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VOYAGER SPACECRAFT SYSTEM STUDY

(PHASE I-TITAN IIIC LAUNCH VEHICLE)

FINAL REPORT
VOLUME IIa

7 AUGUST, 1964

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1. INTRODUCTION

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

The objectives of the study were to:

1. Conduct a conceptual design of both a Bus/Lander and an orbiting spacecraft for Mars 1971.
2. Estimate their performance for Mars 1973 through 1977 and for Venus 1972.
3. Estimate the performance for a combined Orbiter/Lander system.
4. Compare the Titan IIIC and the Saturn 1B + SVI Voyager Spacecraft Systems.
5. Estimate the cost and development cycle for the Mars 1971 systems.

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

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

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

2. MISSION AND SYSTEM ANALYSIS

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

2.1 RECOMMENDED SYSTEMS

As can be seen in prior sections of this report, and as expected, the combined Orbiter/ Lander system proved to be markedly inferior to either a Bus/Lander system or an All-Orbiter spacecraft in the weight of useful scientific payload delivered and placed in operation and in quantity of data returned to earth by the prime telemetry modes. Therefore, the overall recommendation is to use separately-launched systems on the Titan IIIC launch vehicle.

Various combinations of Orbiters and Landers are compared with the Saturn 1B systems of the prior study in Sections 2.7 and 5. Attainable mission values which are used for this comparison are based on separately launched systems with payloads that are not greatly extended over the prior study and on the application of Rover concepts to lander payloads. These systems are based on a decision made during the study to utilize a 144-inch-diameter shroud to accommodate 134-inch-diameter fixed-geometry Landers of a weight that could be also delivered in the later opportunities. Later in the study an evaluation of Landers with variable geometry, i.e., movable flaps, which would permit heavier vehicles with the required W/C_DA of 15 to be carried inside the standard 120-inch-diameter Titan IIIC aerodynamic shroud was performed. These systems appear to be promising and are discussed in Section 3.2 and can affect final system selection. These systems were not costed for purposes of a comparison with Saturn 1B Voyager systems.

No such perturbation appeared in the All-Orbiter and Orbiter/Lander study areas.

System recommendations and major study results for separate and combined systems are discussed below.

2.1.1 BUS/LANDER

A review of injection energy requirements for Bus/Lander systems, with an estimated Bus weight of 450 pounds, yielded the table of Lander weights and opportunities shown in Table 2.1-1. A ballistic coefficient ($W/C_D A$) of 15 pounds/feet^2 was required for entry corridor and retardation system constraints. The drag coefficient of the selected sphere/cone configuration, maximum Lander weights and a nominal Lander-to-shroud clearance of 5 inches were used to determine required shroud diameters for various Lander weights in Table 2.1-1.

This was the basis for the 144-inch diameter shroud and the 2042-pound Lander that was studied in detail for the 1971 opportunity. This Lander could be flown for all the opportunities of interest without shroud or Lander changes.

Landers with movable flaps that would fit within the standard 120-inch-diameter shroud, requiring only changes in length of the shroud, were evaluated.

Gross payload, which includes scientific payload, power supply, thermal control, communication system, deployment hardware and vehicle electrical harnessing are compared for flapped and fixed-flare Landers in Table 2.1-2. It can be seen that flaps permit the maximum payload in 1971 with the 120-inch shroud and incur a small penalty for the 11 mb atmosphere in 1973. If it is found that Mars atmosphere is 15 mb, then the flaps used in 1971 could be removed and the 120-inch core vehicle could land the maximum payload for 1973. Flaps are attractive and further study of the reliability, design and implementation of flaps systems is recommended. It should be noted that the development costs of flapped landers should be compared with costs and overall program usefulness of the 144-inch-diameter shroud. Shroud development information was not available during the study.

Two Landers on one Bus were considered but were discarded in favor of the larger payload of the 2042-pound Lander which could easily provide rover capability and could be flown in succeeding opportunities, whereas the two Lander system was attractive only in 1971.

TABLE 2.1-1. ENTRY/LANDER WEIGHT RESTRICTIONS

Weight Restrictions

- Injection Energy Requirements
- Reasonable Trip Times
- Shroud Diameter

	Trip Time (Max.) (Days)	Max. Lander Weight (lb)	Type Traj.
1971	225	2960	I
	160	2042	
1973	195	2042	I
1975	420	2300	II
	336	2042	
1977	387	2570	II
	297	2042	

Shroud Diameter (in.)	Max. Weight (lb)
120	1380
144	2042
170	2960

The Bus functions are partially integrated with the Lander. The power supply and communication system are in the Lander only, with hard wire connection through the sterilization barrier to the Bus. See Section 2.3.5 for a discussion of the integrated versus separate Bus.

Flyby aiming point trajectories are recommended for Bus/Landers because of the lack of a requirement for more accuracy than can be provided by terminal guidance on a flyby course, and the lowest probability of the non-sterile Bus impacting the planet. See Section 2.3.4.

Bus/Lander Mission sequence is summarized in Table 2.1-3. Bus/Lander weights and payload are summarized in Table 2.1-4.

Communication modes and data rates, shown graphically in Figure 2.1-1 are discussed more fully in Section 2.4.2.

TABLE 2.1-2. EVALUATION OF LANDER FLAPS

Mars 1971

	Flaps	Lander Weight (lb)	Gross Payload (lb)	Shroud Dia. (in.)
I	No	2042	858	144
II	Yes	2042	782	120
III	Yes	2960	980	120

Mars 1973

	Flaps	Lander Weight (lb)	Gross Payload (lb)	Req. Atmos. (mb)	
				γe 20-35	γe 20-90
I	No	2042	858	11	20
II	Yes	2042	782	11	20
III	No	2042	870	15	28

Conclusion

Flaps Very Attractive From System Viewpoint

TABLE 2.1-3. BUS/LANDER MISSION SEQUENCE

Phase		
Transit	225 Days	Flyby Aiming Point
Separation	150,000 n. mi.	Bus Dead
Lander Cruise		Direct Diag. Telemetry
Entry		Telemetry Blackout
Descent		Direct Diag. and Science Telemetry
Impact	Deploy Aft Cover	
Day	Deploy Aft Cover Deploy Pan. TV Deploy Hi-Gain Antenna Orient Hi-Gain Antenna Direct Telemetry (66 TV Frames/Day)	
Night	Drill Record TV Microscope Data	
Dawn	Direct Science Telemetry	

TABLE 2.1-4. BUS/LANDER WEIGHT AND PAYLOADS

1971 Bus Lander Capability

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

Payload

Biological	Geophysical-Geological	Atmospheric
Growth	Surface Penetrability	Temperature
Metabolic Activity	Soil Moisture	Pressure
Existence of Organic Molecules	Seismic Activity	Density
Existence of Photo-Autotroph	Surface Gravity	Composition
Turbidity and PH Changes		Altitude
Microscopic Characteristics		Light Level
Organic Gases		Electron Density
Macroscopic Forms (TV)		
Surface Sounds		

Surface Roving Vehicle

Launch window for the 1971 Bus/Lander is 9 May 1971 to 8 June 1971 with a maximum trip time of 225 days.

2.1.2 ORBITER

The preliminary estimates of weights and propellant requirements for a Titan IIIC launched all orbiting system indicated that a low circular orbit could be achieved. However, payload considerations, communication data rate problems, and synchronous mapping situations led to the recommended 1,000 x 2,278 nautical mile orbit, more fully discussed in Section 2.5.

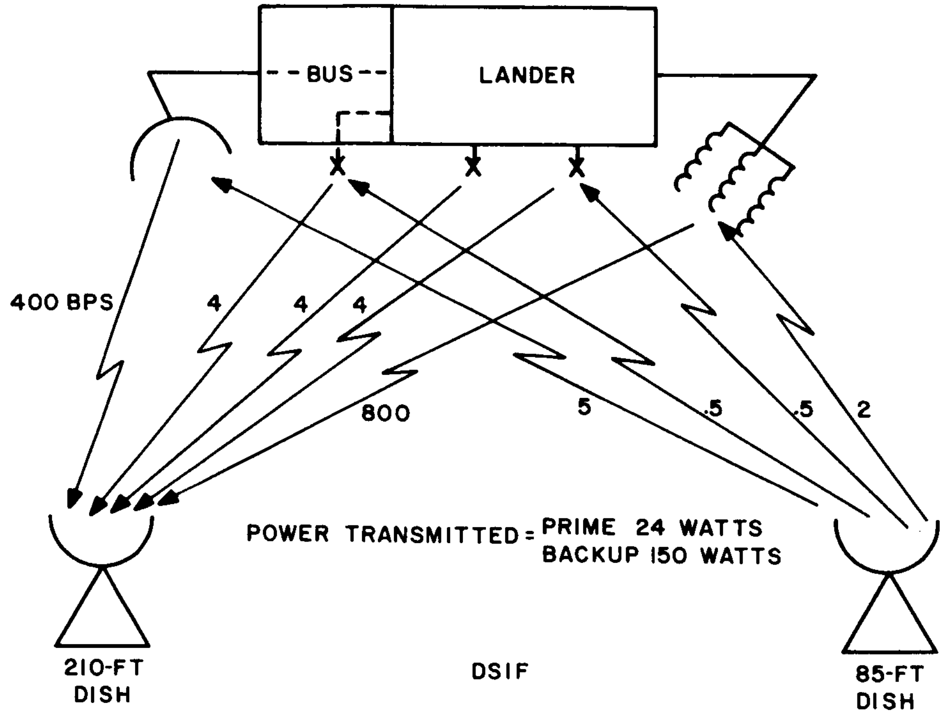

Figure 2.1-1. Bus/Lander Communication Links and Nominal Data Rates

The power supply could be an RTG in the 1971 opportunity because of predicted improvement in fuel availability. A 600-watt RTG was compared to a 600-watt solar cell power supply for the 1971 Orbiter. It was found that the weight savings of the RTG power supply alone were not significant, but that if the Orbiter configuration were changed from Sun to planet oriented, a considerable weight saving with the RTG could be realized. However, the problem of continuously orienting the high-gain antenna for the Earth communication link and reacquiring Earth view after occultation of Earth by Mars in each orbit, coupled with a lower reliability for the RTG Orbiter system, led to a decision to use solar power.

The Orbiter can be easily accommodated by the standard 120-inch diameter shroud and an appropriate extension in length.

The same Orbiter can be flown in successive opportunities by varying the orbit eccentricity to accommodate variations in aerocentric approach velocities. See Table 2.1-5.

TABLE 2.1-5. VARIATION OF ORBITER PERFORMANCE WITH OPPORTUNITY

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

Orbiter weights and payloads, mission sequence, and communication links and data rates are summarized in Tables 2.1-6 and 2.1-7, and Figure 2.1-2 respectively. Launch window for the all orbiter system is from 6 May 1971 to 5 June 1971, with a maximum trip time of 225 days.

TABLE 2.1-6. ORBITER CAPABILITY

Weight Statement

Orbiting Weight		1815
Payload	347	
Fuel (1000 x 2278 n.mi.)		1634
Injected Weight		3449 Pounds
Adapter and Shroud		151 Pounds
		3600 Pounds

Scientific Capability

Television 1 KM Stereo Map
140 M Blue
Red
Green-Yellow
3 - 20M B&W

Upper Atmosphere Composition and Density

Ionosphere Profile

Particles and Fields

UV and IR Radiation

TABLE 2.1-7. ORBITER MISSION SEQUENCE

Time	Event
0	Orbit Insertion
	Stereo Mapping (45%)
	24 High Resolution and Color TV Frames/Orbit (Random)
	Other Science
8.4 Days	
	20 High Resolution and Color TV Frames/Orbit (Aimed)
	Other Science
17 Days	Change Periapsis (100-400 n. mi.)
	3-7 M B&W TV } 16 Frames/Orbit
	45 M Color TV } 16 Frames/Orbit
	Other Science
90 Days	

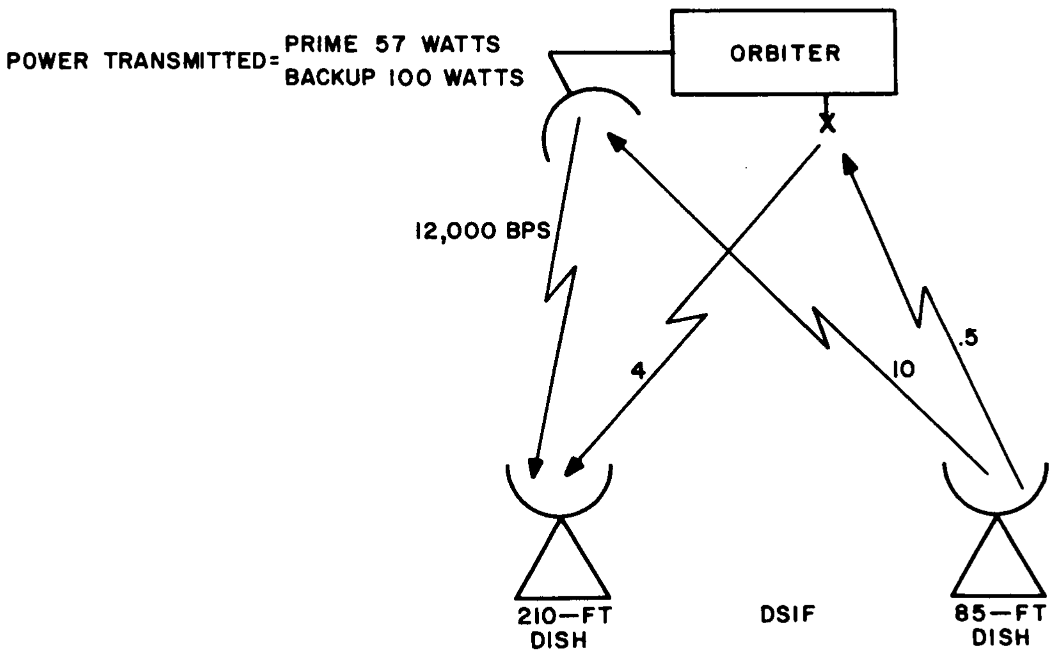

Figure 2.1-2. Orbiter Communication Links and Data Rates

2.1.3 ORBITER/LANDER

The weight limited combination system can carry a much smaller payload in either module than the weights delivered by the separately launched systems. The payload of the Orbiter was established first by eliminating the nadir vidicon and the high resolution image orthicon TV camera and optics in order to reduce payload weight in large increments. Other instruments were removed to reduce the payload to 123 pounds, compared to 215 pounds for the Orbiter in the prior study.

The Orbiter and propellant weight was then estimated and all remaining weight that could be injected, less midcourse correction fuel allowance, was assigned to the Lander. Lander payload allowance was expended in order of descending instrument priority with the exception of TV microscope and subsurface sampler which could not be accommodated. The orbit was planned to be the same 1000 x 19,000 n.mi., selected in the prior study, to maximize Lander weight.

Combination Orbiter/Lander systems can be flown in opportunities after 1971, but, as shown in Table 2.1-8, orbiter weights deteriorate to unattractive levels.

A relay link is incorporated in the combination in order to maximize data returns from the Lander and to acquire entry and atmospheric data before impact.

Weights for the combination system are compared with separate systems and with the prior study in Tables 2.1-8 and 2.1-9.

This system is not recommended because of maximum development cost, minimum data returned and its applicability to a single opportunity in 1971.

TABLE 2.1-8. ORBITER/LANDER WEIGHTS FOR OPPORTUNITY YEARS

	1971	1973	1975	1977
Orbiter, lb	1440	1970	1000	1250
Propellant, lb	684	404	624	474
Lander, lb	1284	1284	1284	1284
Mid-Course Fuel and ΔW Shroud, lb	192	192	192	192
Injected Wt, lb	3600	2850	3100	3200

TABLE 2.1-9. SYSTEM PERFORMANCE SUMMARY

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

2.2 SCIENTIFIC MISSION AND PAYLOADS

2.2.1 GENERAL

The mission values and scientific priorities recommended in the previous GE Voyager report were utilized without change for the Titan IIIC, except for the Titan IIIC combined Orbiter/Lander mission. It was assumed that the 1971 mission would either be the first substantial entry capsule or that any atmospheric information from a possible small Lander in the 1969 opportunity would be corroborated by another determination, thereby providing a mission value equivalent to the original.

The same primary objectives, biological, atmospheric, planetological, geophysical, environmental, and support of future manned missions were held for the Titan IIIC missions as for the previous study. Scientific payload capabilities of candidate Titan IIIC spacecraft configurations were compared with the previous instrument complements, and as much of the original list as possible, with the original priorities, was to be included in these missions. No additional experiments were analyzed or proposed during this study.

2.2.2 BUS/LANDER (SCIENTIFIC MISSION AND PAYLOAD)

The payloads of the prior study were developed for a series of missions in successive opportunities starting with 1969. The Lander instruments from these complements were combined in a general priority list. See Table 2.2-1. Duplications were eliminated and priorities were decided for instruments originally planned to go in missions later than 1969. The size and payload capability of the single Lander of the prime configuration for the Bus/Lander system can easily accommodate the entire payload of the heaviest dual Landers in the prior Voyager study. This capability could be used for additional payload, or more favorably, from the point of view of maximizing mission value, by incorporating limited roving ability in the payload. This was done and the identified payload complement for the 1971 Bus/Lander with the single 134-inch diameter Lander with a W/C_DA of 15 are listed in Table 2.2-2.

The landing sites, Syrtis Major and Pandorae Fretum, selected in the prior study, were retained for these Landers.

TABLE 2.2-1. TITAN IIIC LANDER SCIENTIFIC PAYLOAD PRIORITY LIST

Priority	Name	Inst. No.	Weight (Pounds)	Accum. Weight (Pounds)	Power (Watts)	Year Orig. Planned to Fly
1	Temperature	I-24	0.3	0.3	0.07	1969
2	Sounds	I-34	0.5	0.8	1	1969
3	Pressure	I-17	0.3	1.1	0.10	1969
4	Density	I-20	1.5	2.6	2	1969
5	Multiple Chamber	I-54	4.0	6.6	2	1969
6	Surface Penetrability/Hardness	I-25	4.5	11.1	0.1	1969
7	Photoautotroph Detector	I-62	3.0	14.1	1	1969
8	Light Intensity (Sun Sensor)	I-84	0.5	14.6	0.1	1969
9	Composition, H_2O	I-44	1.5	16.1	1	
10	Composition, O_2	I-45	1.5	17.6	1	
11	Turbidity and PH Growth Detector	I-53	4.0	21.6	1	1969
12	Wind Speed and Direction	I-67	2.0	23.6	0.5	1969
13	Gas Chromatograph	I-8	7.0	30.6	4.5	1969
14	Composition, N_2	I-48	1.0	31.6	1	
15	Composition, CO_2	I-49	1.0	32.6	1	
16	Soil Moisture	I-70	2.0	34.6	25	1969
17	TV Camera, Panorama		20.0*	54.6	20	1969
18	Radioisotope Growth Detector	I-19	6.0	60.6	3	1969
19	Composition, O_3	I-46	1.5	62.1	1	
20	Composition, A	I-47	1.5	63.6	1	
21	Precipitation	I-36	1.0	64.6	1	1969
22	Electron Density (Langmuir Probe)	I-39	3.0	67.6	3	1969
23	Surface Gravity	I-72	3.0	70.6	3	1969

*Includes 10 Pounds TV Deployment

TABLE 2.2-1. TITAN IIIC LANDER SCIENTIFIC PAYLOAD PRIORITY LIST (Continued)

Priority	Name	Inst. No.	Weight (Pounds)	Accum. Weight (Pounds)	Power (Watts)	Year Orig. Planned to Fly
24	Radar Altimeter	I-5	15.0	85.6	25	1969
25	TV Microscope and Subsurface Group	I-71	75.0	160.6	200	1969
26	Seismic Activity	I-21	8.0	168.6	1	1969
27	Mass Spectrometer	I-43	6.0	174.6	6	1969
28	UV Multichannel Radiometer	I-78	1.5	176.1	1.5	1971
	1215 Å (Lyman a) 1026 Å (Lyman B) 972 Å (Lyman y) 584 Å (HeI) 304 Å (HeII)		Sharp Filters			
	1445 Å - 1500 Å 2500 Å - 3000 Å		Band Filters			
29	8446 Å Radiometer	I-40	0.3	176.4	0.2	1971
30	Polarimeter (Skylight Analyzer)	I-68	4.5	180.9	4.5	1971
31	X-Ray Diffracto-meter	I-32	10	190.9	15	1971
32	Alpha - Particle Scattering	I-57	7	197.9	2	1971
33	Thermal Diffusivity of Ground	I-64	1	198.9	25	1971
34	Electrical Conduc-tivity of Ground	I-65	1	199.9	1	1971
35	(Insolation) Pyrheliometer	I-16	1	200.9	1	1971
36	Surface Radio-activity	I-13	8	208.9	2	1971
37	Meteor Trails	I-15	2.5	211.4	2.5	1971
38	Ionospheric Pro-file: Bottomside Sounder	I-87	50	261.4	25	1971
39	Sferics	I-82	3	264.4	2	1971
40	Eclipse by Phobos	-	1	265.4	0.3	1971

TABLE 2.2-1. TITAN IIIC LANDER SCIENTIFIC PAYLOAD PRIORITY LIST (Continued)

Priority	Name	Inst. No.	Weight (Pounds)	Accum. Weight (Pounds)	Power (Watts)	Year Orig. Planned to Fly
41	Insect Attractor	I-69	.1	265.5	1	1971
42	Pulse Light	I-75	1	266.5	0.1	1971
43	UV Solar Spectrum	I-81	22	288.5	12	1973
44	Seismic Properties					
	Natural	I-91	34	322.5	4	1973
	Induced	I-90	90	412.5	5	1973
45	Aerosol Profile	I-99	3	415.5	2	1975
46	Solar 3-Channel Radiometer	I-29	1.5	417.0	1	1975
47	Laser-Induced Gaseous Emission Spectra	I-100	50	467.0	2	1975
48	Laser Atmospheric Backscatter Probe	I-101	20	487.0	15**	1975

**Intermittent Operation

TABLE 2.2-2. SCIENTIFIC PAYLOAD FOR 2042-POUND LANDER

Priority	Name	Inst. No.	Weight (Pounds)	Accum. Weight (Pounds)	Power (Watts)
1.	Temperature	I-24	0.3	0.3	0.07
2.	Sounds	I-34	0.5	0.8	1
3.	Pressure	I-17	0.3	1.1	0.10
4.	Density	I-20	1.5	2.6	2
5.	Multiple Chamber	I-54	4.0	6.6	2
6.	Surface Penetration Hardness	I-25	4.5	11.1	0.1
7.	Photoautotroph	I-62	3.0	14.	1
8.	Light Intensity (Sun Sensor)	I-84	0.5	14.6	0.1
9.	Composition, H_2O	I-44	1.5	16.1	1
10.	Composition, O_2	I-45	1.5	17.6	1

TABLE 2.2-2. SCIENTIFIC PAYLOAD FOR 2042-POUND LANDER (Continued)

Priority	Name	Inst. No.	Weight (Pounds)	Accum. Weight (Pounds)	Power (Watts)
11.	Turbidity and PH Growth Detector	I-53	4.0	21.6	1
12.	Wind Speed and Direction	I-67	2.0	23.6	0.5
13.	Gas Chromatograph	I-8	7.0	30.6	4.5
14.	Composition, N_2	I-48	1.0	31.6	1
15.	Composition, CO_2	I-49	1.0	32.6	1
16.	Soil Moisture	I-70	2.0	34.6	25
17.	TV Camera, Panorama TV		20.0*	34.6	20
18.	Radioisotope Growth Detector	I-19	6.0	60.6	3
19.	Composition, O_3	I-46	1.5	62.1	1
20.	Composition, A	I-47	1.5	63.6	1
21.	Precipitation	I-36	1.0	64.6	1
22.	Electron Density (Langmuir Probe)	I-39	3.0	67.6	3
23.	Surface Gravity	I-72	3.0	70.6	3
24.	Radar Altimeter	I-5	15.0	85.6	25
25.	TV Microscope and Subsurface Group	I-71	75.0	160.6	200
26.	Seismic Activity	I-21	8.0	168.6	1

* Includes 10 Pounds TV Deployment

2.2.3 ORBITER

The payload capability of the Titan IIIC All-Orbiter mission is substantially greater than that of the Orbiter designed for the Saturn 1B. In order to take advantage of this increased scientific potential, the scientific payload complements from the previous Voyager study for the years 1969, 1971 and 1973 were combined into one payload for the all orbiting mission.

The prime task for the Orbiter is still the acquisition of a survey map of the Martian surface. The necessity for the nadir camera, which was required with the highly eccentric 1000 x 19,000 nautical miles orbit selected in the previous Voyager study, was eliminated by the 1000 x 2278 nautical mile orbit that can be easily achieved by the Titan IIIC mission All-Orbiter. The two stereo vidicon television cameras, mounted at the stereo slant angle of 19.79 degrees from the vertical, will give adequate coverage at the maximum altitude at which they will be employed. Orbiter payload priorities are listed in Table 2.2-3.

The scientific payload that could be carried in the all Orbiter mission in 1971 on the Titan IIIC launch vehicles is 347 pounds. A conclusion was drawn that the mission value would be greater with very high resolution television pictures of a small portion of the planet than by using the payload allowance for a large number of lower priority instruments. Consequently the "retro rocket and high resolution package" weighing an arbitrary 146 pounds was incorporated in the instrument complement, for the purpose of lowering the periapsis altitude after the initial map is acquired and providing an additional telephoto lens for the 20 meter image orthicon TV camera. It is estimated that resolutions of 3 to 7 meters could be achieved at the periapsis of approximately 330 nautical miles. Sterilization of the orbiter would be required for this kind of mission but the effects on propellant specific impulse, and on the image orthicon television cameras, etc. were not considered here. The mass spectrometer and electron probe were added to enable analysis of the upper atmosphere if it extended to the lowered periapsis altitude. See Table No. 2.2-4.

TABLE 2.2-3. ALL-ORBITER PAYLOAD PRIORITY LIST

No.	Name	Inst. No.	Weight (Pounds)	Accum. Weight (Pounds)	Power (Watts)	Accum. Watts	Year Orig. Planned to Fly
1.	Magnetometer	I-23	5	5	5	5	1969,1971,1973
2.	IR Multichannel Radiometer Flux	I-2	3	8	3	8	1969,1971
3.	Solar Multichannel Radiometer	I-79	3	11	3	11	1969,1971
4.	Television 4 IO, 2 Vidicon		115	126	(140)	(151)	1969,1971
5.	Charged Particle Flux: Geiger Tubes and Ion Chamber	I-12	55	132	1	12	1969

TABLE 2.2-3. ALL-ORBITER PAYLOAD PRIORITY LIST (Continued)

No.	Name	Inst. No.	Weight (Pounds)	Accum. Weight (Pounds)	Power (Watts)	Accum. Watts	Year Orig. Planned to Fly
6.	Far UV Radiometer	I-96	6	138	3	15	
7.	Micrometeoroid Flux	I-55	8	144	1	16	1969
8.	Bistatic Radar (Ionospheric Profile)	I-85	13	159	2	18	1969
9.	Polarimeter-Skylight Analyzer	I-68	4.5	163	4.5	23	1969
10.	IR Spectrometer	I-1	29	192	7	30	1969
11.	Sferics	I-82	3	195	2	32	1969
12.	X-Ray Flux for Sun	--	3	198	1	33	1969, 1973
13.	Electron Spectra and Direction	I-10	2	200	1	34	1971, 1973
14.	Proton Spectra and Direction	I-11	3	203	1	35	1971, 1973
15.	Cosmic Dust	I-37	2.5	206	.2	35	1971
16.	Radar Altimeter	I-5	15	221	25	60	1971
17.	UV Multi-Channel Radiometer	I-78	1.5	222	1.5	62	1971
18.	UV Solar Spectrometer	I-81	22	244	12	74	1971
19.	Faraday Cup	--	20	264	4	78	1971, 1973
20.	γ-Ray Spectrometer	--	14	278	5	83	1973
21.	Payload Computer		20	298	20	103 +(140 for TV) = 243 w.	--

TABLE 2.2-4. ALL-ORBITER PAYLOAD

No.	Name	Inst. No.	Accum. Weight (Pounds)	Accum. Weight (Pounds)	Power (Watts)	Accum. Watts	Year Orig. Planned to Fly
1.	Magnetometer	I-23	5	5	5	5	1969,1971,1973
2.	IR Multichannel Radiometer Flux	I-2	3	8	3	8	1969,1971
3.	Solar Multichannel Radiometer	I-79	3	11	3	11	1969,1971
4.	Television 4 IO 2 Vidicon		115	126	(140)	(151)	1969,1971
5.	Charged Particle Flux: Geiger Tubes and Ion Chamber	I-12	55	132	1	12	1969
6.	Far UV Radiometer	I-96	6	138	3	15	
7.	Micrometeoroid Flux	I-55	8	144	1	16	1969
8.	BS Radar (Ionospheric Profile)	I-85	13	159	2	18	1969
9.	Polarimeter-Skylight Analyzer	I-68	4.5	163	4.5	23	1969
10.	IR Spectrometer	I-1	29	192	7	30	1969
11.	Retro Rocket and Hi Resolution Package		146	338			
12.	Mass Spectrometer	I-43	6	344	6	36	1973
13.	Electron Probe (Langmuir Probe)	I-39	3	347	3	39 +(140 for TV) = 179 watts	1973

2.2.4 COMBINED ORBITER/LANDER

The instrument priorities for this combined mission, which has less weight capacity in each module of the system than either the All-Orbiter or Bus/Lander with a single Lander, were modified slightly. Since the major emphasis of the study was to be placed on the Orbiter and Bus/Lander configurations, it was decided that a simple and logical method of balancing scientific payload, and consequently overall weights of the Orbiter and the Lander, was to establish the Orbiter instrument complement on the basis of two high priority items as follows:

1. A stereoscopic map combined with medium resolution television coverage of a portion of the mapped area. A number of color pictures should be included in the television coverage.
2. Measurements from a group of planet-scanning instruments for providing critical environmental data.

It was already decided that the orbit would be set at the same practical maximum eccentricity (1000 x 19,000 nautical miles) that was used in the previous Voyager study for these combined Orbiter/Lander missions, to maximize the weight that could be allowed for Landers. See Table 2.2-5.

An ordered list of Lander instruments was prepared. In proceeding with the present study, it was decided that when the Lander weight allowance was determined, the Lander design group would proceed as far down this list as was possible within either the weight or volume restrictions of this small entry vehicle, thereby using the payload capability of the Lander for the most critical measurements.

The payload in the Lander of the Orbiter/Lander combination includes the instruments on the large Lander list down to priority number 26, with the exception of the TV microscope and subsurface group (drill, sample handler and pulvizer). The weight of both the appropriate power supply and the TV microscope group could not be accommodated, so that direct communication data rate was cut in half for this small Lander, and the TV microscope was omitted, allowing room in the payload for the seismograph. See Table 2.2-6.

TABLE 2.2-5. SCIENTIFIC PAYLOAD FOR ORBITER (ORBITER/LANDER)

Priority	Name	Inst. No.	Weight (Pounds)	Accum. Weight (Pounds)	Power (Watts)	Year Orig. Planned to Fly
1	2 Vidicon Cameras	TV	83.0	83.0	25.0	1969
2	3 IO Cameras	TV				
3	IR Flux	I-2	3.0	86.0	3.0	1969
4	Visible Radiometer	I-79	3.0	89.0	3.0	1969
5	Magnetometer	I-23	5.0	94.0	5.0	1969
6	Far UV Radiometer	I-96	3.0	97.0	3.0	--
7	Micrometeoroid Flux	I-55	3.0	100.0	0.5	1969
8	Charged Particle Flux	I-12	5.5	105.5	1.0	1969
9	Polarimeter	I-95	4.5	110.0	4.5	1969
10	Bistatic Radar	I-85	13.0	123.0	2.0	1969

TABLE 2.2-6. SCIENTIFIC PAYLOAD FOR LANDER (LANDER/ORBITER COMBINATION)

Priority	Name	Inst. No.	Weight (Pounds)	Accum. Weight (Pounds)	Power (Watts)	Year Orig. Planned to Fly
1.	Temperature	I-24	0.3	0.3	0.07	1969
2.	Sounds	I-34	0.5	0.8	1	1969
3.	Pressure	I-17	0.3	1.1	0.10	1969
4.	Density	I-20	1.5	2.6	2	1969
5.	Multiple Chamber Growth Detector	I-54	4.0	6.6	2	1969
6.	Surface Penetrability/ Hardness	I-25	4.5	11.1	0.1	1969
7.	Photoautotroph Detector	I-62	3.0	14.1	1	1969
8.	Light Intensity (Sun Sensor)	I-84	0.5	14.6	0.1	1969
9.	Composition, H_2O	I-44	1.5	16.1	1	--
10.	Composition, O_2	I-45	1.5	17.6	1	--
11.	Turbidity and pH Growth Detector	I-53	4.0	21.6	1	1969

TABLE 2.2-6. SCIENTIFIC PAYLOAD FOR LANDER (LANDER/ORBITER COMBINATION) (Cont'd)

Priority	Name	Inst. No.	Weight (Pounds)	Accum. Weight (Pounds)	Power (Watts)	Year Orig. Planned to Fly
12.	Wind Speed and Direction	I-67	2.0	23.6	0.5	1969
13.	Gas Chromatograph	I-8	7.0	30.6	4.5	1969
14.	Composition, N_2	I-48	1.0	31.6	1	--
15.	Composition, CO_2	I-49	1.0	32.6	1	--
16.	Soil Moisture	I-70	2.0	34.6	25	1969
17.	TV Camera, Panorama	-	20.0*	54.6	20	1969
18.	Radioisotope Growth Detector	I-19	6.0	60.6	3	1969
19.	Composition, O_3	I-46	1.5	62.1	1	--
20.	Composition, A	I-47	1.5	63.6	1	--
21.	Precipitation	I-36	1.0	64.6	1	1969
22.	Electron Density (Langmuir Probe)	I-39	3.0	67.6	3	1969
23.	Surface Gravity	I-72	3.0	70.6	3	1969
24.	Radar Altimeter	I-5	15.0	85.6	25	1969
25.	Seismic Activity	I-21	8.0	93.6	1	1969

* Includes 10 Pounds TV Deployment

2.3 SYSTEM CONSTRAINTS AND REQUIREMENTS

The capability of the Titan IIIC launch vehicle, including the standard Titan IIIC shroud, is given by Figure 2.3-1. It was determined during the study that the Titan IIIC shroud seriously restricted the spacecraft design and that a new shroud/adapter design would be necessary.

The choice of the transit trajectory for each of the spacecraft systems followed the method and analyses in the Voyager Design Study except for minor modifications to allow for All-Orbiter and All-Lander systems. The resulting trajectory characteristics are shown in Table 2.3-1; the two trajectories shown for the 1975 and 1977 All-Lander Mars missions correspond to a minimum energy trajectory and a reduced trip-time trajectory. The shorter trip-time trajectories could not be used for an All-Orbiter mission.

The guidance system is essentially the same as in the previous Voyager study. Approach guidance is required and obtained by viewing the planet against the star background with a TV camera and transmitting the picture to earth for processing. With approach guidance, a 0.99 probability of meeting the required entry angle corridor of

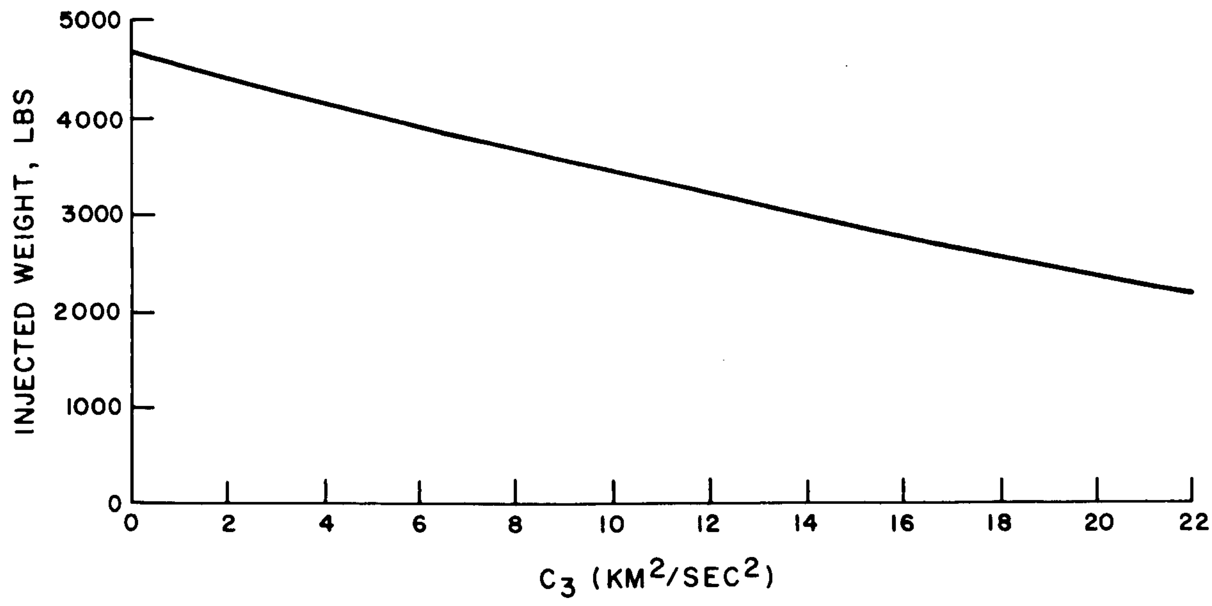

Figure 2.3-1. Titan IIIC Performance

TABLE 2.3-1. ALL-LANDER, ALL-ORBITER AND ORBITER/LANDER TRAJECTORY CHARACTERISTICS

	MARS 1971	MARS 1973	MARS 1975		MARS 1977		VENUS 1972
ALL-LANDER							
C_3, $\frac{Km^2}{Sec^2}$	8.8	16.6	14.3	16.6	11.8	16.6	9.1
Maximum Trip Time	225	195	420	336	387	297	177
Launch Window	9 May 8 June	13 July 12 Aug	5 Sept 5 Oct	24 Aug 23 Sept	26 Sept 26 Oct	14 Sept 14 Oct	23 May 22 April
Maximum Entry Velocity (At 10^6 Ft), Ft/Sec	19,950	21,300	20,500	18,600	19,950	19,300	38,200
Trajectory Type	I	I	II		II		II
ALL-ORBITER & ORBITER/LANDER							
C_3 (Maximum), $\frac{Km^2}{Sec^2}$	8.8	15.6	13.2		12.58		8
Maximum Trip Time	225	202	385		332		186
Launch Window	6 May 5 June	20 July 19 Aug	23 Aug 22 Sept		22 Sept 22 Oct		17 Mar 16 April
Maximum Hyperbolic Excess Velocity at Arrival, Ft/Sec	10,000	10,000	9,940		8,340		17,600
Trajectory Type	I	I	II		II		II

20° to 35° is assured. With the elimination of a synchronized Orbiter, line-of-sight between the Earth and the Lander must be maintained during Lander entry for transmittal of entry data.

Three possible planet approach trajectories were considered as follows: 1) flyby trajectory with the Bus/Lander always on a miss trajectory with a velocity impulse applied to the Lander after separation; 2) impact trajectory with the Bus/Lander always on an impact trajectory with a velocity impulse applied to the Bus after Lander separation; and 3) flyby/impact trajectory with the Bus/Lander on a flyby trajectory until the approach correction maneuver and on an impact trajectory thereafter with a velocity impulse applied to the Bus after Lander separation. Since error analyses showed the capability of meeting the entry corridor and landing-site dispersion requirements with a flyby trajectory, and reliability analyses showed a requirement for propulsion and communication redundancy, the flyby trajectory was selected as a basis for system design.

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

2.3.1 LAUNCH VEHICLE PERFORMANCE

The injected weight capability of the Titan IIIC launch vehicle that was utilized in this study is shown in Figure 2.3-1 as a function of C_3 (hyperbolic excess velocity squared). This injected weight capability assumes that the standard shroud (591 pounds) given in Figure 2.3-2 is utilized and no adapter is required. If a different shroud and/or adapter is used, the injected weight capability must be changed.

While it is desirable to attempt to utilize the standard shroud for the Titan IIIC Voyager Spacecraft, it was found early in the study that such a restriction would severely compromise the spacecraft design.

2.3.2 TRAJECTORY ANALYSIS

A. ORBIT INSERTION VELOCITY REQUIREMENTS

Since the Voyager Spacecraft will not approach the planet at the velocity required for a planetary orbit, a velocity change must be made. The velocity correction required

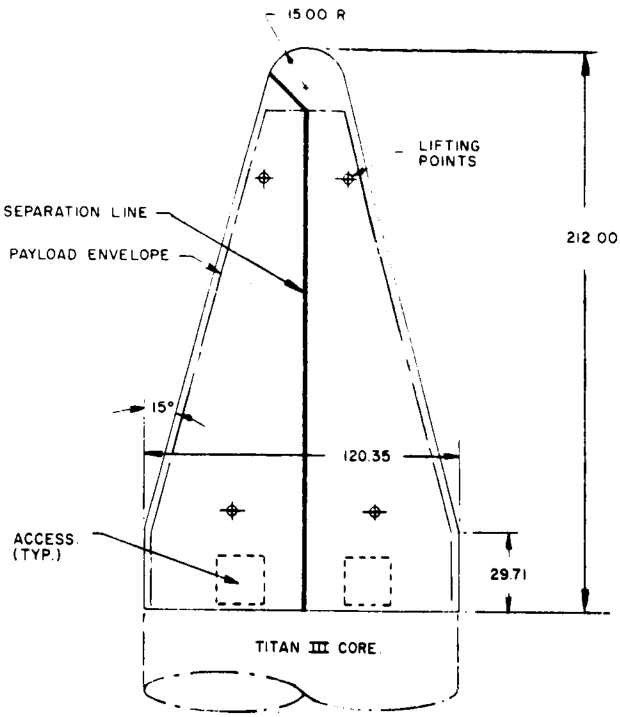

Figure 2.3-2. Standard Shroud

is a function of the hyperbolic excess velocity of the spacecraft, the particular planet in question, and the final planetary orbit desired.

One factor that will place a restriction on the planetary orbit to be utilized is the question whether it is possible to sterilize the Orbiter. For the purposes of this study NASA has specified that if the minimum altitude for the Mars circular orbit is approximately 1000 nautical miles or more, the Orbiter will not require sterilization (for a highly elliptical orbit the perifocus can be as low as 800 nautical miles before sterilization of the orbiter would be required). Therefore, the minimum altitude for all Mars orbits was set at 1000 nautical miles, irrespective of whether it was circular or elliptical. This conservatism was employed because of guidance uncertainties which would necessitate biasing the aiming point.

The velocity required for orbit insertion is equal to the difference between the approach velocity and the perifocal velocity of the particular orbit. The approach velocity (V_a) is determined from the relationship;

$$V_a = \sqrt{V_h^{\,2} + V_e^{\,2}}$$

where V_h = hyperbolic excess velocity

V_e = planetary escape velocity at the altitude for orbit insertion,

while the perifocal velocity (V_p) is given by

$$V_p = \sqrt{g_s R^2 \left(\frac{2}{h_p + R} - \frac{1}{\frac{h_p + h_a}{2} + R} \right)}$$

where g_s = 12.56 ft/sec^2 for Mars

g_s = 28.3 ft/sec^2 for Venus

R = 1830 nautical miles for Mars

R = 3340 nautical miles for Venus

h_p = perifocal altitude

h_a = apifocal altitude

The resulting velocity curves required as a function of hyperbolic excess velocity and planetary orbit desired are given in Figures 2.3-3 and 2.3-4 for Mars and Venus respectively. It is to be noted that for a given hyperbolic excess velocity and desired

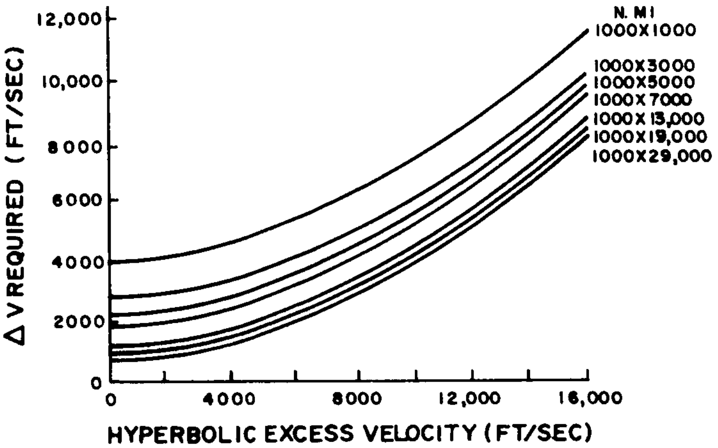

Figure 2.3-3. Martian Orbit Insertion Velocity Requirements

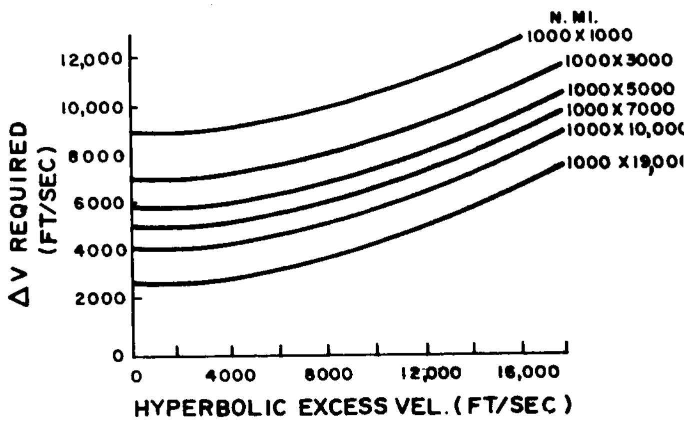

Figure 2.3-4. Venusian Orbit Insertion Velocity Requirements

orbit, the insertion requirement for Venus is significantly higher than for Mars. In addition, the hyperbolic excess velocity at arrival is generally (but not always) higher for Venus than Mars.

B. INTERPLANETARY TRAJECTORY ANALYSIS

The choice of the transit trajectory for each of the spacecraft systems and the various opportunities was made for the most part on the basis of the analysis in the Voyager Design Study Final Report. However, since an All-Orbiter and an All-Lander system was considered for this study some minor changes in the location of the launch window (assumed to be 30 days) were necessary in order to maximize the injected or orbiting weight.

The resulting trajectory characteristics are given in Table 2.3-1. For the All-Lander mission in 1975 and 1977, two levels of energy were considered: One representing a minimum energy trajectory and another an energy equal to the level required in 1973. This latter one has the advantage of reducing the trip time. For the All-Orbiter mission this is not possible.

It is to be noted in Table 2.3-1 that the entry velocity changes with opportunity, reaching a maximum of 21,300 feet per second in 1973. In order to design entry vehicles that could perform during all opportunities with a minimum of changes a conservative design entry velocity of 21,500 feet per second was utilized for all entry vehicles.

2.3.3 GUIDANCE ANALYSIS

A. OBJECTIVE

The approach to the transit and orbit injection phases of the guidance studies was to determine the points of difference that would be found between Titan boosted and Saturn 1B spacecraft.

In the case of the Landers, however, the previous Voyager Design Study was based almost entirely on a Schillings atmosphere. Only brief consideration was given to the case of an atmosphere in the 10-40 millibar range. Accordingly, the Lander studies cover the overall Lander guidance problem and the results are directly applicable to the Saturn 1B studies in all respects except for the small differences inherent in the trajectories of the two years.

The primary effect of the thin atmosphere is to result in a specified narrow entry angle corridor so as to give the Landers the maximum payload weight capability.

The results of the Lander guidance studies for the most part supersede the results of the previous Voyager study.

B. GROUND RULES

The studies were carried out on the basis of the following ground rules:

The guidance system remains essentially the same as in the previous Voyager study. By definition, any needed Approach Guidance information will be obtained by viewing the planet against the star background with a TV camera and transmitting the picture to Earth without processing. Alternate sensors or means for reducing the amount of transmitted data were omitted from the scope of this study.

Maintaining line-of-sight contact from the Orbiter (or Bus) to the Lander until the Lander reaches the surface is no longer required. Line-of-sight contact from the Earth to the landing site at Lander touchdown, on the other hand, is important.

The thin atmospheres under consideration impose an entry angle corridor from 20 degrees to 35 degrees.

The mission can be optimized to favor a single Lander. Accordingly, the approach trajectory plane can be chosen to permit an in-plane landing.

Nominal values were chosen for the following parameters for use in other subsystem studies and to provide a basis for sizing of tanks, vehicle configurations, etc. These represent early estimates rather than tradeoffs made from the completed studies.

Nominal ΔV requirements are:

Flyby Trajectory	
Midcourse plus approach corrections	100 ft/sec
Separation	300 ft/sec
Impact Trajectory	
Pre-separation	250 ft/sec
Post-separation	500 ft/sec

C. RESULTS OF STUDY

As shown in Figure 2.3-5 there is little freedom of choice when landing at a selected spot within a restricted entry corridor. The locus of all points having a given entry angle is a circle centered on the line of sight from the center of the planet to the probe at the separation point. A landing at a specific point occurs where the latitude of that point intersects the entry corridor. The landing then must be timed to coincide with the arrival of the desired spot at the entry corridor.

For a given opportunity the orientation of the approach asymptote relative to the Mars-Earth line is fixed.

As indicated in Figure 2.3-5(a), an arbitrary criterion was chosen in which the Earth is considered visible if it appears a minimum of 15 degrees above the Mars horizon. This excludes all points inside the shaded circle around the horizon. For a sunny side approach and a daylight landing, the requirement that the Earth must be a minimum of 15 degrees above the horizon results in the trip-time constraint shown in Figure 2.3-5(b). By cross-plotting, the corresponding constraint on ζ_p is derived as shown in Figure 2.3-6. In both cases these were determined in terms of an entry angle of 20 degrees

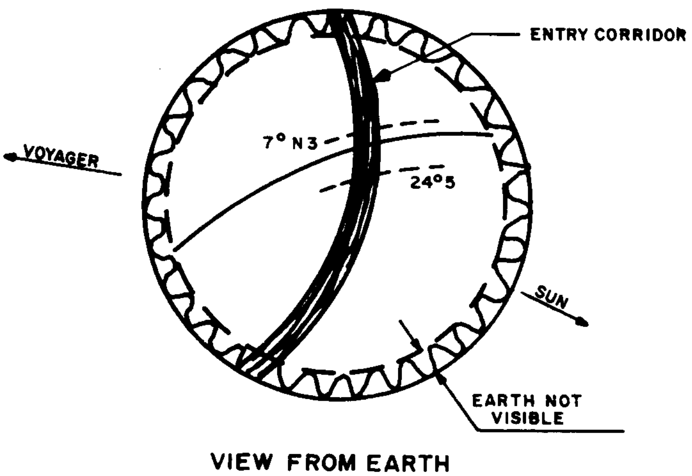

Figure 2.3-5(a). Earth Visibility

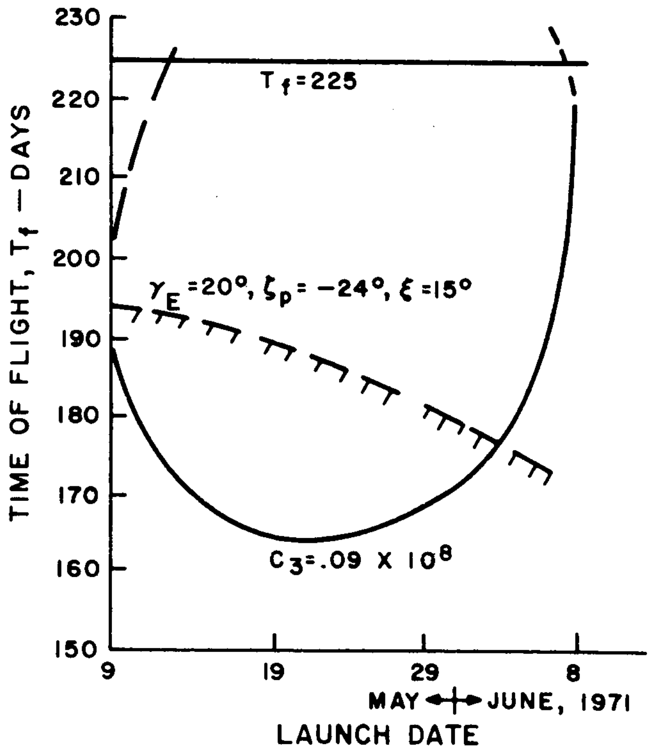

Figure 2.3-5(b). Trip-time Constraint (Earth minimum of 15° above Martian Horizon)

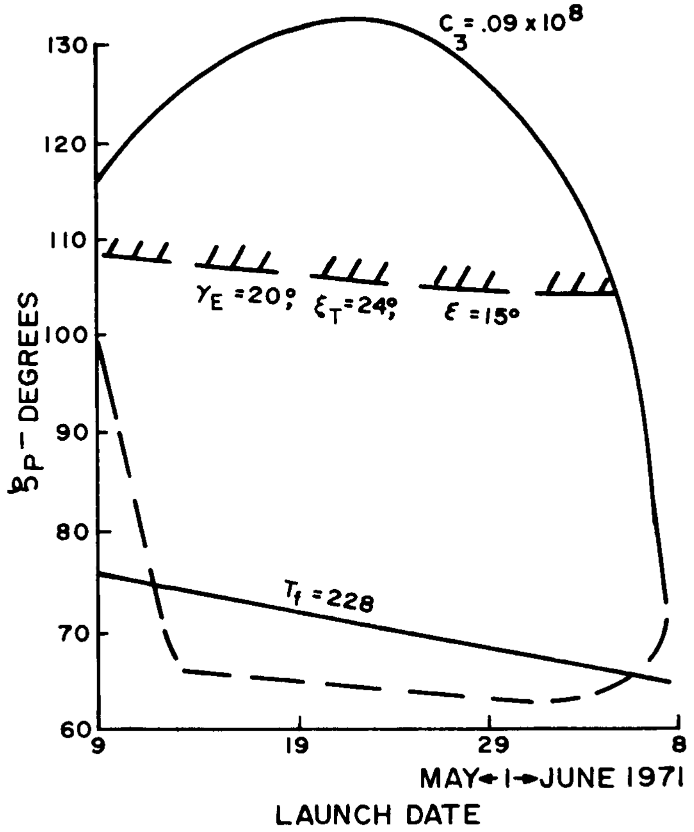

Figure 2.3-6. Trip-Time Constraint (Launch Date versus ζ_p)

(the worst case), and for the Pandorae Fretum location at 24 degrees south latitude.

Figure 2.3-7 shows the range of elevation angles represented by the time of flight and energy constraints for the two limiting values of entry angle, 20 degrees and 35 degrees. In the case of the 20 degree entry angle it is seen that in the absence of a limit on ζ_p the Earth elevation angle would go below 15 degrees. For higher entry angles this constraint disappears.

However, since ζ_p is determined at launch the constraint is present even in those cases where higher entry angles are realized. The upper dotted line, which pertains only to the 35 degree entry angle contours, shows the lower limit of elevation that can be reached for a 35 degree entry angle when observing the ζ_p constraint that is necessary for a 20 degree entry.

Figure 2.3-7 pertains to a 24 degree south latitude landing site. The equivalent curves for a 7 degree north latitude are shown in Figure 2.3-8.

Figures 2.3-9 and 2.3-10 show the equivalent information in terms of time of Martian day at landing.

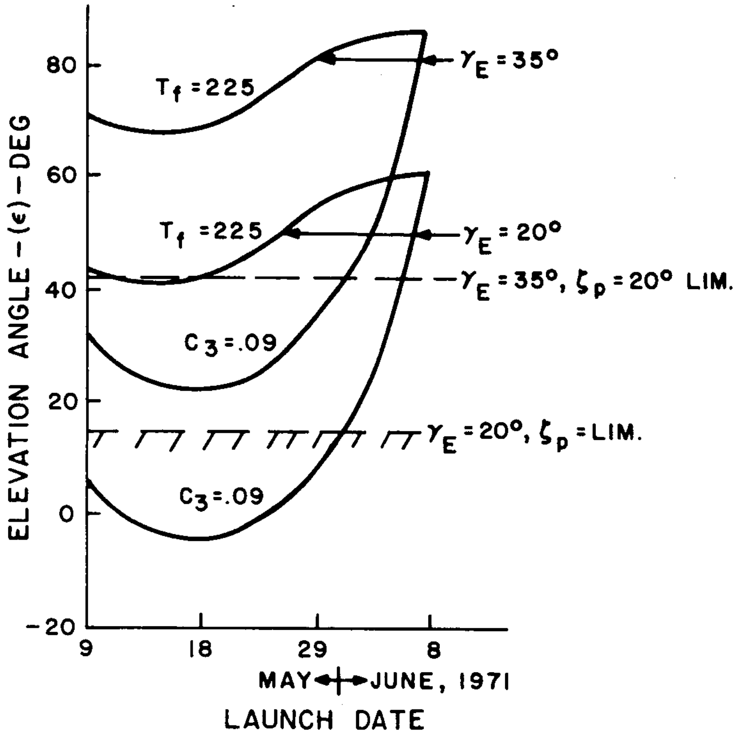

Figure 2.3-7. Trip-Time Constraint (Martian 24° South Landing Site)

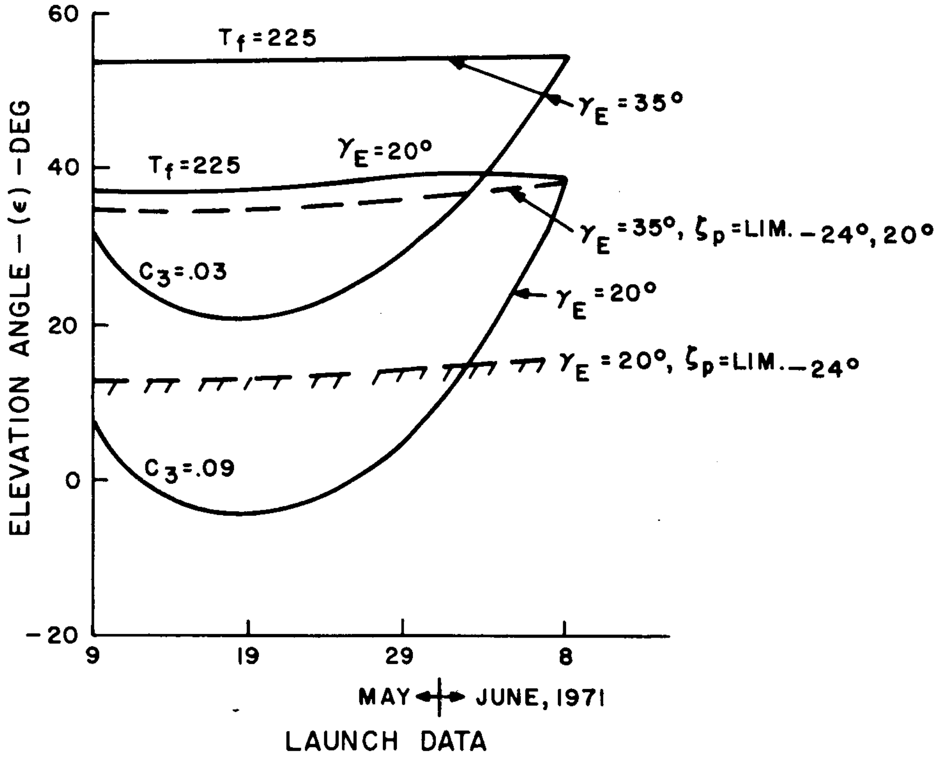

Figure 2.3-8. Trip-Time Constraint (Martian 7° North Landing Site)

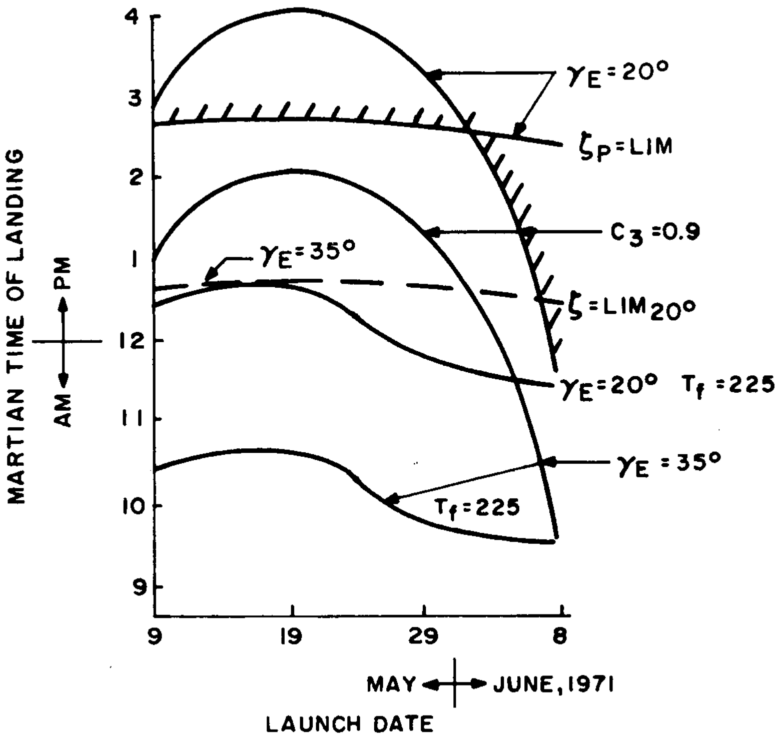

Figure 2.3-9. Time of Martian Day At Landing (Target Latitude = 24° South)

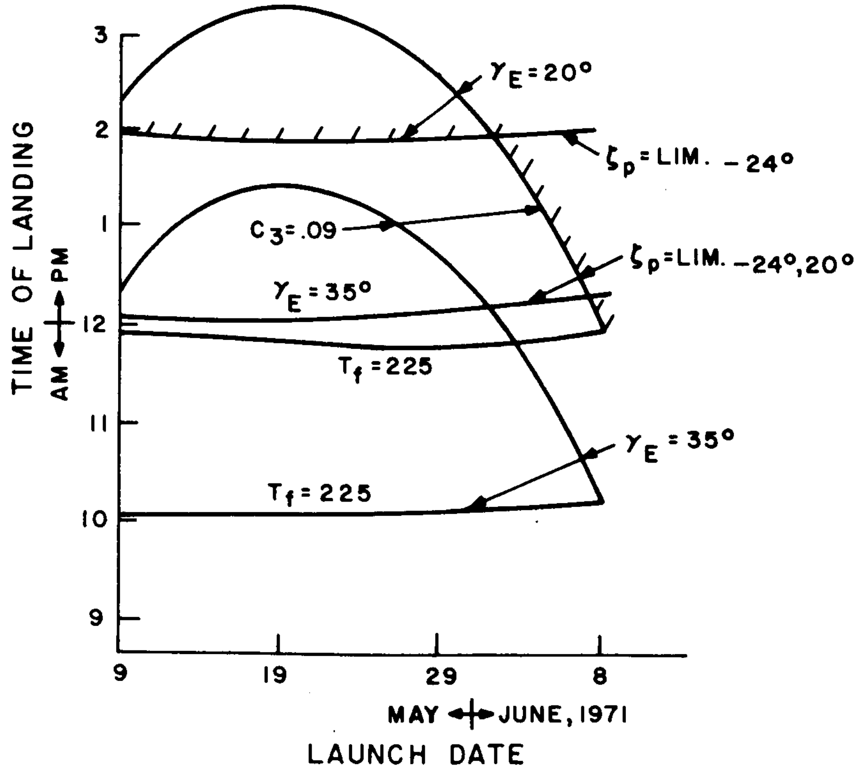

Figure 2.3-10. Time of Martian Day At Landing (Target Latitude = 7° North)

D. TRAJECTORY ANALYSIS

The nominal trajectory used for the approach guidance study was selected to conform to the following requirements:

1. A Mars 1971, Type I trajectory within the May 6 - June 5 launch period
2. Flight time less than 225 days
3. C_3 less than 10 $(Km/sec)^2$
4. Low approach velocity (3.0 Km/sec or less)
5. Landing site visible from Earth (Earth 15 degrees above the horizon at impact)

The selected trajectory has a launch date of May 19, 1971 and a flight time of 200 days. The geocentric hyperbolic asymptote for this trajectory is:

C3	-	8.08 $(Km/sec)^2$
Right Ascension	-	336.8°
Declination	-	26.2°

A trajectory was generated with these characteristics using the GE N-body interplanetary trajectory program.

The trajectory passes Mars at an altitude of 1000 miles on the sunny side and near the equator.

E. APPROACH NAVIGATION ERROR ANALYSIS

As a result of DSIF tracking and concurrent trajectory computation, the position of the Bus/Lander (with respect to Mars) several days before arrival will be known to an accuracy of about 700 kilometers*; that is, the major axis of the 1σ position uncertainty ellipsoid will be 700 kilometers. The uncertainty in the impact parameter obtained by projecting the uncertainty ellipsoid onto the impact parameter plane, is about 400 kilometers. The major sources of this uncertainty are:

1. Tracking errors
2. AU uncertainty
3. Ephemeris errors
4. Uncertainty in solar radiation pressure
5. Unbalanced attitude control torques and gas leaks.

*This data provided by JPL

As the bus approaches Mars and its motion is influenced by the gravitational attraction of the planet, continued DSIF tracking can detect the resulting change in velocity, allowing an improvement in the accuracy of the trajectory determination. At a distance of 150,000 miles from Mars, however, the improvement is negligible. The uncertainty in the predicted impact parameter at the time of capsule separation should be less than 100 kilometers in order to ensure at least a 0.99 probability of hitting the required entry angle corridor of 20 degrees to 35 degrees. Some kind of Approach Guidance is therefore necessary.

An error analysis was made of the approach navigation scheme used in the previous Voyager study, consisting of measurements of the direction of the line-of-sight from the Bus to the center of the planetary disk with respect to a celestial reference coordinate system.

The following assumptions were made:

1. Measurements begin when the vehicle is two million miles from the planet
2. The angular accuracy of each measurement is 1 milliradian
3. Measurements are made at 8-hour intervals

The initial uncertainty at the commencement of the approach phase was obtained from an error analysis of the DSIF tracking during the midcourse phase. Range rate tracking from a single ground station was assumed starting at a point 1 million miles from the Earth (after the first midcourse correction).

A tracking accuracy of 0.1 meter per second and a data rate of 1 measurement per hour were assumed together with an AU uncertainty of 1000 kilometers. The resulting standard deviations in the position and velocity uncertainties, and the correlation matrix, at the beginning of the approach phase are shown in Table 2.3-2. These served as the initial conditions for the Approach Guidance error analysis.

Figure 2.3-11 indicates the reduction of the uncertainty in the predicted position at 140,000 nautical miles from Mars, as successive determinations of line-of-sight to the planet are made during the approach phase. One determination is made every eight hours with a 1σ uncertainty of 1 milliradian. (DSIF tracking is maintained during this period.)

The values shown at 140,000 nautical miles are, of course, the uncertainties in position when that point is reached, rather than predictions.

TABLE 2.3-2. INITIAL CONDITIONS FOR APPROACH GUIDANCE ERROR ANALYSIS

Position Uncertainties (n.mi.)		Velocity Uncertainties (ft/sec)	
σ_x	367	$\sigma_{\dot{x}}$	.364
σ_y	191	$\sigma_{\dot{y}}$	.038
σ_z	203	$\sigma_{\dot{z}}$	.145

CORRELATION MATRIX

$$
\begin{array}{c}
x \\ y \\ z \\ \dot{x} \\ \dot{y} \\ \dot{z}
\end{array}
\begin{pmatrix}
1.0 & -.9634 & -.1448 & .9995 & .1774 & -.4769 \\
 & 1.0 & -.0281 & -.9607 & -.1244 & .5037 \\
 & & 1.0 & -.1587 & .4897 & -.6125 \\
 & \text{(Symmetric)} & & 1.0 & .1684 & -.4578 \\
 & & & & 1.0 & -.8196 \\
 & & & & & 1.0
\end{pmatrix}
$$

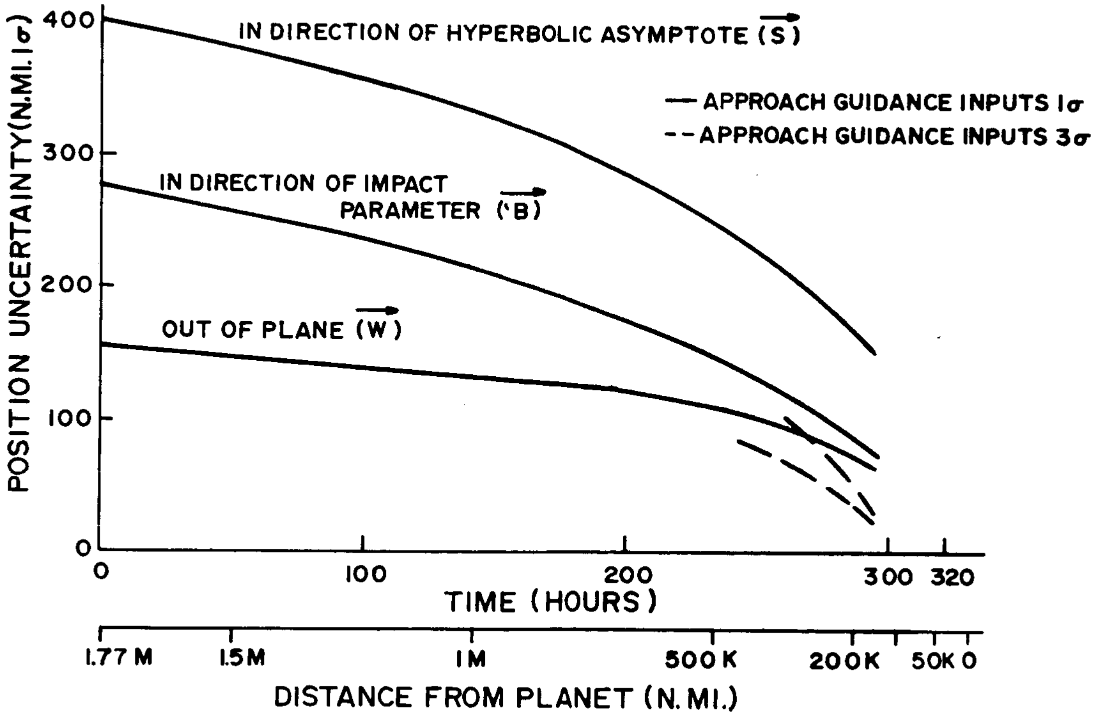

Figure 2.3-11. Time Versus Position Uncertainty

If the uncertainties at 140,000 nautical miles are propagated to the point of closest approach, they are:

In plane (radial)	70 nautical miles
Out-of-plane	45 nautical miles
In direction of velocity	253 nautical miles

These errors are correlated as in Table 2.3-3.

TABLE 2.3-3. CORRELATION MATRIX

	Radial	Out of Plane	Time of Arrival
Radial	1.0	-.021	.733
Out-of-plane		1.0	-.453
Time of arrival	(Symm)		1.0

It may be of interest to compare Figure 2.3-11 with Figure 2.3-12 which was obtained in the previous Voyager study. The two are not exactly comparable in that Figure 2.3-11 pertains to uncertainties as of 140,000 nautical miles, while Figure 2.3-12

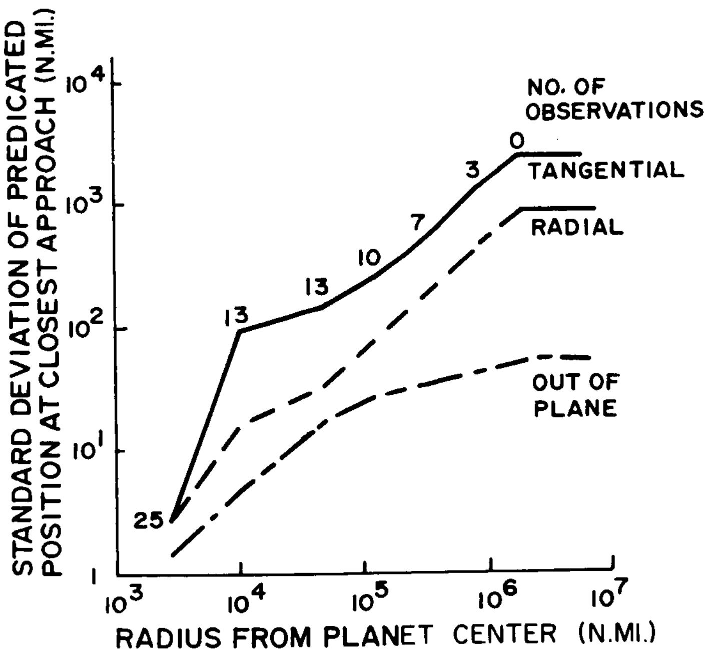

Figure 2.3-12. Mars Trajectory Determination (DSIF plus Line-of-Sight Observations)

projects to the point of closest approach. This difference is small, however, at the scale to which the curves are drawn.

There is a significant difference, however, in the initial uncertainties before beginning to take Approach Guidance data. Figure 2.3-11 reflects considerably lower initial errors, in line with the present JPL estimates of DSIF - based trajectory determinations.

After completion of the above analyses, it was determined that the 1 milliradian figure for line of sight accuracy was a 3 σ value rather than 1 σ as it had been considered. Accordingly, another run was made using the same initial covariance matrix and reducing the measurement uncertainty to 1/3 milliradian, 1 σ. The resulting position uncertainties at 140,000 miles are shown in Table 2.3-4.

TABLE 2.3-4. POSITION UNCERTAINTIES AT 140,000 MILES

		Standard Deviation (n. mi.)
ERRORS IN IMPACT PARAMETER PLANE	in-plane	26.2
	out-of-plane	24.4
	Time of arrival	59 seconds

For comparison, these in turn scale down as follows when projected ahead to perifocus:

In plane (radial)	24.3 nautical miles
Out of plane	16.4 nautical miles
In direction of velocity	136.0 nautical miles

with the following correlations

$$\begin{pmatrix} 1.0 & -.014 & .610 \\ & 1.0 & -.237 \\ & & 1.0 \end{pmatrix}$$

In both cases, the velocity uncertainty was less than 0.1 foot/second which was considered to have a negligible effect on entry dispersion.

In both the previous Voyager study and this present study, the effect on navigation accuracy resulting from an approach trajectory correction was not analyzed. It is recognized that a correction, which may occur in the region of 1 to 2 million miles from the planet, will introduce new uncertainties into the trajectory. These can be evaluated only after further tracking. The rapidity with which they can be determined is of considerable importance to this type of mission. It is strongly recommended that a study of this problem be initiated.

F. LANDER GUIDANCE

Surface dispersion calculations which were carried out under the previous Voyager study are in themselves correct. At that time, however, dispersion of entry angle per se was not considered important and it was not separately determined. In addition, it is recognized that both entry angle and surface dispersion become increasingly sensitive to guidance errors as the entry angle decreases, hence it was necessary to carry out the error analysis again in order to determine dispersions to be expected for entry within the 20-degree to 35-degree entry corridor.

The studies carried out here made use of the specific approach navigation accuracy information from this study, together with the appropriate entry velocity for a 1971 mission.

1. Entry Angle Dispersion

The case studied was for a nominal 30-degree entry angle, with execution errors at separation of 1 percent in Δ V magnitude and 3-degree error in orientation of thrusting at separation. Based on these data we obtain the following which are 3σ numbers:

• Due to navigation errors	2.4 degrees
• Total including execution errors	2.43 degrees

In the previous study it was inferred that the entry angle dispersion would be of approximately the same magnitude as the surface dispersion. As discussed in the following section, these values were found to be pessimistic and would compare appropriately with those tabulated above.

NOTE. Details of this analysis and that of the following section are covered in GE report PIR 9733-SFA-10 "Voyager-Mars 1971 Mission Lander Error Analysis" by M. Levinson.

2. Surface Dispersion

The entry trajectory, when propagated to the planet surface results in surface dispersion as shown in Table 2.3-5.

In the previous study atmospheric errors were found to be negligible and were not recalculated for this case.

For comparison the values of surface dispersion attained in the previous Voyager study are included here. A direct comparison between the present and the previous results is obtained by comparing Table 2.3-5 with Table 2.3-6. It will be observed that the total surface dispersion has increased somewhat in Table 2.3-5 which re-reflects in part the greater downrange sensitivity to entry trajectory errors, that results from shallower entry angles. This is offset in part by the lower arrival velocities characteristic of the 1971 opportunity. An indication of the importance of these two factors can be obtained by observing the decrease in crossrange errors (in which the shallower entry angle is not a significant factor) with the increase in down range dispersion (in which the effect of entry angle is prominent).

In the previous study the values in Tables 2.3-6, 2.3-7, and 2.3-8 were given as 1σ. It has since been found that in that study the Approach Guidance inputs were mistakenly treated as 1σ inputs in the interplanetary trajectory analysis computation. These inputs were in fact 3σ values and recent trajectory runs have shown that the results scale almost linearly. Accordingly, the navigation errors in Tables 2.3-6, 2.3-7, and 2.3-8 are essentially three sigma numbers and since they predominate in Table 2.3-6, Table 2.3-6 becomes directly comparable to the more recent studies in Table 2.3-5.

TABLE 2.3-5. SURFACE DISPERSION (3σ)

PANDORAE FRETUM, 1971 In-plane Landing
Separation from 1000 n.mi. flyby ($\gamma_E = 30^o$)

$\Delta V_T = 0$ $\Delta V_N = 93$ ft/sec Total $\Delta V = 93$ ft/sec

Error Source	Down Range Errors (Degrees)	Cross Range Errors (Degrees)
Navigation Errors	3.1	1.2
Execution Errors ($\Delta V = 1\%$, Misalignment $= 3^o$)	0.58	1.3
Total	3.2	1.8

RSS = 3.66^o

TABLE 2.3-6. PANDORAE FRETUM, 1969 MINIMUM ΔV (From Previous Study)

$\Delta V_T = 0 \quad \Delta V_N = 144$ ft/sec Total $\Delta V = 144$ ft/sec

Error Source	Down-Range Errors (Degrees)		Cross-Range Errors (Degrees)	
1. Separation Errors				
ΔV Magnitude (1%)	0.67	0.67	0	0
ΔV Direction (1^o)	0.05	----	0.64	----
(4^o)	----	0.19	----	2.56
RSS	0.68	0.70	0.64	2.56
2. Navigation Errors	2.66	2.66	2.80	2.80
3. Entry Errors				
Atmosphere	0.27	0.27	0	0
W/C_DA (20%)	0.07	0.07	0	0
4. Total RSS	2.75	2.75	2.88	3.79

TABLE 2.3-7. SYRTIS MAJOR, 1969 (From Previous Study)

$\Delta V_T = \Delta V_N = 147$ ft/sec Total $\Delta V = 281$ ft/sec

Error Source	Down-Range Errors (Degrees)		Cross-Range Errors (Degrees)	
1. Separation Errors				
ΔV Magnitude (1%)	0.24	0.24	0.58	0.58
ΔV Direction (1^o)	2.23	----	0.70	----
(4^o)	----	8.93	----	2.80
RSS	2.24	8.93	0.91	2.90
2. Navigation Errors	2.88	2.88	2.14	2.14
3. Entry Errors				
Atmosphere	0.27	0.27	0	0
W/C_DA (20%)	0.07	0.07	0	0
4. Total RSS	3.64	9.35	2.34	3.62

Note: $1^o = 10\pi$ n. mi. on the surface of Mars.

TABLE 2.3-8. PANDORAE FRETUM, 1969 (From Previous Study)

ΔV_T = 238 ft/sec ΔV_N = 150 ft/sec Total ΔV = 281 ft/sec

Error Source	Down-Range Errors (Degrees)		Cross-Range Errors (Degrees)	
1. Separation Errors				
ΔV Magnitude (1%)	0.68	0.68	0	0
ΔV Direction (1^o)	1.90	----	1.23	----
(4^o)	----	7.72	----	4.92
RSS	2.05	7.75	1.23	4.92
2. Navigation Errors	2.66	2.66	2.80	2.80
3. Entry Errors				
Atmosphere	0.27	0.27	0	0
W/C_DA (20%)	0.07	0.07	0	0
4. Total RSS	3.37	8.20	3.06	5.67

It was shown in the previous Voyager study that surface dispersion is strongly affected by the magnitude of the velocity increment imparted to the Lander at separation, and during that study the case shown in Table 2.3-6, where a correctly sized rocket was assumed, was considered to be of more or less academic interest only. Two factors were considered pertinent: 1) the rocket would undoubtedly be oversized to cover contingencies so that a nominal sized rocket would not be available; 2) some acceleration of the Lander was considered desirable in order to maintain line-of-sight contact between the Lander and the Orbiter to permit the Orbiter to act as a communication relay.

In the present study, a relay link from the Lander via Orbiter to Earth is not contemplated. Consequently, Lander acceleration is no longer of interest.

As to oversizing the separation rocket, it is now recognized that this is of no importance. It is still true that execution errors at separation are a function of the rocket size. However, dispersion is affected not by the size of the rocket, but by how much oversize it is. The large errors occur when it is necessary to fire the rocket other than normal to the trajectory in order to accommodate excess impulse. But as is well known, the impulse needed at separation is a function of range from the planet at which the separation takes place. This is shown in Figure 2.3-13.

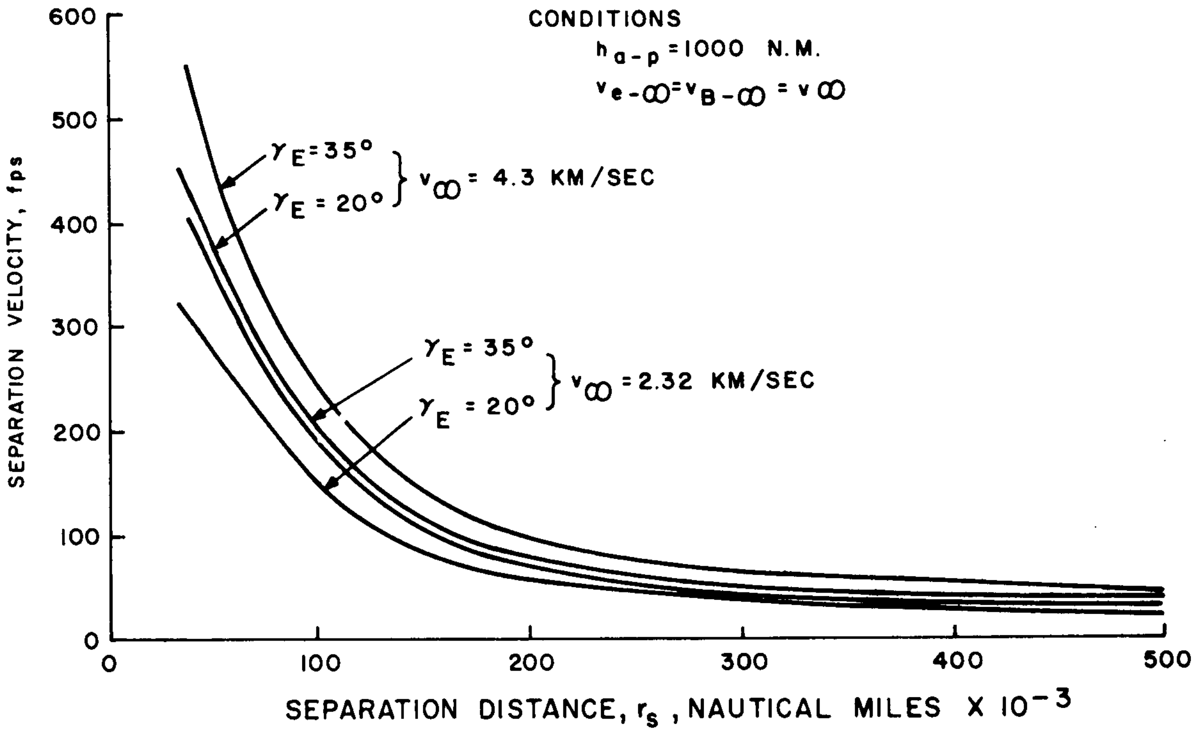

Figure 2.3-13. Impulse Needed at Separation versus Range from Planet

Accordingly, if a mission is planned with a nominal separation point of 150,000 miles, and if for contingency a rocket is provided with enough impulse to separate at 100,000 miles, all that is necessary in the case where the actual trajectory coincides with the nominal is to wait until 100,000 miles to separate. For these reasons, the situation in Table 2.3-8 is unrealistic.

An out-of-plane landing, of course, does require a large rocket to make the plane change. Accordingly, the figures in Table 2.3-7 still stand. They also can be considered approximately three sigma values.

G. FLY-BY VERSUS IMPACT TRAJECTORY

The question of placing an entire probe on an impacting trajectory prior to separation of the Lander is pertinent only to the case of the Bus/Lander. No serious consideration was given in this study to an impacting trajectory for an Orbiter/Lander combination where the Orbiter would be required to separate the Lander and then go into a path which would give it an opportunity to inject into orbit.

In the case of the Bus/Lander, the trade off is between increased Lander accuracy and the reliability penalty for a Bus that must either dodge the planet or be sterile. The subject of reliability is discussed elsewhere.

From the standpoint of accuracy there are two potential sources of improvement with an impacting trajectory. First, the execution errors at separation will decrease. This is because there are no tip-off and spin-up errors associated with the orientation of the separation impulse. Second, although an impulse will be required at separation, it will be small since it is needed only to remove the remaining trajectory errors, as contrasted with the flyby case where the total flyby bias must also be removed.

As seen in Table 2.3-5, however, the total execution errors are small, so little case can be made for an impacting trajectory on this count. The remaining source of accuracy improvement is to reduce the navigation errors. There is a possibility here, particularly in the case of an impactable Bus, in that the final correction and separation can be delayed until very close to the planet at which time navigation errors have become very small. This consideration would appear to be of importance only to future missions since as shown in Table 2.3-5 the total surface dispersion now anticipated is well within the requirements both for hitting the entry corridor and for impacting in the desired areas.

