schedule shown.

The spacecraft hardware to be delivered in Saturn 1B and Titan IIIC equivalent programs are listed in Table 7.1-7.

TABLE 7.1-7. SPACECRAFT HARDWARE FOR THE SATURN 1B and TITAN IIIC EQUIVALENT PROGRAMS

Saturn 1B		Titan III-C
	Flight Systems	
2 Orbiters		2 Orbiters
4 Landers		3 Landers
		3 Buses
	Backup Flight System	
1 Orbiter		1 Orbiter
2 Landers		1 Lander
		1 Bus
	Spares	
1 Lander (sterile)		1 Lander (sterile)
1 set replaceable components for Orbiter		1 set replaceable components for Orbiter & Bus
	Totals	
3 Orbiters		3 Orbiters
7 Landers		5 Landers
1 set replaceable components for Orbiter		4 Buses
		1 set replaceable components for Orbiter & Bus

The costs and schedules presented for equivalent programs are based on the above totals.

C. 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.

7.1.4 DEVELOPMENT PROBLEM AREAS

Critical problems in development of the Titan IIIC Voyager are summarized in Figure 7.1-1 Program Plan Summary, by subsystem affected.

General development problem areas were studied during the Voyager Design Study. These have been reviewed for applicability to the Titan IIIC Voyager system and an updated summary of them is presented in Figure 7.1-8, Development Problems Summary.

DEVELOPMENT PROBLEMS SUMMARY

PROBLEM AREA	PROBLEMS	AFFECTED OR RELATED ITEMS & ACTIVITIES
LANDER STERILIZATION EFFECTS & REQUIREMENTS	EFFECTS ON MATERIALS	TV CAMERA TUBE SCREENS - PARACHUTE MATERIAL - SOLID PROPELLANT - ADHESIVES
	EFFECTS ON PARTS & COMPONENTS	DATA RECORDER - ELECTRONIC PARTS - VIDICON - IMAGE ORTHICON - PYROTECHNICS - SCIENTIFIC EXPERIMENTS
	EFFECTS ON LANDER SYSTEM DESIGN	LANDER STRUCTURE - CHECKOUT & SERVICING PROCESSES & ACCESS - STERILE BARRIER DESIGN - INTERFACES
	EFFECTS ON PROD. PROCESSES & FACILITIES	CLEAN AREAS & PROCESSES - TRAINING - HANDLING & PACKAGING
	EFFECTS ON FIELD PROCESSES & FACILITIES	FIELD ASSEMBLY & CHECKOUT PROCEDURES - FIELD STERILIZATION FACILITY - REMOTE HANDLING
LONG-TERM SPACE SOAK EFFECTS	PROPELLANT CORROSIVE EFFECTS	ENGINE RESTART - PROPELLANT STORAGE TANK MATERIALS VALVES & FITTINGS
	BATTERY & PNEUMATIC SYST. LEAKAGE	SEALING METHODS & MATERIALS
	OPERATION OF MECHANISMS	BEARING & ACTUATOR DESIGN - DEVELOPMENT OF LUBRICANTS - BEARING MATERIALS
	SPACECRAFT THERMAL CONTROL	THERMAL CONTROL COATINGS - ACTIVE CONTROL SYSTEM DESIGN
	SIMULATED SPACE ENV. TESTING	THERMAL VACUUM TESTS - STIMULATION OF CONTROL SENSORS
HIGH RELIABILITY REQUIREMENTS	SYSTEM RELIABILITY	REDUNDANCY OF SYSTEMS AND COMPONENTS - ANALYSIS - APPORTIONMENT - TESTS - TRADEOFFS
	COMPONENT RELIABILITY & LIFE	TEST & SELECTION OF COMPONENTS, PARTS & MATERIALS - VENDOR SELECTION & CONTROL - STANDARDS
	MANUFACTURING & TESTING	CLEAN FACILITIES - TRAINING - QUALITY ASSURANCE PROCEDURES - QUALIFICATION TESTS
	HANDLING, SERVICING, CHECKOUT	EFFECTS OF FIELD OPERATIONS ON RELIABILITY - GROUND SUPPORT PROCEDURES & EQUIPMENT
ENTRY TO, LANDING & OPERATION ON MARS	ENTRY OF LANDER	ENTRY SENSING METHOD - PARACHUTE DEVELOPMENT & TESTS - HEAT SHIELD DEVELOPMENT & TESTS - RETROROCKET
	IMPACT SURVIVAL & ENVIRONMENT	SHOCK ABSORPTION STRUCTURE - ORIENTATION & THERMAL CONTROL DEVELOPMENT & TESTS
	EXPERIMENTS & DATA PROCESSING	DEVELOPMENT & TESTS OF DEPLOYMENT METHODS - DIRECT EARTH COMMUNICATION LINK DEVELOPMENT
OTHER PROBLEM AREAS	RADIOISOTOPE THERMOELECTRIC GEN.	DEVELOPMENT & TESTING - SIMULATION OF RADIOISOTOPE CAPSULE - HANDLING - INSTALLATION - AVAILABILITY
	DATA STORAGE & COMMUNICATION	DEVELOPMENT OF DATA RECORDER - DEVELOPMENT OF KLYSTRON

