The present report is comprised of four volumes:

| | |
|---|---|
| **Volume I** | Sensitivity Analysis |
| **Volume II** | Experiment Requirements |
| **Volume III** | Unmanned Spacecraft Design |
| **Volume IV** | Summary |

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# 1. SUMMARY

The present report contains a description of unmanned spacecraft designed to evaluate the Martian and Cismartian environments in preparation for subsequent manned Mars landing missions. The design concepts include orbiter/bus (or flyby) systems, entry capsules and survivable landers. In addition, a high resolution photo mapper package is defined. Payloads were selected on the basis of detailed requirements analyses reported in Volumes I and II of the present report (see Foreword).

The evaluation of the Martian environment requires monitoring of the interplanetary fields and particles, measurement of the atmosphere of Mars, measurement of the surface properties in potential landing sites, and requires a capability to perform general mapping augmented by detailed photo analysis over areas of interest on the surface ( the photo reconaissance can be augmented with TV based on lander systems). Hence, a lander is mandatory plus orbiter photo mappers of high resolution for evaluation of the Mars environment.

Preliminarydesigns indicate that an orbiter bus weighs approximately 1850 pounds , a survivable lander 700 pounds, and a photo mapper package for use aboard an orbiter approximately 400 pounds. Eighty pounds of experiments are included or can be supported on the bus/orbiter, and 56 pounds on the landers; these scientific payloads include all priority experiments as defined during prior phases of the study.

Preliminary designs indicate that a highly effective spacecraft system can be launched by an Atlas Centaur plus kick stage booster. It is possible using this launch system to inject an orbiter plus lander plus high resolution mapper packages in 1971 and 1973; however, in 1975 the payload capability is sufficient to launch an orbiter only, with intermediate resolution mapper capability. The Saturn IB Centaur, can launch the orbiter-mapper plus lander in all years and may be required in 1975 depending upon the results of the earlier missions.

The orbiter bus design selected for use with the Atlas Centaur plus kick stage boost system uses an Earth pointing mode with body fixed antenna and solar arrays, the latter having somewhat reduced output because of the orientation mode throughout the duration of the mission. A communication capability of 4,000 bits per second is available in the orbiter bus system with a 10-watt transmitter and a 9-foot diameter antenna. A solid retro propulsion system was used to place the spacecraft in a 2,000 by 20,000 km orbit about Mars, which was selected to give a 50-year lifetime, based on noncontamination constraints. A monopropellant midcourse and thrust vector control system were used. It was found that the radioisotope thermal electric power generators were not available in adequate quantities for the orbiter/bus power system; the use of solar cells was indicated for this reason. The total weight of the orbiter bus system is 1850 pounds including retro for the orbiting maneuver. Approximately 80 pounds of experiments are supported by the orbiter/bus system.

The lander, which was designed for the Model 3 (10 millibar) atmosphere, uses an Apollo-shaped shield, two parachutes (a supersonic and subsonic unit), and is designed for impact and lateral velocities of 50 fps, with 100 earth g's impact. Selective elements within the lander system may require additional shock attenuation. It was found that the drift velocities cannot be removed without the use of elaborate sensing equipment and velocity removable techniques. The lander is capable of self-righting after touchdown and subsequent tumbling. Approximately 55 pounds of experiments are supported by the lander, with data being relayed back to the orbiter by a 200 mc communication subsystem. A television system is incorporated for mapping during subsonic descent through the atmosphere, and for photoanalysis of the surrounding areas after touchdown and subsequent activation of the lander system. The television pictures acquired during descent are stored on a tape recorder for subsequent transmission to the orbiter spacecraft.

## 2. INTRODUCTION

The present report contains a description of unmanned spacecraft systems designed to evaluate the Martian and Cismartian environments in preparation for subsequent manned Mars landing missions. The design concepts include orbiter/bus (or flyby) systems, entry capsules and survivable landers. In addition a high resolution photo mapper package is defined.

Prior phases of the present study contract were devoted to analyzing the sensitivity of the manned Mars system to the Martian and Cismartian environment, and on the basis of the results establishing priorities for the measurement of the environmental factors. The results of these foregoing analyses were presented in the form of experiment lists for each of the unmanned spacecraft systems, orbiter, entry capsule and landers. Within each experiment list the priority experiments of importance to the design of the manned mission systems were indicated.

The individual experiments were defined in sufficient detail so that an indication of the scientific payload weight for the unmanned systems could be determined. The results indicated that the scientific payload for an orbiter system would require a payload weight capability of approximately 80 pounds, 18 pounds for a nonsurvivable entry capsule to probe the Mars atmosphere, and 56 pounds for survivable landers. On the basis of these results specific spacecraft designs were evolved to accomplish the required mission goals.

Having established specific spacecraft designs with the related scientific payloads, the launch capabilities of several boost systems were examined to select the most effective Earth boost system for each of the precursor spacecraft. The boost systems included Atlas Centaur, Atlas Centaur with a high energy kick stage, and the Saturn IB Centaur system. A highly useful launch system is the Atlas Centaur plus a high energy kick stage, which is capable of injecting an orbiter plus lander with a high resolution mapping telescope during the favorable mission opportunities of 1971 and 1973.

In general, the usefulness of a single lander mission seemed open to question from the standpoint of landing site selection. Surface property measurements at a single point may not be representative of general conditións within a proposed exploration area, and detailed television coverage of one landing site may have limited applicability to other sites under consideration. Two means of extrapolating the data from one lander site seemed reasonable. First, the TV scan around the lander base would aid in "extrapolating" the data to a wider area; deployment of a miniature surface rover, properly instrumented to measure surface properties, also offer possibilities for extending the usefulness of a single lander system. However, these modes at best yield limited data, and it seems assured that heavy reliance will be placed on extensive mapping (general and detailed) from a Mars orbiter photographer. This mapper spacecraft must be equipped with high resolution telescope systems capable of achieving surface resolutions of a few feet. Based upon the results of coarse resolution mapping, also performed from an orbiter, selected areas on the surface of Mars could be selected for detailed high resolution reconaissance for landing site selection.

The unmanned spacecraft described herein can accomplish all priority experiments required for the necessary evaluation of the Martian and Cis-martian environment in preparation for the eventual manned landing missions. Additional design studies should be carried out to verify the usefulness of these types of spacecraft for the mission goals considered herein.

# 3. MISSION EXPERIMENTS

A complete set of experiment programs for the measurement of Mars environmental factors by unmanned precursor missions is analyzed and described in Volume II. A brief summary of these experiments is presented in this section in order to establish a continuity between the experiments and the resulting unmanned spacecraft designs.

In order to define and select the instruments or experiment packages which would be required, it was first necessary to determine the interactions of environmental factors and the effects due to uncertainties of these factors on the manned Mars mission systems. A reference manned mission system design was selected based upon a design study conducted previously for the NASA Ames Research Center[1], and utilizing the Mars orbiting rendezvous mode, analogous to that being planned for the Apollo lunar program. The design was established for the opposition-class missions of approximately 14 months duration, and was capable of either aerodynamic or retro capture at Mars; chemical and nuclear propulsion systems were considered.

A numerical rating system was used to aid in establishing priorities for the measurements of the Martian and Cismartian environment factors. Each environmental factor was rated in each of the categories of design feasibility, system weight, mission operations, and system development.

The results of this analysis were used to establish a list of priority experiments designed to measure the environmental factors which are of major importance to the manned Mars mission system.

The various mission modes considered for the unmanned precursor flights include the following:

- Flyby
- Flyby plus capsule/lander
- Orbiter
- Orbiter plus capsule/lander

1. R. L. Sohn, ed., "Study of a Manned Mars Landing and Return Mission." TRW/STL Report 8572-6011-RU000, 24 March 1964. Contract NAS 2-1409

For present purposes the direct lander systems were not considered, although these may be advantageous when considered as one phase in a sequence of missions.

The various mission options were reviewed to indicate the most effective means of accomplishing the experiment goals. It was noted that a majority of the high-priority experiments could be accomplished with fairly small experiment payloads, permitting many of the experiments to be accomplished with the Atlas-based launch systems. Because of small payload weights many of the high-priority experiments could be repeated on many missions to Mars. For example, measurements of the solar cosmic radiation environment are repeated on all missions to the planet, thus, building up the necessary volume of data to develop statistically meaningful models of the solar cosmic radiation environment.

The summary list of experiments is presented in Table 3.1 in the order of the priorities established on the sensitivity analysis of the manned Mars mission. The experiments are grouped by mission mode. The weights for each experiment are tabulated, and the accumulated payload weights obtained by summing the experiments in the order of their priorities.

It is possible to accommodate many of the high-priority experiments with a moderate total payload weight, such that it is possible to accomplish these experiments with an Atlas Agena or Atlas Centaur launch system. The Saturn IB launch system is required only for heavy landers. The high-priority experiments on the interplanetary bus/orbiter systems constitute a basic experiment package that probably can be incorporated on all missions on Mars.