For those cases where for any reason it may be desirable to place the Lander on an impact trajectory and then deflect the Bus, the required velocity increments to do so were calculated. This is shown in Figure 2.3-14. Figures 2.3-15 and 2.3-16 show the additional impulse that would be required in case the orientation accuracy of the bus has deteriorated appreciably at the time it performs this, its last function of the mission.

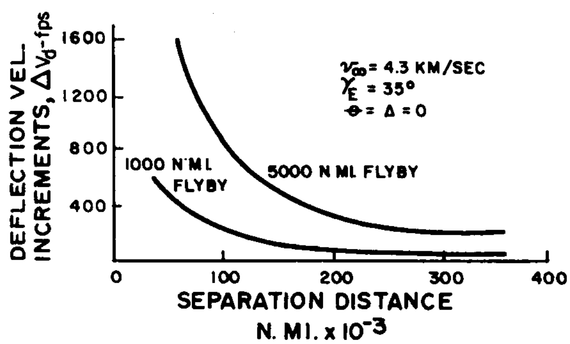

Figure 2.3-14. Deflection Velocity Increment

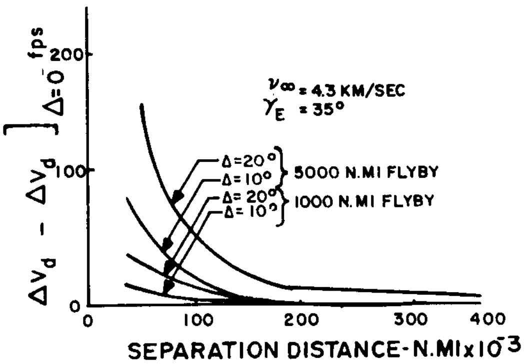

Figure 2.3-15. Increase in Deflection Velocity Increment Due to In-plane Error (Δ)

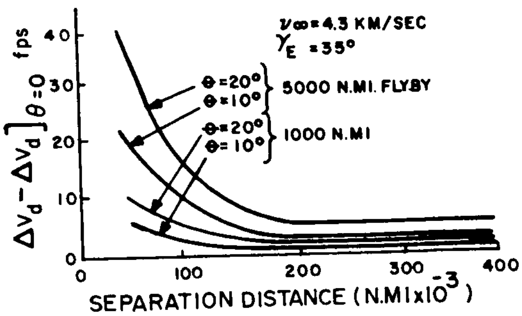

Figure 2.3-16. Increase in Deflection Velocity Increment Due to In-plane Error (θ)

2.3.4 BUS/LANDER AIMING POINT

The approach trajectory for the Orbiter/Lander Saturn 1B Voyager study was constrained to have the closest approach to the target planet of 1000 nautical miles so that there would be no possibility of the unsterilized orbiter striking the surface of the planet either because of navigation errors before orbit insertion or because of a decaying orbit due to aerodynamic drag at a lower periapsis. This same restriction applies to the All/Orbiter or combined Orbiter/Lander systems launched by the Titan IIIC booster. The same restriction, applied to the Bus/Lander configuration, would then result in the unsterilized Bus having a closest approach distance of 1000 nautical miles. The Bus is not injected into orbit at this point, but continues past the planet on what is termed a flyby trajectory. A velocity impulse is imparted to the Lander to change its trajectory from one that enters the atmosphere within the required corridor and landing site limits.

This scheme cannot be altered for Orbiter/Lander combinations and is not relevant to an All-Orbiter mission. However, consideration of the Bus/Lander mission revealed several criteria that could possibly lead to a different kind of trajectory for this mission.

- The weight of the solid rocket for diverting the Lander to impact is a function of required ΔV and the weight of the Lander. The bus is far lighter than the Lander i.e., ~ 600 pounds versus ~ 1900 pounds and thus if the bus were diverted rather than the Lander, the rocket would be lighter.
- The Bus may not have a mission after separation from the Lander and therefore the accuracy of its diversion maneuver could be much less than for the Lander.
- The 11 mb atmosphere assumed in this study from the beginning requires a narrow, shallow, entry corridor in order to achieve usable ballistic parameters ($W/C_DA = 15$) and the required velocity at altitude for parachute deployment (Mach 2.5 at 20,000 feet). If the Bus were diverted instead of the Lander it appeared that the ΔV execution error in Lander entry and landing accuracy could be eliminated; thus increasing accuracy where it might be needed, either for atmospheric entry probability or for scientific mission landing site requirements.
- However, placing an unsterilized Bus on an impact trajectory with the atmosphere and/or surface of Mars requires more reliance on the reliability of the Bus's impact avoidance maneuver than is required on the fly-by trajectory. If the Bus fails to function while on a fly-by trajectory, the mission is lost but planet biological isolation is still preserved. If control is lost, while the spacecraft is on an impact trajectory, isolation is lost unless the Bus is as sterile as the Lander.

These considerations led to the identification of three possible planet approach trajectories for the Bus/Lander mission, see Figure No. 2.3-17. The first is the previously selected flyby trajectory where the Bus/Lander is always on a miss trajectory and after the usual approach guidance and correction maneuver, and separation of the Lander from the Bus a velocity impulse is applied to the Lander. The alternate to this is the second and is termed the impact trajectory. The Bus/Lander is on an impact trajectory for the entire trip after either injection or the first successful, midcourse correction. In this trajectory the approach guidance and correction maneuver are utilized to refine the impact trajectory of the entire spacecraft. Immediately after confirmation of an accurately executed approach correction, the Lander is separated from the Bus with a very small force (about 1 foot per second velocity) and is spun up to maintain the desired orientation for the correct angle of attack at atmospheric entry. After the Lander drifts far enough away from the Bus, a velocity increment is applied to the Bus to preclude planetary impact by the Bus.

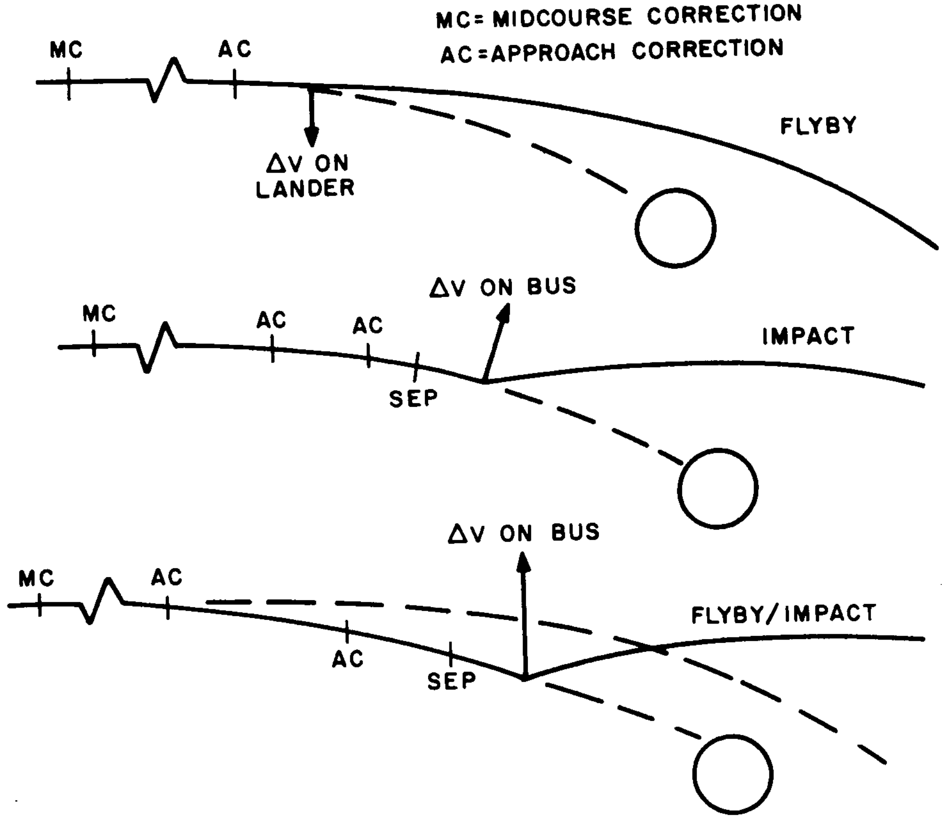

Figure 2.3-17. Lander/Bus Trajectories

It is apparent that in the impact trajectory there exists a risk that Mars would be contaminated if control of the spacecraft is lost during the long transit period of the mission. Bus function must be maintained throughout this period, which places a burden on the reliability of the spacecraft. However, it was noted that this risk could be greatly reduced if the spacecraft were on a flyby trajectory until the approach correction maneuver. This velocity impulse would be applied far enough from the planet to be economical but within the functional range of the approach guidance system, and would place the Bus/Lander on an impact trajectory. This maneuver would and could only be executed if spacecraft control was still available. Thus an in-transit failure would not cause a non-sterile impact. After an additional approach correction maneuver, the Lander would be separated on a very accurate impact trajectory and then an impact avoidance maneuver would be executed by the Bus. This trajectory is termed flyby/impact. It was expected that that this variation could provide maximum entry and landing accuracy with minimized risk to planet biological isolation.

Reliability analysis was applied to the flyby/impact trajectory with the required goal of satisfying the NASA requirement of 10^{-4} probability of a non-sterile impact. It soon appeared that two solid ΔV rockets would be required on the Bus in order to provide backup for the midcourse engine. The Bus would have to have its own power supply and be able to proceed to change its attitude, if the midcourse engine failed to ignite, in order to present the backup solid rockets in the proper orientation to execute the impact avoidance maneuver. In addition it was felt that a communications system would have to be provided on the bus that would function after Lander separation in order to report the successful execution of the maneuver, and if required, provide command capability to backup the pre-programmed series of impact avoidance maneuvers to be performed by the Bus programmer. This assurance was not readily available.

In addition, error analysis, applied to the flyby trajectory showed that with approach guidance, the 3σ error in entry corridor was only ± 2.43 degrees with guidance and separation execution errors, which is well within the previously selected corridor of γ_e = 20 degrees to 35 degrees, measured at 10^6 feet altitude above Mars surface. The landing site dispersion is calculated to be ± 3.2 degrees down range and ± 1.8 cross range.

The down range direction will be within 20 degrees of the lines of latitude (parallels) because of the approach geometry for 1971. The landing site limits, set in the Voyager study are:

Pandorae Fretum	Lat. 24°S $\pm 4^\circ$
	Long. 310° $\pm 20^\circ$
Syrtis Major	Lat. 7°N $\pm 7^\circ$
	Long. 285° $\pm 5^\circ$

It can be seen that the down range dispersion of ± 3.2 degrees is less than the smallest desired foot print of ± 5 degrees of longitude, likewise for cross range dispersion of ± 1.8 degrees and ± 4 degrees for the latitude error of the landing site.

The flyby trajectory was selected because: 1) guidance would be quite adequate for entry corridor and presently defined landing site limits, 2) planet isolation is assumed without risk and 3) possible weight savings disappeared due to the requirement of 2 solid ΔV rockets for impact avoidance maneuvers by the bus.

2.3.5 INTEGRATED VERSUS SEPARATE BUS

A. PURPOSE OF BUS

The main purpose of the Bus is to deliver the Lander to its impact trajectory. Bus functions include propulsion, guidance, attitude control and communication. These functions require a power supply, thermal control and structural integrity of Bus/Lander geometry from the launch pad through separation. It is readily apparent that some of these same functions are also included in the Lander.

A Bus that is capable of operating as a spacecraft without using services of any Lander components is termed a separate Bus. A Bus that utilizes some of the Lander components during transit is termed an integrated Bus.

Factors considered in determining the degree of integration are:

1. Reliability
2. Applicability to mission phase
3. Weight
4. Post separation Bus mission

B. RELIABILITY

Critical components that require redundancy to provide adequate reliability and are also necessary for both Lander and Bus function may be duplicated by placing one in each module, Bus and Lander, or by placing two units in either or both modules. It was noted that while one unit in each module would provide redundancy, through Bus/Lander connections during transit, the redundant component mounted on the Bus would not be able to back up the lander function after separation from the Bus. It was concluded that greatest reliability was obtained for the total mission by including all redundant components in the Lander.

C. APPLICABILITY TO MISSION PHASE

It is also apparent that some Bus functions have no application to the Lander mission after separation.

Questions of component allocation could often be easily decided on the basis of mission phase applicability. When the usefulness was totally confined to one phase or the other, the component was allocated to either the Bus or the Lander.

The weight of those components which had no surface phase application, such as Bus structure, mid-course propulsion, vehicle attitude control external antennas that could not withstand atmospheric entry environment, approach guidance, attitude control programmer and logic unit, was substantial and would decrease the Lander payload capability. A decision was made to jettison these items at separation and inject only the basic Lander into the atmosphere.

D. WEIGHT

When a component would serve both transit and surface functions, putting only one in the Lander would save weight and this was done if no other criteria were significant.

E. POST SEPARATION BUS MISSION

In a Bus/Lander configuration there exists no strong scientific mission incentive for the Bus to be able to operate as an independent spacecraft after the Lander is separated from the Bus and launched on its impact trajectory and a decision was made to eliminate any post Lander separation mission for the Bus.

F. ALLOCATION OF COMPONENTS

The decision to utilize the flyby trajectory (See Section 2.3.2(E) simplified the required Bus functions and eliminated any need for independent operational functioning by the Bus after Lander separation. Thus, the Bus can be "dead" after Lander separation.

The communication system is entirely within the Lander with the exception of the two omni-antennas for command reception, back-up transmission, and the 3-feet diameter parabolic high-gain antenna, which are all used during transit. All the remaining communications components are used during transit and during the Lander entry and surface phases.

The Lander power supply is an RTG and, since its output is continuous and use does not degrade its reliability, it is used to power the bus functions during transit. Hard-wire connection is supplied through the Lander biological isolation cover.

Guidance and control components have no surface application and are thus all in the Bus.

Propulsion has no surface application and is all in the Bus with the exception of the Lander ΔV solid fueled motor which is attached to the Lander.

The Bus structure supports the Lander in the boost phase and provides required functions during the transit phase as discussed above.

Thermal control components are dependent on individual components and are deployed as required.

2.4 COMMUNICATION MODES AND DATA RATES

2.4.1 GENERAL

This section deals with establishment of data requirements for each mission phase of each system. Prime and secondary or "back-up" modes were selected in order to accommodate the respective data requirements. Back-up modes were designed only to provide much lower data rates for such purposes as in-transit or surface phase emergencies and are not to be construed as having full mission capability. In the case of the Lander descent phase, there is only one mode which provides the most essential entry dynamics, diagnostic, and prime atmospheric parameter data.

Back-up modes are intended for use when failure of attitude control, high-gain antenna or pointing mechanism, or when maneuvering requirements preclude transmitting through the narrow beam width antennas. Data rates are drastically reduced to barest essentials of critical diagnostic and non-pictorial scientific data of highest interest and lowest bit requirements.

Back-up mode data rates were usually near marginal for the distance prevailing in the mission, since these modes use very broad beam "omni-directional" antennas in order to have the mode operate independent of vehicle attitude.

In most cases, prime data rates were established on the basis of mission requirements. It is obvious that high resolution television systems can provide virtually unlimited non-repetitive information. These high volume information generators can be used on a flexible basis, dependent on the varying Earth/Mars communication distance and the obtainable data rates. These data rates were varied by factors of two, in accordance with varying distance during a particular mission.

Prime data rates were established by balancing high data volume generating payload, such as television cameras, with reasonable power supply weights and parabolic antenna sizes compatible with the spacecraft sizes. Command receiving modes in the spacecraft have, of course, very low rates and the spacecraft are capable of receiving commands in any attitude. Command back-up links are marginal but prime command links are operational out to maximum distances expected during the missions.

2.4.2 BUS/LANDER COMMUNICATION MODES AND DATA RATES

A. TRANSIT

The high power transit link uses a 3-foot diameter, parabolic, high-gain antenna, which is limited by the physical dimension of the Lander/Bus. The link can transmit a terminal guidance TV frame in 45 minutes through this dish at a data rate of 400 bits/second.

It should be noted here that Bus/Lander Titan IIIC missions differ from the prior Voyager spacecraft mission in that the transit high-gain antenna is not stowed and deployed every time there is a trajectory correction. In the Saturn 1B spacecraft, the 9-foot orbiter antenna was too large and flimsy to be allowed to overhang on its long boom while the mid-course or orbit insertion velocity increments were being executed. In the Bus/Lander, the dish size is only three feet in diameter and the boom is very short. The thrust is only 50 pounds and since it is only used when the Lander is attached to the Bus, g forces are small. Therefore, the 3-foot diameter dish antenna is deployed once after injection and is left in that position for the entire trip, thus eliminating many motions of the antenna and increasing its reliability and simplifying the design of the deployment mechanism. It is still oriented on two axes.

B. POST SEPARATION

After the Lander is separated from the Bus, the Bus ceases to function, and diagnostic telemetry is transmitted directly from the Lander to Earth. The 150 degree beamwidth antenna in the center of the lander aft cover is blanked by the solid rocket which imparts the required velocity impulse to the Lander. After burn-out, the rocket is jettisoned, thus clearing the antenna pattern.

Earth will be within this antenna beam because in separating from a flyby trajectory, the velocity impulse will be imparted normal to the velocity vector in order to minimize both the impulse required and the Lander dispersion. Since the velocity vector of the probe at this point is in no case colinear with the Mars/Earth line, the Lander's axis is never normal to the Mars/Earth line. The spin-stabilized Lander maintains this orientation until, at entry, it is oriented along the velocity vector (zero angle of attack).

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The spacecraft is guided so that the Earth is a minimum of 15 degrees above the horizon at impact. Hence, in the extreme case, the Lander axis during vertical descent will be a minimum of 15 degrees from normal to the Mars/Earth line.

It is planned that this link will be utilized for only one transmission of separation and separate cruise diagnostic telemetry data, ending at least one hour before entry to allow time for the batteries to be recharged.

As soon as entry black-out is ended, the "descent link" transmits engineering and critical atmospheric data to earth. The battery is large enough to provide 10 minutes transmission at 4 bits per second; minimum descent time in the worst case is 35 seconds; seventy bits of scientific data would be transmitted in that case. Nominal descent time in the steepest entry angle in the 11 millibar atmosphere is 75 seconds. Longer times would result from shallower entry angles.

C. SURFACE PHASE

In the previous Voyager study, it was suggested that a continuous low-power link through a constantly earth-oriented helical array might prove advantageous to a Lander mission. This link was evaluated in this study.

The system requirement for the data rate for this link is dictated by the requirements for TV pictures, which was set at 5000 TV frames in the first 90 days. A rate of 800 bits/second will meet this requirement and was selected. Because of the increasing communication distance during the surface phase, Figure 2.4-1, the data rate is reduced to 400 bits/second after 62 days. A total of 8350 TV frames can be transmitted during the total mission time of 180 days.

2.4.3 ORBITER COMMUNICATION MODES AND DATA RATES

Continuous telemetry and tracking is provided by the 57 watt klystron through an omni-antenna up to 1/3 AU distance. The command receiver operates with an omni- or the high-gain parabolic antenna. Battery power is provided for short transmission of diagnostic data in the emergency mode.

The terminal guidance data rate is 6000 bit/second or one-half (for lower error rate) of the minimum data rate to be used in orbit. A single TV frame can be transmitted in 3 minutes.

The prime orbiting data rate is based on the rate of information produced by the stereo mapping pair of vidicons and the high resolution image orthicon cameras. The rate is strongly influenced by the orbit selected (see Section 2.4 - "Orbit Analysis and Selection") and by the size of the largest rigid dish that can be installed in the Orbiter and still fit in the 120-inch Titan IIIC shroud.

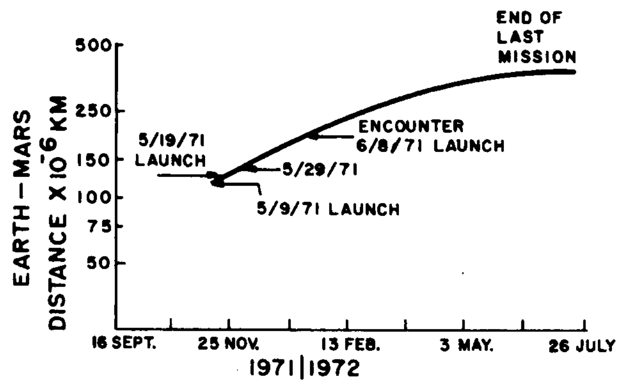

Figure 2.4-1. 1971 Mars Mission Communication Distance

The selected orbit, 1,000 x 2,278 nautical miles has a period of 4.3 hours. With 50 percent overlap required by the shape and lay of the swaths, it takes 47 orbits to complete the initial map. This is accomplished in 8.4 days. An overlap of 10 percent along the swath requires 76 individual Vidicon frames to acquire the stereo information. If the transmitter were to operate for the entire 4.3 hours of this orbit period, it would require 5400 bits/second just to handle the mapping data. Due to the high ratio of mapping time to orbit period, 2.16 hour:4.3 hour, the transmitter is operated as long as Earth is in sight.

The maximum diameter of the parabolic reflector is limited to about 9 feet and an optimized solar cell power supply allows about 50-60 watts RF power. With a ± 1 degree antenna pointing error, and an 8 db margin, a data rate of 12,000 bits/second can be obtained with a transmitter power of 57 watts. This rate allows six sets of high resolution TV frames per orbit in addition to the map pictures.

It is noted that early in the launch window, Earth is not always available to the orbiter. However, here the communication distance is less and the link can operate at 24,000 bits/second to satisfy the requirements.

At completion of the initial mapping phase of the mission, the increasing Earth/Mars communication distance causes the margin to drop below 8 db and the data rate is reduced to 6,000 bits/second. The mapping cameras are turned off and the Orbiter continues to obtain 5.4 sets of three medium resolution (140 meters) color and one high-resolution (20 meters) black and white image orthicon TV frames per orbit.

2.4.4 ORBITER/LANDER COMMUNICATION MODES AND DATA RATES

The size and weight constraints of the combined Orbiter/Lander Titan IIIC system severely limits the data rates. However, the same communication modes were incorporated in the Titan IIIC combined system as were used in the Saturn 1B Voyager study. This included a relay link from Lander to Orbiter, as well as direct link from Lander to Earth, and a high rate in the Orbiter through an orienting high-gain parabolic dish antenna.

A. ORBITER OF ORBITER/LANDER COMBINATION

The Orbiter communication link that affects the design of the rest of the communication system is the high-rate telemetry during the orbiting phase of the mission. The data rate for this link is constrained somewhat by the maximum size (8-foot diameter) of rigid parabolic antenna that can be fitted inside the cone section of the 120-inch diameter shroud. The optimized power supply for this size antenna allows 30 to 40 watts of radiated power.

Table 2.4-1, shown below is based on an orbit period of 27.6 hours with the telemetry system operating for 25.8 hours per orbit.

TABLE 2.4-1. DATA RATE REQUIREMENTS FOR VARIOUS TELEVISION FRAME RATES

Vidicon Sets 2 frames	IO Sets 3 frames	Bits/Orbit x 10^{-8}	Bits/ Second
1	0	.48	519
3	1	1.43	1540
2	1	1.9	2040
1	1	3.32	3680
2	3	4.73	5080
1	2	6.16	6630
1	3	8.98	9650
1	4	11.81	12720

A nominal rate of 6000 bits per second was selected for this system.

Since the map could not be completed before 28 days after encounter, the longest communication distance for the 6000 bit rate is 1.57 AU and 43 watts of RF power are required.

A back-up mode with 100 watts is supplied to transmit diagnostic telemetry through the omni-directional antennas.

The orbiter is equipped with the VHF relay equipment, including a Yagi antenna mounted on the PHP.

B. LANDER OF ORBITER/LANDER COMBINATION

This Lander was initially equipped with the same continuous low-power direct-communication link used in the 2042-pound Lander of the Bus/Lander system. However, detailed analysis of the Orbiter/Lander system showed that this communication link could not be used because of the limitations on the gross payload. Therefore, the RF power was reduced to 12 watts, with a 400-bit/second data rate. This late change is not reflected in the discussion of the communication subsystem in Section 4.1.3(C).

2.4.5 DISCUSSION OF A RELAY LINK

The question of whether or not to use a relay link is complicated, in this study, by the fact that the Orbiters and Landers are separate missions utilizing separate launch vehicles.

The capacity of the relay link in the prior study was restricted by the frequency and duration of line-of-sight opportunities between Lander and Orbiter and by the range at which these opportunities occurred. The total capability, with the 1000 x 19000 nautical mile orbit selected for maximum Lander weight in the prior study, was 5000 Lander TV frames over the entire 90-day lifetime of the Orbiter. However, the 1000 x 2278 orbit selected for the All-Orbiter mission in the Titan based system meant that Lander to Orbiter line-of-sight opportunities would occur more frequently with the 4.3-hour period of that orbit than with the 27.6-hour period of the 1000 x 19000-nautical mile orbit. A computer run was made based on the 4.3-hour period and the two prime Lander sites, at 7 degrees north and 24 degrees south latitude, selected in the previous

Voyager study, for a simulated period of 100 hours. These opportunities for line-of-sight between Lander site and the Orbiter were plotted. See Figure 2.4-2. The ordinate is the slant range between Orbiter and Lander and the abscissa is time in minutes. Because of the strong effect that communication distance has on allowable data rates for fixed power communication links, it was decided that the relay link would only be utilized in those passes where slant range is less than 2000 nautical miles. The data rate is based on this range, on the 10 db Yagi antenna mounted on the PHP and a 25-watt VHF solid-state transmitter with an omni antenna on the Lander, as in the prior study. The allowable data rate, in Lander TV frames per 1/2 hour (which was the duration of each period of relay link operation) is plotted against Orbiter altitude for a Lander on the horizon with respect to the Orbiter in Figure 2.4-3. With the beam of the PHP-mounted Yagi antenna pointed straight down along the local vertical through the Orbiter position, the antenna pointing loss and the slant range are at a maximum for that altitude. If the Lander is above the horizon, as viewed from the Orbiter, the Lander is closer to the antenna beam center line and the slant range is less than for the horizon situation and the data rate is, therefore, higher than for the horizon case. The minimum data rate for the 1000-nautical mile

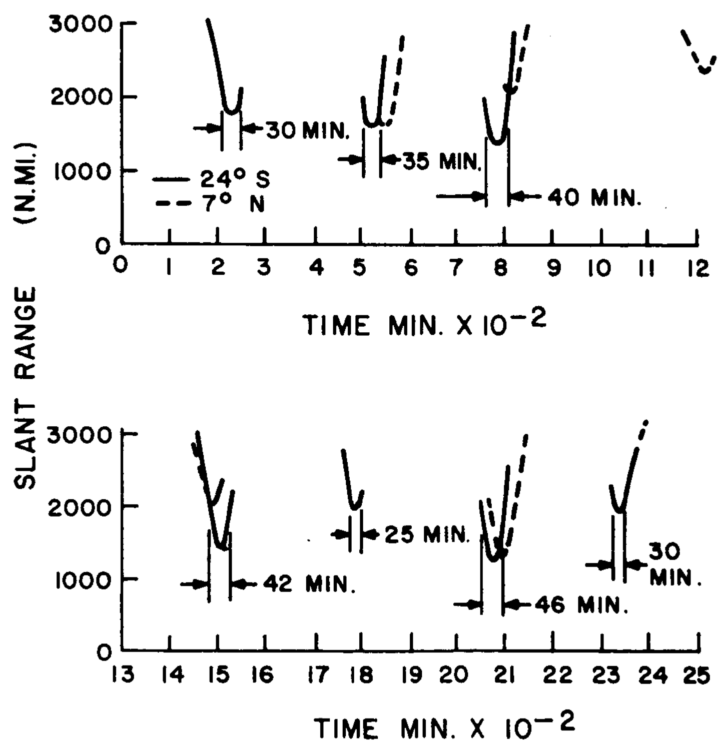

Figure 2.4-2. Relay Capability

altitude and a slant range of 2140 nautical miles with the Lander on the horizon is 22,000 bits/second or 95 frames per 1/2 hour.

The line-of-sight plots were examined for the number and duration of opportunities with slant ranges of less than 2000 nautical miles. These numbered 15 for a total duration of 535 minutes, and a total information relayed of 36,000 Lander TV frames for a 90-day mission. If a tape recorder speed of 16,000 bits/second is used for practical reasons, it still amounts to 26,000 frames per 90-day Orbiter mission with increased communication margin in the relay link.

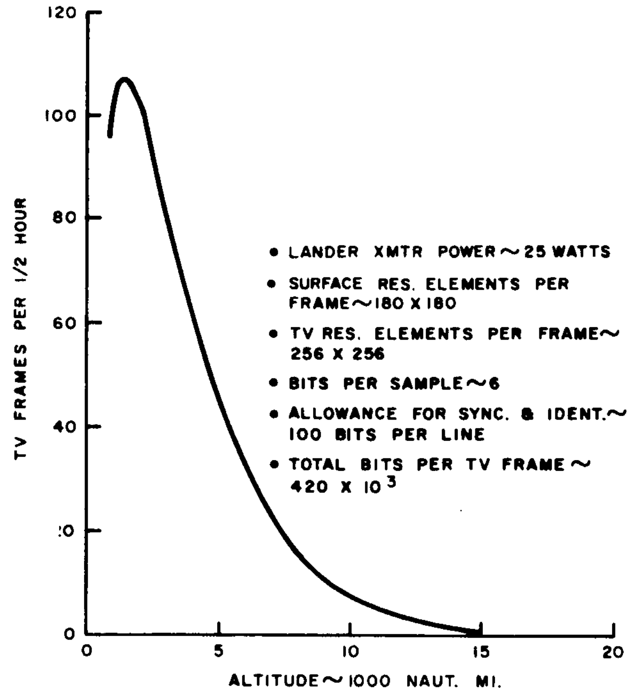

Figure 2.4-3. Lander to Orbiter TV-Frame Rate

This makes the relay link for co-ordinated Bus/Lander and Orbiters appear to be very worthwhile if both missions can be conducted. The weight penalty is 30 pounds in the Orbiter and 20 pounds in the Lander, which is small when compared to the payload capacities of each system.

The Orbiter would require about 10 percent of its communication time to send the relayed information if the basic prime rate of 12,000 bits/second were retained.

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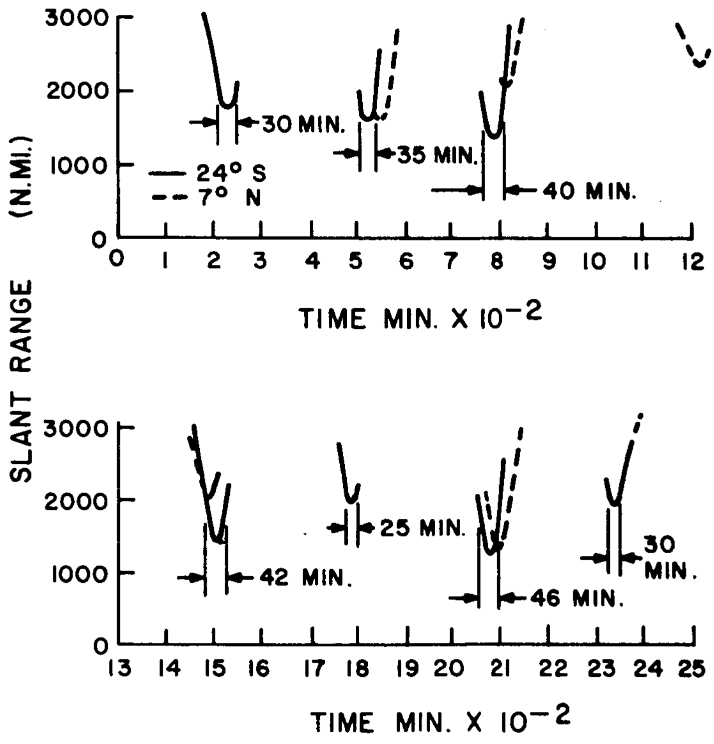

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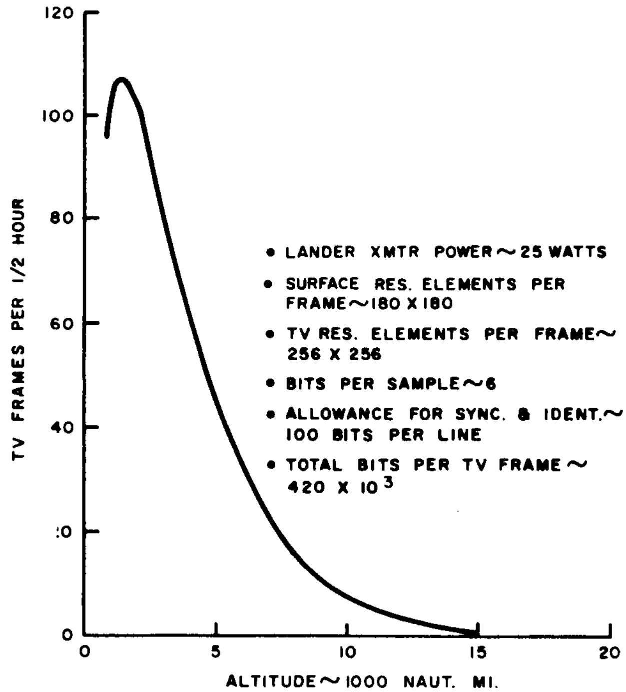

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2.5 ORBIT ANALYSIS AND SELECTION

Initial estimates of the All-Orbiter configuration showed that a 1000 x 1000 nautical mile circular orbit could be achieved with the original 215 pounds of Orbiter payload from the prior study. However, this orbit required very high data rates which could not be efficiently obtained. Consequently, the orbit period was increased to accommodate the weight of a larger power supply, to provide communication time for the television data and to provide more weight allowance for the "retro rocket and high resolution package." The 4.3 hour 1000 x 2278 nautical mile orbit was selected to provide the minimum possibility of the occurrence of a synchronous relationship between orbit and Mars rotational period. See Figure 2.5-1.

The orbit of the Orbiter/Lander combination system is constrained by weight limitations to an 1000 x 19,000 mautical mile orbit.

The All-Orbiter system was initially designed and weights were estimated on the basis of attaining, in 1971, a 1000 nautical mile altitude circular orbit. The circular orbit is very favorable for stereo mapping purposes because of the constant, minimum Orbiter altitude. Other surface scanning instruments and the TV cameras yield constant resolution and in general the highest amount of useful information, although field and particle instruments can yield more interesting information from eccentric orbits.

However, requirements for the communication link for the 1000 nautical mile orbit are severe. For the circular orbit, the constant, low altitude permits many more stereo vidicon mapping frames per orbit than the variable altitude of the 1000 x 19,000 orbit.

The higher number of frames from each orbit must be transmitted to earth during a much shorter orbital period, i.e., 3.16

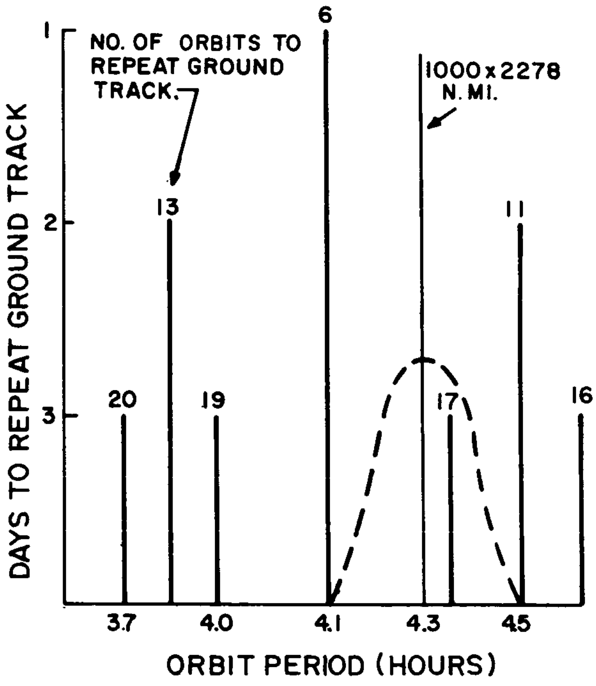


Figure 2.5-1. Orbit Selection

hours for the 1000 nautical mile circular orbit versus 7.6 hours for the 1000 x 19000 nautical mile orbit. In the prior study, the peak power requirement for the Orbiter was reduced by not transmitting during the portion of the orbit which is over the illuminated surface of mars. If this scheme were to be applied to the 1000 nautical mile circular orbit, a data rate of 19,600 bits/second would be required just to keep up with the requirements of the stereo-mapping vidicon television cameras.

Since the Orbiter system design would not permit so high a data rate, a degree of eccentricity was introduced to permit more payload and relieve the data rate requirements for the mapping cameras (by a longer period and fewer frames per orbit). This would permit operation of the three medium resolution, color, and one high-resolution black and white image orthicon TV cameras through the mapping phase of the orbiting mission and simultaneous transmission of the data.

The effects of varying eccentricity are shown in Table No. 2.5-1. Values for the listed parameters are also shown for the 1000 x 19000 orbit of the prior study for comparison.

TABLE 2.5-1. EFFECT OF ORBIT ECCENTRICITY ON MAPPING MISSION

Booster	TITAN IIIC		SATURN C1B
Year	1971		1969
System	All-Orbiter		Orbiter and 2 Landers
Orbit (n.mi.)	1,000 x 1,000	1,000 x 2,000	1,000 x 19,000
Orbit Period, hours	3.2	4.05	27.6
Orbiter Weight, lb	1660	1730	2059
Net Scientific Payload, lb	215	287	215
Mapping Time, Days	8	10	26
Data Storage, Bits	6×10^8	6×10^8	2×10^9
No. & Type of Recorder	3 TR	3 TR	2 TPR
Nominal Data Rate, Bit/Sec	12,000	12,000	14,000
Peak Power, Watts	608	608	434
Orbit Inclination	67°	67°	55°
Precession Rate, degrees/Day	1.8	1.5	.65
Precession in 90 Day	168°	135°	58°
Max. Area Mapped in Stereo			
(% of Planet)			
Initial Map	54	54	35
At 90 Days	88	88	35
Days to Complete Max. Map	61	73	26

If the period of a mapping Orbiter divides evenly into some integral multiple of the rotational period of the planet being mapped, the Orbiter will begin repeating orbit tracks or swaths that have already been photographed. This "synchronous" situation must be avoided or the effect must be minimized in order to acquire a complete map of the surface available to the Orbiter.

The worst case, of course, occurs when the Orbiter repeats its tracks every Martian day. Synchronous orbit periods are shown in Figure 2.5-1 for the low eccentricity orbits of interest in this study. The tallest bar at 4.1 hours represents the worst case of daily repetition and the numeral on top of the bar indicates the number of orbits between repeated tracks. Since map completion times for these orbits are 4 to 10 days, periods that repeat in 4 or more days are not shown. If, due to guidance error, a 4 day (or higher) repeating cycle occurs, the complete map can be acquired by tilting the PHP 5.4 degrees (equal to the field-of-view of the stereo vidicon optics) to one side or the other of the orbit track after repeating starts. The error induced in the pictures for this small angle can probably be neglected. The period of 4.3 hours has a dotted line indicating probable distribution of orbit periods if 4.3 hours is the intended period of the orbit. The extremeties of the distribution curve are based on an expected maximum velocity error of 1 percent. This period of 4.3 hours is centered between daily repetition (6 orbits/day) and an orbit that repeats every two days (11 orbits/2days). If the 4.34 hour orbit that repeats after 17 orbits in 3 days occurs, the PHP can be directed to both sides of the orbit track, successively at the end of the third day and at the end of the sixth day. This map will be complete in less than nine days. The 4.3-hour orbit has an apoapsis altitude of 2,278 nautical miles and is the orbit selected for the all Orbiter mission on the Titan IIIC launch vehicle.

The All-Orbiter mission is not affected by choice of Lander sites when a relay link is not employed, and orbit inclination can be selected on other criteria.

Higher inclinations than the 55 degrees selected in the prior study were desirable because the 1,000 x 2,278 nautical mile orbit precesses much faster than the 1,000 x 19,000 nautical mile orbit of the prior Orbiter, and the total precession during the 90 day life of the Orbiter could be more than 180 degrees. This would require extreme look angles on the PHP and high-gain antenna.

Orbit geometry was appraised early in the study on the basis of 1,000 x 1,000 nautical mile circular orbit, and the comments on the 1,000 x 2,278 nautical mile orbit are based on this work. The arrival geometry for the first day of the launch window showed that no inclination of interest would prevent occultation of earth by mars during a portion of the orbit, so this aspect did not determine inclination. It was found that 67 degrees inclination would afford an orbit with uninterrupted solar illumination of the Orbiter. This inclination limits the total progression to less than 180 degrees and was selected for the All-Orbiter mission. The all-sunlit orbits should persist through the mapping period. This inclination may provide an opportunity to take TV pictures of the receding Northern Martian ice cap.

The Orbiter/Lander system, when launched on a Titan IIIC booster, is seriously weight limited. Therefore, the Orbiter is constrained, in order to limit or minimize orbit insertion propellant requirements, to the 1,000 x 19,000 nautical mile orbit selected in the prior study. The orbit inclination for 1971 should be the same 67 degrees selected for the 1,000 x 2,278 nautical mile orbit for the All-Orbiter system.

2.6 POWER PROFILES

The major events of the transit phase of the Titan IIIC Voyager missions are the same as for the prior Saturn 1B Voyager; i.e., injection, orientation, midcourse correction, cruise, terminal guidance observation of target planet, approach correction, Lander separation, and orbit injection.

The orbiting phase of the Titan IIIC All-Orbiter mission is altered from that of the prior study because of the change in orbit, discussed in Section 2.5, Orbit Analysis and Selection, and the elimination of the relay mode of Lander/Orbiter communication. The mission tasks of map acquisition, medium and high resolution television, IR scanning of the planet surface and other orbit science are the same as for the prior study.

The descent phase of the Landers is the same except for direct transmission of entry diagnostic and atmospheric scientific information, already discussed in Section 2.4.2. The surface phase of the Lander mission has the same objective as the Landers in the prior study, namely, life detection, landscape television, and geological and atmospheric determination. However, the relay communication mode is eliminated and the prime mode of telemetry to earth is by a continuous direct transmission, discussed in Section 2.4.2. Operation of the Lander TV system or the subsurface drill requires interruption of the telemetry or postponement until after earthset. This is discussed in Section 2.6.1.

All mission phases of the Titan IIIC Orbiter/Lander combination system are the same as for the Saturn 1B Voyager system with the exception of the substitution of continuous direct communication for the intermittent system used in the prior study.

2.6.1 BUS/LANDER

The major difference in the transit phase between the combined Orbiter and two Landers of the prior study and the Bus/Lander mission in this study is the elimination of the need for the integrated Bus to function after separation. Consequently, the separation event and Lander cruise diagnostic telemetry must be transmitted directly from the Lander through an omni antenna.

The power requirement for orienting the high-gain antenna during transit is reduced from a constant drain of 29 watts for the Saturn 1B Voyager to one five-minute period per day. The spacecraft stabilization limits are ± 1 degree and if the Bus attitude control system is functioning correctly, there is no need to "true" up the earth pointing antenna on a continuous basis because the earth/spacecraft/sun angle is changing very slowly during transit. This comment applies to the transit phase for the All-Orbiter and the combined Orbiter/Lander systems as well.

It is noted in the transit power matrix, Table 2.6-1, that the communication system is not transmitting during a course correction maneuver. This is necessary because the Bus obtains its electrical power from the Lander RTG, which is sized for the surface mission (discussed below) and which is not large enough to satisfy the power requirements for both the Bus guidance and control system and communication during a course correction maneuver. Consequently, diagnostic information is stored as the maneuver is executed, and, upon the completion of the maneuver, is transmitted to earth. If the correction maneuver is the first or second mid-course and the Bus/Lander is still close enough to Earth to be within range of the 15-watt amplitron transmitting through the omni-antenna on the Bus, then this communication can be continuous and may be accomplished without requiring a successful re-orientation of the Bus/Lander.

If the correction was applied near the end of transit, then re-orientation is required for the normal mode of transmission. If orientation is not successful, then the Lander programmer senses a deviation from programmed events and the back-up communication mode is energized. The secondary battery supplies electrical energy in addition to the output of the Lander RTG. The programmer controls the duration of this transmission so that the secondary battery is not excessively discharged. When the battery is again charged by the RTG and if control of the spacecraft has still not been acquired, then the diagnostic transmission is repeated. This cycle continues until control is reacquired or until communication is lost.

The terminal guidance television pictures of Mars and the star background are obtained when the Bus/Lander is less than 2×10^6 nautical miles from the planet and the communication distance at that time is too great for communication through the omni-antenna. One picture frame is transmitted in $\sim$45 minutes using the secondary battery to supplement the RTG. Since the deficit is only 20 watts, 4 or 5 frames can be transmitted before the battery must be recharged, which can be done while the TV

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

	Launch and Injection	Initial Orientation	Initial Orientation Earth Link to Verify Canopus	Cruise	Cruise and Earth Link	Midcourse Maneuvers and Reorientation	Cruise	Term. Guidance Obser.	Cruise and Earth Link Xmit Term. Guid. Info.	Final Trajectory Correction	Cruise and Term. Guidance Obser.	Cruise and Earth Link Xmit Term. Guid. Info.
1. Communications												
· Earth Link			75		75				120			120
· Data Storage & Processing		4	4	4	4	4	4	4	4	4	4	4
· Tape Recorder								5	2		5	2
· Commands and Tracking		8	9	8	9	8	8	8	9	8	8	9
· Power Conv. & Control		10	10	10	10	10	10	10	10	10	10	10
2. Guidance and Control												
· Attitude Control		65	65	28	28	65	28	28	28	65	28	28
· Gyro and Electronics	17	17	17			17				17		
· Hi Gain Antenna Point.				29 watts for 5 minutes once per day								
· Body Mounted I.O. Camera & Electronics								25			25	
3. Thermal Control												
· Bus		5	5	15	15	15	15	15		15	15	
· Landers	15	15	15	15	15	15	15	15		15	15	15
4. Engineering Data												
· Bus	2	1	1	1	1	1	1	1	1	1	1	1
· Lander		1	1	1	1	1	1	1	1	1	1	1
5. Science												
· Descent												
· Surface												
Total Power Required	34	126	202	82	158	136	82	112	190	136	112	190
Duration	50 min	15 min	1 hr	Cruise	Cont.	46 min	Cruise	15 min	1 hr	46 min	15 min	1 hr

frames are being correlated and the approach guidance decision is being formulated on Earth. See the Transit Phase Power Profile, Figure 2.6-1.

The Lander, as stated above, transmits separation and Lander condition diagnostic information. Power is supplied by the RTG and the battery. The battery is recharged before entry so that the critical entry diagnostic and atmospheric scientific information can be transmitted during descent. This transmission continues until impact or, if a denser atmosphere is encountered, to a maximum duration of 10 minutes to avoid battery damage.

Upon impact, the aft cover is deployed and the helical array is deployed and oriented toward earth. After initial science and television information has been recorded, the remainder of the day is spent in transmitting direct to earth through the low power continuous direct link. The total power requirements for this operational mode of the Lander are 164 watts and this sizes the Lander RTG at 170 watts. Direct telemetry is interrupted to refill recorders by acquiring additional television frames from the landscape panorama system or the petrographic microscope.

Subsurface sample acquisition is accomplished at night when direct telemetry is not possible and when the battery is fully recharged to help the RTG meet the peak power demand of 256 watts during drilling.

The power profile, Figure 2.6-2, is a graphical presentation of the major modes shown in the Power Matrix, Table 2.6-2.

2.6.2 ORBITER

The transit phase of the mission differs from the transit phase of the combined Orbiter/ Landers in the prior study in that the Lander separation maneuvers are eliminated.

Since the Orbiter spacecraft is equipped with a solar cell power supply, cruise mode requires solar orientation of the spacecraft. If a course correction velocity increment requires deviation from solar orientation, the spacecraft must be powered by secondary (rechargeable) batteries throughout the maneuver until solar orientation is reacquired.

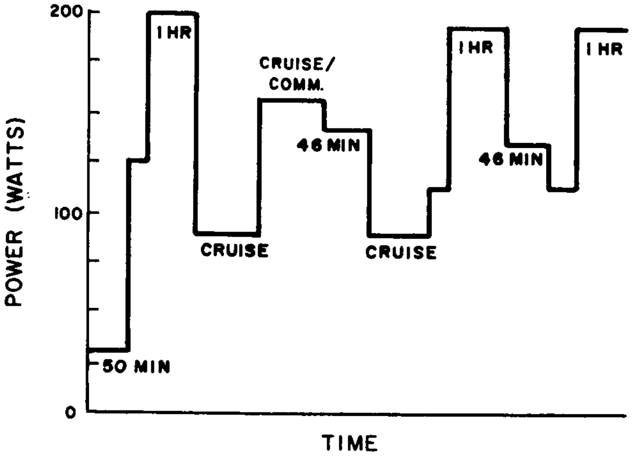


Figure 2.6-1. 2042-Pound Lander/Bus Transit Phase Power Profile

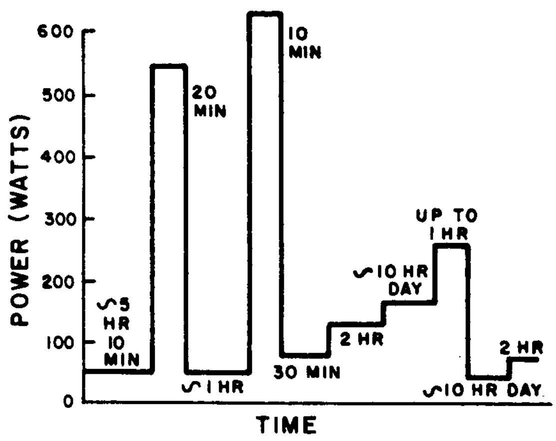


Figure 2.6-2. 2042-Pound Lander Separate Phase Power Profile

The high-gain parabolic antenna is preferably stowed during a powered maneuver in order to minimize risk of structural damage to the antenna and boom. This stowing precludes low-power telemetry during an approach correction maneuver, when the distance from spacecraft to Earth approximates encounter distances of as much as 1.33 AU, because of the power requirements for even low data rates through the low-gain omni-directional antenna.

It is planned that vehicle engineering and sequence history data generated during a powered maneuver shall be stored in the buffer storage unit of the communication system for later transmission to earth after the maneuver has been completed and sun and earth orientation of the spacecraft and high-gain antenna have been accomplished. This minimizes spacecraft attitude restrictions and battery requirements for course correction maneuvers. A minimum of 260 watt-hours is required to provide real time, continuous, diagnostic telemetry at encounter ranges through a correction maneuver and this would require 48 pounds of secondary batteries for repeated maneuvers for this purpose alone.

TABLE 2.6-2. 2042-POUND LANDER (SEPARATE PHASE) POWER MATRIX

	Lander Separation (Bus Dead)	Lander Spin Up, ΔV	Lander - Spinning - Cruising	Lander Earth Link Direct Hi Power - Omni	Lander - Cruise	Descent Earth Link - Hi Power Omni to Impact	Vehicle Orientation to Surface	Surface Science Antenna Orientation	Lander Link Direct Continuous	Night, Serious Drilling	Monitor & Recharge Batt.	Surface TV
1. Communications												
• Earth Link				500		500			120			
• Data Storage & Processing	4	4	4	4	4	4	4	4	4	4	4	4
• Tape Recorder								5	2			5
• Commands and Tracking	8	8	8	9	8	9	8	6	6	6	6	6
• Power Conv. & Control	10	10	10	10	10	10	10	10	10	10	10	10
2. Guidance and Control												
• Attitude Control												
• Gyro and Electronics	9	9	9	9	9	9	9					
• Hi Gain Antenna Pointing												
• Body Mounted I.O. Camera & Electronics												
3. Thermal Control												
• Bus												
• Landers	15	15	15	15	15	15	15	15	15	15	15	15
4. Engineering Data												
• Bus												
• Landers	1	1	1	1	1	1	1	1	1	1	1	1
5. Science												
• Descent						41						
• Surface							30	88	6	220	6	31
Total Power Required	47	47	47	548	47	589	77	129	164	256	42	72
Duration	5 min	5 min	~5 hr	20 min	~1 hr	10 min	30 min	2 hr	~10 hr day	up to 1 hr	10 hr day	2 hr

However, system malfunctions can occur causing deviation from the programmed sequence. If the spacecraft cannot complete a programmed maneuver, the deviation is sensed by the spacecraft programmer and an automatic sun acquisition is initiated. If this succeeds, stored diagnostic information is transmitted through the omni-antenna using power from the solar cells. If it does not, battery energy is used to transmit vehicle state information through the omni-antenna. Batteries are sized for this "emergency" sequence on the basis of the power profile for this sequence. See Figure 2.6-3. Primary batteries are used to supplement the secondary battery sized for the orbit shadow portion of the orbiting phase power profile. The emergency power profile (Figure 2.6-3) is a suggested energy and spacecraft management scheme. The total stored energy requirements are based on the worst case of not acquiring solar illumination of the solar cells at the very end of the maneuver. The spacecraft will already have been operating on battery power for 46 minutes. Then the automatic diagnostic communication is initiated, while at the same time, the sun acquisition sequence is still being attempted by the spacecraft with full power being supplied to the gyros and attitude control subsystems. At the end of 20 minutes, the transmitter and the guidance and control subsystems are shut down, except for diagnostic gyros for vehicle rates. Command receivers and command demodulators are kept on until the battery is exhausted or until solar power is again available. If the automatic diagnostic communication is not commanded by the programmer, lack of this message or the normal end-of-maneuver message at the expected time is a signal for earth to send a command to the spacecraft to transmit maneuver history and vehicle state information. These powers are based on radiating 100 watts through the omni-antenna.

Although the duration of the standby portion of the suggested "emergency" power profile is shown as 4 hours, this does not mean the mission is lost because the primary battery has a capacity of 20 percent in excess of that shown on the profile, and the secondary battery still has available energy if 100 percent depth of discharge is utilized.

It is noted that the power for thermal control, listed in the power matrix (Table 2.6-3) during transit is shown at maximum expected value, but this power requirement is obviously a variable, dependent on the insolation which varies with the square of the spacecraft/sun distance. See Figure 2.6-4 for a graphical summary of the power profile.

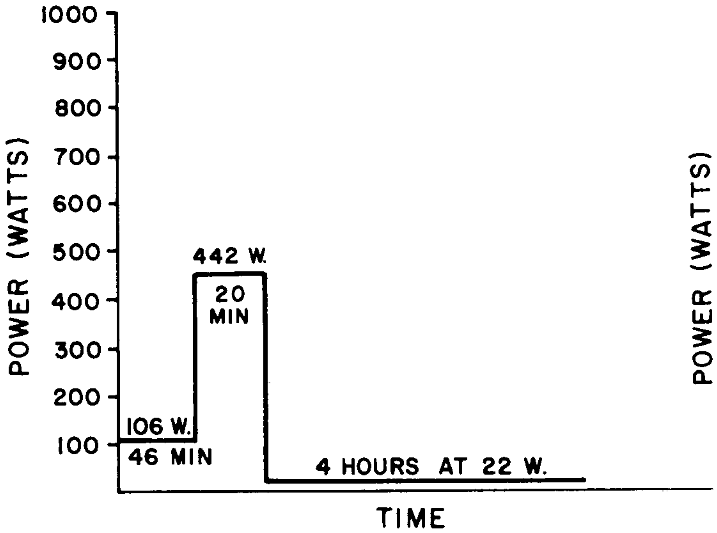


Figure 2.6-3. Emergency Power Requirements (No Solar Power)

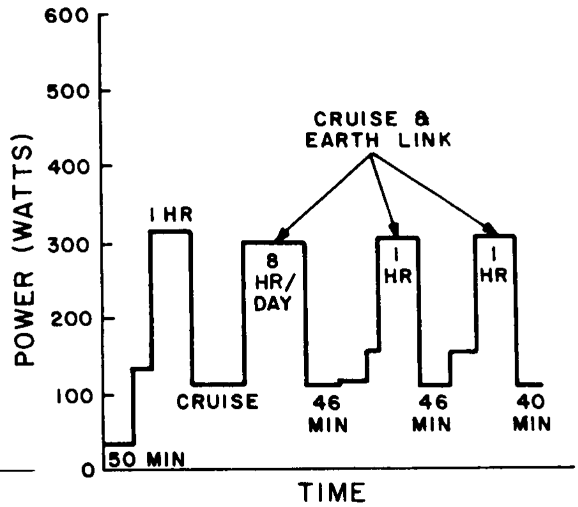


Figure 2.6-4. All-Orbiter Transit Phase Power Profile

As discussed in Section 2.5, Orbit Analysis and Selection, the favorable low eccentricity and period achievable by the Titan IIIC All-Orbiter system, requires that high data rate telemetry be continuously employed even while the television system is being used over the illuminated portion of the planet's surface. This requires peak power from the solar array of 545 watts. The net solar cell output is higher because of battery charging requirements when the Orbiter is passing through the maximum shadow of Mars.

The power matrix, Table 2.6-4, shows a maximum of 30 watts for orbit science, other than the television subsystem. If the maximum amount of the payload in the alternate All-Orbiter payload list is incorporated in the Orbiter, not enough power is available to turn all of them during each orbit. The 30 watts will allow each instrument to be used for one out of every three orbits, on a time-sharing basis.

TABLE 2.6-3. ALL-ORBITER (TRANSIT PHASE) POWER MATRIX

	Launch and Injection	Initial Orientation	Initial Orientation Earth Link to Verify Canopus	Cruise	Cruise & Earth Link	Midcourse Maneuvers and Reorientation	Cruise	Term. Guidance Obser.	Cruise and Earth Link	Final Trajectory Correction	Cruise and Term Guidance Obser.	Cruise and Earth Link
1. Communications												
• Earth Link			234		234				234			234
• Data Storage & Processing		4	4	4	4	4	4	4	4	4	4	4
• Tape Recorder								15	7		15	7
• Commands and Tracking		8	9	8	9	8	8	8	9	8	8	9
• Power Conv. & Control		10	10	10	10	10	10	10	10	10	10	10
2. Guidance and Control												
• Attitude Control		65	65	28	28	65	28	28	28	65	28	28
• Gyro and Electronics	17	17	17			17				17		
• Hi Gain Antenna Pointing			29		29 watts for 5 minutes once per day when used							
• Body Mounted I.O. Camera & Electronics								25			25	
3. Thermal Control		25	25	50	50		50	50	50		50	50
4. Engineering Data	2	2	2	2	2	2	2	2	2	2	2	2
5. Science												
• Transit				10	10		10	10	10		10	10
• Orbit												
Total Power Required	34	131	435	112	347	106	112	152	345	106	152	353
Duration	50 min	15 min	1 hr	Cruise	8 hr day	46 min	Cruise	15 min	1 hr	46 min	15 min	1 hr
Sun Available	No	No	Yes	Yes	Yes	No	Yes	Yes	Yes	No	Yes	Yes

TABLE 2.6-4. ALL-ORBITER ORBITING PHASE POWER MATRIX

	Orient for Orbit Injection and re-orient to Sun	Deploy PHP	Optimum Orbit: Orbit over Dark Surface and in Sunlight	Optimum Orbit: Orbit over Light Surface and in Sunlight	Worst Orbit: Orbit over Light Surface and in Sunlight	Worst Orbit: Orbiter in Shadow	Worst Orbit: Orbiter in Light and Over Dark Surface w/re-charge Battery
1. Communications							
· Earth Link			234	234	234		234
· Data Storage & Processing	4	4	4	4	4	4	4
· Tape Recorder			7	22	22	5	7
· Commands and Tracking	9	9	9	9	9	9	9
· Power Conversion & Control	10	10	10	10	10	10	10
2. Guidance and Control							
· Attitude Control	65	28	28	28	28	28	28
· Hi Gain Antenna Pointing		(29)			5 min	Once/Orbit	
· Gyro and Electronics	17						
· Body Mounted I.O Camera and Electronics							
· PHP Pointing		41	41	41	41	41	41
3. Thermal Control		25	25	25	25	25	25
4. Engineering Data	2	2	2	2	2	2	2
5. Science							
· Orbit			30	30	30	30	30
· Orbit TV				140	140		
Total Power Required	107	148	390	545	545	154	390
Duration	40 min	25 min	2.6 hr	1.7 hr	1.7 hr	0.7 hr	1.9 hr
Sun Available	No	Yes	Yes	Yes	Yes	No	Yes

Orbit Period = 4.3 hr

See Figure 2.6-5 for a graphical summary of the power profile.

2.6.3 ORBITER/LANDER

The transit mission for this combination system is the same as for the Saturn 1B Voyager spacecraft with the exception of having lower power requirements for the prime telemetry link. See the power matrix, Table 2.6-5, and the graphical summary of the power profile, Figure 2.6-6.

The communication requirements for maneuver diagnostic data reporting are similar to the All-Orbiter system discussed in Section 2.6.2. However, emergencies, due to loss of attitude control during an attempted correction maneuver, are somewhat less disastrous to the mission than in the All-Orbiter case because the Radioisotope Thermoelectric Generator, in the attached Lander, can be utilized to recharge the secondary batteries in both the Lander and Orbiter. Therefore, the spacecraft can be programmed to "stay alive" continuously in the event of loss of attitude control by cycling the diagnostic transmission to off periods to permit recharging the batteries.

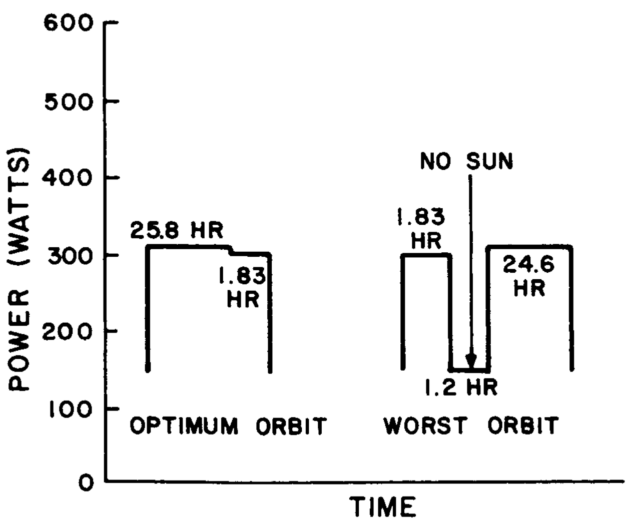


Figure 2.6-5. Orbiter/Lander Orbiting Phase Power Profile

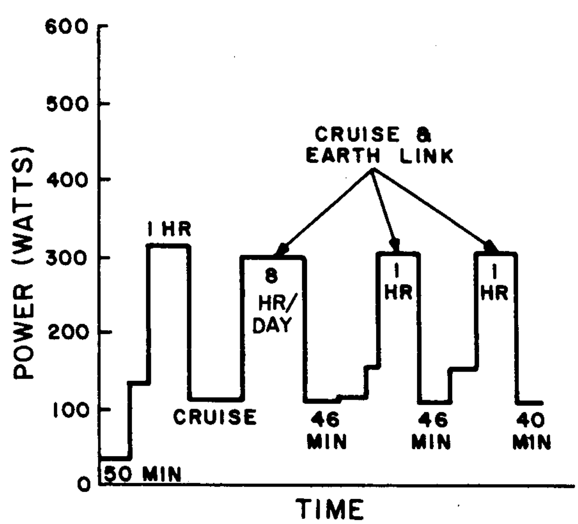


Figure 2.6-6. Orbiter/Lander Transit Phase Power Profile

TABLE 2.6-5. ORBITER/LANDER (TRANSIT PHASE) POWER MATRIX

	Launch and Injection	Initial Orientation	Initial Orientation Earth Link to Verify Canopus	Cruise	Cruise and Earth Link	Midcourse Maneuvers and Reorientation	Cruise	Term. Guidance Obser.	Cruise and Earth Link Xmit Term. Guid. Info.	Final Trajectory Correction	Cruise and Term. Guidance Obser.	Cruise and Earth Link Xmit Term. Guid. Info.	Orient for Lander Separation	Cruise and Earth Link
1. Communications														
· Earth Link														
· Data Storage & Processing		4	4	4	4	4	4	4	4	4	4	4	4	4
· Tape Recorder								15	7		15	7		
· Commands and Tracking		8	9	8	9	8	8	8	9	8	8	9	8	9
· Power Conv. & Control		10	10	10	10	10	10	10	10	10	10	10	10	10
2. Guidance and Control														
· Attitude Control		65	65	28	28	65	28	28	28	65	28	28	65	28
· Gyro and Electronics	17	17	17			17				17			17	
· Hi Gain Antenna Pointing				29 watts for 5 minutes once per day										
· Body Mounted I.O Camera & Electronics								25			25			
3. Thermal Control		25	25	50	50		50	50	50		50	50		50
4. Engineering Data	2	2	2	2	2	2	2	2	2	2	2	2	2	2
5. Science														
· Transit				10	10		10	10	10		10	10		10
· Orbit														
Total Power Required	34	131	312	112	295	106	112	152	302	106	152	302	106	295
Duration	50 min	15 min	1 hr	Cruise	8 hr day	46 min	Cruise	15 min	1 hr	46 min		1 hr	50 min	17 hrs
Sun Available	No	No	Yes	Yes	Yes	No	Yes	Yes	Yes	No	Yes	Yes	No	Yes

This operating mode would be available until the Lander is separated. A 5-pound primary battery can be utilized for post separation and orbit injection "emergencies" as in the All-Orbiter system.

In the orbiting mission phase, the prime telemetry mode is not utilized while the television cameras are employed during the small portion of the orbit over illuminated surface, and the data rate is lower than for the All-Orbiter system, resulting in a peak power demand of only 305 watts. See the power matrix, Table 2.6-6 and the power profile, Figure 2.6-7.

The combined system incorporates VHF Relay equipment in each module. The Orbiter will have a line-of-sight with the Lander within a favorable range of ~2,500 nautical mile during the daylight portion of the orbit, with the Orbiter near periapsis and over illuminated surface. During this mode, the solar array output of 360 watts is sufficient to power the TV system, the VHF receiver, and a tape recorder. The TV mapping need not be interrupted for relay link operation. Power for the TV system is 115 watts as compared with 140 watts for the TV system in the All-Orbiter.

The Lander of this system starts operating upon separation from the Orbiter. However, its RTG power supply, coolant pump, and programmer have been operating since launch. During transit the RTG is a standby power supply for the Orbiter until separation. Post-separation diagnostic telemetry, entry and descent engineering and scientific information are transmitted to the Orbiter by VHF relay equipment. A secondary battery in the Lander supplements the output of the 110-watt RTG for these requirements and for descent radar. A low-power continuous telemetry system is incorporated in this Lander to increase the amount of data returned to earth in the planned mission (with relay link operative) and in the case of no relay link. The size and weight of

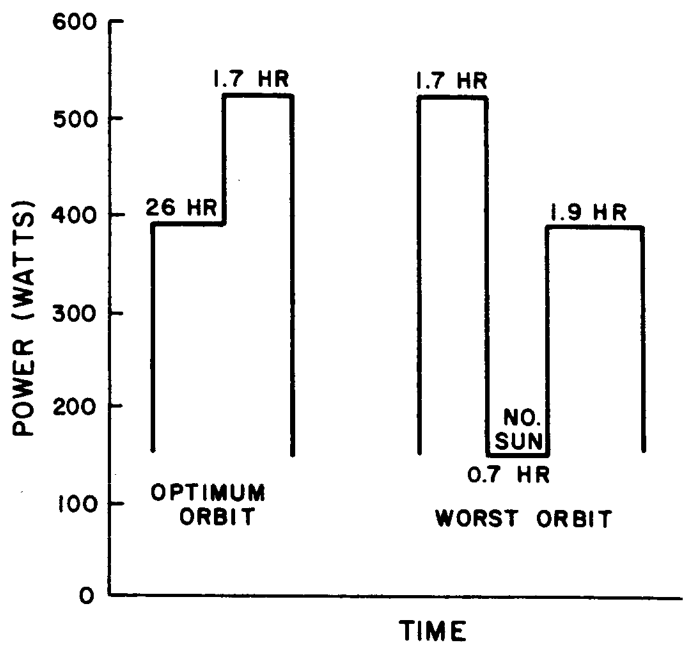


Figure 2.6-7. All-Orbiter Orbiting Phase Power Profile

TABLE 2.6-6. ORBITER OF ORBITER/LANDER COMBINATION (ORBITING PHASE) POWER MATRIX

			Optimum Orbit		Worst Orbit		
	Orient for Orbit Injection and re-orient to Sun	Deploy PHP	Orbit over Dark Surface and in Sunlight	Orbit over Light Surface and in Sunlight + Relay Receiver	Orbit over Light Surface and in Sunlight + Relay Receiver	Orbiter in Shadow	Orbiter in Light and Over Dark Surface w/Re-charge Battery
1. Communications							
· Earth Link			182				182
· Data Storage & Processing	4	4	4	4	4	4	4
· Tape Recorder			7	37	37	5	7
· Commands and Tracking	9	9	9	9	9	9	9
· Power Conversion & Control	10	10	10	10	10	10	10
· Relay Receiver & Demodulator				4	4		
2. Guidance and Control							
· Attitude Control	65	28	28	28	28	28	28
· Gyro and Electronics	17						
· Hi Gain Antenna Pointing		29			5 min	Once/Orbit	
· Body Mounted I.O Camera & Electronics							
· PHP Pointing		41	41	41	41	41	41
3. Thermal Control		25	25	25	25	25	25
4. Engineering Data	2	2	2	2	2	2	2
5. Science							
· Orbit			22	22	22	22	22
· Orbit TV				115	115		
Total Power Required	107	148	305	297	297	146	305
Duration	40 min	25 min	25.8 hr	1.83 hr	1.83 hr	1.2 hr	24.6 hr
Sun Available	No	Yes	Yes	Yes	Yes	No	Yes

Orbit Period = 27.6 hr

this Lander compelled the elimination of the petrographic TV microscope package from the payload, and the reduction of the direct link data to one-half of that in the large Lander. Thus the direct link requires 60 watts instead of 120 watts and the RTG output at the load is 110 watts instead of 170 watts. See power matrix, Table 2.6-7.

Relay link operation requires interruption of operation of the direct link to earth whenever the earth and Orbiter line-of-sight coincide. Peak power demands of other payload operation are not critical in this small Lander (no Drill) and the secondary battery used for descent is adequate.

TABLE 2.6-7. LANDER OF ORBITER/LANDER COMBINATION (SEPARATE PHASE) POWER MATRIX

	Lander Separation	Lander Spin Up, ΔV	Lander - Spinning - Cruising	Relay Link to Orbiter	Lander - Cruise	Descent Relay Link to Orbiter	Vehicle Orientation to Surface	Surface Science & Antenna Deployment	Direct Continuous Communication to Earth	Night, Scientific Inst. Peak Power	Monitor, Scientific Inst. and Recharge Battery	Surface, Record TV
1. Communications												
• Earth Link				100		100			60			
• Data Storage & Processing	4	4	4	4	4	4	4	4	4	4	4	4
• Tape Recorder								5	2			5
• Commands and Tracking	8	8	8	9	8	9	8	6	6	6	6	6
• Power Conv. & Control	10	10	10	10	10	10	10	10	10	10	10	10
2. Guidance and Control												
• Attitude Control												
• Gyro and Electronics	9	9	9	9	9	9	9					
• Hi-Gain Antenna Pointing												
• Body Mounted I.O. Camera & Electronics												
3. Thermal Control												
• Bus												
• Landers	15	15	15	15	15	15	15	15	15	15	15	15
4. Engineering Data												
• Bus												
• Landers	1	1	1	1	1	1	1	1	1	1	1	1
5. Science												
• Descent						41						
• Surface							30	88	6	100	6	31
Total Power Required	47	47	47	148	47	189	77	129	104	136	42	72
Duration	5 min	5 min	~5 hr	20 min	~1 hr	10 min	30 min	2 hr	~10 hr day	up to 1 hr	10 hr day	2 hr

2.7 RELIABILITY AND VALUE ANALYSIS

2.7.1 GENERAL

The Titan IIIC Study supplements the study completed in October, 1963, and provides comparative data for use in trade-off analyses and system configuration selections. Reliability analyses have been carried out to part and component level on the basis of the best values obtainable under high reliability program controls at the present state of the art.

Many components and subsystem elements are provided in redundant or switchover, back-up combinations to obtain a well balanced, optimal system. Further improvements in reliability can be made by specific component developments, by the further use of redundancy or by materials and part research and development.

It should be noted that the major design improvements or system configurations affecting the reliability of the Titan IIIC-Voyager system are also applicable to the improvement of the Saturn 1B-Voyager reliability estimates. The principal difference results from the use which is made of the larger payload capability of the Titan IIIC system when applied in multiple launch opportunities.

2.7.2 SCIENTIFIC VALUE ASSIGNMENT

The values which have been applied to the various scientific payload items in the conduct of this study are fully consistent with those used in the Saturn 1B-Voyager study. Where these instruments are the same (and it has been a requirement of this study that they be so wherever possible) and where their deployment and use were directly comparable, the same relative point number was used to indicate their contribution to the overall scientific value of the mission.

In those instances in which the Titan IIIC study made possible the consideration of new or unique configurations or deployments, as in the case of the controlled, roving instrument carrier, values consistent with those used for instrument and site locations in the earlier study were assigned for the purpose of providing a method of comparison.

Details of these scientific value assignments are provided in Section 5.3 of this Document.

2.7.3 RELIABILITY ANALYSIS

This section contains reliability analyses of the three major spacecraft systems investigated during this study. These spacecraft systems are required to have the capability or orbiting or landing, or both, on Mars during the time period of 1969-75.

The three spacecraft systems are:

1. Bus/Lander system
2. Orbiter system
3. Orbiter/Lander system.

The results of these system reliability analyses have been summarized and presented in Table 2.7-1.

TABLE 2.7-1. MARS 1971 RELIABILITY SUMMARY

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

Any number of missions and mission-profile variables can be applied to these various requirements and capabilities, since the mission times are distributed according to selected launch dates. The mission transit period of 225 days for the 1971 launches to Mars is used throughout this reliability analysis to present an indication of the probability of success than can be expected with present day part and design technology.

The system reliability analyses represent the summation and interpretation of the many subsystem and component analyses including the effects of their individual operating times, environments, and the effects of backup modes and redundancies incorporated in the system design as a result of the failure effects analyses.

In order to develop a basic reliability analysis which is adaptable to any particular mission, the individual analyses are prepared relative to mission phases. Two principal "cut-off" or evaluation points in the mission cycle are taken to be at 100 hours and at 3 months after the end of the transit period.

To insure the success of each system, various features have been incorporated into the design of the vehicle to maintain total uninterrupted or degraded operation in the event of partial or complete component failure.

Three methods are employed to sustain operational continuity: 1) complete redundancy of components, 2) internal circuit redundancy (majority logic), or 3) programming of alternate functional loops (stand-by redundancy). The definition of these features and their areas of use are shown in Table 2.7-2.

Mathematical models are used to describe the contribution of each functional equipment's success probability relative to the overall system success probability. The relative values employed in the model for the functional elements are based on the variables associated with a particular mission use of the equipment.

TABLE 2.7-2. METHODS EMPLOYED TO SUSTAIN OPERATIONAL CONTINUITY

Complete Redundancy	Redundancy (Majority Logic)	Stand-By Redundancy
Pitch, Yaw and Roll Amplifiers	Command and Computer Equipment (Comm.) Data Processor (Comm.) Storage and Logic Unit (G&C)	Star Trackers Earth Trackers Hot Gas System

All models used in this analysis are based on the assumption of the exponential distribution since the failure distributions of electronic equipment in the time domain generally exhibit the characteristics of this distribution:

All reliability values were estimated from:

1. Duty cycle of individual components in the mission.
2. Estimated parts complexity of each component.
3. Thermal control to maintain ambient part case temperatures.
4. Partial or complete redundancy, where applied.

During the Voyager mission, all equipments are either fully energized, cycled, or in the "off" state, according to the sequence in which their function is required. Thus, the parts and circuits within the equipment are subjected to various degrees of stress, relative to the operational state they are in. Recognizing that the lifetime of a part is a function of the stress level and the interval of the applied stress, modifying factors are employed herein to account for the operational state of the parts throughout the mission. The "effective time" used in the following reliability tables and calculations is derived from the actual operating time and the use of modifying factors where required.

A. BUS/LANDER SYSTEM

1. System Definition

The Voyager Bus/Lander System is required to have the capability of transporting a vehicle to Mars and landing it in such a manner as to have it function properly in the scientific investigation of the planetary surface and atmosphere.

The bus is designed primarily for the transportation of the Lander vehicle and for Lander orientation to a predetermined impact trajectory.

The Lander is designed to acquire information about the space environment prior to impact on Mars and subsequent scientific investigations after impact. This acquired data is recorded and periodically communicated to earth. The Lander will have the capability for six months of operation after impact, although the majority of the required scientific data can be obtained in a shorter time interval.

2. Reliability Analysis

Earth communication is maintained by the Lander for command reception and data acquisition and transmission.

Many modes of Bus/Lander communications are provided for particular time phases in the mission for Bus/Lander vehicle to Earth. (See Table 2.7-3.)

The system model for the Voyager Mars 1971 mission shows the Bus in operation only during the launch and transit phases to the point of separation from the Lander. Some of the Lander subsystems are contributory during the transit phase.

For a more detailed examination and breakdown of this system and its subsystems, see Section 5.2.

3. Mathematical Model and Reliability Computations

The probability of success of the Bus/Lander system for the 1971 Mars Mission is given by the following Mathematical Model.

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

where

$$R\ (\text{Bus}) = R\ (\text{Communications}) \cdot R\ (\text{Guidance and Control}) \cdot R\ (\text{Hot Gas Propulsion}) \cdot R\ (\text{Cold Gas Propulsion})$$

and

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

Using the Bus subsystem reliability values tabulated in Table 2.7-4 gives

$$(\text{Transit})\ R\ (\text{Bus}) = (0.999)\ (0.920)\ (0.999)\ (0.997) = 0.915$$

Using the Lander subsystem reliability values tabulated in Table 2.7-4, gives

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

TABLE 2.7-3. MODES OF BUS/LANDER COMMUNICATION

Phase	Mode	Primary Loop	Back-up
1. Transit 10-2880 Hours	Vehicle to Earth	Omni	Hi-Gain
2. Transit 2880 to separation	Vehicle to Earth	Hi-Gain	Omni
3. Pre-entry and Descent	Lander to Earth	VHF Omni	
4. Surface Operation	Lander to Earth	Hi-Gain	Omni (Degraded Operation)

TABLE 2.7-4. 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 and Control	0.920	EP & D	0.970	0.959
Hot Gas Propulsion	0.999	Propulsion and Separation	0.972	0.972
Cold Gas Propulsion	0.997	Thermal Control	0.957	0.947
		Retardation	0.984	0.984
		Orientation	0.993	0.993
Bus Vehicle Reliability	0.915	Lander Vehicle Reliability	0.760	0.704

$$(3\ \text{Months})\quad R\,(\text{Lander}) = (0.815)\,(0.959)\,(0.972)\,(0.947)\,(0.984)\,(0.993)$$

$$= 0.704$$

Entering the above values for the reliability of the Bus and the Lander into the equation for the reliability of the complete Bus/Lander system gives

$$(100\ \text{Hours})\quad R\,(\text{System}) = (0.915)\,(0.760)$$

$$= 0.696$$

$$(3\ \text{Months})\quad R\,(\text{System}) = (0.915)\,(0.704)$$

$$= 0.645$$

For a summary of the Bus/Lander system reliability, see Table 2.7-4.

B. ORBITER SYSTEM

1. System Definition

The Voyager Orbiter system is composed of a single vehicle with the capability of orbiting Mars for a three-month time period during which it will acquire scientific information about the Martian atmosphere and the space environment. Information about the space environment is also acquired during transit. The Orbiter contains data conversion and storage capability, plus command reception and data transmission.

2. Reliability Analysis

The Orbiter vehicle in this system contains five major functional subsystems. The Communications subsystem has only two modes of communications, the omni link which is mainly used for the first 120 days of transit and the high gain link which is used after the 120 days transit point. The guidance and control subsystem contains a 3-axis PHP in place of the 2-axis PHP used in the previous Voyager-Saturn 1B Study.

For a more detailed examination and breakdown of this system and its subsystems, see Section 5.2.

3. Mathematical Model and Reliability Computations

Since this system comprises only the Orbiter vehicle, the mathematical model of the system and the probability of success of the mission would be the reliability of the Orbiter vehicle which is

$$\begin{aligned} R\text{ (System)} = R\text{ (Orbiter)} = {} & R\text{ (Communications)} \cdot R\text{ (Guidance and Control)} \\ & \cdot R\text{ (Power Supply)} \cdot R\text{ (Hot Gas Propulsion)} \\ & \cdot R\text{ (Cold Gas Propulsion)} \end{aligned}$$

Substituting computed reliability values in the above equation gives

$$\begin{aligned} \text{(100 Hours) } R\text{ (System)} &= (0.866)\,(0.912)\,(0.980)\,(0.998)\,(0.996) \\ &= 0.768 \\ \text{(3 Months) } R\text{ (System)} &= (0.793)\,(0.831)\,(0.973)\,(0.998)\,(0.990) \\ &= 0.633 \end{aligned}$$

This is summarized in Table 2.7-5.

TABLE 2.7-5. RELIABILITY SUMMARY FOR ORBITER SYSTEM

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

C. ORBITER/LANDER SYSTEM

1. System Definition

The Voyager Orbiter/Lander System is required to have the capability for both orbiting a vehicle around Mars and landing another vehicle on the surface of Mars. The Orbiter has multiple functions in the mission. During the transit phase, it is the earth vehicle communications link, performs maneuvers, and transmits diagnostic data. In the orbiting phase, it acquires and transmits scientific information to earth, maintains back-up communication relay with the Lander and exercises stabilization and control of the vehicle.

The Orbiter is designed to acquire information about the space environment during transit and in its orbiting interval in the same manner and to the same degree required of the Orbiter System described in Section 2.7.3(B). The Lander is designed to acquire information about the space environment prior to impact on Mars and subsequent scientific investigations after impact in the same manner and degree required of the Lander in the Bus/Lander system described in Section 2.7.3(A). This acquired data is recorded and periodically communicated to earth.

2. Reliability Analysis

Many modes of Orbiter/Lander communications are provided for particular time phases in the mission for Earth to Orbiter and/or Lander links. (See Table 2.7-6.)

The Orbiter portion of the Voyager vehicle is the only communication link during transit for commands and data transmission. Equipment within the orbiter is energized periodically by command or pre-programming for maneuvers or diagnostic data transmission.

2.7.4 ATTAINABLE MISSION VALUES

A. GENERAL

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

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

The product of the mission values available from a particular lander or orbiter complement of scientific instruments multiplied by the probability of its successful completion of the mission is a measure of the mission value most likely to be attained. This value for a single launch is less than 100 percent of the basic unit mission value defined above. Where more than one launch is involved, and thus the possibility of more than one successful orbiter and more than one successful lander with different orbits and different landing sites is involved, the values attainable exceed those available from a single launch.

Thus, in multiple launches, more than 100 percent of a single basic unit mission value is attainable. And, the attainable mission values in the various figures and tables correspondingly show figures of greater than 100 percent where more than one system (or more than one Lander) is launched.

1. Attainable Mission Effectiveness

The approach used to evaluate multiple launch opportunities is discussed below. It is considered general in applicability and is included here to provide additional information on the approach and application made of the Attainable Mission Value concept during this study.

a. Attainable Mission Value

V_x = Value per experiment

(R_x) = Product summation of all reliability factors* upon which the success of this experiment depends.

$$(R_x) = (R_1 \times R_2 \times R_3 \times \text{----} \; R_N)x$$

$(R_x)\ V_x$ = (AMV) = The attainable mission value using the system specified.

b. Attainable Mission Value/Launch

Attainable Mission Value/Launch = The arithmetic sum of the attainable mission values for each experiment carried.

$$(R_x)_1\ V_1 + (R_x)_2\ V_2 + \text{----}\ (R_x)_N\ V_N = \text{AMV}$$

c. Attainable Mission Effectiveness

Attainable Mission Effectiveness = Attainable mission value divided by the sum of all costs** directly applicable to the mission(s) specified.

d. Probability of Success for Missions Involving Multiple Launches

1. Attainable Mission Value is equal to the sum of the probability of exactly one success multiplied by the value of one mission, plus the probability of exactly two successes multiplied by the value of two missions . . . , plus the probability of exactly N successes multiplied by the value of N missions.

   Example: 5 Launches — Mission Reliability per Launch of 0.75

Number of Successes	Probability of Success	Mission Value	AMV
EXACTLY 0	0.00098	0	0.000
EXACTLY 1	0.015	1	0.015
EXACTLY 2	0.088	2	0.176
EXACTLY 3	0.264	3	0.792
EXACTLY 4	0.396	4	1.584
EXACTLY 5	0.237	5	1.185
Total	1.000		3.752

*For this purpose---the reliability values at 50 percent confidence are used as having the highest probability of being true for any given flight.

**This includes capital investment amortization, ground support and logistic support costs, etc., as well as the more obvious R&D, systems equipment and operational costs applicable to a given launch or series of launches.

TABLE 2.7-6. MODES OF ORBITER/LANDER COMMUNICATIONS

Phase	Mode	Primary Loop	Back-Up
Transit 10-2880 Hours	Vehicle - Earth	Omni	High-Gain
Transit 2880 Hours to Separation	Vehicle - Earth	High-Gain	Omni
Orbit	Orbiter - Earth Orbiter - Lander	High-Gain VHF Yagi	Omni
Separation to Impact	Orbiter - Lander	VHF Omni	
Surface Operation	Lander - Earth	High-Gain	Lander-Orbiter $\begin{pmatrix}\text{VHF}\\ \text{Omni}\end{pmatrix}$

For a more detailed examination and breakdown of this system and its subsystems, see Section 5.2.

3. Mathematical Model and Reliability Computations

The probability of success of the Orbiter/Lander system for the 1971 Mars mission is given by the following Mathematical Model.