Figure 7.1-8. Development Problems Summary

7.2 PROGRAM COST AND SCHEDULE COMPARISONS

7.2.1 SUMMARY

A. PROGRAM COSTS AND SCHEDULES

The comparisons of Titan IIIC and Saturn 1B Voyager program costs and schedules are summarized in Figure 7.2-1.

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 Lander/Bus 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 IB	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

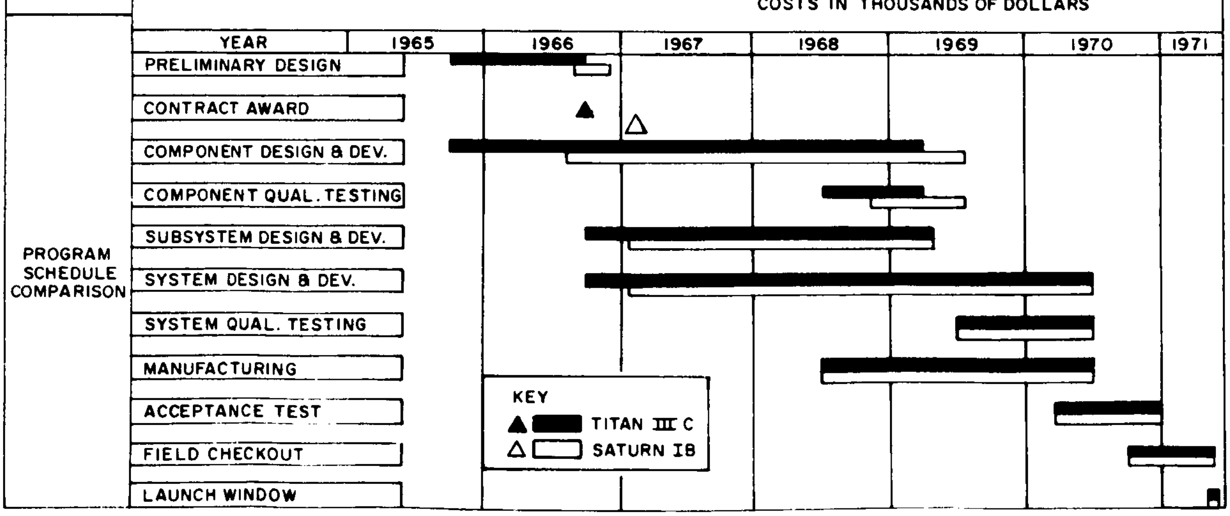


Figure 7.2-1. Summary of Program Costs and Schedule Comparisons

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.

B. 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 Section 5. of this report). They indicate that mission values attained by a Titan IIIC Voyager system, consisting of 2 Orbiter and 3 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 2 Orbiters and 4 Landers (2 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)

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 5 launches (2 Orbiters and 3 Lander/Buses) with a Saturn program of 2 launches (2 Orbiters and 4 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