Table 3.1 Preliminary Scientific Payload List
(Note: Experiment Number refers to item in Table 3.4, Vol. II)

BUS/ORBITER

| | Experiment No. (See Note) | Rating (1→3) | Weight (lbs) | Weight Summation (lbs) |
|---|---|---|---|---|
| Particle Flux (High Energy) | 7 | 3 | 10 | 10 |
| Particle Flux | 8* | 3 | 2.5 | 12.5 |
| Ion Chamber | 9* | 3 | 1.3 | 13.8 |
| Trapped Radiation Detector | 10* | 3 | 4 | 18 |
| Magnetometer | 11* | 3 | 5 | 23 |
| Meteoroid Environ. (TRW) | 19 | 3 | 5 | 28 |
| Micrometeoroid Environ. | 17* | 3 | 8 | 36 |
| TV | 20* | 3 | 17 | 53 |
| UV Spectrometer | 4 | 2 | 22 | 75 |
| Ionosphere Exp. | 15 | 2 | 3 | 78 |
| IR Radiometer | 23 | 1 | 3 | 81 |
| IR Spectrometer | 5 | 1 | 29 | 110 |

*Mariner

DESCENT CAPSULE

| | Experiment No. | Rating | Weight (lbs) | Weight Summation (lbs) |
|---|---|---|---|---|
| Accelerometers | 1 | 3 | 1.0 | 1.0 |
| Pressure, Temp. | 2 | 3 | 0.5 | 1.5 |
| Gas Composition | 3 | 3 | 2.0 | 3.5 |
| TV | 25 | 3 | 15 | 18.5 |

LANDER

| | Experiment No. | Rating | Weight (lbs) | Weight Summation (lbs) |
|---|---|---|---|---|
| Solar Cosmic Radiation | 12 | 3 | 1.3 | 1.3 |
| Cell Growth | 30 | 3 | 4 | 4.3 |
| Turbidity & PH | 31 | 3 | 4 | 8.3 |
| TV | 25 | 3 | 17 | 25.3 |
| Mass Spectrometer | 6 | 2 | 6 | 31.3 |
| Anemometer | 21 | 2 | 1 | 32.3 |
| UV Detector | 14 | 2 | 0.5 | 32.8 |
| Surface [illegible]ation | 16 | 2 | 1.5 | 34.3 |
| Surface Properties | 26 | 2 | 13 | 47.3 |
| Seismometer | 27 | 2 | 8 | 55.3 |
| Visible Intensity | 13 | 1 | 0.5 | 55.8 |
| X-Ray Diffractometer | 28 | 1 | 10 | 65.8 |
| Core & Mill | 29 | | 30 | 95.8 |

# 4. LAUNCH VEHICLE PERFORMANCE

## 4.1 Transfer Trajectory Characteristics

Launch vehicle performance for one-way transfers between Earth and Mars were determined as a function of launch date for the years 1971, 1973 and 1975. The launch constraints included were: 30-day launch opportunity, minimum daily launch window of 30 minutes, and launch azimuths of 90° to 114° from ETR.

Reference 4.1 presents the characteristics of ballistic interplanetary trajectories to Mars for the launch dates considered. The analytical model used consists of (1) an escape hyperbola near Earth; (2) heliocentric transfer to Mars; and (3) terminal hyperbolic motion near Mars. Transfer trajectory characteristics based on this model are given in Figures 4.1 to 4.3.

The method utilized was to minimize the injection velocity (maximize launch vehicle payload capability) required for the transfer trajectory, observing the three launch constraints previously mentioned. The flight time and the asymptotic speed with respect to Mars are then determined. It should be noted that while this analysis maximizes launch vehicle payload capability, terminal conditions for Mars orbiters might dictate slightly different launch dates, particularly for the unfavorable years of 1973 and 1975. For an orbiter mission in 1975, a 30-day launch hold is very difficult to accommodate without increasing the eccentricity of the Mars parking orbit.

The minimum hyperbolic-excess injection velocity requirements were determined from Reference 4.1 observing the constraints on the declination of the geocentric asymptote as determined from Reference 4.2 to insure that a minimum daily launch window of 30 minutes is maintained. Once the

4.1 V. C. Clarke, W. E. Bollman, R. Y. Roth, W. J. Scholey, "Design Parameters for Ballistic Interplanetary Trajectories, Part I. One-way Transfers to Mars and Venus," JPL Technical Report No. 32-77, 16 January 1963.

4.2 R. L. Wolpert, "Interplanetary Launch Windows," 9861.5-199, 16 April 1963.

minimum injection velocities as a function of launch date were determined, transfer times and hyperbolic approach speed with respect to Mars were determined.

Figure 4.1 presents minimum hyperbolic-excess injection velocity, transfer time, and hyperbolic approach speed with respect to Mars as a function of launch date for the year 1971. A hyperbolic-excess injection velocity of approximately 9850 feet per second would be required to insure a 30-day launch opportunity. Similar data are given in Figure 4.2 for the 1973 launch opportunity.

Figure 4.3 presents trajectory data for the year 1975. Due to geometrical relationships, the minimum hyperbolic-excess injection velocities cannot be obtained within the launch constraints discussed. Consequently, the hyperbolic-excess injection velocity presented are for the minimum 30-minute launch window. A hyperbolic-excess injection velocity of approximately 17,750 feet per second is required for the 30-day launch opportunity. This data is based upon Type I trajectories ($< 180^{\circ}$). Type II trajectories have lower launch velocity requirements, however, Type I trajectories were used because of shorter flight times and lower sensitivity to injection errors.

## 4.2 Launch Vehicle Payload Performance

The payload capability for several launch vehicles has been determined for the hyperbolic excess velocity requirements presented in the above section for the launch opportunities of 1971, 1973 and 1975. All vehicles were assumed to depart from a 100 nmi parking orbit. The payload values are based on a 30-day launch period, a 30-minute launch delay, and include a 3 $\sigma$ flight performance reserve.

The high energy kick stage added to the Atlas Centaur vehicle has the following characteristics (the velocity increment was added impulsively, without gravity losses: [4.3]

4.3 D. R. Pence, "HEUS Weight Comparisons," TRW Memorandum 9862.7-285, 26 July 1965.

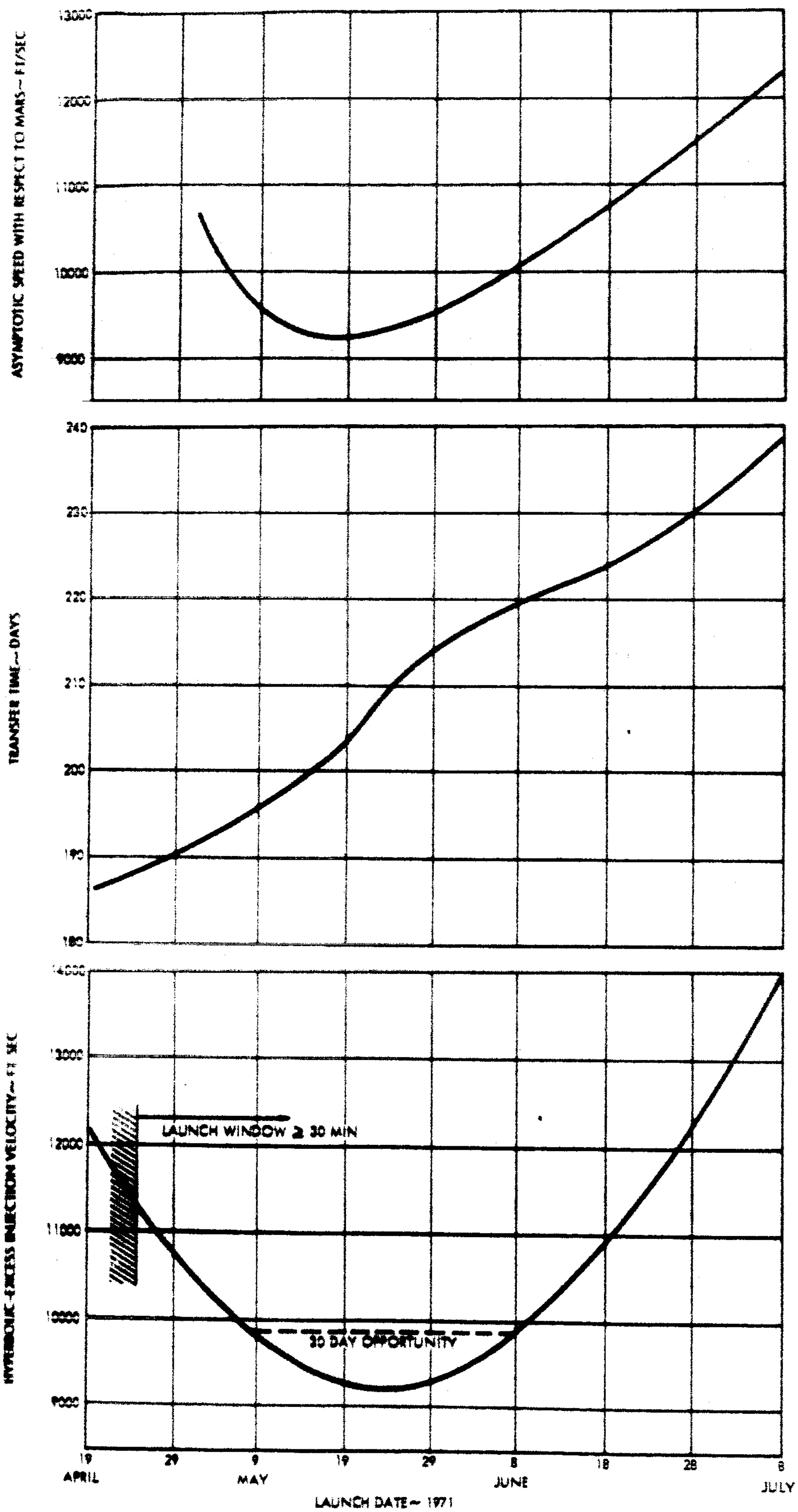



Figure 4.1 Transfer Trajectory Characteristics (1971)