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

where

$$R\text{ (Orbiter)} = R\text{ (Communications)} \cdot R\text{ (Guidance and Control)} \cdot R\text{ (Power Supply)} \cdot R\text{ (Hot Gas Propulsion)} \cdot R\text{ (Cold Gas Propulsion)}$$

and

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

Using the Orbiter subsystem reliability values tabulated in Table 2.7-7, gives

$$\text{(100 Hours) } R\text{ (Orbiter)} = (0.856)\ (0.912)\ (0.980)\ (0.998)\ (0.996) = 0.758$$

TABLE 2.7-7. RELIABILITY SUMMARY FOR THE ORBITER/LANDER SYSTEM

Orbiter			Lander		
Subsystem	Reliability		Subsystem	Reliability	
	100 Hours	3 Months		100 Hours	3 Months
Communications	0.856	0.742	Communications	0.989	0.952
Guidance and Control	0.912	0.831	EP & D	0.970	0.959
Power Supply	0.980	0.973	Propulsion and Separation	0.972	0.972
Hot Gas Propulsion	0.998	0.998	Thermal Control	0.957	0.957
Cold Gas Propulsion	0.996	0.990	Retardation	0.984	0.984
			Orientation	0.993	0.993
Orbiter Vehicle Reliability	0.758	0.587	Lander Vehicle Reliability	0.872	0.822

$$\text{(3 Months) } R\text{ (Orbiter)} = (0.742)(0.831)(0.973)(0.998)(0.990) = 0.587$$

Using the Lander subsystem reliability values tabulated in Table 2.7-7, gives

$$\text{(100 Hours) } R\text{ (Lander)} = (0.989)(0.970)(0.972)(0.957)(0.984)(0.993) = 0.872$$

$$\text{(3 Months) } R\text{ (Lander)} = (0.952)(0.959)(0.972)(0.957)(0.984)(0.993) = 0.822$$

Entering the above values for the reliability of the Orbiter and the Lander into the equation for the reliability of the complete Orbiter/Lander system gives

$$\text{(100 Hours) } R\text{ (System)} = (0.758)(0.872) = 0.661$$

$$\text{(3 Months) } R\text{ (System)} = (0.587)(0.822) = 0.482$$

For a summary of the Orbiter/Lander system reliability, see Table 2.7-7.

Example: 5 Launches — Mission Reliability per Launch of 0.60

Number of Successes	Probability of Success	Mission Value	AMV
EXACTLY 0	0.010	0	0.000
EXACTLY 1	0.077	1	0.077
EXACTLY 2	0.230	2	0.460
EXACTLY 3	0.346	3	1.038
EXACTLY 4	0.259	4	1.036
EXACTLY 5	0.078	5	0.390
Total	1.000		3.001

2. Attainable Mission Value can also be equal to the sum of the probability of at least one success multiplied by the value of the first successful mission, plus the probability of at least two successes multiplied by the value of the second successful mission, plus the probability of at least N successes multiplied by the value of the Nth successful mission.

Example: 5 Launches — Mission Reliability per Launch of 0.60

Number of Successes	Probability of Success	Mission Value	AMV
AT LEAST 1	0.990	1.0	0.990
AT LEAST 2	0.913	0.9	0.822
AT LEAST 3	0.683	0.8	0.557
AT LEAST 4	0.337	0.7	0.233
AT LEAST 5	0.078	0.6	0.047

B. BUS/LANDER

The larger payload capability of the Titan IIIC single lander has been evaluated for the alternative of using all the additional payload capability for reliability improvement. This was evaluated under the constraint of improvement via redundancy rather than assuming any specific or general reliability changes in the state of the art. The use of high reliability parts, materials and process controls and the full implementation of reliability programs in accordance with NASA documents has already been considered in preparing the estimates for the basic systems.

The attainable mission values have been developed using the methods outlined and illustrated in Section 2.7.4(A)(1). The reliability values applied for the Bus/Lander are shown in Table 2.7-8.

C. ORBITER

The method and approach is the same as noted above. The reliability values for the orbiter are provided in Table 2.7-9.

TABLE 2.7-8. RELIABILITY SUMMARY

Lander	Saturn 1B Orbiter/Dual Lander	Titan IIIC Bus/Lander
Launch and Transit		
Communication	99.0%	86.6%
Elec. Power and Distr.	96.3	97.1
Propulsion and Sep.	98.2	98.6
Thermal Control	94.6	95.8
Retardation	99.9	99.9
Orientation	99.9	99.9
Communication		99.9%
Guidance and Control		92.0
Hot Gas		99.9
Cold Gas		99.7
Total Bus		91.5%
Total Lander* (each)	88.3%	72.5%
Separation, Impact and 1st 100 Hours	95.9	96.0
Total Lander (each)	84.7%	69.6%
3-Month Mission --2200 Hours	90.5	88.9
Total Lander (each)	80.3%	64.5%

*Saturn Lander is dependant upon Orbiter

TABLE 2.7-9. RELIABILITY SUMMARY

Orbiter	Saturn 1B Lander/Orbiter	Titan IIIC All-Orbiter
Launch and Transit		
Communication	85.1%	86.9%
Guidance and Control	90.9	91.8
Power Supply	98.5	98.1
Hot Gas	99.9	99.9
Cold Gas	99.7	99.7
Total Orbiter	76.8%	77.9%
Separation and 1st 100 Hours	98.6	98.6
Total Orbiter	75.7%	76.8%
3-Month Mission --2200 Hours	81.0	81.2
Total Orbiter	62.2%	63.3%

D. ORBITER/LANDER

Various system configurations have been analysed. The one of principal importance is that of a single lander or a single orbiter per Titan IIIC booster. The reliabilities of the orbiter and lander are combined with the reliabilities of the booster, the terrain and scientific instrument as shown in Table 2.7-10.

With various booster reliabilities, the probability sum applicable to various combinations is shown in Table 2.7-11. At the bottom of this table, a sample calculation of Attainable Mission Value (AMV) is provided for the 2 Lander + 2 Orbiter combination.

TABLE 2.7-10. SYSTEM RELIABILITY -- SINGLE LAUNCH
(Launch -- Through 100 Hours After Arrival)

	Saturn 1B		Titan IIIC
Lander "Surface" Data			
Martian Terrain Suitability	90.%		90.%
Lander Reliability	84.7		76. *
Lander Instrument Reliability	96.5		96.5
Orbiter through Transit	76.8	Bus	91.5
Orbiter during 1st 100 Hours	98.6	Incl. in	--
Booster	80.0	Lander	80.0
Subsystem	44.6%		48.3%
Lander "Entry" Data			
Lander through Entry	88.3		79.2
Lander Instrument	99.5		99.5
Orbiter into Orbit	76.8	Bus	91.5
Booster	80.0		80.0
Subsystem	54.0%		57.7%
Orbiter Data			
Orbiter through 100 Hours	75.7		76.8
Orbiter Instrument	96.5		96.5
Booster	80.0		80.0
Subsystem	58.4%		59.2%
Effective Single System Reliability	49.7%		52.5%

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

*Lander through Transit = 79.2%, and 79.2% x 96.0% = 76%

TABLE 2.7-11. RELIABILITY OF LAUNCH VEHICLES VERSUS ATTAINABLE MISSION VALUES

Booster Reliability	70%	80%	90%
Orbiter			
Probability of	51.9	59.3	66.7
Success: 1 of 2 ---	76.9	83.5	88.8
(At Least) 2 of 2 ---	26.8	35.0	44.2
Probability Sum	103.7	118.5	133.0

NOTE:

7/8 x 35 = 30.6

7/8 x 118.5 = 103.7

Lander "Surface"			
Probability of	42.3	48.3	55.4
Success: 1 of 2	66.5	73.3	80.1
(At Least) 2 of 2	17.7	23.2	30.5
Probability Sum (2)	84.2	96.5	110.6
Probability Sum (3)	127.0	145.0	163.0
Probability Sum (4)	168.7	203.1	217.2
Lander "Entry"			
Probability of Success	50.5	57.7	64.9
At Least: 1 of 2	75.5	82.2	87.7
2 of 2	25.3	33.1	41.9
At Least: 1 of 3	87.8	92.4	95.7
2 of 3	51.0	61.6	71.6
3 of 3	12.8	19.2	27.4

Attainable Mission Value (Example at 80%)

2 Orbiters------	118.5% x	54% V.	=	64.0% AMV
2 Landers "Surface"--	96.5% x	138.5% V.	=	133.6% AMV
Landers --"Entry"---	82.2% x	10.0% V.	=	8.2% AMV
	33.1% x	5.0% V.	=	1.7% AMV
				207.5% AMV

The results of these analyses are provided in Figures 2.7-1 through 2.7-4. Figure 2.7-1 illustrates the mission values attainable using the Titan IIIC-Voyager systems recommended by this study for the 1971 opportunity. This system configuration includes a small, controlled, roving instrument carrier in the large lander and the use of high resolutions, one meter resolution mapping capability as well as an upper atmospheric sampling capability in a sterilized orbiter as the most valuable use of most of

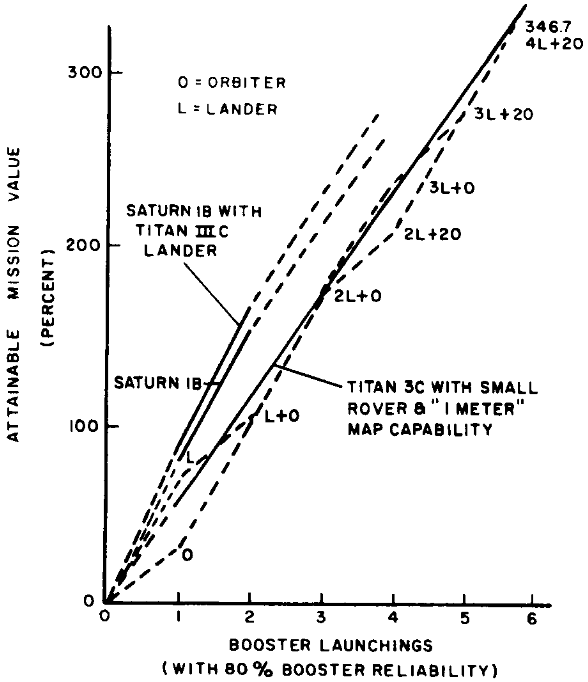


Figure 2.7-1. Attainable Mission Value

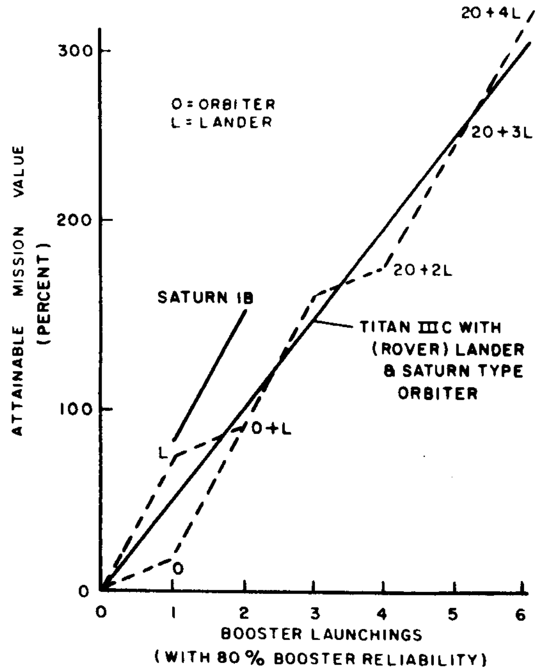


Figure 2.7-2. Attainable Mission Value

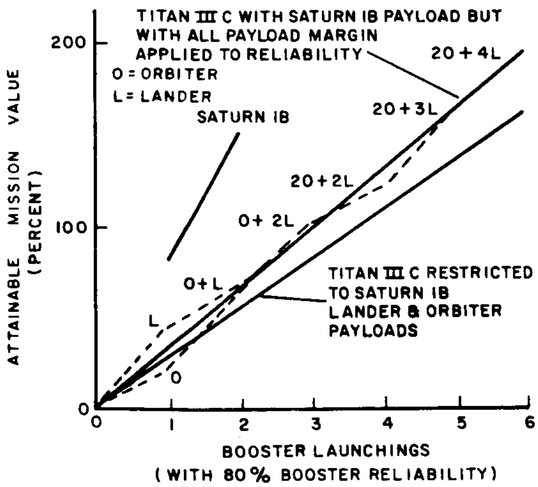


Figure 2.7-3. Attainable Mission Value

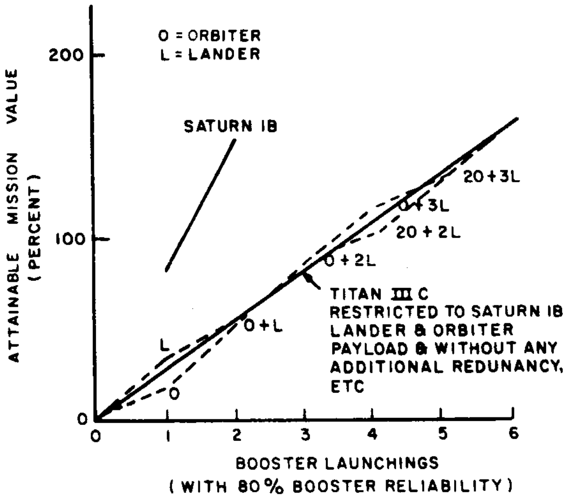


Figure 2.7-4. Attainable Mission Value

their extra payload carrying capabilities. A significant portion of the extra payload remains available for some further improvement in reliability. The values obtainable using the Saturn 1B and the same large single lander are also provided for comparison.

Figure 2.7-2 illustrates a similar system but one in which sterilization of the orbiter is not required and in which the high resolution mapping and upper atmospheric data values are not obtained.

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

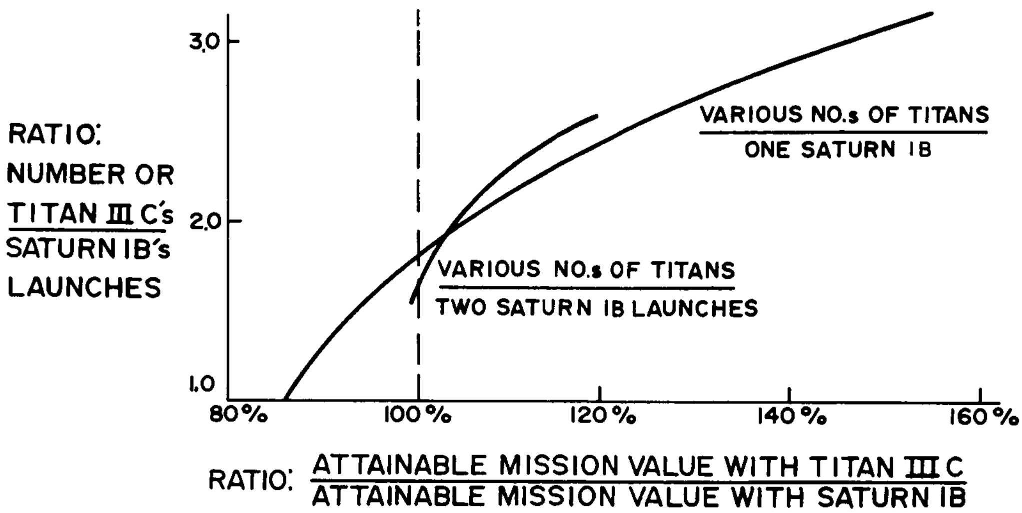
The summary data presented in this section is considered representative and illustrative both of the best estimates and of the range of values involved in comparing Titan IIIC-Voyager capabilities with those of the Saturn 1B-Voyager study. Additional detail is provided in Section 5.3.

E. "SINGLE" SUCCESS CRITERIA

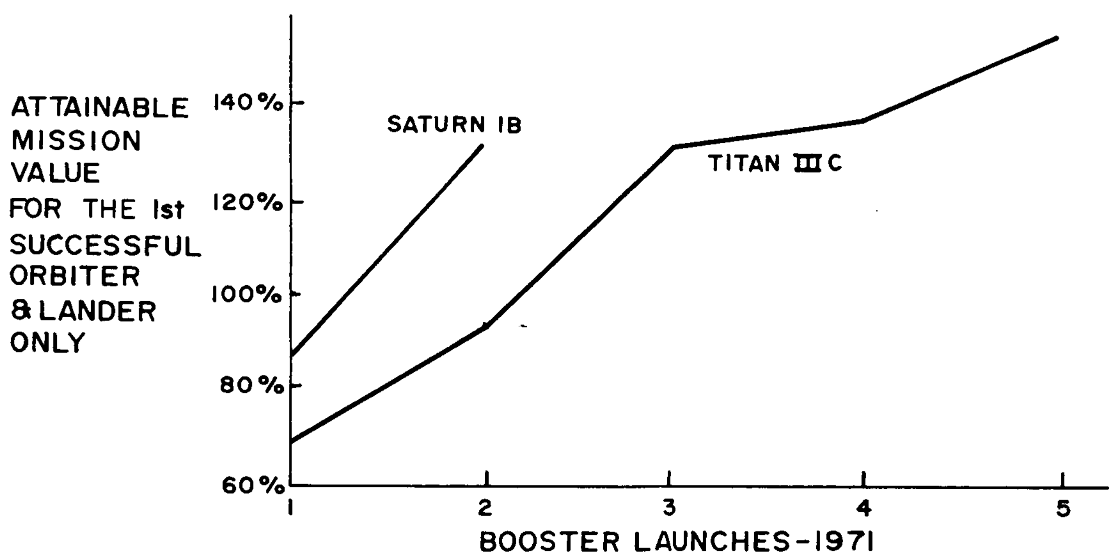
The Attainable Mission Values summarized above are the total, cumulative values resulting from a consideration of all chances for success and the values attached to each such success. These are considered the best criteria for decision since all factors are included and because specific alternatives are evaluated to the greatest practicable degree before final comparisons and judgements are made.

It was also felt pertinent to consider the first "single" success of an Orbiter/Lander combination as a supplementary decision criteria. Since the Orbiter/Lander combination designed for both to be launched with a single Titan IIIC booster resulted in a very reduced lander payload capability, it was not included. The same combinations of multiple launchings, as are shown in Figure 2.7-1 through 2.7-4, were summarized and compared. The results are shown in Figure 2.7-5.

These indicate equivalence at approximately 1.8 Titan IIIC per Saturn 1B IF the Saturn 1B subsystems are updated from the 1969 designs (reported in October 1963) so as to result in subsystems of equal reliability to those of the Titan IIIC system and if the Titan IIIC large lander and rover are included on both the Titan IIIC and Saturn 1B as indicated in Figure 2.7-4. A comparison is made for the 1971 opportunity in both



FOR VARIOUS LAUNCH COMBINATIONS. ONLY THE AMV'S FOR THE 1st SUCCESSFUL LANDER AN THE 1st SUCCESSFUL ORBITER ARE.INCLUDED. (BOTH WITH SINGLE LANDERS & ROVERS)



(WITH 80% BOOSTER RELIABILITY & WITH LARGE SINGLE LANDER & ROVER ON SATURN IB's AS WELL AS TITAN III C's)

Figure 2.7-5. Single-Success Criteria

cases. In this comparison, the effects of the 192 day transit time for Titan IIIC landers in 1971 are included. Saturn 1B Orbiter/Lander and Titan IIIC Orbiter transit times were included at 225 days.

If no advantage is taken (either for reliability improvement or for payload improvement) of the increased payload capability of the Titan IIIC, equivalence occurs at 2.1 Titan IIIC per Saturn 1B.

F. REMARKS

It may be noted that a comparison has been made of the 1969 Saturn 1B system with the 1971 Titan IIIC system during this study. This has been done to provide continuity in accordance with the guide lines received for this study contract. The magnitudes of the differences in reliability between corresponding subsystems of these two systems (once technological updating of the Saturn 1B has been made) results almost entirely from the differences in transit times.

2.8 APPLICABILITY TO MARS 1969

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

Table 2.8-1 shows the possibility of utilizing the same Orbiter design for both 1969 and 1971. Payload and subsystem simplifications could be effected without altering the basic similarity of the two systems.

The Lander could have many variations in size and payload. It is assumed that the Mariner 1969 mission would have a more conservative design which would permit unrestricted entry corridors into the 11 mb atmosphere. Wide entry corridors would reduce the required guidance accuracy and dependence upon sophisticated terminal guidance and approach correction techniques. Payload of such Landers would be minimal with the emphasis being upon the determination of atmospheric characteristics and of basic life detection experiments. Table 2.8-2 shows two possibilities of using variations of the basic Lander design for both 1969 and 1971. The first system uses the 1971 Lander design with the gross payload reduced to 367 pounds. This reduces the W/C_DA sufficiently to increase the acceptable entry corridor to a range of 20-60 degrees in the 11 mb atmosphere. The second system uses the 1971 Lander design with the gross payload reduced to 222 pounds and modification to the retardation and structural subsystems sufficient to reduce the entry weight to 1094 pounds. This would permit unrestricted entry into the 11 mb atmosphere.

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

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

TABLE 2.8-1. ADAPTABILITY TO MARS 1969

(ORBITER)

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

TABLE 2.8-2. ADAPTABILITY TO MARS 1969

(BUS/LANDER)

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

3. SPACECRAFT SYSTEM DESIGN STUDIES

3.1 SUMMARY

Three systems were considered for Mars 1971 using the Titan IIIC launch vehicle. These were:

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

The Bus/Lander system that evolved during the study considered the use of two possible Landers, a 134-inch base diameter Lander and a 110-inch base diameter Lander with extensible flaps. A Bus with one Lander was selected as the most optimum arrangement. Table 3.1-1 shows in condensed form the weight breakdown and payload capability of the Bus/Lander system. Section 3.2 gives detailed information concerning the Bus/Lander system.

The All Orbiter system was designed to permit the largest scientific payload possible consistent with required reliability. Table 3.1-2 indicates the weights and payload obtainable. Figure 3.3-1 shows the Orbiter during transit to Mars with the high-gain antenna and solar panels deployed.

The design of the Orbiter is such that variations in experiments and payload may be easily accommodated for other years and missions. Section 3.3 gives the detailed information concerning the Orbiter system.

The Orbiter/Lander system selected for Mars 1971 has an Orbiter with 123 pounds of payload and a 1284 pound Lander. However, with a reduction in injected weight for later years the system shows a marked decline in capability. Table 3.1-3 shows the weight allotment for the Orbiter/Lander system for Mars 1971.

TABLE 3.1-1. WEIGHT BREAKDOWN AND PAYLOAD CAPABILITY OF THE BUS/LANDER SYSTEM

		Bus/Lander Capability
Bus		455
Lander		2042
Entry Weight	1830	
Scientific Payload	387	
Fuel		49
Injected Weight		2546 Lbs

Payload

Biological	Geophysical-Geological	Atmospheric
Growth	Surface Penetrability	Temperature
Metabolic Activity	Soil Moisture	Pressure
Existence of Organic Molecules	Seismic Activity	Density
Existence of Photo-Autotroph	Surface Gravity	Composition
Turbidity and PH Changes		Altitude
Microscopic Characteristics		Light Level
Organic Gases		Electron Density
Macroscopic Forms (TV)		
Surface Sounds		

TABLE 3.1-2. ORBITER CAPABILITY

Weight Statement

Orbiting Weight		1815
Payload	347	
Fuel (1000 x 2278 n.mi)		1634
Injected Weight		3449 Lbs
Adapter and Δ Shroud		151 Lbs
		3600 Lbs

Scientific Capability

Television	1 KM	Stereo Map
	140 M	Blue Red Green-Yellow
	3-20 M	B&W

Upper Atmosophere Composition & Density

Ionosphere Profile

Particles & Fields

UV & IR Radiation

TABLE 3.1-3. WEIGHT ALLOTMENT FOR ORBITER/LANDER SYSTEM (MARS '71)

Orbiter Weight, lb		1440
Payload, lb	123	
Lander Weight, lb		1284
Payload, lb	110	
Injected Weight, lb		3600

Section 3.4 gives detailed data on the Orbiter/Lander system.

3.2 BUS / LANDER SPACECRAFT SYSTEM

3.2.1 SYSTEM CONFIGURATION STUDY AND ANALYSIS

A. BUS/LANDER SYSTEM SUMMARY

The system described in this section is the one selected as the prime approach for a mission involving a landing on the planet. Reasons for selection of this specific approach are presented in earlier sections on mission and payload analysis, value analysis and system selection.

The prime system consists of an Integrated Bus/Lander which is suitable for launch in any window from 1971 through 1977. The maximum payload that could be launched in 1971 is not utilized on this system because of decreased capability in 1973, which would mean redesigning the Titan IIIC shroud for only one opportunity. However, the Lander shown carries all of the presently identified payload with adequate margins; therefore, it would be necessary to provide two Landers to make full use of the Titan IIIC 1971 launch capability. This approach has not been selected because of the diminished capability in 1973 and the changes that would be necessary to the system at that time.

The Lander vehicle presented in this section has been designed to meet the requirements and ground rules noted in Section 3.2.2 with the maximum of reliability and payload. Parametric analyses were performed in the areas of structural and impact attenuation material design, parachutes, terminal retrorockets, heat shield and thermal control systems. Prime attention has been given to the retardation system wherein four combinations of parachutes, retrorockets, sensors and impact attenuation were considered.

Alternate analyses were conducted in the areas of:

1. The effect of variation of lateral wind
2. Design for a 90 degree entry (W/C_DA = 8 PSF)
3. Payload penalty in designing to a range of atmospheres
4. The effect of firm definition of the Martian atmosphere during a hardware program
5. The extensible flare lander designed to permit packaging within a 120-inch diameter shroud.

Table 3.2-1 identifies the prime vehicle subsystems, the reasons for selection and past work on the problem. Two alternate designs were also prepared for use on the Lander/Bus system; the Extensible Flare Lander and the Limited Rover Lander and are presented with differences from the prime vehicle noted. The Extensible Flare Lander has a folding flare section consisting of four flap type surfaces with support structure and linkage, all of which are contained within a 110-inch diameter for launch on a booster with a 120-inch diameter shroud. Immediately after the shroud is removed on leaving the Earth's atmosphere, the flaps are extended and locked in place to become fixed structure for the remainder of the mission. The extensible flare section is jettisoned when the decelerator chute is deployed after entry to reduce chute and impact loads and to eliminate chute fouling problems. The Limited Rover Lander design was prepared as a conceptual approach to using a small wheeled vehicle to obtain additional mission value. Adequate payload capability is available to provide the mobile vehicle which carries the surface sampling instruments over an area limited by a trailing cable attached to the main vehicle for power supply and communication.

The Bus of the Integrated Bus/Lander uses the maximum of Lander equipment during transit to the vicinity of Mars. All power supply and communication equipment, except the transit antenna, are located in the Lander. The Bus then consists primarily of guidance and control systems, mid-course propulsion, the antenna for use in transit and the necessary structure to support these components and attach the Lander to the launch vehicle. After the Lander is separated, the Bus no longer has electrical power or communication capability and hence, becomes inoperative.

TABLE 3.2-1. LANDER SUBSYSTEM SUMMARY

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

B. LANDER MISSION PROFILE

Separation of the Spacebus and Entry/Lander is programmed to occur at 150,000 nautical miles from the planet as shown in Figure 3.2-1 on command from Earth. Physical detachment will be made by initiating four tie-in explosive bolts. The nitrogen cold gas system will fire through two canted nozzles to give a separation rate of one foot per second and a roll rate of one revolution per second. After sufficient distance is reached between the Spacebus and Lander to prevent particle impingement, the Lander delta velocity solid rocket is fired. At separation + 22 minutes, the rocket is ejected. The Lander then proceeds on an impact trajectory telemetering directly to Earth diagnostic information and separation sequence data. Just prior to entry (entry sequence begins at 10^6 feet altitude) the thermal control system switches from the space radiator to an internal evaporative heat exchanger. The radiator is then ejected. Entry experiments and the telecommunications system will be operating at this time. A buffer storage unit is provided for recording of events which take place during communication blackout and as a backup to real time descent telemetry. At a preselected descending g level, the retardation deployment sensor will be activated. When Mach 2.5 is reached a signal from this sensor will fire a mortar charge which deploys the decelerator parachute. Should there be a malfunction in the redundant programmer, the radar altimeter will deploy the parachute at an altitude of 20,000 feet. If an extensible flare design is used, the flare section is separated just prior to decelerator parachute deployment. After a timed interval to allow the vehicle to decelerate to or below Mach 1.0, the main parachute is deployed. The radar altimeter will be further called upon to signal retrorocket firing altitude. The retrorockets will fire to minimize impact velocity. Upon impact, the parachute harness lines will be severed and if it is found necessary, a small solid rocket may be used to spill the chute to one side of the Lander.

After all vehicle motion has stopped, the Lander aft cover will unlock and open, stabilizing the vehicle. The cover actuator will be designed to operate even if the Lander comes to rest on its aft cover. Once open, harpoons will be fired to stake the vehicle to the ground. The antenna boom and TV camera can now be erected and the surface scientific experiments can be deployed overboard. The radioisotope thermoelectric generator is now exposed to the Martian atmosphere and uses direct thermal radiation as a cooling mode. Surface operation of the payload can proceed under a pre-programmed

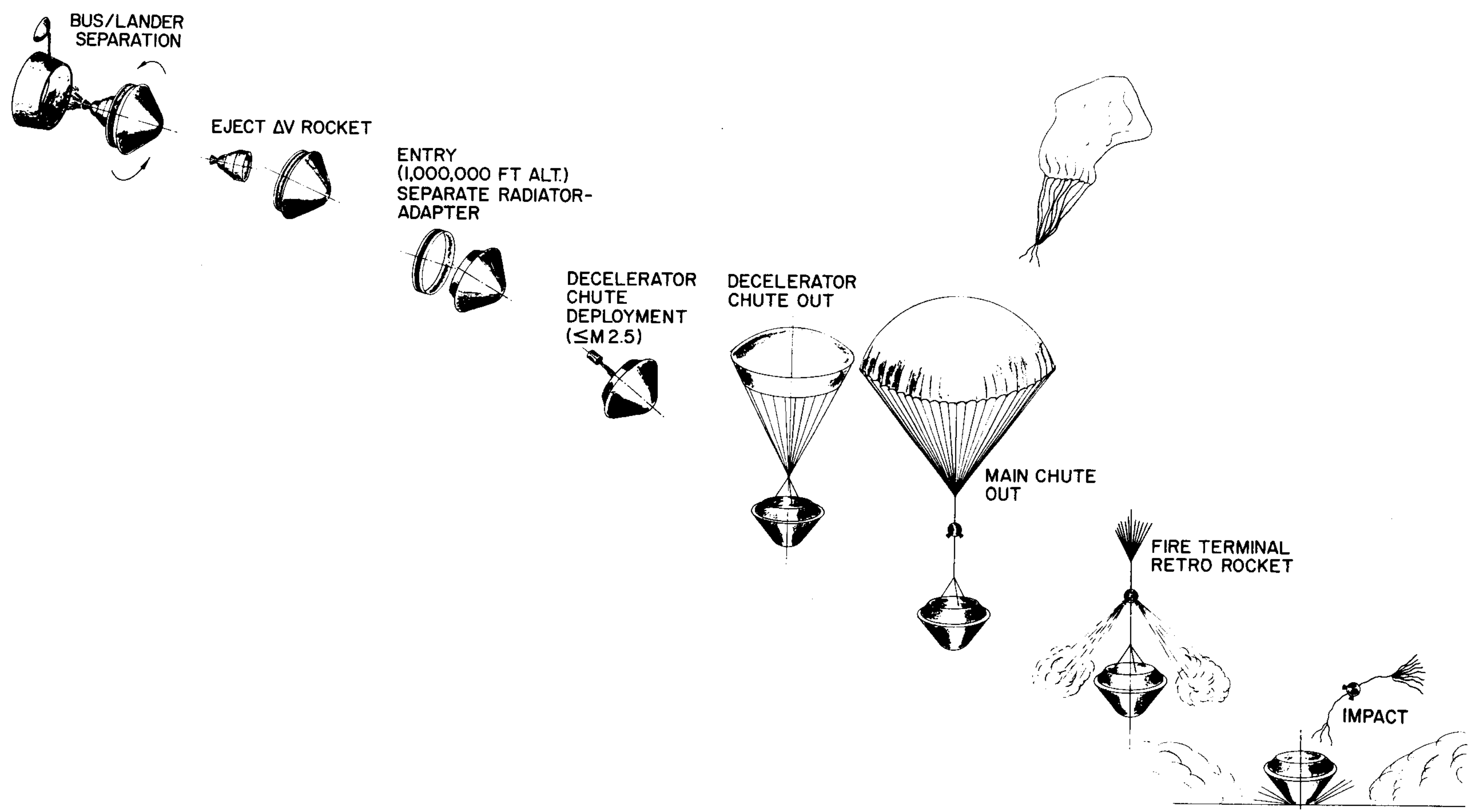


Figure 3.2-1. Sequence Diagram - Entry/Lander

arrangement or can be activated through Earth command as required. Surface life is planned for six months. Throughout the entire mission, the payload is maintained at a proper operating temperature through the use of a secondary heat exchanger loop between the RTG heat source and the payload.

C. SEQUENCE OF EVENTS

A detailed sequence of events covering items pertinent to the Lander from prelaunch checkout to operation on the surface of Mars is presented in Table 3.2-2.

D. SUMMARY WEIGHT STATEMENT

A summary of the preliminary weight statement is shown in Table 3.2-3 for the prime configuration (134-inch base diameter). For comparison, weights are also shown for the alternate configurations studied, Extensible Flare, and Limited Rover.

3.2.2 LANDER CONFIGURATION DESIGN

A. LANDER SYSTEM REQUIREMENTS & DESIGN PHILOSOPHY

In general the Titan IIIC Lander is based on the same philosophy and design criteria as used in the Saturn 1B-Voyager Study. The following points formed general and specific ground rules for this study:

1. Designs are based on 1965 state of the art
2. Lander is capable of operating in 11-30 mb model atmospheres
3. Prime system is designed to enter the above atmospheres at angles of 20 to 35 degrees down from local horizontal
4. An alternate study was conducted to show the effect of unrestricted entry angle on Lander design
5. The parametric and analytical results obtained in the Voyager Study were utilized to permit the maximum progress in new analysis and design
6. A Mach number of 2.5 at 20,000 feet for adequate chute deployment was taken as criteria for trajectory and W/C_DA optimization
7. Shock attenuation material was provided to limit impact loads to 125 g's
8. Six months lifetime is required on surface of Mars
9. Lateral wind velocity at impact is 40 mph

TABLE 3.2-2. SEQUENCE OF EVENTS - VOYAGER TITAN IIIC

Lander Events	Signal	Time to Begin	Operating Time	Cumulative Time	Remarks
1.0 Prelaunch Checkout	Command thru Orbiter or Bus				
1.1 Communications system check		P	10 min	10 min	
1.2 Power Supply and Thermal Control System Checkout		P + 10 min	10 min	20 min	
1.2.1 RTG Inlet and Outlet Temperatures					
1.2.2 Cooling system pump-motor temperature					
1.2.3 Cooling water temperature and flow rate (sensor on AGE)					
1.2.4 Modulation valve temperature					
1.2.5 Water boiler pressure					
1.2.6 Differential pressure across pump					
1.2.7 Pump-motor power & RPM check					
1.3 Pyrotechnic Devices Checkout		P + 20 min	5 min	25 min	
1.3.1 Arming check of all explosive squibs & other pyrotechnics					
1.3.2 Disarm all devices					
1.4 Propulsion System Checkout		P + 25 min	5 min	30 min	
1.4.1 Cold gas temperature					
1.4.2 Ignition checkout (ΔV and retardation retrorocket)					
1.4.3 Spin & separation system valve checkout					

TABLE 3.2-2. SEQUENCE OF EVENTS - VOYAGER TITAN IIIC (Cont'd)

Lander Events	Signal	Time to Begin	Operating Time	Cumulative Time	Remarks
1.5 Scientific Payload Checkout		P + 30 min	10 min	40 min	
1.5.1 Temperature checkout of experiments					
1.5.2 Power check					
1.6 Vehicle Structure Checkout					
1.6.1 Monitor structural strain reference		Continuous			
1.7 Aerothermodynamic Sensor Checkout		P + 40 min	15 min	55 min	Continuity & Power Checks
1.7.1 Pressure transducer & temperature sensor check					
1.7.2 Char sensor check					
1.7.3 Ablation sensor check					
1.8 Motion Sensor Checkout		P + 55 min	10 min	65 min	
1.8.1 Three-axis rate gyro checkout - fine & coarse					
1.8.2 Three-axis acceler-ometer checkout - fine & coarse					
2.0 Launch Phase					
2.1 Motion sensors on & Telemeter-ing Thru In-flight Disconnect	Command thru Orbiter or Bus	L - 10 min			
2.2 Selected Diagnostic Instruments on and Telemetering Thru IFD	Command thru Orbiter or Bus	L - 10 min			
2.3 Launch		L			
2.4 Release of Nose Fairing	Command to Launch Vehicle	300 K Ft.			
2.5 Extension and Lock of Lander Flap Section	Sequenced with Event 2.4				Extensible Flare Design Only

TABLE 3.2-2. SEQUENCE OF EVENTS - VOYAGER TITAN IIIC (Cont'd)

Lander Events	Signal	Time to Begin	Operating Time	Cumulative Time	Remarks
5.10 Switch from Radiator to Boiler	Command thru Lander Programmer - Stored Command	E - 1 min	3 sec		
5.11 Arm Explosive Bolts	Command thru Lander Programmer - Stored Command	E - 1 min			
5.12 Fire Explosive Bolts - Eject Radiator	Command thru Lander Programmer - Stored Command	E			Solid Flare Design Only

TABLE 3.2-2. SEQUENCE OF EVENTS - VOYAGER TITAN IIIC (Cont')d

Lander Sequence of Events	Signal	Time: Shortest	Time: Longest	Altitude Ft.: Lowest	Altitude Ft.: Highest	Remarks
6.0 Entry Phase						
6.1 Buffer Storage On	Stored Command Timed From Separation	E		1 Million		"g" Switch Backup
6.2 Motion Sensors On	↓	↓		↓		
6.3 Diagnostic Instruments On						
6.4 Entry Scientific Experiments On						
6.5 Communication System On & Transmitting						
6.6 Enter Communications Blackout		E + 60	E + 79	300,000	500,000	
6.7 Arm Retardation Programmer	"g" Switch	E + 83	E + 154	130,000	182,000	
6.8 Start Radar Altimeter	"g" Switch	E + 83	E + 154	130,000	182,000	
6.9 End Communication Blackout		E + 80	E + 154	132,000	182,000	
6.10 Peak Heating		E + 87	E + 156	98,000	176,000	
6.11 Peak "g"		E + 90	E + 169	77,000	145,000	
6.12 Activate Retardation Sensor Timers	"g" Switch					At Occurrence of Reference "g" Level
6.13 Jettison Atmospheric Instrument Cover Jettison Extensible Flare & Radiator Assembly	"g" Switch ↓					Extensible Flare Design Only
6.14 Fire Mortar - Decelerator Parachute Deployed	Retardation Programmer	E + 110	E + 225	20,000	93,000	Mach 2.5
6.15 Jettison Parachute Well Cover Main Parachute Deployed (Reefed)	Timer	E + 125	E + 240	10,000	86,600	Mach 1.0 or Below
6.16 Main Parachute Dereefed	Timer	E + 131	E + 246	8,000	84,600	
6.17 Switch Radar to Low Altitude Phase	Timer	E + 131	E + 246			

TABLE 3.2-2. SEQUENCE OF EVENTS - VOYAGER TITAN IIIC (Cont'd)

Lander Events	Signal	Time to Begin	Operating Time	Cumulative Time	Remarks
3.0 Parking Orbit & Entry Into Transit Phase		L + 6 min			
3.1 Motion Sensors On & Telemetering thru IFD	Orbiter or Bus Programmer				
3.2 Selected Diagnostic Instruments On and Telemetering thru IFD					
4.0 Transit Phase					
4.1 Motion Sensors Off	Command thru Orbiter or Bus	L + 40 min			
4.2 Periodic Diagnostic Monitoring	Command thru Orbiter or Bus				
4.3 Exercise Thermal Control Correction if Required	Command thru Orbiter or Bus				
4.4 Assume Terminal Mode of Transit Phase - Start Separation Sequencer	Command thru Orbiter or Bus	S - 4 min			Located On Lander
4.5 Motion Sensors & Diagnostic Inst. On	Command thru Orbiter or Bus	S - 3 min			
4.6 Arm In-Flight Disconnect (IFD)	Command thru Separation Sequencer	S - 2 min			
4.7 Fire IFD	Command from Separation Sequencer	S - 1 min			
4.8 Arm Explosive Bolts	Command from Separation Sequencer	S - 1/2 min			
4.9 Fire Explosive Separation Bolts		S			
4.10 Arm Spin & Separation System Reset Sequencer, Read Out Acceleration	Separation Switch	S			

TABLE 3.2-2. SEQUENCE OF EVENTS - VOYAGER TITAN IIIC (Cont'd)

Lander Events	Signal	Time to Begin	Operating Time	Cumulative Time	Remarks
5.0 Approach Phase					
5.1 Fire Cold Gas Spin & Separation System	Command from Separation Sequencer	S + 1 sec	30 sec	224 Days	150,000 nm from Planet
5.2 Arm Delta V rocket	Command from Separation Sequencer	S + 10 min			
5.3 Fire Delta V rocket	Command from Separation Sequencer	S + 17 min	30 sec		
5.4 Arm Rocket Adapter Explosive Bolts	Command from Separation Sequencer	S + 20 min			
5.5 Fire Explosive Bolts-Eject Rocket Motor & Adapter	Command from Separation Sequencer	S + 22 min			
5.6 Recorder Playback of Separation Events	Command thru Lander Programmer - Stored Command	S + 25 min	22 min		
5.7 Telemetry of Selected Diagnostic Instruments	Command thru Lander Programmer - Stored Command				
5.8 Activate Retardation Battery	Command thru Lander Programmer - Stored Command	E - 2 min			
5.9 Read Out Thermal Control System Diagnostic Instruments	Command thru Lander Programmer - Stored Command	E - 2 min			

TABLE 3.2-2. SEQUENCE OF EVENTS - VOYAGER TITAN IIIC (Cont'd)

Lander Sequence of Events	Signal	Time Shortest	Time Longest	Altitude Ft. Lowest	Altitude Ft. Highest	Remarks
7.0 Descent Phase						
7.1 Transmitter Turn Off	Timed From Entry	E + 10 Minutes		—		Required if Entry Exceeds 10 Minutes
7.2 Playback Entry Data						
7.3 Fire Retardation Retrorockets	Initiation Radar	—		h = 90 Ft.		
7.4 Impact		E + 185	E + 530			
7.5 Spill & Release Main Parachute, Arm Relay	Impact Switch					

Lander Sequents of Events	Signal	Time to Begin Sequence	Remarks
8.0 Surface Phase			
8.1 Upset Vehicle to Nose (Aft Cover Door Opened, Harpoons Fired)	Surface Sequencer	I + 1 minute	Surface Sequencer Initiated By Impact Switch Vehicle Motion Will Delay Event 8.1
8.2 Stabilizing Legs Extended		I + 1.3 minutes	
8.3 S-Band Antenna and Panorama TV Deployed		I + 2 minutes	
8.4 Deployment and Initiation of Surface Scientific Experiments		I + 3 minutes	
8.5 Begin Surface Telemetry Sequence		As Required	

TABLE 3.2-3. LANDER SUMMARY WEIGHT STATEMENT

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

B. BOOSTER & SHROUD LIMITATION

The Titan IIIC Launch Vehicle has the interface diameter of 120 inches which limits vehicles inside a straight shroud to an approximately 110-inch diameter. However, as discussed in Section 3.2.3, various larger diameter shrouds have been proposed which lead to a bulbous configuration. The system selected for the prime Lander study as discussed previously, consists of a 134-inch diameter Lander and integrated Bus which necessitate a 144-inch outside diameter shroud. However, because of cost and other considerations, it was deemed desirable to determine whether an equal gross weight Lander could be contained within the 110-inch diameter shroud by folding some of the required drag surface. This requirement has resulted in the Extensible Flare Configuration which is presented later in this section. This concept would permit use of a straight shroud and flexibility in changing drag area relatively late in a hardware program.

C. TECHNICAL ANALYSIS

Basic technological analyses have been performed as a part of this study to determine requirements for the vehicle during the entry and surface operation modes. The results are contained in this section.

1. Configuration Study & Selection

In determining a Mars entry configuration for Voyager application, one can narrow the range of configuration classes to ballistic vehicles on the basis of "state of the art" considerations and uncertainties surrounding early planetary missions (Reference 1). Within the general class of flight tested ballistic configurations are sphere-cones and sphere-cone-cylinder-flares. Possible adverse aerodynamic stability effects on the flared shapes, due to atmospheric CO_2 content, and overall higher design confidence dictate the choice of sphere-cones. Limits of the sphere-cone configurations can vary from sharp-pointed cones to very blunt segmented spheres. Both of these extremes are worthy of comment for specific applications.

Pointed sphere cones, which maintain an attached bow shock wave, have been suggested by Allen of NASA Ames as desirable, if not the required, entry shapes when the heating due to radiation becomes dominant. For Voyager applications, extremely high entry velocities are not encountered and overwhelming radiative heating is not expected.

Therefore, the lower drag of these pointed bodies (and their associated higher ballistic parameter) makes them appear non-optimum for this mission.

Very blunt configurations, such as the Apollo type, are also of interest for Mars entry. The high drag shapes lower the ballistic coefficient so that the retardation problem is eased. However, they also present large areas with near stagnation values of loads and heating. Perhaps the largest question mark on these shapes is their dynamic characteristics and the resultant capability to converge to near zero angle of attack during regions of high loads and heating. During initial portions of the entry while the dynamic pressure is increasing, density damping will occur. When the dynamic pressure rises to significant values, the aerodynamic characteristics of the configuration will influence oscillation convergence. After peak dynamic pressure any aerodynamic instability will be even more pronounced. While some differences in test data exist concerning the exact value of aerodynamic damping ($C_{m_q} + C_{m_{\dot{\alpha}}}$) for these shapes at high Mach numbers, they at best have near zero damping for small angles of attack. Combining this with negative values of C_{L_α}, the dynamic stability factor $C_D - C_{L_\alpha} + (C_{m_q} + C_{m_{\dot{\alpha}}}) (D/\sigma)^2$ will be positive for these high drag shapes indicating poor angle of attack convergence. While the situation may improve at higher angles of attack, six-degree-of-freedom trajectories have shown with relatively small initial pitch rates ($q < 10$ degrees/sec.), high angles of attack or even tumbling can occur well into the entry trajectory even with reasonable spin rates. In view of the existing uncertainties in the Voyager mission (i.e., atmosphere, possible Lander orientation at initial entry etc.), it is felt that unless absolutely necessary for accomplishment of mission requirements, very blunt segmented sphere shapes should be avoided. This process of elimination leaves a rather broad family of blunted sphere-cone configurations for possible lander configurations.

A matrix of these configurations were parametrically investigated to determine optimum configurations (Reference 1). The results of this study indicate an optimum configuration at a bluntness ratio of approximately 0.6 and a half cone angle near 50 degrees. The upper limit on cone angle is a function of acceptable packaging density and adverse angle of attack convergence of blunter shapes as discussed previously. Fortunately, early re-entry technology (when heat sink approaches were being followed) has provided a

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Reference 1. Voyager Design Study, Volume IV, System Design, GE-MSD 63SD801, October, 1963.

near optimum configuration for Mars entry in the flight-proven Mark 2 vehicle. This configuration has a half cone angle of 51.5 degrees and a bluntness ratio of 0.47. The center of pressure location for this vehicle is 64 percent of the base diameter. Center of gravity estimates for the prime vehicle which was designed as a part of this study show $X_{CG} = 0.288D$, hence, more than adequate static margin is provided. Since extensive data are available on this configuration and on the basis of trade-off studies it appears near optimum, it has been selected as the Voyager Mars Lander.

2. Aeromechanics

a. Flight Dynamics

(1) Planet Characteristics - The planetary entry trajectory data presented herein assumed a round, non-rotating Mars with radius of 11,128,000 feet and surface gravity of 12.24 feet/second^2. No winds were considered and several model atmospheres were assumed. The density variations with altitude for these atmospheres which are based on Kaplan's recent investigations appear in Figures 3.2-2 and 3.2-3 and are designated according to their surface pressures. Table 3.2-4 shows a comparison of the density gradients expressed in terms of the atmospheric density parameter, β, which were estimated by averaging several values of β calculated at several altitudes.

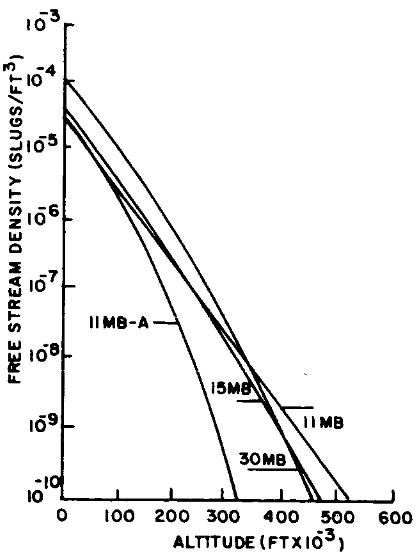


Figure 3.2-2. Martian Model Atmospheres

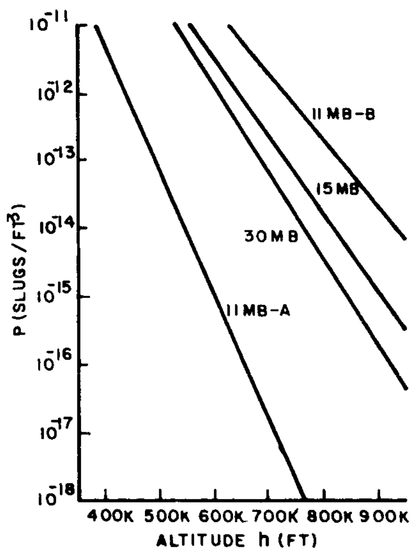


Figure 3.2-3. Mars Density Profile

TABLE 3.2-4. DENSITY GRADIENT COMPARISON

Model Atmosphere	$\beta = \frac{1}{h} \ln \frac{\rho}{\rho_{reference}}$
11 mb-A	$0.44 \times 10^{-4}\ ft^{-1}$
11 mb-B	$0.25 \times 10^{-4}\ ft^{-1}$
15 mb	$0.29 \times 10^{-4}\ ft^{-1}$
30 mb	$0.32 \times 10^{-4}\ ft^{-1}$

(2) Capture Angle - In order to trap a vehicle flying at a small path angle (measured down from local horizontal) in a planet's gravitational field, the vehicle must experience enough atmospheric braking to remove the hyperbolic excess velocity. To accomplish this atmospheric braking, the vehicle must enter the planet's atmosphere at an entry path angle greater than the capture angle. For this study, the capture angle was defined as the path angle at entry (10^6 ft) above which the trajectory would have a monotonic decreasing altitude history. Capture angles for several atmospheres, entry velocities, and W/C_DA's have been determined in earlier studies (see references). Figures 3.2-4 and 3.2-5 show the effect of velocity on capture angle for several atmospheric models for a W/C_DA = 10 and 15 psf. The data are based on a vehicle with zero angle of attack at entry. Uncertainties in the Martian atmosphere make it unwise to design a vehicle to enter too near the capture angle since the total integrated heating becomes quite large, and a small guidance error could cause an "overshoot", resulting in the vehicle escaping. Consequently, the minimum path angle considered for this study was 20 degrees down from horizontal.

(3) Selection of Ballistic Parameter - From the initial work which was done on the Saturn 1-Voyager study using Kaplan's low density atmospheres, it became apparent that higher entry path angles were incompatible with retardation requirements. Studies of retardation methods have led to a two-parachute (high speed decelerator and terminal parachute) system (See Section 3.2.2-C-5). State-of-the-art considerations show that a deployment Mach number of 2.5 is attainable for the decelerator chute and that 20,000 feet represents a near minimum safe deployment altitude. From these retardation system requirements, a tradeoff of entry path angle, γ_e, and ballistic parameter, W/C_DA, can

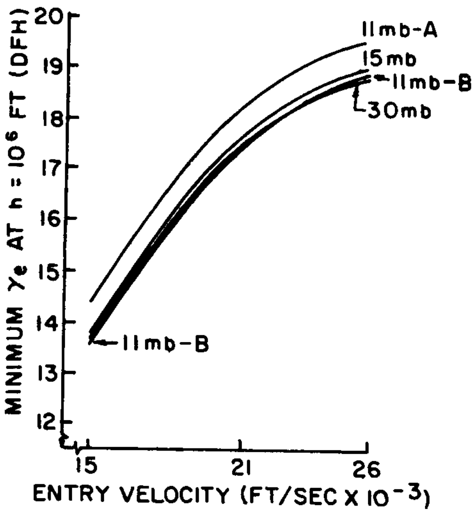


Figure 3.2-4. Capture Angle for Martian Entry. W/C_DA = 10PSF

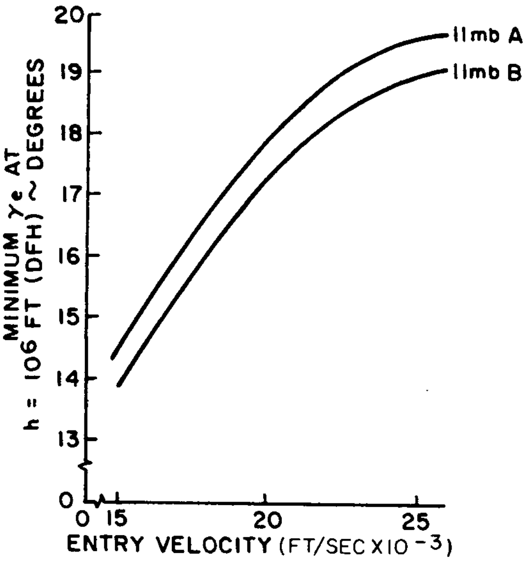


Figure 3.2-5. Capture Angle for Martian Entry. W/C_DA = 15 PSF

be made. Figure 3.2-6 shows the W/C_DA and γ_e required for the 11 mb-A, 15 mb and 30 mb atmospheres to allow the vehicle to decelerate to Mach 2.5 at 20,000 feet altitude.

An entry path angle of 20 degrees represents the design skip limit and establishes the lower limit of the entry corridor. Based on system criteria of the Voyager S-1B Study a maximum path angle of 35 degrees was specified which is compatible with a ballistic parameter of 16 psf. For design purposes and as a hedge against a potential weight growth, a W/C_DA = 15 psf was chosen. Recent error analyses show that $\pm$2.4 degrees (3σ value) is a reasonable tolerance on entry path angle and according to Figure 3.2-6 would allow a selection of W/C_DA = 27 psf. However, in view of the uncertainty of the atmosphere and the sensitivity of required W/C_DA to it, it seems unrealistic to design a vehicle to this W/C_DA; therefore, a conservative value of W/C_DA = 15 psf has been recommended. The ballistic parameter required can be directly affected if higher Mach number parachutes are developed and if further confidence is found by confirmation of the atmosphere with prior Mariner flights.

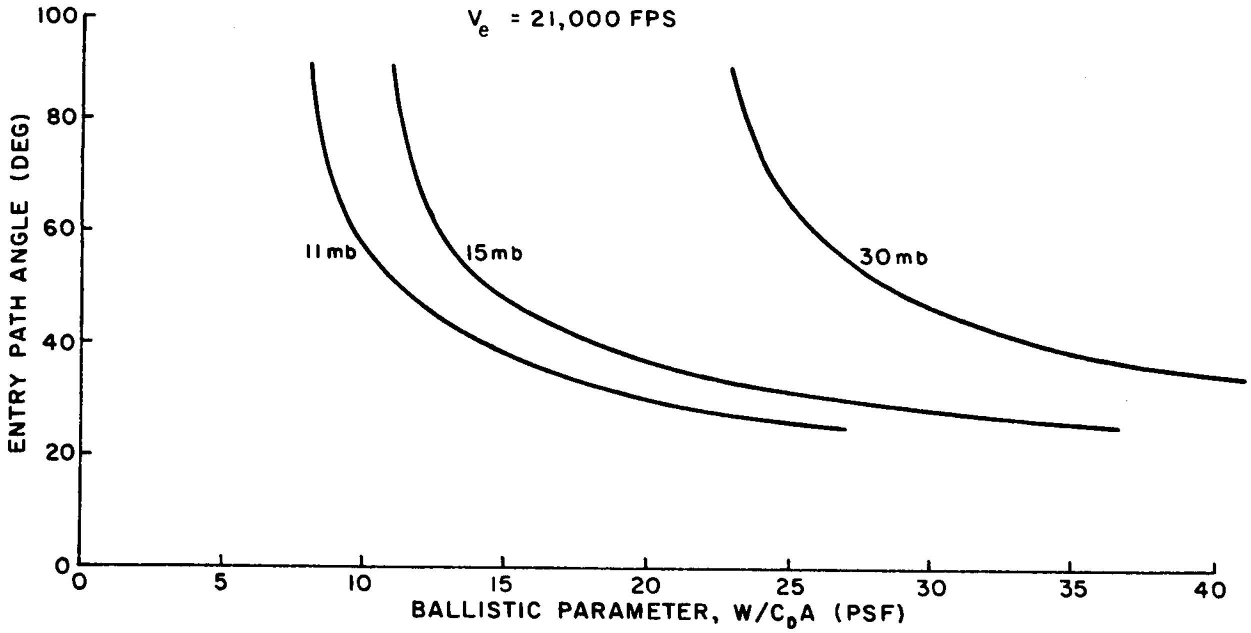


Figure 3.2-6. Allowable Entry Corridor and Atmospheric Limits to Obtain M 2.5 at 20,000 Feet

(4) Point-Mass Trajectories — Drag coefficients for all point-mass trajectories are based on information contained in Section b below and are used as a function of both Mach Number and altitude. The several C_D-vs-h variations were constructed by matching Earth and Martian densities and adjusting the altitudes accordingly (Reference 2). Trajectory calculations use linear interpolation between tabular values of speed of sound and logarithmic interpolation between tabular values of density. Entry is assumed to be 10^6 feet.

Nominal entry velocity and angle for the Voyager vehicle are 21,000 fps and 27 degrees, respectively. Retardation and aerodynamic considerations have led to a design vehicle with a W/C_DA of 15 psf. Trajectory parameters for an entry into the 11 mb-A atmosphere for the above nominal case are presented in Figures 3.2-7 and 3.2-8. The 11 mb-A atmosphere causes the largest entry loads since it has the largest density gradient of all the atmospheric models considered. The largest peak heating and the lowest altitude of occurrence for Mach 2.5 (deceleration chute deployment) also occur for this atmosphere.

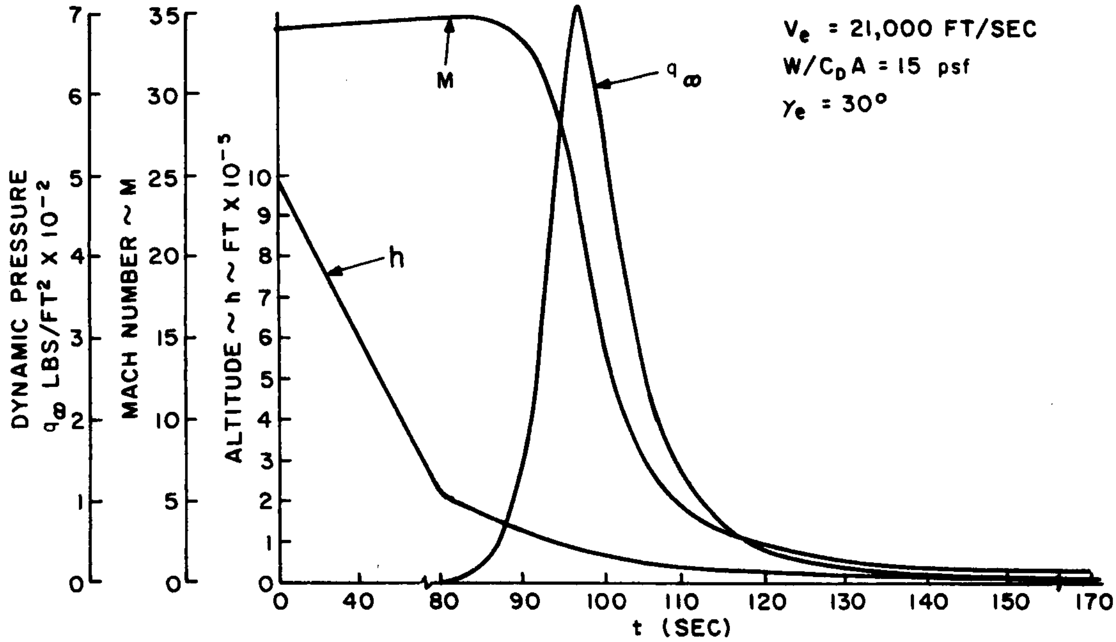


Figure 3.2-7. Trajectory Parameters for Entry into Martian 11 mb-A Atmosphere

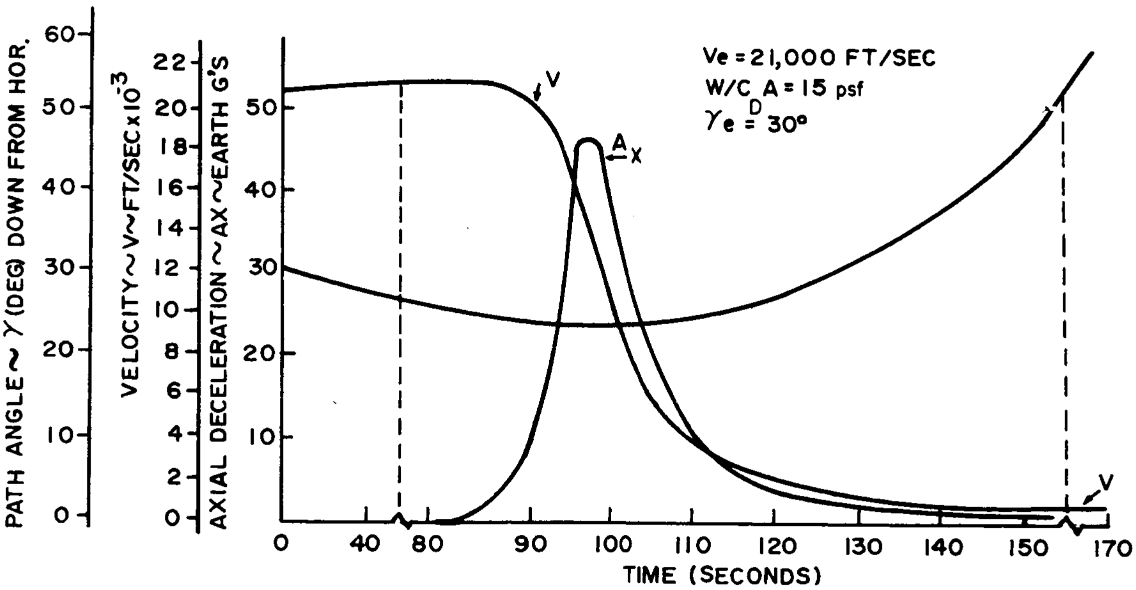


Figure 3.2-8. Trajectory Parameters for Entry into Martian 11 mb-A Atmosphere

(5) Maximum Loads — The maximum axial deceleration which the vehicle will experience under the above nominal entry conditions is approximately 47 Earth g's; Ax_{max} for a trajectory entering at the maximum design γ_e (35 degrees) is approximately 58 Earth g's. The effect of entry path angle and atmosphere on peak axial deceleration for the nominal condition of V_e = 21,000 ft/sec is shown in Figure 3.2-9. The density-gradient effect mentioned earlier is evident, that is, the level of peak axial deceleration varies directly as the atmospheric density parameter, β. An increase in entry velocity causes an increase in maximum g's, and for W/C_DA's less than 60 pounds/feet^2, there is an increase in maximum g's as the W/C_DA decreases.

(6) Six-Degree-of-Freedom Trajectories — The point-mass trajectories yield no information about lateral loads, effects of spin, or effects of vehicle configuration. Consequently, analyses have been conducted in six degrees of freedom to describe the motion of and about the vehicle's CG. The aerodynamic coefficients were used as a function of altitude and angle of attack as well as Mach number and angle of attack.

A spin rate of 60 rpm is adequate to minimize the effects of undesired transverse rates due to errors in the separation system or induced transverse rates due to CG offset or products of inertia by minimizing the vehicle's precession cone prior to entry. Following

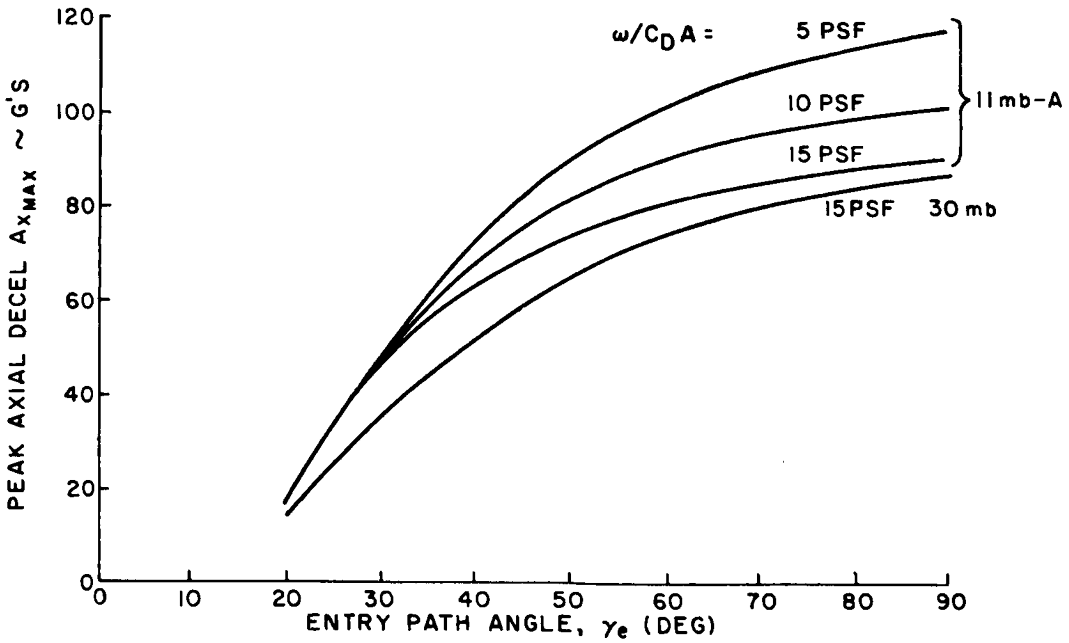


Figure 3.2-9. Maximum Deceleration Loads versus Entry Angle; V_e = 21,000 feet per second

entry, however, the angle-of-attack convergence will vary inversely with the spin rate, and too large a spin rate will cause the vehicle to maintain a fixed spatial orientation, resulting in large angles of attack at lower altitudes.

Angle-of-attack convergence will be most rapid for the model atmosphere having the largest density gradient. If the vehicle is de-spun to a small roll rate prior to entry, the motion will tend to become planar and angle-of-attack oscillations will converge rapidly.

Maximum loads normal to the vehicle's center-line (which vary directly with entry angle of attack) will be largest for the 11 mb-A atmosphere which has the largest density gradient.

Shallow entry path angles cause a slight increase in entry angle of attack due to the change in path angle and inertial central angle while the vehicle is above the sensible atmosphere. This increase seldom exceeds approximately five degrees but the subsequent angle-of-attack convergence is delayed to a much lower altitude. Reference 2 indicates that for vehicles of this shape, there is sufficient dynamic stability to ensure that convergence continues after the density damping decreases. There is a possibility of dynamic instability if the roll rate becomes much larger than 60 rpm.

(7) Spin Stabilization — The successful flight of the Lander requires the transfer of the vehicle from a hyperbolic orbit to an elliptical orbit which intersects the planet. This transfer is achieved by the addition of an incremental velocity to the Lander at a predescribed position along the hyperbolic orbit. While the spacebus (or orbiter) control system can align the Lander initially to the proper orientation for velocity addition, during and following separation the vehicle may be subjected to unwanted torques which destroy this orientation, with the result that the desired velocity and direction are not attained.

The motion of the vehicle may be contained by spinning the vehicle about its longitudinal or thrusting axis. The gyroscopic forces associated with spin prevent the vehicle from tumbling and thereby permit the attainment, within limit, of the desired velocity vector. In addition, spin stabilization insures proper vehicle attitude at entry.

The system producing the spin up is also used to provide an initial separation rate from the parent spacecraft. Presently envisioned is a pair of canted, cold gas jets which provide sufficient thrust to produce a separation velocity of 1 foot/sec. and a rotational rate of 60 rpm. Preliminary studies have been conducted (see Voyager Design Study, Volume

IV, Section 1.3.2-J) to determine the effects of tip off errors, rotational rates, torque times, and mass asymmetry on vehicle dynamics. Results indicate that a preferred spin up rate lies between 40 and 80 rpm, the former associated with vehicles with significant products of inertias and the latter choice for vehicles where no mass asymmetries are present. Nominally, a 60 rpm spin rate and a torqueing time of 30 seconds was chosen and has been used for the six-degree of freedom trajectory analysis.

b. Aerodynamics

Aerodynamic characteristics of Voyager, which is a 51.5-degree cone with a bluntness ratio of 0.47 are presented in Figures 3.2-10 to 3.2-22. These data have been mainly derived from available ground test data and supplemented with Newtonian and Free Molecular analyses.

Figures 3.2-10 and 3.2-11 present the zero-angle-of-attack drag coefficient variations with Mach number and altitude. The Mach number variation has been calculated assuming continuum flow at altitudes below 240,000 feet in the Martian 11 mb atmosphere. The altitude effects have been determined on the basis of a rarefied-gas Knudsen number analogy. Transition from continuum to free-molecule flow occurs between 250,000 and

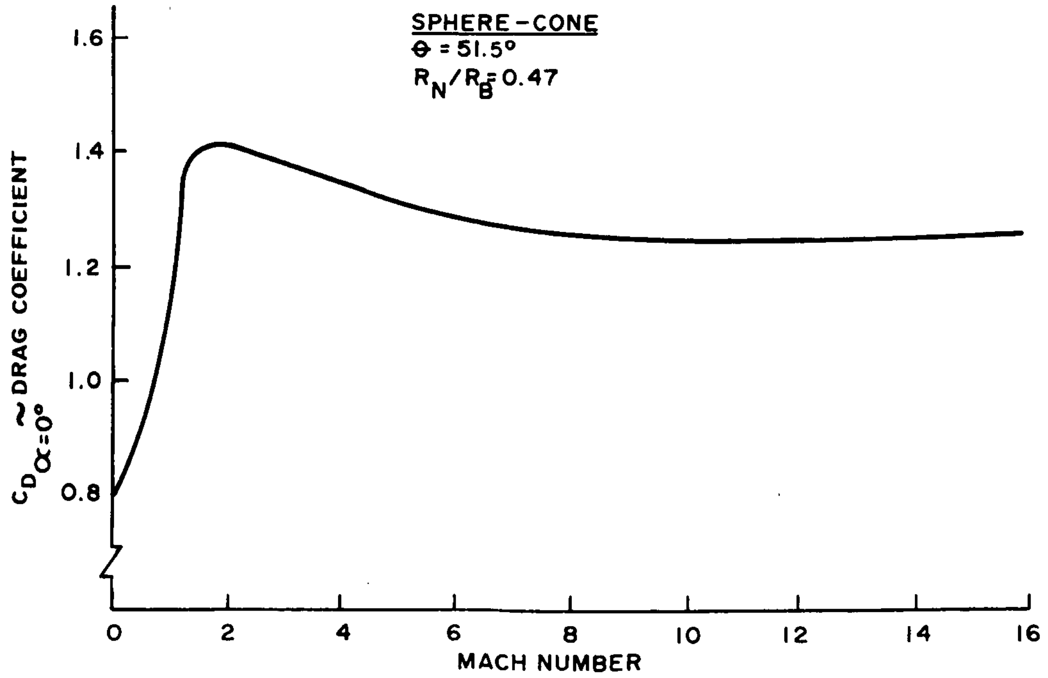


Figure 3.2-10. Variation of Drag Coefficient with Mach Number

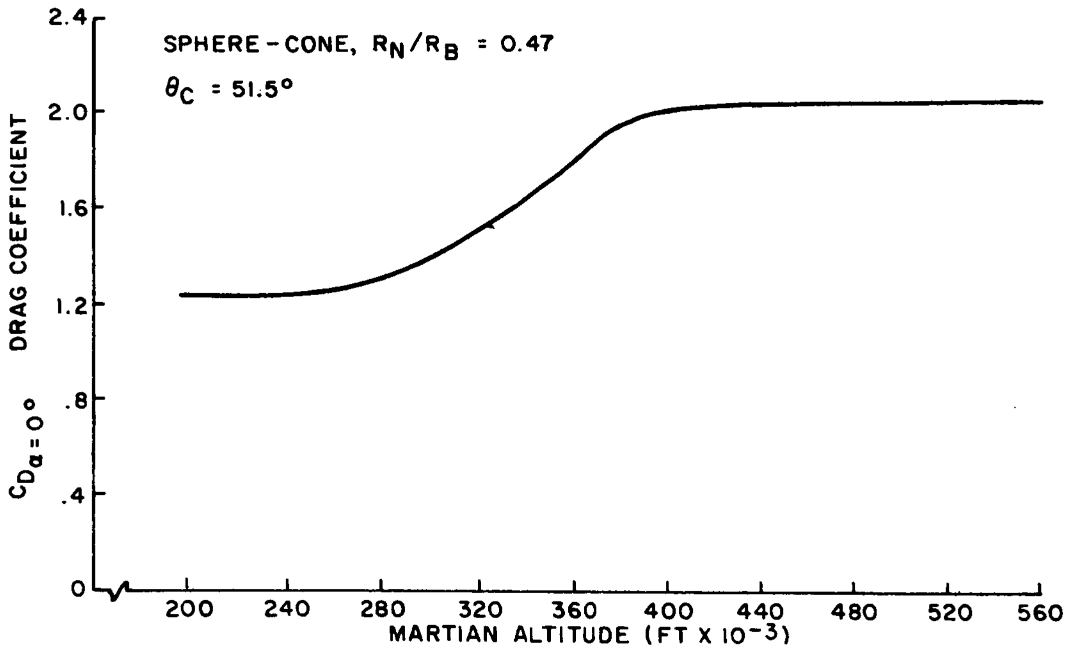


Figure 3.2-11. Variation of Drag Coefficient with Martian Altitude
Mars 11mb-A Atmosphere

400,000 feet in the 11 mb atmosphere. The limiting value of free-molecule flow has been calculated by a digital computer program, and all altitude effects (e.g., mean free path) have been referred to Earth's atmosphere and converted to the Martian 11 mb model on the basis of a density ratio. Viscous effects, within the continuum-flow regime, are relatively small for this type configuration.

The variations of axial force with angle of attack are presented in Figure 3.2-12. The hypersonic variation has been determined using a digital computer program based on Newtonian theory, and the supersonic axial force with angle of attack has been assumed to vary similarly to the hypersonic. Altitude effects on axial force as a function of angle of attack have been computed by a Knudsen number analysis and are presented in Figure 3.2-13.

Figure 3.2-14 shows the normal force variations with angle of attack and Figure 3.2-15 shows the normal force coefficient slope variations with Mach number. These data have

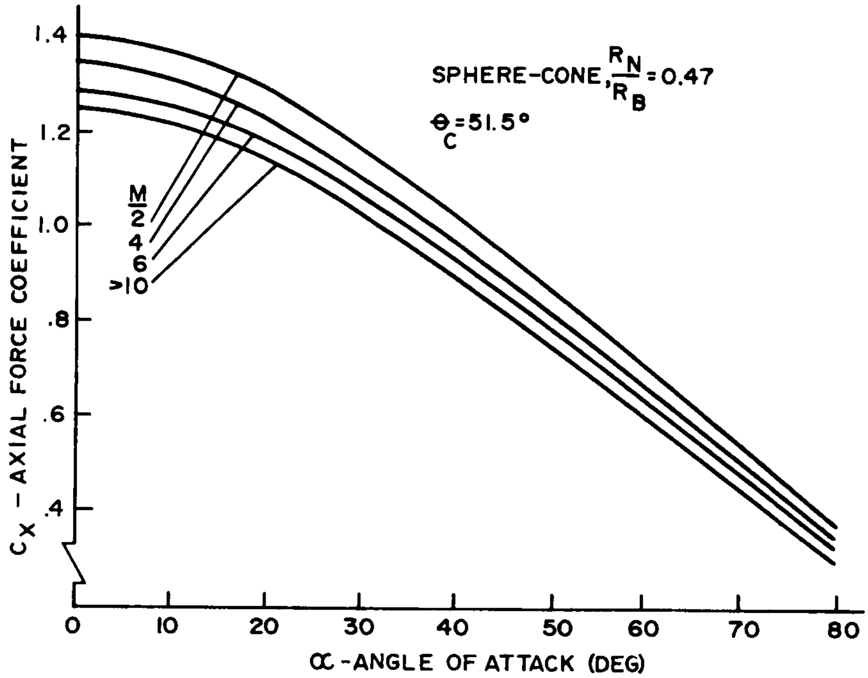


Figure 3.2-12. Variation of Axial Force Coefficient with Angle of Attack

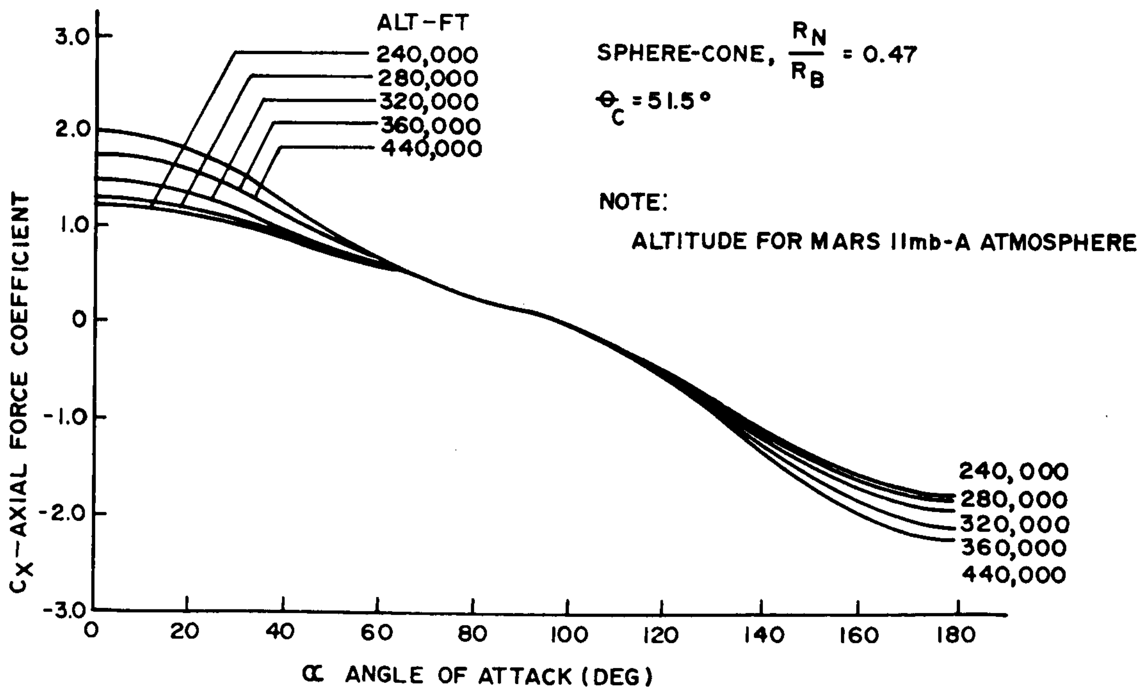


Figure 3.2-13. Variation of Axial Force Coefficient with Angle of Attack M > 10

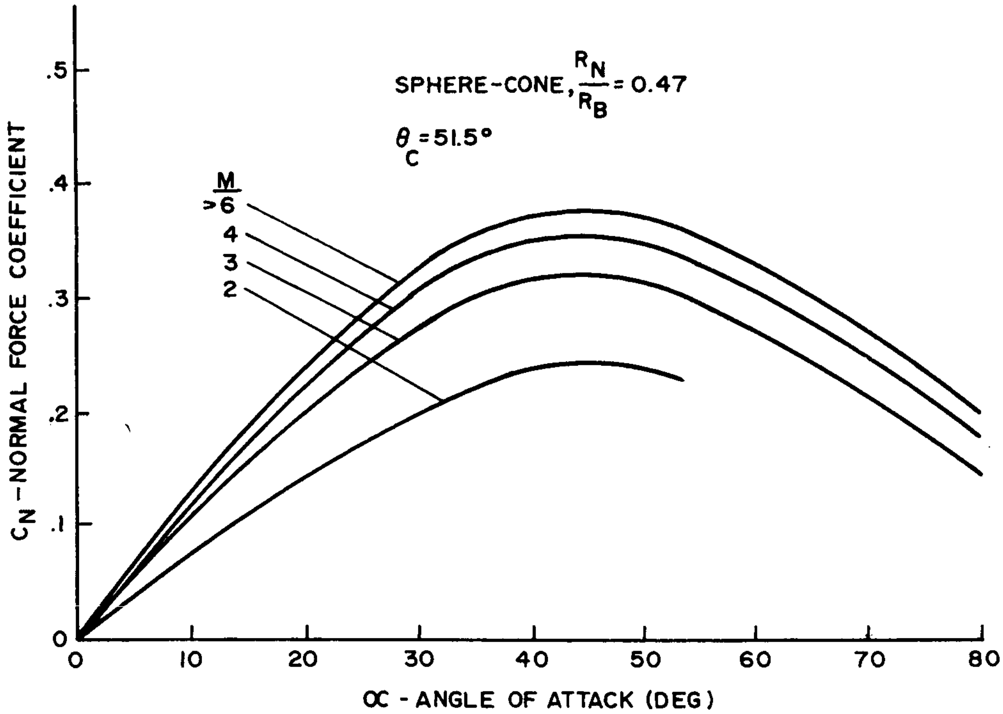


Figure 3.2-14. Variation of Normal Force Coefficient with Angle of Attack

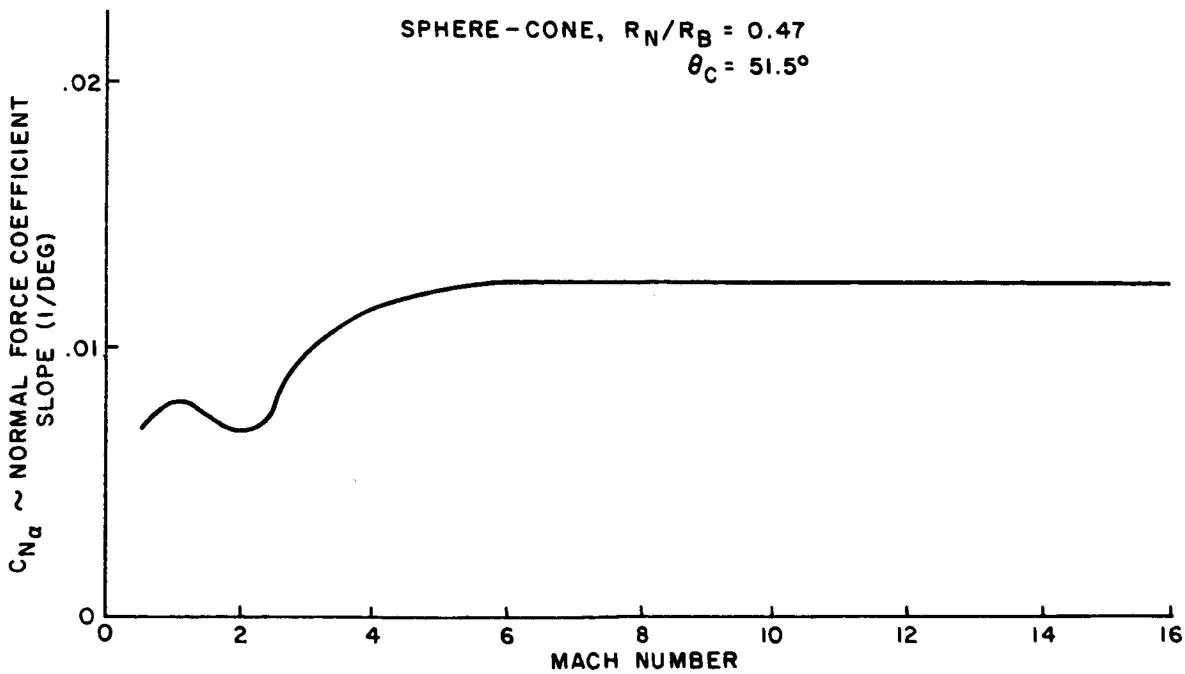


Figure 3.2-15. Variation of Normal Force Coefficient Slope with Mach Number

been determined utilizing Newtonian analyses for the hypersonic case and assuming similar variations for the supersonic cases. In the case of C_N, this was determined from available data correlations. The effects of altitude (only in free-molecule flow) on normal force variation with angle of attack are contained in Figure 3.2-16.

Figure 3.2-17 presents the variation of pitching moment about the nose with angle of attack for various Mach numbers, and Figure 3.2-18 shows the pitching moment coefficient slope about the nose as a function of Mach number. Altitude effects, presented in Figure 3.2-19, are derived from a Knudsen number analogy. Past experience with this configuration indicates that backward stability can be obtained with certain CG locations. A more detailed design would require further analysis to ensure backward instability.

Dynamic-damping characteristics for this configuration are presented in Figure 3.2-20.

Estimated pressure distributions at $\alpha = 0$ degrees and 15 degrees for a Mach number of 5 are shown in Figure 3.2-21. Figure 3.2-22 presents a hypersonic pressure distribution for this configuration at zero angle of attack.