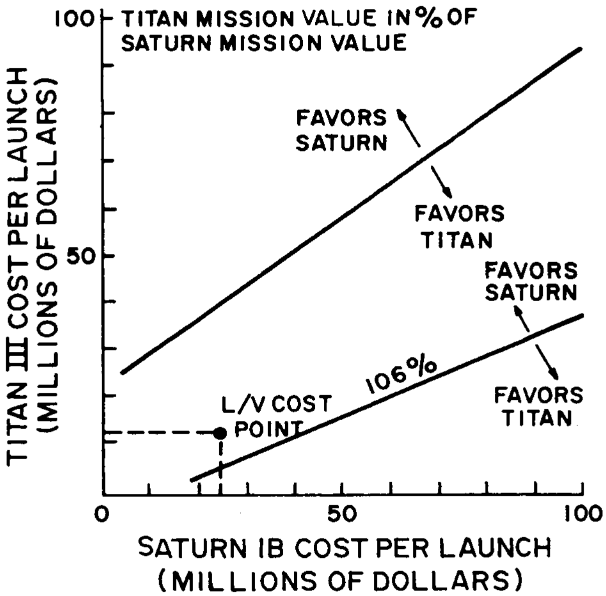
V_r = 1.8 where Titan IIIC program yields 180 percent of Saturn 1B program attainable mission value.

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 7.2.1(B), the above equation has been plotted in Figure 7.2-2, 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 a particular value line, 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 7.2-3.



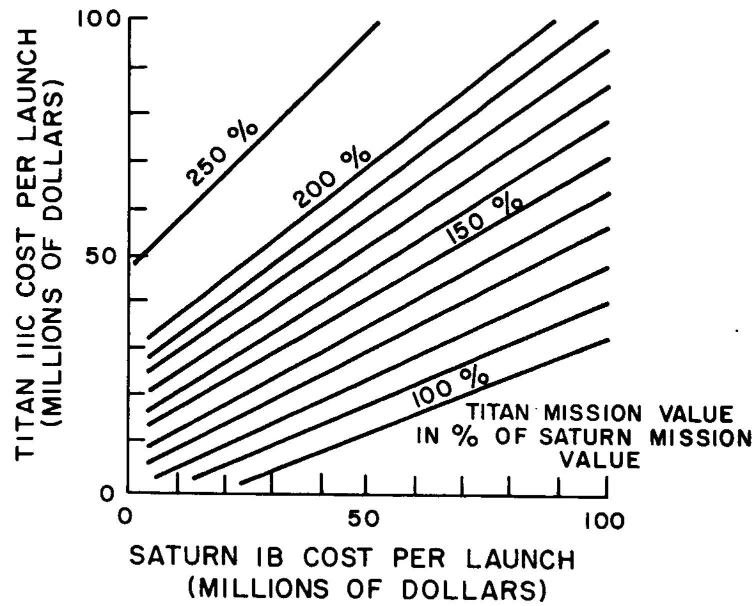


Figure 7.2-2. Comparison of 5 Titan IIIC Launches (3 L/B +2 ORB.) versus 2 Saturn 1B Launches (4L + 2 ORB)

Figure 7.2-3. Comparison of 5 Titan IIIC Launches (3 L/B +2 ORB.) versus 2 Saturn 1B Launches (4L +2 ORB)

Figures 7.2-4 and 7.2-5 present similar data for Titan IIIC programs employing four and three launches, respectively.

7.2.2 SPACECRAFT COST COMPARISONS

The cost comparisons which follow in Tables 7.2-1, 7.2-2, and 7.2-3 apply to spacecraft programs of similar attainable mission values utilizing Saturn 1B and Titan IIIC launch vehicles.

The costs shown for the Saturn 1B Voyager are those obtained during the Voyager Design Study. Those for Titan IIIC were obtained by incremental adjustment of the previous Voyager estimates based on changes to major components, subsystems and systems as dictated by the Titan IIIC Voyager design.

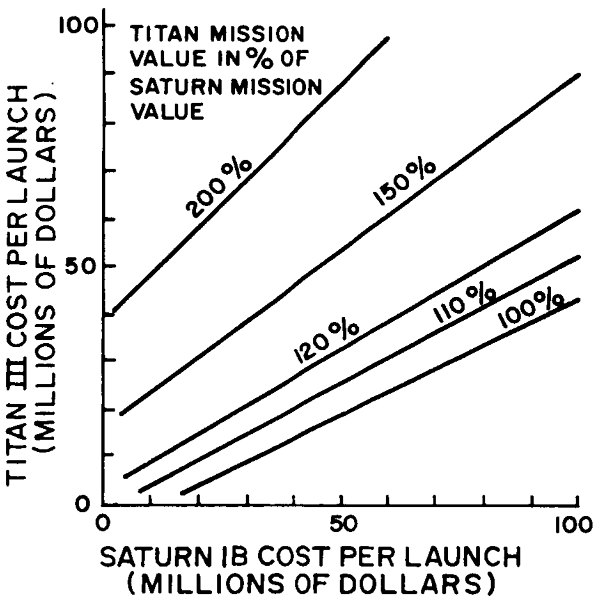


Figure 7.2-4. Comparison of 4 Titan IIIC Launches (3 L/B +1 ORB) versus 2 Saturn 1B Launches (4L +2 ORB)