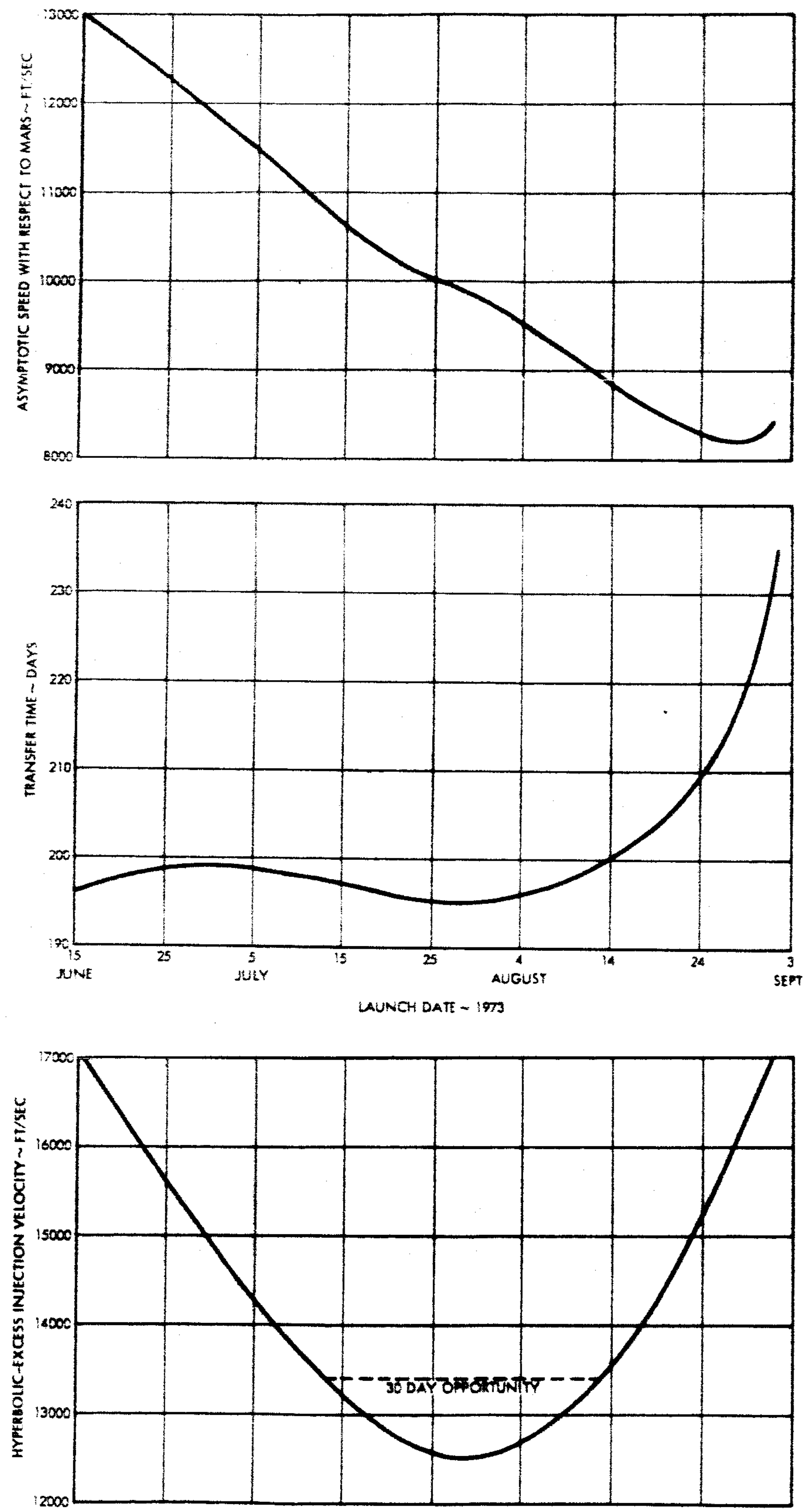



Figure 4.2 Transfer Trajectory Characteristics (1973)

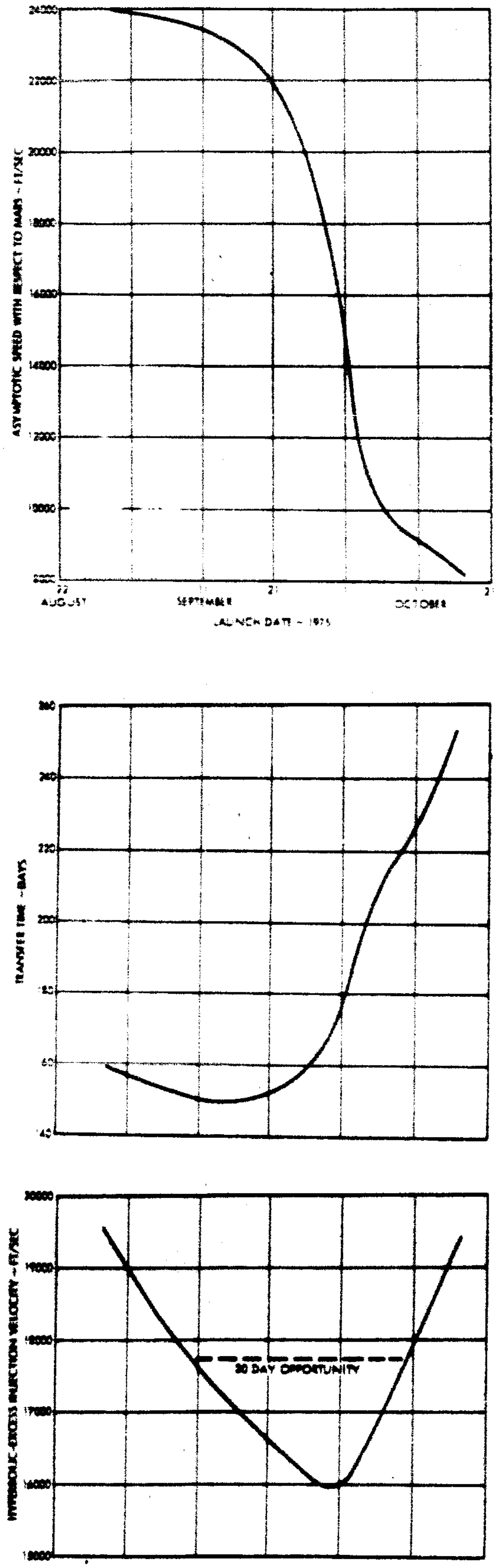



Figure 4.3 Transfer Trajectory Characteristics (1975)

| | |
|---|---|
| Stage Weight | 7267 lbs |
| Propellant Weight | 6104 lbs |
| Specific Impulse | 440 sec |
| Propellants | $H_2$ - $F_2$ |

The kick stage was not used in establishing the initial parking orbit. A launch vehicle payload summary is given below.

Table 4.1 Launch Vehicle Payloads (lbs)

| Booster | 1971 | 1973 | 1975 | |
|---|---|---|---|---|
| Atlas-Centaur | 1650 | 1200 | 560 | |
| Atlas SLV (Advn)-Centaur | 2250 | 1700 | 910 | |
| Atlas-Centaur-Kick Stage | 3650 | 3200 | 2550 | (2080)* |
| SIB-Centaur | 9900 | 8600 | 6600 | |

*Launch dates shifted to maximize weight in Mars orbit (26 day launch window)

## 4.3 Mars Arrival Velocities

Mars arrival velocities are shown in Figure 4.4 for the launch windows shown in Figures 4.1 to 4.3. Sufficient retro propulsion has been included in the orbiter vehicle designs to place the spacecraft in a nominal 2,000 x 20,000 km orbit at Mars.

In 1971 and 1973, very nearly maximum payload is placed in Mars orbit by maximizing the launch payload capability. In 1975, however, Mars arrival velocities become exorbinant under these conditions and it becomes necessary to shift the launch window to obtain more favorable arrival conditions. Launch payload performance is degraded as a result, but retro propulsion requirements are reduced markedly. With an Atlas-Centaur-Kick Stage launch system, a nominal orbiter can be placed in a 2,000 x 20,000 km orbit about Mars, with sufficient payload remaining to accommodate an entry capsule (nonsurvivable).

A summary of the payload capabilities for the various launch opportunities is given in Section 7.

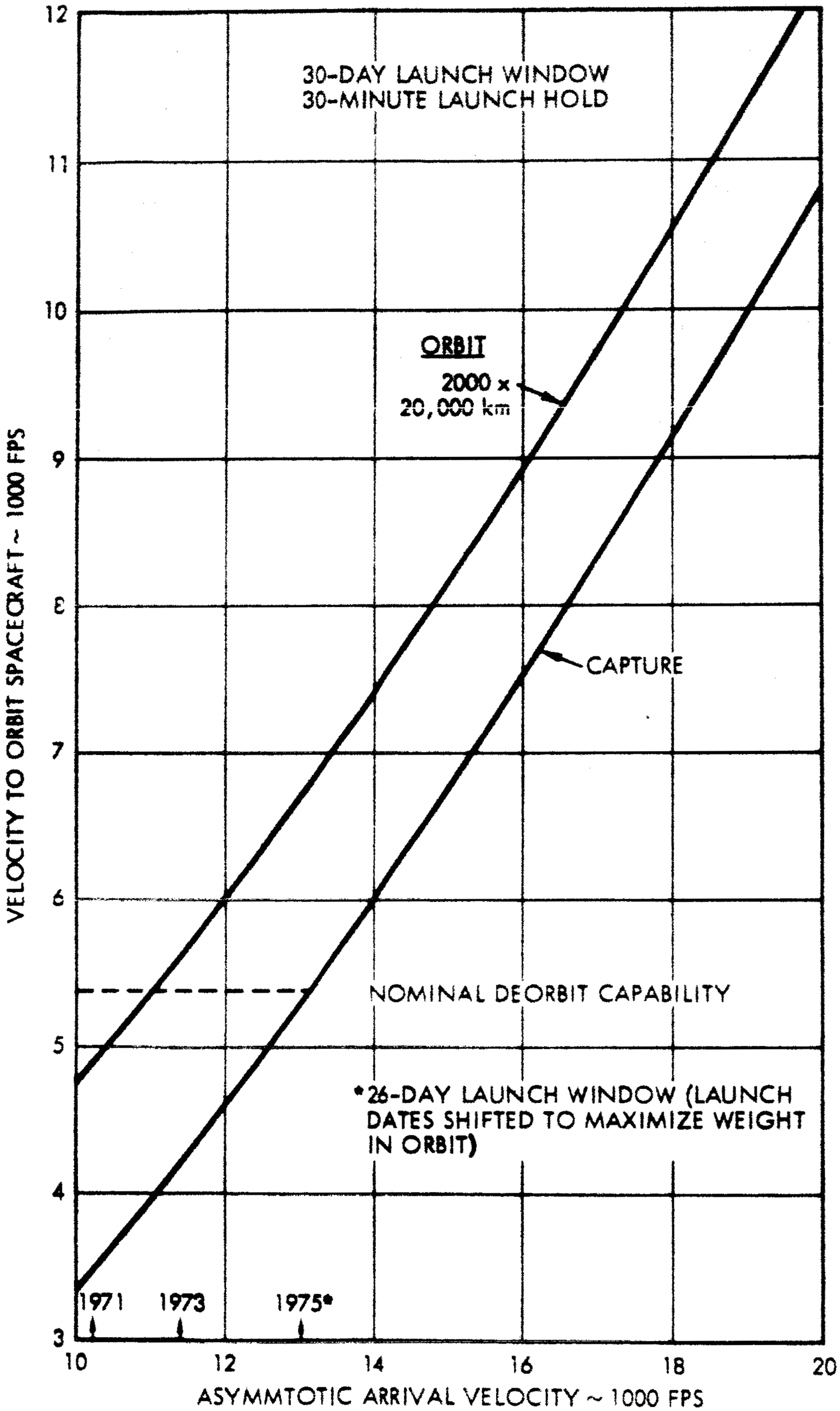



Figure 4.4 Velocity Required to Orbit Spacecraft at Mars

# 5. SATURN 1B/CENTAUR SPACECRAFT CONCEPTS

Two Saturn 1B/Centaur with 260-inch diameter shroud spacecraft concepts have been studied; one using solar power, the other using radioisotope thermoelectric generators (RTG's). The solar powered version explores the conventional (Mariner) concept and the RTG powered version explores the possible advantages associated with the removal of the solar pointing constraint.