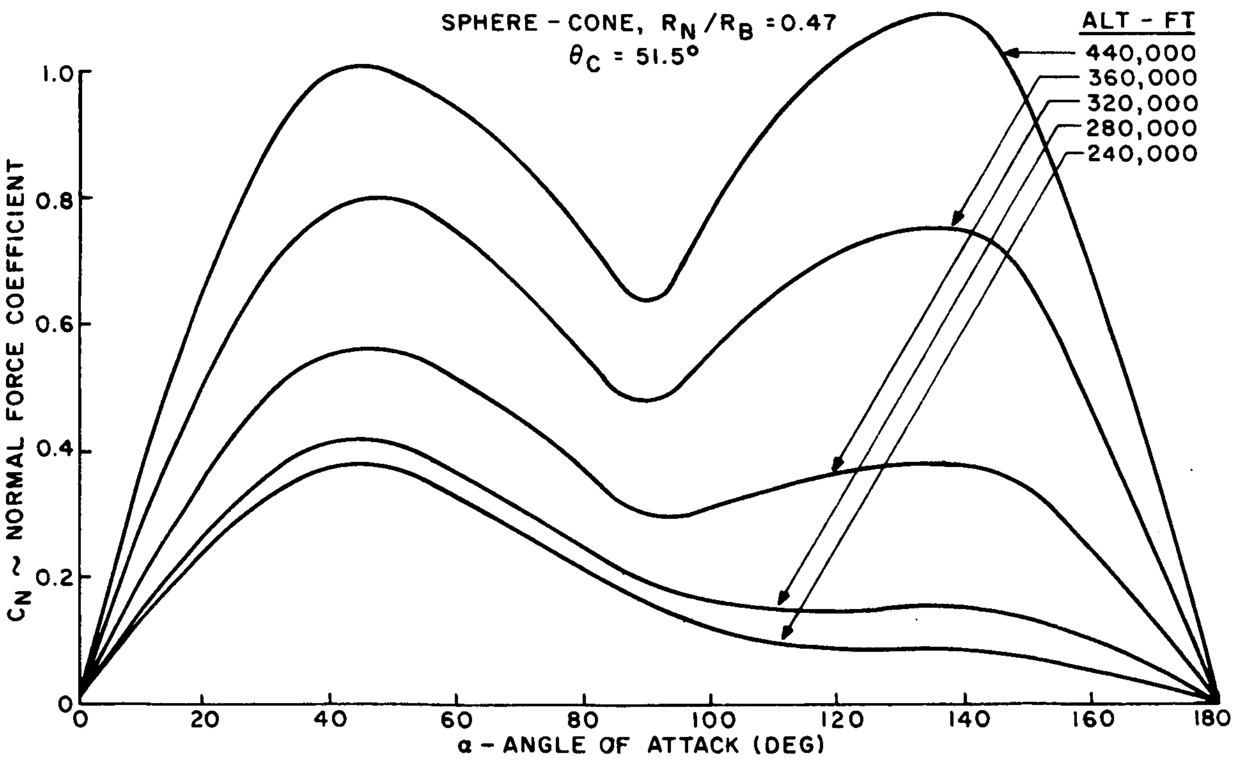


Figure 3.2-16. Variation of Normal Force Coefficient with Angle of Attack M > 10

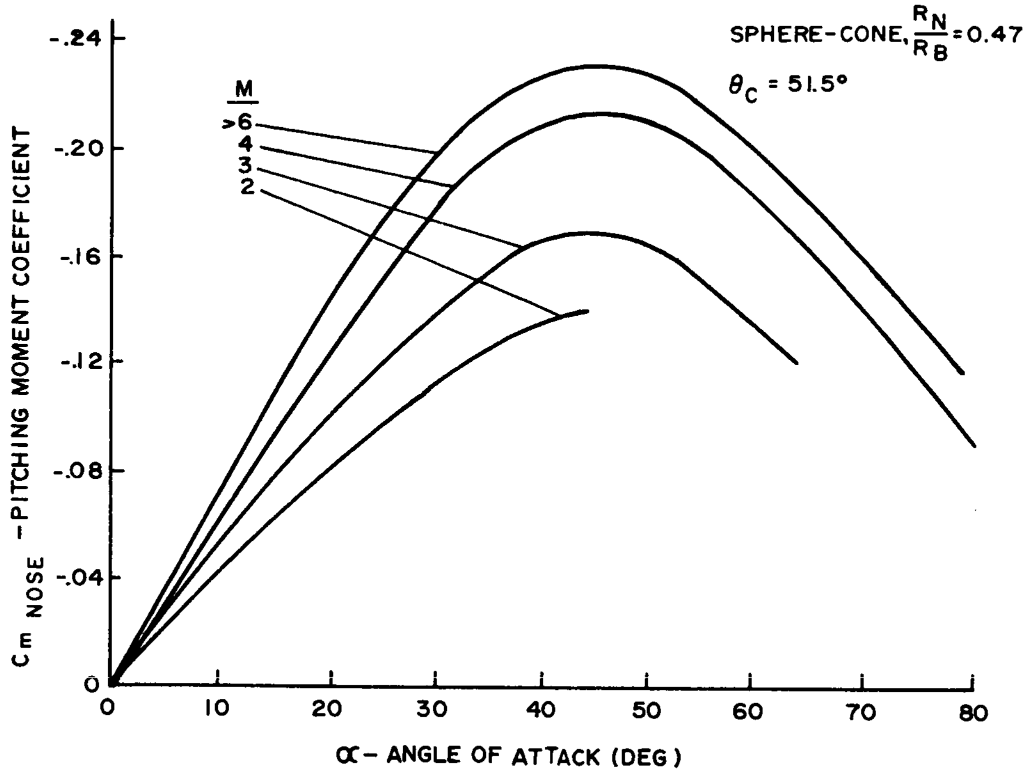


Figure 3.2-17. Variation of Pitching Moment Coefficient with Angle of Attack

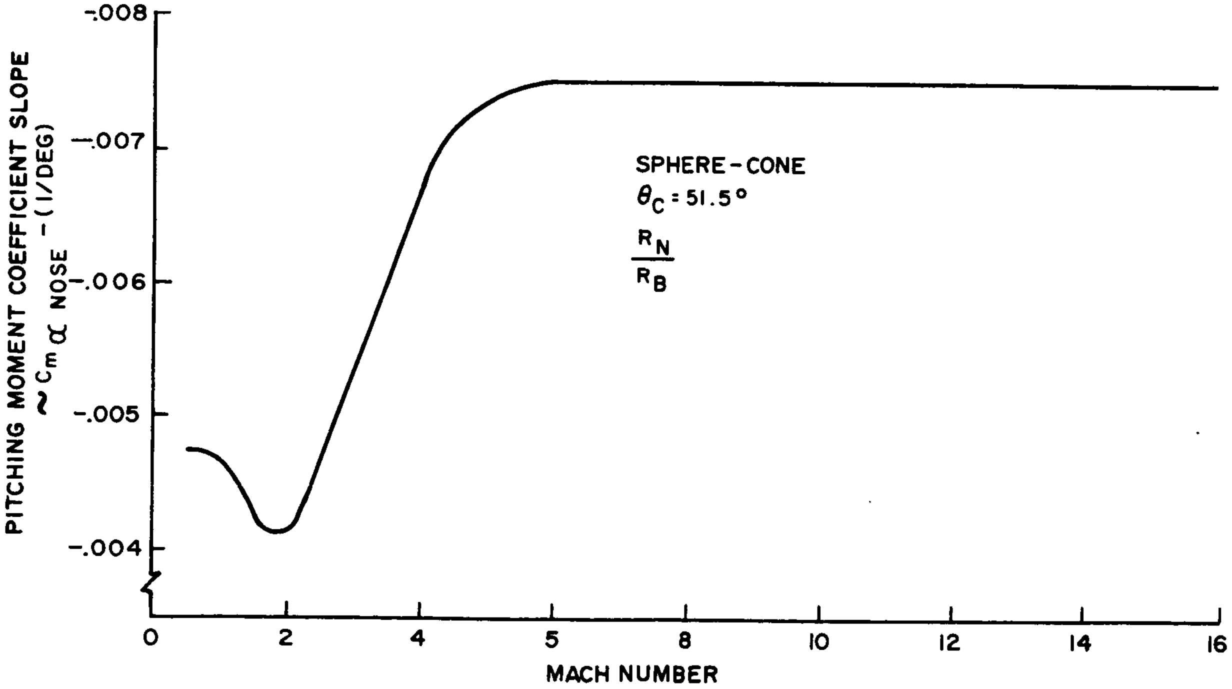


Figure 3.2-18. Variation of Pitching Moment Coefficient Slope with Mach Number

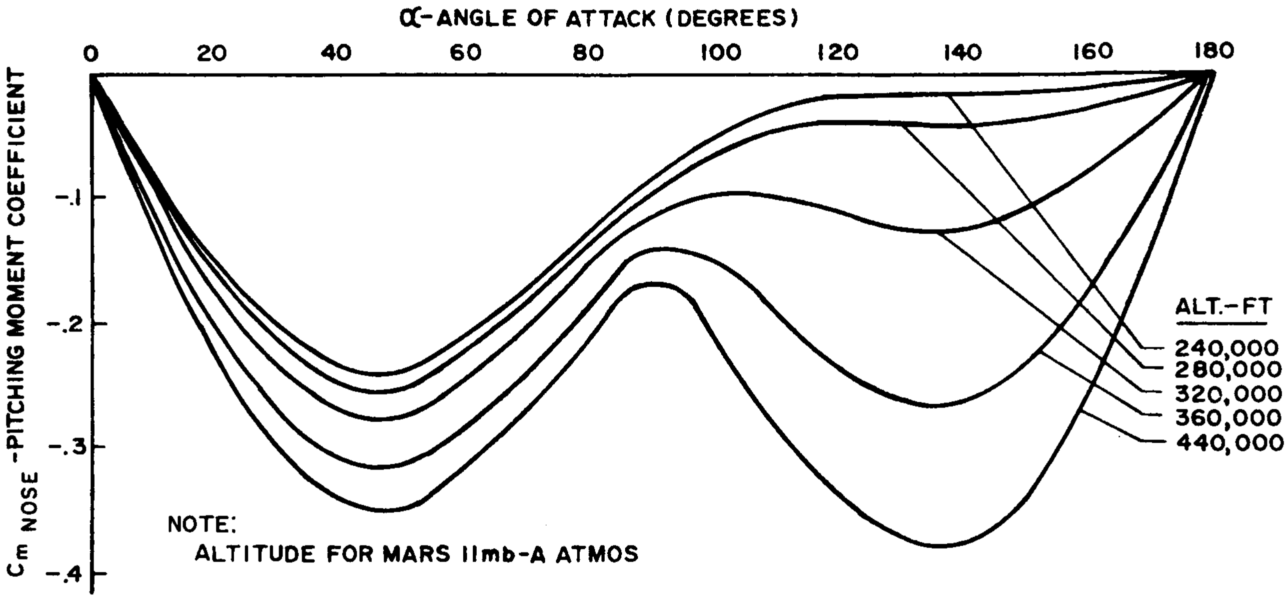


Figure 3.2-19. Variation of Pitching Moment Coefficient with Angle of Attack M > 10

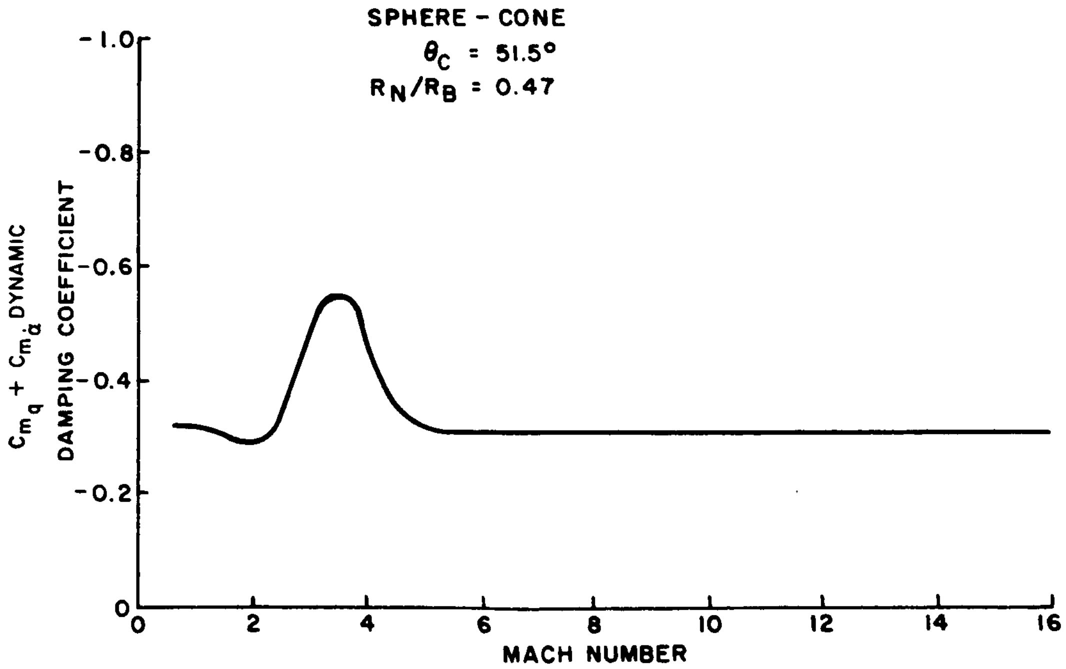


Figure 3.2-20. Variation of Dynamic Damping Coefficient with Mach Number

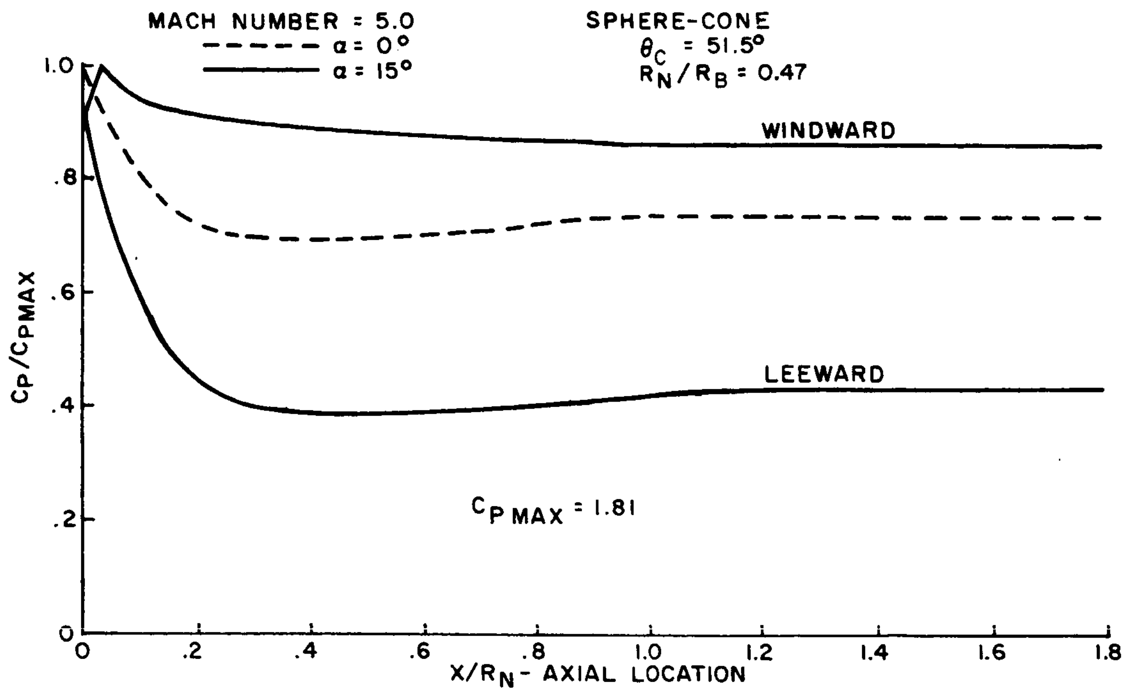


Figure 3.2-21. Pressure Distribution

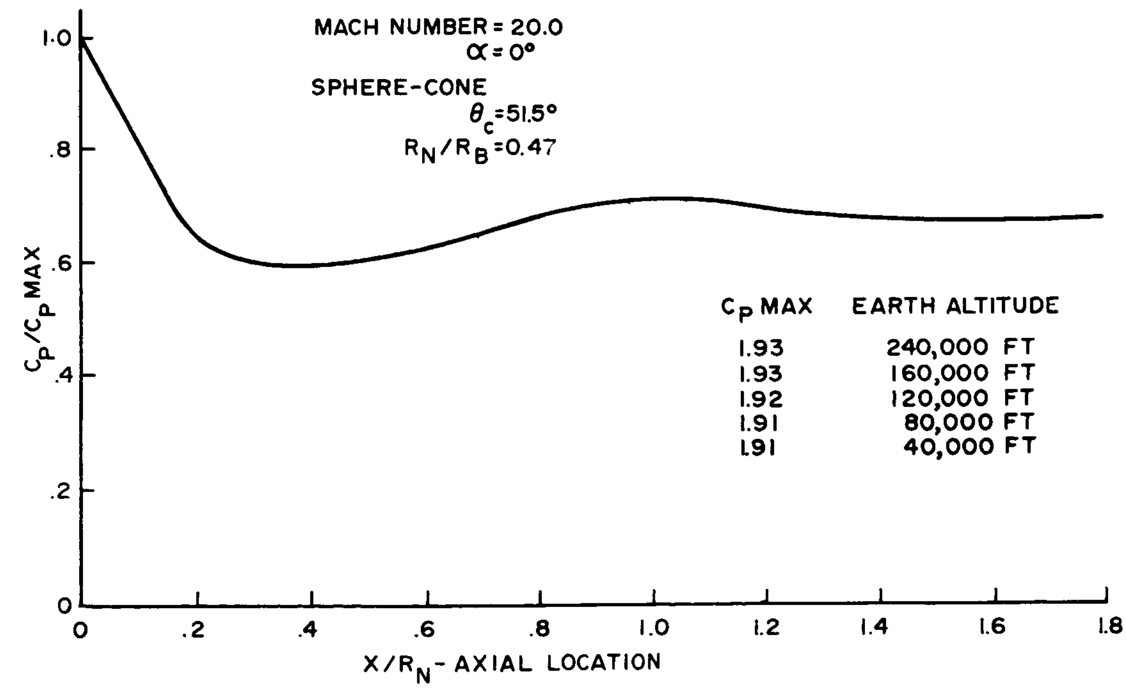


Figure 3.2-22. Pressure Distribution

c. References

(1) Voyager Design Study, Vol. IV, System Design, GE-MSD Document No. 63SD801, 15 October 1963.

(2) Mariner B Entry Vehicle, Vol. I, Technical Study, 26 November 1963.

(3) Mundy, A. M., Jr., "Mars Entry Vehicle Study Flight Mechanics and Control Section," GE-MSD, FMDM 238, 27 April 1962.

3. Entry Heat Protection

a. Summary

The external thermal environment of an entry vehicle with survival capability has been analyzed for entry in the Martian 11 mb-B model atmosphere. The entry vehicle considered was a sphere-cone configuration with a bluntness ratio of 0.47, nose radius of 2.62 feet, and a half-cone angle of 51.5 degrees. The specific trajectory analyzed was for entry at an altitude of 1,000,000 feet with a velocity of 21,000 feet per second, and an entry angle of 20 degrees. Stagnation and conical heating rates were evaluated using the results of Scala and Gilbert (Reference 1). The heat fluxes thus obtained and the thermal properties of Elastomeric Shield Material, were then used to obtain ablation and insulation requirements based on parametric one-dimensional conduction solutions with melting. Employing the results of the current and previous Martian entry studies (References 2, 3, and 4), an ESM heat shield with a 50 per cent ablation margin is presented for Martian entry conditions of:

$$W/C_DA = 15 \text{ psf}$$

$$20^o \leq \gamma_e \leq 90^o$$

$$u_e = 21{,}000 \text{ fps}$$

b. Entry Environment

The determination of heat shield requirements for a Martian entry mission is complicated by uncertainties in the prediction of Martian atmospheric characteristics. Within the scientific community, several atmospheric models have been advanced which differ both in chemical composition and physical structure. Of the numerous models available, aerodynamic heating during entry was investigated as part of the Voyager and Mariner B studies for the seven atmospheric models in Figure 3.2-23. Results from the Voyager and Mariner B studies (References 2 and 3) indicated that the Mars 135 mb (Upper Limit Model) and 11 mb-B model atmospheres present the most severe thermal environments. The heating environment was governed not only by absolute free-stream density levels, but also by the atmospheric density gradient through its effect on the trajectories. Consequently, heat shield requirements have been based on entry into the Martian 11 mb-B model atmosphere consisting of 65 per cent carbon dioxide and 35 per cent argon.

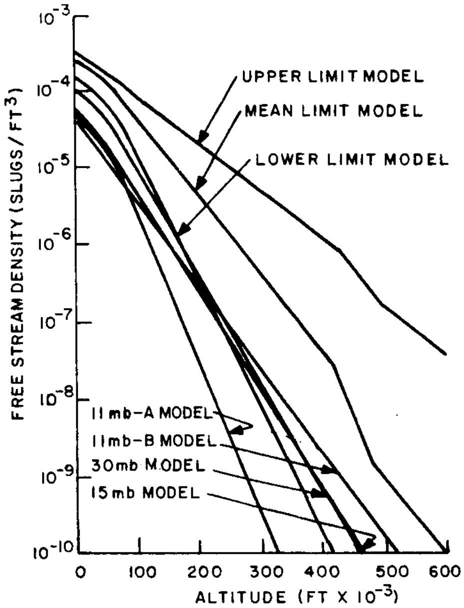


Figure 3.2-23. Martian Model Atmospheres

Stagnation-point aerodynamic convective heating was obtained through the heating parameter available in the Flight Mechanics Round Earth Point Mass Program. This heating parameter was developed by G. Walker (Reference 5) for application to Earth re-entry and is an approximate laminar stagnation point heat transfer equation of the form:

$$\dot{q}_L = f_L \; \rho_\infty^{\;0.5} \; u_\infty^{\;3}$$

where

$$f_L = \frac{1.67 \times 10^{-5}}{P_N^{\;2/3}} \left(\frac{\gamma_\infty - 1}{\gamma_\infty}\right)^{0.25} \left(\frac{\gamma + 1}{\gamma - 1}\right)^{0.25} \left(\frac{u_\infty}{a_\infty}\right)^{0.5} \left(R_N\right)^{-0.5}$$

In order to apply Walker's results to entry in a Martian atmosphere, a comparison was made with the results of S. Scala (References 1 and 6) which consider "the thermochemical effects of foreign planetary atmospheres upon hypersonic stagnation region laminar heat transfer." These results indicated a laminar heat transfer equation of the same form as Walker's equation, but with a correction for the molecular weight of the planetary gas being considered:

$$\dot{q}_L = C \; \rho_\infty^{\;0.5} \; u_\infty^{\;3}$$

Where C is a function of the molecular weight given as:

$$C = (9.18 + 0.663 \ \bar{m}_{\infty}) \ 10^{-10}$$

For application to the Martian 11 mb-B model atmosphere, the coefficients in both Walker's and Scala's heating parameter were evaluated. Walker's equation was evaluated at free-stream conditions of $\gamma_{\infty} = 1.4$, $\gamma = 1.2$*, Pr = 0.72 and $T_W = 560^{o}R$, whereas Scala's equation was evaluated for the Martian 11 mb-B atmospheric model with an ambient free stream molecular weight of 42.6.

Walker (Earth)

$$\dot{q}_L \ R_N^{0.5} = 3.16 \times 10^{-9} \ \rho_{\infty}^{0.5} \ u_{\infty}^{3}$$

Scala (Mars)

$$\dot{q}_L \ R_N^{0.5} = 3.74 \times 10^{-9} \ \rho_{\infty}^{0.5} \ u_{\infty}^{3}$$

From a comparison of the two equations it is apparent that Walker's heating parameter may be readily applied to Martian entry, provided his constant is increased by a factor of 1.18. The ratio of local to stagnation heating for a point on the skirt of the vehicle ($X/R_N = .75$), was obtained based on a Lee's hemispherical distribution using a Prandtl-Meyer pressure distribution. Local laminar heating for the skirt point was then obtained by multiplying stagnation heating by the skirt/stagnation heating ratio.

The earlier Beagle study for a similar geometry ($R_N = 7.73$ feet) indicated the highest integrated heating occurred for an entry angle of 20 degrees, even though entry at an angle of 90 degrees resulted in primarily turbulent heating. In addition peak radiative heating for a 90 degree entry was found to only be about 30 percent of the convective value, with a smaller ratio pertaining to a 20 degree entry angle. For the current vehicle with the smaller nose radius, aerodynamic convective heating at given body stations increased. However radiative heating would be considerably reduced by the reduction in nose radius, and the decreased wetted length at given X/R_N values would delay transition to turbulent heating. Consequently, heat shield design for the current vehicle based on laminar heating during a 20 degree entry, is even more justified than in earlier studies.

*Ratio of specific heats behind normal shock wave.

c. Heat Shield Design

Having determined the heating for the design entry conditions, it was necessary to apply the results to heat shield design. The previously referenced studies indicated Elastomeric Shield Material as suitable for both ablation and insulation protection during Martian entry and indicated the material properties of Table 3.2-5 as being applicable to Martian entry. Reference 7 has indicated heats of degradation in the 8000-10,000 Btu/lb range for ESM with honeycomb in a low shear environment. ($0 \leq \tau \leq 2$ psf). Corresponding flux levels in the referenced document were $3.5 \leq \dot{q} \leq 6$ Btu/ft^2 sec, with a stagnation enthalpy of 4500 Btu/lb, and $80 \leq \dot{q} \leq 120$ Btu/ft^2 sec at a stagnation enthalpy of 13,000 Btu/lb based on tests in supersonic and hypersonic arc tunnels. Previous studies indicated that the shear levels to be encountered for the type of entry being considered would fall within the quoted range. Although ESM performance has been found to decrease with increasing shear levels, the ablation safety margin recommended in the design (see Table 3.2-6) is deemed adequate to account for any decrease in performance at steeper entry angles. Using the ESM properties in Table 3.2-5, ablation and insulation requirements were determined for the stagnation point and a point on the cone (X/R_N = .75) using the GE-RSD One-Dimensional Conduction Solution Program.

TABLE 3.2-5. MATERIAL PROPERTIES (ELASTOMERIC SHIELD PROPERTIES)

Density (lb/ft^3)	40
Specific Heat (Btu/lb^{o}F)	0.34
Thermal Conductivity (Btu/hr-ft^2-^{o}F/ft)	0.000026
Heat of Ablation (Cold Wall) (Btu/lb)	7560
Degradation Temperature (^{o}F)	1240

d. One-Dimensional Conduction Solution

To provide a rapid means of determining ablation and temperature response during entry, GE-RSD developed the one-dimensional conduction melting solution (Reference 8). This program solves the one-dimensional heat-flow equation implicitly, using material thermal properties which are considered a function of temperature. Assuming that heat flows normal to the outer surface and no heat flows past the last layer, a heat balance is obtained across each node. During melting, the surface boundary condition is changed to

TABLE 3.2-6. INSULATION AND ABLATION REQUIREMENTS

	Axial Location	Ablation	Ablation & 50% Margin	Insulation	Insulation Ablation & 50% Margin
SPHERE	0	.131	.197	.220	.417
	.054	.115	.173	.235	.408
	.109	.092	.138	.240	.378
	.163	.085	.128	.245	.373
CONE	.218	.080	.120	.250	.370
	.397	.067	.100	.250	.350
	.576	.059	.088	.250	.338
	.750	.053	.080	.250	.330
	.935	.049	.074	.250	.324
	1.110	.045	.068	.250	.318

All dimensions in inches, based on peak ESM backface temperature of 300^{o}F.

account for the phase change. The program as described provides temperature and ablation histories, as well as temperature profiles through the shield.

Using this program and predicted cold wall heat fluxes, ablation and temperature response were determined for several different thicknesses of ESM for the body points previously indicated (Figure 3.2-24). Stagnation and conical insulation requirements were evaluated and uniformly tapering insulation applied over the spherical section, with a constant insulation on the conical section. Off-stagnation-point ablation thicknesses were obtained by multiplying total stagnation-point ablation by the ratio of off-stagnation heating at the point in question to stagnation-point heating. Due to uncertainties in heating and material properties, ablation values have been increased by a 50% margin to allow for a design safety margin.

Total shield thickness presented in Table 3.2-6 has been obtained by adding local ablation requirements (including 50 percent safety margin) to the insulation requirements on the sphere and cone.

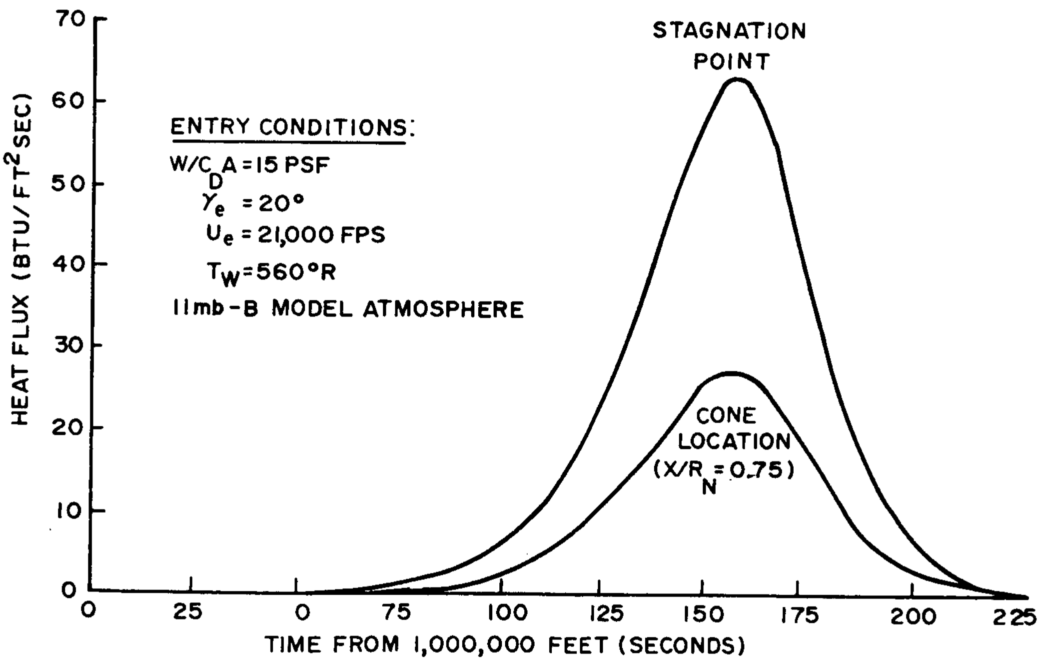


Figure 3.2-24. Martian Entry Approximate Aerodynamic Convective Heating

e. Conclusions

The heat shield design presented is suitable not only for an entry angle of 20 degrees, but for a 90-degree entry angle as well. Entry at steeper angles will result in higher heating rates and shear stresses, but at a lower value of total integrated heating. Although material performance may be less favorable than at the design conditions, the effective heat of ablation used was conservative, resulting in unchanged shield requirements for steeper entry angles.

f. Symbols

$\bar{m}$	mean molecular weight
Pr	Prandtl number
$\dot{q}$	heat transfer rate
R	radius
u	velocity
W/C_DA	ballistic parameter
x	axial distance from stagnation point, ablation thickness
y	radial co-ordinate of body location
γ_e	entry angle
ρ	density
μ	viscosity

g. Subscripts

e	entry conditions
L	laminar
N	nose
O, S	stagnation point
∞	free stream

h. References

(1) Scala, S., and Gilbert, L., "Theory of Hypersonic Laminar Stagnation Heat Transfer in Dissociating Gases," AETM No. 188, 11/1/62.

(2) Voyager Design Study - Vol. IV, "System Design," GE No. 63SD801, 10/15/63.

(3) Simons, R., "Mariner - B Proposal Input (External Thermodynamics)," Pir-ARSTA-8151-026, 11/5/63.

(4) Simons, R., "Heat Shield Analysis for a Large Martian Lander (BEAGLE), PIR-ARSTA-8151-027, 12/10/63.

(5) Walker, G. K., "Some Comments on Laminar and Turbulent Heat Transfer Equations," AETM 147, December 1959.

(6) Scala, S., "Hypersonic Heat Transfer in a Planetary Atmosphere During High Altitude Flight," Contributed to Voyager Study by High Altitude Aerodynamics Operation, 10/1/63.

(7) "Study of Elastomeric Shield Materials (ESM) for Apollo," RSD Prop. No. N30122, 2/5/63.

(8) Di Christina, V., "A One-Dimensional Conduction Melting Solution with Internal Air Gap," AETM No. 174, 11/18/60.

4. Structural Analysis

Structural analysis performed on the Voyager study consisted of determining structural weight and design characteristics for:

a. impact attenuation system

b. primary structural shell

   W/C_DA = 8, 15 psf

c. extensible-flare configuration

   W/C_DA = 15 psf

The studies of b and c above provide a sound basis of comparison between the standard and extensible flare vehicles in order to determine the payload penalty associated with the latter concept. Data has been extrapolated to provide results in parametric form. The impact attenuation system design and the shell structures are based on the configuration shown on Figure 3.2-25.

a. Impact Attenuation

Impact attenuation to limit the shock loads transmitted to the payload is provided through the use of fiberglass honeycomb crushable material. Fiberglass was selected because of its high specific energy absorption capacity and its transparency to radio frequency, a requirement set by the use of a radar altimeter. The studies assumed a ground slope of 30 degrees and a design surface wind of 40 mph although, parametric investigations were made to determine the effects of higher wind velocities.

The required impact material thickness for a 125 g shock limitation is shown on Figure 3.2-26 as a function of wind velocities for various descent rates. The energy absorption characteristics of fiberglass are shown on Figure 3.2-27. With these basic system design requirements and specific vehicle geometry, the total impact attenuation system weight as a function of vehicle weight and descent velocity is shown on Figure 3.2-28. The effect of lateral wind velocity on system weight is shown on Figure 3.2-29 for the 1830 pound (entry weight) Lander.

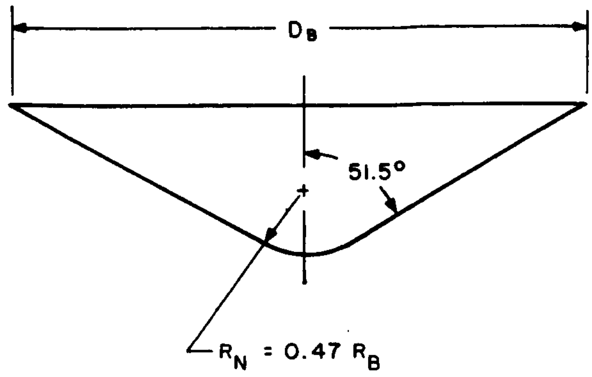


Figure 3.2-25. Voyager Geometry

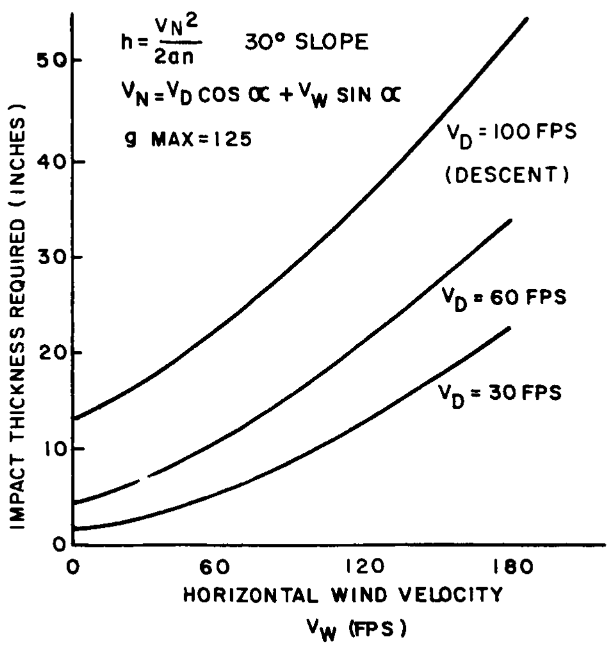


Figure 3.2-26. Impact Thickness Required

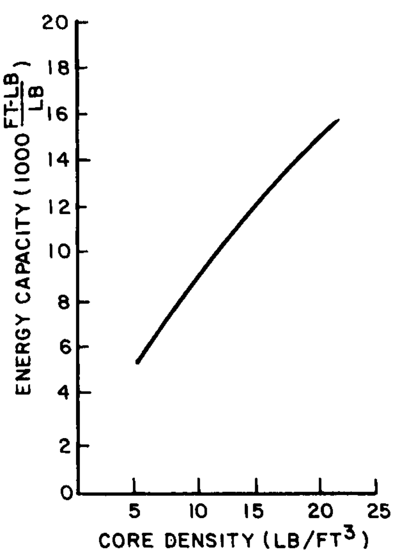


Figure 3.2-27. Energy Absorption Property of Fiberglass Honeycomb

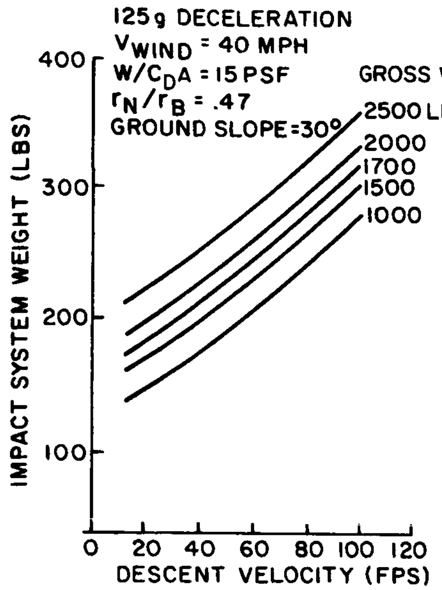


Figure 3.2-28. Voyager System Impact Weight

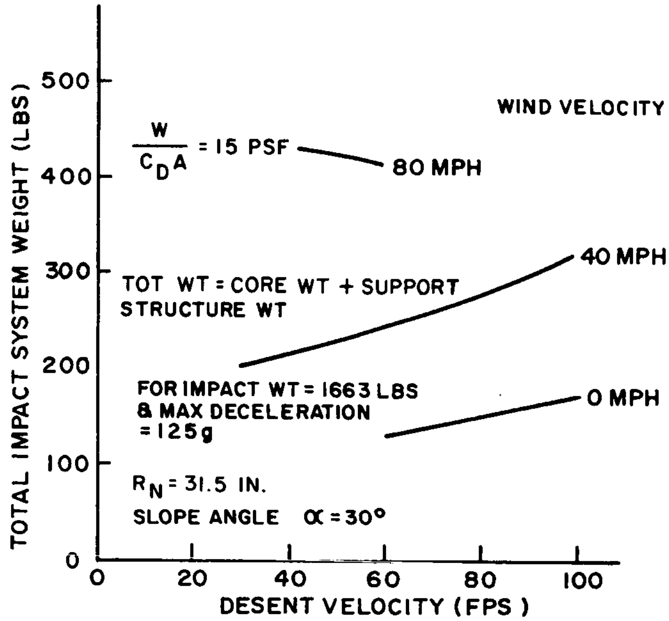


Figure 3.2-29. Total Shock Attenuation System Weight Versus Descent Velocity

b. Primary Structural Shell

Structural shell analyses were conducted for two values of W/C_DA; 15 and 8 psf. Both structures assumed a base diameter of 134 inches, the first W/C_DA representing the prime vehicle under study, and the W/C_DA = 8 psf representing the allowable ballistic parameter if the entry corridor ranged to a 90 degree path angle. The shell material, aluminum honeycomb sandwich, was selected as a result of tradeoffs performed on the earlier Voyager-Saturn 1B study. Peak load levels are based on a trajectory into the 11A mb atmosphere and from previous studies, a maximum outer temperature (backface temperature) of 300°F and an inner temperature of 100°F was selected. The sandwich face skins were determined from the yield condition whereas, the core depths were critical in shell buckling. The equivalent cylinder method coupled with the Garber-Hess analysis was used to determine the sandwich size and weights. Shell weight as a function of base diameter for a W/C_DA = 15 psf is shown in Figure 3.2-30. The weight breakdown for the W/C_DA = 8 psf vehicle is shown in Table 3.2-7. In all cases, a core density of 0.0035 pounds per inch^3 and a fabrication factor of 1.9 were used.

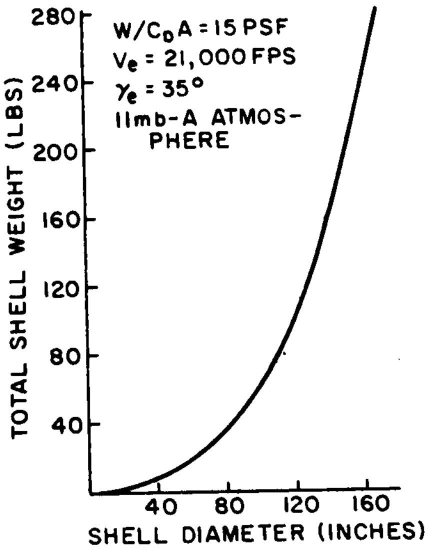


Figure 3.2-30. Structural Shell Weight

c. Extensible Flare Configuration

Several extensible flare configurations of various base diameters were analyzed to determine the total structural weights of these vehicles. A core vehicle diameter of 110 inches was chosen as the maximum compatible with the Titan IIIC interface and shroud limitation. Figure 3.2-31 shows a conceptual design of a Lander vehicle with an equivalent base diameter of 157 inches. Sizes and weights of flaps and the support structure have been determined for the configurations shown in Figure 3.2-32. Actuator system and associated hardware weights were estimated. Figure 3.2-33 shows the total structural weight of the flare section as a function of equivalent base diameter.

TABLE 3.2-7. SHELL ANALYSIS RESULTS FOR W/C_DA = 8 PSF
(Shell Diameter = 134 inch; Gross Weight = 975 lb)

	Nose	Fwd. Cone	Aft Cone
Shell Weight, lb	7.1	23.6	117.3
Total Shell Weight, lb		148	

5. Vehicle Retardation Analysis

a. Background and Requirements

The recent revision of the predicted Mars atmosphere from a nominal surface pressure of 85 mb to 15 mb has placed tremendous emphasis upon the selection and design of the

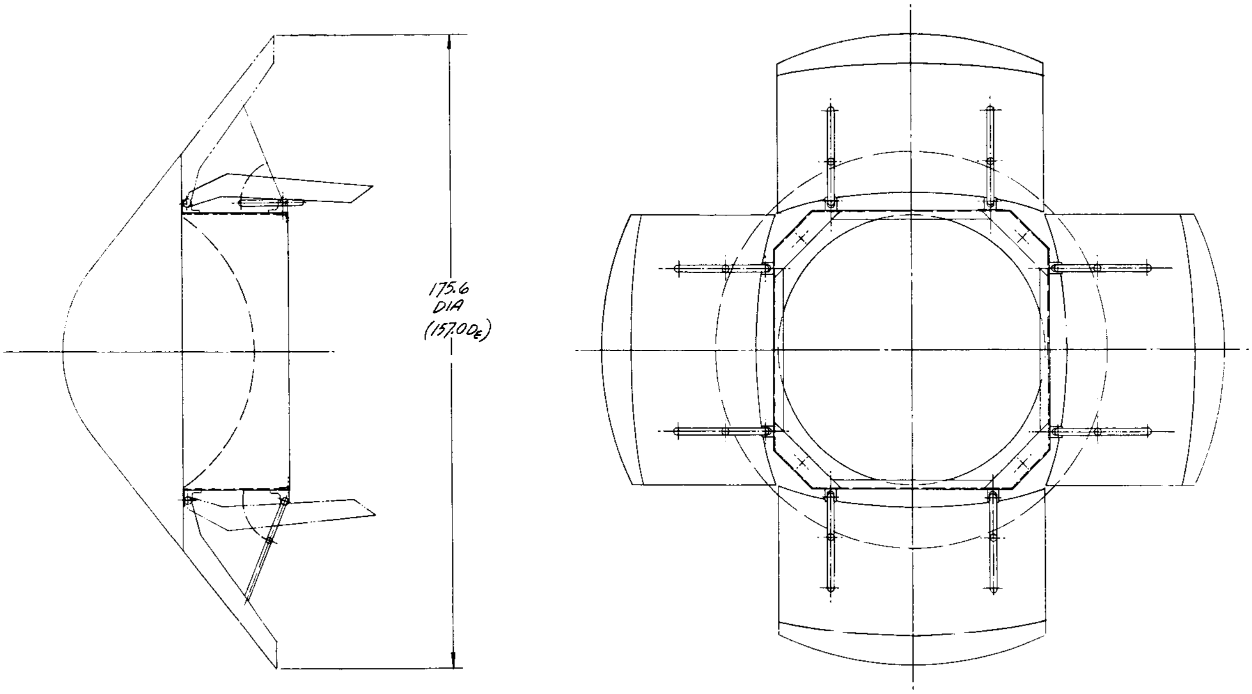


Figure 3.2-31. Extensible Flare Configuration

vehicle retardation system. Complicating the choice and design of a reliable and light-weight system is the uncertainty attached to the new surface pressure and density gradient. Until better definition is made either through preceeding Mariner probes, or Earth based observations, the retardation systems recommended for the Voyager Entry/ Lander must be designed to operate successfully throughout a range of predicted atmospheres.

The retardation system, as well as the vehicle trajectory analysis, has been based upon entry into the group of atmospheres based on Kaplan's work and issued in the JPL Inter-office Memo 313-1222; namely, group 2, atmospheres G-K. Density profiles of these atmospheres are shown on Figures 3.2-2 and 3.2-3. The 11 mb-A model represents the severest atmosphere from a retardation standpoint since it allows the deepest penetration into the atmosphere in the shortest amount of time.

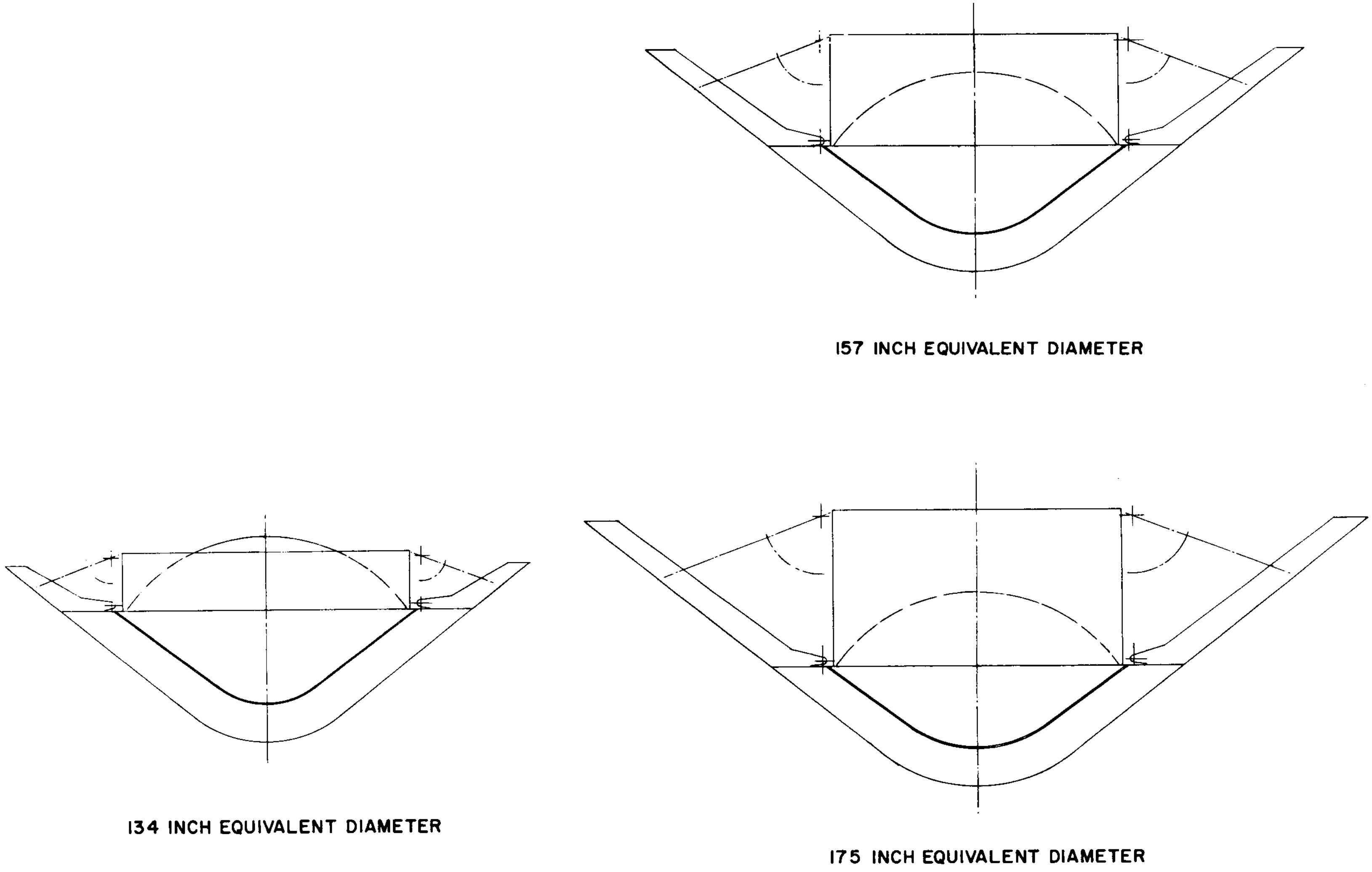


Figure 3.2-32. Extensible Flare Configurations

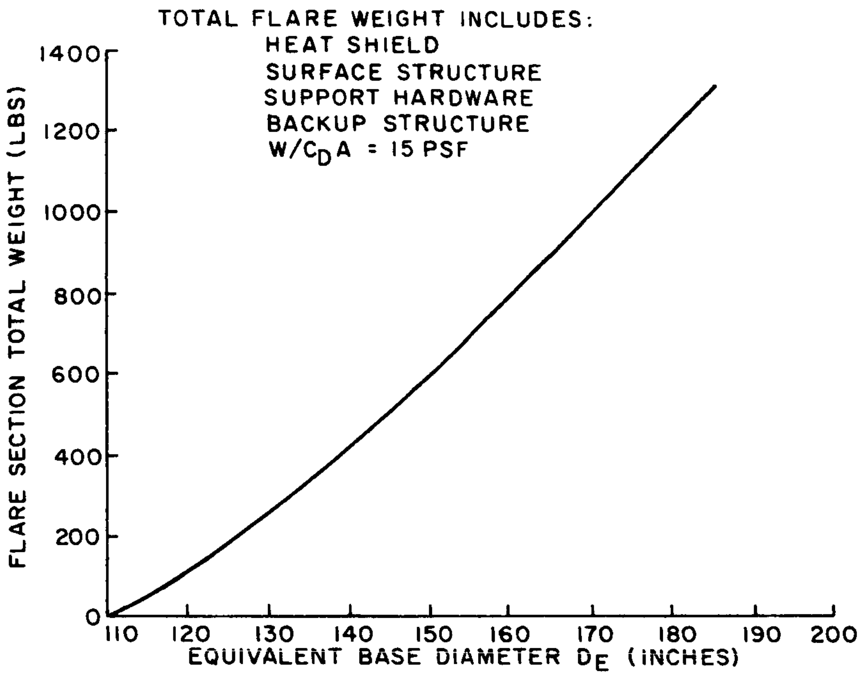


Figure 3.2-33. Extensible Flare Weight vs Equivalent Base Diameter

Although aerodynamic retardation removes a significant portion of the vehicle's velocity (90-98 percent), it is not sufficient to allow the vehicle to land on the surface of Mars at reasonable impact velocities. Terminal, or equilibrium velocities are very nearly reached in the 30 mb atmosphere regardless of the entry path angle, however, equilibrium velocity is approached in the 11 mb atmosphere only for very shallow entry path angles. A worst case entry, 90 degree path angle in the 11 mb-A atmosphere, results in an impact velocity of approximately 2700 feet per second. Hence, there is a requirement for supplementary retardation.

The selection of the vehicle's ballistic parameter is closely associated with the retardation system selection, as is the range of acceptable entry path angles. This subject is treated in Section 3.2.2(C)(2) and is based upon the selected retardation system which is presented elsewhere in this section.

b. Retardation Methods

Auxiliary retardation can occur at both high and low altitudes. Several high altitude retardation schemes were investigated (See Voyager Design Study, Volume IV) but were rejected because of inefficiencies, weight penalties, or unnecessary complexities. These included aerobraking, high altitude retrorockets, and trailing drag bodies. Investigation of low altitude retardation led to the choice of parachute systems as being a highly reliable and competitive method of assuring safe landing of the Entry/Lander.

Studies have been conducted on the Voyager - Titan IIIC program to investigate the merits of several candidate parachute systems. These include the combination of one or two parachutes, shock attenuation material, and possible use of low altitude (terminal) retrorockets. The parachute's function is to remove the bulk of the vehicle's descent velocity and serve to stabilize and orient the Lander for correct attitude at impact. The shock attenuation material, when properly placed, limits to some predetermined value the shock loads that are transmitted to the payload. For these low density atmospheres, it is found that retrorockets used to remove any residual descent velocity greatly reduce the system weight.

A landing system can be optimized based on weight and/or volume. For this study, the optimization was based on weight with a secondary objective of maintaining a reasonable impact attenuation stroke. The systems which have been evaluated are all based on an 1830 pound Lander and are as follows:

- Case A. Supersonic decelerator parachute + Main parachute + shock attenuation material
- Case B. Supersonic decelerator parachute + shock attenuation material + constant thrust retrorocket
- Case C. Supersonic decelerator parachute + main parachute + shock attenuation material + constant thrust retrorocket
- Case D. Supersonic decelerator parachute + shock attenuation material + controllable retrorocket.

For Cases B, and C above, the system was synthesized by allowing the main parachute and retrorocket combination to nominally provide zero impact velocity in the 30 mb atmosphere and sizing the impact attenuation material to absorb the energy resulting from the residual descent velocity which occurs in the lower density 11 mb atmosphere. For Case A, the impact attenuation material is sized to absorb the residual impact energy resulting from the main chute equillibrium velocity.

In Case D, it was assumed that a velocity sensor could be provided which could control a throttlable rocket which would essentially give a zero impact velocity in any atmosphere encountered throughout the 11 to 30 mb range. Enough attenuation material was added to absorb energy resulting from the effects of a 40 mph cross wind.

The impact attenuation material has been selected on the basis of previous work and consists of crushable fiberglass honeycomb material located between the heat shield and substructure of the vehicle. In all cases, an impact shock limitation of 125 g was chosen. Additional Mars surface conditions were imposed in the form of a 40 mph cross wind and a local terrain slope of 30 degrees.

For the two-parachute system, the purpose of the decelerator chute is to quickly decelerate the vehicle to Mach 1.0 so that the larger main parachute can be reliably deployed. The decelerator is sized to decelerate from Mach 2.5 to Mach 1.0 in a maximum of 10,000 feet. It is felt that decelerator parachutes capable of being deployed at Mach 2.5 or greater will be available for a Voyager launch in the late 1960s. Parachutes such as the Hyperflo are currently under development and show strong promise as high speed decelerators.

Figure 3.2-34 shows the retardation system weight as a function of impact velocity (which, in this case is the same as terminal velocity on the parachute) in the 11 mb atmosphere for Case A, i.e., two chutes + crushup. Minimum weight for this system is approximately 460 pounds. However, the minimum weight system requires a crushup stroke of about 27 inches which is considerably higher than can be accepted for vehicle design.

Figure 3.2-35 presents retardation system weight versus impact, or residual velocity for Case B; one decelerator chute, crushup and retrorocket combination. Minimum weight is approximately 450 pounds requiring an impact stroke of 22 inches. The system weight versus impact velocity for the two parachute-crushup-retrorocket concept (Case C) is shown in Figure 3.2-36. Minimum weight of approximately 364 pounds occurs at an optimum impact velocity of 45 fps in the 11 mb atmosphere. Associated with this impact velocity is an impact thickness of 7.5 inches.

Case D was studied to show what the minimum possible weight system would be using one parachute and a controllable retrorocket to give a nominal zero impact velocity. In this case, the only crushup material which was provided gave secondary protection against a possible 40 mph cross wind. The system weight as a function of terminal velocity (since impact velocity is zero) is shown on Figure 3.2-37.

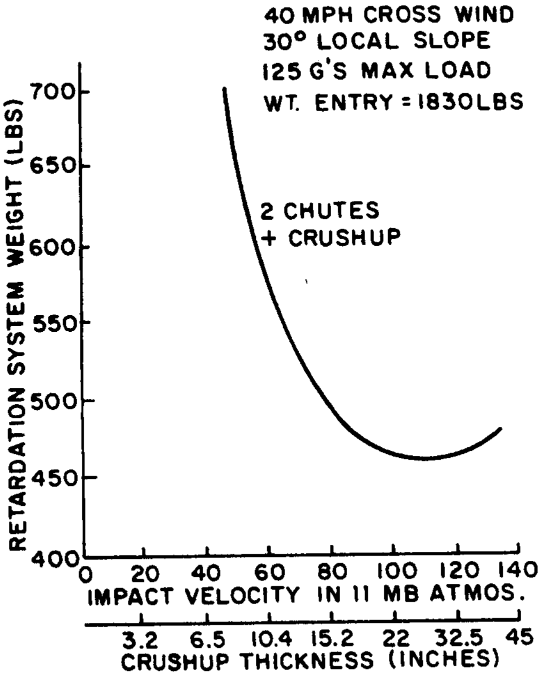


Figure 3.2-34. Retardation System Weight Optimization

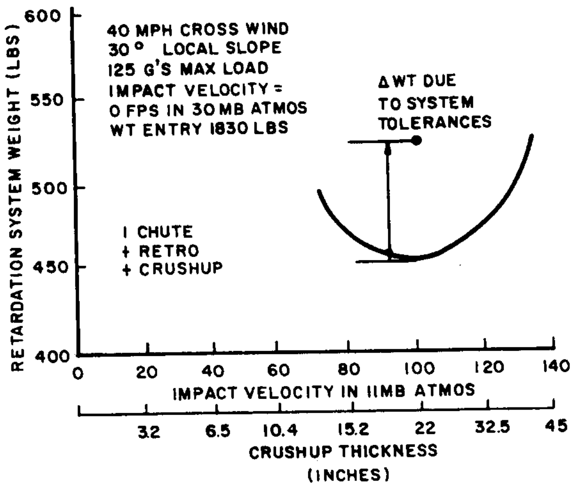


Figure 3.2-35. Retardation System Weight Optimization

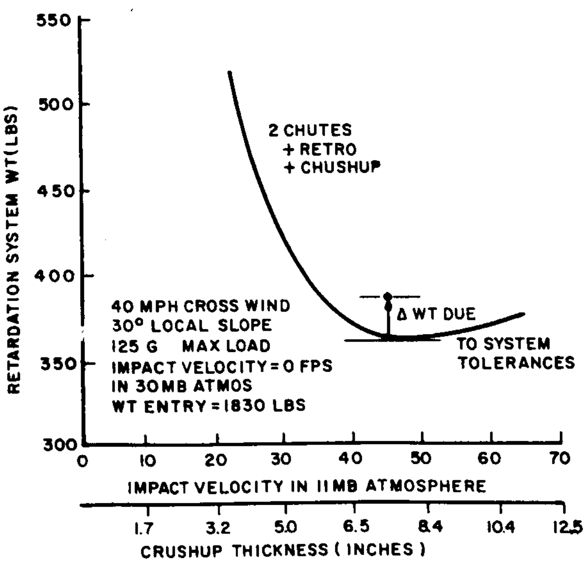


Figure 3.2-36. Retardation System Weight Optimization

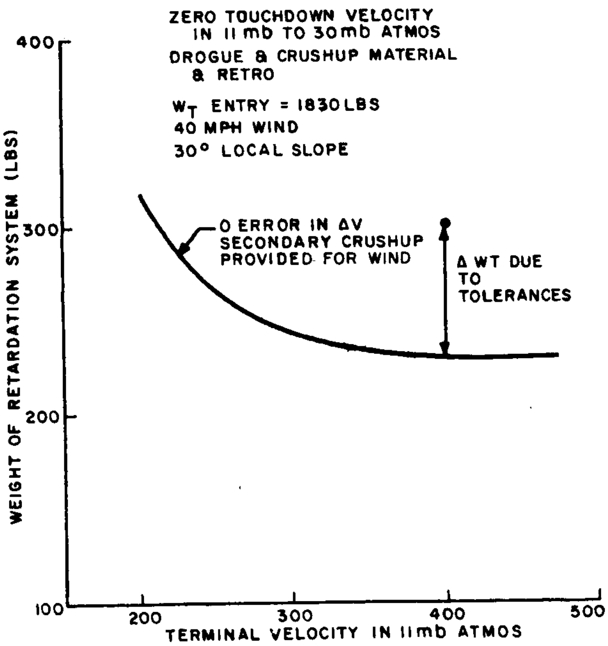


Figure 3.2-37. Retardation System Weight Optimization

c. Error Analysis

The above weight tradeoffs are based on nominal descent and landing operation. It is realized, however, that the use of retrorockets and the associated altitude sensing and initiation procedures will give rise to tolerances on these systems. It is of interest to investigate these tolerances and to apply their effects on the systems for a more realistic comparison.

The following tolerances have been selected for the purposes of identifying typical system effects.

Retrorocket burning time	$\pm$10%
Retrorocket total impulse	constant
Retrorocket initiating altitude	$\pm$1 ft or 3% whichever is greater
Parachute descent velocity	$\pm$3 fps

Two combinations of these tolerances provide the largest increases in impact velocity. They are 1, the combination of longer burning time (+10 percent), lower initiating altitude (-1 ft or -3 percent) and a higher descent velocity (+3 fps) that result in incomplete burning at impact and 2, the combination shorter burning time (-10 percent), higher initiating altitude (+1 ft or +3 percent), and decrease descent velocity (-3 percent) which gives retrorocket burn out before reaching the surface. The increase due to these two combinations of tolerances is given in Figure 3.2-38 as a function of nominal retrorocket initiating altitude for the 1830-pound prime Entry/Lander configuration. Considering the upper curve at any initiating altitude it can be seen that an initiating height of approximately 90 feet results in the smallest increase in impact velocity. The increase in impact velocity requires additional crushup material thus resulting in a larger landing system weight. This increase for the 1830-pound Lander, using a Case C system, is presented in Figure 3.2-39 as a function of initiating altitude. The total weight including tolerances is 388 pounds at an initiating height of 90 feet, as compared to the system weight of 363 pounds if the tolerances are neglected.

A realistic comparison can now be made of the various systems under consideration. Figure 3.2-40 shows three discrete points representing the minimum weight retardation systems for each of the alternatives considered which do not use retrorockets. An optimization curve is shown for Case A. Each system, except Case A, has been adjusted to include the effects of system tolerances. Comparison can now be made on a weight basis.

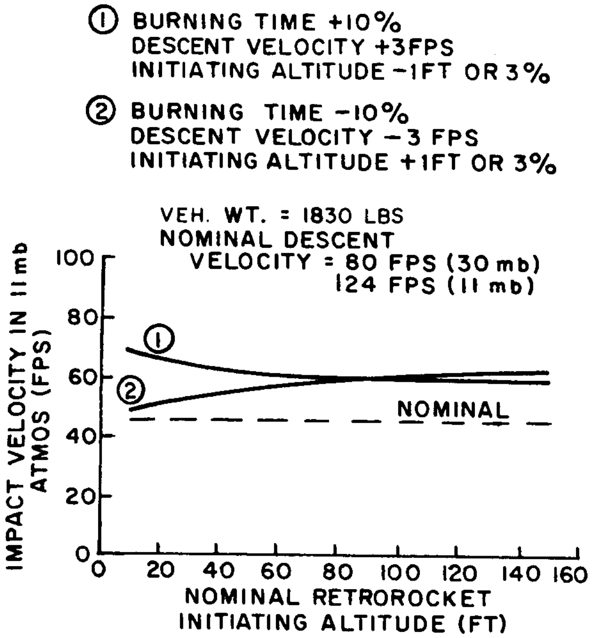


Figure 3.2-38. Effect of System Tolerance on Impact Velocity

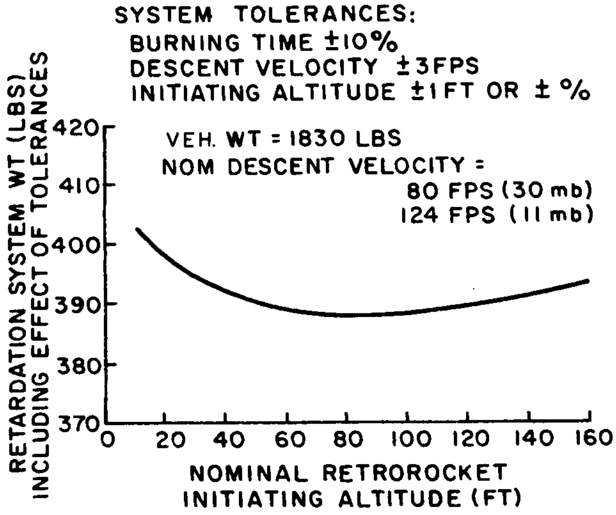


Figure 3.2-39. Retardation System Weight versus Retrorocket Initiating Altitude

d. System Comparison & Selection

Figure 3.2-40 shows that the retardation system utilizing a controllable retrorocket is the lightest weight system available. However, this system is not recommended because of the added complexities associated with the throttlable rocket, sensing and discrimination components and their potentially degrading effect on reliability. The sophistication of this system might possibly look attractive for post Voyager missions, particularly if the atmosphere is found to be 11 mb or lower.

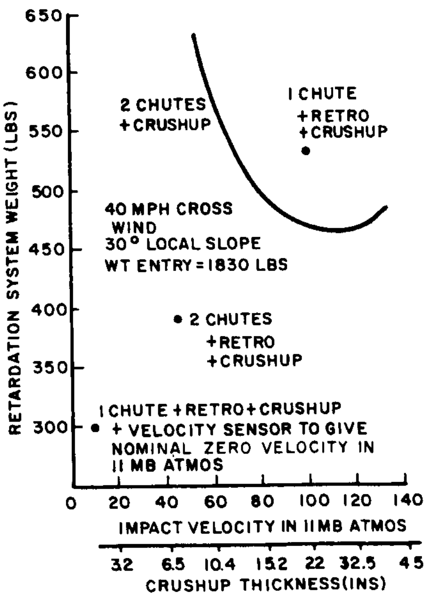


Figure 3.2-40. Retardation System Comparison

The two chutes-retrorocket and crushup system combines the best features of the previous case in that it utilizes the efficiency of the retrorocket to decrease velocity near the surface without resorting to a complex controllable rocket. Initiation and sensing can be provided by the radar altimeter which is already a part of the payload. This system when compared to the one chute-retrorocket-crushup scheme, looks attractive from a weight standpoint. It should also be pointed out that the optimum impact velocity for the latter system requires a crushup thickness of nearly 22 inches which becomes unreasonable for vehicle design. If a shorter crushup stroke is chosen, say 15 inches which compares to an impact of 77 fps, the system weight would increase to approximately 475 pounds without tolerances which accentuates the choice of Case C. Although the addition of the second parachute requires an additional sequence operation, this is not considered a serious drawback because of the accumulated experience of successful operation on present vehicles.

Comparison of the two chutes-retrorocket-crushup system with its equivalent without retrorockets (Case A) provides an even stronger case for the former system. Again, crushup thickness is unrealistic for design purposes and a shift to lower impact velocities would have to be made. If, for example, the thickness of crushup were limited to 12 inches the weight for Case A according to Figure 3.2-40 increases to approximately 535 pounds.

On the basis of the comparisons made in this section, and upon the results of previous studies, the recommended retardation system is the two parachute-retrorocket-crushup combination.

e. Materials

The subject of materials selection and materials application for Mars entry vehicles has been fully covered in the Voyager Saturn 1B study. Since the materials selected for application on the Titan III-C design are identical to those previously selected, the interested reader is referred to Volume IV, System Design, Section 1.3.7. It can be stated in summary that no insurmountable materials problems have been found although in some areas, extensive development and evaluation work is required.

D. VEHICLE DESIGN

In this section will be found summary statements describing the various subsystems which make up the Entry/Lander vehicle. Emphasis has been placed on the prime configuration, that is, a vehicle weighing 1830 pounds at entry having a base diameter of 134 inches. Contained in subsection 3.2.2D-9 is an overall description of the prime configuration, and two alternates, namely, the extensible flare concept and a Rover carrying Lander.

1. Retardation System

The recommended retardation system has been based on the following major design constraints:

1.	Vehicle Entry Weight:	1830 lb
2.	Atmospheric Surface Pressure Range:	11 mb to 30 mb
3.	Impact g Load Limit:	125 g max.
4.	Maximum Decelerator Deployment Condition:	Mach 2.5

To accomplish retardation with state-of-the-art devices, a parachute system, solid-fuel retro-rocket, and honeycomb crush-up material is recommended. The parachute system includes a Hyperflo supersonic decelerator to reduce the vehicle velocity from supersonic to subsonic and a main chute to obtain the required terminal descent rate. This descent rate will be further reduced immediately prior to impact by a solid fuel rocket motor. Parachutes and retrorocket have been optimized to produce a zero impact velocity in the 30 mb atmosphere. Sufficient honeycomb crush-up material has been incorporated in the vehicle design to reduce impact load levels below 125 g for impact velocities resulting from the residual velocity in the 11 mb atmosphere and rocket sensing and firing tolerances.

A summary of the retardation system specifications follows:

Hyperflo Decelerator:

Diameter	18.5 ft
No. Req'd	1
Weight	27 lb
Pack Volume	.97 ft^3
C_DA	134 ft^2

Ringsail Main Chute:

Diameter	72 ft
No. Req'd	1
Weight	66 lb
Pack Volume	.97 ft^3
C_DA	134 ft^2
Pack Volume	2.35 ft^3
C_DA (Reefed)	404 ft^2
C_DA (Open)	2040 ft^2

Crush-up Material

Material	Fiberglass honeycomb

Retrorocket

Fuel	Solid
Weight	41 lb
I_{sp}	230 sec

Additional hardware items include

- Decelerator ejection mortar
- Mortar squib and charge
- Main chute reefing cutters
- Cover release bolts
- Programmer with g sensors and timers
- Battery power supply
- Main chute swivel
- Decelerator and main chute cut-off fittings.

2. Functional Description

The retardation system is self-operating except for two electrical signals that must be provided by other vehicle subsystems. The first is a "battery-activate" signal prior to the time the vehicle enters the atmosphere; a second signal must be provided by the radar altimeter/atmosphere sensor to ignite the retrorockets immediately prior to impact. This latter event is particularly critical since the proper time-distance relationship must be achieved to obtain the required impact velocity.

The primary events which must be completed to accomplish the entire sequence of events are as follows:

(a) Battery Activate Event - At the time of entry, an electrical signal activates the battery power supply. The battery is selected with sufficient capacity to operate electronic circuitry within the recovery programmer as required to complete the sequence. The regular RTG power source serves as a back-up.

(b) Arm Event - As the vehicle enters the planet atmosphere, deceleration is experienced which is utilized to apply battery power to the programmer electronic circuitry. Arming is accomplished by activation of an inertial switch which permanently closes an arming relay, applying voltage to the appropriate elements of the programmer. The g-load level selected for arming will be below the minimum peak value expected for all possible trajectories.

(c) Hyperflo Decelerator Ejection Event - An electrical signal is provided by the programmer which ignites an explosive charge to eject the Hyperflo chute pack from its ejection mortar. The method of sensing this event will be largely dependent on the accuracy at which the entry trajectory can be controlled and the expected atmospheric conditions. For a defined trajectory, this could be sensed simply by means of an inertia switch and time-delay as may be required to provide an ejection point above the minimum altitude required for deceleration and below the maximum chute deployment Mach number. More complex sensing methods would be required where wide variations in entry angles and atmospheric density profiles are expected. A back-up signal will be provided by the radar altimeter at 20,000

(d) Decelerator Drag Interval Time Delay Event - Simultaneous with the Hyperflo decelerator ejection event, a time-delay is activated which prevents release of the decelerator (and hence delays main chute deployment) until the vehicle has reached subsonic velocity.

(e) Decelerator Release - At the end of the decelerator drag interval time delay, a signal is provided by the programmer to activate the release bolts which tie the decelerator riser to the vehicle structure. This riser is mechanically connected to the main chute. Aerodynamic forces acting on the decelerator cause it to move aft of the vehicle, extracting the main chute pack from the stowage compartment. When the main chute pack is separated from the vehicle at a distance equal to its suspension line length, the deployment bag is stripped from the canopy and inflates to a reefed configuration.

(g) Reefed Main Chute Event - The main chute canopy is reefed to control opening loads and reduce the vehicle velocity to a point where dynamic pressures will permit full canopy inflation. Mechanically activated reefing cutter are provided on the main chute canopy to control the reefed drag time interval. An opening load balance may be obtained by proper sizing of the reefing line length (and hence reefed drag area) as well as by incorporating more than one reefing stage if required.

(h) Disreefing Event - After the reefing time interval has elapsed, the reefing cutters sever the reefing line on the main chute canopy skirt, allowing the canopy to inflate to its fully open configuration. The vehicle then decelerates to its equilibrium descent velocity.

(i) Retro-Rocket Extraction Event - During main chute extraction, the rocket motors are released and separated from the structure. This is accomplished by rigging the main chute suspension lines such that under steady-state descent conditions, these lines extend with the rocket motors attached. The separation distance between the motor and the vehicle will be dependent on clearance required to divert the nozzle exhaust such that the hot gasses will not impinge on the aft face of the vehicle.

(j) Retro-Rocket Ignition Event - Immediately prior to impact, the retro-rockets are ignited to accomplish a velocity reduction. With the proposed chute system, this event will occur 90 feet above the impact point.

(k) Impact and Main Chute Release Event - At impact, the main chute is released from the vehicle structure to prevent the uninflated canopy from falling on the vehicle and obstructing optical sensors or radio transmission. Residual velocity due to tolerances of the rocket system is absorbed by the honeycomb crush-up.

The final event, as defined above, will require further study to determine a satisfactory method of preventing the main chute canopy from falling on and covering the vehicle. Directing the canopy away from the vehicle may be accomplished by spilling the entrapped air in such a way as to produce a side force component, or possibly a small rocket can be used to carry the canopy away from the vehicle after it has collapsed. Also, it is important that some provision be made either for propelling the expended retro-rocket case to one side or for supporting them by some structure to prevent dropping on the aft face of the vehicle.

3. Descent Times

Time to impact is of importance from a communications standpoint. Emergence from communication blackout has been estimated at a minimum of 130,000 feet. Figure 3.2-41 shows descent time from end of blackout to impact. Also shown are comparable times should only a portion of the retardation system function.

4. Power Supply

Power is supplied for all electrical functions of the Lander and spacebus by a radioisotope thermoelectric generator and a rechargable Nickel-Cadmium battery. The RTG, size at 170 watts, is the primary source of power and is used throughout the entire mission. In certain instances, short bursts of power are required greater than the RTG output and in these cases, the battery will supplement the RTG. Typical of a high power mode is the direct link transmission of data through the omni antenna during Lander

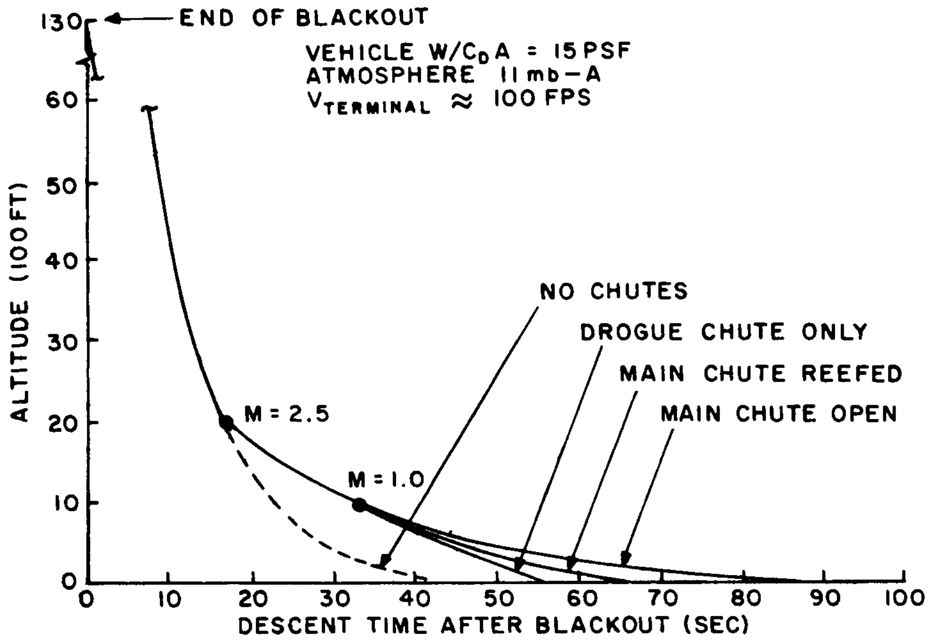


Figure 3.2-41. Descent Time

descent. Peak loads also occur when soil sample drilling is performed or, if the omni directional surface antenna is used as a backup.

The RTG was chosen as the primary power supply because of its inherent reliability, its long life capability, and the function it plays in the thermal control system. The fuel recommended for the 1971 flight is Curium 244. The RTG is cooled conductively by the thermal control system until the vehicle has landed on Mars. Cooling is then accomplished through natural radiation to the Mars atmosphere.

The selected battery is an 8 ampere-hour, heat sterilizable unit which is trickle charged from the RTG during the transit phase. Temperature control is maintained through a two-phase wax envelope. The power supply subsystem also includes a charge regulator and the power conversion and control unit. Specific details of the power supply subsystem are contained in Section 4.3.

5. Communication

The telecommunications subsystem recommended for the 1971 Bus/Lander uses a direct Earth link operating at 2300 mc. This system will serve the dual role of in-transit communications and telemetry requirements during the approach, descent, and surface mission modes.

Once separated from the spacebus, the Lander will periodically transmit engineering information and event signals. Transmission is through a quarter wave length turnstile antenna located on the aft cover of the vehicle. This antenna is also used throughout the entry trajectory for telemetry of scientific and diagnostic data. Because the antenna must survive entry heating, it is encapsulated in foam and coated with a transparent covering. The encapsulation will allow a minimization of breakdown which is due to the low ambient pressures.

Surface communications use a high gain, tracking helix array antenna which is erected immediately after landing. The system is designed to give a minimum of 10 hours per day of communications capability. The helix array drive system could be designed to incorporate an equatorial mounting so that once the Lander has stabilized and the antenna locks on to Earth, only planar motion would be required to maintain lock. Backup to the primary antenna system is through the use of an omni directional turnstile located inside the Lander and exposed when the aft cover is opened. Bit rate through this system is four bps. Data storage is accomplished through the use of two tape recorders and a buffer storage unit. For a detailed analysis of the communications subsystem see Section 4.1.

6. Ground Orientation

The mechanisms designed to bring the impacted Lander into surface operating condition, level and secure it with respect to local vertical and otherwise prepare it for experimentation constitute the ground orientation system. The general requirements are as follows:

1. Determine when landing shocks or post-landing tumbling have ceased
2. Open Lander as required to permit deployment of communications antenna and experimental equipment
3. Level Lander if required
4. Secure Lander to prevent shifting due to winds or other surface conditions
5. Prevent inadvertent mis-orientation during above procedure
6. Permit thermal radiation of the RTG unit to space
7. Perform adequately under the maximum number of adverse conditions.

Some of the types of orientation approaches considered during the Voyager and other studies are illustrated in Table 3.2-8. Of these, type III, the clam shell, was selected as the type most suitable for the present study for the following reasons:

1. The relatively flat shape of the Lander configuration is most adaptable to the clam shell.
2. The Lander will inherently come to rest on the one of the two halves of the clam shell.
3. This type is simpler and more reliable than types requiring rockets or mechanisms that could jam on impact.

The sequence of operation of the system is as follows:

1. Accelerometers determine that the Lander has come to rest and which side is up.
2. Pyrotechnic harpoons are fired from the down side to secure that portion against further motion.
3. The base door of vehicle is opened exposing the internal components and experiment equipment. If the base is down, the main portion of the vehicle is swung open leading to the same opened "clam shell."
4. The second half of the clam shell is staked down by harpoons.
5. Legs are extended to level the vehicle if required.
6. The high gain antenna is erected.
7. Experimental equipment is deployed and put into operation.

7. Propulsion

Propulsive devices required on the Entry/Lander include:

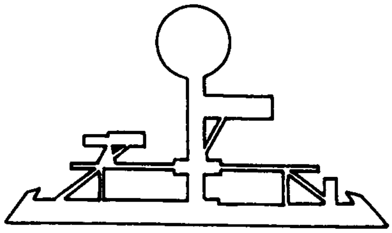
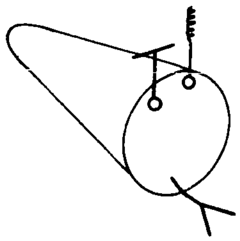
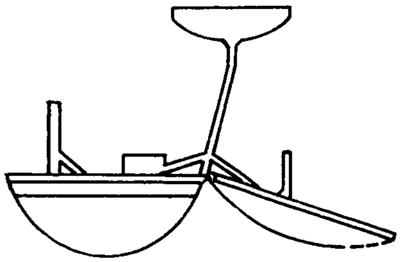
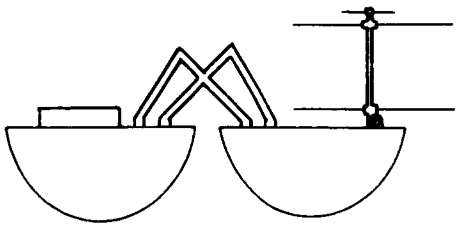
(a) delta velocity rocket
(b) spin and separation subsystem
(c) low altitude retardation retro-rockets

a. <u>Delta Velocity Rocket</u>

The ΔV rocket is used just after separation of the spacebus (or orbiter) and Entry/ Lander to change the flyby trajectory to an impact trajectory. The rocket is sized to give a ΔV of 300 fps based on a specific impulse of 230 seconds.

Initiation of the unit will be by signal from the separation sequence programmer. If it is found necessary, thermal control of the motor can be provided. The ΔV rocket and

TABLE 3.2-8. LANDER ORIENTATION SYSTEMS

Type	Title		Comments
I	Base Down		Requires tip over rocket or bar + shield ejection after impact
II	Rotating Bulkhead		Most suitable for relatively sharp vehicle
III	Clamshell		Most suitable for blunt nosed vehicle. Cover acts as tip over bar
IV	Ball Clamshell		For spherical vehicle of simple construction

support structure is separated from the Entry/Lander after firing by means of three pyrotechnic bolts and given a differential velocity by compression springs.

b. Spin and Separation Subsystem

The spin and separation subsystem provides both spin stabilization of the Entry/Lander and an initial separation rate between the Lander and Bus after detachment. Spin-up of the vehicle is required to negate potential velocity errors caused by angular tip off rates at separation and by the ΔV rocket thrust vector misalignment. Favorable entry attitude can be predicted and maintained with a spin stabilized vehicle. A spin rate of 60 rpm has been established for this application and the nozzles are canted to give a separation rate of 1 fps.

The selected system is a cold gas assembly using nitrogen gas as the propulsive medium. The system is designed with adequate safety margins to allow it to be heat sterilized. Two tanks are provided to increase reliability. The entire assembly including nozzles is attached to the radiator-adapter section and is ejected prior to entry. Total weight of the system excluding attachment hardware is 47.9 pounds.

c. Terminal Retrorocket

Retardation studies have indicated that a minimum weight system requires the use of terminal braking rockets to decrease the impact velocity. The 1971 Entry/Lander utilizes a solid rocket system capable of providing a velocity decrease of 80 fps. The propulsion assembly is fastened in series to the main parachute riser, see Figure 3.2-1. The motor consists of a spherical chamber and two nozzles 90 degrees apart which are angled to prevent impingement on the Lander and contamination of the ground directly below the descending vehicle and to minimize misalighment effects due to parachute oscillation. Initiation of the retrorocket is by means of a radar altimeter. Delta velocity level has been selected to give a theoretical zero velocity impact in a 30 mb atmosphere.

8. Environmental Control

The system presented provides for cooling of the RTG power supply (which constitutes the design load) as well as for the Lander systems and temperature control of the scientific laboratory. The estimated system weight based on a 170 watt RTG is 161 pounds.

a. Description

The system's main feature is a dual-loop coolant system with the high temperature loop providing RTG cooling; Figure 3.2-42. The coolant is Monoisopropylbiphenyl which can operate at temperatures up to 500°F and in a nuclear radiation environment. A waterboiler is provided in the RTG loop to provide cooling during the launch and entry phases of the mission. A radiator provides the sink for the RTG during the Mars transit. Payload temperature control is provided by the second and lower temperature coolant loop which is coupled to the RTG coolant loop by a heat exchanger. Control is provided by a bypass valve across the heat exchanger.