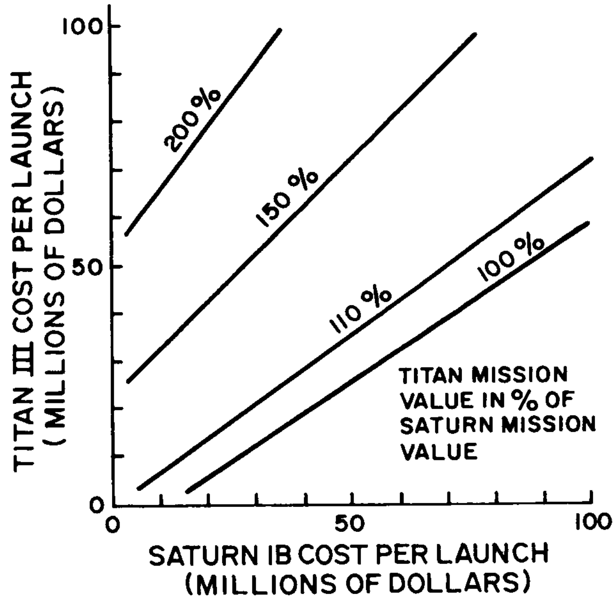


Figure 7.2-5. Comparison of 3 Titan IIIC Launches (2 L/B +1 ORB) versus 2 Saturn 1B Launches (4L +2 ORB)

Table 7.2-1 shows a comparison of overall program costs while Tables 7.2-2 and 7.2-3 compare cost estimates of corresponding Orbiter and Lander elements of the programs down to the subsystem level.

The nature of the major changes which affect the cost estimate differences are shown in Table 7.2-4.

TABLE 7.2-1. COMPARISON OF SPACECRAFT COSTS FOR EQUIVALENT PROGRAMS (COSTS IN THOUSANDS OF DOLLARS)

Cost Item	Saturn 1B Voyager	Changes Dominating Cost Difference	Titan IIIC Voyager
Voyager System			
Program Management	11,790	Addition of bus vehicles	12,270
Reliability	9,259	Addition of bus vehicles	10,800
System Test	9,407	Separate Orbiter & Lander Tests	14,110
Syst. Total	30,456		37,180
Orbiter System			
Design & Development	51,857	No relay communication link	50,299
Qual. Testing	14,234	No relay link; smaller engine	13,562
Mfg. Hardware	17,210	No relay link; no TPR	15,873
Prod. Testing	7,782	No relay link	7,665
Orb. Total	91,083		87,399
Lander System			
Design & Development	14,002	Antenna pointing req'ts; retrorocket	16,142
Qual. Testing	7,269	Antenna pointing req'ts; retrorocket	8,137
Mfg. Hardware	8,008	Antenna pointing req'ts; retrorocket	8,398
Prod. Testing	6,188	Reduced quantity	6,094
Lander Total	35,467		38,771
Bus System			
Design & Development		No	12,140
Qual. Testing		Bus	2,656
Mfg. Hardware		Previously	5,099
Prod. Testing		Required	4,218
Bus Total			24,113
GSE			
Design & Development	17,570	More vehicle types to support	19,295
Mfg. Hardware	5,385	More support equipment required	5,915
Operational I & C/O	468	More equipment & vehicles	514
GSE Total	23,423		25,724
PROGRAM TOTAL	180,429	ADDITION OF BUS TO SYSTEM	213,187

TABLE 7.2-2. COMPARISON OF ORBITER COSTS FOR EQUIVALENT PROGRAMS (COSTS IN THOUSANDS OF DOLLARS)