## 5.1 Summary of Configuration

### 5.1.1 Solar Powered Configuration

The key functions which must be performed by any spacecraft include those necessary to assure that the entry vehicle arrives at Mars within a suitable entry corridor. (A desirable function is to relay the entry data from the entry vehicle to the spacecraft and then to earth, giving a higher bandwidth than is possible with a direct entry vehicle to earth link, and without requiring a high gain antenna on the entry vehicle or, alternatively, data storage with post landing survival data transmission.) The second function is to perform imaging, i. e., mapping experiments in orbit, and the third is to perform other experiments in orbit. Within these functional constraints Figure 5.1 shows a possible configuration.

This example spacecraft is nominally pointed toward the sun to maximize solar cell power. It mounts a high gain antenna which is capable of being pointed at the earth for an arbitrary spacecraft roll angle about the sun line. A single gimbal then allows the imaging package to be pointed at Mars when in Mars' orbit. This particular concept is closely related to the current Mariner concept, but the requirements for long duration orbital operations and for the delivery of entry vehicle results in the requirement for a more versatile spacecraft.

The basic ground rule that has been used is that the spacecraft should be versatile in the performance of its mission. This versatility takes the following forms:

1) No restrictions on the allowable orbit about Mars are inherent in the spacecraft except for propulsion $\Delta V$ limitations.

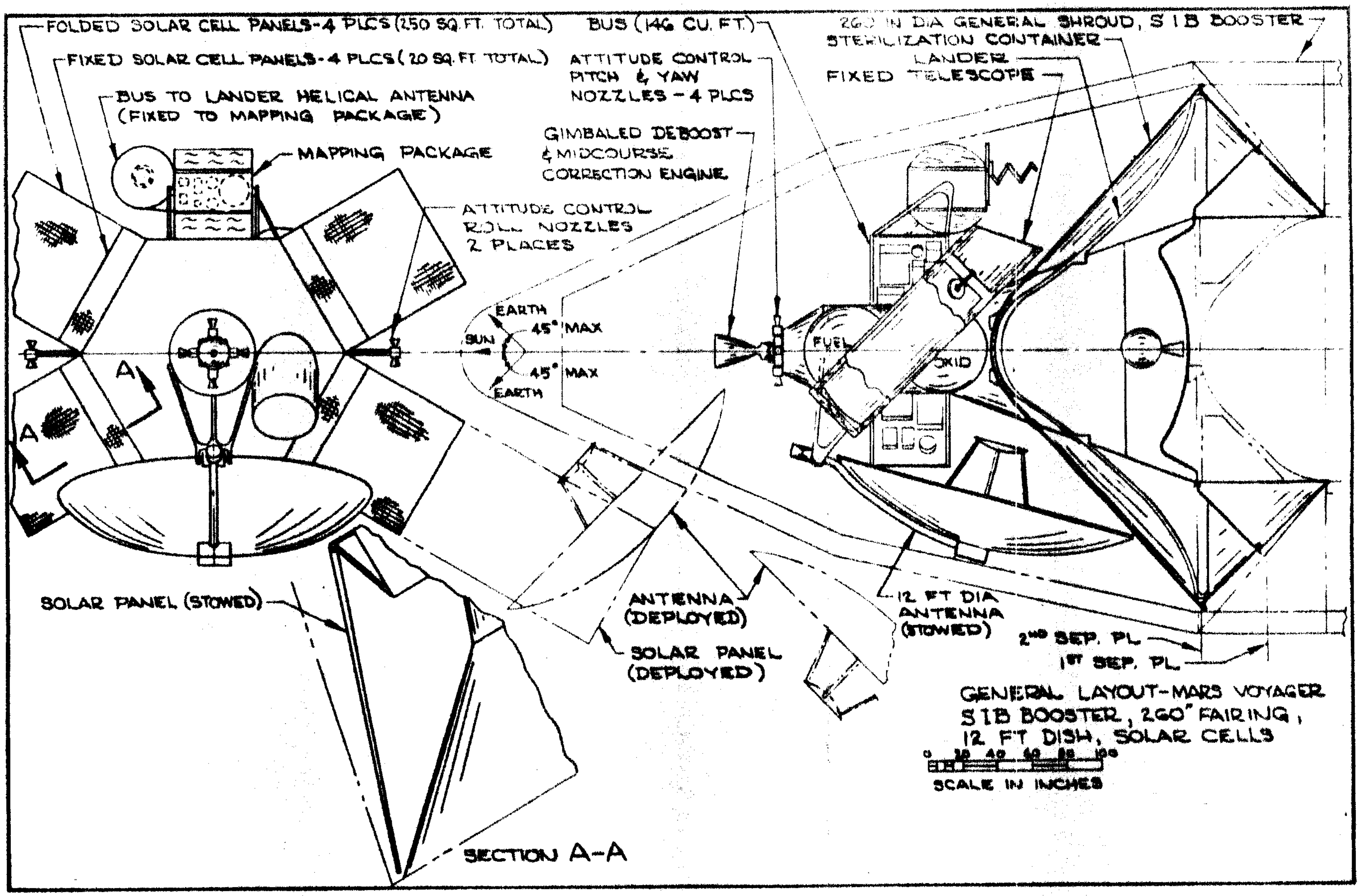



Figure 5.1 General Layout - Mars Voyager SIB Booster, 260-inch Fairing, 12-ft. Dish, Solar Cells

2) The entry vehicle should be deliverable to any spot on Mars from any approach trajectory, even including partial failure mode approach trajectories (i. e., second midcourse failed).

3) Maximum versatility should be available in the use of the imaging package (i. e., pointing capability at any visible portion of Mars from any point in any orbit).

4) Orientation maneuver capability is available to fire midcourse or deboost in any desired direction.

5) The spacecraft should be able to survive Martian eclipse (even though such an eclipse might not occur until after six months of orbital operations for selected orbits).

The entry vehicle should have the maximum convenient drag coefficient, consistent with entry system weight. This allows a maximum entry corridor for a given weight of entry vehicle or, conversely, a maximum weight of entry vehicle for a given corridor. This led to the selection of the 260-inch shroud for the Saturn IB as opposed to the alternate 154-inch shroud, and to positioning the entry vehicle at the base of the shroud. Subsidiary ground rules are that the entry vehicle should be spin-stabilized and propelled by solid rocket for separation from the spacecraft, thus minimizing the number of components required in the entry vehicle. It is also assumed that the entry vehicle will be landed by a parachute plus retros. This latter assumption cannot, of course, be verified at the present time but seems to lead to the simplest lander concept since it does not require doppler lateral velocity sensors, or lander rockets to control lateral velocity. Another assumption is that the landing vehicle will be enclosed in a sterile container which will be opened late in the flight, thus leading to the minimum contamination interface between spacecraft and lander and the maximum possibility of insuring adequate sterilization.

The largest solid dish (12-ft diameter) which could be conveniently accommodated in this configuration has been shown. It is possible to consider larger dishes which can be unfolded or otherwise erected; however, consideration of such dishes has been deferred to Case 2) (the RTG powered version) where it appears to be much more appropriate. The dish is double gimbaled to allow earth coverage when the vehicle is rolled from the nominal transit sun-Canopus orientation. It was felt that it was easier to double gimbal the dish and single gimbal the small imaging package than to single gimbal the dish and double gimbal the imaging package. With the capability of vehicle roll to allow the single gimbaled imaging package to point at any visible spot on Mars, it was felt that a high gain relay link antenna between the spacecraft and the landing vehicle could also be incorporated on this package.

The concept chosen has deployable solar panels but these panels are not gimbaled. It is, of course, possible to gimbal the panels and point the vehicle in some other direction but no concept has yet been generated which makes such an option appear more attractive.

It is assumed that the imaging subsystem represents one of the major experiments on the orbiting spacecraft and that much could be sacrificed in order to attain high resolution and adequate coverage for these subsystems. It is assumed that imaging under favorable illumination angles and at various seasons of as much of Mars as possible is desired, and it is also assumed that, within this broad coverage, nested high resolution coverage is also desired, with the value of the mission being to a significant extent related to the resolution limit. This combination of broad coverage and high resolution coverage led to the concept of two imaging packages. The first is a versatile (single gimbal) imaging system having several telescopes operating in several wavelengths, giving both broad and moderate coverage (low and moderate resolution). The second package consists of the largest telescope which can conveniently be mounted. Because of its size, this large telescope is shown fixed as part of the spacecraft body and the angle between the telescope axis and the spacecraft body was chosen to provide suitable suborbital illumination for those regions in which the telescope could be pointed at the center of Mars.

The question of whether film or TV should be used has not been fully settled, but it is currently contemplated that the small imaging package would use TV and that the large telescope might use film if it turns out to be advantageous over TV.

A rollable vehicle with a single gimbaled imaging package allows that package to look down toward the center of Mars at almost any time in the orbit, and the 2-gimbaled antenna gives earth coverage at any time. This concept also allows the fixed telescope to look down when the illumination is favorable, i. e., 45 degrees. The major weakness of this design concept lies in the in-line reliability requirement for the antenna drive. Further consideration must be given to failure modes and also to possible simpler systems. There is a minor problem associated with the cg shift due to the movable antenna.

Accommodating all of the required systems within the shroud contour represents one of the major problems in the configuration design. One aspect of this is the competing desire to measure several items along the vehicle's centerline. Thus it is desirable to have the landing vehicle, the landing vehicle separation propulsion, the midcourse and deboost fuel tanks, and the deboost engine all on-axis. The configuration shown represents one of the ways in which all of these constraints can be met. Probably the weakest of these constraints is the requirement for the fuel tanks to be on axis; however, this is desirable and seems achievable.