RTG cooling on the Mars surface is passive. The transit radiator is not required and can therefore be ejected before entry. Cooling fins on the RTG surface are required. The coolant loop continues to receive heat from the RTG to provide payload temperature control for the potentially low surface temperatures (i.e., - 184°F). At the high temperature condition of 117°F maximum, the controlling thermal resistance of the crush-up material has been minimized to reduce internal temperatures due to internal heat dissipation. Telecommunication equipment temperatures do not exceed the 100°F

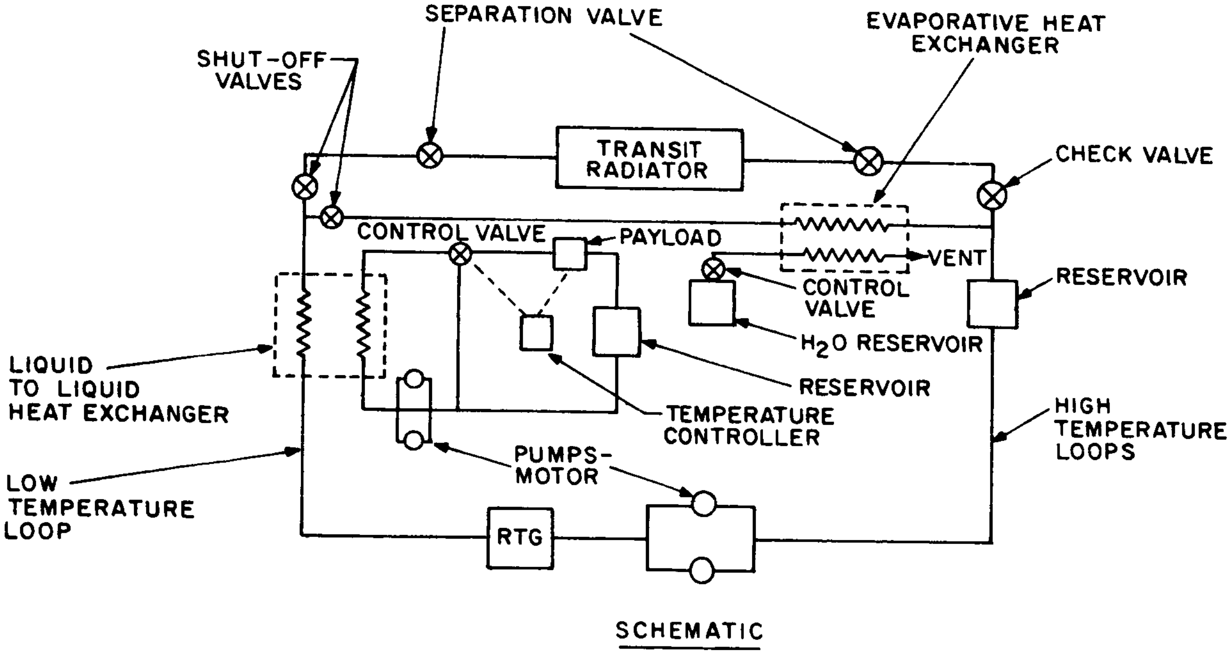


Figure 3.2-42. Titan IIIC Voyager Temperature Control

operating temperature during maximum temperature environment conditions. An octadecane wax is provided in the battery and biological experiment area to maintain temperatures below 100°F.

b. Operation

1. Pre-Launch

   During the checkout phase after installation of the power source, RTG cooling is provided by the water boiler through the high temperature coolant loop. Water is provided from a ground source. The low temperature loop is operated independently from a ground coolant source.

2. Launch

   RTG cooling is provided by the water boiler utilizing an on-board water supply. Sufficient thermal mass exists in the payload to limit the temperature rise until the launch shroud is ejected.

3. Transit

   During transit, RTG heat rejection is through the transit radiator. With the Entry/Lander in the shade, heating is always required by the payload during transit. The high temperature loop serves as the heat source for this requirement.

4. Mars Surface Operation

   The RTG is passively cooled on the Mars surface but the coolant loop still serves as a heat source for the payload during night operation. The inter loop heat exchanger provides for heating of the payload coolant loop. A bypass valve controls temperature at the payload. During high temperature ambient operation, the bypass restricts flow to the payload. Telecommunication system heat rejection would be passive through the aft cover. Octadecane wax will prevent overheating of the battery and laboratory through a phase change at about 85°F.

c. Requirements

Requirements and system characteristics are given in the following table:

1. Temperature	
RTG (170 watts)	450 - 600°F
Electronic Equipment	50 - 150°F
Batteries and Laboratory	75 - 90°F
2. Coolant	Monoisopropylbiphenyl
3. Coolant Flow Rate	250 lb/hour
4. System Pressure Drop	10 - 15 psi

5. System Power Consumption — 35 watts

6. Transit Radiation Surface Area — 50 ft^2

7. Mean Radiator Temperature — 400^oF

8. System Weight

2	Modulation Valves	6
1	Reservoir	3
1	Elect. Temp. Controller	2
2	Motor - Pumps	6
2	Separation Valves	6
4	Shut-off and Relief Valves	6
1	Liquid-Liquid Heat Exchanger	8
1	Evaporative Heat Exchanger (Water Boiler)	8
1	Transit Space Radiator	36
1	Water Storage Vessel + H_2O	26
	Insulation, tubing and coolant	24
	Octadecane Wax and Enclosure	30
		161 pounds

9. Vehicle Description

Three preliminary designs have been generated for the Lander/Bus concept during the Voyager - Titan IIIC study. The first design is that of the Solid Flare Vehicle. A detailed layout of the Mars 1971 solid flare Lander is shown on Figure 3.2-43. To reiterate, the Lander has a half cone angle of 51.5 degrees, a nose radius of 31.5 inches and a base diameter of 134 inches. To the aluminum honeycomb shell structure is bonded the fiberglass honeycomb shock attenuation material. The heat shield is the GE developed ESM ablative material ranging in thickness from 0.417 inches at the stagnation point to 0.318 inches on the aft end of the cone. The radar dish is located in the crush-up material at the nose and views through the fiberglass honeycomb crush-up, nose cap and ESM all of which are transparent to RF signals. The one dish serves both the radar altimeter subsystem and the retrorocket initiation sensor.

An aluminum honeycomb cruciform structure provides support for the thermal control and telecommunications components and selected scientific payload. In addition,

KEY FOR FIGURES 3.2-43, -44, AND -45

1. Heat Shield
2. Radar Altimeter Antenna
3. Crush Up (Fiberglass Honeycomb)
4. Structural Shell
5. Delta "V" Rocket
6. Adapter Section (Transient Radiator)
7. Spin & Separation System
8. Omni Antenna
9. Harpoon
10. Crush Up
11. Parachute Package (Decelerator)
12. Parachute Package (Main)
13. Helix Array Antenna
14. Shelf
15. RTG Unit
16. Gas Reservoir
17. Aft Cover
18. Photoautotroph
19. Turbidity
20. TV Camera (Panorama)
21. Light Intensity (Sun Sensor)
22. Transponder
23. Motor & Pumps
24. Evaporative Heat Exchanger
25. Isolator & Load
26. Battery
27. Liquid Heat Exchanger
28. Modulation Valve
29. Coolant Reservoir
30. Power Control & Conversion Unit
31. Modulation Valve
32. Temperature Control
33. Buffer Storage
34. Shelf
35. Tape Recorder
36. Shelf
37. Surface Gravity
38. Power AMP (24W)
39. Diplexer
40. H. V. PWR Supply (24W)
41. H. V. PWR Supply (150W)
42. H. V. PWR Supply (15W)
43. Transmitter (200 MW)
44. Command Decoder
45. Surface Roughness Altimeter Electronics
46. Power AMP (15W)
47. Power AMP (100W)
48. R. F. Switch
49. Water
50. Command & Computer Equipment
51. Data Processing Unit
52. Precipitation
53. Wind Speed & Direction
54. Surface Penetration Hardness
55. Electron Density (Langmuir Probe)

55A. Electronics for Electron Density

56. Seismic Activity
57. Microscope (Including TV Camera, Drill Handling Pulverizer, Sampler)
58. Multiple Chamber
59. Sounds

59A. Electronics for Sounds

60. Sample Gatherer
61. Soil Moisture
62. Radioisotope

62A. Electronics for Radioisotope

63. Composition O_3
64. Pressure
65. Temperature
66. Composition A
67. Composition H_2O
68. Composition CO_2
69. Density
70. Composition N_2
71. Composition O_2
72. Gas Chromatograph

thermal control coolant lines route through the structure and allow selective temperature control to be maintained. Other payload experiments which must view the atmosphere during the entry and descent phases are located in a cluster on the aft cover (17) under an ejectable protective cover; see Figure 3.2-44, section F-F. Instruments which must be exposed to the Martian soil, such as the soil analyzer and biological life detector are also located on the aft cover.

Because of degrading effects which integrated nuclear radiation causes on certain solid state electronic components, a minimum distance of 28.6 inches has been maintained from the center of the RTG source (15) to these components during the transit phase. This distance is sufficient to keep the integrated dosage at or below the 10^{12} neutrons/cm^2 level for the length of the entire mission from prelaunch phase to the end of the six month Martian surface operation. The 170-watt RTG is cooled by passive thermal radiation to the atmosphere during the surface phase and is located on the aft cover so that adequate cooling is assured.

Communications with Earth during planetary approach and entry is maintained through the use of an encapsulated turnstile antenna (8). The antenna is located on the centerline of the Lander on the aft end so that a symmetrical pattern is provided. Around the antenna, the main parachute (12) and the terminal retrorockets are packed in an aluminum cannister. The decelerator parachute and mortar assembly (11) are located in a well next to the main chute support structure. An ejectable cover protects the retardation system from entry heat.

The delta velocity solid rocket (5) is supported by an aluminum monocoque structure which is fastened to the aft cover with three explosive bolts. Electrical quick disconnects are provided for heater power and squib lines. Redundancy in ejecting the ΔV rocket assembly is provided by securing the supporting structure to the parachute housing cover which is ejected just prior to the parachute sequencing. Entering with the ΔV rocket attached increases the W/C_DA to approximately 15.9 psf and decreases the altitude at which Mach 2.5 is reached to 18,000 feet. Analysis indicates that this is still adequate for full parachute sequencing and, a successful landing would be accomplished.

The space radiator and adapter assembly (6) is jettisoned just prior to entry. The spin and separation system including the redundant cold gas reservoirs (16), tubing and nozzles (7) are attached to the adapter structure and are jettisoned at the same time. The adapter serves to attach the Lander system to the spacebus.

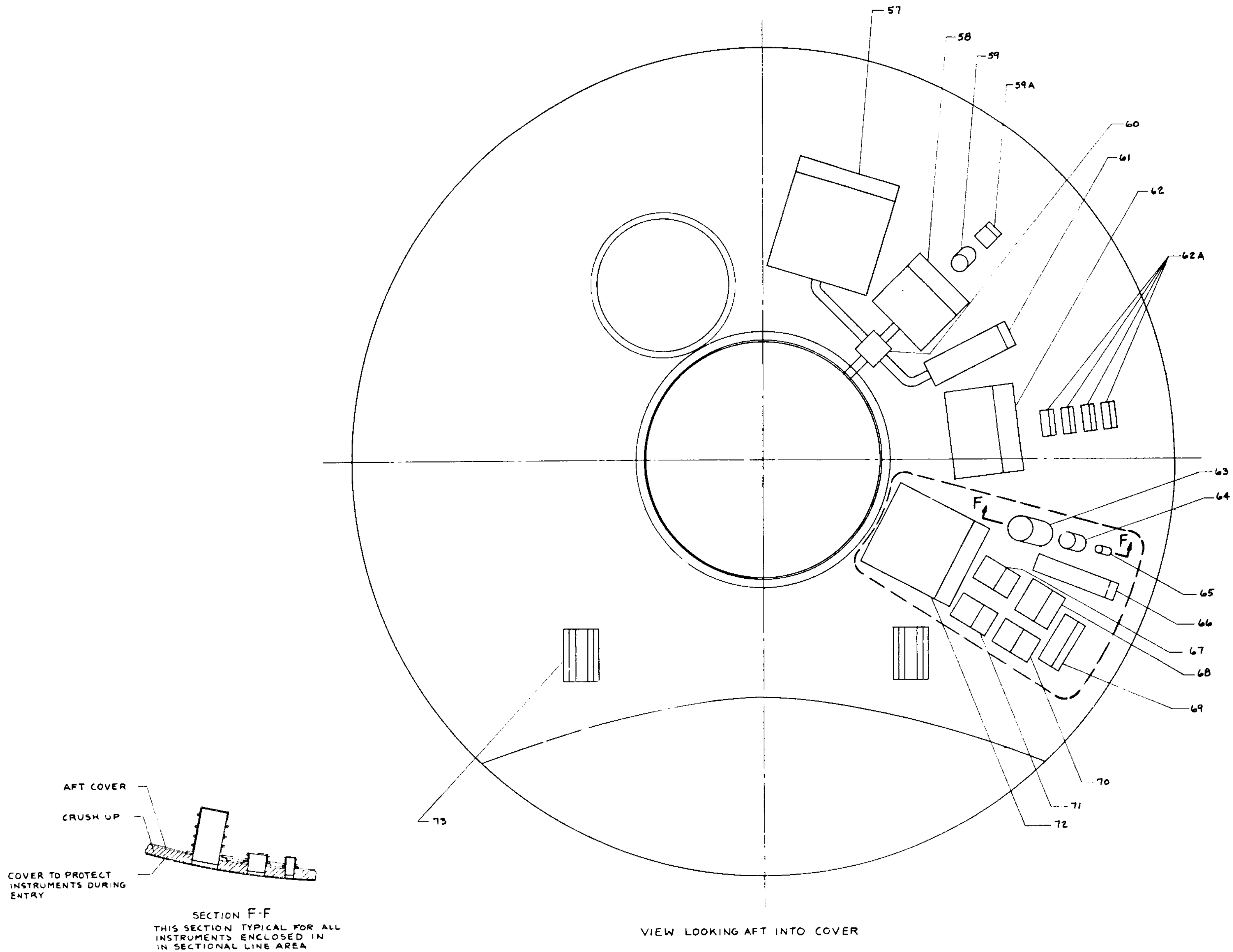


Figure 3.2-44. 1971 Entry/Lander - Solid Flare

The direct link helix array antenna is mounted in such a fashion that it will erect after the aft cover is opened and secured. The antenna is capable of tracking Earth in a hemispherical pattern. The panorama television camera is mounted on a boom which is erected after opening of the cover. Several experiments are located on a deployable beam as shown in Figure 3.2-45 section E-E. These experiments which must be dropped to the surface or be completely exposed to the atmosphere include the soil hardness tester (54), seismic activity experiment (56), anemometer (53), and precipitation gauge (52). Figure 3.2-46 shows the Lander in a deployed configuration. A notch is provided in the aft cover to allow it to fully open during surface orientation. The notch is covered with a frangible section which protects the vehicle interior from entry heat. A system block diagram for the Solid Flare Vehicle is presented in Figure 3.2-47.

A second configuration which has been studied, consists of a sphere-cone with a 110-inch base diameter to which has been added a movable flare section to attain the necessary drag area to reach Mach 2.5 at a minimum of 20,000 feet in the 11 mb-A model atmosphere. The folding flap concept, mentioned in Section 3.2.2-b, provides a high payload capacity Entry/Lander which is compatible with the Titan IIIC interface diameter. Flaps have been sized to provide the correct amount of drag area necessary to produce a ballistic coefficient of 15 psf. Figure 3.2-48 shows the relationship between the geometric and equivalent base diameters and the slant length of the flap. The equivalent diameter refers to the equivalent aerodynamic reference area which is to be matched.

In operation, the flaps are extended and locked just after the launch vehicle shroud has been ejected. Deployment could be achieved by explosive charge, one-way actuators. The flare section remains attached throughout the mission until just prior to parachute deployment. At that time, explosive bolts will sever the attachment and the extensible flare section will be separated from the basic vehicle.

Structural analyses have been performed on the extensible flare for several sized vehicles and are reported in Section 3.2.2-c. A preliminary design for an equivalent base diameter of 138 inches was conducted to show:

1. design feasibility
2. payload packaging

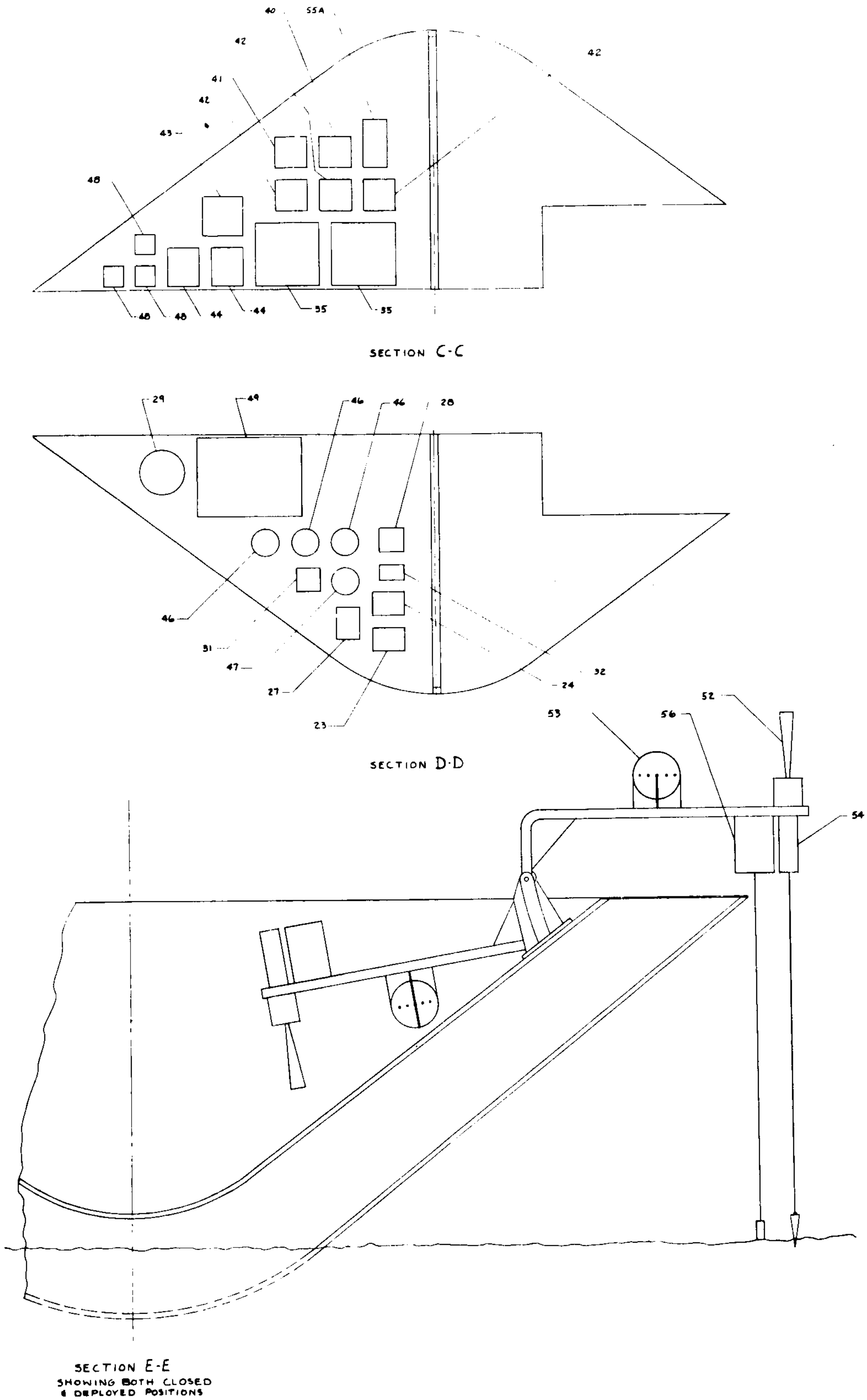


Figure 3.2-45 1971 Entry/Lander - Solid Flare

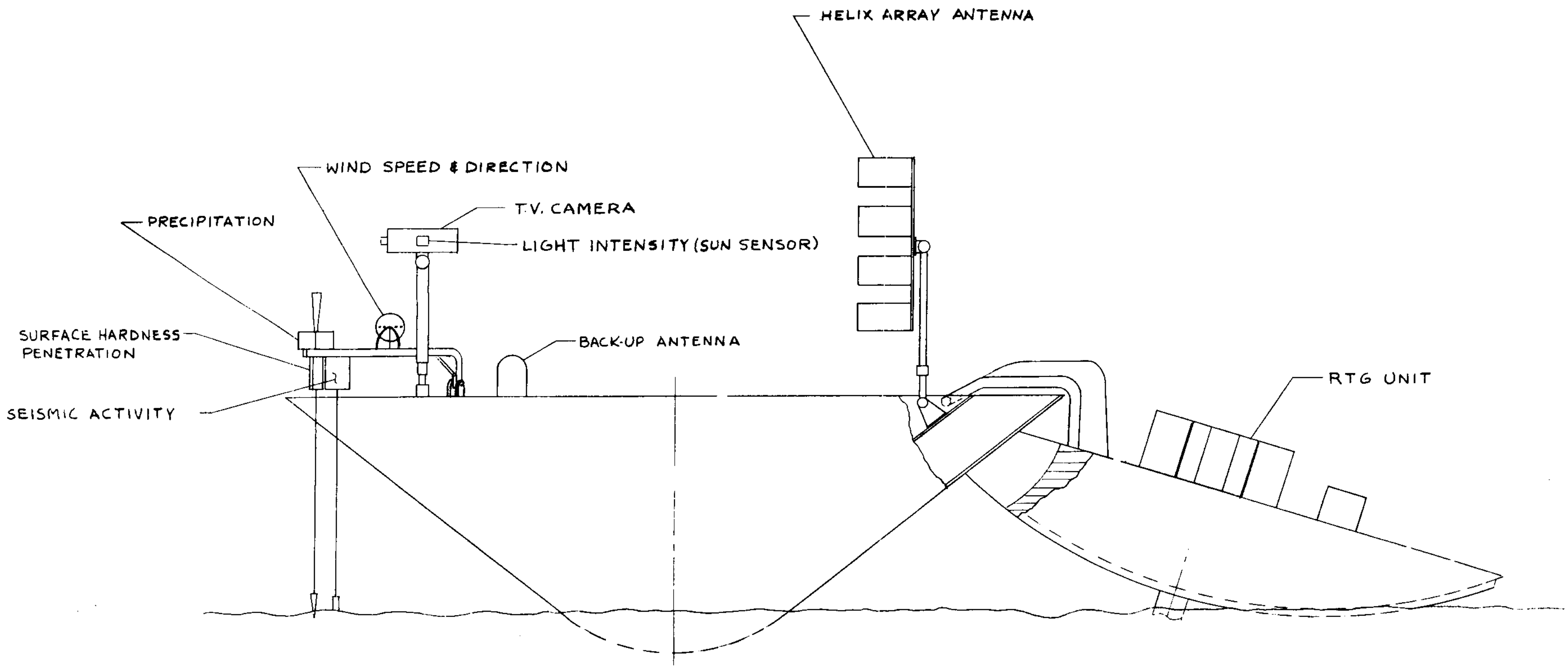


Figure 3.2-46. 1971 Entry/Lander - Solid Flare, Surface Deployed Configuration

3. basis for weight estimation
4. comparison with the Solid Flare Vehicle

The design layout is shown in Figure 3.2-49 for a vehicle based on an entry weight of 1942 pounds and a ballistic parameter of 15 psf. This entry weight results in a total Lander weight (at bus separation) the same as for the 1830 pound Solid Flare Vehicle. The basic vehicle, that is, the portion of the Entry/Lander which is landed on the surface, has a base diameter of 110 inches. The subsystems contained in this vehicle are identical to those carried on the Solid Flare configuration and discussions concerning those items are also applicable here.

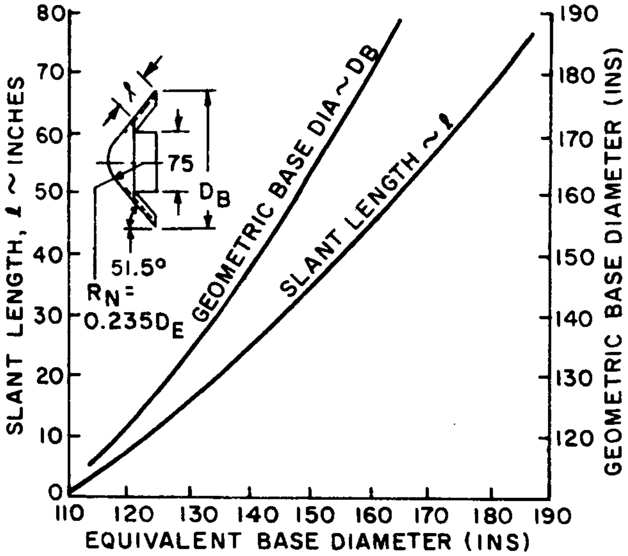


Figure 3.2-48. Extensible Flare Lander - Equivalent Base Diameter Relationship

Since the extensible flare vehicle has been designed as an alternate concept to be used with a spacebus, a direct link antenna used during entry and descent is furnished and is located on the rear of the aft cover. Component packaging in the extensible flare vehicle is basically the same as in the solid flare concept except that the parachute housing has been located outside the aft cover to facilitate packaging of the helix array antenna.

The extensible flare section consists of four movable flaps which are hinged to a center supporting, cylindrical structure. Actuation is by explosive charge mechanisms, two per flap to increase reliability. The cylindrical supporting section serves to provide hinge and actuating surfaces and also provides the Bus/Lander interface adapter function. The thermal control space radiator and spin and separation system are attached directly to this structure and, unlike on the solid flare vehicle, do not separate prior to entry. This different separation sequence causes the difference in entry weight noted earlier. The extensible flare section including the space radiator and spin subsystem are released at the time decelerator parachute deployment. This removes approximately 460 pounds from the vehicle retarded by the parachutes and

KEY FOR EXTENSIBLE FLARE LANDER, FIGURE 3.2-49

1. Heat Shield
2. Radar Altimeter Antenna
3. Crush Up (Fiberglass Honeycomb)
4. Structural Shell
5. Delta "V" Rocket
6. Adapter Section (Transient Radiator)
7. Spin & Separation System
8. Crush Up (Fiberglass Honeycomb)
9. Aft Cover
10. Parachute Package (Decelerator)
11. Parachute Package (Main)
12. Omni Antenna
13. Diplexer
14. Power Supply
15. Helix Array Antenna
16. Driver Amplifier & Power Supply
17. Flap Actuator
18. Electron Density (Langmuir Probe)
19. Battery
20. R. F. Switch
21. Command Detector
22. Power Amp.
23. Isolator & Load
24. Power Amp.
25. Temperature Control
26. Isolator & Load
27. Command & Computer Equip.
28. Power Amp.
29. Power Supply
30. Data Processing
31. Power Conversion & Control
32. Tape Recorder
33. Power Supply
34. Power Amp.
35. Tape Recorder
36. Shelf
37. Motor & Pumps
38. Evaporative Heat Exchanger
39. Modulation Valve
40. Modulation Valve
41. Liquid Heat Exchanger
42. Coolant Reservoir
43. Water
44. Buffer Unit
45. Transponder
46. Electronics for Electron Density
47. Photoautotroph
48. Altimeter Electronics
49. Radar Altimeter Electronics
50. Turbidity
51. Surface Gravity
52. Gas Chromatograph
53. Pressure
54. Density
55. Temperature
56. Composition A
57. Composition O_3
58. Composition N_2
59. Composition CO_2
60. Composition H_2O
61. Composition O_2
62. Sounds
63. Electronics for Sounds
64. Multiple Chamber
65. Soil Moisture
66. Radioisotope
67. TV Microscope & Sub-Surface Group
68. Electronics for Radioisotope
69. Wind Speed & Direction
70. Seismic Activity
71. Precipitation
72. Surface Penetration Hardness
73. Shelf
74. RTG Unit
75. TV Camera (Panorama)
76. Light Intensity (Sun Sensor)
77. Gas Reservoir
78. Extensible Flap

impact subsystem. The weight saving thus effected partially offsets the weight added due to the inherently inefficient structure of the movable flaps. The resultant gross payload of the Extensible Flare Vehicle is only 75 pounds less than that of the Solid Flare Vehicle.

Advantages attributable to the Extensible Flare configuration:

1. Permit use of the 120-inch diameter shroud
2. Permit flexible program for change from year to year or due to changes required because of better definition of atmosphere

Surface orientation of this vehicle is identical to that of the solid flare vehicle. The aft cover is used to stabilize the Lander and harpoons are provided to anchor the vehicle to prevent vehicle motion due to surface winds. As before, the helix array antenna, TV camera, and surface experiment beam will deploy after the aft cover has opened and all vehicle motion has stopped. A preliminary weight statement is shown in the following section.

The third Lander design that is presented takes advantage of the additional payload capability which is available in the Lander of the Bus/Lander spacecraft. Analysis has shown a significant increase in attainable mission value (see Section 2.6) if a Rover vehicle were provided to transport selected scientific instruments to several exploration sites. The unspecified payload weight is estimated at approximately 156 pounds which was deemed sufficient to include a roving capability. A preliminary design was made to determine the feasibility of this concept.

Figure 3.2-50 shows the basic solid flare, 134 inch Entry/Lander with the Rover (9) located in the aft cover. The gross vehicle weight is the same; 2042 pounds.

The Rover was selected to be a limited range vehicle which does not carry its own communications or power supply. These functions are provided by the parent Lander through a cable attached to the Rover. The Rover is a three-wheeled vehicle which is powered by an electric motor. It serves as a transport platform for the soil analysis group including the analysis TV, microscope, and sample handler. The multiple chamber biological experiment would also be included on the Rover.

Upon landing, one set of data would be taken before the Rover is released to insure that information could be obtained at the landing site without involving the probable success estimates associated with the Rover and terrain. On command, the exit ramp would be deployed and the Rover would be released, reeling out its umbilical as it proceeds, see Figure 3.2-51. Aiming of the vehicle could be accomplished by using the panorama TV camera and commands from Earth or possibly through a self-contained terrain sensing system. The use of a Rover appears to be an attractive capability which could be provided on the 1971 Bus/Lander configuration. A preliminary weight statement appears in the following Section.

10. Weight Statements

Weight and balance properties have been estimated in detail for each of three Lander configurations and are presented in the following tables.

a. Table 3.2-9. Voyager Bus/Lander.

This is the prime configuration presented and consists of a solid flare with D_B = 134 inches, W/C_DA = 15 psf, and W_{entry} = 1830 pounds. Gross weight including adapter, radiator, spin and separation and ΔV rocket is 2,042 pounds.

The gross payload capability of this configuration is 857.8 pounds and includes 156.3 pounds of capability which is unassigned. The gross payload includes the scientific experiments, thermal control, electrical power supply and communication equipment.

Moments and products of inertia have been calculated for this configuration in addition to the detail weights.

b. Table 3.2-10. Voyager Bus/Extensible Flare Lander.

This vehicle has a basic solid flare main body of 110" dia. and an extensible flare to achieve a $D_B$effective = 138 inches, with W/C_DA = 15 psf and W_{entry} = 1942 pounds. Entry weight includes the radiator and spin and separation components. These are attached the adapter section which provides structural support for the extensible flaps and actuators.

The extensible flaps thus form a unit, with adapter, radiator and spin and separation, which is ejected after entry and prior to chute deployment.

Gross payload capability of this vehicle is 782 pounds.

The gross vehicle weight including ΔV rocket is 2042 pounds.

TABLE 3.2-9. VOYAGER BUS/LANDER

LANDER PRELIMINARY C G AND MOMENTS OF INERTIA

Condition	Weight Lb.	C G Station (Inches)			Products of Inertia ($Slug-Ft^2$)			Moments of Inertia ($Slugs-Ft^2$)		
		Axial (X)*	Vert. (Z)	Lat. (Y)	I_{XZ}	I_{XY}	I_{ZY}	I_X (Roll)	I_Z (Yaw)	I_Y (Pitch)
Lander (Entry)	1830.0	38.6	2.0	0.6	18.35	19.45	0.66	322	201	203
Lander (Gross)	2042.0	41.4	1.8	1.5	16.20	-14.75	13.60	367	270	253

*Inches From Stag Point

W/C_DA = 15 PSF

D_B = 134 Inches

θ (1/2) = 51.5^o

C_D = 1.25 (Hypersonic)

TABLE 3.2-9. VOYAGER BUS/LANDER

LANDER WEIGHT ESTIMATE

Component	Weight (lb)	C G Sta. (in) *
Shield	(165.5)	26.9
Structure	(227.2)	35.1
Honeycomb sandwich	143.8	33.9
Ring-fwd.	1.7	14.2
Ring-mid	9.7	29.5
Ring-aft	16.0	44.5
Cruciform	34.0	33.5
Fittings-chute (4)	10.0	45.0
Brackets & Fasteners	12.0	35.1
Aft Cover	(139.0)	52.0
Shield	30.8	52.0
Crush Up	10.2	52.7
Skin (incl. doublers)	75.0	51.7
Attaching angles	7.6	53.0
Hinges & Fittings	12.0	52.0
Fasteners	3.4	52.0
Retardation	(414.5)	34.0
Crush Up	244.0	24.0
Decel. Chute	27.0	48.8
Main Chute	66.0	49.6
Bags, risers, etc.	10.0	49.0
Chute Housing	9.5	49.6
Retrorockets	41.0	49.6
Prog., Batt., Switch, mortar	9.0	40.0
Harnesses	4.5	43.0
Fasteners	3.5	33.5

*From Stag Point

TABLE 3.2-9. VOYAGER BUS/LANDER (Cont'd)

LANDER WEIGHT ESTIMATE

Component	Weight (lb)	C G Sta. (in) *
Scientific Payload	(387.9)	48.4
Instrumentation		
Temperature	0.3	56.0
Sounds	0.5	58.0
Pressure	0.3	57.0
Density	1.5	57.0
Multiple Chamber	4.0	57.0
Surface Penetration Hardness	4.5	38.2
Photoautotroph	3.0	38.5
Light Intensity (sun sensor)	0.5	49.0
Composition - H_2O	1.5	57.0
Composition - O_2	1.5	57.0
Turbidity & P_H	4.0	22.7
Wind Speed & Direction	2.0	33.3
Gas Chromatograph	7.0	58.0
Composition - N_2	1.0	57.0
Composition - CO_2	1.0	57.0
Soil Moisture	2.0	57.0
TV Camera, Panoramic	10.0	48.0
Radioisotope	6.0	57.0
Composition - O_3	1.5	57.0
Composition - A	1.5	57.0
Precipitation	1.0	32.0
Electron Density	3.0	45.0
Surface Gravity	3.0	30.0
Surface Roughness Altimeter	15.0	38.5
Microscope (Inc. TV Camera, etc.)	75.0	57.0
Seismic Activity	8.0	38.9

*From Stag Point

TABLE 3.2-9. VOYAGER BUS/LANDER (Cont'd)

LANDER WEIGHT ESTIMATE

Component	Weight (lb)	C G Sta. (in) *
Scientific Payload (Cont'd)		
Instrumentation (Cont'd)		
Ablation Sensors (Diagnostic)	1.0	24.0
Temp. Sensors (Diagnostic)	0.5	24.0
Accelerometers (Diagnostic)	2.0	27.0
Ablation Converter (Diagnostic)	1.5	27.0
Additional Payload (Capability)	156.3	51.0
Deployment & Installation		
TV	10.0	48.0
Surface Hardness	3.0	38.2
Brackets & Fasteners	15.0	36.0
Harnesses	40.0	36.0
Thermal Control	(129.0)	31.2
Modulation Valves	6.0	25.6
Reservoir	3.0	39.2
Temperature Controller	2.0	26.4
Motor Pumps (2)	6.0	18.0
Separation Valves	6.0	21.0
Shutoff & Relief Valves (4)	6.0	21.0
Heat Exchanger (Liq. - Liq.)	8.0	20.1
Heat Exchanger (Evaporative)	8.0	22.5
H_2O & Tank	26.0	38.6
Insulation	12.0	44.0
Tubing	4.0	44.0
Coolant	6.0	44.0
Octadecane Wax & Enclosure	30.0	27.9
Brackets, Fittings & Fasteners	2.5	31.8
Harnesses	3.5	31.8

*From Stag Point

TABLE 3.2-9. VOYAGER BUS/LANDER (Cont'd)

LANDER WEIGHT ESTIMATE

Component	Weight (lb)	C G Sta. (in) *
Electrical	(143.0)	39.2
RTG	89.2	42.6
Battery	24.6	27.9
Battery Regulators	3.0	25.6
Power Controller	9.0	38.0
IFD	1.5	45.0
Brackets & Fasteners	7.2	40.0
Harnesses	8.5	40.0
Ground Orientation	(26.0)	46.8
Clamshell Actuating Mech.	13.5	52.0
Accelerometer & Controls	2.0	41.0
Harpoons	3.0	42.0
Fasteners	4.0	41.0
Harness	3.5	41.0
Communications	(197.9)	37.0
Transmission		
Diplexers (2)	2.0	29.3
Helix Array Antenna	10.0	39.0
Omni Antennas (2)	10.0	59.3
Transponders (2)	10.8	38.5
Power Amp. - 24W (2)	6.0	26.4
Power Supply (2)	12.0	26.4
Power Amp. - 15W	3.0	30.3
Power Supply	6.0	32.0
Power Amp. - 150W (2)	8.0	27.3
Power Supply (2)	16.0	26.4
Driver Amp. & Power Supply	4.0	34.6

*From Stag Point

TABLE 3. 2-9. VOYAGER BUS/LANDER (Cont'd)

LANDER WEIGHT ESTIMATE

Component	Weight (lb)	C G Sta. (in) *
Communications (Cont'd)		
Transmission (Cont'd)		
Command Detector (2)	6.0	41.0
RF Switch (3)	3.0	40.8
Isolator & Load	3.0	31.1
Data Handling		
Data Processing Unit	16.0	29.7
Buffer Unit	4.0	41.0
Tape Recorders (2)	16.0	39.4
Command		
Command & Computer Equipment	14.0	41.3
Power Converter & Controller	7.0	38.0
Antenna Controls		
Omni Switch	2.0	42.0
Sun Sensor	2.3	42.0
Electronic Gimbal Control	1.4	42.0
Amplifier	1.4	40.0
Drive Motors	4.0	42.0
Mode Control Electronics	0.5	40.0
Vertical Switch	2.0	40.0
Antenna Deployment Mech.	6.0	48.0
Brackets & Fasteners	9.5	42.0
Harnesses	12.0	42.0
Total Lander (Entry Condition)	(1,830.0)	38.6

*From Stag Point

TABLE 3. 2-9. VOYAGER BUS/LANDER (Cont'd)

LANDER WEIGHT ESTIMATE

Component	Weight (lb)	C G Sta. (in) *
Adapter		
Structure	(30. 0)	54. 1
Skin	7. 5	54. 3
Longerons	1. 2	54. 3
Stiffeners	1. 4	54. 3
Ring - Fwd.	8. 3	44. 7
Ring - Aft	8. 3	63. 7
Explosive Bolts (4)	0. 4	44. 0
Fasteners	2. 9	54. 1
Thermal Control	(31. 0)	54. 3
Skin	6. 4	54. 3
Insulation	3. 9	54. 3
Spacers	4. 1	54. 3
Tubing	4. 7	54. 3
Fittings & Connectors	3. 1	54. 3
Coolant	5. 1	54. 3
Fasteners	3. 7	54. 3
Spin & Separation	(53. 0)	53. 0
Tanks (2)	30. 2	54. 4
N_2	13. 4	54. 4
Squib Valves	1. 5	45. 0
Tubing	2. 0	48. 0
Nozzles & Fittings	0. 8	54. 3
Support Struct. & Fasteners	3. 0	45. 0
Harness	2. 1	45. 0

*From Stag Point

TABLE 3.2-9. VOYAGER BUS/LANDER (Cont'd)

LANDER WEIGHT ESTIMATE

Component	Weight (lb)	C G Sta. (in) *
Adapter (Cont'd)		
ΔV Rocket Installation	(98.0)	78.9
Rockets	92.0	79.8
Support Struct. & Fasteners	6.0	65.0
Total Adapter	(212.0)	65.3
Total Lander (Gross)	(2,042.0)	41.4

*From Stag Point

TABLE 3.2-10. VOYAGER BUS/EXTENSIBLE FLARE LANDER

LANDER WEIGHT ESTIMATE

Component	Weight (lb)	C G Sta. (in.)
Shield	(111.0)	24.5
Structure	(126.7)	25.2
Honeycomb Sandwich	75.0	26.4
Ring - Fwd.	2.1	12.0
Ring - Mid	5.2	23.6
Ring - Aft	10.4	35.4
Cruciform	26.0	23.5
Fasteners	8.0	24.0
Aft Cover (Same As Orbiter/Lander)	(83.1)	44.9

TABLE 3.2-10. VOYAGER BUS/EXTENSIBLE FLARE LANDER (Cont'd)

LANDER WEIGHT ESTIMATE

Component	Weight (lb)	C G Sta. (in.)
Retardation	(360.0)	34.3
Impact Crush Up	212.0	19.6
Decel. Chute	21.0	54.5
Main Chute	56.0	54.5
Bags, Risers, Etc.	10.0	54.5
Chute Housing	9.0	54.5
Prog., Batt., Switch, Mortar	9.0	28.0
Retrorockets	35.0	67.5
Fasteners	3.5	48.0
Harnesses	4.5	36.0
Flare	(375.0)	42.2
Flap Thermal Protection	58.5	42.0
Flap Struct.	202.5	41.0
Actuators	58.0	46.0
Mounting & Support Struct.	56.0	42.5
Radiator	(31.0)	42.0
Skin	6.4	42.0
Insulation	3.9	42.0
Spacers	4.1	42.0
Tubing	4.7	42.0
Fittings & Connectors	3.1	42.0
Coolant	5.1	42.0
Fasteners	3.7	42.0
Spin & Separation	(53.0)	45.1
Tanks	30.2	45.5
N_2	13.4	45.5

TABLE 3.2-10. VOYAGER BUS/EXTENSIBLE FLARE LANDER

LANDER WEIGHT ESTIMATE

Component	Weight (lb)	C G Sta. (in.)
Spin and Separation (Cont'd)		
Squib Valves	1.5	45.5
Tubing	2.0	38.0
Nozzles & Fittings	0.8	48.0
Support Struct. & Fasteners	3.0	45.5
Harness	2.1	42.0
Ground Orientation (Same As Orbiter/Lander)	(20.0	38.0
Gross Payload	(782.0)	32.4
Scientific P/L	312.1	37.6
Thermal Control	129.0	27.1
Electrical	143.0	32.6
Communications	197.9	28.5
Total Lander (Entry Condition)	(1,942.0)	34.6
Δ V Rocket Installation	(100.0)	66.5
Rocket	92.0	66.8
Support Struct. & Fasteners	8.0	62.0
Total Lander (Gross)	(2,042.0)	36.2

c. Table 3.2-11. Voyager Bus/Lander (with Rover)

This configuration is basically the Bus/Lander (Table 3.2-9) in which part of the unassigned payload capability is used to provide a roving vehicle. On this rover have been placed the microscope (including TV camera and other ancillary items), surface penetration hardness, multiple chamber and soil moisture experiments.

TABLE 3.2-11. VOYAGER BUS/LANDER (WITH ROVER)

LANDER (WITH ROVER) WEIGHT ESTIMATE

Component	Weight (lb)	C G Sta. (in.)
Shield (Same as Bus/Lander)	(165.5)	26.9
Structure (Same as Bus/Lander)	(227.2)	34.8
Aft Cover	(154.0)	55.1
Shield	34.8	55.2
Crush Up	11.2	55.3
Skin (Incl. Doublers)	85.0	55.1
Attaching Angles	7.6	55.1
Hinges & Fittings	12.0	54.5
Fasteners	3.4	54.5
Retardation (Components Same As Bus/Lander)	(414.5)	39.4
Scientific Payload (Excluding Rover)	(180.9)	43.4
Instrumentation		
Temperature	0.3	56.0
Sounds	0.5	58.0
Pressure	0.3	57.0
Density	1.5	57.0
Photoautotroph	3.0	38.5
Light Intensity (Sun Sensor)	0.5	49.0
Composition - H_2O	1.5	57.0
Composition O_2	1.5	57.0
Turbidity & pH	4.0	22.7
Wind Speed & Direction	2.0	33.3
Gas Chromatograph	7.0	58.0
Composition - N_2	1.0	57.0
Composition - CO_2	1.0	57.0
TV Camera, Panoramic	10.0	48.0

TABLE 3.2-11. VOYAGER BUS/LANDER (WITH ROVER) (Cont'd)

LANDER (WITH ROVER) WEIGHT ESTIMATE

Component	Weight (lb)	C G Sta. (in.)
Scientific Payload (Excluding Rover) (Cont'd)		
Instrumentation (Cont'd)		
Radioisotope	6.0	57.0
Composition - O_3	1.5	57.0
Composition - A	1.5	57.0
Precipitation	1.0	32.0
Electronic Density	3.0	45.0
Surface Gravity	3.0	30.0
Surface Roughness Altimeter	15.0	38.5
Seismic Activity	8.0	38.9
Ablation Sensors (Diagnostic)	1.0	24.0
Temp. Sensors (Diagnostic)	0.5	24.0
Accelerometers (Diagnostic)	2.0	27.0
Ablation Converter (Diagnostic)	1.5	27.0
Add'l. Payload (Capability)	41.8	51.0
Deployment & Installation		
TV	10.0	48.0
Brackets & Fasteners	13.0	36.0
Harnesses	38.0	36.0
Rover Experiments	(192.0)	53.0
Vehicle	(145.0)	52.5
Structure	16.0	54.6
Wheels	19.0	51.9
Motors & Controls	5.5	51.9
Soil Moisture	2.0	52.8
Microscope (Incl. TV Camera, Etc.)	75.0	52.2

TABLE 3.2-11. VOYAGER BUS/LANDER (WITH ROVER) (Cont'd)

LANDER (WITH ROVER) WEIGHT ESTIMATE

Component	Weight (lb)	CG Sta. (in.)
Rover Experiments (Cont'd)		
Vehicle (Cont'd)		
Surface Pen. Hardness (Incl. Instl.)	7.5	52.8
Multiple Chamber	4.0	52.8
Harnesses	2.0	52.8
Battery	12.0	52.4
Brackets & Fasteners	2.0	52.5
Ramp	(12.0)	51.0
Reel	(10.0)	55.8
Cable	(25.0)	55.8
Thermal Control (Same As Bus/Lander)	(129.0)	31.2
Electrical (Same As Bus/Lander)	(143.0)	39.2
Ground Orientation (Same As Bus/Lander)	(26.0)	46.8
Communications (Same As Bus/Lander)	(197.9)	37.0
Total Lander (Entry)	(1,830.0)	40.1
Adapter (Components Same As Bus/Lander)	(212.0)	65.9
Total Lander (Gross)	(2,042.0)	42.8

E. PARAMETRIC RESULTS AND SUPPLEMENTAL STUDIES

1. Parametric Results

Presented in this section are parametric data which will enable the reader to determine the payload capability for a range of Entry/Lander weights. Curves are shown for both the solid and extensible flare vehicle which have been based on a ballistic parameter of 15 psf, a bluntness radio (R_N/R_B) of 0.47 and a half cone angle (θ_c) of

51.5 degrees. The parameter "gross weight," which is presented herein includes the following:

Scientific instruments

Deployment mechanisms, hardware, cabling

Communications

Power supply

Thermal control

Since these items are directly related to one another, it was felt that they should be treated as a single quantity which may be separately interchanged as required.

Figure 3.2-52 presents both gross payload and gross payload/entry weight ratio as a function of entry weight for the solid flare configured vehicle. Payload fraction is seen to vary only slightly between entry weights of 500 to 3000 pounds, with a nominal value of 47 percent. Also shown is a single point representing the payload fraction for a vehicle designed specifically for the 30 mb, see section e-2-a. The payload capability for vehicles using the extensible flare design is shown in Figure 3.2-53. These vehicles as presented in the previous section are based on a 110 inch core diameter to

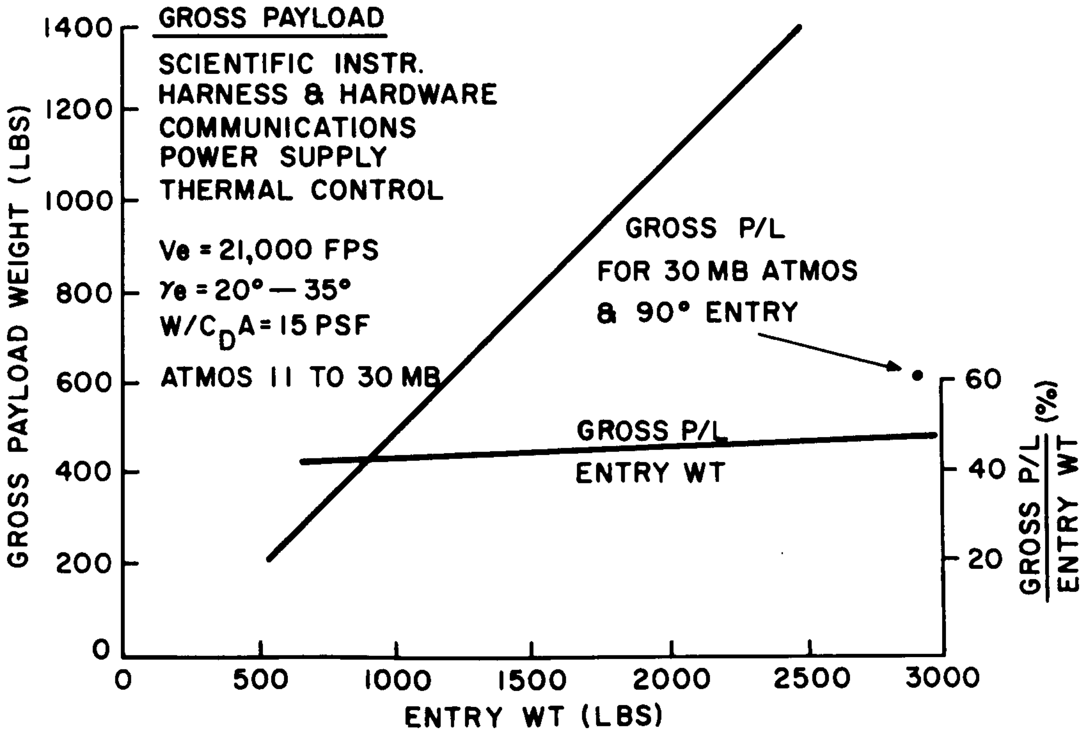


Figure 3.2-52. Gross Payload Capacity

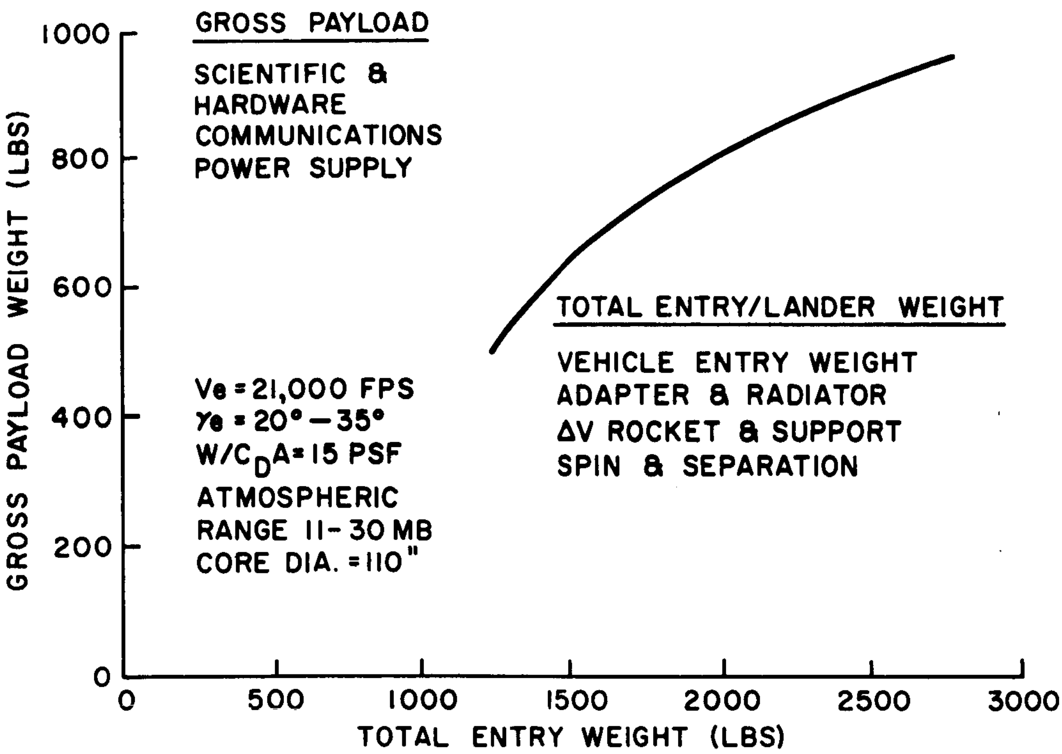


Figure 3.2-53. Gross Payload Capacity - Extensible Flare

which the flaps are attached. On Figure 3.2-54 the gross payload fraction for both fixed and extensible designs are shown as a function of total vehicle system weight. Total weight is the vehicle weight at the time of Bus/Lander separation. The curves are seen to converge at a total weight of 1240 pounds which is equivalent to a vehicle base diameter of 110 inches, the core diameter of the extensible flare design. The trend of decreasing payload fraction of the extensible flare vehicle is expected because of the rapid growth of the flare weight with total vehicle weight. A point will be reached at some gross vehicle weight where the weight of the added flap area to maintain a constant W/C_DA will result in no increase in gross payload.

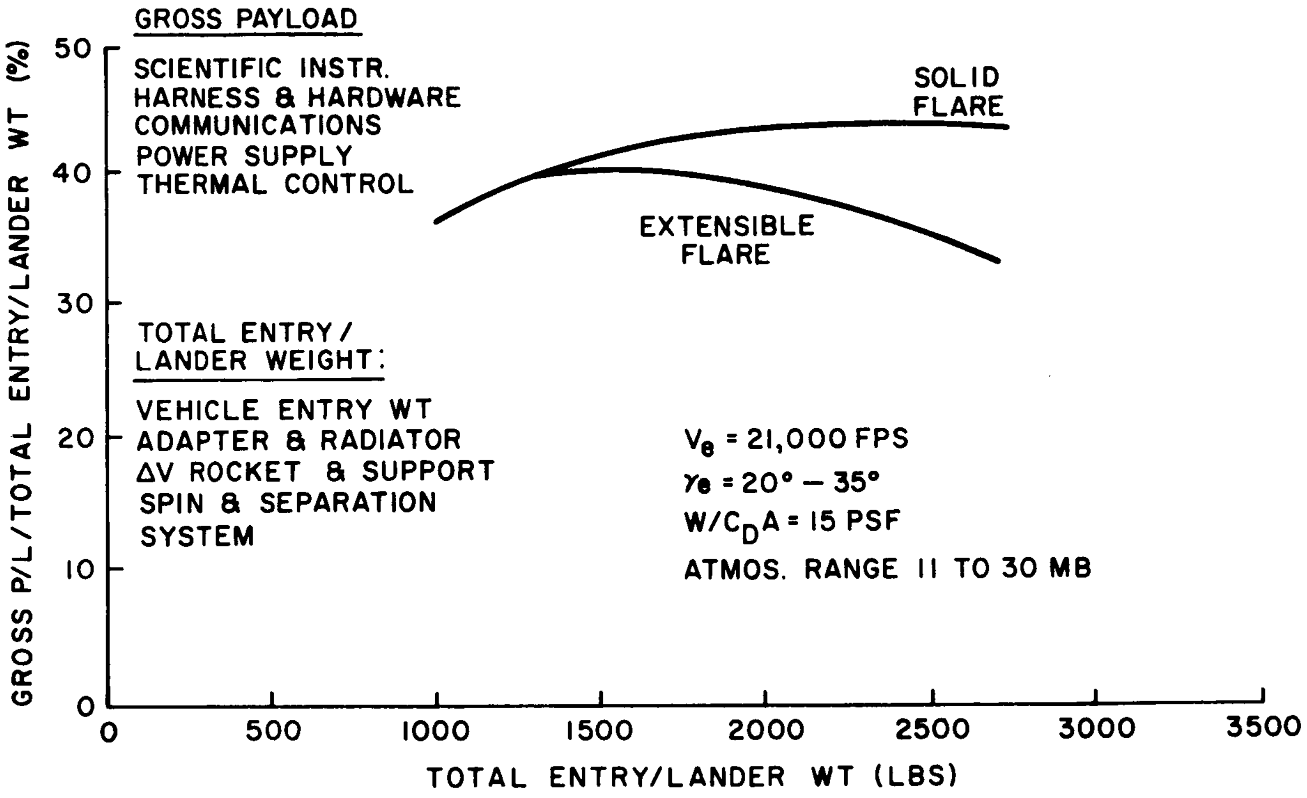


Figure 3.2-54. Gross Payload Capacity

2. Supplemental Studies

a. Vehicle Requirements for an Unlimited Entry Path Angle

If, for some reason, either because guidance accuracies are not attainable or because of the desirability of limiting to one the number of midcourse maneuvers, the entry corridor were opened to range from the skip limit to 90 degrees, it is of interest to investigate the effects on vehicle design and payload capability.

To meet the requirement of a 90 degree entry, the W/C_DA must be 8 psf or less to insure reaching Mach 2.5 at 20,000 feet altitude. Based on this and the MK II configuration, a vehicle has been synthesized and a summary weight statement assembled. The vehicle assumes a base diameter of 134 inches so that direct comparison can be made with the prime 1830-pound configuration of this study. A comparative weight statement is shown in Table 3.2-12. Gross payload is seen to decrease by 635 pounds. However, as was shown in the Mariner B Entry Vehicle study, a useful gross payload could be assembled at approximately 200 pounds for a short duration surface mission.

TABLE 3.2-12. ENTRY VEHICLE FOR 90 DEGREE ENTRY TO REACH MACH 2.5 AT 20,000 FT. COMPARING WITH 20-35 DEGREE ENTRY VEHICLE

Allowable γ_E	20-90°	20-35°
Allowable W/C_DA, psf	8	15
Entry Wt. for DB = 134 inches, lb	975	1830
Structure	309	367
Heat Shield	141	166
Retardation	283	414
Ground Orientation	20	26
Gross Payload Incl. Elect., Comm., Therm. Control	222	857
Radiator & Adaptor	41	61
ΔV Rocket & Spin & Sep.	78	151
Entry/Lander Total	1094	2042

b. Effects of Mars Surface Winds

The vehicles presented in this study have been designed to survive impact in a 40 mph cross wind. Because of the uncertainty associated with the determination of the wind velocity, it was thought to be of interest to investigate the effects of other values on payload. Wind velocity directly affects the amount of secondary crushup material provided for possible horizontal impact. Hence, for a constant weight Lander the additional weight required for the crushup will be taken from the gross payload. From the results of an analysis which was performed to determine impact attenuation requirements, several vehicles were investigated as a function of surface wind velocity. The results appear in Figure 3.2-55 where for an 1830-pound Lander, the gross payload/entry weight ratio is shown as a function of surface wind.

c. Payload Penalty in Designing to a Range of Martian Atmospheres

The prime systems analyzed in this study have been based on the presumption that the atmosphere encountered would fall in the range between 11 and 30 mb. This range of pressure that must be designed for has caused certain design penalties which would not be incurred if a specific atmosphere were identified and adequately defined. To determine the penalties attendant to design for the 11-30 mb range a brief study was made of designs for two specific cases, 30 mb and for 11 mb.

The 30 mb case will obviously result in a lighter system since the 11 mb was the dedesign case for structure and retardation system design. The following modes of comparison were considered in making this study:

A. Equal entry weight and entry angle

B. Equal base diameter and entry angle

C. Equal base diameter and changed entry angle.

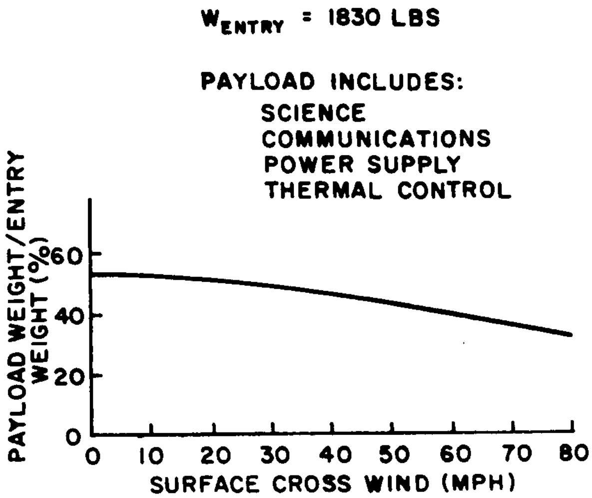


Figure 3.2-55. Payload Entry Weight vs. Surface Wind

Of these, A leads to a greatly increased W/C_DA allowable (for parachute deployment criteria) and requires a reduction in vehicle size to where it would be volume limited and inadequate to carry the payload allowable. Hence, it was discarded as a trivial solution. Cases B and C were studied and resulted in the system comparisons summarized in Table 3.2-13. Note that in Case C the allowable entry corridor has been widened to 20-90 degrees while W/C_DA has also been increased to allow a Lander gross weight of 2920 pounds within the base diameter of 134 inches. Case B resulted in a less realistic system which might also be volume limited. The significant parameter in this analysis is the gross payload/entry weight factor which has increased from 46.8 percent for the prime system to 59.8 percent for Case C. This discrete point has been plotted on the curve of gross payload/entry weight presented in

TABLE 3.2-13. PAYLOAD PENALTY IN DESIGNING TO RANGE OF ATMOSPHERES

GROUND RULE

CONSTANT BASE DIAMETER = 134 INCHES

	Case A	Case C	Case B
Atmosphere	11-30 MB	30 MB	30 MB
γ_e Range	20-35°	20-90°	20-35°
W/C_DA	15	24	50
Vehicle Entry Weight	1830	2920	6100
Gross Payload			
With Retro	857	1744	4025
w/o Retro	602	1104	2818
Gross PL/EW			
With Retro	46.8%	59.8%	66.0%
w/o Retro	32.9%	37.8%	46.2%

Figure 3.2-50. Thus the penalties in designing to a range of 11-30 mb atmospheres rather than to 30 mb specifically are:

a. Decrease in gross payload/entry weight ratio of over ten percent
b. Reduction of the entry corridor from 20-90 degrees to 20-35 degrees.

Investigation of the specific case of the 11 mb atmosphere yields a less significant penalty. However, it can be identified that although the retardation system for the 11-30 mb studies is based on a retrorocket sized to yield an impact velocity of zero in a 30 mb atmosphere, this system constraint no longer holds. More retro impulse can be added with a consequent reduction in crushup weight. If the system is designed for an impact velocity (at retro burnout) of zero in the 11 mb atmosphere, the only crushup required will be that for lateral winds and system tolerances. A preliminary analysis indicates that the weight saving may be approximately five percent of entry weight. Thus, the penalty in this case is a gross payload reduction of approximately five percent.

d. Effect of Firm Definition of Mars Atmosphere

The possibility exists that during the development stage of the Voyager program the characteristics of the Martian atmosphere would be more fully defined. Assuming that the Lander was designed for the 11 to 30 mb range, the following effects and changes are anticipated if a specific atmosphere were conclusively determined one year prior to launch:

A. Atmosphere found to closely resumble the 30 mb model.

1. Retrorockets could be eliminated.
2. Parachute sizes would change, however, this is not recommended because requalification would be required.
3. Payload would increase proportionately to a decrease in retardation system weight.
4. No change in configuration would be made.
5. Parachute deployment sensor would be modified.
6. Longer descent times might prompt a change in data bit rate.
7. If desired, the W/C_DA could be increased to approximately 50 psf for the same entry corridor. Payload would be increased proportionately. See section 3.2.2(E)(2)(c) for additional comments.

B. Atmosphere found to be on the order of 5 mb.

1. Additional velocity decrement would have to be supplied by larger retrorockets.
2. Present decelerator parachute would reach Mach 2.5 at approximately 15,000 feet, however, this is still considered adequate although, descent time would decrease.
3. Peak deceleration is a direct function of the atmospheric density parameter β. Should the defined atmosphere have the same, or a less steep density gradient as that of the present 11 mb-A atmosphere (11 mb-A presents the severest case), the system would be able to withstand the loading. However, if the defined atmospheric density profile were found to be much steeper, the structure would have to be redesigned for the additional loading.

C. Atmosphere found to resemble the "old" Upper Limit Model (135 mb).

1. Vehicle configuration would be grossly inefficient.
2. Recommend that present system be flown without modification to subsystems except for the deletion of the retrorockets.
3. Additional payload weight would be available subject to vehicle volume and communications limitation.

e. Plan for Program Decisions on Extensible Flare Vehicle

The Extensible Flare Configuration has been identified as having features which permit the vehicle to be readily modified leading to highly flexible hardware program. The flaps can be so designed that their size can be easily reduced or they can be completely eliminated if the change in W/C_DA is justified on the basis of new information on the atmosphere. If the Extensible Flare Configuration is selected to be used with the 120 inches OD shroud, the following steps should be taken to establish the program most certain to provide maximum effectiveness:

1. Determine W/C_DA for 110-inch diameter core section (without the extensible flare) as a function of Lander weight.
2. Determine core structural beef-up required as function of W/C_DA to permit use of that W/C_DA at the same and increased values of entry angle.
3. Determine W/C_DA required in various atmospheres to meet retardation requirements.
4. Design the extensible flare so that the flaps can be reduced on short notice or completely deleted.
5. Design the 110-inch diameter core structure so that it can be quickly strengthened to meet the loads resulting from higher W/C_DA or higher entry angles.
6. If the atmosphere on Mars is more fully defined during the program, trade-offs and designs will be available to make a decision to:
   a. Delete all or part of flaps
   b. Beef-up the structure
   c. Increase entry angle
   d. Increase payload

Planning potential changes as outlined will result in a program readily adaptable to changes long after the program start.

The Extensible Flare Configuration also offers flexibility in changing Lander gross weight from one mission to the next. This could be based on a low weight system designed to operate at a reduced W/C_DA and increased entry angle in the 1969 opportunity. In the 1971 opportunity the flare section could be added (or increased) to permit maximum weight allowable. Again in 1973, a reduced weight Lander could be based on the 110-inch diameter core, to meet reduced injection capabilities.

F. STERILIZATION

The system approach to sterilization is the same as used on the Saturn I Voyager Study, i.e., the Lander system which impacts the planet must be sterilized and the Bus or Orbiter which is never on an impact trajectory does not require sterilization. Hence, additional work in this area has not been required since the concepts developed during the Voyager Study are applicable.

The Lander system is sterilized in a ground facility and completely enclosed in a sterile container. After an operational check through the umbilical, it is mated to the Bus and the assembly installed on the launch vehicle. After launch the sterile container is removed only after completely leaving the Earth's atmosphere. The RTG unit may be installed during assembly on the ground and cooled by a water boiler until after launch or may be installed through a sterile lock on the launch vehicle shroud as late as possible in the launch sequence. Further details of these concepts may be found in the Voyager Study Report, Vol. V.

3.2.3 BUS CONFIGURATION DESIGN

A. CONFIGURATION STUDY AND SELECTION

During the Voyager-Titan study, the requirements for the Bus/Lander configuration varied considerably. The final systems decision was to study two types of Bus/Lander configurations.

1. Integrated Bus/Lander with the Lander having a base diameter of 134 inches.
2. Integrated Bus/Lander with the Lander having flaps and a base diameter of 110 inches.

Figure 3.2-56 shows the Bus/Lander combination with the 134-inch base diameter Lander and Figure 3.2-57 shows the Bus/Lander combination with the 110-inch base diameter Lander with flaps.

The systems decision to use an integrated Bus/Lander determined the design of the Bus. Refer to Section 2.2.6 for a complete discussion on integrated versus separate Bus designs.

The following subsystem constraints were imposed on the Bus design:

Lander:	Sterilization containter required so that a biological barrier is maintained between the Lander and the Bus.
Communications:	High-gain antenna required. No electronics required in Bus.
Power:	No power required in Bus.
Propulsion:	Midcourse correction engine and associated equipment required.
Guidance and Control:	Required in the Bus in order to obtain the correct entry corridor for the Lander.

The Bus is designed to meet the same Structural Design Criteria listed for the Orbiter. (Reference Section 3.3.)

The Bus configuration selected for the 134 inch-diameter Lander effectively utilizes available volume to mount required equipment and provide room for the Lander retro-engine. (Reference Figure 3.2-56.)

The shroud required for the Bus/Lander configuration is shown in Figure 3.2-58. Five inches per side are allowed in order to provide for shroud structure and deflections of the spacecraft due to launch vibrations.

The Lander attachment to the Bus is through four points spaced so that they match the location of attachment points to the booster. A biological barrier is provided for the Lander. Separation of the Lander from the Bus is arranged such that the Lander remains sterilized. The only junctions through the biological barrier are the four attachment points and an in-flight disconnect.

The high-gain antenna is mounted inside the Bus. The maximum size possible with easy packaging is three feet. The antenna has a fixed feed and is latched in place

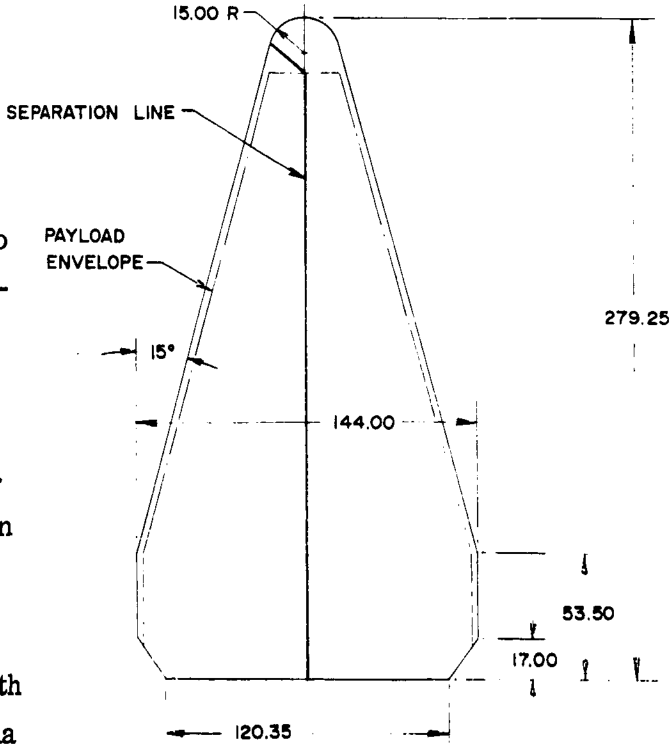


Figure 3.2-58. 134.0-Inch Base Diameter Lander Shroud

during launch. Immediately after separation from the booster the antenna is deployed and is used as the principal means of communication for the remainder of the mission.

The electronics necessary for the Bus operation are mounted within the structure against the outer skin. In addition, the Image Orthicon camera required by the Guidance and Control subsystem is mounted within the structure. The camera is mounted such that adjustments may be made in camera viewing direction in order to update the camera according to the launch date.

The midcourse correction engine (50 pounds thrust) is mounted on the lower edge of the Bus structure such that the plume will not impinge on the spacecraft. The thrust vector is oriented through the spacecraft CG. The maximum angular thrust vector misalignment is expected to be about 7 minutes of arc. The engine capability for thrust vector misalignment is ±6 degrees.

Omni-antennas and attitude control nozzles are located on four deployable booms located on the external surface of the Bus. (Reference Figure 3.2-56.) The two omni-antennas are located such that communications with Earth may be had regardless of spacecraft orientation. See Section 4.1 for more complete details.

The attitude control nozzles are located so as to have a failure mode of operation. In the nominal deployed position the nozzles will give a couple about the spacecraft CG. However, if the booms fail to deploy, the stowed position of the nozzles is such that a couple about the spacecraft CG may still be obtained as illustrated in Figure 3.2-59.

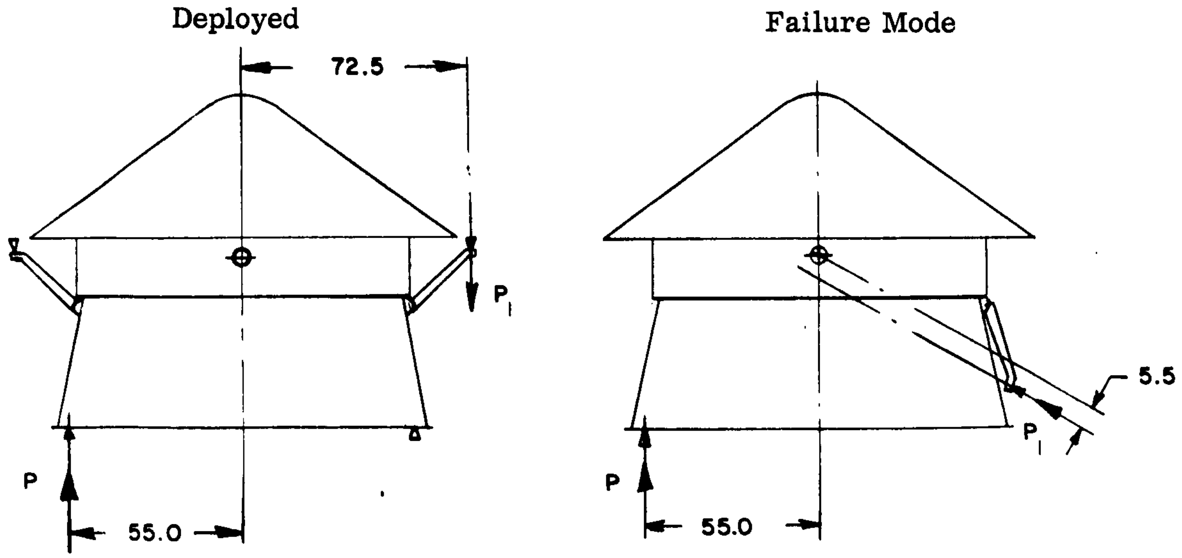


Figure 3.2-59. Couple Remaining After Boom Deployment Failure

Figure 3.2-60 shows the Bus/Lander configuration in the transit configuration. The booms are shown in the deployed position with the omni-antennas and attitude control nozzles. Figure 3.2-61 shows the Lander immediately after ejection from the Bus at approximately 150,000 n.mi. from the planet.

1. Separation

Separation of the spacecraft from the booster is by means of explosive nuts and bolts and spring actuators. The Titan IIIC booster has eight attachment points. Four of these attachments will be used to transfer tension and compression loads between the spacecraft and the booster. The remaining four attachments will transfer compression loads from springs built into fittings on the Bus structure. These springs will serve to provide the separation force between the launch vehicle and the spacecraft.

2. Weights

Detailed weights of the subsystem components have been calculated in order to determine payload capabilities and launch weights for the various opportunities. A weight statement for the Bus/Lander (134-inch diameter) combination is presented in Table 3.2-14.

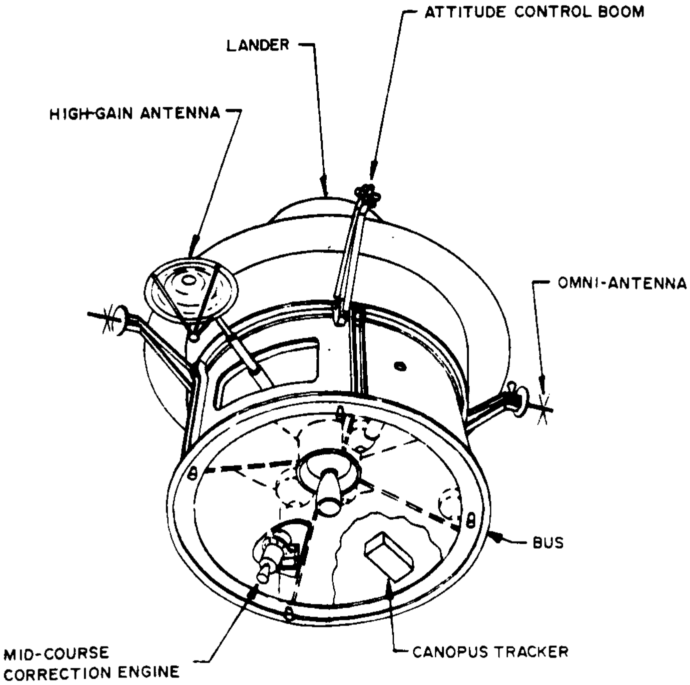


Figure 3.2-60. Bus/Lander Transit Configuration

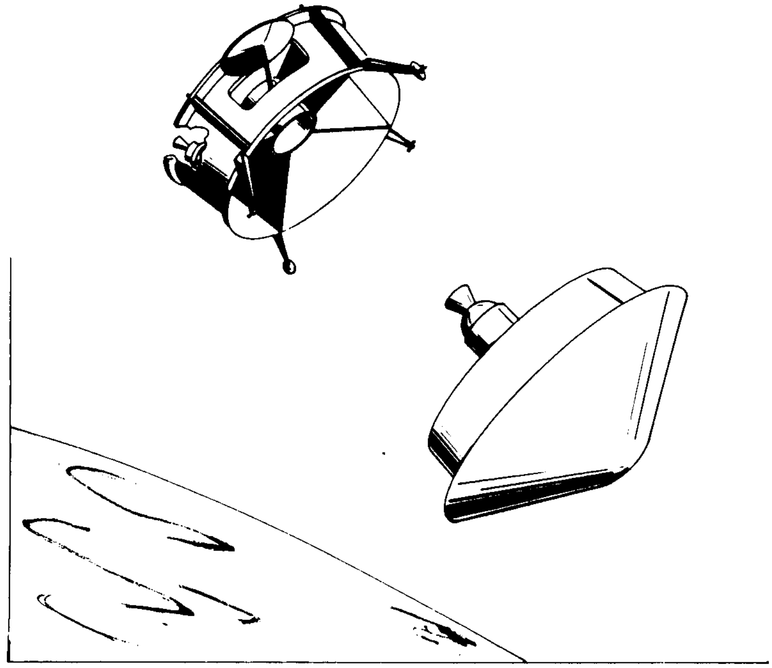


Figure 3.2-61. Lander Immediately after Ejection from Bus

TABLE 3.2-14. BUS/LANDER - BUS WEIGHT ESTIMATE

<u>Guidance and Control</u>			(154.55) lb
Image Orthicon		22.00 lb	
Optics	5.00 lb		
Head	4.00		
Electronics	13.00		
Switching Amp.		1.00	
Gyro Control		1.10	
Auto Pilot		2.00	
Antenna Drive Electronics		2.00	
Actuator Hinge (Ant.)		7.50	
Acutator Elevation (Ant.)		4.00	
Logic, Storage and Relays		14.25	
Power Supply		20.00	

TABLE 3.2-14. BUS/LANDER - BUS WEIGHT ESTIMATE (Cont'd)

Earth Sensor		6.50	
Canopus Scanner		5.50	
Gyro (Roll)		2.00	
Gyro (Yaw)		2.00	
Gyro (Pitch)		2.00	
Accelerometer		3.00	
Sun Sensors (Fine & Coarse) (7)		.80	
Payload Compartments Structure		18.00	
Pneumatic System		40.90	
Regulators (2)	4.00		
Solenoid Valves (12)	5.20		
Filters (2)	.80		
Latch & Check Valves	1.90		
High Pressure Transducer (2) Low Pressure Transducer (2)	1.00		
Temperature Sensors (4)	1.00		
Nozzles (12)	1.20		
Tubing	2.80		
Fitting and Bracketry			
Tanks (2)	6.40		
Gas	16.50		
<u>Bus Power Supply</u>			(1.50)
Inflight Disconnect		1.50	
170 Watts Power from Lander RTG			
<u>Communications</u>			(22.00)
Omni Antennas (2)		4.00	
Omni Cabling		6.00	
High Gain Antenna 3' Dia.		7.00	
Cabling		5.00	

TABLE 3.2-14. BUS/LANDER - BUS WEIGHT ESTIMATE (Cont'd)

Diagnostic Instrumentation		(12.00)
Propulsion		(39.33)
Hydrazine		
N_2H_4 Tank	3.40	
Thrust Chamber	2.50	
Residual	4.00	
Insulation	.10	
Fill Valve	.20	
Propellant Valves (4)	2.00	
N_2 Pressure Transducers	.10	
N_2H_4 Pressure Transducers	.30	
N_2 Sensors	.10	
Harness	1.33	
Lines, Fittings & Manifold	3.20	
Brackets	4.00	
N_2H_4 Temp. Sensors	.10	
Jet Vane System	2.30	
Chamber Pressure Transducer	.30	
Bladder	1.00	
Burst Discs (2)	.20	
Filters (2)	.40	
N_2 Relief Valve (4)	1.00	
N_2 Hand Valve	.30	
N_2 Solenoid Valve (2)	1.00	
N_2 Regulators (4)	4.80	
N_2 Filters (2)	.20	
N_2 Fill Valves (2)	.40	
N_2 Tank	4.00	
Gas (N_2)	2.10	

TABLE 3.2-14. BUS/LANDER - BUS WEIGHT ESTIMATE (Cont'd)

Thermal Control		(16.00)
Misc. Insulation	10.00	
Biological Barrier	6.00	
Vehicle Harnessing		(21.00)
Structure		(188.57)
Orbiter Body		
Honeycomb Shell	24.95	
Lander Support Fittings	14.44	
Top Ring	7.19	
Bottom Ring	9.34	
Inner Cone	9.91	
Tank Support - Ring	3.02	
Tank Support Ftgs.	2.00	
Top Cone	9.80	
Bulkheads	9.40	
Mid-Course Engine Support	0.54	
Ring	0.07	
Bottom Support	10.21	
Splice Rings	11.91	
Bulkhead Angles	6.62	
N_2 Tank Supports	.46	
Doublers	.63	
Antenna Support Structure, Stowed	5.74	
Top Support	4.83	
F-14 Tank Support	5.68	
Antenna Boom and Actuation Support	9.01	
Tie Down Fittings	1.40	
Gussets	1.20	
Att. Control Booms	15.68	
GSE Fittings, Hard Points	3.00	

TABLE 3.2-14. BUS/LANDER - BUS WEIGHT ESTIMATE (Cont'd)

I. O. Camera Support Str.	2.00	
Omni Supports	.40	
Misc. Supports	2.00	
Orbiter Hardware	17.14	
Total Bus		454.95
Lander		2042.00
Propellant		48.90
Δ Shroud Wt.		86.00
Launch Weight		2631.85 lb

3. Spacecraft Mass Properties

The Mass properties of the Bus/Lander are listed in Table 3.2-15 and Figure 3.2-62.

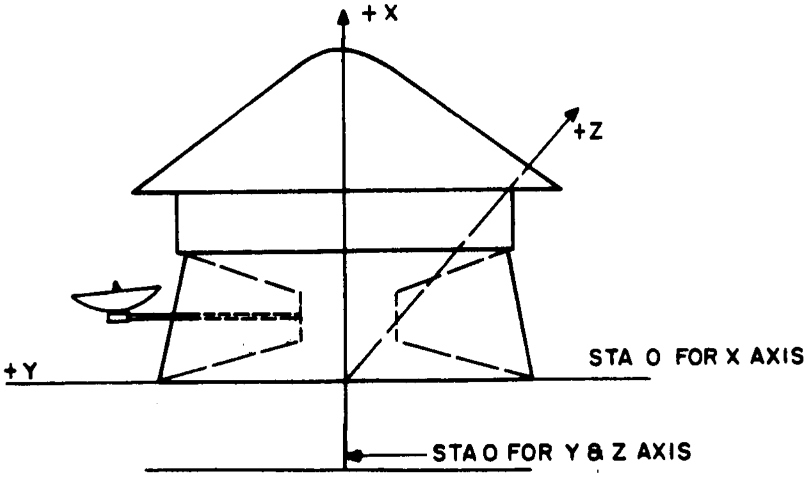


Figure 3.2-62. Bus/Lander Reference Data

TABLE 3.2-15. MASS PROPERTIES OF BUS/LANDER

Condition	I_oX	I_oZ	I_oY
	←	Slug - Ft^2	→
Launch	530.32	597.18	587.25
After Midcourse	525.99	582.94	570.40
After Lander Separation	251.71	143.54	130.82

4. Sequence of Events

The sequence of events for the Bus/Lander (134-inch diameter) is presented in Table 3.2-16. The block diagram illustrating the relationship between the various subsystems is shown in Figure 3.2-63.

TABLE 3.2-16. SEQUENCE OF EVENTS FOR 134-INCH BASE DIAMETER BUS/LANDER

The basic assumption is made that a successful ascent and injection into transit trajectory, with successful separation from the launch vehicle, will have been completed.