Cost Item	Saturn 1B Voyager	Changes Dominating Cost Difference	Titan IIIC Voyager
Orbiter System			
Design & Development	11,818	No relay link	11,570
Qual. Testing	4,289	No relay link	4,200
Mfg. Hardware	874	No relay link	855
Prod. Testing	986	No relay link	965
Syst. Total	17,967		17,590
Structure			
Design & Development	5,690	Deployable solar panels; 3 axis PHP	6,008
Qual. Testing	341	Deployable solar panels; 3 axis PHP	349
Mfg. Hardware	1,937	3 axis PHP	1,961
Prod. Testing	1,929	Deployable solar panels; 3 axis PHP	2,116
Struct. Total	9,897		10,434
Communication			
Design & Development	7,433	No relay link; no TPR	5,485
Qual. Testing	1,575	No relay link	1,335
Mfg. Hardware	5,353	No relay link; no TPR	3,560
Prod. Testing	1,460	No relay link	1,168
Comm. Total	15,821		11,448
Power Supply			
Design & Development	1,817		1,817
Qual. Testing	582		582
Mfg. Hardware	3,147	More solar cells	3,609
Prod. Testing	793	More solar cells	826
Pwr. Supp. Total	6,339		6,834
Guidance & Control			
Design & Development	17,704	3 axis PHP	18,794
Qual. Testing	2,563	3 axis PHP	2,723
Mfg. Hardware	4,490	3 axis PHP	4,655
Prod. Testing	1,915	3 axis PHP	1,980
Guid. & Cont. Total	26,672		28,152
Propulsion			
Design & Development	7,395	Smaller main engine; no trim rockets	6,625
Qual. Testing	4,884	Smaller main engine; no trim rockets	4,373
Mfg. Hardware	1,409	Smaller main engine; no trim rockets	1,233
Prod. Testing	699	Smaller main engine; no trim rockets	610
Prop. Total	14,387		12,841
ORBITER TOTAL	91,083	NO RELAY LINK; REDUCED PROP. REQ.	87,399

TABLE 7.2-3. COMPARISON OF LANDER COSTS FOR EQUIVALENT PROGRAMS (COSTS IN THOUSANDS OF DOLLARS)

Cost Item	Saturn 1B Voyager	Changes Dominating Cost Difference	Titan IIIC Voyager
Lander System			
Design & Development	3,990		3,990
Qual. Testing	4,077		4,077
Mfg. Hardware	316	Reduced quantity	284
Prod. Testing	1,831	Reduced quantity	1,575
Syst. Total	10,214		9,926
Structure			
Design & Development	4,312	Simplified design	4,275
Qual. Testing	718	Simplified design	610
Mfg. Hardware	2,585	Simplified design; reduced quan.	2,380
Prod. Testing	1,757	Simplified design; reduced quan.	1,615
Struct. Total	9,372		8,880
Communication			
Design & Development	2,936		2,936
Qual. Testing	1,685		1,685
Mfg. Hardware	3,636	Reduced quantity	3,340
Prod. Testing	2,023	Reduced quantity	1,860
Comm. Total	10,280		9,821
Power Supply			
Design & Development	917		917
Qual. Testing	107		107
Mfg. Hardware	233	Reduced quantity	214
Prod. Testing	59	Reduced quantity	55
Pwr. Supp. Total	1,316		1,293
Earth Antenna			
Design & Development	1,007	Ant. Pointing Req'ts	2,114
Qual. Testing	252	Ant. Pointing Req'ts	378
Mfg. Hardware	828	More complex design	1,240
Prod. Testing	251	More complex design	377
Ant. Total	2,338		4,109
Propulsion			
Design & Development	840	Retrorocket req'd	1,910
Qual. Testing	430	Retrorocket req'd	1,280
Mfg. Hardware	410	Retrorocket req'd	940
Prod. Testing	267	Retrorocket req'd	612
Prop. Total	1,947		4,742
LANDER TOTAL	35,467	ANT. POINTING REQ'TS; RETROROCKET	38,771

TABLE 7.2-4. MAJOR CHANGES AFFECTING COST ESTIMATE DIFFERENCES

Item	Major Titan IIIC Prog. Changes	Cost Change (Thous.)
Voyager System	Additional spacecraft type (bus); increased number of spacecraft delivered; increased number of flights; program lengthened 5 months	$ 6,724 +
Orbiter System	No relay communications link; smaller main engine; tape recorders instead of TPR; 3-axis PHP; more solar cells.	3,684 –
Lander System	More elaborate direct communications; no relay communications; larger lander; fewer landers required; retro-rocket required.	3,304 +
Bus System	No bus previously required.	24,113 +
Support Eng.	More spacecraft types to handle, service and checkout; two flight configurations instead of one; increased number of launches.	2,301 +
	Net Change in Prog. Cost	$ 32,758 +
	Saturn 1B Prog. Cost	180,429
	Titan IIIC Prog. Cost	$213,187

7.2.3 SPACECRAFT SCHEDULE COMPARISON

The schedule comparison which follows in Figure 7.2-6 compares the schedule developed for the Saturn 1B Voyager, Mars 1969, with the Titan IIIC schedule, Mars 1971, with the schedules transposed to meet a common launch window.