The entry vehicle heat shield shape has been selected from the type being studied by Langley. This shape has the advantages of high aerodynamic stability and high drag coefficient, but requires that the entry vehicle be mounted upside down on the spacecraft so that the skirt can be in the region of maximum shroud diameter. As is discussed later, this introduces an additional propulsion maneuver following separation of the landing the vehicle so that it does not run into the spacecraft when the landing vehicle separation propulsion is fired. Further study needs to be given to the necessity of this maximum diameter entry vehicle or, conversely, to alternate shapes with high drag coeffieients. Actually, the shape chosen has about the highest drag coefficiency achievable within the overall system constraints.

Thermal control represents another major aspect of configuration concept. In particular, the entry vehicle is expected to be powered by radioisotope thermonuclear generators which, because of their low efficiency, have an extremely high thermal output. This leads to another reason for inverting the entry vehicle in relation to the spacecraft since then the RTG units radiate toward the Centaur instead of toward the spacecraft and minimize the transit thermal control problem. Another thermal control aspect was the desire to have the midcourse and deboost engine located in the sun so that fuel freezing problems would be eliminated.

Lastly, the spacecraft body is so designed that the sides of the spacecraft are shielded from the sun (up to 20-deg pointing error from the sun), giving a convenient region on which to mount electronic components. The large telescope would communicate with the thermally controlled spacecraft body but would be structurally isolated from the spacecraft and would have subsidiary thermal control heaters to keep its structure at a very uniform temperature. The louvers on the outside sides of the spacecraft would be used to keep the interior temperature at the desired value. The small mapping package has its own thermal requirements which are probebly more stringent than those associated with electronics: The package shown is cylindrical in shape, thus keeping the solar input to the package constant independent of the package gimbal angle. This feature may not actually be required but is shown for conservatism.

Structurally, the compression loads are carried from the top of the Centaur through the landing vehicle and thence to the spacecraft. Tension loads, associated with the Saturn 1B structure rebound at cutoff, are taken through the outer sterilization can up to the spacecraft. A separation joint is provided between the Centaur and the landing vehicle sterilization can as shown, and at the extreme diameter is a separation joint allowing the sterilization can to be opened. It is currently contemplated that the upper half of the sterilization can would be retained with the spacecraft. Cg control is assured by the axial location of most critical components and by the fact that the small mapping package is gimbaled at its cg. The only element which can cause a shift of the cg is the high gain antenna, which will reflect into an engine gimbal angle requirement.

Various possible locations for the high gain earth antenna have been investigated and the one shown seems to give the maximum diameter of antenna and also allows a convenient stowage position. It is contemplated that the antenna would have a monopulse feed, allowing the antenna to be servoed directly to earth from the rf signal. This capability would serve as a backup to gimbal angle sensors which would allow the antenna to be pointed at earth from a known spacecraft body attitude.

Other features of the configuration, such as the locations of the cold gas attitude control jets, can be readily seen by an examination of 6-1.

### 5.1.2 RTG Powered Configuration

One of the main problems associated with Case 1) lay in the requirement for multiple gimbaled elements. For example, in the sample design the antenna required two gimbals (with in-line reliability). The case we are now considering represents an attempt to find a simpler way of doing the mission to reduce the gimbaling requirements. It is based upon the substitution of RTG power for solar power (which is not feasible for the 1971 - 1975 missions, but may be feasible for later missions).

Use of RTG power removes the solar pointing constraint. This reduction significantly simplifies the overall configuration but leads to additional tradeoffs in comparing the two cases. RTG power is significantly less efficient in terms of watts per pound than is solar power, even at Martian distances from the sun. This means that an RTG powered spacecraft must have a greater antenna gain in order to achieve comparable data rates and overall performance. Figure 5.2 represents a possible RTG powered earth-oriented configuration. Advantage is taken of the TRW Sunflower dish development to achieve a 36-ft diameter antenna (as compared to the 12-ft in Case 1). This gives 9.5 db more antenna gain and allows a commensurate reduction of transmitter power for the same data rate. A 32-ft diameter antenna has been made and deployment demonstrated for a solar collector application by TRW Tapco. Thus, an antenna of this size is quite possible even with surface dimensional accuracies higher than those required for an antenna application.

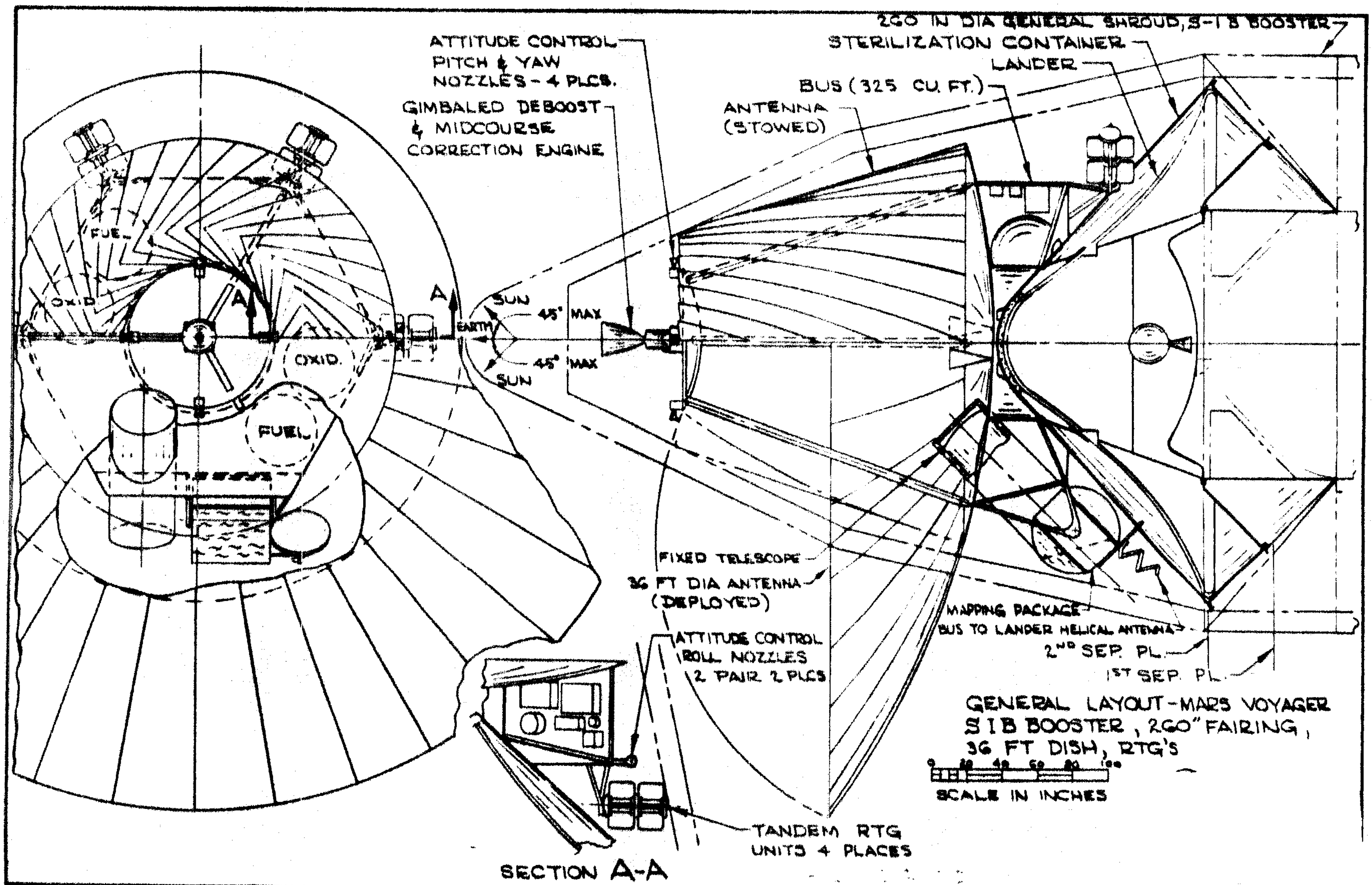



Figure 5.2 General Layout - Mars Voyager SIB Booster, 260-in. Fairing, 36-ft. Dish, RTG Power

Since the antenna is the dominant feature of this configuration, it was found necessary to change the spacecraft body concept from a comparatively small compact body to a toroidal body which surrounds the entry vehicle. Again, two separate imaging packages are used, a low resolution single gimbal package and a high resolution fixed package. The use of earth pointing does not affect the operation of the gimbaled package but does result in various illuminations of the surface at the time the high resolution telescope is looking directly down. However, an examination of typical orbits shows that reasonable illuminations will be available for use by the high resolution telescope.

In this configuration antenna receiving power becomes in-line in a reliability sense in order to conveniently keep the antenna rf axis pointed at earth. Of course, additional small gimbaled sun sensors could be used as a backup, if desired. It is felt, however, that the reliability of this monopulse scheme will be acceptable through the use of redundancy.

Again, the spacecraft is rolled to give the imaging packages their appropriate coverage. The midcourse and main deboost engine is conveninetly mounted on top of the cassegranian element of the main antenna, as well as the cold gas attitude control jets.

Thermal control considerations are essentially the same as for Case 1) and the basic interstage and entry vehicle sterilization container structures are retained. Louvers on the exterior of the toroidal spacecraft body are used for thermal control; there is more than adequate mounting area for all of the electronic components. There are no elements which shift the cg as in Case 1), hence, the gimbal limits of the engine can be smaller.

All in all, a comparison between Case 1) and Case 2) rests upon the procurement of adequate numbers of RTG units and upon more detailed reliability/performance analyses. The preliminary weight estimates given later show the RTG powered spacecraft to be slightly heavier.

It does not appear that RTG units of the wattage required will be available within the next 10 years (in any case the cost of long half life isotope material is extremely expensive).