Launch Date: 9 May - 8 June 1971

Operation Sequence	Time to Complete Operation	Time to Begin Operation
A. Entry Into Transit Mode -		Time 0 (Immediately after separation from launch vehicle)
1. Turn on transponder	1 min	
2. Establish round trip phase lock	5 min	
3. Turn on attitude control subsystem	1 sec	
4. Deploy attitude control nozzle booms	30 sec	
5. Orient to Sun	16 min	
6. Deploy High-Gain Antenna and pre-program to point in Earth direction	10 min	
7. Switch to High-Gain from Omni by means of Earth Communication	1 sec	

TABLE 3.2-16. SEQUENCE OF EVENTS FOR 134-INCH BASE DIAMETER BUS/LANDER (Cont'd)

Operation Sequence	Time to Complete Operation	Time to Begin Operation
8. Orient to Canopus	48 min	
9. Earth verification of reference acquisition	60 min	
10. Shut down Gyros	1 sec	
B. First Mid-Course Correction - - - - - - - - - - - - - - - - - -		Time 0 + 1-2 weeks
1. Switch on Gyros	1 sec	
2. Commands received from Earth, acknowledged and verified by spacecraft and stored in the Programmer	7 min	
3. Orientation of spacecraft by means of attitude control subsystem to required orientation	12 min	
4. Firing of Main Engine	30 sec	
C. Reorientation to the Sun -		Immediately following engine firing
1. Commands read out by Programmer	10 sec	
2. Orientation to Sun and verification	6 min	
3. Orientation to Canopus and verification	6 min	
4. Sensor errors telemetered to Earth upon completion	1 min	
5. Shut down Gyros	1 sec	
D. Terminal Guidance Observation (2×10^6 nm from Planet)- - -		Time 0 + 213 days
1. Commands transmitted to the spacecraft and verified	40 min	
2. Body mounted I. O. camera turned on	5 min	
3. TV pictures taken of planet and background		

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9. Earth verification of reference acquisition	60 min	
10. Shut down Gyros	1 sec	
B. First Mid-Course Correction - - - - - - - - - - - - - - - - - -		Time 0 + 1-2 weeks
1. Switch on Gyros	1 sec	
2. Commands received from Earth, acknowledged and verified by spacecraft and stored in the Programmer	7 min	
3. Orientation of spacecraft by means of attitude control subsystem to required orientation	12 min	
4. Firing of Main Engine	30 sec	
C. Reorientation to the Sun -		Immediately following engine firing
1. Commands read out by Programmer	10 sec	
2. Orientation to Sun and verification	6 min	
3. Orientation to Canopus and verification	6 min	
4. Sensor errors telemetered to Earth upon completion	1 min	
5. Shut down Gyros	1 sec	
D. Terminal Guidance Observation (2×10^6 nm from Planet)- - -		Time 0 + 213 days
1. Commands transmitted to the spacecraft and verified	40 min	
2. Body mounted I. O. camera turned on	5 min	
3. TV pictures taken of planet and background		

TABLE 3.2-16. SEQUENCE OF EVENTS FOR 134-INCH BASE DIAMETER BUS/LANDER (Cont'd)

Operation Sequence	Time to Complete Operation	Time to Begin Operation
E. Final Trajectory Correction (1×10^6 nm from Planet) - - - -		Time 0 + 219 days
1. Switch on Gyros	1 sec	
2. Commands transmitted to spacecraft and verified	40 min	
3. Orientation of spacecraft by means of attitude control subsystem to required orientation	12 min	
4. Firing of Main Engine	30 sec	
F. Reorientation to Sun -		Immediately after engine firing
1. Commands read out by Programmer	10 sec	
2. Orientation to Sun and verification	6 min	
3. Orientation to Canopus and verification	6 min	
4. Sensor angles telemetered to Earch	10 min	
5. Shut down Gyros	1 sec	
6. TV pictures taken of planet and background		
G. Lander Ejection (150,000 nm from planet) - - - - - - - - - -		Time 0 + 224 days
1. Switch on Gyros	1 sec	
2. Commands transmitted to the spacecraft and verified	40 min	
3. Orientation of spacecraft to desired orientation	12 min	
4. Physical attachment of Lander to Orbiter broken	1 sec	
5. Lander is separated from Orbiter $\Delta V = 1$ ft/sec	1 sec	
6. Lander spin up to 60 RPM	30 sec	
7. Lander ΔV rocket engine fires	15 sec	
8. Bus is dead.		

B. THERMAL CONTROL

The Voyager Bus temperature control system will utilize a passive design concept complemented by heater power to be contributed by the Lander RTG. Electronics equipment of the Guidance and Control subsystem to be used in the Bus is identical to that used in the Orbiter (Reference Section 3.3). The propellant tanks will have temperature controlled by heaters such that the tanks will be kept within the operating extremes of temperature. The payload packages will be mounted against the Bus structure such that a view to free space is obtained.

Freon 14 tanks for attitude control and N_2 tanks for propellant tank pressurization have specific temperature limitations. These temperatures will be obtained by means of insulation blankets and temperature controlled heaters.

With the Lander facing the sun during transit, the Bus and its contents will remain in complete shadow. In order to obtain appropriate temperatures for the equipment located in the Bus, maximum use will be made of energy emanating from radiator surfaces. To transfer this energy to the aft of the vehicle, the upper cone exposed to the radiator will be covered, on both of its sides, with a high emittance coating. The lower cone and the shell structure will also have a high emittance coating applied to their internal surfaces; on the other hand, those surfaces viewing space will have a coating of low emittance, on the order of 0.1.

With high internal emittances, temperature gradients will be reduced; low values for external skins will tend to reduce heat losses and still create a permissible temperature environment for piping, valves, and other ancillary components. The temperature requirements for the individual nitrogen, freon, and hydrazine tanks will be satisfied by providing energy via strip heaters whenever the natural Bus thermal environment alone cannot suffice. Tanks demanding heater power will be enveloped in a blanket of multilayer insulation to make effective use of this power; incoming energy will be continuously available from the Lander RTG.

The guidance and control components will be packaged within the Bus in such a manner as to allow the 28 watts of constant heat generation to maintain components of intermittent duty cycles within acceptable temperature limits.

Should the insulation, required on the RTG radiator inner surfaces to protect the Lander aft section against excessive temperatures, deprive the Bus of sufficient heat inputs, consideration would be given to pointing the lower Bus cone towards the sun

TABLE 3.2-16. SEQUENCE OF EVENTS FOR 134-INCH BASE DIAMETER BUS/LANDER (Cont'd)

Operation Sequence	Time to Complete Operation	Time to Begin Operation
E. Final Trajectory Correction (1 x 10^6 nm from Planet) - - - -		Time 0 + 219 days
1. Switch on Gyros	1 sec	
2. Commands transmitted to spacecraft and verified	40 min	
3. Orientation of spacecraft by means of attitude control subsystem to required orientation	12 min	
4. Firing of Main Engine	30 sec	
F. Reorientation to Sun -		Immediately after engine firing
1. Commands read out by Programmer	10 sec	
2. Orientation to Sun and verification	6 min	
3. Orientation to Canopus and verification	6 min	
4. Sensor angles telemetered to Earch	10 min	
5. Shut down Gyros	1 sec	
6. TV pictures taken of planet and background		
G. Lander Ejection (150,000 nm from planet) - - - - - - - - - -		Time 0 + 224 days
1. Switch on Gyros	1 sec	
2. Commands transmitted to the spacecraft and verified	40 min	
3. Orientation of spacecraft to desired orientation	12 min	
4. Physical attachment of Lander to Orbiter broken	1 sec	
5. Lander is separated from Orbiter ΔV = 1 ft/sec	1 sec	
6. Lander spin up to 60 RPM	30 sec	
7. Lander ΔV rocket engine fires	15 sec	
8. Bus is dead.		

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Should the insulation, required on the RTG radiator inner surfaces to protect the Lander aft section against excessive temperatures, deprive the Bus of sufficient heat inputs, consideration would be given to pointing the lower Bus cone towards the sun during transit. This technique does not have the advantage of providing a constant thermal environment for the Bus, because of the variation of the solar constant. Therefore, this concept will not be employed unless other means of meeting Bus temperature requirements are not available. Another possible technique would consist in transporting a portion of the RTG excess energy, by means of a liquid loop, to the Bus shell structure, which would then act as a radiator (see Section 3.4). By thus raising the shell structure temperature, an adequate thermal environment would be provided to the Bus equipment.

C. STRUCTURAL DESIGN

The Bus structure supports and protects the Entry Lander vehicle along with components and fuels necessary for this mission under the environmental conditions expected. In this capacity, the Bus assumes the aspects of an adapter between Lander and booster. The design is basic, and entails the provisions of suitable load paths from the lander to the booster for reacting loads which occur under the lateral and longitudinal dynamic load conditions.

The lateral load condition which induces overturning moments and side shears on the bus shell is the most critical. However, a suitable design for longitudinal vibrations must be adhered to or this condition would become limiting.

This is accomplished by dynamically uncoupling the lander and the bus. The Lander, through its inherent stiffness for withstanding entry loads has a comparatively high natural frequency. Therefore, by designing the Bus at a suitably lower natural frequency, it will act as a filter for loads transmitted through it at frequencies approaching the Lander natural frequency and a low transmissibility factor will be obtained. At the same time, at the Bus system resonance (approximately 50 cps) the Lander modes are primarily rigid body motions which are not significantly amplified.

1. Load Paths

To distribute loads into the booster in the prescribed manner, the only practical configuration for the Bus is that of a shell. Longerons are spaced so that axial and overturning couple loads are transmitted into the eight tension compression fittings provided by this booster. (Reference Figure 3.2-56 Titan III Interface Plane.) A ring at the base transfers shears from the shell skin to the 24 shear pins, and a ring at the top plane helps distribute the localized lander loads into the shell in a uniform manner.

The Lander is supported at four tie down points, which are oriented with four of the shell longerons. These four longerons form the major load path to the booster. The remaining longerons are of lighter construction and receive only loads distributed through the shell by shear lag phenomena and from some of the internally mounted bus sub-systems. To effectively package and deploy the high-gain antenna, a cutout at a longeron location must be provided. This cutout is reinforced by a ring and at this location the cut longeron acts basically as a panel breaker.

Loads transmitted to the booster with this configuration are, of course, unequal. However, in the interest of a lightweight structure, there is no justification to provide the internal structure to both the Lander and the Bus necessary to distribute loads equally to all eight tie-down points. The maximum compressive load at a tie-down point, which occurs at one of the main longeron locations under a combined lateral and longitudinal load, is well within the attachment capability. Any attempt to equalize loads must also consider an eight point tie-down and separation arrangement for the Lander which is more complex than the present proposed system.

The outer shell of the Bus is aluminum honeycomb construction. The superiority of honeycomb for shell stabilization has been discussed previously, and is comparable to the lander. The like construction methods and materials will keep thermal stresses and deformations to a minimum.

D. DEPLOYABLE DEVICES

There are three items which require deployment during a Bus/Lander Mars 1971 mission.

1. Attitude control nozzles and onmi antennas
2. The high-gain antenna
3. The Lander which must be separated from the Bus with minimum disturbances.

The booms, with attitude control nozzles and omni antennas attached, are in a stowed position (Reference Figure 3.2-56). Immediately after separation from the launch vehicle, an explosive actuator releases the boom tie down from the Bus structure. Torsion springs are employed to deploy the boom to a fixed position.

TABLE 3.2-17. WEIGHTS FOR THE TWO-LANDER SYSTEM

Guidance and Control	154 lb
Power	5
Communications	22
Diagnostic Instrumentation	12
Propulsion	39
Thermal Control	29
Harness	21
Structure	260
Total Bus	542 lb
Landers - 2 at 1386.5 Pounds	2773
Mid-Course Propellant	49
Adapter and Δ Shroud Weight	236
Launch Weight	3600 lb

TABLE 3.2-18. WEIGHT PENALTIES

Design	One Lander	Two Landers
Individual	Basic Unit Wt.	Basic Unit + 237 Pounds
Unified	Basic Unit +147 Pounds	Basic Unit +237 Pounds

Power Supply	3 lb
Structure	71
Adapter	100
Thermal Control	13
Shroud Δ Wt.	50
TOTAL	237 lb

Adapter	47 lb
Structure	71
Shroud Δ Wt.	29
TOTAL	147 lb

3.3 ORBITER SPACECRAFT SYSTEM

The Orbiter is designed to be launched on a Titan IIIC, and to carry a selected group of experiments and instrumentation to Mars. The selected launch window is May 6, 1971 to June 5, 1971.

3.3.1 SYSTEM CONFIGURATION STUDY AND ANALYSIS

Figure 3.3-1 shows the selected transit configuration and the axes of the Orbiter.

The orbit selected for the Orbiter phase of the Voyager program is 1000 n. mi. perigee by 2278 n. mi. apogee. This places definite constraints on the Orbiter system design. The instrumentation package requires a continuous view of the planet and the selected orbit gives a precession rate of the orbit plane about the planet of 1.72 degrees per day.

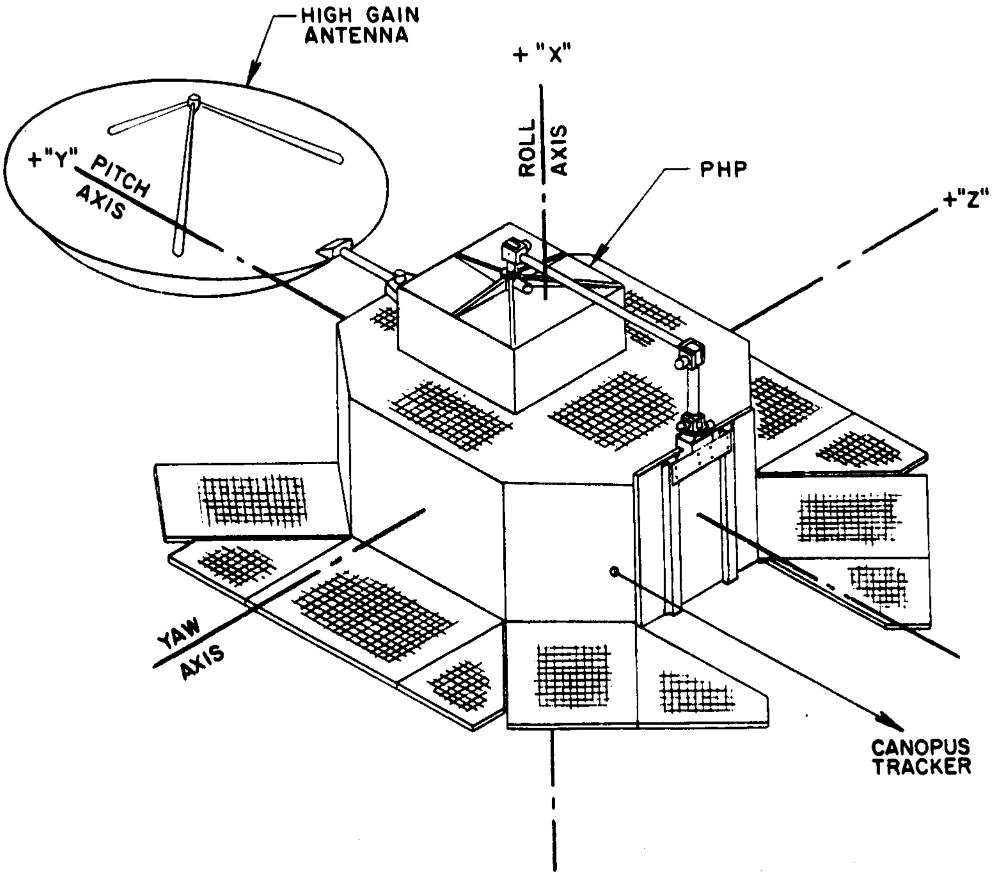


Figure 3.3-1. Orbiter Axes

This has a great influence on the design of the Planet Horizontal Package (PHP) actuation mechanism. (See Section 3.3.2 (A)(2)(d).

Figure 3.3-2 defines a propellant factor used in determining the propellant required for various missions. For example, the 1971 mission has an orbit of 1000 x 2,278 n.mi. By entering the curve at the 2,278 n.mi. apogee, this propellant factor is 0.88. Since the Orbiter weight is 1815 pounds, the propellant needed for orbit insertion is 1598 pounds.

A. SHROUD LIMITATION

The Titan IIIC has been designed for Dynasoar bending moments and shears. In addition, a standard shroud is available and is being used during the launching of the development vehicles. The standard shroud is shown in Figure 3.3-3 as well as several other shrouds considered for the Titan IIIC. The problem inherent in the use of larger diameter or bulbous shrouds is the buffeting on the booster due to the change in diameter. For this study it was decided that no shroud would be considered greater than the 154-inch diameter MORL shroud suggested by the booster manufacturers.

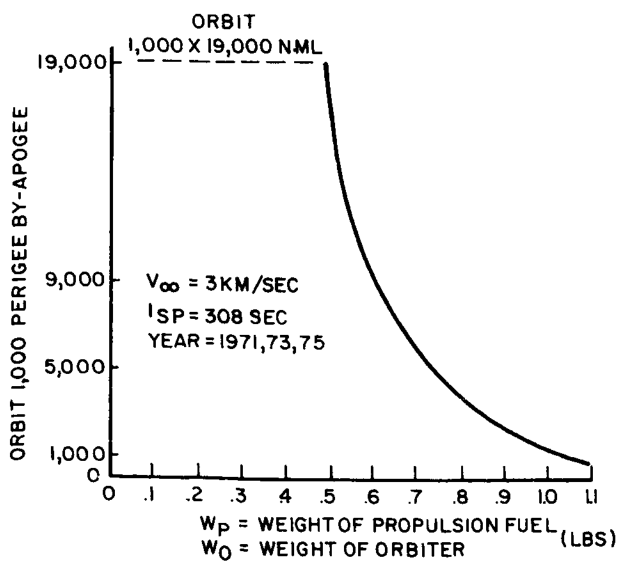


Figure 3.3-2. Propellant Factor

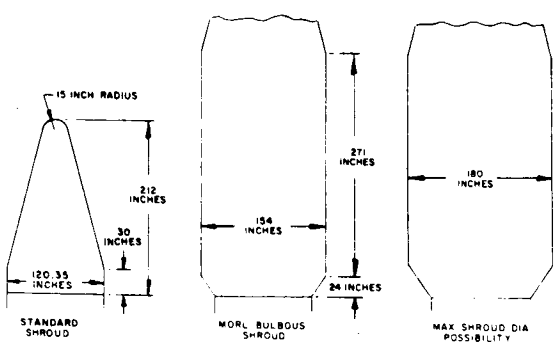


Figure 3.3-3. Titan IIIC Shroud

1. Shroud Data — Titan IIIC

In order to predict the changes in weight for the various shrouds under consideration, a group of curves were developed based on knowledge of the standard shroud. Figure 3.3-4 has been developed based on several assumptions. These assumptions are:

1. The shroud thickness for larger diameters will be the same as for the standard shrouds.
2. The slope of the larger diameter shrouds will be the same as for the standard shroud.

To use Figure 3.3-4, proceed as follows:

1. Determine shroud outside diameter
2. Select required cylindrical section
3. Determine shroud weight
4. Take 10 percent of (shroud weight - 591 pounds)
5. Subtract weight calculated from gross payload.

2. Booster Limitations

The launch vehicle considered (Titan IIIC) limits the Orbiter design in several ways. It limits the amount of injected weight, and the launch vehicle - spacecraft interface controls the basic configuration of the spacecraft body. Figure 3.3-5 indicates the method of attachment to the booster. The attachment points indicated dictate the location of the eight main longerons on the Orbiter body and on the adapter. Shear loads are transmitted to the launch vehicle through 24 shear pins at the launch vehicle-spacecraft interface (Missile Station 77). The combination of these two factors dictates a semi-monocoque type of structure capable of transmitting concentrated loads through longerons or fittings, and transmitting shear loads through the shell.

A further limitation is caused by the launch vehicle configuration. No spacecraft structure can project below the launch vehicle-spacecraft interface at Missile Station 77. This factor, along with the main engine nozzle configuration, determines the adapter size.

3. Subsystem Limitations

Certain limitations are placed on an Orbiter design by the requirements of the various subsystems. These limitations are listed following Figures 3.3-4 and 3.3-5.

The high-gain antenna is stowed inside the spacecraft during launch. The antenna is in a locked position so that survival during launch environment is ensured. Release of the high-gain antenna is by means of an explosive actuator and occurs immediately after separation of the spacecraft from the launch vehicle.

Separation of the Lander from the Bus is by means of a cold gas system using canted nozzles. See section 3.2.2 for further details of the separation system. Release of the Lander will occur immediately before the spin rockets fire. Lander release is by means of explosive bolts and nuts designed so that the biological barrier between the Lander and the Bus is not violated. The first incremental motion of the Lander is produced by means of flat leaf springs. This ensures a positive separation in conjunction with the Lander cold gas system.

E. BUS/LANDER (110-INCH DIAMETER WITH FLAPS)

Figure 3.2-60 shows the Bus/Lander configuration with the 110-inch diameter flapped Lander. The subsystems and the design of the Bus are identical to that required for the 134-inch base diameter Lander. The end configuration, however, is slightly different in order to allow for the variations in the Lander design. See Section 3.2.2 for a more complete description of the 110-inch diameter flapped Lander.

One advantage of this configuration is that the same shroud used for the all Orbiter can be used for the Bus/Lander. There will be difficulties, however, in designing a Lander sterilization container which will provide an effective biological barrier between the Bus and the Lander.

Attachment of the Landers to the Bus, thermal control, structural design, communications, and separation from the launch vehicle are all similar to the Bus/Lander (134-inch diameter) design. Because of the similarity between designs, the injected weight is very close, 2546 pounds for the 134-inch diameter Bus/Lander combination versus 2524 pounds for the 110-inch diameter flapped Bus/Lander combination.

F. TWO LANDER CONFIGURATION

In order to utilize the total launch vehicle payload capability, the possibility of launching two Landers was studied. Figure 3.2-64 indicates the proposed configuration. Since the spacecraft is a fly-by Bus configuration, each Lander has a ΔV rocket motor. The dimensions of the required ΔV motors determine the configuration of the Bus.

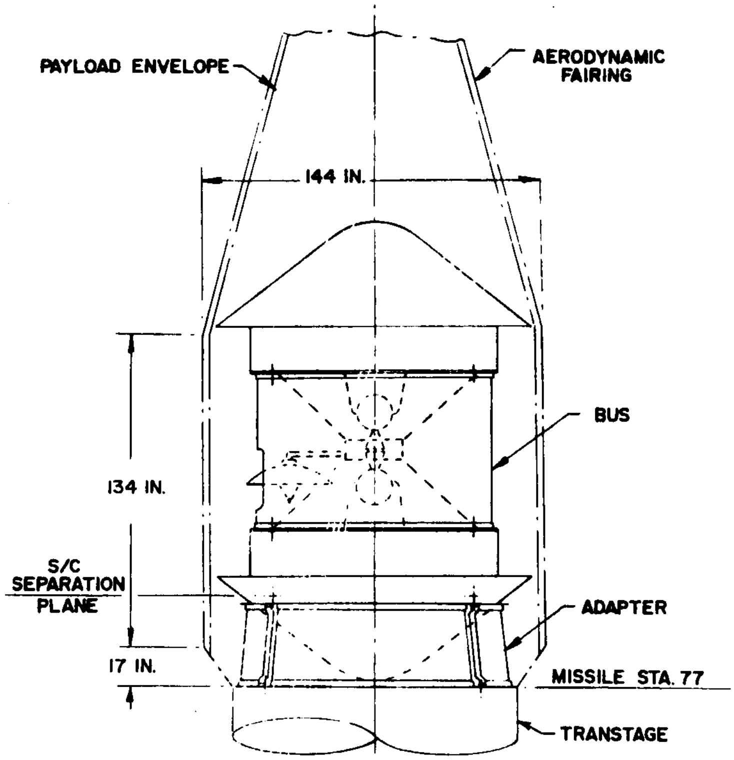


Figure 3.2-64. Integrated Bus/2 Landers — 134-Inch Diameter Launch Configuration

Two 134-inch base diameter Landers are used and are mounted one below and one above the Bus. For this configuration, an adapter is required. The adapter mounts to the launch vehicle at Missile Station 77 and attaches to the lower Lander at four attachment points which protrude through the heat shield.

The Bus is located between the two Landers. Subsystem requirements for this Bus are the same as previously determined for the Bus/Lander (134-inch diameter) and Bus/Lander (110-inch diameter, with flaps) configurations. The weights for the two Lander system are shown in Table 3.2-17.

It is obvious that a weight penalty will occur if a design is made which is suitable to both one or two Landers.

Table 3.2-18 shows the weight penalties if it is required to design to carry either one or two Landers.

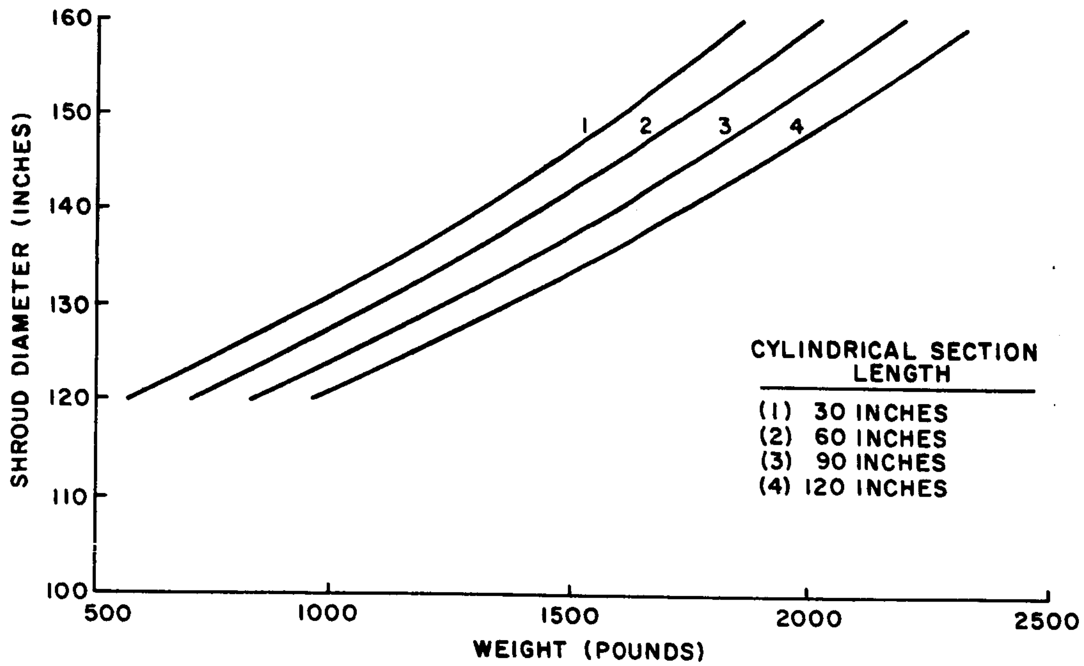


Figure 3.3-4. Shroud Weight vs. Shroud Diameter for Various Length

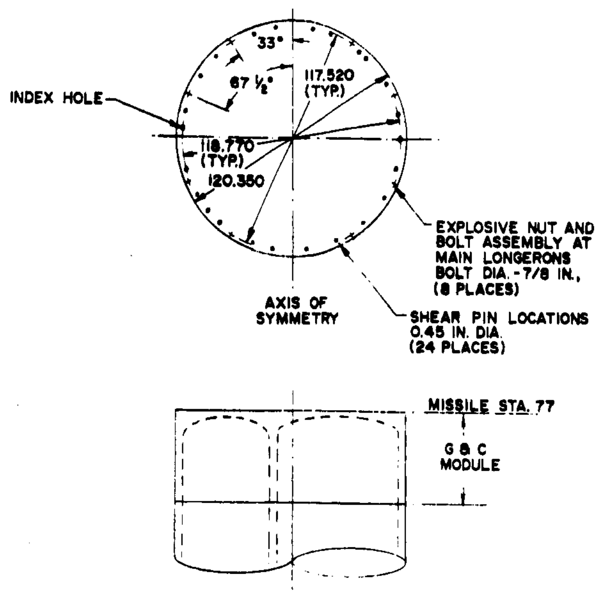


Figure 3.3-5. Titan IIIC Attachment Interface

1.	Power:	Solar power requires a large area for solar cells.
		RTG power requires specific location of the unit.
2.	Communications:	High-gain antenna required. Omni-antennas required for communications when high-gain antenna is stowed.
3.	Propulsion:	Main engine thrust must go through the spacecraft CG. Provisions must be made for attitude control jets.
4.	Guidance and Control:	Sensors must be located with open field of view.
5.	Structure:	Spacecraft must be mounted on the Titan IIIC and be packaged within an extended standard shroud.

B. RTG ORBITER AND SOLAR POWERED ORBITER

Two separate types of Orbiter design were considered. These types were solar powered with solar cells and RTG powered with a Radioisotope Thermoelectric Generator unit. Figures 3.3-6 through 3.3-9 indicate the various differences inherent to the two conceptual designs.

A comparison of the two designs is shown in Table 3.3-1 in a condensed sequence of events:

TABLE 3.3-1. SEQUENCE OF EVENTS

Solar Power Orbiter	RTG Orbiter
Launch	Launch
Deploy Solar Panels	No Maneuver
Orient to Sun/Canopus	Orient to Sun/Canopus
Deploy Antenna	Deploy Antenna
Midcourse	Midcourse
Terminal Guidance	Terminal Guidance
Orbit Insertion	Orbit Insertion
Deploy PHP	No Maneuver
Sun Oriented Spacecraft Planet Oriented PHP Earth Oriented Antenna	Planet Oriented Spacecraft Earth Oriented Antenna

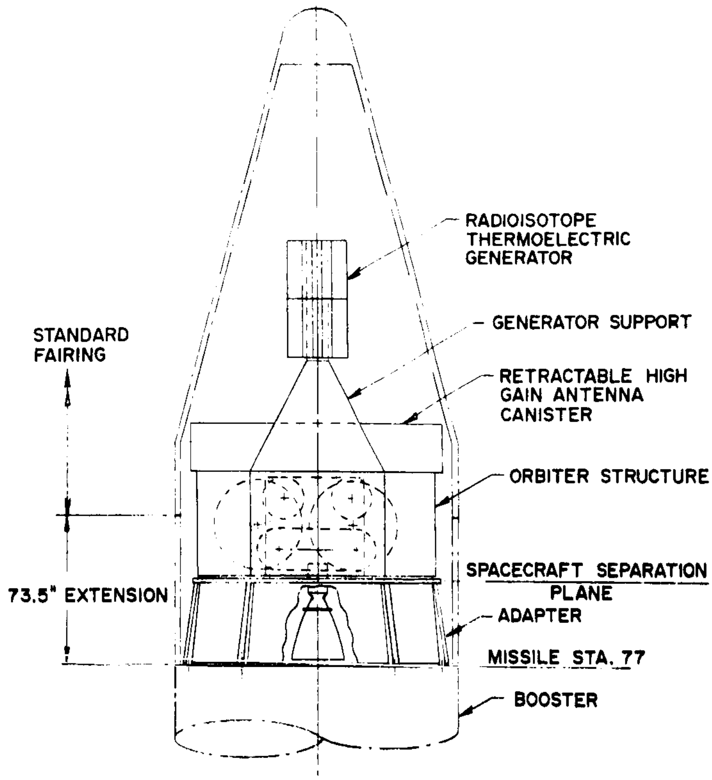


Figure 3.3-6. 600-Watt RTG Orbiter (Launch Configuration)

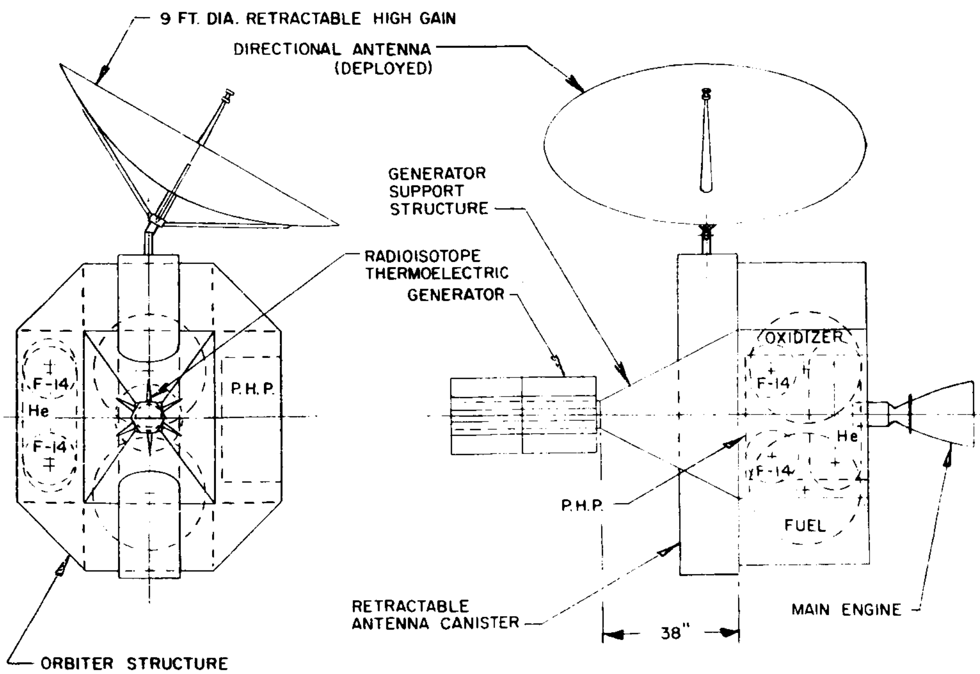


Figure 3.3-7. 600-Watt RTG Orbiter (Orbiting Configuration)

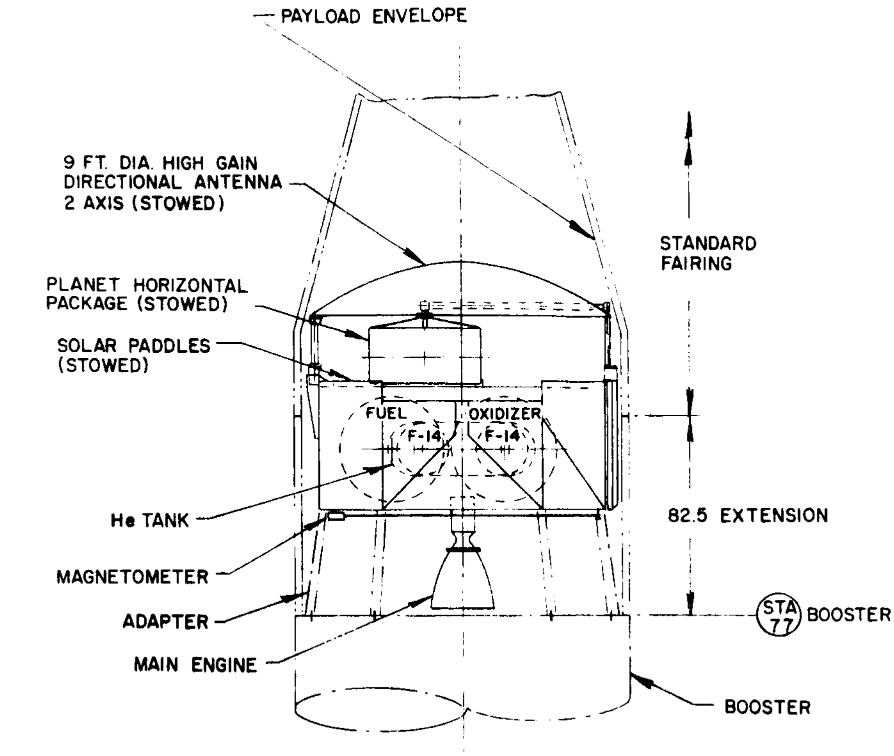


Figure 3.3-8. 600-Watt Solar Orbiter (Launch Configuration)

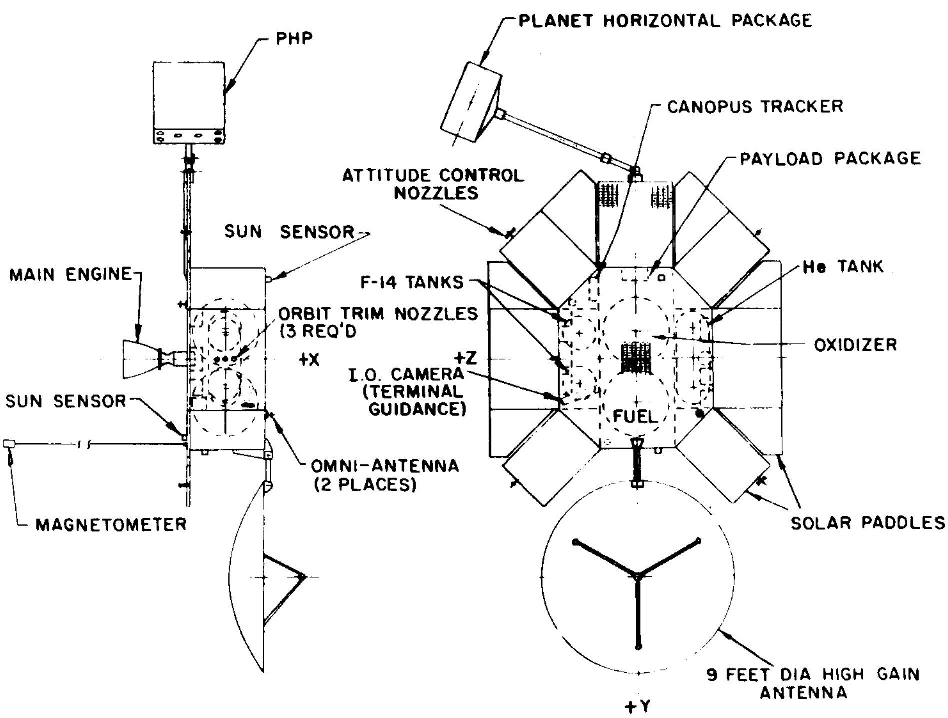


Figure 3.3-9. 600-Watt Solar Orbiter (Orbiting Configuration)

The decision between the Solar powered and RTG powered Orbiters was based on the following considerations:

1. System Weight

There is a 347-pound payload capability for the solar design; versus 409-pound payload capability for the RTG design. This capability favors the RTG design even though payload has not been identified as yet for the total 347-pound capability of the solar design. A detailed weight statement is shown for the RTG powered Orbiter in Table 3.3-2. A similar weight statement for the solar powered Orbiter is shown in Section 3.3-2.

2. Availability

The capability of solar cell producers to provide cells is not presently taxed to the maximum. Although the requirements of the 600 watts would require planning and preparation, the capability to produce the cells is in existence. On the other hand, the isotope required is not available, nor is a readily available production capability. The AEC states that isotope could be available; however, there is no program at present requiring the production of the isotope.

3. Reliability

The Solar powered Orbiter has a vehicle reliability of 0.628 for 3 months versus a reliability of 0.583 for the RTG Orbiter. (Reference Section 5.1).

4. Mission Variations

Each design of the Orbiter has distinct variations, planet oriented for the RTG, and sun oriented, with PHP, for the solar power design. However, these variations have been considered in arriving at a reliability number for the two designs. Either design will obtain and transmit the required information to Earth.

5. Life Time

Each design is limited by attitude control gas requirements. However, in growth potential, the RTG Orbiter could be passively oriented in a circular orbit about the planet thus giving an extremely long life time. This still leaves the problem of antenna pointing to Earth to be solved.

6. Multi-Planet Capability

The Solar powered Orbiter is extremely limited in capability of travel to other planets unless revisions in power requirements are made. On the other hand, the RTG Orbiter design is ideally suited for multi-planet capability. The RTG will supply heat for thermal control and a constant level of power regardless of distance from the Sun.

7. Costs

Although complete costs are not available, there is sufficient information to make a comparison. Data available indicates a total cost of $6,800,000 for 600-watt solar power. This cost is for one opportunity with four units. On the same basis, the cost of the isotope alone (Cu 244) for the RTG is $24,000,000. This does <u>not</u> include any costs for design and development of the RTG unit. Obviously, the cost figures favor the Solar powered Orbiter.

8. Special Ground Handling Conditions

Other than careful handling a Solar cell power supply offers no particular ground handling problems. However, because it contains radioactive material there are significant ground handling problems associated with a Radioisotope Thermoelectric Generator. The safety requirements and potential problems associated with this phase of the mission are not unique to the Voyager application and are the same as those that would be encountered in the handling, shipping, and installation of any large isotope source. Considerable experience exists in this area since similar problems have been encountered and solved on other programs.

The fuel would be encapsulated in a hot cell and the capsule then transported to the launch site in a suitable shipping container. A permit from the Bureau of Explosives would be required to transport the encapsulated fuel from the point of origin to the launch site. The shipping container would have to provide, (1) the biological shielding required by the Interstate Commerce Commission regulations, and (2) protection for the isotope capsule under all credible transportation accidents. Having arrived at the launch site the capsules would be transferred from the shipping container to a heat exchanger that would also serve as the generator loading mechanism. If possible the generator would be loaded after it was installed onboard the launch vehicle. All or most of this work would be done remotely to minimize the hazard to ground personnel.

Details of ground handling would be available because of the use of an RTG in the Entry Lander. However, it is obvious that ground handling of an RTG unit is considerably more difficult than ground handling of a Solar cell power supply.

Certain items were deemed of greater importance than others. Availability, reliability, and cost were ranked ahead of the other items considered.

Based on evaluation of all items, the Solar powered Orbiter was selected as the optimum design for this study.

TABLE 3.3-2. DETAILED WEIGHTS 600-WATT RTG ORBITER

Guidance and Control			(188.25) lb
Image Orthicon		22.00 lb	
Optics	5.00 lb		
Head	4.00		
Electronics	13.00		
Switching Amp.		1.00	
Gyro Control		1.10	
Auto Pilot		2.00	
Antenna Drive Electronics		2.00	
Actuator Hinge (Ant.)		7.50	
Actuator Elevation (Ant.)		4.00	
Logic, Storage and Relays		14.25	
Power Supply		20.00	
Earth Sensor		6.50	
Canopus Scanner (+Pitch)		5.50	
Canopus Scanner (- Pitch)		5.50	
Horizon Scanner		13.00	
Gyro (Roll)		2.00	
Gyro (Yaw)		2.00	
Gyro (Pitch)		2.00	
Accelerometer		3.00	
Sun Sensors (Fine & Coarse) (7)		.80	
Payload Compartments Structure		18.00	

TABLE 3.3-2. DETAILED WEIGHTS 600-WATT RTG ORBITER (Cont'd)

Pneumatic System			56.10	
Regulators	(2)	6.20		
Solenoid Valves	(12)	5.20		
Latch Valve		1.80		
Filters	(2)	.80		
Check Valves	(2)	.10		
High Pressure Transducer	(2))	1.00		
Low Pressure Transducer	(2))			
Temperature Sensors	(4)		1.00	
Nozzles	(12)		1.20	
Tubing			2.80	
Shut-off Valves			5.00	
Tankage			7.60	
Gas F 14			23.40	
Orbiter Power Supply				(209.92)
600 Watt RTG (2 at 43.2 lb each)			186.40	
Regulator (Power Control)			3.52	
Inflight Disconnect	(3)		4.50	
RTG Support Str.			20.00	
Communications				(226.65)
S-Band Diplexer	(2)		2.00	
Omnidirectional Antenna	(2)		4.00	
Transponder	(2)		10.80	
Pre-Amplifier			2.00	
Power Amplifier	(3) (57W)		9.00	
High Voltage Power Supply	(3) (57W)		18.00	
Power Amplifier	(45W)		2.50	
Power Supply	(45W)		4.50	
Command Demodulator	(2)		6.00	
R. F. Switch			1.00	

TABLE 3.3-2. DETAILED WEIGHTS 600-WATT RTG ORBITER (Cont'd)

Isolator and Load		(2)		1.50	
Data Processor				12.25	
Multiplexer		(PHP)		10.00	
Buffer Unit				4.00	
Tape Recorder		(3)		45.00	
Command Decoder		(PHP)		4.00	
Programmer Unit				20.00	
Power Conversion & Control (Orbiter)				12.00	
Power Conversion & Control (PHP)				2.00	
Coax Cabling				12.30	
High Gain Antenna (9-ft)				28.00	
Payload Compartments Package Structure				15.80	
Diagnostic Instrumentation					(30.00)
<u>Payload</u>					(409.26)
Scientific				111.00	
U.V. Spectrometer		I-81	22.00		
Radio Altimeter		I-5	15.00		
I.R. Flux		I-2	3.00		
I.R. Spectrometer		I-1	29.00		
Magnetometer		I-23	5.00		
Micrometeroid Flux		I-55	2.50		
Mounting Provisions			5.50		
B.S. Radar & Antenna		I-85	13.00		
Charged Particle Flux		I-12	5.50		
Polarimeter		I-68	4.50		
Far UV Radiometer		I-96	3.00		
Visible Radiometer		I-79	3.00		
Television				114.80	
Image Orthicons	(4)		95.80		
Optics 20M	(1)	18.80			

TABLE 3.3-2. DETAILED WEIGHTS 600-WATT RTG ORBITER (Cont'd)

Optics 140M	(3)	3.00			
Camera Heads	(4)	16.00			
Electronics	(4)	52.00			
Mirror		2.00			
Misc. Controls		4.00			
Vidicons	(2)		19.00		
Optics	(2)	2.00			
Camera Heads	(2)	5.00			
Electronics	(2)	8.00			
Misc Controls		4.00			
Unidentified (Scientific)				183.46	
<u>Propulsion</u>					(340.91)
Fuel and Oxidizer System (Dry)					
Tanks 2 at 35.00 inches Diameter		75.00			
Residual		56.00			
Thrust Chamber		42.00			
Filters	(4)	.60			
Main Valves	(4)	10.00			
Fill & Purge Valves	(12)	6.00			
Filters & Orifices	(4)	2.50			
Latch Valves	(4)	4.50			
Transducers	(8)	3.20			
Shielding		4.00			
Harness		3.00			
Lines		4.00			
Brackets		5.00			
Gimbal System		50.00			
Includes: Hwd. Power Pk.					
Actuators					
Servo Valves					
Accumulator					

TABLE 3.3-2. DETAILED WEIGHTS 600-WATT RTG ORBITER (Cont'd)

Plumbing			
Oil, Ring, Bearings			
Pressurization System		74.11	
Tank	53.20		
Gas (He)	5.60		
Tubing & Connectors	10.00		
Clamps	.10		
Regulator	3.25		
Filter	.25		
Relief Valve	.31		
Squib Valve Norm. Open (2)	.75		
Squib Valve Norm. Close (2)	.75		
Orbit Correction Nozzle		10.00	
<u>Thermal Control</u>			(60.00)
Insulation: Orbiter		19.25	
Active Control Orbiter		10.00	
Equipment Insulation		6.00	
Equipment Active Control		7.75	
Timers		1.00	
Paint		5.00	
Grease		2.00	
Heaters (at .1 lb each)		4.00	
Misc.			
<u>Vehicle Harnessing</u>			(106.26)
<u>Structure</u>			(243.75)
Orbiter Body		199.75	
Honeycomb Sides	27.84		
Top Honeycomb Panel	16.10		
Edging Members	12.54		
Tank Supports	13.24		

TABLE 3.3-2. DETAILED WEIGHTS 600-WATT RTG ORBITER (Cont'd)

Honeycomb Bulkheads (Main)	29.42		
Honeycomb Bulkheads (Secondary)	17.32		
Tie Down Fittings	1.40		
Corner Gussets	1.20		
G.S.E. Fittings	3.00		
Misc. Support Bhd. for Engine Support	2.00		
Antenna Support Str.	4.86		
Magnetometer Boom & Support	2.50		
I.O. Camera Support Str.	2.00		
Misc. Supports & Brackets	2.21		
Boost Extrusions	22.56		
Hardware Including Solar Array	21.81		
Bottom Panel	22.69		
Equipment Housing Str.		44.00	
Framing L's	4.00		
Honeycomb Component Mounting	17.00		
Fittings	3.00		
Clips, Gussets, Supports	10.00		
Honeycomb Sides	6.00		
Hardware	4.00		
Total Orbiter			1815.00
Fuel (.88)			1598.00
Arrival Weight			3413.00
Mid-Course Correction Fuel			36.00
Injected Weight			3449.00
Adapter and Δ Shroud Weight			151.00
Total Payload			3600.00 lb
Orbit (n.mi.) 1000 x 2278			

3.3.2 ORBITER CONFIGURATION DESIGN

A. CONFIGURATION STUDY AND SELECTION

With the Solar powered Orbiter an obvious selection, it was required that all subsystems be integrated into a complete operating system. The major areas for design are itemized in the remainder of the report. The final selected configuration is then shown with the pertinent parameters.

B. DESIGN CRITERIA AND GENERAL DESIGN PHILOSOPHY

The basic spacecraft structure shall be of sufficient strength and rigidity to survive the critical loading conditions and environments, including, but not limited to, those enumerated in this section, without reducing the probability of the successful completion of the mission. Wherever feasible, the structure shall be designed to achieve minimum weight consistent with the high reliability inherently required of all spacecraft. The structure shall be designed by the critical flight, transit and orbital conditions, limiting the influence of non-flight conditions and environments such as handling and transportation loads by use of appropriate packaging and handling techniques.

C. DESIGN CONDITIONS AND ENVIRONMENTS

1. Purpose

This section defines the loads and environments for use in preliminary studies of an interplanetary spacecraft, using the Titan IIIC launch vehicle. The spacecraft shall suffer no degradation when exposed to the following environments.

2. Pre-Launch (Shipping, Handling and Storage)

a. Steady-State Accelerations

Table 3.3-3 indicates steady-state accelerations in the N_x, N_y, and N_z directions.

TABLE 3.3-3. STEADY-STATE ACCELERATION

	N_x	N_y	N_z
1. Hoist	+3.0	$\pm$0.5	$\pm$0.5
2. Air Transportation	$\pm$3.62	$\pm$1.82	$\pm$3.62
3. Ground Transportation -	Special Handling procedures will be adhered to so that above load factors are not exceeded.		

NOTE: The spacecraft is not complete when shipped (i.e., fuel tanks empty, major mass items are not installed, etc.)

b. Shock

Shock loading transmitted to the spacecraft from the shipping contractor shall be attenuated such that the loads in the spacecraft structure do not exceed the powered flight and air transportation steady state loads.

Allowable load limits will be based on fatigue considerations. A free drop of 1/2-inch maximum can be expected on the complete spacecraft under normal handling conditions.

c. Vibration

Vibration loads will also be attenuated by the shipping container so that the structural member loads do not exceed those experienced under the launch load conditions.

d. Temperature

Temperature extremes of -80°F to 125°F are to be expected during all phases of shipping, handling and storage. For the specific components that cannot withstand this environment, special handling techniques and packaging specifications that limit the temperature from -35°F to +125°F will be specified.

e. Pressure

Stored in container at 2.5 psig and 5 percent relative humidity referred to 70°F.

15.4 to 10.2 psia (0-10000 feet)

10.2 to 1.69 psia (10000-50000 feet) (air transport).

3. Powered Flight (Launch)

a. Steady State Accelerations

The expected steady state accelerations for the Titan IIIC are given in Tables 3.3-4a and 3.3-4b.

TABLE 3.3-4. VOYAGER-TITAN IIIC — EXPECTED POWERED FLIGHT ENVIRONMENTAL LOAD FACTORS

(a) STEADY STATE ACCELERATIONS

EVENT	N_x	N_{yz}
Max q	2.8	0.7
1st Engine Cutoff	3.9	Neg.

(b) GROUND WIND (PAYLOAD PROTECTED BY SHROUD)

CONDITION	N_{yz}
Steady State Wind-100%	0.158
2/3 Steady - 1/3 Gust	0.192

The vibration environment presented in Figures 3.3-10 through 3.3-13 is for the most part assumed as being representative. The information available at this time for the Titan III launch vehicle is not applicable to the range of payload weights which would include the 3600 lb of the Voyager spacecraft. Extrapolation of the available data to higher payloads is not acceptable since a known correlation between payload weight and expected random environment does not exist. It is known that a random vibration environment predominates, and that the Titan III contractor does not design for a sinusoidal environment. However, it is not recommended that sinusoidal loadings be neglected completely. It has been determined from other programs that sinusoidal vibratory loadings are much more significant to structural design than random vibration loadings, and in keeping with the philosophy of an operability assurance test program to reliably qualify the structure, system sinusoidal longitudinal and lateral vibration environments are presented (Figures 3.3-10 and 3.3-11).

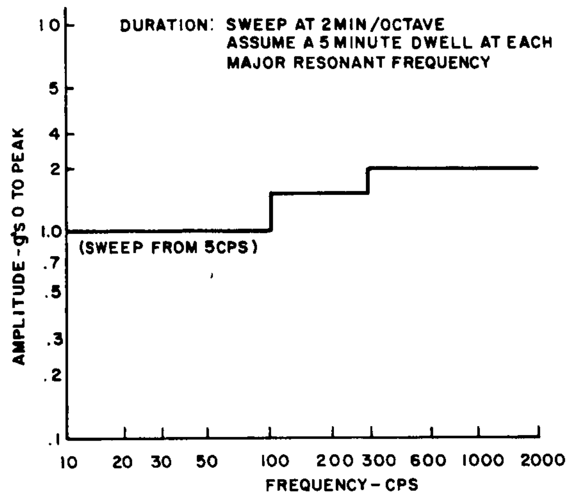


Figure 3.3-10. System Longitudinal Sinusoidal Vibration

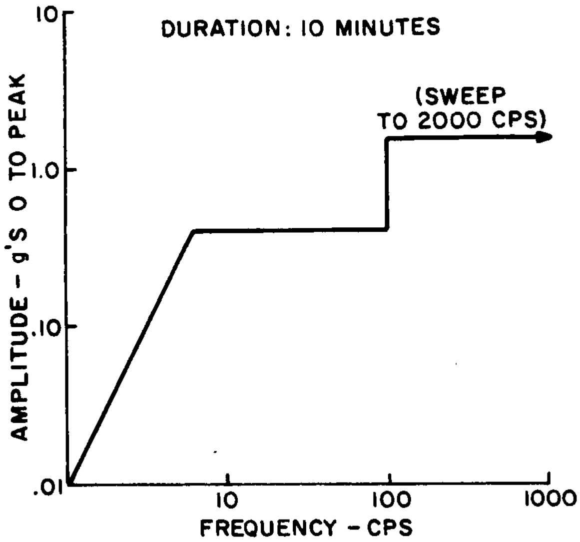


Figure 3.3-11. System Lateral Sinusoidal Vibration

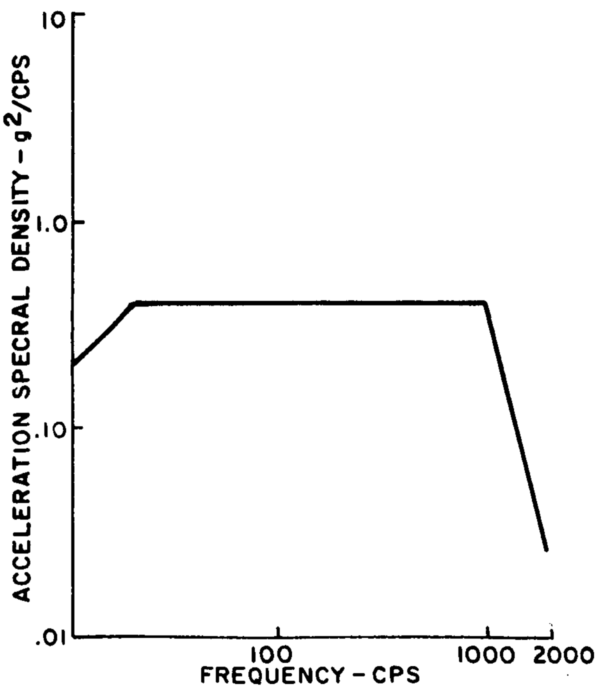


Figure 3.3-12. Sytem Longitudinal Random Vibration

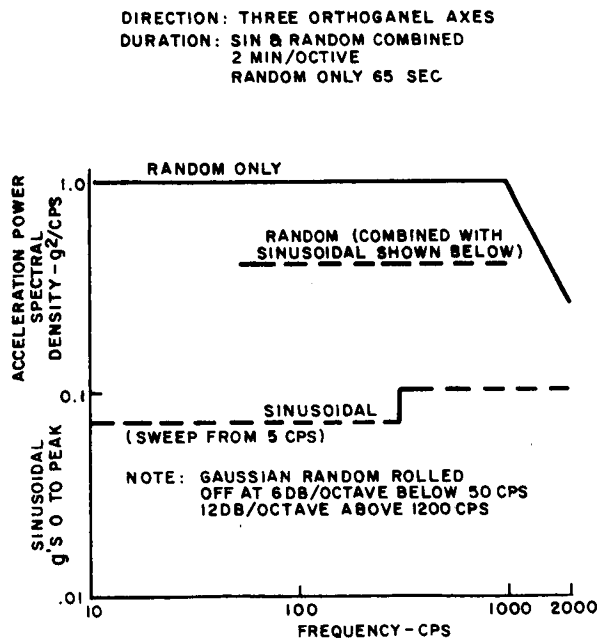


Figure 3.3-13. Component Vibration Qualification Levels

The power spectral density curve given in Figure 3.3-12 shows the order of magnitude of the random environment considered for the Titan IIIC. While the levels presented here tend toward conservatism, it is expected that the predominance of random vibrations in the loading spectrum will significantly affect the design, and, unless a realistic approach is taken on the determination of flight levels, an unnecessary structural weight penalty may be incurred.

Shock data is not presented at this time, since shock loadings can be greatly influenced by the method of tie down to the booster and the actual separation hardware used. Available Titan III shock data appears quite severe, and it is not known at this time if the levels are characteristic of the payloads considered and if these shock load "spikes" are also reflected in the relatively high random environment.

Figure 3.3-13 indicates the levels of vibration testing necessary to qualify components. Hard mounted components shall be capable of withstanding a 100 g sawtooth shock pulse with a 10.5 ms rise time and a 0.5 ms decay time. The shock pulse shall be assumed to act along each of the three mutually perpendicular axes. Appropriate attenuation will be accounted for in components isolated by brackets or vibration isolators.

b. Acoustical Field

Maximum sound pressure levels of 145 db at lift-off may be encountered. The sound will be quite random over a broad spectrum with the octave spectral maximum at about 100 cps. Levels inside spacecraft compartments may be about 10 db less.

c. Pressure

1. Aerodynamic Pressure at maximum q (estimated at 650 lb/ft^2) will be taken by the shroud and is not applicable for spacecraft design.
2. Internal pressure will reduce from 15.4 psia to a vacuum of 10^{-6} mm Hg during boost.

d. Separation Loads

Separation from the booster will be at a rate which minimizes the effect of separation loads on the design.

4. Planetary Transit and Orbital Environment

Vibratory and shock loads during orbit injections rocket engine firing are assumed less severe than during boost flight.

Other minor vibrations and accelerations as excited during PHP and Antenna orientation maneuvers and attitude and control gas jet firing will be encountered. These are not considered to be significant in the structural design of the vehicle.

a. Life Time

The operational life of the Orbiter is considered to be 225 days for transit and 90 days for planetary orbit.

b. Thermal

Component thermal environment will be passively and actively controlled to maintain temperature within specified design limits.

Extremes of -100°F to +250°F can be expected on structural items. Actual temperature distributions will be a function of:

1. Spacecraft Orientation
2. Solar Irradation
3. Planetary Flux
4. Planetary Albedo
5. Internal Power Dissipation
6. Thermal Radiation to Cold Walls and Free Space.

c. Radiation and Solar Flares

Charged particles, resultant X-rays, and gamma rays associated with Van Allen radiation belts and solar flares will be encountered. Cosmic rays are considered to have negligible effect. Some of the constituent protons will penetrate the structure to the equipment. Some of the energetic protons striking the structure may cause penetrating gamma rays of energy approaching their incident energies. Almost all of the electrons will be absorbed with generation of ion pairs, X-rays, or Gamma rays of smaller individual energy (per particle).

d. Meteoroid Protection

The meteoroid protection requirements for the Voyager spacecraft are not as severe as for manned missions, and no weight expenditure of any consequence is anticipated in designing for adequate protection.

The current knowledge on penetration mechanics recommends a bumper or split skin design. By its construction, the Voyager vehicle affords adequate protection to critical electronic components and fuel tankage since the outer shear panels in most cases act as effectively spaced bumpers. In the case of panel mounted electronics, if the base plate required for thermal reasons is inadequate for meteoroid shielding, it may be necessary to foam fill the honeycomb sandwich core immediately behind it since it is known that the core cells channel of the energy of impact, and the full benefit of the split face sheets is not realized. A polyurethane foam of as little as 1.2 pounds per cubic foot density can be used.

e. Scientific Instrumentation

The Orbiter has been designed to accommodate the instruments and experiments shown in Table 3.3-5.

TABLE 3.3-5. INSTRUMENTS AND EXPERIMENTS ACCOMODATED BY ORBITER DESIGN

No.	Name	Inst. No.	Wt.	Accum. Wt.	Power Watts
1	Magnetometer	I-23	5	5	5
2	I.R. Multichannel Radiometer Flux	I-2	3	8	3
3	Solar Multichannel Radiometer	I-79	3	11	3
4	Television 4 I.O. 2 Vid.		115	126	140
5	Charged Particle Flux Geiger Tubes and Ion Chamber	I-12	55	132	1
6	Far UV Radiometer	I-96	6	138	3
7	Micrometeroid Flux	I-55	8	144	1
8	B.S. Radar (Ionospheric Profile)	I-85	13	159	2
9	Polarimeter-Skylight Analyzer	I-68	4.5	163	4.5
10	I.R. Spectrometer	I-1	29	192	7
11	Retro Rocket and Hi Resolution Package		146	338	
12	Mass Spectrometer	I-43	6	344	6
13	Electron Probe (Langmuir Probe)	I-39	3	347	3

D. PHP — TWO AXIS VERSUS THREE AXIS

The experiments and instrumentation planned for the Mars 1971 Orbiter require that continuous viewing of the planet be maintained while in orbit. The planet pointing capability is obtained by mounting all equipment requiring planet viewing in a Planet Horizontal Package (PHP). This equipment package would at all times point towards the planet while in orbit except during the unwind operation which occurs once each orbit.

The difficulty of maintaining a continuous viewing of the planet is directly related to the amount of precession the orbit plane makes about the planet. The Voyager Design Study shows that the amount of precession is greatest for low circular or slightly elliptical orbits.

The orbit selected for the Orbiter is a posigrade orbit 1000 x 2278 n. mi. (Reference Section 2.4) with an inclination to the planet equator of 67 degrees. The nodal regression for this orbit is 1.22 degree/day (Reference Voyager Report) and the mean motion of the planet about the Sun is about 0.50 degree/day. Since, for a posigrade orbit, these two factors add, the average rotation of the orbit plane about the polar axis is 1.72 degree/day. For a ninety day lifetime, the total motion is 155 degrees.

The 155 degree total motion was then investigated to determine its effect on the design of the PHP actuating mechanism. Layouts and models were made to investigate the problems.

The questions to be answered were essentially two: Can the planet be viewed throughout a 155 degree orbit plane rotation, and if so, does the mechanism require a two axis or three axis control?

Figure 3.3-14 indicates the rotation of the orbit plane about the polar axis of the planet. Figure 3.3-15 shows position requirements of the PHP for orbit plane rotation angles of 0 degree, 90 degree and 180 degree. The position of the orbit plane at planet encounter is considered to be at a zero angle of rotation.

It was clearly evident from the layouts and from the models that three axes of control are required. With this knowledge, the Orbiter design could be continued.

Reference Section 4.2.2 for more complete discussion of the guidance and control of the PHP.

During launch and transit the PHP is in a stored position on the sun side of the spacecraft. The area of the PHP containing the TV lenses and other planet-pointing instruments is oriented along the pitch axis. The flight direction is along the yaw axis. This specific arrangement precludes the sun's rays impinging on the TV lenses except possibly for a short period of time during maneuvers. It also tends to reduce the possibility of micro-meteoroid impact because of other equipment being interposed between the TV lenses and micro-meteoroids.

E. SOLAR CELL PANELS

Power required for the Orbiter is 592 watts. The basic Orbiter design is mounted within the confines of the 120-inch O.D. shroud. Because of interference problems, five inches per side were allowed for clearance. The design selected is a octagon configuration modified to match the attachment points on the launch vehicle. Figure 3.3-16 shows the area of cells available if mounted on the structure (158 watts).

Using the maximum amount of body mounted solar cells still leaves 434 watts needed from another source. The remainder of the required power must be obtained from deployable solar panels. In order to reduce the number of hinges, it was decided to use only panel area which mounted directly to the Orbiter structure. When these panel sizes were determined and the required length of the PHP boom determined for effective view angles, it was immediately evident that all available area must be used in order to reduce PHP boom dimensions.

Figure 3.3-17 indicates the area needed to obtain the remaining required power. Figure 3.3-18 shows the final selected design of the Orbiter spacecraft. This design has seven deployable panels attached to the spacecraft structure; four of these

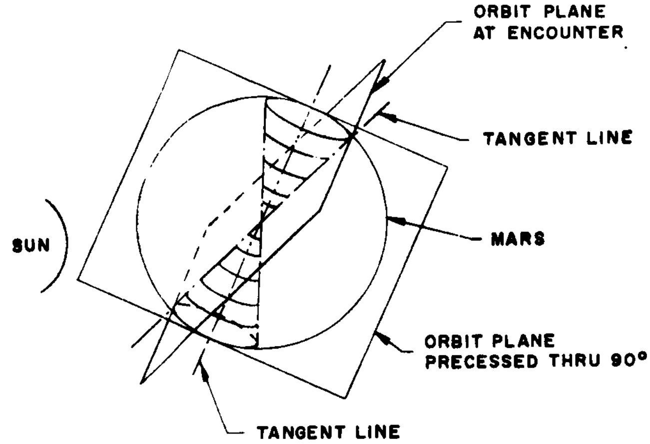


Figure 3.3-14. Rotation of Orbit Plane About Polar Axis

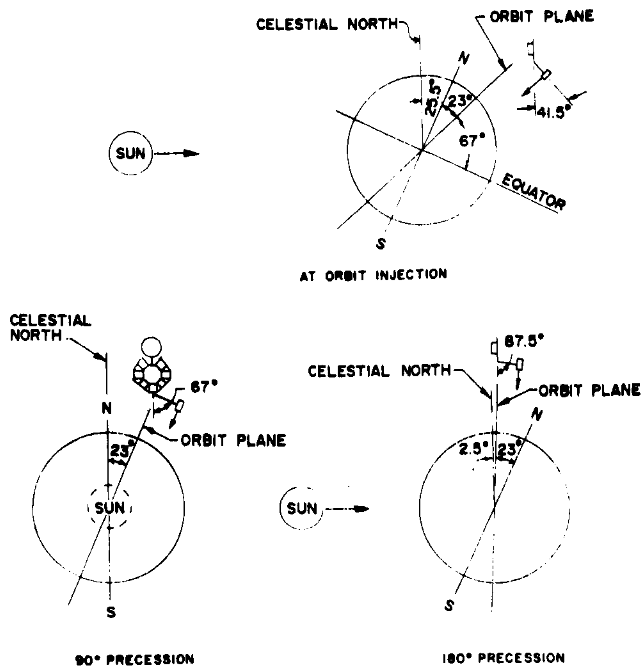


Figure 3.3-15. Position Requirements of PHP for Orbit Plane Rotation Angles

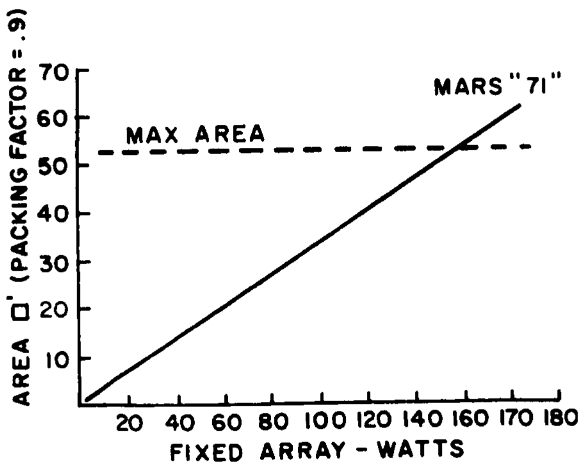


Figure 3.3-16. Watts vs. Area - Fixed Array

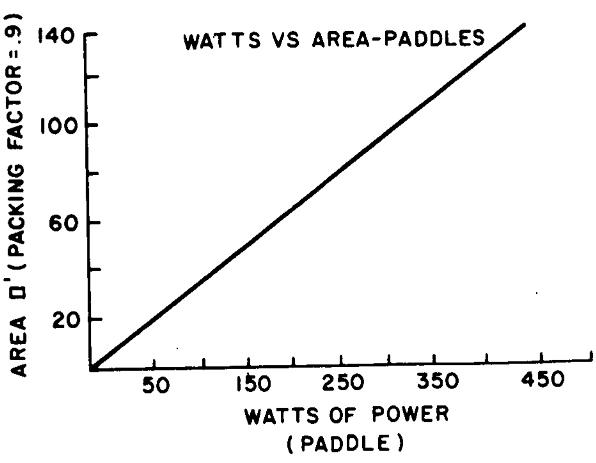


Figure 3.3-17. Watts vs. Area - Paddle

panels have additional deployments which are required in order to fill the area between panels. It should be noted that all panels are deployed immediately after injection into transit trajectory except the panel attached to the PHP boom support structure. This panel will not be deployed until after orbit injection. Power required during transit can readily be supplied by those panels which are deployed, with 511 watts being available.

The following factors have been used to calculate the required area of solar cells:

Year	Body Mounted	Panel Mounted
1971	3.02 watts/sq ft	3.21 watts/sq ft

Reference Section 4.3.2(B) for further details concerning this subject.

E. ATTITUDE CONTROL NOZZLES

The Guidance and Control subsystem for the Voyager-Titan is designed for an active attitude control system. This involves the use of gas jets located so as to efficiently utilize the available impulse to orient the spacecraft. As the Orbiter design evolved, it became evident that the most optimum location of the jets was on the outboard edges of the solar panels. This location gives a center-to-center distance of the jets of 196 inches. However, the design does require a flexible joint in the attitude control lines at the base of the solar panel to allow for expected motion when the solar panel is deployed.

The final precise location of the jets posed a problem. It was desired to locate them on the pitch and yaw axes. This, however, was not possible because of the problems caused by the High-Gain Antenna and PHP deployment. As shown in Figure 3.3-18, the final selected location is at 45 degrees to the pitch and yaw axes and on the outboard edges of the solar panels. This location was selected after due consideration of the problems inherent in the Guidance and Control subsystem. Section 4.2 discusses this problem in more detail.

The Mars All-Orbiter for 1971 carries 23.4 pounds of Freon 14 control gas. This gas weight is equated from a total impulse required of 1082 lb/sec. Reference Section 4.2 for determination of disturbances and Section 4.4 for calculations of gas weight.

F. PROPULSION SUBSYSTEM

For a Mars 1971 trip and a 1000 x 2278 nautical mile orbit the orbit insertion propellant to Orbiter weight ratio is 0.88 (See Figure 3.3-2). The propellant weight of 1598 pounds plus 92 pounds for mid-course correction and residual give a total propellant weight of 1690 pounds. The oxidizer to fuel weight ratio is 1.618.

The fuel and oxidizer tanks are located about the X-X Axis in accordance to the oxidizer/fuel weight ratio. The tanks are mounted so as to keep the center of gravity travel during burn to a minimum.

The main engine for mid-course control and orbit insertion is located at the base of the Orbiter structure. A space framework of titanium tubes is provided for the 900-pound thrust of the engine. Gimbaling is provided by hydraulic actuators.

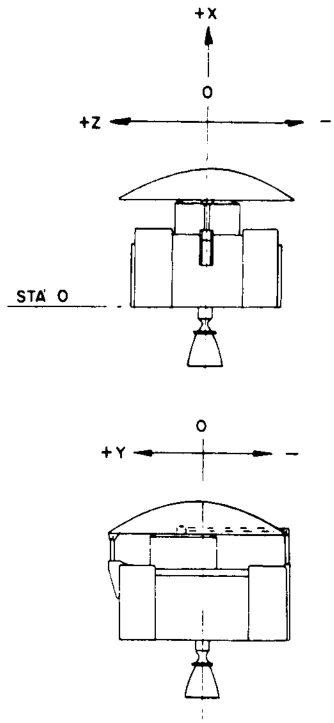


Figure 3.3-19. Orbiter Reference Data

Gimbaling of the engine allows ±6 degrees of engine thrust orientation. The maximum CG shift of the spacecraft is equal to about 10 minutes of arc thus allowing for minimum required control of the spacecraft.

G. SYSTEM WEIGHTS

A detailed weight statement (see Table 3.3-6) has been prepared for the 1971 Mars opportunity. Table 3.3-7 and Figure 3.3-19 show the mass properties of the Orbiter and indicate the probable CG shift during the mission.

H. SEQUENCE OF EVENTS AND BLOCK DIAGRAM

The sequence of events is presented in order to specify actions required of the Orbiter during the mission.

Figure 3.3-20 is the Orbiter block diagram delinating the relationships between the various subsystems.

TABLE 3.3-6. SOLAR POWERED ORBITER - WEIGHT ESTIMATE

Guidance and Control				(212.25)
Image Orthicon			22.00	
Optics		5.00		
Head		4.00		
Electronics		13.00		
Switching Amp.			1.00	
Gyro Control			1.10	
Auto Pilot			2.00	
Antenna Drive Electronics			2.00	
Actuator Hinge (Ant.)			7.50	
Actuator Elevation (Ant.)			4.00	
Logic, Storage and Relays			14.25	
Power Supply			20.00	
Earth Sensor			6.50	
Canopus Scanner (+ Pitch)			5.50	
Canopus Scanner (- Pitch)			5.50	
Horizon Scanner			13.00	
Gyro (Roll)			2.00	
Gyro (Yaw)			2.00	
Gyro (Pitch)			2.00	
Accelerometer			3.00	
Sun Sensors (Fine and Coarse)	(7)		.80	
Payload Compartments Structure			18.00	
PHP Drive Electronics			2.00	
Actuator			7.50	
Actuator			7.50	
Actuator - Amplifier-Drive Motor Logic (Third Axis)			7.00	
Pneumatic System			56.10	
Regulators	(2)	6.20		
Solenoid Valves	(12)	5.20		
Latch Valve		1.80		
Filters	(2)	.80		

TABLE 3.3-6. SOLAR POWERED ORBITER - WEIGHT ESTIMATE (Cont'd)

Check Valves	(2)	.10		
High Pressure Transducer	(2))	1.00		
Low Pressure Transducer	(2))			
Temperature Sensors	(4)	1.00		
Nozzles	(12)	1.20		
Tubing		2.80		
Shut-off Valves		5.00		
Tankage		7.60		
Gas F 14		23.40		
<u>Orbiter Power Supply</u>				(246.01)
Secondary Battery			26.30	
Regulator (Power Control)			3.52	
Charge Control (Based on 14 lb/Kw)			7.08	
Diodes			1.00	
Inflight Disconnect (3)			4.50	
Harness (Solar Array)			10.93	
Solar Array			192.68	
Body Panel Str. .415 lb/sq. ft.		22.69		
Paddle Str.		62.65		
Cells (187.52 sq. ft. x .5724) =		107.34		
52.26 sq. ft. at 3.02 W/sq. ft.		157.83 watts		
135.26 sq. ft. at 3.21 W/sq. ft.		434.17 watts		
	=	592.00 watts		
<u>Communications</u>				(226.65)
S-Band Diplexer (2)			2.00	
Omnidirectional Antenna (2)			4.00	
Transponder (2)			10.80	
Pre-Amplifier			2.00	
Power Amplifier (3) (57W)			9.00	
High Voltage Power Supply (3) (57W)			18.00	

TABLE 3.3-6. SOLAR POWERED ORBITER - WEIGHT ESTIMATE (Cont'd)

Power Amplifier (45W)			2.50	
Power Supply (45W)			4.50	
Command Demodulator (2)			6.00	
R.F. Switch			1.00	
Isolator and Lead (2)			1.50	
Data Processor			12.25	
Multiplexer (PHP)			10.00	
Buffer Unit			4.00	
Tape Recorder (3)			45.00	
Command Decoder (PHP)			4.00	
Programmer Unit			20.00	
Power Conversion and Control (Orbiter)			12.00	
Power Conversion and Control (PHP)			2.00	
Coax Cabling			12.30	
High-Gain Antenna (9 ft.)			28.00	
Payload Compartments Package Structure			15.80	
Diagnostic Instrumentation				(30.00)
<u>Payload</u>				(347.64)
Scientific			229.00	
Mass Spectrometer	I-43	6.00		
Electron Probe	I-39	3.00		
I.R. Flux	I-2	3.00		
I.R. Spectrometer	I-1	29.00		
Magnetrometer	I-23	5.00		
Micrometeoroid Flux	I-55	2.50		
Mounting Provisions		5.50		
B.S. Radar and Antenna	I-85	13.00		
Charged Particle Flux	I-12	5.50		
Polarimeter	I-68	4.50		
Far UV Radiometer	I-96	3.00		
Visible Radiometer	I-79	3.00		

TABLE 3.3-6. SOLAR POWERED ORBITER - WEIGHT ESTIMATE (Cont'd)

Retro Rocket and High Resolution Package			146.00		
Television				114.80	
Image Orthicons	(4)		95.80		
Optics -20M	(1)	18.80			
Optics 140M	(3)	3.00			
Camera Heads	(4)	16.00			
Electronics	(4)	52.00			
Mirror	(1)	2.00			
Misc. Controls		4.00			
Vidicons	(2)		19.00		
Optics (1000M)	(2)	2.00			
Camers Heads	(2)	5.00			
Electronics	(2)	8.00			
Misc. Controls		4.00			
Unidentified (Scientific)				3.84	
<u>Propulsion</u>					(340.91)
Fuel and Oxidizer System (Dry)				259.80	
Tanks 2 at 35.00" Diameter		75.00			
Residual		56.00			
Thrust Chamber		42.00			
Filters	(4)	.60			
Main Vlaves	(4)	10.00			
Fill and Purge Valves	(12)	6.00			
Filters and Orifices	(4)	2.50			
Latch Valves	(4)	4.50			
Transducers	(8)	3.20			
Shielding		4.00			
Harness		3.00			
Lines		4.00			
Brackets		5.00			

TABLE 3.3-6. SOLAR POWERED ORBITER - WEIGHT ESTIMATE (Cont'd)

Gimbal System	50.00			
Includes: Hwd. Power Pk. Actuators Servo Valves Accumulator Plumbing Oil, Ring, Bearings				
Pressurization System			74.11	
Tank		53.20		
Gas (He)		5.60		
Tubing and Connectors		10.00		
Clamps		.10		
Regulator		3.25		
Filter		.25		
Relief Valve		.31		
Squib Valve Norm. Open (2)		.75		
Squib Valve Norm. Close (2)		.75		
Orbit Correction Nozzle			10.00	
<u>Thermal Control</u>				(48.75)
Insulation: Orbiter			8.00	
Active Control Orbiter			10.00	
PHP Insulation			6.00	
PHP Active Control			7.75	
Timers			1.00	
Paint			5.00	
Grease			2.00	
Heaters (at .1 lb each)			5.00	
Misc.			4.00	
<u>Vehicle Harnessing</u>				(106.26)
<u>Structure</u>				(256.53)
Orbiter Body			212.53	
Honeycomb Sides		27.84		

TABLE 3.3-6. SOLAR POWERED ORBITER - WEIGHT ESTIMATE (Cont'd)

Top Honeycomb Panel		16.10	
Edging Members		12.54	
Tank Supports		13.24	
Honeycomb Bulkheads (Main)		29.42	
Honeycomb Bulkheads (Secondary)		17.32	
Tie Down Fittings		1.40	
Corner Gussets		1.20	
G.S.E. Fittings		3.00	
Misc. Support Bhd. for Engine Support		2.00	
PHP (Stowed Position) Support Str.		5.28	
Antenna Support Str.		4.86	
Magnetometer Boom & Support		2.50	
PHP Support Hinge		4.57	
I.O. Camera Support Str.		2.00	
Misc. Supports and Brackets		2.21	
Boost Extrusions		22.56	
Hardware Including Solar Array		21.81	
PHP Boom and Support		22.50	
PHP Hardware		3.12	
PHP			44.00
Framing L's	4.00		
Honeycomb Component Mounting	17.00		
Fittings	3.00		
Clips, Gussets, Supports	10.00		
Honeycomb Sides	6.00		
Hardware	4.00		
Total Orbiter			1815.00
Fuel (.88)			1598.00
Arrival Weight			3413.00

TABLE 3.3-6. SOLAR POWERED ORBITER - WEIGHT ESTIMATE (Cont'd)

Mid-Course Correction Fuel	36.00
Adapter and Δ Shroud Weight	151.00
Total Payload	3600.00

Orbit (n.mi.) 1000 x 2278

TABLE 3.3-7. MASS PROPERTIES OF THE MARS 1971 ORBITER

Condition	Weight	$\overline{X}$	$\overline{Z}$	$\overline{Y}$	I_0X	I_0Z	I_0Y
After Launch	3483.0	27.55	-	-	1196.5	825.1	741.7
After Midcourse	3447.0	26.47	-	1.00	1292.7	908.3	759.4
Before Orbit Injection	3447.0	27.60	-	-	1193.6	822.0	741.5
After Orbit Injection	1768.0	30.90	.11	.06	1054.7	669.7	752.6
After Equipment Deploy	1768.0	18.26	.09	-34.02	3400.3	3200.4	934.9

TABLE 3.3-8. SEQUENCE OF EVENTS

A basic assumption is made that a successful ascent and injection into transit trajectory, with successful separation from the launch vehicle adapter, will have been completed.

Launch Date: 6 May - 5 June 1971 Operation Sequence	Time to Complete Operation	Time to Begin Operation
A. Entry Into Transit Mode		Time 0 (Immediately after separation from launch vehicle.
1. Turn on transponder	1 min	
2. Establish round trip phase lock	5 min	
3. Turn on attitude control subsystem	1 sec	
4. Deploy Solar Panels	10 min	
5. Orient to Sun	16 min	
6. Deploy High-Gain Antenna and preprogram to point in Earth direction	10 min	
7. Switch to High-Gain from Omni by means of Earth communication	1 sec	

TABLE 3.3-8. SEQUENCE OF EVENTS (Cont'd)

Operation Sequence	Time to Complete Operation	Time to Begin Operation
8. Orient to Canopus	48 min	
9. Earth verification of reference acquisition	60 min	
10. Switch to Omni	1 sec	
11. Stow Antenna	10 min	
12. Shut down Gyros	1 sec	
B. First Mid-Course Correction------------------------------		Time 0 + 1-2 weeks
1. Switch on Gyros	1 sec	
2. Commands received from Earth, acknowledged and verified by spacecraft and stored in the Programmer	7 min	
3. Orientation of spacecraft by means of attitude control subsystem to required orientation	12 min	
4. Firing of Main Engine	30 sec	
C. Reorientation to the Sun----------------------------------		Immediately following engine firing
1. Commands read out by Programmer	10 sec	
2. Orientation to Sun and verification	6 min	
3. Orientation to Canopus and verification	6 min	
4. Sensor errors telemetered to Earth upon completion	5 min	
5. Shut down Gyros	1 sec	
D. High-Gain Antenna Deployment----------------------------		Time 0 + 120 days
1. Commands transmitted to spacecraft and verified	30 min	
2. High-Gain Antenna pointed to Earth using sensor corrected on programmed angles	10 min	
3. Switch to High-Gain Antenna	1 sec	
E. Terminal Guidance Observation (2 x 10^6 nm from Planet)----		Time 0 + 213 days
1. Commands transmitted to the spacecraft and verified	40 min	

TABLE 3.3-8. SEQUENCE OF EVENTS (Cont'd)

	Operation Sequence	Time to Complete Operation	Time to Begin Operation
	2. Body mounted I.O. camera turned on	5 min	
	3. TV pictures taken of planet and background		
F.	Final Trajectory Correction (1 x 10^6 nm from Planet)-------		Time 0 + 219 days
	1. Switch on Gyros	1 sec	
	2. Commands transmitted to spacecraft and verified	40 min	
	3. Switch to Omni	1 sec	
	4. Store High-Gain Antenna	10 min	
	5. Orientation of spacecraft by means of attitude control subsystem to required orientation	12 min	
	6. Firing of Main Engine	30 sec	
G.	Reorientation to Sun------------------------------------		Immediately after engine firing
	1. Commands read out by Programmer	10 sec	
	2. Orientation to Sun and verification	6 min	
	3. Orientation to Canopus and verification	6 min	
	4. Deploy High-Gain Antenna	10 min	
	5. Switch to High-Gain Antenna	1 sec	
	6. Sensor angles telemetered to Earth	10 min	
	7. Shut down Gyros	1 sec	
	8. TV pictures taken of planet and background		
H.	Orbiter Orientation and Injection into Orbit----------------		50 minutes before retro engine firing
	1. Switch on Gyros		
	2. Commands transmitted to spacecraft and verified		
	3. Switch to Omni Antenna	1 sec	
	4. Stow High-Gain Antenna	10 min	

TABLE 3.3-8. SEQUENCE OF EVENTS (Cont'd)

	Operation Sequence	Time to Complete Operation	Time to Begin Operation
	5. Orientation of spacecraft to required orientation	10 min	
	6. Firing of Main Engine	10 min	
I.	Orbiter in Orbit		Time 0 + 225 days
	1. Commands read out by Programmer	10 sec	
	2. Orientation to Sun and verification	4 min	
	3. Orientation to Canopus and verification	6 min	
	4. Deploy High-Gain Antenna	10 min	
	5. Switch to High-Gain Antenna	1 sec	
	6. Sensor angles telemetered to Earth	15 min	
	7. Deploy PHP	10 min	
	8. Deploy any other instrumentation required for the specific mission	10 min	
	9. Shut off Gyros	1 sec	

J. THERMAL CONTROL

Payload scientific instruments and electronic components can either be mounted on those spacecraft panels which always lie parallel to the sun's rays, or on the shadowed face normal to the roll axis. None of the heat rejection capability of these surfaces is impaired by Mars albedo and planetary fluxes. The logical location of payload components will therefore depend on which of the two factors, solar cells or rocket skirt heating, produces the least detrimental effect on surface heat rejection.

Thermal inputs to side panels from solar paddles are due to infrared radiation, and solar reflection. Radiation can be minimized by placing the high power density components as far away from paddles as feasible, and designing a louver arrangement which will have the maximum view to black space. The solar reflection onto the panels should be negligible, assuming that the state of the art of P on N cells will advance to the point where their surfaces will reflect in a purely specular manner. If this improvement cannot be achieved (present silicon solar cells do reflect diffusely approximately 10 percent of normal incident solar energy), serious consideration should be given to locating the payload components on the aft face of the spacecraft. The possible damage to louver surface finishes caused by prolonged exposure to rocket skirt heating must then be assessed.

The basic temperature variations on tankage resulting from the continuously changing vehicle to Sun distance cannot be avoided, unless the tanks are largely isolated from indirect solar effects by means of radiation insulation barriers placed on the back surface of the body mounted solar cells. In order to minimize heater power necessary to maintain tanks above freezing levels, heat will be allowed to flow into internal portions of the spacecraft from the back of the cells; this method will permit an important reduction in solar cell temperatures throughout the entire mission, leading to more favorable electrical power outputs than if the back of the cells were for all purposes adiabatic.

Most beneficial thermal control of both cells and tanks can be achieved by placing a light weight cover across the aft section of the vehicle. This cover, acting as a radiation shield, prevents large temperatures differences from developing within the tanks, and reduces the heater power required to maintain their liquids within their temperature limits (see Voyager Study Report, Document No. 63SD801). Tanks will be individually enveloped in light weight, multilayer insulation.