It will be noted that the major schedule difference is the longer time span for the Titan IIIC program with the additional time allocated to system integration and development testing. This also increases the time available for the integration and acceptance testing of qualification test systems hardware prior to qualification tests.

The Titan IIIC and Saturn 1B spacecraft schedules are considered to be equally attainable with no new critical areas apparent which would jeopardize performance on the schedules shown. However, the development of critical components and techniques during the preliminary design period is considered essential to the performance of either program on the schedule shown.

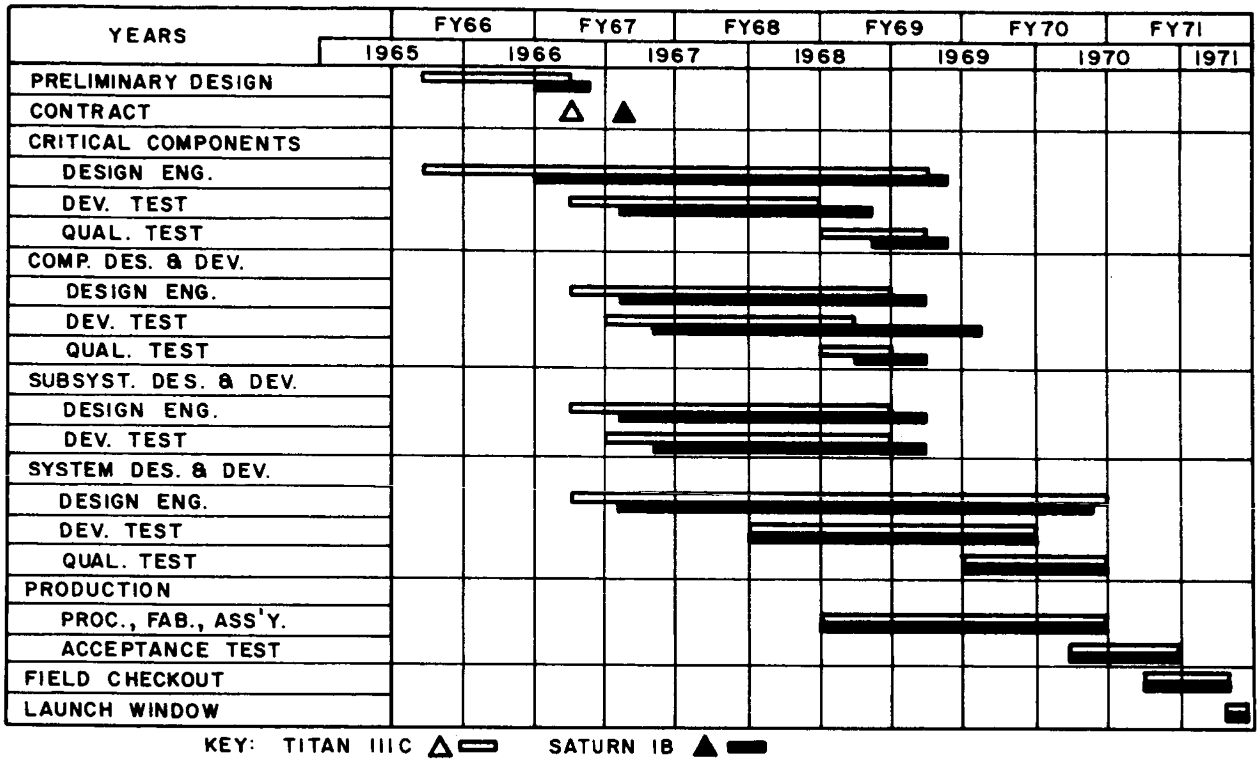


Figure 7.2-6. Program Schedule Comparison