## 5.2 System Analysis

In the configurations discussed, a large lander is separated from the spacecraft bus at some time preceding Mars encounter, and is directed on an impact course. The spacecraft bus tracks and monitors the lander through impact, and then uses its own propulsion system to enter an elliptical orbit about Mars. The experimental objectives of the lander are directed toward measurements made while descending through the atmosphere, as well as extensive experimentation performed on the surface of Mars. A major experimental objective of the orbiter is to obtain comprehensive mapping coverage of the surface of the planet.

The spectrum of logical Mars missions and configurations includes other combinations, e.g., flyby configurations, flyby landers, and orbiters with small landers or no landers. However, the orbiter with a large lander has requirements which encompass those of the other configurations. So by concentrating on the orbiter-lander, we are developing all the elements necessary to satisfy the requirements of the lesser missions.

The major requirements of the orbiter-large lander configurations are:

1) performance of all major subsystems for a period of 7 to 14 months after injection;

2) an attitude control subsystem which maintains cruise attitude during 7 to 9 months transit to Mars, stable commanded attitude during the separation of the lander and during the propulsive maneuvers of midcourse correction and deboost, and stable commanded attitude consistent with the requirements of high-resolution mapping operations;

3) a communications subsystem capable of transmitting the large amounts of data obtained in a comprehensive mapping of Mars over distances to earth of up to 400,000,000 kilometers; also capable of relaying data to and from the lander;

4) a space power subsystem and a thermal control subsystem which cope with levels of solar intensity which diminish to less than half the initial value during transit, and later, when in orbit about Mars, are subject to periodic interruptions by eclipse of the sun by Mars;

5) maintenance of the lander in a state of biological sterility, from before launch through transit and separation from the orbiter;

6) a guidance system which conducts the spacecraft to Mars, places the lander on an impact trajectory which is in a corridor whose width is several hundred kilometers, and injects the spacecraft bus into a predetermined orbit about Mars; yet this trajectory must avoid at all times even a remote possibility of unintentional entry into the Martian atmosphere by unsterile components; and

7) a complement of spacecraft and lander experiments which perform their function after a 7 to 9 month transit period.

The above requirements are those of the spacecraft bus and mission operations. The subsystems of the lander are subjected to the following environmental conditions:

1) Sterilization of the entire lander, including soaking at temperature of 135$^{\circ}$ C;

2) Maintenance of sterilization during transit by enclosure in a hermetically-sealed container;

3) Acceleration levels of the order of 50 g, and high thermal inputs, during entry into Mars' atmosphere;

4) Laning impact of 100 g; and

5) The low temperature and pressure environment of the Martian surface.

## 5.2.1 Mission Profile

The sequence of the principal events associated with a Mars mission comprised of a large lander and a spacecraft which is injected into orbit is as follows:

The events occurring from launch through transit include these phases:

Launch
Separation from launch vehicle
Acquisition of attitude references and orientation
First two midcourse corrections

The events occuring near encounter with Mars are detailed in Table 5.1 and include these phases:

Terminal guidance sensing of Mars
Terminal propulsion correction maneuver
Lander separation and propulsion; spacecraft separation propulsion
Lander entry
Injection of spacecraft into orbit

The events listed for the encounter phases indicate a mission profile with a moderate degree of complexity; however, they serve to accommodate a number of mission constraints, some based on assumptions which may well prove less severe than presently assumed, and others introduced because they are deemed desirable but not essential to the success of the mission.

Table 5.1 Preliminary Flight Sequence
(Events associated with encounter of Mars)

| EVENT | TIME | COMMENTS |
|---|---|---|
| A. Terminal guidance sensing of Mars | | |
| 1. Set mapping package pitch angle | E-150.2 hours | This aims mapping package at Mars. Mapping package and Canopus sensor have appropriate relative longitude on spacecraft so that Canopus lock is retained for this event. |
| 2. Take identification TV picture ($15^{o}$ field) | E-150.1 hours | |
| 3. Take measurement picture ($5^{o}$ field) | E-150 hours | |
| 4. Process photograph | E-149.9 hours | |
| 5. Transmit to Earth | E-149 hours to E-140 hours | |
| 6. Repeat 39 to 41 | E-140 hours to E-130 hours | |
| 7. Repeat 39 to 41 | E-130 hours to E-120 hours | |
| 8. Return mapping package pitch angle to stowed position | E-120 hours | |
| B. Terminal correction maneuver (if necessary) | | |
| 9. Third propulsion maneuver | E-96 hours to E-93.7 hours | |
| 10. Arm fourth propulsion maneuver | | |
| C. Terminal guidance sensing of Mars | | |
| 11. (Events 37 to 41, and Event 44) | E-72.2 hours to E-62 hours | Evaluation of accuracy of terminal correction maneuver and determination of new trajectory |

Table 5.1 (Continued)

| EVENT | TIME | COMMENTS |
|---|---|---|
| D. Checkout lander | | |
| 12. Lander subsystem checked from spacecraft | E-54 hours | Final check which is made while lander is in sterilization container |
| E. Separation of lander | | |
| 13. Input separation data | E-50 hours | Provides quantitative commands for spacecraft orientation (Events 51 and 56) and duration of propulsion (Event 57) |
| 14. Separate and eject outer half of sterilization container | E-48.9 hours | |
| 15. Orient spacecraft to separation attitude | E-48.8 hours | Spacecraft roll axis is aligned with direction of $\Delta V$ (lander) vector |
| 16. Separate and eject lander from spacecraft | E-48 hours | Inner half of sterilization container remains with spacecraft |
| 17. Spinup lander | E-47.99 hours | Initiated by lander timer started by Event 52 |
| 18. Turn on lander power amplifier | E-47.98 hours | Initiated by lander timer |
| 19. Turn on spacecraft power amplifier (to transmit to lander) | E-47.98 hours | |
| 20. Orient spacecraft to attitude for retro propulsion | E-47.97 hours | Spacecraft roll axis is aligned with direction of trajectory (relative to Mars). This is $180^{\circ}$ from $\Delta V$ (spacecraft) vector. |
| 21. Spacecraft propulsion | E-47 hours | |

Table 5.1 (Continued)

| EVENT | TIME | COMMENTS |
|---|---|---|
| 22. Reorient spacecraft to cruise attitude | E-46.95 hours | |
| 23. Arm fifth propulsion maneuver | E-46 hours | |
| 24. Ignite lander propulsion | E-46 hours | Fixed impulse solid propellant. Burn to depletion. Initiated by lander timer. |
| 25. Set mapping package pitch angle | E-45.9 hours | Establishes communication link between lander and spacecraft |
| F. Lander entry | | |
| 26. Enter Mars atmosphere | E-1.00 hours | Detected by on-board accelerometer |
| 27. Fire de-spin rockets | E-0.999 hours | Initiated by accelerometer |
| 28. Arm parachute deployment | E-0.998 hours | Effected by accelerometer, as deceleration increases |
| 29. Deploy parachute; separate lander capsule from heat shield | E-0.99 hours to E-0.90 hours | Initiated by accelerometer, as deceleration decreases to a fixed low value |
| 30. Impact | E-0.98 hours to E-0.50 hours | Detected by on-board instrumentation. Time depends on atmospheric density profile and angle of entry. |
| G. Injection of spacecraft into orbit | | |
| 31. Input deboost data | E-12 hours | Provides quantitative commands for spacecraft orientation (Event 69), spacecraft reorientation (Event 75), and duration of propulsion (Event 73) |
| 32. Maneuver sequence start | E-2 hours | |
| 33. Return mapping package pitch angle to stowed position | E-0.50 hours | Events 68 to 71 are commanded from Earth to occur at predetermined times. It is expected that Event 66 occurs before Event 69, so that the spacecraft receives lander data through impact. |

Table 5.1 (Continued)

| | EVENT | TIME | COMMENTS |
|---|---|---|---|
| 34. | Orient spacecraft to deboost attitude | E-0.49 hours to E-0.13 hours | An orientation mode with higher angular velocity rates is required. |
| 35. | Ignite spacecraft engine | E-0.12 hours | |
| 36. | Spacecraft passes point of closest approach to Mars (periareum of subsequent orbit) | E | E stands for "Encounter," defined by this event |
| 37. | Terminate spacecraft engine | E+0.12 hours | |
| 38. | Reacquire Sun | E+0.13 hours to E+0.46 hours | Events 74 to 77 describe process of attaining normal orbiting cruise attitude |
| 39. | Roll spacecraft to put mapping package motion in Sun-spacecraft-Mars plane | E+0.47 hours to E+0.97 hours | |
| 40. | Scan with mapping package to acquire Mars with horizon scanners | E+1.00 hours to E+1.10 hours | |
| 41. | Spacecraft roll angle and mapping package gimbal angle are controlled by lock of horizon scanners on Mars | E+1.10 hours | |

For example, it is assumed that guidance of the spacecraft in the approach phase cannot, if it is based solely on DSIF tracking, guarantee a knowledge of the location of the trajectory relative to Mars with required precision. This assumption is certainly valid for narrow corridors (50 to 100 km) associated with landers of high ballistic coefficient, $W/C_D A$ (30 to 40 lb/ft$^2$). However, with large diameter landers, $W/C_D A$ is typically 10 lb/ft$^2$, and the corridor may be as wide as 1,400 km, so that DSIF tracking may be adequate. However, the sequence of events provides for terminal sensing of Mars, and a corresponding propulsive correction to refine the trajectory.

It is also assumed that the spacecraft must receive communication from the lander (to relay it to Earth) until the lander impacts. To satisfy this assumption it is necessary to use propulsion to slow down the spacecraft relative to the lander after separation, and to compress the time available after lander impact for orientation of the spacecraft preparatory to execution of its orbit-injection propulsion. These requirements tend to complicate this phase of the mission sequence. It should be recognized that they contribute to a backup mode of operation not essential (in itself) to the success of the mission. For the data recorded by the lander during its entry (assuming successful operation of lander subsystems) will survive impact, and may later be transmitted to Earth either directly through the lander-Earth communications system or by relay via the orbiting spacecraft.