Preliminary calculations have been carried out to indicate approximate emittance values which will maintain tanks below maximum tolerable temperatures:

	Emittances	
	Internal Surface	External Surface
Solar Cells (mounted)	0.8	Fixed
Side Panels	0.3	0.3
Aft Cover	0.3	0.15

With such coating properties, solar cell temperatures would be on the order of 210^{o}F at Earth depature, and 112^{o}F at Mars orbit. Approximate tank heater requirements at Mars arrival are listed on the following page.

	Watts	Insulation Weight (pounds)
He	8	1.0
Fr or N_2	3	0.5
Fuel or Oxidizer	6	3.75
Total	26	9.5

Insulation thicknesses are chosen to obtain realistic power figures; an increase in such thicknesses would decrease power needs.

The location of the PHP during transit will allow continuous solar impingement on one face, and re-radiated energy from solar cells to strike those faces parallel to the sun's rays. As the PHP will have portions of its periphery insulated, others covered with louvers, mainly as a result of solar flux histories in orbit as well as component duty cycles, strip heaters will need to supply energy during part of the transit time. Continuous heat leaks are created by exposed camera lenses and other apertures, which cannot be solely compensated by inputs from the Sun and solar cells, at least not to the extent necessary to maintain internal PHP equipment above $0^{o}F$.

When the PHP is deployed and its components are activated, an active temperature control system, consisting of louvers, will be able to vary the skin emittance and maintain internal equipment within the specified temperature range. The parallelpiped configuration not only facilitates the mounting of components to obtain maximum thermal conductance from base plates to PHP skin, but also eases the attachment of anticipated temperature control mechanics.

K. STRUCTURAL DESIGN

The overall structural design philosophy for the Orbiter is predicated by the goal to design an optimum, high strength, low weight, state of the art structure which offers mission flexibility and high reliability through simplicity of design. Since the dynamic environment during powered flight is limiting in most cases, a structure which keeps dynamic amplifications to a minimum through stiffness and superior damping characteristics is mandatory. It has been proven experimentally that semi-monocoque shear structure provides a greater degree of damping through its basic construction than bending or truss-type structure. For example, a damping factor of 13 percent (as against 6 percent for truss type) was determined by test for the Advent Communications Satellite which utilized similar construction methods to those presented here.

The structural arrangement consists of a flat panel box structure which offers the advantages of semi-monocoque construction without compromising ease of thermal control, packaging of components and manufacture of structure.

For actual sizing of members, a natural frequency criterion must be introduced with the expected load levels. Modal coupling of masses must be avoided to help keep dynamic amplifications low. For example, calculating a system resonance of 60 cps results in designing the rocket engine support truss to a natural frequency criteria (approximately 125 cps) rather than to actual load levels during powered flight. This method gives rather conservative results compared to a pure load level type of design.

A deflection criteria was also used in the design to ascertain that shroud-spacecraft interferences under lateral loads were avoided. In this type of structure, bending deflections are negligible, whereas shear deformations are significant. A deflection analysis by the matrix force method showed that in one lateral direction the structure did not deflect excessively under the applied loads; however, in a perpendicular direction, additional stiffening members (in the form of shear bulkheads) had to be added to keep deflections at an acceptable level. The final design deflected 0.56 inches/g side load. Data obtained from the launch vehicle manufacturer indicated that clearance required for the payload was 3.50 inches per side from the outside contour. Earlier data, however, had indicated 2.50 inches required from the outside contour. With an utlimate lateral loading of 2.5 g's, the required clearance equated to $(0.56 \times 2.5) + 3.5$ inches $= 1.4 + 3.5 = 4.9$ inches/side. Therefore, for all future calculations, 5.0 inches per side or 10 inches per diameter was used to obtain either required outside shroud diameter or maximum allowable spacecraft dimensions.

1. Titan III-Voyager Interface

The attachment locations provided on the Titan III are shown in Figure 3.3-5. Eight tension-compression fittings and 24 shear pins are utilized to effectively distribute loads into the booster structure. To allow clearance for the main rocket engine, an adapter, as shown in Figure 3.3-18 must be utilized. This adapter consists of eight main longerons which transmit axial and overturning couple loads into the booster attachment fittings. A ring at the base is used to distribute side load shears to the 24 shear pins. A ring at the top of the adapter helps distribute the orbiter side loads in a uniform manner into the adapter skin. The adapter is slightly tapered since the orbiter attachment points must be at a smaller diameter than the booster attachment points in order to provide an adequate packaging envelope for the solar panels.

An all magnesium construction is recommended for the adapter since the shell longerons and skin are designed by stability considerations. Incompatibility stresses from thermal expansions will not be a significant factor during launch. The shell proper can be of rib-stiffened sheet construction; however, a honeycomb construction may offer some slight weight advantage.

2. Load Paths

The primary load carrying structure of the Orbiter consists of two full depth transverse beams which transfer the inertia loads of the high mass items (predominantly the fuel and oxidizer tanks and the Planet Horizontal Package) to four of the eight attachment points provided by the adapter.

These internal beams are constructed of aluminum honeycomb sandwich. The core provides stabilization to the face sheets and allows the sheet to be stressed to its yield point in shear. Bulkheads which frame the tanks, are used where necessary to provide a load path for lateral loads normal to the main interior beams. These bulkheads transfer lateral loads into the top and bottom panels as shear flows. They also help distribute part of the internal tank loads into the four longerons not common to the main beams.

The fuel and oxidizer tanks are supported through trunnion fittings on the sandwich beams. This is a rather efficient method of tank support, since only a local build-up of tank material is required at the trunnions. Applied concentrated thrust loads and bending moments decay quite rapidly, and the predominant stresses in the tank proper are the membrane stresses induced by internal pressure. By providing lateral thrust capability at one trunnion only, the tank is free to grow under pressure, and no discontinuity stresses are induced in the tank as would be the case with girth ring or multiple tension strap support.

One disadvantage to trunnion support is that the tank loads are introduced into the support structure as concentrated loads. These loads must be transferred into the shear beams through full length stiffeners of sufficient stiffness to distribute the loads in a uniform manner so that the thin gage honeycomb face sheets are not locally overstressed. The advantages of packaging ease and tank structure weight more than offset the local buildup required in the shear beams. Additionally, many optimization studies performed at GE previously for a variety of spacecraft configurations have shown that the higher strength/weight ratios afforded by honeycomb as compared to sheet stiffener construction more than compensate for the extra buildup required at loading locations. The use of sandwich construction is further enhanced by the availability of chemical milling processes for reducing high strength sheet from minimum mill stock gages to foil thickness.

With the exception of the rocket engine support truss, aluminum is used throughout the Orbiter spacecraft structure. The choice of aluminum is an obvious one, since higher strength/weight ratios are obtained for the fully stabilized structure than with the lighter alloys. To preclude the development of thermal stresses under orbital heat flux, mating of dissimilar metals has been avoided.

In keeping with simplicity of design, the side honeycomb panels are designed flat, resulting in an octagonally shaped rather than conic shaped structure. Flat panels, besides having the obvious advantages of ease and cost of manufacture, are desirable for component mounting and thermal control. From a load path point of view, the outer panels form the basic shell for transfer of the loads from the additional mass itesm to the hard points. This arrangement of mounting on the exterior panels and the inclusion of the interior bulkheads tends to load all eight hard points uniformly.

The top panel serves a dual purpose, that of a solar cell mounting panel, and also as a bulkhead for transmitting side loads to the longerons and side panels. The deployable solar paddles are also constructed of solid honeycomb panels and are preloaded in the stowed position to provide the capability for tensile and compressive normal loading and shear and moment capability.

The Planet Horizontal Package mounts in the stowed position on four fittings which protrude through the top solar cell panel. Loads are distributed through stiffeners into the main transverse shear beams in a uniform manner.

In the deployed position, the Planet Horizontal Package boom mounts on an extended solar paddle. Analysis has shown that extremely low loads are induced at the hinge fitting during orbital maneuvers; however, to positively maintain deflections at a minimum, this panel is reinforced by a frame of sheet metal channel sections.

The rocket engine support is basically a tubular truss structure. Truss construction is chosen here because only a limited number of tie-down points which have load reaction capability in any direction are available. Titanium is chosen as the truss tube material in consideration of the high temperature environment (700^{o}F) prevalent during rocket engine firing. At this temperature the reduction in modulus of elasticity of Titanium is only 22 percent, resulting in a more efficient design for column buckling than for aluminum.

L. DEPLOYABLE DEVICES

1. High-Gain Antenna

The high-gain antenna is attached to the Orbiter structure during launch at three points (120 degrees apart). Two of these three attachments are explosively actuated and are fired shortly after separation of the spacecraft from the launch vehicle. The third point is the hinge about which the antenna rotates. A compression support is provided on the lower portion of the spacecraft in order to provide support during main engine firing. The antenna is normally stowed during midcourse maneuvers and retro firing.

A folded feed deployed by a spring-latch assembly is provided on the antenna and is actuated by antenna motion. This feed locks permanently in a fixed position after the first antenna deployment.

The high-gain antenna is stored during the launch and insertion into orbit. Immediately after separation from the booster, the antenna is deployed, used to verify the Sun/Canopus orientation, and is then stowed until the 120th day of transit. As shown in Figure 3.3-21, the true Earth-Spacecraft-Sun angle is 30^{o} at 120 days. The Earth-Spacecraft-Sun distance at 120 days is then 26 million n. mi. which is well within the communication capability of the omni-antennas which are used up to 120-day transit time. (Ref. Figure 3.3-22).

During the orbiting period, the Earth-Mars-Sun angle changes from about 44 degrees to about 34 degrees. This averages to about 0.1 degree/day antenna motion.

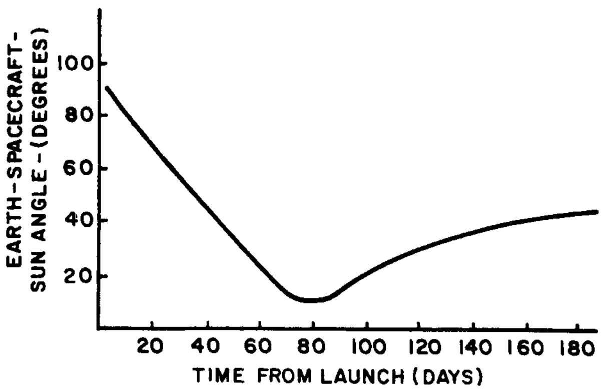


Figure 3.3-21. Earth-Spacecraft-Sun Angle vs. Time From Launch

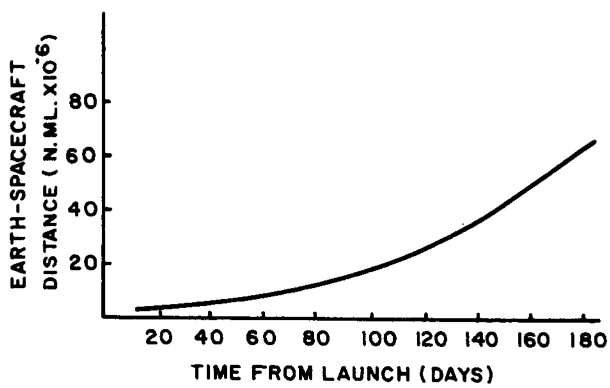


Figure 3.3-22. Earth-Spacecraft Distance vs. Time From Launch

2. Planet Horizontal Package (PHP)

The PHP is rigidly attached to the spacecraft during launch, transit and orbit insertion. Attachment to the spacecraft is by means of four explosive actuators. These attachments provide the capability of taking tension, compression and shear loads into the spacecraft structure.

The PHP and the two shear beams supporting the propellant tanks have been sized so that attachment points are readily accessible.

Since the PHP is not deployed until after orbit insertion, there is a potential problem with the explosive actuators. These actuators must be in a controlled environment at the time of firing. The Orbiter with the PHP is so designed that the explosive actuators will be located within the body of the Orbiter and in a controlled environment.

After insertion into orbit the actuators are fired and the PHP begins to deploy. In order to package the structure for launch within the confines of the extended standard shroud, several extra joints had to be added. During the PHP deployment three hinges are rotated and permanently locked in place. One hinge is at the base of the solar panel and the remaining two hinges are located on the boom mechanism.

After deployment, the PHP is free to rotate 360 degrees at the end of the boom. The boom can be positioned such that it will be perpendicular to the orbit plane. In this manner only rotation of the instruments is needed.

3. Magnetometer Boom and Radio-Propagation Experiment

The magnetometer is mounted at the end of a sixteen-foot boom. This boom is hinged at the base and the center in order to package within the confines of the shroud. During launch, transit, and orbit insertion the boom and magnetometer will be fixed to the base structure of the spacecraft as shown in Figure 3.3-18.

After orbit insertion an explosive actuator releases the boom and a spring deploys it to its operating position. Both hinges are locked in a fixed position during this deployment operation.

Provisions have been made to orient the magnetometer for the calibration requirement. This requirement called for physical polarity reversal of each of the three sensors daily without using magnetic field producing devices.

The boom length provided is the maximum capable on this spacecraft without adding another hinge joint. However, past studies have shown that distances in the order of 13 feet or more are satisfactory.

A radio-propagation experiment antenna (bi-static radar) is attached to the magnetometer boom and deployed when the magnetometer boom is deployed and locked in place. The radio-propagation experiment antenna consists of 10-foot and three-foot elements which are spring loaded to the magnetometer boom.

M. SOLAR PANELS

With a total power requirement of 592 watts at load and a maximum Orbiter diameter of 110 inches, the need for deployable solar panels becomes evident. The Orbiter has 135 square feet of deployable solar cell area. This area is divided into seven panels fixed to the Orbiter and six panels fixed to the deployable panels. These secondary panels are needed to fill the voids between the rectangular shaped panels which are attached to the Orbiter structure.

The primary panels are fixed to the structure by a hinge at the base of the Orbiter. For the launch configuration the panels are attached at the outboard end to the Orbiter structure by means of explosive actuators. After booster separation the explosive actuators fire, and the primary panels are deployed and locked in place. At the same time the secondary panels are moved to their correct location by means of spring actuators.

One solar panel is not deployed until after orbit insertion. This is the panel attached to the PHP boom support.

3.4 ORBITER / LANDER SPACECRAFT SYSTEM

3.4.1 SYSTEM CONFIGURATION STUDY AND ANALYSIS

The requirement for an Orbiter/Lander design was evident in order to make system trade-offs between the three systems, All Orbiter, Bus/Lander and Orbiter/Lander. The spacecraft was to mount on the Titan IIIC and be packaged within an extended standard shroud.

Power requirements for this Orbiter were 328 watts. This power was to be supplied by means of solar panels.

The Orbiter mission is TV mapping, requiring orientation of an instrument package to the planet.

Relay capability, from Lander to Orbiter to Earth, is required. This relay capability will operate during Lander descent and during the orbiting period. The orbit will be 1,000 x 19,000 n.mi.

The Lander will be designed to enter the planet atmosphere, impact, and survive for a six-month period.

3.4.2 LANDER CONFIGURATION DESIGN

A. LANDER DESIGN

The Entry/Lander system which has been designed for use with an Orbiter is identical aeromechanically with the Entry/Lander used with the spacebus configuration hence, much of what was presented in Section 3.2.2 (d) applies here. Since the ballistic parameter is the same (15 psf), trajectory characteristics will be the same. The Orbiter/ Lander, however, has a base diameter of 106 inches and an entry weight of 1137 pounds.

Significant differences in the vehicle subsystems lie mainly in communications. This Lander is equipped with both a relay and direct link. The additional relay link is a VHF, 100 mc system using a transmission line antenna during entry and descent and a 5 foot cross dipole antenna for surface operations. The direct system acts as a backup mode during the life of the Orbiter.

Power requirements will be lessened because of the smaller payload capability of this vehicle although, as before, an RTG will be used in conjunction with a rechargeable

Nickel Cadmium battery. RTG power output for this application is estimated at 110 watts. The payload included on the Lander can be found on Table 3.4-1.

Because of the change in total vehicle weight, differences in the ΔV rocket and spin and separation system will be reflected only as a change in those components' weight. Operation and requirements of the propulsive items are the same as for the Bus/Lander vehicle.

An inboard profile layout of the vehicle has been generated for design and packaging studies and is shown in Figure 3.4-1. The parachutes (item 10 and 11) and cannister assembly have been located outside of the aft cover to facilitate packaging of the helix array antenna (29). The transmission line relay antenna (35) has replaced the cross dipole direct antenna which was located at the center of the aft cover. As before, selected atmospheric and soil experiments are grouped in a cluster on the aft cover, readily accessible to the environment once the protective cover is removed.

The five-foot cross dipole antenna (62) used for surface relay communications is shown as a folding umbrella which is extended after the aft cover opens. The antenna is raised sufficiently far enough above the vehicle surface to produce a good transmission pattern.

As before, the aft cover is notched to allow it to open fully during surface orientation. The notch is covered with a frangible section which protects the vehicle interior from entry heat.

B. WEIGHT STATEMENT

A detailed weight estimation was conducted for a Lander which is used in conjunction with a Mars Orbiter. This is shown in Table 3.4-1.

This is a solid flare vehicle with D_B = 106 inches, W/C_DA = 15 psf and W_{entry} = 1137 pounds. The gross weight including adapter, radiator, spin and separation and ΔV rocket is 1284 pounds.

This configuration has a gross payload capability of 505.2 pounds.

KEY FOR ORBITER/LANDER, FIGURE 3.4-1

1 - Heat Shield
2 - Radar Altimeter Antenna
3 - Crush-Up (Fiberglass Honeycomb)
4 - Structural Shell
5 - Delta "V" Rocket
6 - Adapter Section (Transient Radiator)
7 - Spin & Separation System
8 - Crush-Up
9 - Cover
10 - Parachute Package (Main)
11 - Parachute Package (Decelerator)
12 - VHF Diplexer
13 - Power Supply
14 - Battery
15 - VHF Transmitter
16 - R. F. Switch
17 - Tape Recorder
18 - Tape Recorder
19 - VHF Transmitter
20 - Temperature Control
21 - Power Supply
22 - Command & Computer Equipment
23 - Power Supply
24 - Power Amplifier
25 - Power Supply
26 - Power Amplifier
27 - Electron Density (Langmuir Probe)
28 - Isolator & Load
29 - Helix Array Antenna
30 - Power Supply
31 - Diplexer
32 - Wind Speed & Direction
33 - Seismic Activity
34 - Precipitation
35 - Transmission Line Antenna
36 - R. T. G. Unit
37 - TV Camera (Panorama)
38 - Light Intensity (Sun Sensor)
39 - Gas Reservoir
40 - Shelf
41 - Motor & Pumps
42 - Evaporative Heat Exchanger
43 - Modulation Valve
44 - Modulation Valve
45 - Liquid Heat Exchanger
46 - Coolant Reservoir
47 - Water
48 - Buffer Unit
49 - Transponder
50 - Electronics for Electron Density
51 - Photoautotroph
52 - Altimeter Electronics
53 - Radar Altimeter Electronics
54 - Turbidity
55 - Surface Gravity
56 - Data Processing Unit
57 - Power Conversion & Control
58 - VHF Receiver
59 - Command Detector
60 - Tape Recorder
61 - Shelf
62 - VHF Turnstile
63 - Surface Penetration Hardness
64 - Composition H_2O
65 - Composition O_2
66 - Composition N_2
67 - Composition CO_2
68 - Composition O_3
69 - Composition A
70 - Gas Chromatograph
71 - Density
72 - Pressure
73 - Temperature
74 - Sounds
75 - Electronics for Sounds
76 - Multiple Chamber
77 - Soil Moisture
78 - Radioisotope
79 - TV Microscope & Sub-Surface Group
80 - Electronics for Radioisotope
81 - Encapsulated Turnstile Antenna

TABLE 3.4-1. VOYAGER-ORBITER/LANDER-LANDER WEIGHT ESTIMATE

COMPONENT	Weight (lb)	C.G. Sta. (Inches)
Shield	(103.5)	24.5
Structure	(113.2)	26.1
Honeycomb Sandwich	65.5	26.4
Ring - Fwd	2.1	12.0
Ring - Mid	5.2	23.6
Ring - Aft	10.4	35.4
Cruciform	22.0	23.5
Fasteners	8.0	24.0
Aft Cover	(83.1)	44.9
Shield	22.6	45.5
Crush Up	6.1	45.0
Skin	41.4	44.7
Hinges & Fittings	10.0	44.7
Fasteners	3.0	44.7
Retardation	(312.0)	33.3
Impact Crush Up	190.0	19.6
Decel. Chute	16.0	54.5
Main Chute	43.0	54.5
Bags, Risers, Etc.	10.0	54.5
Chute Housing	9.0	54.5
Prog., Batt., Switch, Mortar	9.0	28.0
Retrorocket	27.0	67.5
Fasteners	3.5	48.0
Harnesses	4.5	36.0
Scientific Payload	(109.8)	38.0
Instrumentation		
Temperature	0.3	48.5
Sounds	0.5	48.5
Pressure	0.3	48.0
Density	1.5	48.0
Multiple Chamber	4.0	48.0
Surface Penetration Hardness	4.5	37.0
Photoautotroph	3.0	51.0
Light Intensity (Sun Sensor)	0.5	33.0
Composition - H_2O	1.5	48.0
Composition - O_2	1.5	48.0
Turbidity & pH	4.0	31.9

TABLE 3.4-1. VOYAGER-ORBITER/LANDER-LANDER WEIGHT ESTIMATE (Cont'd.)

COMPONENT	Weight (lb)	C.G. Sta. (Inches)
Scientific Payload (Cont'd.)		
Instrumentation (Cont'd.)		
Wind Speed & Direction	2.0	30.0
Gas Chromatograph	7.0	48.0
Composition - N_2	1.0	48.0
Composition - CO_2	1.0	48.0
Soil Moisture	2.0	48.0
TV Camera, Panoramic	10.0	32.0
Radioisotope	6.0	48.0
Composition - O_3	1.5	48.0
Composition - A	1.5	48.0
Precipitation	1.0	29.5
Electron Density	3.0	36.5
Surface Gravity	3.0	31.9
Surface Roughness Altimeter	15.0	31.9
Deployment & Installation		
TV	10.0	32.0
Surface Hardness	3.0	36.0
Brackets & Fasteners	6.2	36.0
Harnesses	15.0	36.0
Thermal Control	(110.5)	27.1
Modulation Valves	6.0	22.4
Reservoir	3.0	45.5
Temperature Controller	2.0	20.0
Motor Pumps (2)	6.0	14.9
Separation Valves	6.0	35.0
Shutoff & Relief Valves (4)	6.0	35.0
Heat Exchanger (liq. - liq.)	7.0	21.2
Heat Exchanger (Evaporative)	7.0	19.3
H_2O & Tank	22.0	29.6
Insulation	10.0	32.0
Tubing	3.5	32.0
Coolant	5.0	32.0
Octadecane Wax & Enclosure	20.0	21.4
Brackets, Fittings & Fasteners	2.5	32.0
Harnesses	4.5	32.0

TABLE 3.4-1. VOYAGER-ORBITER/LANDER-LANDER WEIGHT ESTIMATE (Cont'd.)

COMPONENT	Weight (lb)	C.G. Sta. (Inches)
Electrical	(100.7)	32.6
RTG	58.0	37.4
Battery	16.2	21.4
Battery Regulator	3.0	32.0
Power Controller	8.5	30.0
IFD	1.5	36.0
Brackets & Fasteners	7.0	28.0
Harnesses	6.5	28.0
Ground Orientation	(20.0)	39.7
Clamshell Actuating Mech.	9.0	39.5
Accelerometer & Controls	2.0	35.0
Harpoons	3.0	48.0
Fasteners	3.0	39.5
Harness	3.0	35.0
Communications	(184.2)	28.5
Deep Space Transmission		
Diplexers (2)	2.0	31.8
Helix Array Antenna	10.0	41.0
Encapsulated Turnstile Antenna	5.0	40.0
Transponders (2)	10.8	29.4
Power Amplifiers - 24W (2)	6.0	31.6
Power Supplies (2)	12.0	31.4
Power Amplifier - 140W	4.0	25.5
Power Supply	8.0	26.5
Driver Amp. & Pwr. Supply - 5W	4.0	26.0
Command Detectors (2)	6.0	32.2
RF Switch	1.0	13.5
Isolator & Load (2)	1.5	25.5
Data Handling		
Data Processing Unit	16.0	21.7
Buffer Unit	4.0	20.4
Tape Recorders (2)	16.0	19.0
Command		
Command & Computer Equip.	14.0	25.8
Power Conversion & Control	7.0	26.4

TABLE 3.4-1. VOYAGER-ORBITER/LANDER-LANDER WEIGHT ESTIMATE (Cont'd.)

COMPONENT	Weight (lb)	C.G. Sta. (Inches)
Communications (Cont'd.)		
Relay Transmission		
VHF Antenna	10.0	42.0
VHF Diplexer	1.0	23.3
VHF Transmitter - 25W	1.3	13.7
VHF Transmitter - 5W	0.6	22.9
VHF Receiver	2.0	22.3
Command Detector	3.0	32.2
Antenna Controls		
Switch	2.0	26.5
Sun Sensor	2.3	28.0
Electronic Gimbal Control	1.4	27.0
Amplifier	1.4	18.0
Drive Motors	4.0	35.0
Mode Control Electronics	0.5	30.0
Vertical Switch	2.0	34.0
Antenna Deployment Mech.	6.0	35.5
Brackets & Fasteners	9.4	26.5
Harnesses	10.0	26.5
Total Lander (Entry Condition)	(1,137.0)	31.8
Adapter		
Structure	(27.0)	42.8
Skin	7.5	43.0
Longerons	1.2	43.0
Stiffeners	1.4	43.0
Ring - Fwd.	6.8	36.2
Ring - Aft	6.8	49.5
Explosive Bolts (4)	0.4	36.0
Fasteners	2.9	43.0
Thermal Control	(22.0)	42.0
Skin	5.0	42.0
Insulation	3.0	42.0
Spacers	3.1	42.0
Tubing	3.0	42.0
Fitting & Connectors	1.9	42.0
Coolant	3.5	42.0
Fasteners	2.5	42.0

TABLE 3.4-1. VOYAGER-ORBITER/LANDER-LANDER WEIGHT ESTIMATE (Cont'd)

COMPONENT	Weight (lb)	C.G. Sta. (Inches)
Adapter (Cont'd.)		
Spin & Separation	(32.0)	45.1
Tanks	16.6	45.5
N_2	7.5	45.5
Squib Valves	1.5	45.5
Tubing	1.6	38.0
Nozzles & Fittings	0.8	48.0
Support Struct. & Fasteners	3.0	45.5
Harness	1.0	42.0
ΔV Rocket Installation	(66.0)	66.4
Rockets	60.0	66.8
Support Struct. & Fasteners	6.0	62.0
Total Adapter	(147.0)	53.7
Total Lander (Gross)	(1,284.0)	34.3

3.4.3 ORBITER CONFIGURATION DESIGN

A. CONFIGURATION STUDY AND SOLUTION

The major limitation on the Orbiter design was the requirement for packaging within an extended standard shroud. This required a maximum spacecraft dimension of 110 inches in any direction except the roll axis (launch vehicle thrust axis). As shown by Figure 3.4-2, this packaging problem required the PHP to be mounted within the Lander adapter. In addition, the high-gain antenna had to be located in a position below the main engine during launch. Packaging the PHP within the Lander adapter and the high-gain antenna below the main engine allowed the total configuration to be packaged within a standard shroud extension of 67.5 inches.

The Orbiter is designed in the same manner as previously described in Section 3.3, a semi-monocoque structure with longerons for point loads and sandwich panels for shear capacity. Because of the lessened requirement for propellant storage, due to the greater eccentricity of this orbit, and the reduced payload, the total depth of the Orbiter is reduced from that needed for the All Orbiter.

The solar panels are designed to extend a minimum distance from the Orbiter structure. Reduction of deployed length of solar panels reduces the required length of the PHP boom. Deployment of all panels except that panel attached to the PHP support boom occurs immediately after separation from the launch vehicle. The solar cells which are mounted on the upper surface of the Orbiter do not produce power during transit. These cells are hidden from the sun by the Lander, Lander adapter, and the PHP. After orbit is obtained and the PHP is deployed, these cells begin generating power.

The adapter between the launch vehicle and the spacecraft is manufactured in two sections. Whereas the separation plane for the All-Orbiter design (Section 3.3) was at the base of the Orbiter structure, the separation plane is now 27 inches below the base of the Orbiter. The revised design is necessary since the high-gain antenna is now packaged below the main engine nozzle.

Because available volume in the Orbiter is taken by tankage and payload, an adapter was added in order to raise the Lander and the ΔV rocket motor above the upper surface of the Orbiter. The PHP is located within this adapter. The separation plane for the Lander is at the intersection of the adapter and the Lander radiator.

1. Sequence of Events

The sequence of events for the Orbiter/Lander combination is shown in Table 3.4-2.

2. Block Diagram

Figure 3.4-3 shows the block diagram for the Orbiter of the Orbiter/Lander combination.

3. System Weights

A detailed weight statement is shown in Table 3.4-3 for the Orbiter for Mars 1971. In addition, general subsystem weight statements are shown for 1973, 1975, and 1977 (Tables 3.4-4 through 3.4-6).

TABLE 3.4-2. SEQUENCE OF EVENTS FOR ORBITER/LANDER

The basic assumption is made that a successful ascent and injection into transit trajectory, with successful separation from the launch vehicle adapter, will have been completed.

Launch Date: 6 May - 5 June 1971

Operation Sequence	Time to Complete Operation	Time to Began Operation
A. Entry Into Transit Mode		Time 0 (Immediately after separation from launch vehicle)
1. Turn on transponder	1 min	
2. Establish round trip phase lock	5 min	
3. Turn on attitude control subsystem	1 sec	
4. Deploy Solar Panels	10 min	
5. Orient to Sun	16 min	
6. Deploy High-Gain Antenna and preprogram to point in Earth direction	10 min	
7. Switch to High-Gain from Omni by means of Earth Communication	1 sec	
8. Orient to Canopus	48 min	
9. Earth verification of reference acquisition	60 min	
10. Switch to Omni	1 sec	
11. Stow Antenna	10 min	
12. Shut down Gyros	1 sec	
B. First Mid-Course Correction		Time 0 + 1-2 weeks
1. Switch on Gyros	1 sec	
2. Commands received from Earth, acknowledged and verified by spacecraft and stored in the Programmer	7 min	
3. Orientation of spacecraft by means of attitude control subsystem to required orientation	12 min	
4. Firing of Main Engine	30 sec	

TABLE 3.4-2. SEQUENCE OF EVENTS FOR ORBITER/LANDER (Cont'd.)

	Time to Complete Operation	Time to Begin Operation
C. Reorientation to the Sun		Immediately following engine firing
1. Commands read out by Programmer	10 sec	
2. Orientation to Sun and verification	6 min	
3. Orientation to Canopus and verification	6 min	
4. Sensor errors telemetered to Earth upon completion	5 min	
5. Shut down Gyros	1 sec	
D. High-Gain Antenna Deployment		Time 0 + 120 days
1. Commands transmitted to spacecraft and verified	30 min	
2. High-Gain Antenna pointed to Earth using sensor corrected on programmed angles	10 min	
3. Switch to High-Gain Antenna	1 sec	
E. Terminal Guidance Observation (2×10^6 nm from Planet)		Time 0 + 213 days
1. Commands transmitted to the spacecraft and verified	40 min	
2. Body mounted I. O. camera turned on	5 min	
3. TV pictures taken of planet and background		
F. Final Trajectory Correction (1×10^6 nm from Planet)		Time 0 + 219 days
1. Switch on Gyros	1 sec	
2. Commands transmitted to spacecraft and verified	40 min	
3. Switch to Omni	1 sec	
4. Store High-Gain Antenna	10 min	
5. Orientation of spacecraft by means of attitude control subsystem to required orientation	12 min	
6. Firing of Main Engine	30 sec	

TABLE 3.4-2. SEQUENCE OF EVENTS FOR ORBITER/LANDER (Cont'd.)

	Time to Complete Operation	Time to Begin Operation
G. Reorientation to Sun		Immediately after engine firing
1. Commands read out by Programmer	10 sec	
2. Orientation to Sun and verification	6 min	
3. Orientation to Canopus and verification	6 min	
4. Deploy High-Gain Antenna	10 min	
5. Switch to High-Gain Antenna	1 sec	
6. Sensor angles telemetered to Earth	10 min	
7. Shut down Gyros	1 sec	
8. TV pictures taken of planet and background		
H. Lander Ejection (150,000 nm from planet)		Time 0 + 224 days
1. Switch on Gyros	1 sec	
2. Commands transmitted to the spacecraft and verified	40 min	
3. Orientation of spacecraft to desired orientation	12 min	
4. Physical attachment of Lander to Orbiter broken	1 sec	
5. Lander is separated from Orbiter ΔV = 1 ft/sec	1 sec	
6. Lander spin up to 60 RPM	30 sec	
7. Lander ΔV rocket engine fires. Distance from Orbiter approximately 1000 ft.	15 sec	
I. Orbiter Reorientation to the Sun		Immediately following Lander ejection
1. Commands read out by Programmer	10 sec	
2. Orientation to Sun and verification	6 min	
3. Orientation to Canopus and verification	6 min	
4. Sensor angles telemetered to Earth	10 min	
J. Orbiter Orientation and Injection into Orbit		50 minutes before retro engine firing
1. Commands transmitted to spacecraft and verified		
2. Switch to Omni Antenna	1 sec	

TABLE 3.4-2. SEQUENCE OF EVENTS FOR ORBITER/LANDER (Cont'd.)

	Time to Complete Operation	Time to Begin Operation
3. Stow High-Gain Antenna	10 min	
4. Orientation of spacecraft to required orientation	10 min	
5. Firing of Main Engine	10 min	
K. Orbiter in Orbit -		Time 0 + 225 days
1. Commands read out by Programmer	10 sec	
2. Orientation to Sun and verification	4 min	
3. Orientation to Canopus and verification	6 min	
4. Deploy High-Gain Antenna	10 min	
5. Switch to High-Gain Antenna	1 sec	
6. Sensor angles telemetered to Earth	15 min	
7. Deply PHP	10 min	
8. Deploy any other instrumentation required for the specific mission	10 min	
9. Shut off Gyros	1 sec	

TABLE 3.4-3. ORBITER/LANDER - ORBITER WEIGHT ESTIMATE

Guidance and Control			(210.15)
Image Orthicon		22.00	
Optics	5.00		
Head	4.00		
Electronics	13.00		
Switching Amp.		1.00	
Gyro Control		1.10	
Auto Pilot		2.00	
Antenna Drive Electronics		2.00	
Actuator Hinge (Ant.)		7.50	
Actuator Elevation (Ant.)		4.00	
Logic, Storage and Relays		14.25	
Power Supply		20.00	
Earth Sensor		6.50	
Canopus Scanner (+ Pitch)		5.50	
Canopus Scanner (- Pitch)		5.50	
Horizon Scanner		13.00	
Gyro (Roll)		2.00	
Gyro (Yaw)		2.00	

TABLE 3.4-3. ORBITER/LANDER - ORBITER WEIGHT ESTIMATE (Cont'd.)

Gyro (Pitch)			2.00	
Accelerometer			3.00	
Sun Sensors (Fine & Coarse) (7)			.80	
Payload Compartments Structure			18.00	
PHP Drive Electronics			2.00	
Actuator			7.50	
Actuator			7.50	
Actuator Amplifier-Drive Motor Logic (Third Axis)			7.00	
Pneumatic System			54.00	
Regulators	(2)	6.20		
Solenoid Valves	(12)	5.20		
Latch Valve		1.80		
Filters	(2)	.80		
Check Valves	(2)	.10		
High Pressure Transducer	(2))	1.00		
Low Pressure Transducer	(2))			
Temperature Sensors	(4)	1.00		
Nozzles	(12)	1.20		
Tubing		2.80		
Shut-Off Valves		5.00		
Tankage		7.10		
Gas F 14		21.80		
Orbiter Power Supply				(163.62)
Secondary Battery			34.20	
Regulator (Power Control)			4.32	
Charge Control (Based on 14-lb/Kw)			7.56	
Diodes			1.00	
Inflight Disconnect (3)			4.50	
Harness (Solar Array)			6.02	
Solar Array			106.02	
Body Panel Str. .415 lb/sq. ft.		21.24		
Paddle Str.		24.57		
Cells (190.01 sq. ft. x .5724) =		60.21		
51.19 sq. ft. at 3.02 W/sq. ft.		154.59 watts		
54.00 sq. ft. at 3.21 W/sq. ft.		173.34 watts		
	=	327.93 watts		
Communications				(253.75)
Pre-Amplifier			4.00	
S-Band Diplexer (2)			2.00	
Omnidirectional Antenna (2)			4.00	
Transponder (2)			10.80	
Power Amplifier (3) (43W)			9.00	
Power Supply (3) (43W)			18.00	

TABLE 3.4-3. ORBITER/LANDER - ORBITER WEIGHT ESTIMATE (Cont'd.)

Power Amplifier (60W)				2.50	
Power Supply (60W)				4.50	
Command Demodulator (2)				6.00	
R. F. Switch				1.00	
Isolator and Load				1.50	
Data Processor				12.25	
Multicoder				10.00	
Buffer Unit				4.00	
Tape Recorder (3)				45.00	
Command Decoder				4.00	
Programmer Unit				20.00	
Power Conversion & Control (Orbiter)				12.00	
Power Conversion & Control (PHP)				2.00	
Coax Cabling				12.30	
High Gain Antenna				23.00	
Payload Compartments Package Structure				15.80	
VHF Antenna (Yagi)				16.00	
VHF Antenna (Turnstile)				5.00	
VHF Diplexer				1.00	
VHF Transmitter				.60	
VHF Receiver (2)				4.00	
Data Demodulator				3.50	
<u>Diagnostic Instrumentation</u>					(30.00)
<u>Payload</u>					(123.00)
Scientific				45.00	
I.R. Flux		I-2	3.00		
Magnetometer		I-23	5.00		
Micrometeroid Flux		I-55	2.50		
Mounting Provisions			5.50		
B.S. Radar & Antenna		I-85	13.00		
Charged Particle Flux		I-12	5.50		
Polarimeter		I-95	4.50		
Far UV Radiometer		I-96	3.00		
Visible Radiometer		I-79	3.00		
Television				78.00	
Image Orthicons	(3)		59.00		
Optics 140	(3)	3.00			
Camera Heads	(3)	12.00			
Electronics	(3)	39.00			
Mirror	(1)	2.00			
Misc. Controls		3.00			
Vidicons	(2)		19.00		
Optics	(2)	2.00			
Camera Heads	(2)	5.00			
Electronics	(2)	8.00			
Misc. Controls		4.00			

TABLE 3.4-3. ORBITER/LANDER - ORBITER WEIGHT ESTIMATE (Cont'd.)

Propulsion				(233.04)
Fuel and Oxidizer System (Dry)			197.92	
Tanks (2 at 25.25" Dia.)		46.62		
Residual		16.50		
Thrust Chamber		42.00		
Filters	(4)	.60		
Main Valves	(4)	10.00		
Fill & Purge Valves	(12)	6.00		
Filters & Orifices	(4)	2.50		
Latch Valves	(4)	4.50		
Transducers	(8)	3.20		
Shielding		4.00		
Harness		3.00		
Lines		4.00		
Brackets		5.00		
Gimbal System		50.00		
Includes: Hwd. Power Pk. Actuators Servo Valves Accumulator Plumbing Oil, Ring, Bearings				
Pressurization System			35.12	
Tank		17.81		
Gas He		1.90		
Tubing and Connectors		10.00		
Clamps		.10		
Regulators		3.25		
Filter		.25		
Relief Valve		.31		
Squib Valve Norm. Open (2)		.75		
Squib Valve Norm. Close (2)		.75		
Thermal Control				(54.75)
Insulation: Orbiter			8.00	
Active Control Orbiter			10.00	
Biological Barrier			6.00	
PHP Insulation			6.00	
PHP Active Control			7.75	
Timers			1.00	
Paint			5.00	
Grease			2.00	
Heaters (at .1 lb each)			5.00	
Misc.			4.00	
Vehicle Harnessing				(50.00)

TABLE 3.4-3. ORBITER/LANDER - ORBITER WEIGHT ESTIMATE (Cont'd.)

<u>Structure</u>			(321.42)
Orbiter Body		193.92	
Honeycomb Sides	20.88		
Top Honeycomb Panel	16.10		
Edging Members	11.88		
Tank Supports	13.24		
Honeycomb Bulkheads (Main)	22.06		
Honeycomb Bulkheads (Secondary)	12.99		
Tie Down Fittings	1.40		
Corner Gussets	1.20		
G.S.E. Fittings	3.00		
Misc. Support Bhd. for Engine Support	2.00		
PHP (Stowed Position) Support Str.	5.28		
Antenna Support Str.	4.86		
Magnetometer Boom & Support	2.50		
PHP Support Hinge	4.57		
I.O. Camera Support Str.	2.00		
Misc. Supports & Brackets	2.21		
Boost Extrusions	16.92		
PHP Boom and Support	22.50		
PHP Hardware	3.12		
Ant. Support (Stowed - Mid-Course)	3.00		
Hardware Including Solar Array	22.21		
<u>PHP</u>		44.00	
Framing L's	4.00		
Honeycomb Component Mounting	17.00		
Fittings	3.00		
Clips, Gussets, Supports	10.00		
Honeycomb Sides	6.00		
Hardware	4.00		
Lander to Orbiter Adapter		83.50	
Total Orbiter			1439.73
Fuel - Orbit Insertion (.475)			683.87
Lander			1284.40
Mid-Course Fuel			36.00
Adapter and Δ Shroud			156.00
Launch Weight			3600.00

(Orbit (n.mi.) 1000 x 19,000)

TABLE 3-4-4. ORBITER/LANDER SYSTEM WEIGHTS - MARS - 1973

Guidance and Control	210		
Power Supply	167		
Communications	254		
Diagnostic Instrumentation	30		
Payload	123		
Propulsion	228		
Thermal Control	55		
Vehicle Harnessing	50		
Structure	321		
Total Orbiter		1440	
Orbit-Insertion Fuel (Factor .415)		596	
Lander		622	
Mid-Course Fuel		36	
Adapter and Δ Shroud		156	
Total Launch Weight			2850
Orbit N. Mi. 1000 x 19,000 n.mi.			

TABLE 3.4-5. ORBITER/LANDER SYSTEM WEIGHTS - MARS - 1975

Guidance and Control	210		
Power Supply	165		
Communications	254		
Diagnostic Instrumentation	30		
Payload	123		
Propulsion	234		
Thermal Control	55		
Vehicle Harnessing	50		
Structure	321		
Total Orbiter		1442	
Orbit-Insertion Fuel (Factor .622)		898	
Lander		568	
Mid-Course Fuel		36	
Adapter and Δ Shroud		156	
Total Launch Weight			3100
Orbit N. Mi. 1000 x 19,000 n.mi.			

TABLE 3.4-6. ORBITER/LANDER SYSTEM WEIGHTS - MARS - 1977

Guidance & Control	210		
Power Supply	165		
Communication	254		
Diagnostic Instrumentation	30		
Payload	123		
Propulsion	234		
Thermal Control	55		
Vehicle Harnessing	50		
Structure	321		
Total Orbiter		1442	
Orbit-Insertion Fuel (Factor .38)		548	
Lander		1018	
Mid-Course Fuel		36	
Adapter and Δ Shroud		156	
Total Launch Weight			3200

Orbit N. Mi 1000 x 19,000 n.mi.

Table 3.4-7 is shown in order to indicate what Lander weight may be available for later years. It is apparent that 1971 is an extremely good year for a Mars trip. On this basis, an Orbiter/Lander combination appears to be an excellent choice. However, observation of the years 1973, 1975, and 1977 indicates a rapidly declining capability, thus making the Orbiter/Lander combination a much less likely selection.

TABLE 3.4-7. ORBITER/LANDER COMPARISON

Year	1971	1973	1975	1977
Total Orbiter	1439.73	1437.79	1442.74	1443.00
Orbit - Insertion Fuel	683.87	596.68	897.38	548.00
Lander	1284.40	623.53	567.88	1017.00
Mid-Course Fuel	36.00	36.00	36.00	36.00
Adapter and Shroud	156.00	156.00	156.00	156.00
Total Injected Weight	3600.00	2850.00	3100.00	3200.00

Orbit N. Mi. 1000 x 19,000 n.mi.

B. THERMAL CONTROL

The temperature control system of the orbiter will involve the use of combined active and passive design concepts. During the transit phase, the Orbiter and PHP will be occulted from the Sun, as they are stowed aft of the Lander. In order to satisfy the equipment temperature limits during the entire mission, heat will need to be supplied to both main orbiter shell as well as to the PHP. Indirect effects of deployed solar cell paddles, in conjunction with heat flow from internal lander radiator surfaces may be insufficient to provide an adequate thermal environment for miscellaneous tanks, components, and the PHP. The heating effects of paddles on orbiter shell could be enhanced, should they make an angle less than 90 degrees with incident sun rays. The electrical power penalty suffered by allowing the angle to be 75 degrees for example, may perhaps be tolerable.

If this angle change in the position of the paddles is still short of providing enough energy to the orbiter during transit, then, another design may be in order, although its potential effect on weight and cost is yet to be evaluated. This technique would basically consist in rejecting a portion of the excess RTG energy from the orbiter and adapter outer walls. To transport this energy from the lander to these areas, a liquid loop system is required. This loop would be separate from that in the lander, and would demand its own pump; heat would be picked up from the lander radiator, and ejected by radiation and/or conduction, depending on needs and complexities involved. The pump power requirements would be on the order of 10 to 15 watts, to be continuously delivered during transit. By suitable design, the adapter and orbiter walls, acting as radiators, would maintain internal sections at desired temperatures. At separation time, the two liquid loop systems would separate, and the one in the orbiter would cease functioning, since it is no longer needed.

Payload components will be insulated on all but those facing space, which are covered with louvers. Both payload and PHP will have the same thermal design as described in the Section 3.3.2.(B).

C. STRUCTURAL DESIGN

The structural design of the Orbiter of the Orbiter/Lander system is identical to that of the All-Orbiter as detailed in Section 3.3. In brief, the structure is semi-monocoque with tension and compression carrying longerons, and sandwich panels for shear capability.

The external structure is composed of eight side shear panels and eight longerons plus the seven solar panels attached to the structure. The internal structure consists of two main sandwich beams on which the propellant tanks are supported and five bulkheads to provide support for additional tankage and payload equipment. The structural criteria to which the Orbiter is designed is listed in detail in Section 3.3. In brief, the Orbiter is designed to withstand the Titan IIIC launch environment plus the transit and orbiting environments associated with the required mission.

D. DEPLOYABLE DEVICES

1. Solar Panels

Solar panels are attached to the Orbiter structure by means of a hinge at the base and by explosive actuators at the upper end. Primary and secondary solar panels are used. The primary solar panels are those mounted to the basic Orbiter structure and the secondary solar panels are those attached to the primary panels. (Reference Figure 3.4-2.)

After booster separation, the explosive actuators are fired and the solar panels deployed. As the primary panels are being deployed, the secondary panels are being moved to their location by means of spring actuators.

2. Magnetometer and Magnetometer Boom

The magnetometer and magnetometer boom are attached to the base of the Orbiter. Deployment and operation is identical to that listed for the Orbiter of Section 3.3.2.(D). Release of the boom is by means of an explosive actuator, deployment is by springs, and the boom is locked in place after deployment.

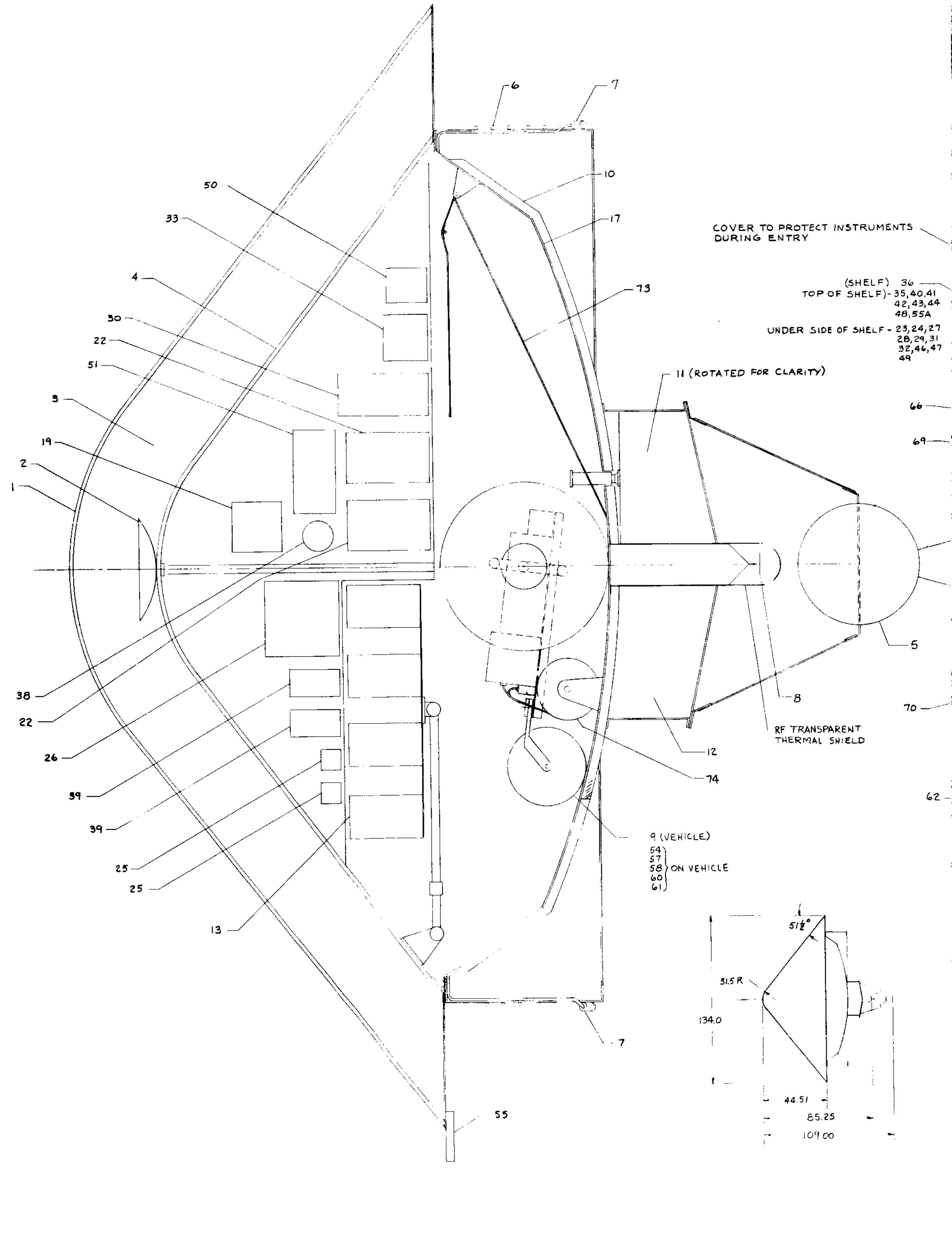
3. Planet Horizontal Package (PHP)

The PHP location, attachment, and deployment is similar to that of the Orbiter of Section 3.3.2.(D). As shown in Figure 3.4-2, the PHP is located within the Lander adapter. After orbit insertion, the PHP is deployed and, at an appropriate time, the adapter is separated from the spacecraft. This allows full view of the planet by the PHP.

4. High-Gain Antenna

An eight-foot high-gain antenna is packaged with the Orbiter/Lander. Shroud limitations made it necessary that the antenna be packaged below the main engine nozzle. Three attachments are provided (120 degrees apart). Two attachments are explosively actuated immediately after spacecraft separation from the launch vehicle. At this time the antenna is used to verify the orientation of the spacecraft. After verification, the antenna is stowed alongside the spacecraft until 120 days. At this time the Earth-Spacecraft-Sun angle is 30 degrees. (Reference Figures 3.3-21 and 3.3-22).

Although the maximum load expected during orbit insertion is only 0.625 g , the antenna is designed to be stowed during orbit insertion and mid-course engine firings.



COVER TO PROTECT INSTRUMENTS DURING ENTRY
(SHELF) 36
TOP OF SHELF)-35,40,41 42,43,44 48,55A
UNDER SIDE OF SHELF - 23,24,27 28,29,31 32,46,47 49
66
69
70
62
6
7
10
17
73
50
33
4
30
22
51
3
19
2
1
11 (ROTATED FOR CLARITY)
5
8
RF TRANSPARENT THERMAL SHIELD
12
74
38
22
26
39
39
25
25
13
9 (VEHICLE)
54 57 58 60 61 ON VEHICLE
7
55
$51\frac{1}{2}°$
31.5 R
134.0
44.51
85.25
109.00

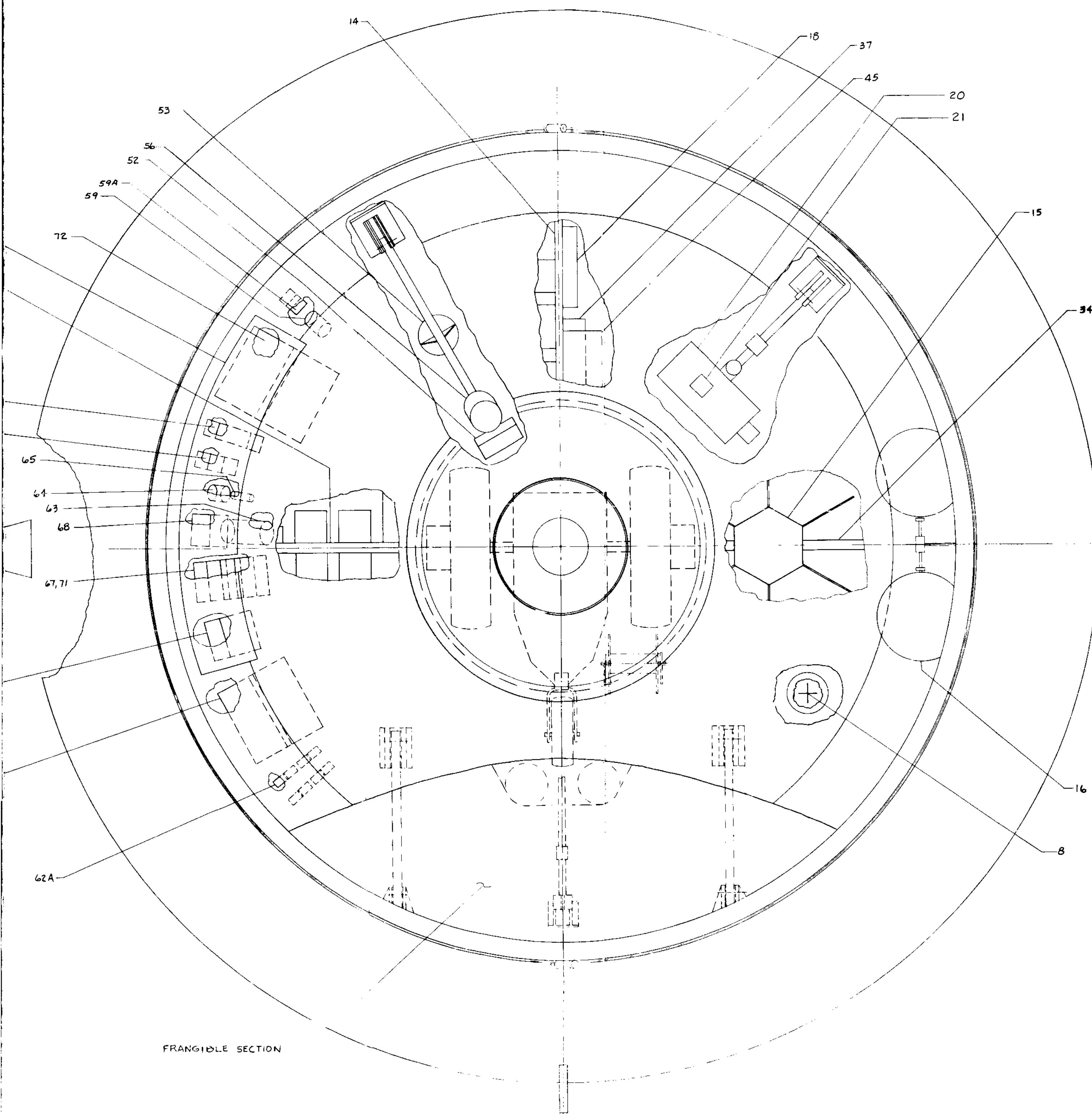
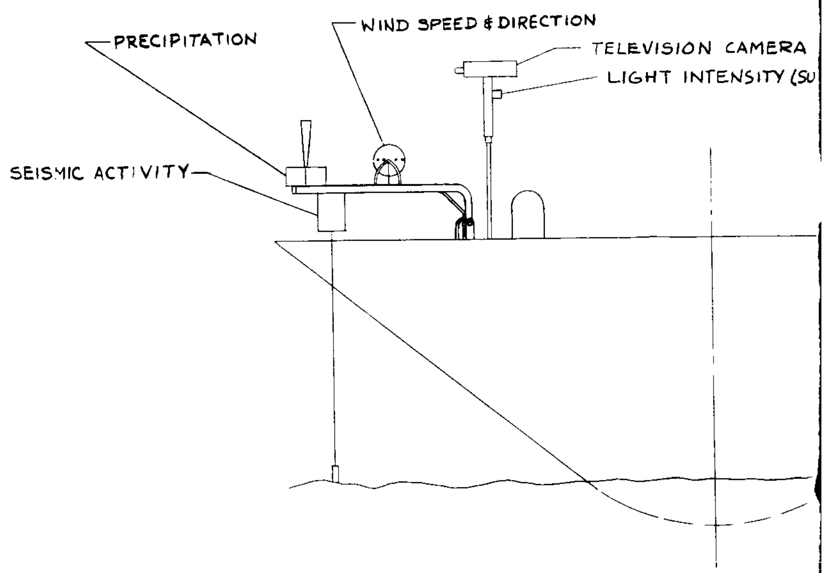


Figure 3.2-50. Entry/Lander with Rove

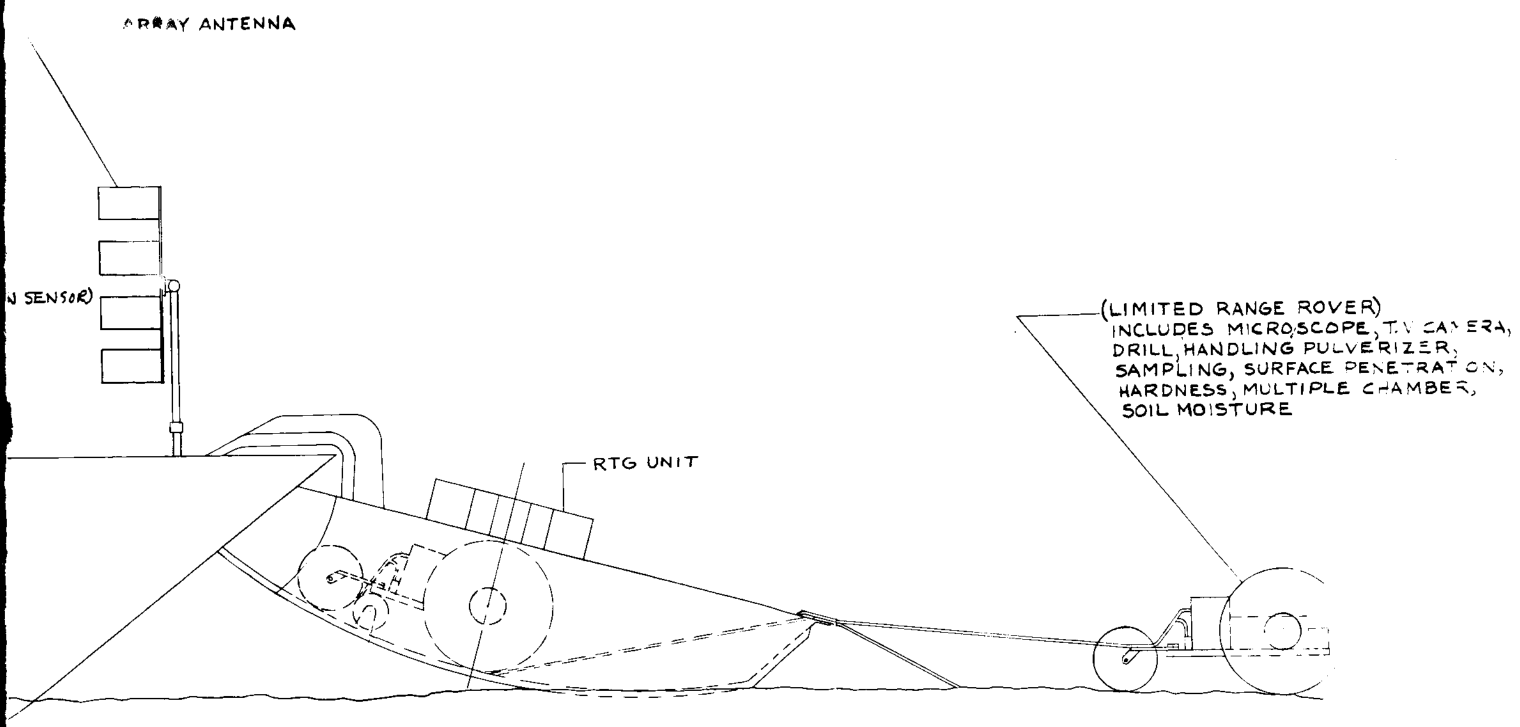


ITEM – DESCRIPTION

1 – HEAT SHIELD
2 – RADAR ALTIMETER ANTENNA
3 – CRUSH UP (FIBERGLASS HONEYCOMB)
4 – STRUCTURAL SHELL
5 – DELTA "V" ROCKET
6 – ADAPTER SECTION (TRANSIENT RADIATOR)
7 – SPIN & SEPARATION SYSTEM
8 – OMNI ANTENNA
9 – LIMITED RANGE ROVER
10 – CRUSH UP
11 – PARACHUTE PACKAGE (DECELERATOR)
12 – PARACHUTE PACKAGE (MAIN)
13 – HELIX ARRAY ANTENNA
14 – SHELF
15 – RTG UNIT
16 – GAS RESERVOIR
17 – AFT COVER
18 – PHOTOAUTOTROPH
19 – TURBIDITY
20 – TV CAMERA
21 – LIGHT INTENSITY (SUN SENSOR)
22 – TRANSPONDER
23 – EVAPORATIVE HEAT EXCHANGER
24 – MOTOR & PUMPS
25 – ISOLATOR & LOAD

ITEM – DESCRIPTION

26 – BATTERY
27 – LIQUID HEAT EXC
28 – MODULATION VA
29 – COOLANT RESE
30 – POWER CONTROL
31 – MODULATION V
32 – TEMPERATURE
33 – BUFFER CONT
34 – SHELF
35 – TAPE RECORDE
36 – SHELF
37 – SURFACE GRA
38 – POWER AMP
39 – DIPLEXER
40 – H.V. PWR SUPP
41 – H.V. PWR SUPP
42 – H.V. PWR SUPP
43 – TRANSMITTER
44 – COMMAND DEC
45 – ALTIMETER AN
46 – POWER AMP
47 – POWER AMP
48 – RF SWITCH
49 – WATER
50 – COMMAND & C

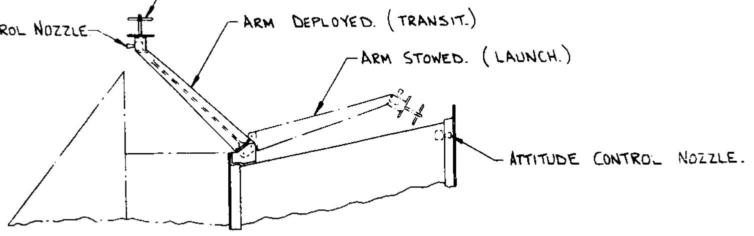


SURFACE DEPLOYED CONFIGURATION

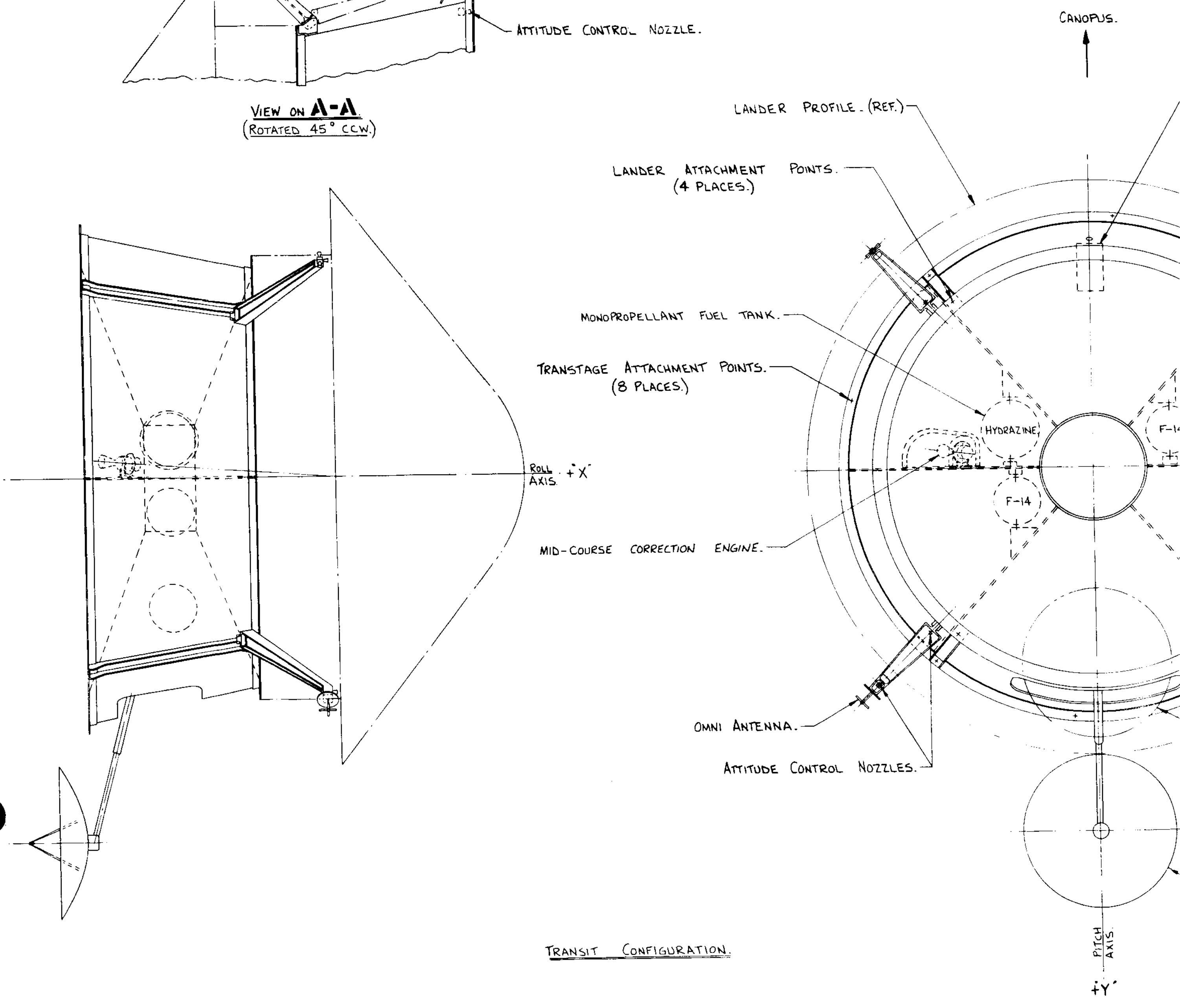
CHANGER
LVE
RVOIR
& CONVERSION UNIT
ALVE
CONTROL
ROL
ER
VITY
LY
LY
LY
ODER
TENNA ELECTRONICS
OMPUTER EQUIP.

ITEM	DESCRIPTION
51	DATA PROCESSING UNIT
52	PRECIPITATION
53	WIND SPEED & DIRECTION
54	SURFACE PENETRATION HARDNESS
55	ELECTRON DENSITY (LANGMUIR PROBE)
55A	ELECTRONICS FOR ELECTRON DENSITY
56	SEISMIC ACTIVITY
57	MICROSCOPE (INCLUDES T.V. CAMERA, DRILL, HANDLING PULVERIZER, SAMPLE)
58	MULTIPLE CHAMBER
59	SOUNDS
59A	ELECTRONICS FOR SOUNDS
60	SAMPLE GATHERER
61	SOIL MOISTURE
62	RADIOISOTOPE
62A	ELECTRONICS FOR RADIOISOTOPE
63	COMPOSITION O_3
64	PRESSURE
65	TEMPERATURE
66	COMPOSITION A
67	COMPOSITION H_2O
68	COMPOSITION CO_2
69	DENSITY
70	COMPOSITION N_2
71	COMPOSITION O_2
72	GAS CHROMATOGRAPH
73	RAMP
74	CABLE REEL

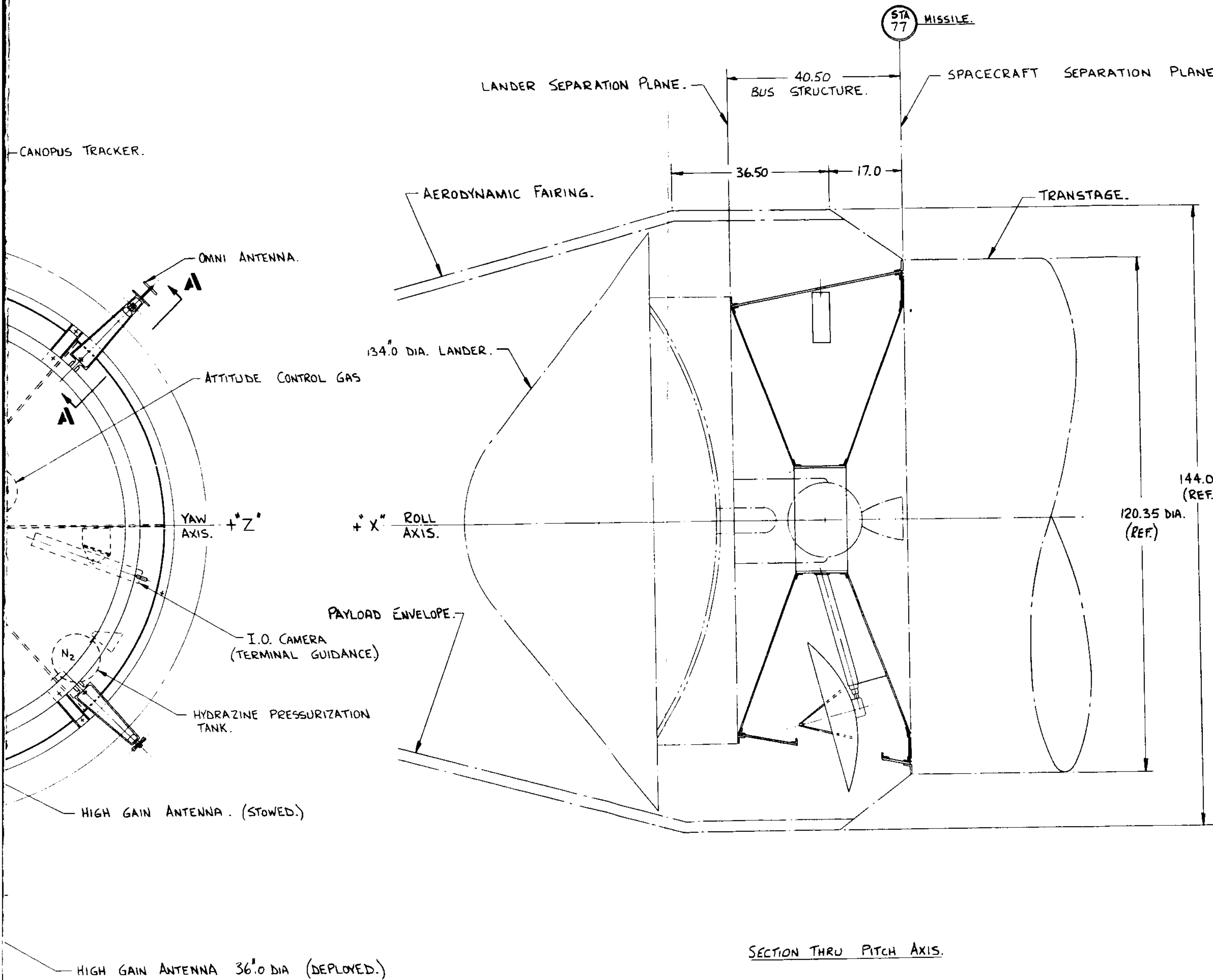
Figure 3.2-51. Entry/Lander with Rove
Surface Deployed Configuration



VIEW ON A-A
(ROTATED 45° C.C.W.)



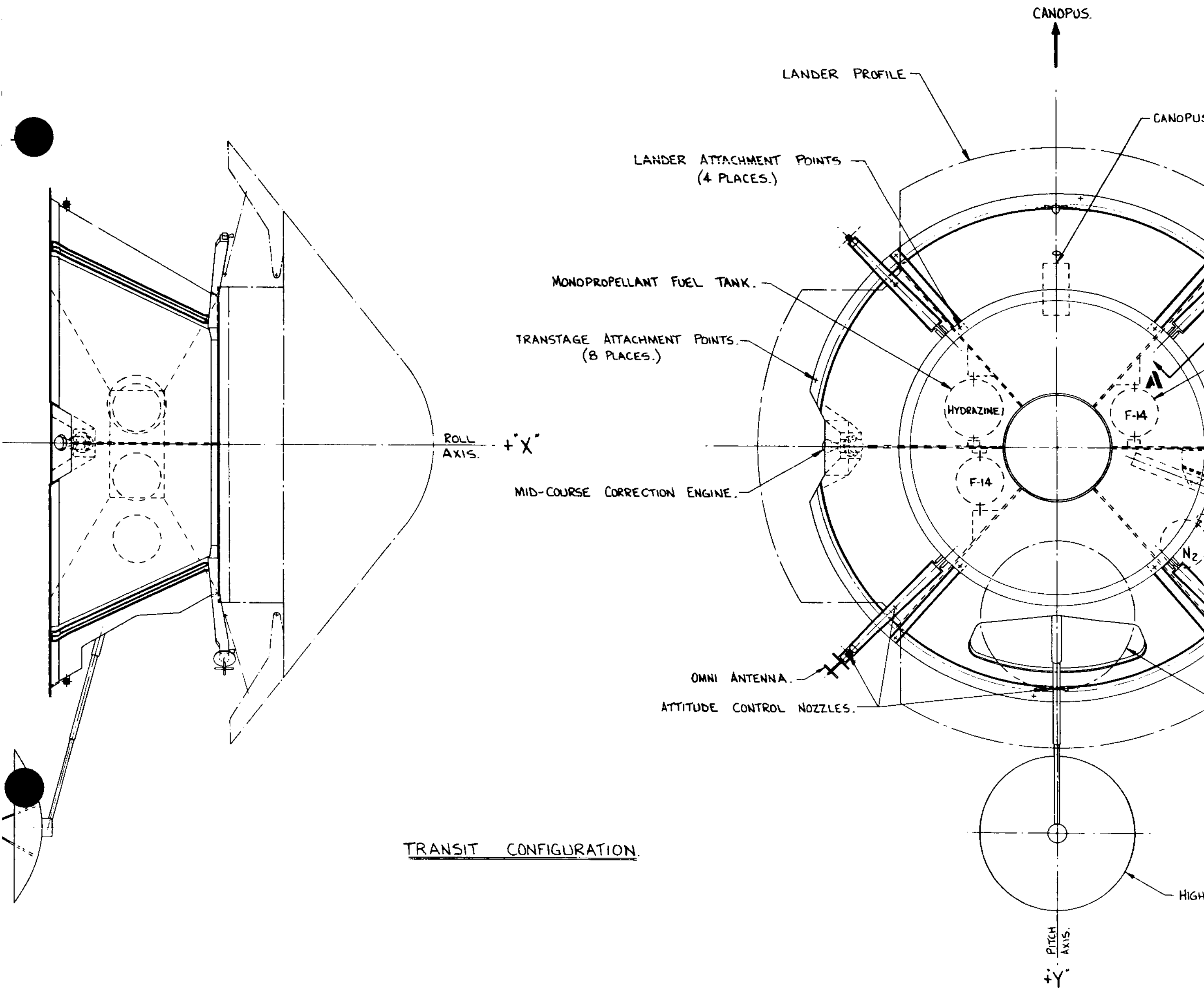
TRANSIT CONFIGURATION.



SECTION THRU PITCH AXIS.

LAUNCH CONFIGURATION.

Figure 3.2-56. Bus/Lander Configuratio
with 134-Inch Base Diameter Lander



TRANSIT CONFIGURATION.

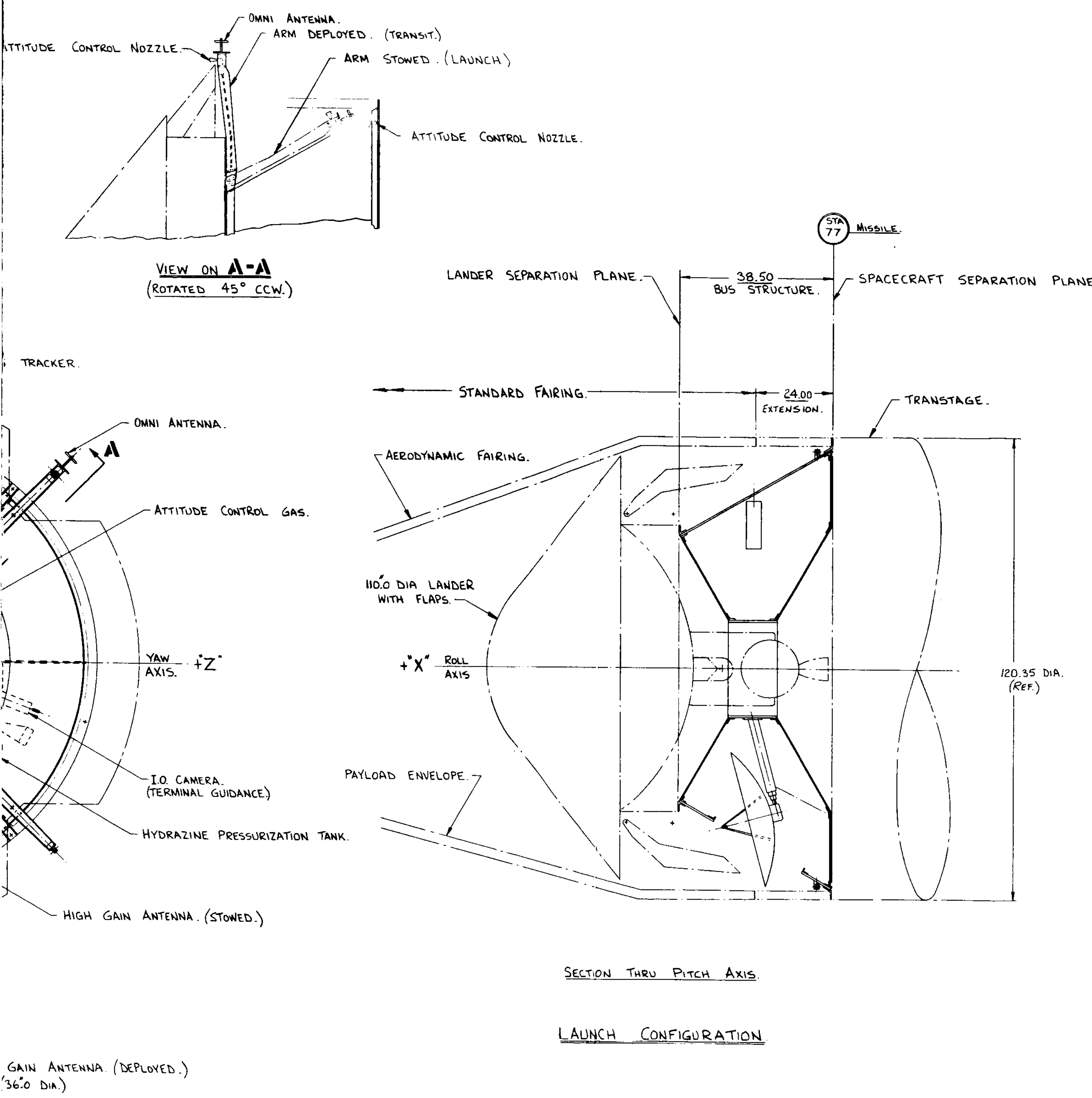
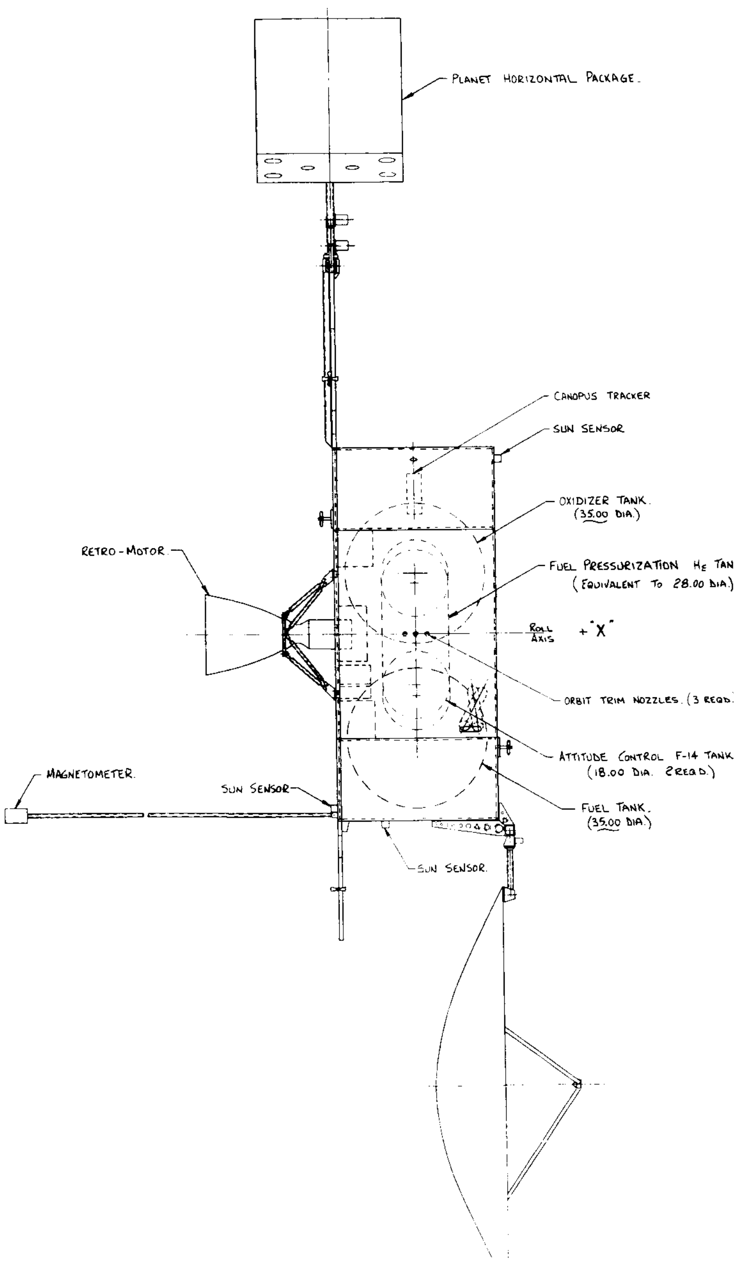
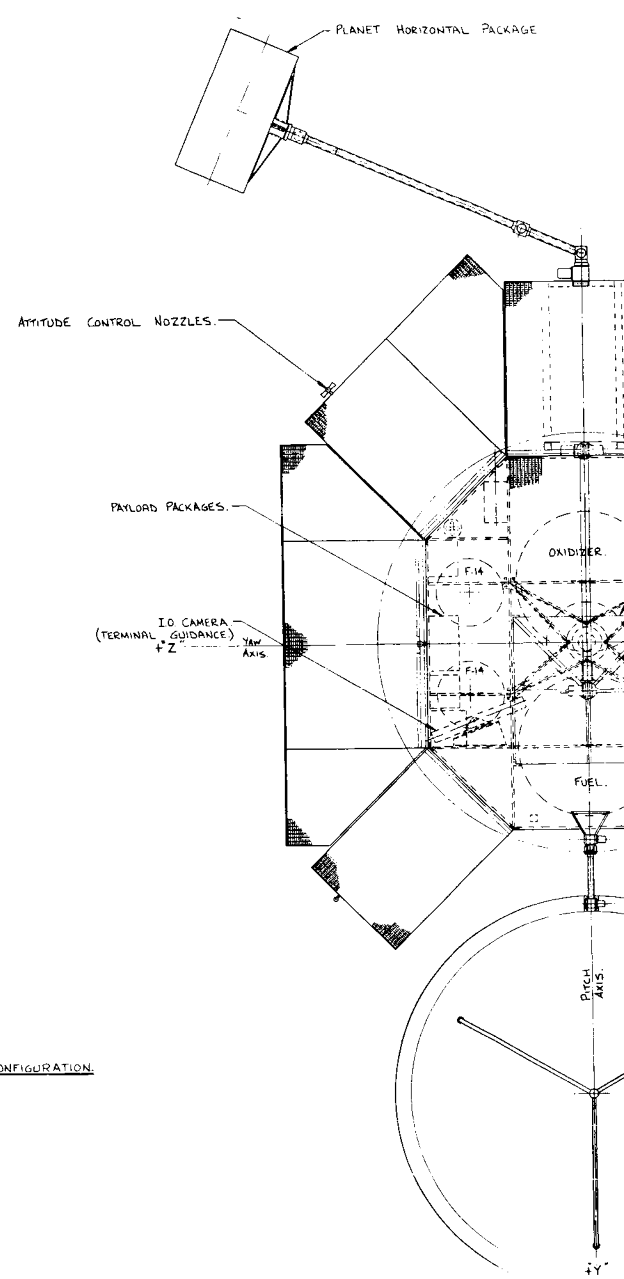
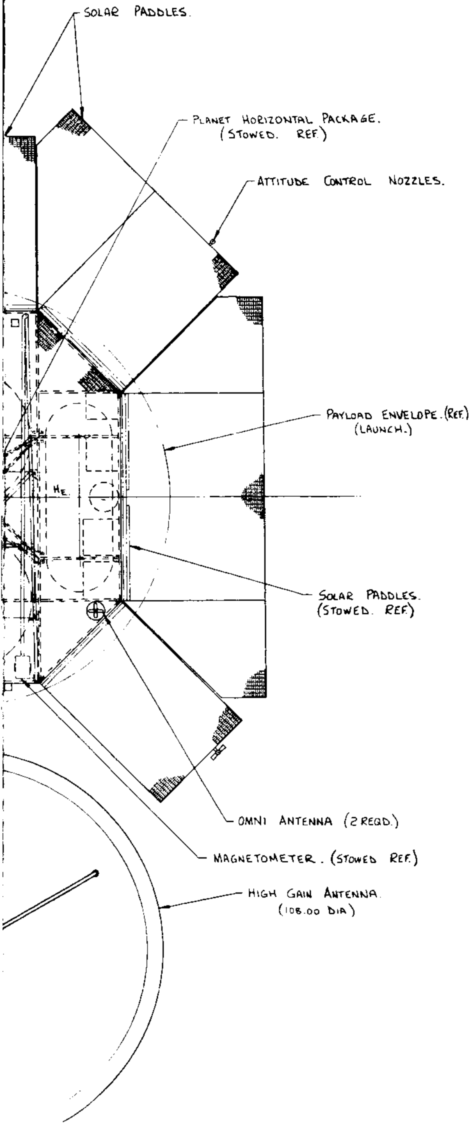


Figure 3-2-57. Bus/Lander Configuration with 110-Inch Base Diameter Lander with Flaps





ORBITING CONFIGURATION.