In connection with the terminal guidance sensing of Mars, the sequence of events presented in Table 5.1 is based on sensing with an optical system employing a 6-inch diameter aperture, f/4, and photographic film. This is an optical system presently under consideration as part of the complement of the mapping package. Compared with the alternate system based on television sensors, the photographic system provides superior angular resolution of Mars against a star background, leading to an earlier assessment of the trajectory accuracy. It is estimated that the photographic system will allow trajectory assessment to 200 km accuracy (one sigma) to be completed when still more than 1,500,000 km from Mars. This allows adequate time to conduct a corrective propulsion maneuver, and to perform the lander separation sufficiently in advance of encounter to avoid propellant weight penalties. With alternate optical systems of inferior resolution, assessment of trajectory accuracy, propulsive trajectory correction, and separation are all delayed, leading to higher $\Delta V$'s and higher propellant weights. At some point, it is worth considering the advantage of designing a sensing subsystem for the sole function of terminal (approach) guidance, without reference to its qualities in a surface mapping function.

### 5.2.2 Spacecraft Weights

Preliminary weight estimates were prepared for the two configurations. Summary weight statements for these three configurations are presented on Table 5.2 assuming entry into a circular Mars orbit (2000 km altitude) and assuming redundancy in the power, propulsion, and communication systems. Summary weight statements for lighter orbiter configurations are presented on Table 5.3 assuming entry into an aliptical orbit at Mars (2000 km by 20,000 km) and eliminating the battery redundancy assumed for Table 5.2.

Lander - A Lander weight of 4000 pounds was assumed for the orbiters.

"Bus" Structure - The structure weights have the following assumptions: The equipment compartment outer shell is designed by meteoroid protection requirements assuming 99% probability of no penetrations for a 270-day mission. Equipment mounts are 10% of the equipment weight and the telescope mount weight is 5% of the telescope weight.

Thermal Control, Communications, Power and Integration, and Central Computer and Sequencer - J-Box, cabling anc connector weights were estimated based on the weight of the components requiring power and electrical connection. The solar array weights are based on the entire spacecraft being oriented to the sun. The total power assumed for the configurations is 500 and 300 watts, respectively. The batteries, battery charger and load conditioning equipment include redundancy.

Deboost and Midcourse Correction Propulsion Systems - Weights for these systems were derived from empirical data. Ground rules for each configuration are listed below.

Table 5. 2 Weight Summary Mars Missions (2000 km Circular Orbit)

| | Configuration | |
|---|---|---|
| | Solar | RTG |
| Structure | 469 | 452 |
| Thermal Control | 64 | 66 |
| Communications | 94 | 352 |
| Power and Integration | 739 | 752 |
| Central Computer and Sequencer | 36 | 36 |
| Deboost and Midcourse Correction Propulsion System (Inert) | 435 | 465 |
| Attitude Control System | 143 | 143 |
| Telescope | 400 | 400 |
| Experiments | 283 | 283 |
| Contingency (10%) | 266 | 295 |
| Weight of Orbiter | 2929 | 3244 |
| *Deboost Propellants | 3354 | 3715 |
| Weight of Orbiter Prior to Deboost | 6283 | 6959 |
| Lander | 4000 | 4000 |
| Weight of Spacecraft at End of Midcourse Correction | 10, 283 | 10, 959 |
| Midcourse Correction Propellants | 331 | 353 |
| Spacecraft Gross Weight | 10, 614 | 11, 312 |

***Inject into circular orbit (2000 km) approaching Mars at $V_\infty$ = 3. 05 km/sec.**

Table 5. 3. Weight Summary Mars Mission
(2, 000 x 20, 000 km orbit)

| | Configuration Solar | RTG |
|---|---|---|
| Structure | 469 | 452 |
| Thermal Control | 64 | 66 |
| Communications | 94 | 352 |
| Power and Integration | 622 | 682 |
| Central Computer and Sequencer | 36 | 36 |
| Deboost and Midcourse Correction Propulsion System (Inert) | | |
| Attitude Control System | 143 | 143 |
| Telescope | 400 | 400 |
| Experiments | 283 | 283 |
| Contingency (10%) | 241 | 273 |
| Weight of Orbiter | 2647 | 3001 |
| *Deboost Propellants | 1670 | 1893 |
| Weight of Orbiter Prior to Deboost | 4317 | 4894 |
| Lander | 4000 | 4000 |
| Weight of Spacecraft at End of Midcourse Correction | 8317 | 8894 |
| Midcourse Correction Propellants | 268 | 287 |
| Spacecraft Gross Weight | 8585 | 9181 |

*Inject into eliptical orbit (2000 KM x 20, 000 KM) approaching Mars at $V_{\infty}$ = 3. 05 KM/Sec.

Table 5.4 Detailed Weight Statement
(2,000 km Circular Orbit)

| | Configuration | |
|---|---|---|
| | Solar | RTG |
| | Weight, Lbs. | |
| Item | | |
| Structure | 469 | 452 |
| Equipment Compartment | 368 | 368 |
| Thrust Structure and Cylindrical Shell | 83 | 56* |
| Mapping Package Mount | 8 | 8 |
| Antenna Mount | 10 | 20 |
| Thermal Control | 64 | 66 |
| Communications | 93.5 | 351.5 |
| (Bus/Earth Equipment w/o Automatic Track) | (77.3) | (335.3) |
| Receiver, Command (2) | 7.0 | 7.0 |
| Decoder, Command | 3.5 | 3.5 |
| Digital Telemetry | 5.2 | 5.2 |
| Modulator | 1.4 | 1.4 |
| Baseband Assembly and Mode Selector | 1.0 | 1.0 |
| Branch Line Coupler | 0.3 | 0.3 |
| Power Amplifier, (2) | 6.2 | 6.2 |
| Antenna Subsystem | | |
| Diplexer (2 each) | 1.6 | 1.6 |
| Circulator (2 each) | 0.8 | 0.8 |
| Antenna (Omni) | 1.0 | 1.0 |
| Antenna (Directional) Including Feed | 42.0 | 300.0 |
| Antenna Drive | 7.3 | 7.3 |
| (Additional Equipment for Automatic Track) | (8.2) | (8.2) |
| Receiver, Error Angle | 2.0 | 2.0 |
| Sum and Diff. Hybrids | 1.3 | 1.3 |
| Bandpass Filter | .3 | .3 |
| Feed (Horn) | 4.6 | 4.6 |

*Includes 20 pounds for RTG support structure

Table 5.4 Detailed Weight Statement (Continued)

| | Configuration | |
|---|---|---|
| | Solar | RTG |
| | Weight, Lbs. | |
| **Communications (Cont'd)** | | |
| (Bus/Lander Equipment) | (8.0) | (8.0) |
| Receiver | 2.0 | 2.0 |
| Transmitter | 2.5 | 2.5 |
| Diplexer | 1.1 | 1.1 |
| Antenna (Helix) | 2.4 | 2.4 |
| **Power and Integration** | 738.5 | 752.1 |
| Solar Array | 260.0 | - |
| RTG's (Snap 19) | - | 465.0 |
| Battery | 235.0 | 141.0 |
| Battery Charger | 33.5 | 20.1 |
| Load Conditioning | 100.0 | 60.0 |
| Source Control | 10.0 | 6.0 |
| J-Box | 25.0 | 15.0 |
| Cabling and Connectors | 75.0 | 45.0 |
| **Central Computer and Sequencer** | 36 | 36 |
| Programmer | 7 | 7 |
| Computer | 20 | 20 |
| Power | 4 | 4 |
| Miscellaneous | 5 | 5 |
| **Deboost and/or Midcourse Propulsion System** (Inert) | 435 | 465 |
| **Attitude Control System** | 143 (1) | 143 (1) |
| **Telescope** | 400 | 400 |
| **Experiments** | 283 | 283 |
| Planet Oriented | 238 | 238 |
| Fixed to Bus Structure | 45 | 45 |

(1) Includes 35.0 lbs. $N_2$

Table 5. 4 Detailed Weight Statement (Continued)

| | Configuration | |
|---|---|---|
| | Solar | RTG |
| Contingency (10%) | 266 | 295 |
| WEIGHT OF BUS | 2929 | 3244 |
| Deboost Propellants | 3354 | 3715 |
| WEIGHT OF BUS PRIOR TO DEBOOST | 6283 | 6959 |
| Lander | 4000 * | 4000 * |
| WEIGHT OF SPACECRAFT AT END OF MIDCOURSE CORRECTION | 10, 283 | 10, 959 |
| Midcourse Correction Propellants | 331 | 353 |
| SPACECRAFT GROSS WEIGHT | 10, 614 | 11, 312 |

*Specified for recent NASA studies.

| | Configuration Solar | RTG |
|---|---|---|
| $\Delta V$, Deboost (ft/sec) | 7218 | 7218 |
| $\Delta V$, Midcourse Correction (ft/sec) | 300 | 300 |
| Type Propellants | $N_2O_4$/(72-25) $N_2H_4$ - MMH | |
| $I_{sp}$ (sec.) | 294 | 294 |
| Mixture Ratio | 1.42 | 1.42 |
| Thrust (lbs.) | 600* | 600* |

*Same system used for both midcourse correction and deboost.

Attitude Control - Attitude control weights were estimated using Mariner C weights as a guide. The weights as listed include the complete system, i. e., the expended nitrogen as well as the sensors and other fixed weight.

A contingency of 10% of the orbiter inert weight is included to account for weight uncertainties during spacecraft development.

## 5.3 Major Tradeoff Considerations and Key Technical Problems

### 5.3.1 Programming Lander Entry and Spacecraft Deboost

The mission profile for the entry and encounter phases of orbiter-lander configuration proposed is based on these ground rules:

a) Terminal guidance sensing is necessary to achieve the accuracy required to place the lander in the entry corridor.

b) The guidance and control of the lander are established by the subsystems of the spacecraft before the lander is separated from the spacecraft.

c) Data taken by the lander during entry should be transmitted before impact, so that this information is not lost in the event of failure at impact.

d) The lander has no high-gain antenna until after it has landed; therefore, its transmission during this period must be directed to the spacecraft.

e) Lander entry is effected by atmospheric braking based on the size of the rigid structure of the lander until its speed is reduced to Mach 2, thence by parachute(s) deployed.

f) The lander entry is to be successful for any Martian atmosphere within the extremes of atmospheric Models (To 10 mb).

g) The lander is separated from the spacecraft before the spacecraft is deboosted into orbit.

h) The spacecraft tracks the received data from the lander via a low gain antenna mounted on a mapping package and aimed open loop in an appropriate direction relative to the spacecraft.

i) The spacecraft should receive data from the lander through lander impact before it starts orientation for the deboost propulsion.

j) The spacecraft deboost is conducted for a duration of approximately 14 minutes, centered about the point of closest approach to Mars. (This corresponds to using 1,700 lbs of propellant ($I_{sp}$ = 300 sec) at a thrust level of 600 lbs to effect a $\Delta V$ of 1.60 km/sec for a spacecraft of 2,300 lbs weight dry)

k) The spacecraft maintains its normal cruise attitude, except as required for 1) separation of lander, 2) propulsion of the spacecraft after lander separation, and 3) deboost operation.