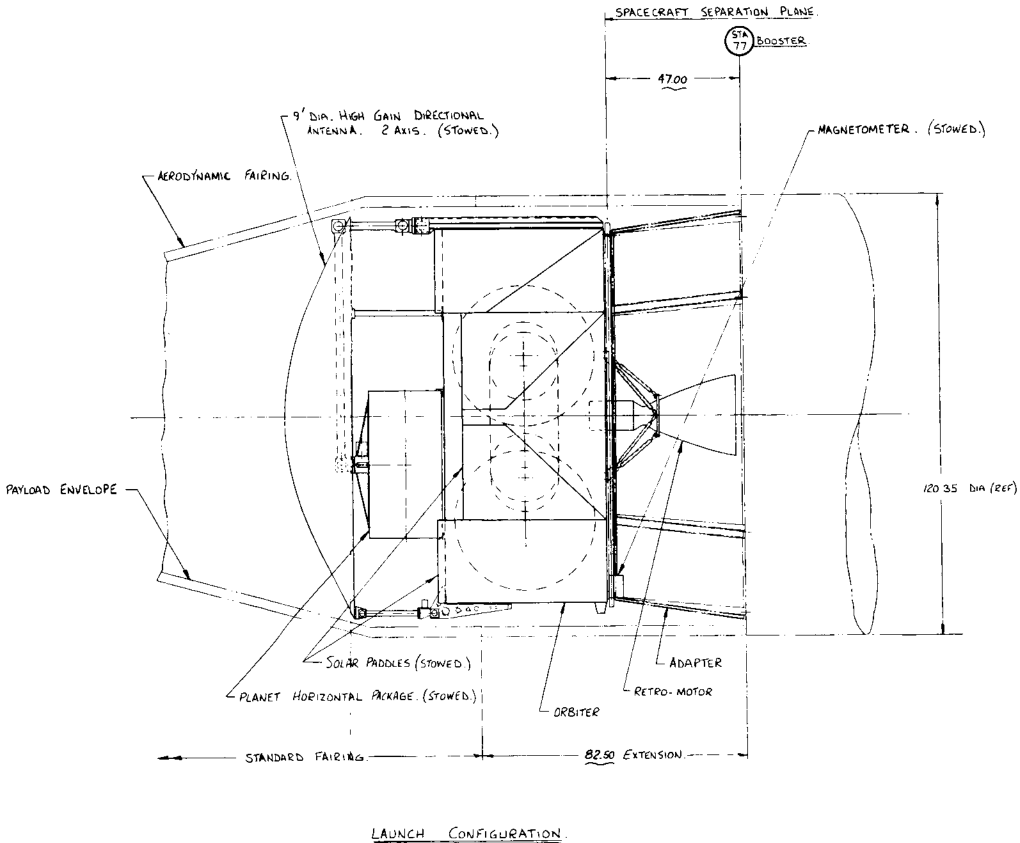
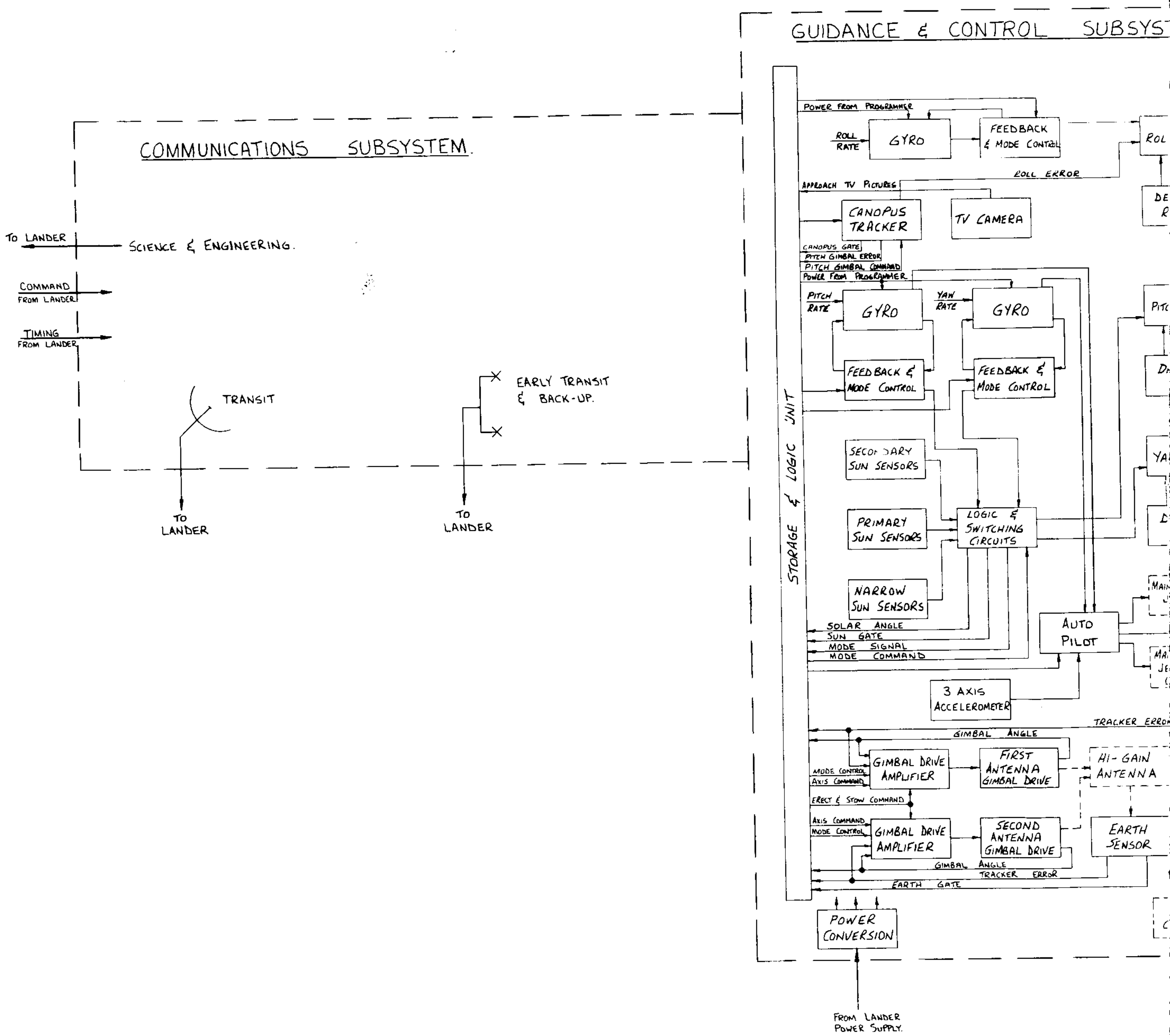


Figure 3.3-18. Solar Powered Orbiter System Configuration



GUIDANCE & CONTROL SUBSYS
COMMUNICATIONS SUBSYSTEM.
TO LANDER
SCIENCE & ENGINEERING.
COMMAND FROM LANDER
TIMING FROM LANDER
TRANSIT
EARLY TRANSIT & BACK-UP.
TO LANDER
TO LANDER
STORAGE & LOGIC UNIT
POWER FROM PROGRAMMER
ROLL RATE
GYRO
FEEDBACK & MODE CONTROL
ROLL ERROR
APPROACH TV PICTURES
CANOPUS TRACKER
TV CAMERA
CANOPUS GATE
PITCH GIMBAL ERROR
PITCH GIMBAL COMMAND
POWER FROM PROGRAMMER
PITCH RATE
GYRO
YAW RATE
GYRO
FEEDBACK & MODE CONTROL
FEEDBACK & MODE CONTROL
SECONDARY SUN SENSORS
PRIMARY SUN SENSORS
LOGIC & SWITCHING CIRCUITS
NARROW SUN SENSORS
SOLAR ANGLE
SUN GATE
MODE SIGNAL
MODE COMMAND
AUTO PILOT
3 AXIS ACCELEROMETER
TRACKER ERROR
GIMBAL ANGLE
MODE CONTROL
AXIS COMMAND
GIMBAL DRIVE AMPLIFIER
FIRST ANTENNA GIMBAL DRIVE
HI-GAIN ANTENNA
ERECT & STOW COMMAND
AXIS COMMAND
MODE CONTROL
GIMBAL DRIVE AMPLIFIER
SECOND ANTENNA GIMBAL DRIVE
EARTH SENSOR
GIMBAL ANGLE
TRACKER ERROR
EARTH GATE
POWER CONVERSION
FROM LANDER POWER SUPPLY.

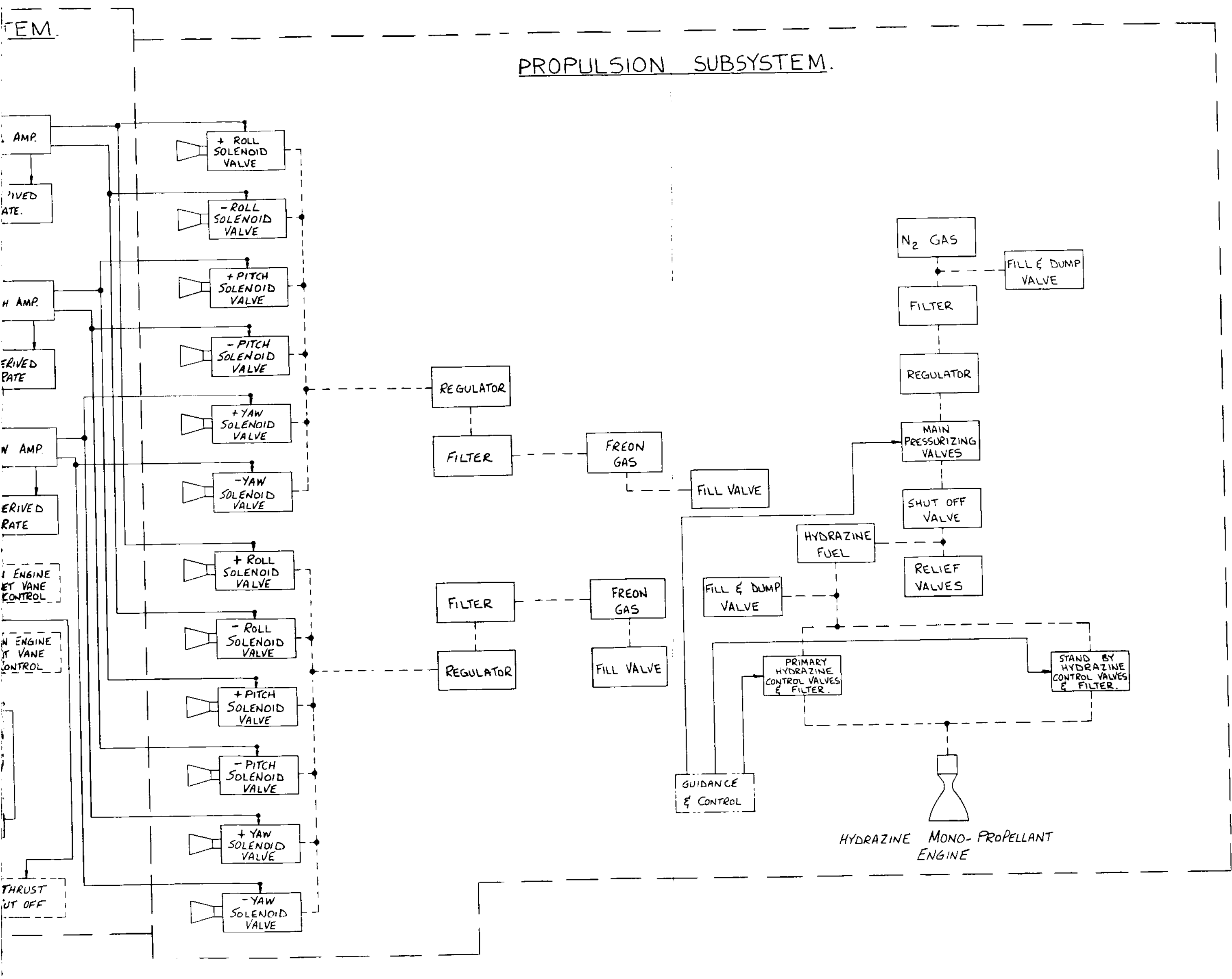
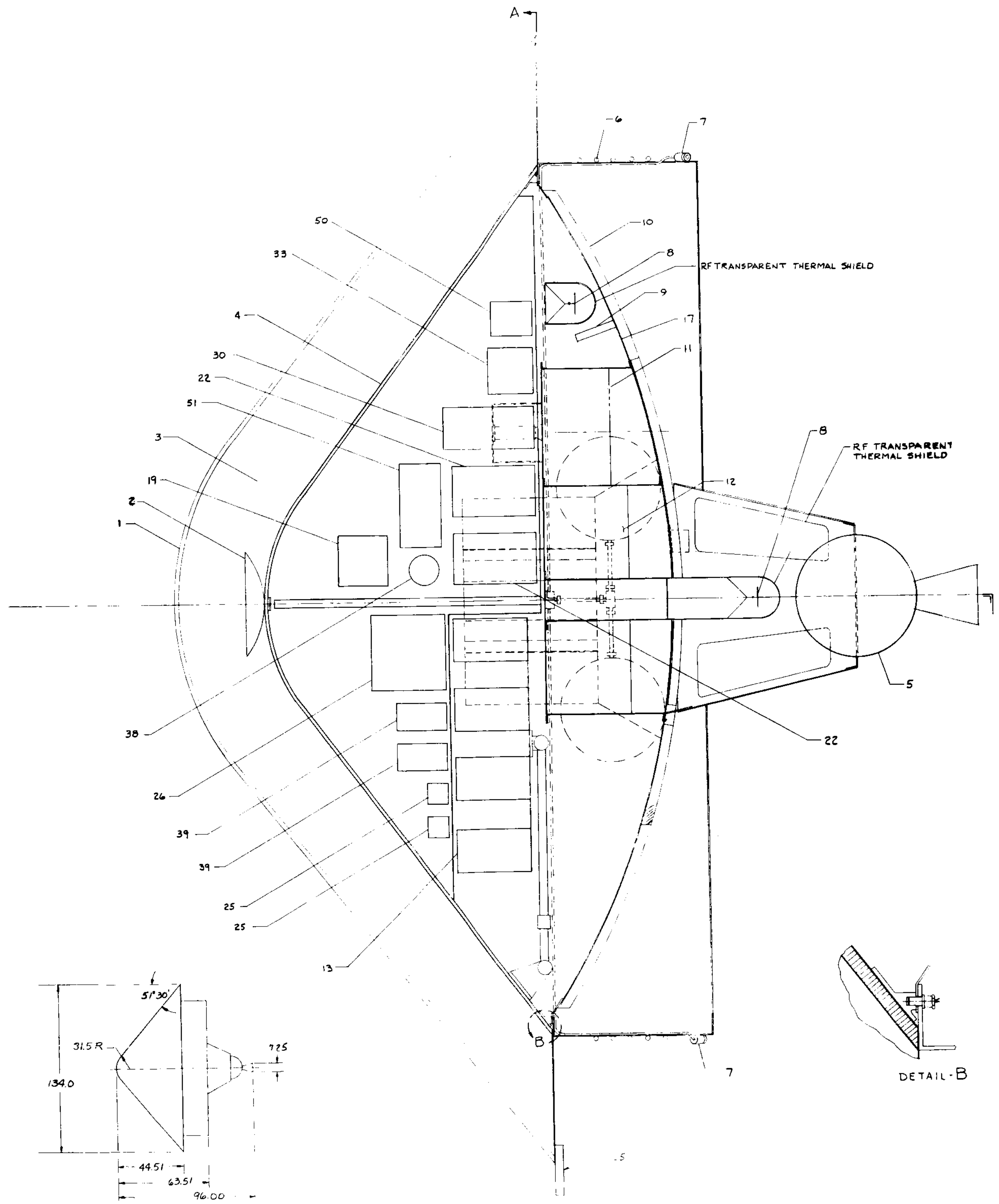


Figure 3.2-63. Bus/Lander Block Diagram



A
6
7
10
50
33
8
RF TRANSPARENT THERMAL SHIELD
9
17
4
30
11
22
51
8
RF TRANSPARENT THERMAL SHIELD
3
12
19
2
1
5
38
22
26
39
39
25
25
13
51°30'
31.5 R
7.25
134.0
44.51
63.51
96.00
B
7
DETAIL-B

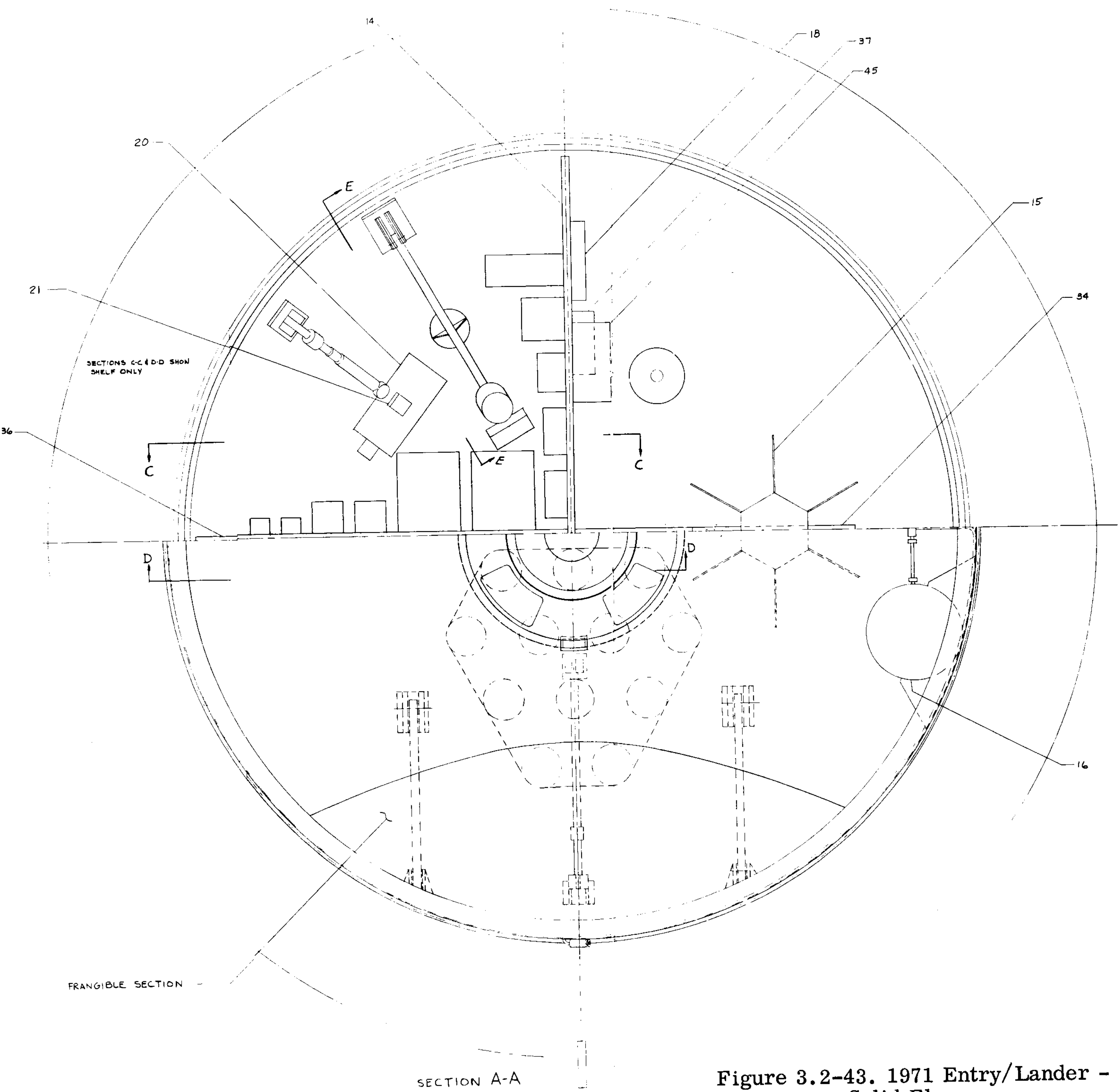
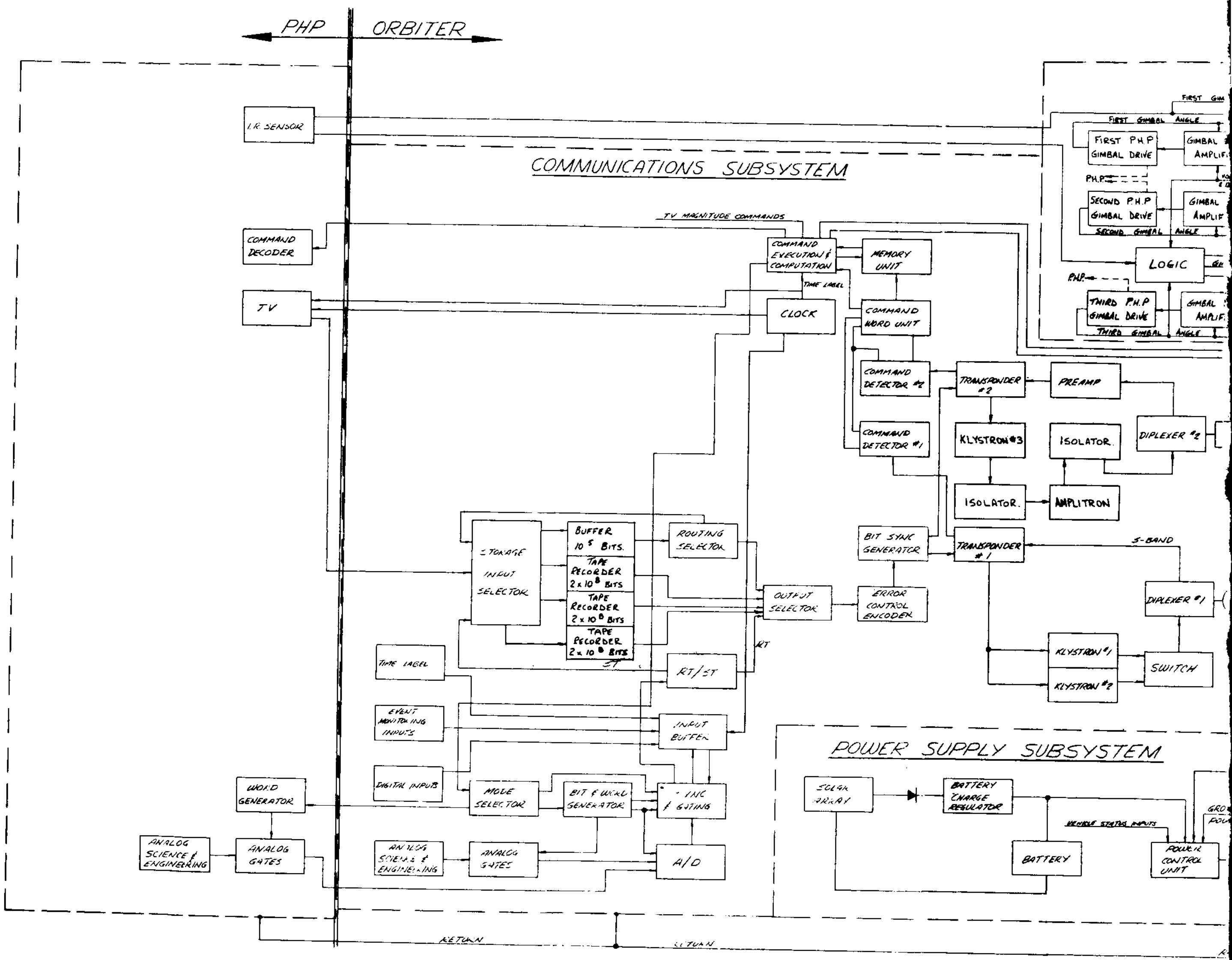
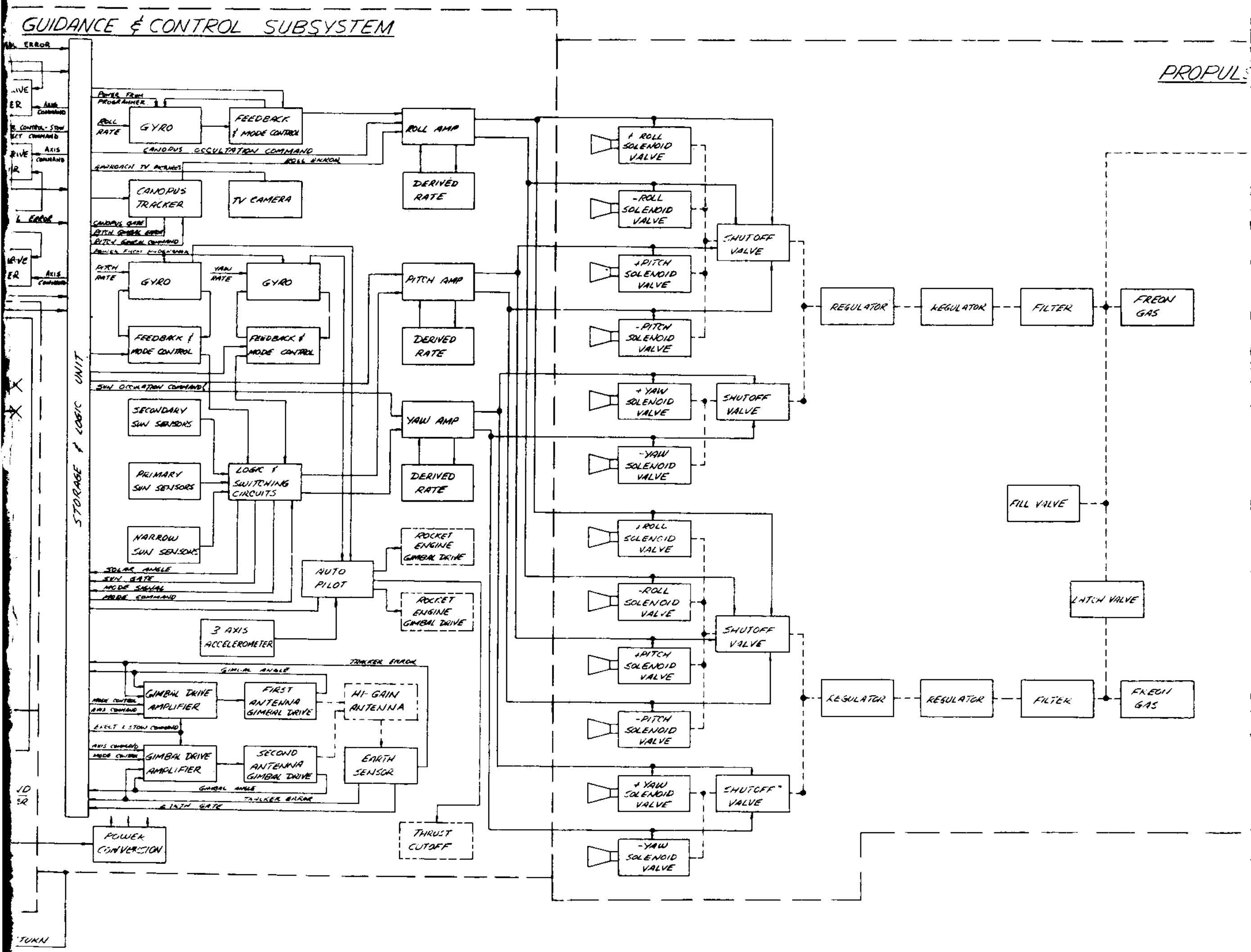


Figure 3.2-43. 1971 Entry/Lander - Solid Flare


PHP
ORBITER
COMMUNICATIONS SUBSYSTEM
I.R. SENSOR
COMMAND DECODER
TV
TV MAGNITUDE COMMANDS
COMMAND EXECUTION & COMPUTATION
MEMORY UNIT
TIME LABEL
CLOCK
COMMAND WORD UNIT
COMMAND DETECTOR #2
COMMAND DETECTOR #1
TRANSPONDER #2
PREAMP
KLYSTRON #3
ISOLATOR
DIPLEXER #2
ISOLATOR
AMPLITRON
FIRST P.H.P GIMBAL DRIVE
SECOND P.H.P GIMBAL DRIVE
THIRD P.H.P GIMBAL DRIVE
LOGIC
FIRST GIMBAL ANGLE
SECOND GIMBAL ANGLE
THIRD GIMBAL ANGLE
STORAGE INPUT SELECTOR
BUFFER 10^5 BITS
TAPE RECORDER 2 x 10^8 BITS
TAPE RECORDER 2 x 10^8 BITS
TAPE RECORDER 2 x 10^8 BITS
ROUTING SELECTOR
OUTPUT SELECTOR
BIT SYNC GENERATOR
ERROR CONTROL ENCODER
TRANSPONDER #1
S-BAND
DIPLEXER #1
KLYSTRON #1
KLYSTRON #2
SWITCH
TIME LABEL
RT/ST
RT
ST
EVENT MONITORING INPUTS
INPUT BUFFER
DIGITAL INPUTS
MODE SELECTOR
BIT & WORD GENERATOR
SYNC & GATING
WORD GENERATOR
ANALOG SCIENCE & ENGINEERING
ANALOG GATES
ANALOG SCIENCE & ENGINEERING
ANALOG GATES
A/D
POWER SUPPLY SUBSYSTEM
SOLAR ARRAY
BATTERY CHARGE REGULATOR
BATTERY
POWER CONTROL UNIT
VEHICLE STATUS INPUTS
RETURN
RETURN


GUIDANCE & CONTROL SUBSYSTEM
PROPULS
STORAGE & LOGIC UNIT
GYRO
FEEDBACK & MODE CONTROL
ROLL AMP
DERIVED RATE
CANOPUS TRACKER
TV CAMERA
PITCH AMP
YAW AMP
SECONDARY SUN SENSORS
PRIMARY SUN SENSORS
NARROW SUN SENSORS
LOGIC & SWITCHING CIRCUITS
AUTO PILOT
ROCKET ENGINE GIMBAL DRIVE
3 AXIS ACCELEROMETER
GIMBAL DRIVE AMPLIFIER
FIRST ANTENNA GIMBAL DRIVE
SECOND ANTENNA GIMBAL DRIVE
HI-GAIN ANTENNA
EARTH SENSOR
POWER CONVERSION
THRUST CUTOFF
+ROLL SOLENOID VALVE
-ROLL SOLENOID VALVE
+PITCH SOLENOID VALVE
-PITCH SOLENOID VALVE
+YAW SOLENOID VALVE
-YAW SOLENOID VALVE
SHUTOFF VALVE
REGULATOR
FILTER
FREON GAS
FILL VALVE
LATCH VALVE
CANOPUS OCCULTATION COMMAND
ROLL ERROR
SUN OCCULTATION COMMAND
SOLAR ANGLE
SUN GATE
MODE SIGNAL
MODE COMMAND
GIMBAL ANGLE
TRACKER ERROR
EARTH GATE

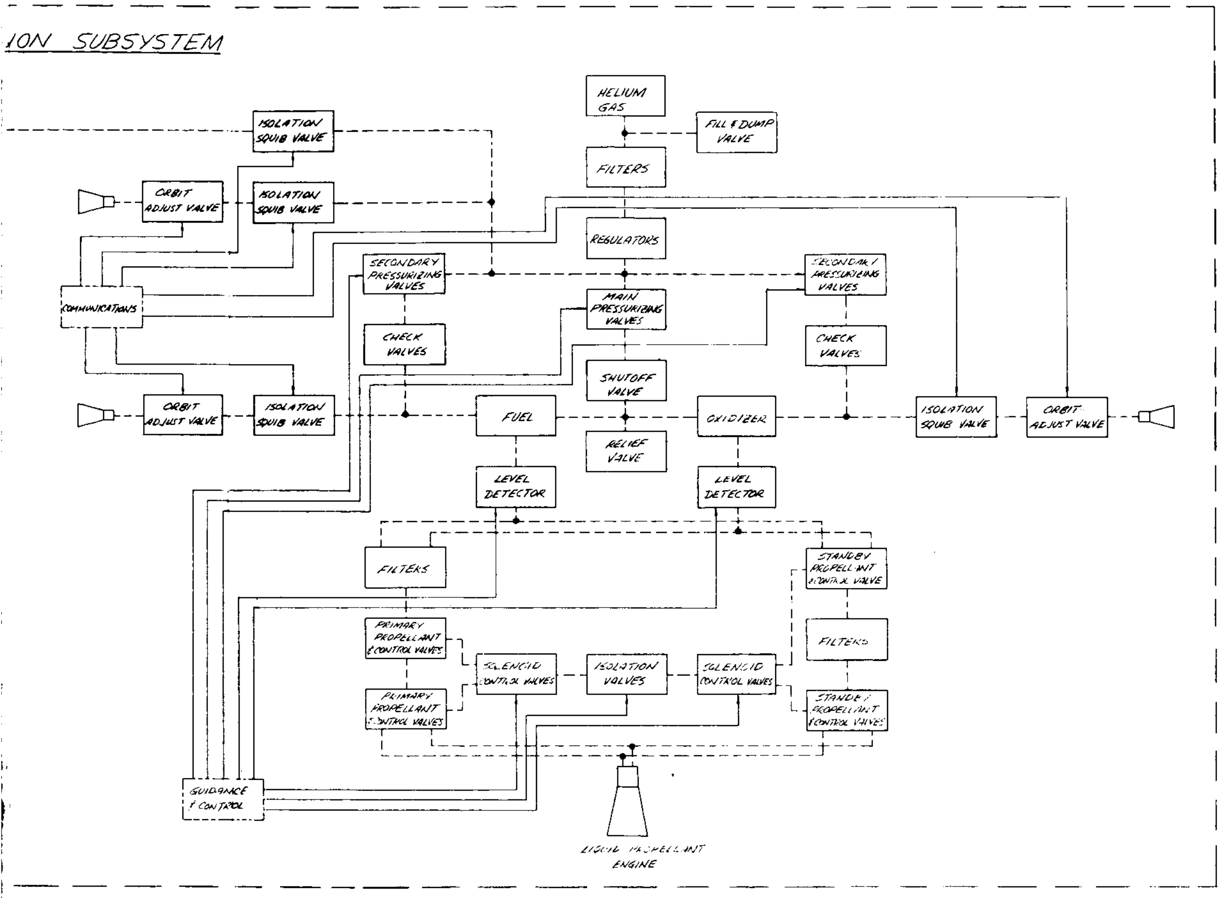
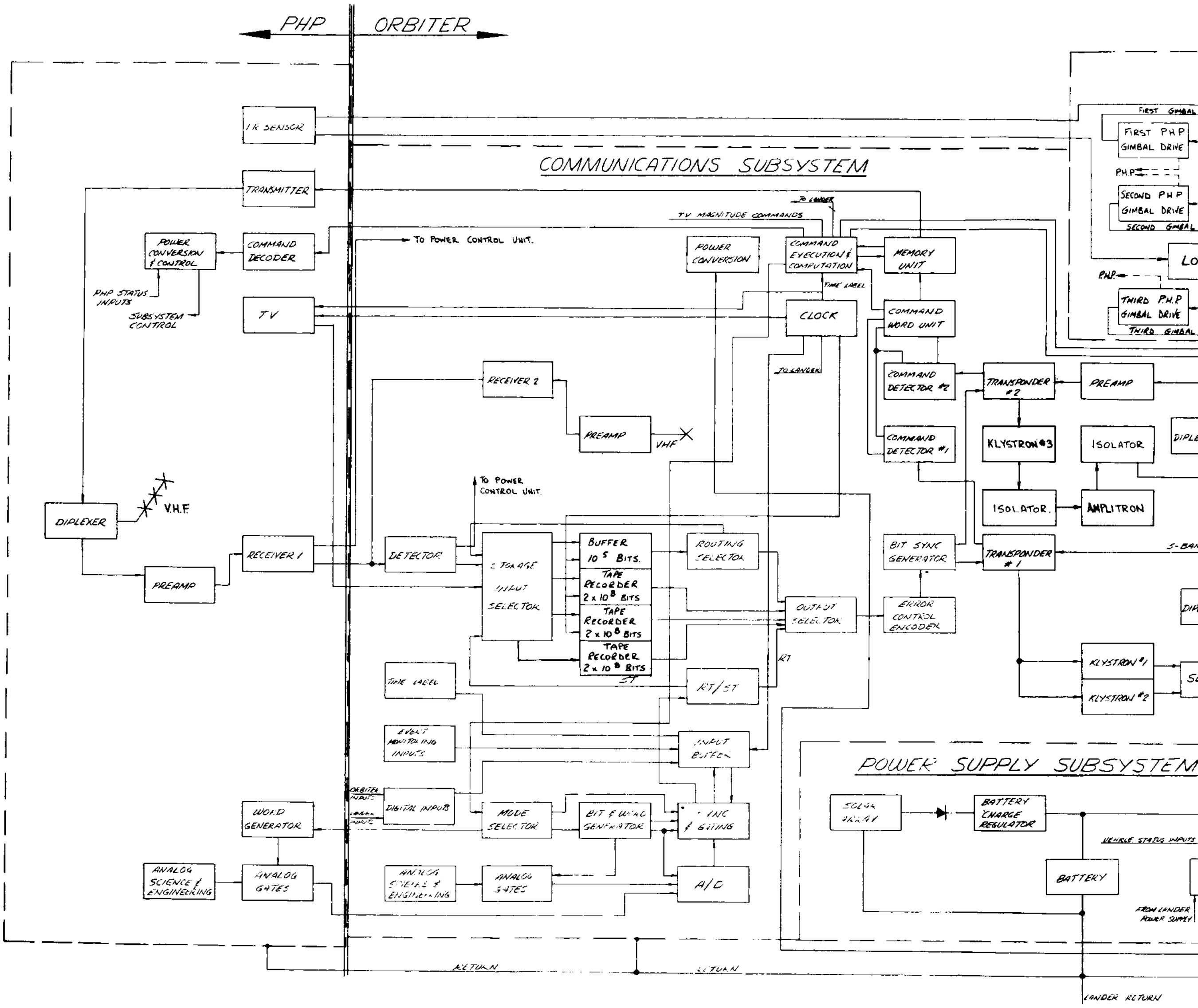
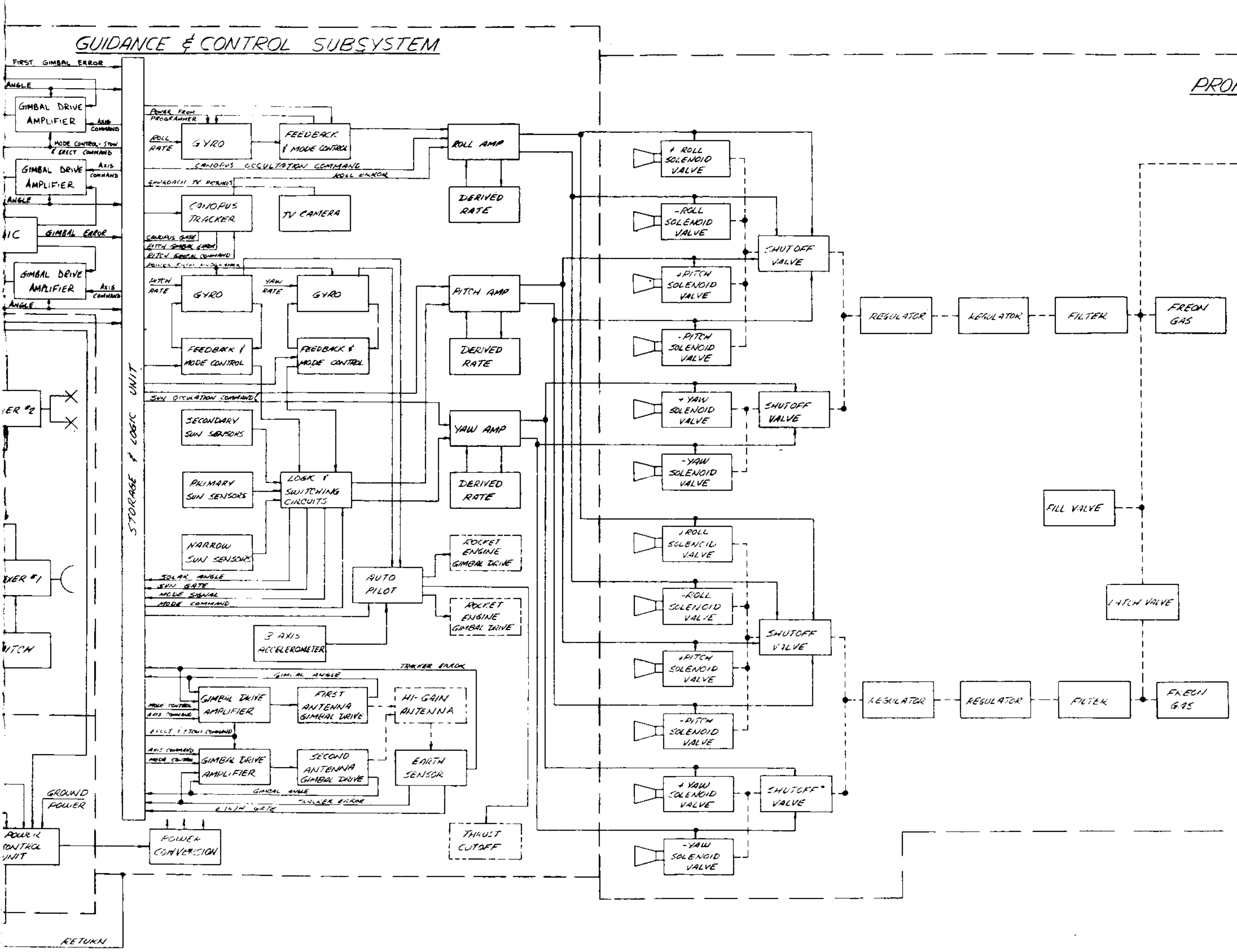


Figure 3.3-20. Orbiter System Block Diagram



PHP
ORBITER
COMMUNICATIONS SUBSYSTEM
POWER SUPPLY SUBSYSTEM
I R SENSOR
TRANSMITTER
COMMAND DECODER
POWER CONVERSION & CONTROL
PHP STATUS INPUTS
SUBSYSTEM CONTROL
TV
TO POWER CONTROL UNIT.
DIPLEXER
V.H.F
PREAMP
RECEIVER 1
RECEIVER 2
VHF
DETECTOR
TO POWER CONTROL UNIT.
STORAGE INPUT SELECTOR
BUFFER 10^5 BITS.
TAPE RECORDER 2 x 10^8 BITS
ROUTING SELECTOR
OUTPUT SELECTOR
RT/ST
RT
ST
TIME LABEL
EVENT MONITORING INPUTS
INPUT BUFFER
ORBITER INPUTS
LANDER INPUTS
DIGITAL INPUTS
MODE SELECTOR
BIT & WORD GENERATOR
SYNC & GATING
A/D
WORD GENERATOR
ANALOG SCIENCE & ENGINEERING
ANALOG GATES
RETURN
TV MAGNITUDE COMMANDS
TO LANDER
POWER CONVERSION
COMMAND EXECUTION & COMPUTATION
MEMORY UNIT
TIME LABEL
CLOCK
COMMAND WORD UNIT
TO LANDER
COMMAND DETECTOR #2
COMMAND DETECTOR #1
TRANSPONDER #2
TRANSPONDER #1
PREAMP
KLYSTRON #3
ISOLATOR
AMPLITRON
BIT SYNC GENERATOR
ERROR CONTROL ENCODER
KLYSTRON #1
KLYSTRON #2
S-BAN
FIRST GIMBAL
FIRST PHP GIMBAL DRIVE
PHP
SECOND PHP GIMBAL DRIVE
SECOND GIMBAL
THIRD PHP GIMBAL DRIVE
THIRD GIMBAL
SOLAR ARRAY
BATTERY CHARGE REGULATOR
BATTERY
VEHICLE STATUS INPUTS
FROM LANDER POWER SUPPLY
LANDER RETURN


GUIDANCE & CONTROL SUBSYSTEM
PRO
GIMBAL DRIVE AMPLIFIER
GYRO
FEEDBACK & MODE CONTROL
ROLL AMP
DERIVED RATE
CANOPUS TRACKER
TV CAMERA
PITCH AMP
YAW AMP
SECONDARY SUN SENSORS
PRIMARY SUN SENSORS
NARROW SUN SENSORS
LOGIC & SWITCHING CIRCUITS
STORAGE & LOGIC UNIT
AUTO PILOT
ROCKET ENGINE GIMBAL DRIVE
3 AXIS ACCELEROMETER
FIRST ANTENNA GIMBAL DRIVE
SECOND ANTENNA GIMBAL DRIVE
HI-GAIN ANTENNA
EARTH SENSOR
POWER CONVERSION
POWER CONTROL UNIT
THRUST CUTOFF
GROUND POWER
RETURN
+ROLL SOLENOID VALVE
-ROLL SOLENOID VALVE
+PITCH SOLENOID VALVE
-PITCH SOLENOID VALVE
+YAW SOLENOID VALVE
-YAW SOLENOID VALVE
SHUTOFF VALVE
REGULATOR
FILTER
FREON GAS
FILL VALVE
LATCH VALVE

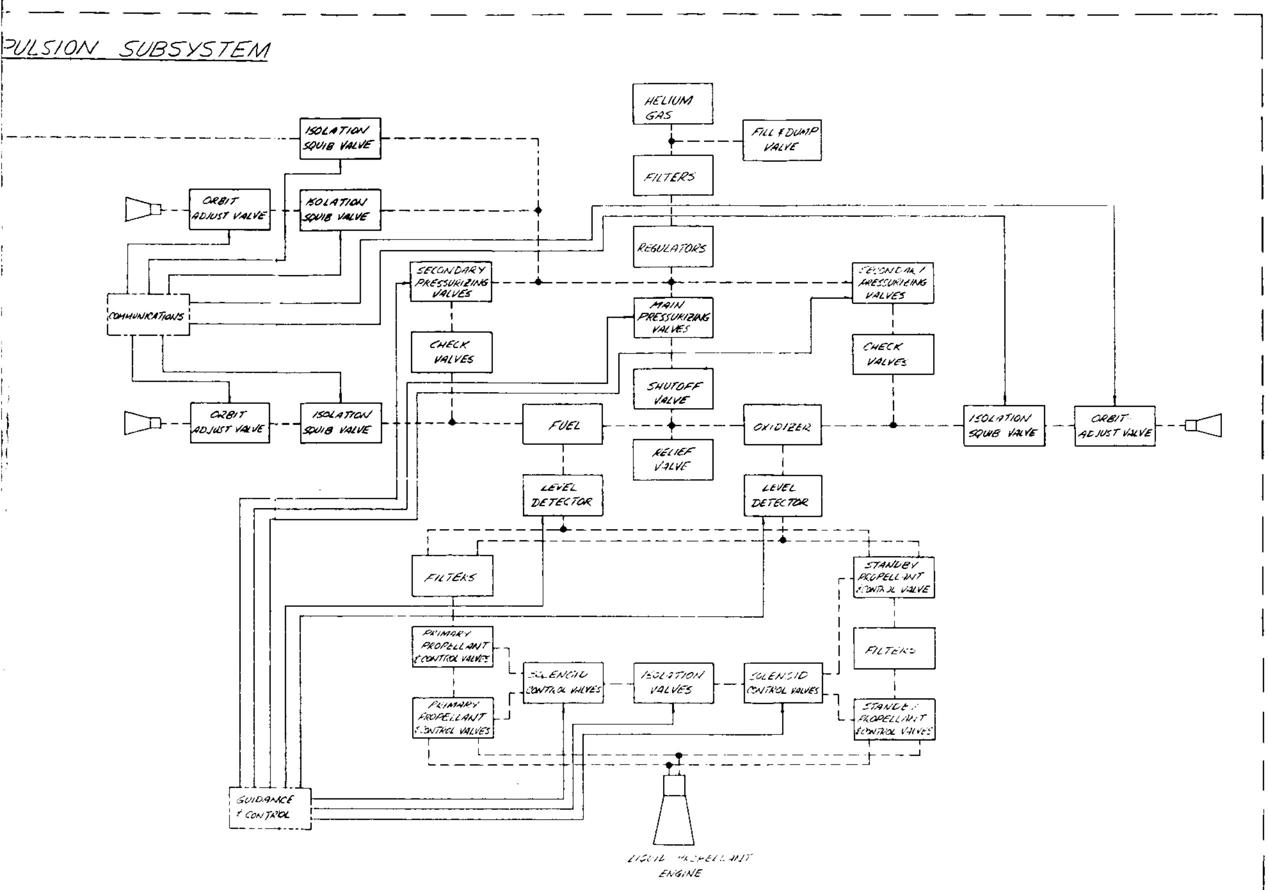
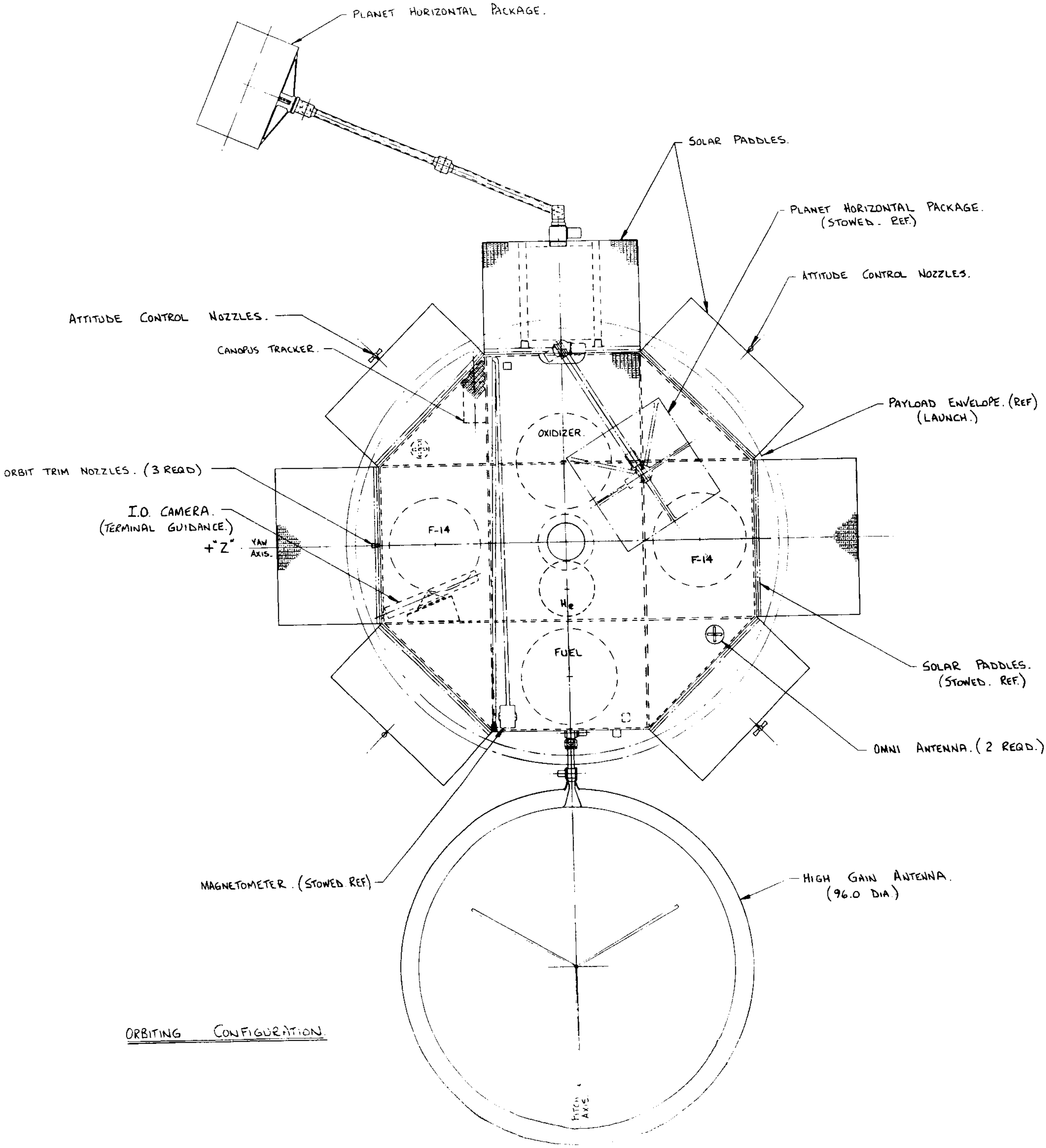


Figure 3.4-3. Orbiter System Block Diagram



ORBITING CONFIGURATION.

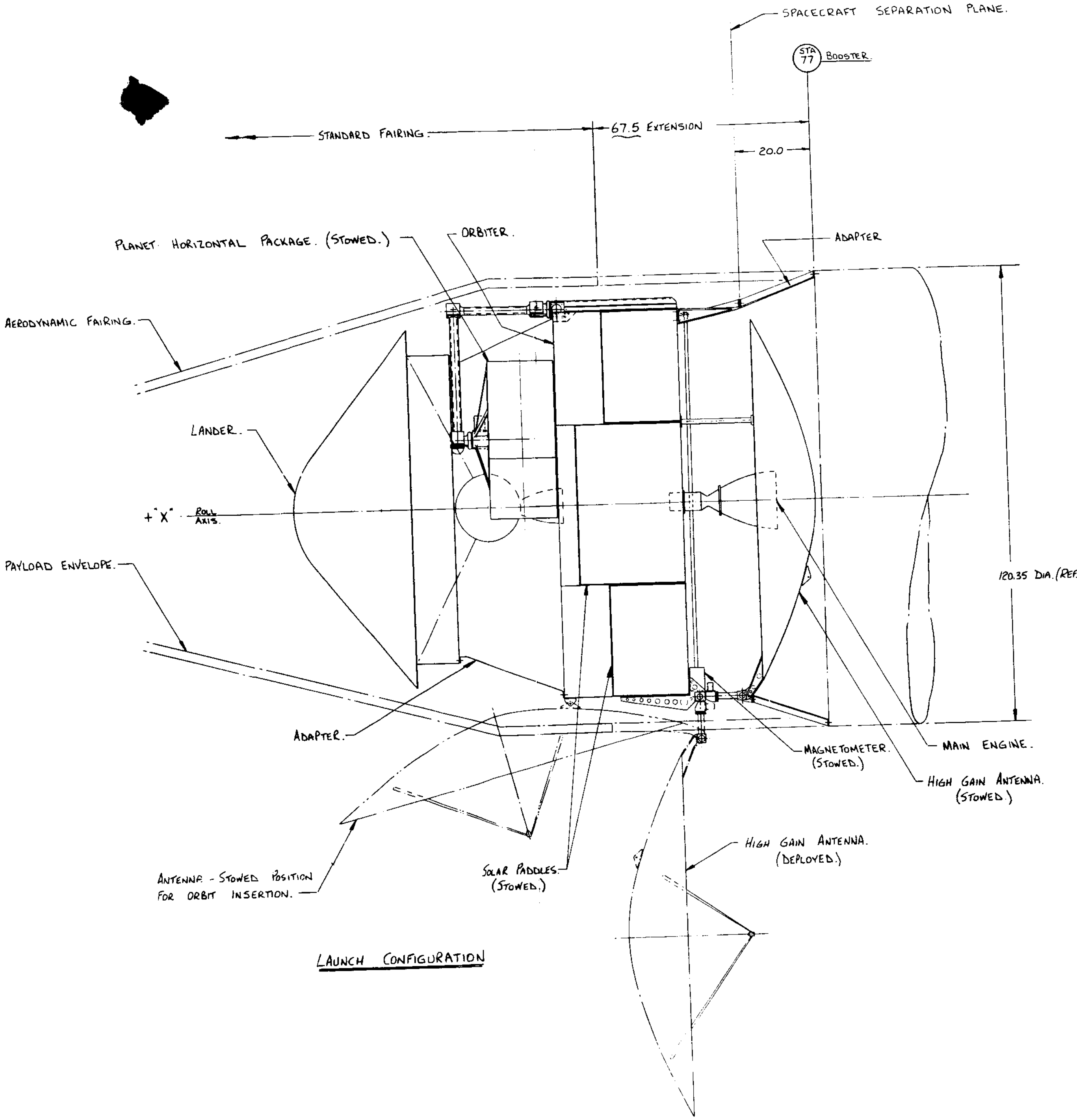
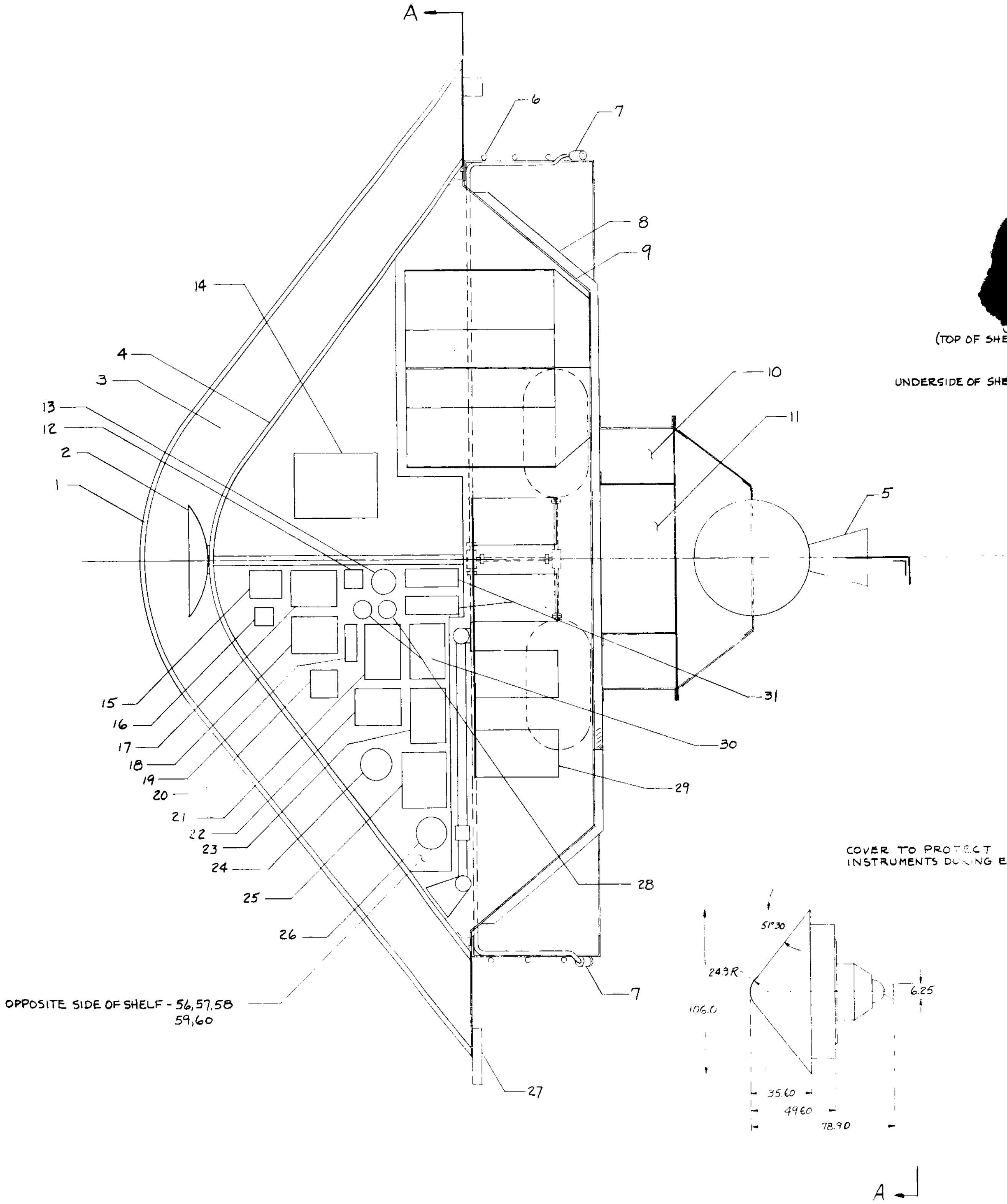


Figure 3.4-2. Orbiter/Lander System Configuration


A
1
2
3
4
5
6
7
8
9
10
11
12
13
14
15
16
17
18
19
20
21
22
23
24
25
26
27
28
29
30
31
7
OPPOSITE SIDE OF SHELF - 56,57,58
59,60
(TOP OF
UNDERSIDE OF
COVER TO PROTECT
INSTRUMENTS
51°30
24.3R
106.0
6.25
35.60
49.60
78.90
A

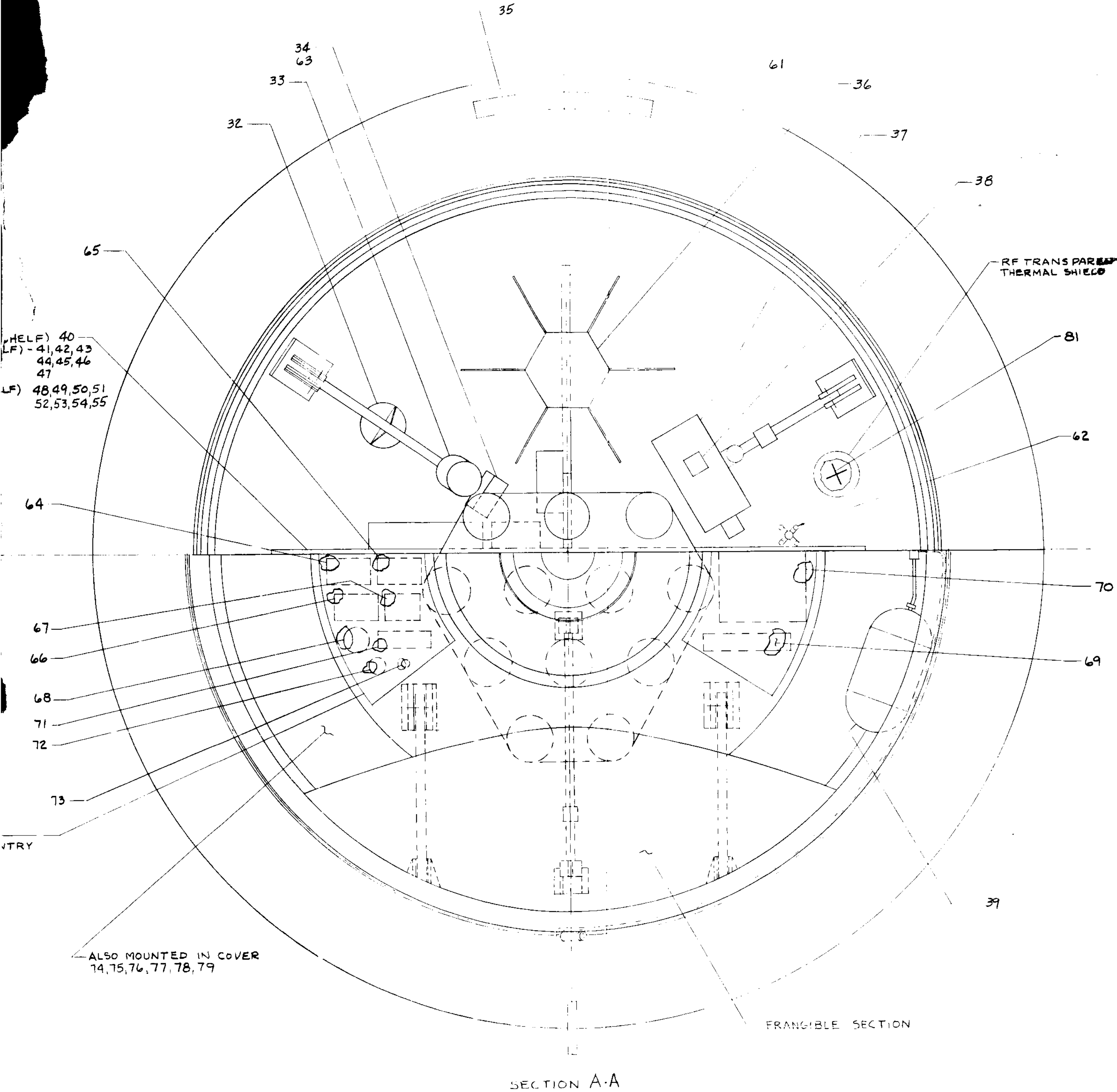
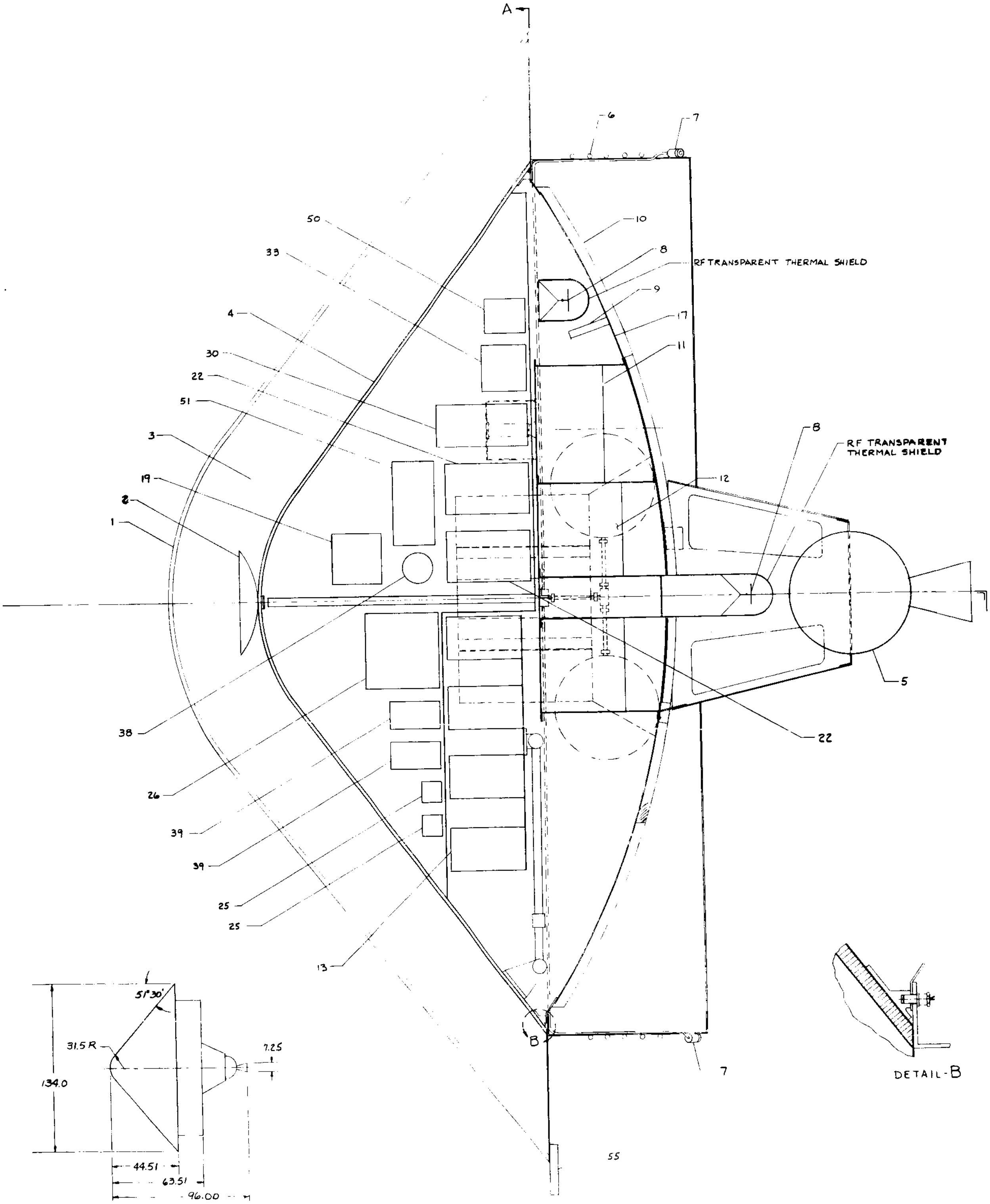
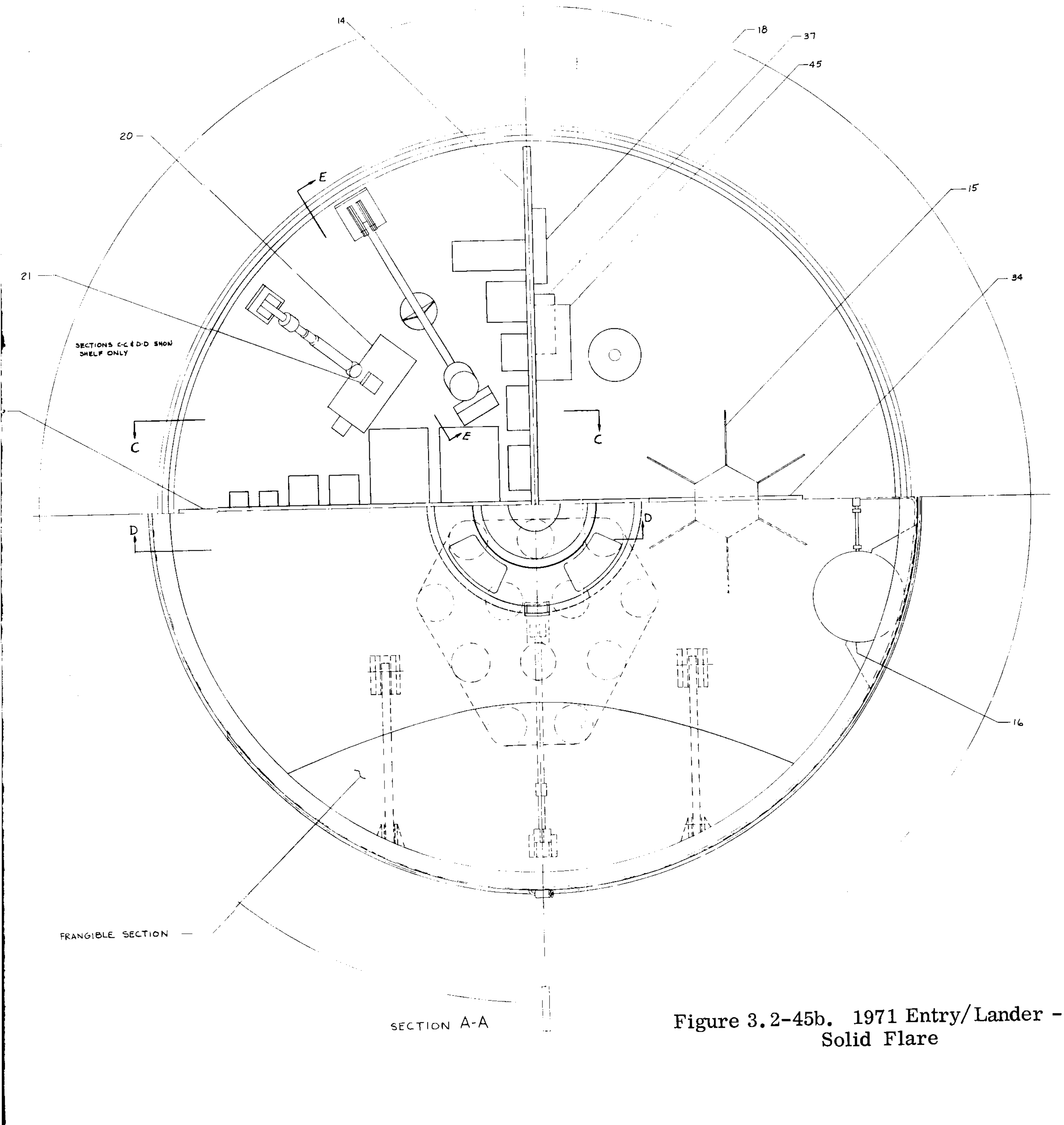


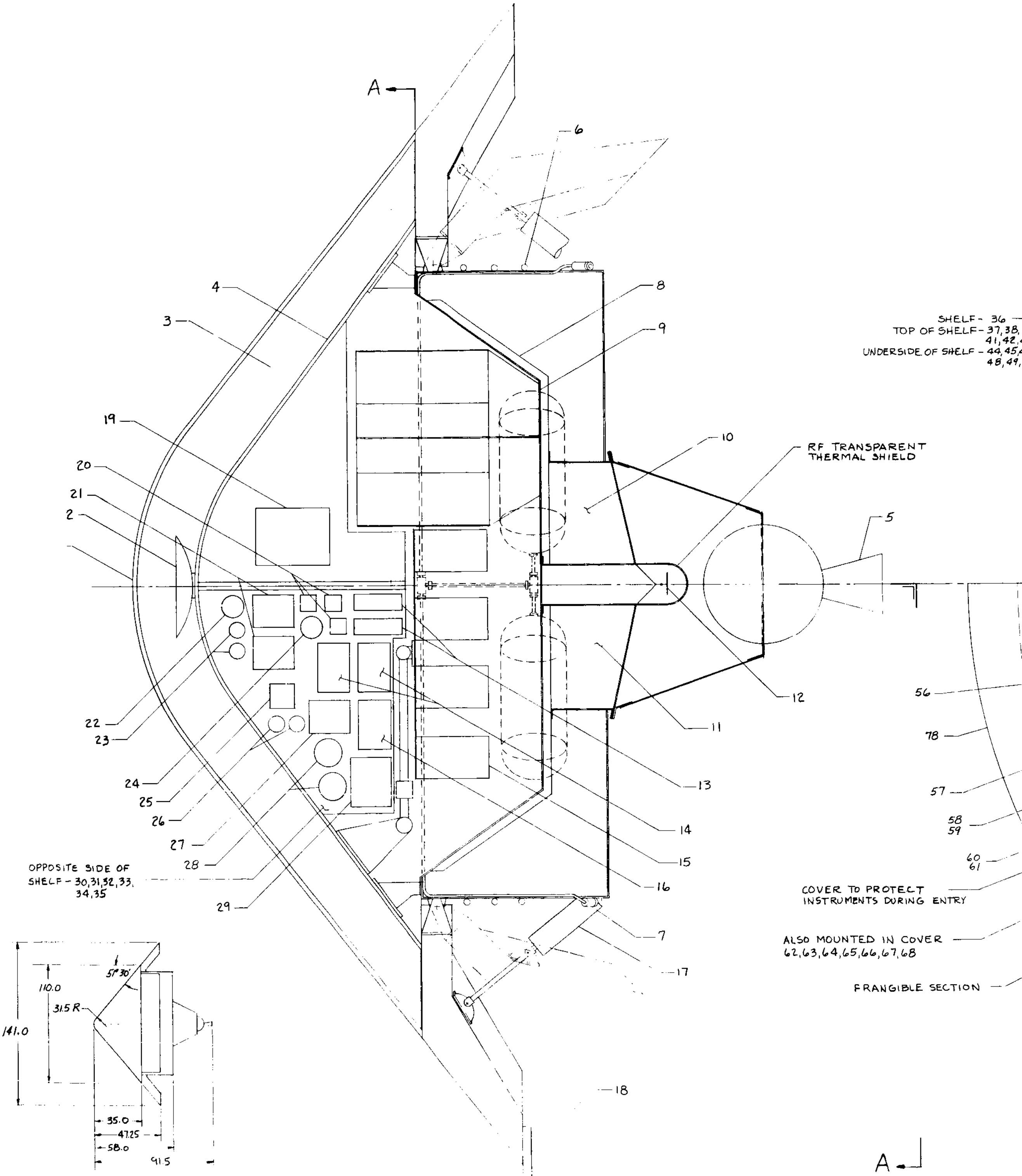
Figure 3.4-1. Entry/Lander for Orbite
Design


A
6
7
10
8
RF TRANSPARENT THERMAL SHIELD
9
17
11
50
33
4
30
22
51
3
19
2
1
8
RF TRANSPARENT
THERMAL SHIELD
12
5
22
38
26
39
39
25
25
13
51°30'
31.5 R
134.0
7.25
44.51
63.51
96.00
B
7
DETAIL-B



SECTION A-A

Figure 3.2-45b. 1971 Entry/Lander - Solid Flare


A
A
SHELF - 36
TOP OF SHELF - 37,38,
41,42,
UNDERSIDE OF SHELF - 44,45,
48,49,
RF TRANSPARENT
THERMAL SHIELD
COVER TO PROTECT
INSTRUMENTS DURING ENTRY
ALSO MOUNTED IN COVER
62,63,64,65,66,67,68
FRANGIBLE SECTION
OPPOSITE SIDE OF
SHELF - 30,31,32,33,
34,35
51°30'
110.0
31.5 R
141.0
35.0
47.25
58.0
91.5

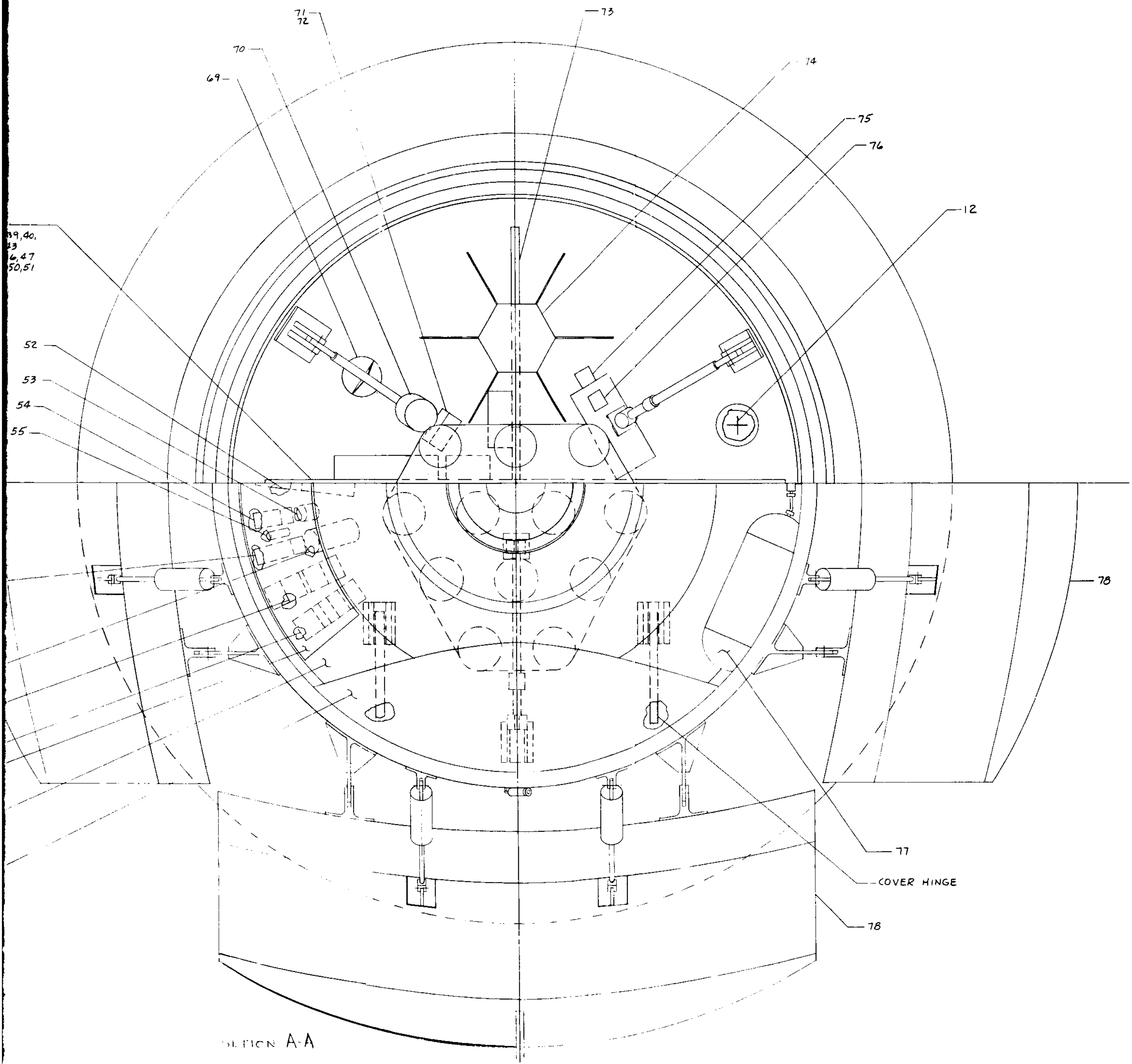


Figure 3.2-49. 1971 Entry/Lander - Extensible Flare Design

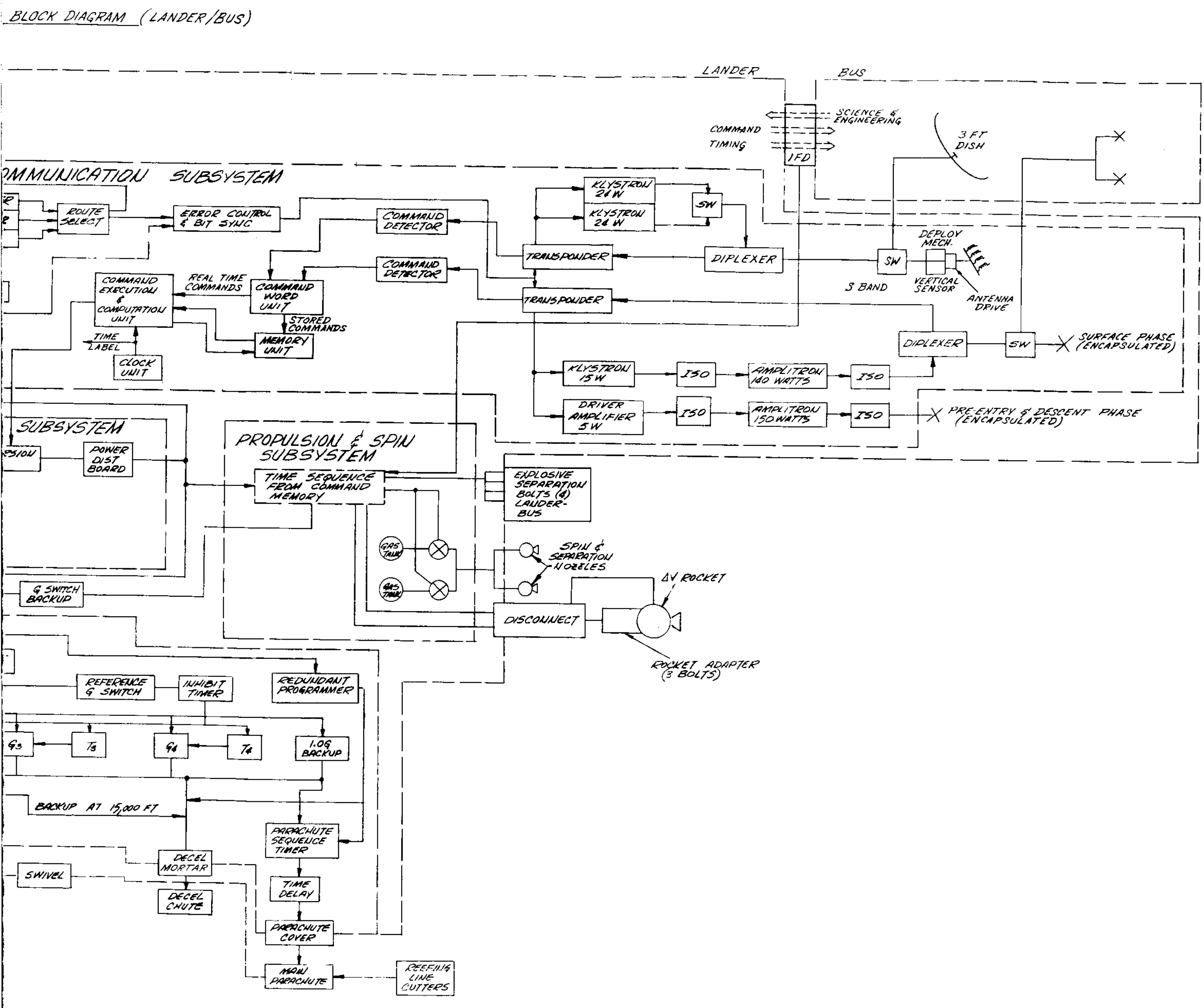


Figure 3.2-47. Entry/Lander System Block Diagram

LANDER SYSTEM