Ground rules e) and f) lead to a lander entry corridor width which varies from 50 to 100 km for landers of high ballistic coefficient, $W/C_DA$, (30 to 40 lb/ft$^2$) to 1,400 km for landers of low coefficient (10 lb/ft$^2$). The corridor width arises from tolerable variations in the angle of entry into the atmosphere. Because the corridor width depends on the cosine of the entry angle, small entry angles contribute little to corridor width. However, they contribute substantially to the range of entry time intervals (time from entry to impact) which exist over the corridor width, because the time of descent increases rapidly as the entry angle approaches grazing incidence. Accommodating high entry time intervals imposes a design penalty, thus, it may be desirable to disqualify from the effective corridor that portion arising from the smallest entry angles.

Due to guidance accuracy limitations, we cannot expect the lander to enter on a prescribed track, but rather in a corridor of finite width. The width depends on the angular resolution and accuracy which can be achieved by an on-board Mars sensor, on the distance from Mars at which the sensing is performed, and on the execution error of trajectory correction and separation $\Delta V$. Because the sensor accuracy and resolution is not unlimited, achieving any required guidance accuracy of corridor entry depends on performing the sensing when the spacecraft is within a certain maximum distance from Mars. To reduce the amount of fuel necessary to perform the separation and trajectory correction maneuvers, it is desirable to perform the terminal sensing when far from Mars. And to achieve this, it is desirable to keep the effective entry corridor as wide as possible. This objective is contrary to that of the preceding paragraph.

Ground rules d), h), i), and j) require the spacecraft to be delayed relative to the lander (as it begins its entry) by a time interval approximately equal to the time necessary for spacecraft orientation before deboost plus the maximum permissible lander entry time interval. The greater this delay time interval, the greater the maximum lander entry time interval which can be permitted, and the wider the effective corridor. But also, when this delay time interval is greater, the lander-spacecraft distance is greater, resulting in lower communication rate and in a greater propulsion requirement to achieve the increased separation distance.

The optimum values for all these variables have not yet been determined, as a sensor for terminal guidance has not been selected or rigorously analyzed.

With regard to the separation velocities to be achieved by lander propulsion and spacecraft propulsion, Figure 6-3 shows the vector relationship indicating the lateral and longitudinal components of the total separation $\Delta V$, and which propulsive impulses contribute to them. The order of magnitude of these velocities is

| | |
|---|---|
| V (spacecraft) before separation | 2.5 to 4.0 km/sec |
| $\Delta$ V (spacecraft) | 50 to 300 meters/sec |
| $\Delta$ V (lander) | 20 to 120 meters/sec |

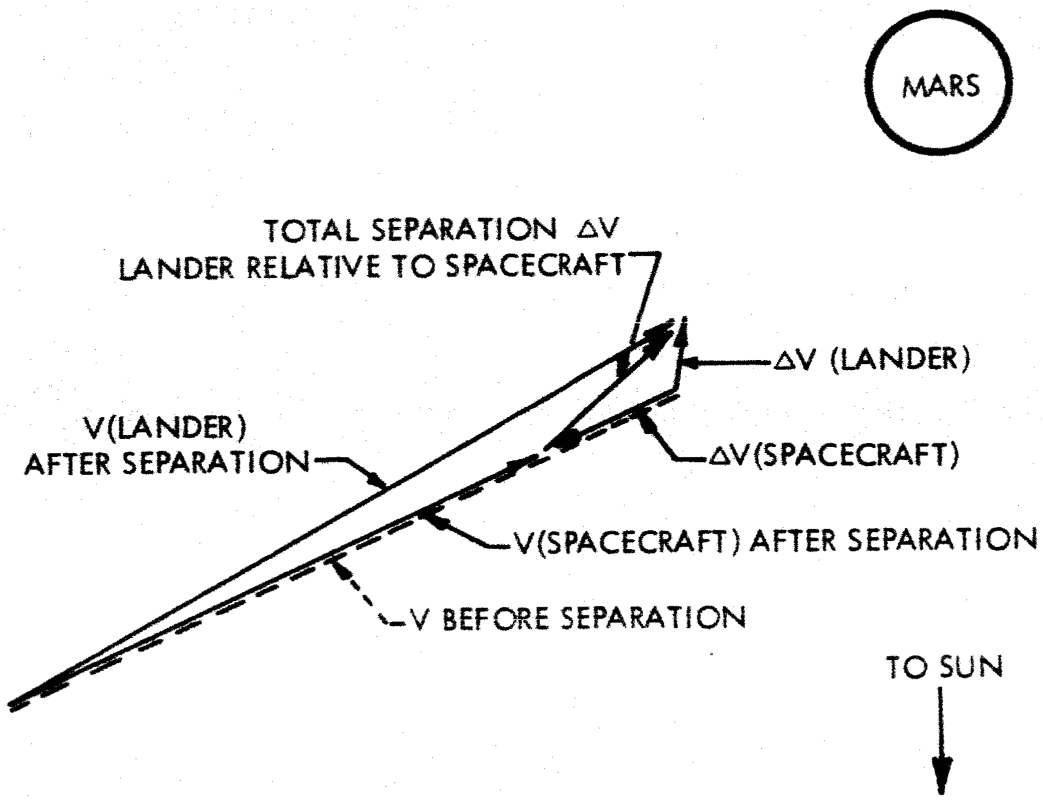



Figure 5.3 Approach Velocities Relative to Mars

The total separation $\Delta V$ produces the different paths and time differences for lander entry and spacecraft encounter. Further analysis must be made to ascertain the optimum division of this $\Delta V$ into that provided by the spacecraft and that provided by the lander. In a sense, the $\Delta V$ provided by the lander is "wasted", whereas the $\Delta V$ provided by the lander is "useful" in that it reduces its velocity and reduces the propellant required for deboost (although separation $\Delta V$ is not as efficient as deboost $\Delta V$ in reducing spacecraft kinetic energy, as it occurs at a lower velocity). Other factors to be considered in this tradeoff are:

- o As lander $\Delta V$ is increased, its direction is more closely aligned with the $V_\infty$ vector, leading to a lower angle of attack at entry (favorable) and to higher absolute values of execution error, both in longitudinal and lateral component (unfavorable).
- o As spacecraft $\Delta V$ is reduced to a smaller portion of total $\Delta V$ the direction from the spinning lander to the spacecraft is more nearly along the lander centerline, making possible the use of a more directional lander antenna. This improves the lander-spacecraft communication link.

### 5.3.2 Solar Power vs. RTG Power

The decision between solar power and RTG power is not simply a matter of watts per pound or watts per unit intercepted area. To a certain extent, by making certain design decisions contingent on the choice of spacecraft power, we have arrived at two systems which differ markedly in many more respects than just the mode of power generation. In particular, these design decisions have affected the attitude control subsystem, the communication subsystem and the thermal control subsystem, as well, as indicated by the configuration drawings.

Aside from purely design tradeoffs, it was determined that the availability of adequate amounts of plutonium precludes consideration of RTG power for the missions considered (to 1975).

### 5.3.4 TV vs Film

The choice of the imaging medium for the mapping coverage of Mars is still to be made. Compromise systems are a possible choice also, as it seems feasible to have both TV and film on separate optical systems, or even sharing the same optical system.

Factors which affect a comparison of TV and photographic film are:

- Achievable resolution, as affected by the light sensitivity (ASA rating) of the imaging media.
- Achievable resolution vs. smear due to relative motion of spacecraft and planetary surface, and the consequent requirement for image motion compensation.
- Resolution expressed as the number of lines (or picture elements) in a single frame or exposure.
- The mechanization of data storage, processing and readout.
- Sensitivity to environment. In particular, the susceptibility of photographic film to the interplanetary radiation fields restricts film types which may be considered, and influences the design decision.
- The usual considerations of weight, size, required power, and reliability.
- Compatibility with methods of reconstructing surface coverage on Earth.

### 5.3.5 Attitude Control in Orbit

The optimum method of attitude control for the spacecraft in orbit has not been completely determined. For Case 1 (solar power) we have assumed that the normal orientation of the spacecraft while orbiting Mars is to have the roll axis pointed at the Sun. The roll attitude and mapping package gimbal angle are controlled to values which vary during the orbit so as to keep the mapping package (boresight) axis aimed at the center of Mars. The Earth-directed antenna has two gimbals so it is able to point at the Earth at all times.

Two requirements to be satisfied by these procedures are:

a) to have the mapping package axis aimed at the center of Mars, and

b) to measure the direction (in celestial coordinates) from the spacecraft to Mars, or at least to take sufficient data that this direction can be reconstructed later.

We can accomplish a) by either open-loop or closed-loop methods. Closed-loop methods, employing Mars horizon sensing elements, seem preferable, because this will depend less on knowing the ephemeris of the spacecraft's orbit, and because it will contribute some information to aid in accomplishing b).

To accomplish b), there are two major alternate approaches. In the first of these, no on-board Mars sensing is required. The ephemeris of the spacecraft's orbit about Mars is determined entirely from DSIF tracking of the spacecraft. DSIF tracking of a spacecraft in orbit about Mars can lead to a determination of five of the six elements comprising the ephemeris, but the sixth element (corresponding to angle of rotation of the entire orbital track about the Earth-Mars line) cannot be determined, and remains unresolved. However, there are two methods for determining the sixth element.

a) By knowledge (obtained in tracking the spacecraft while in transit to Mars) of the direction of the approach $V_\infty$ vector, we can restrict possible resulting orbit planes to a family which is parallel to this vector. This serves to resolve the ambiguity of the sixth element of the orbit ephemeris, although the accuracy with which it does so is degraded if the V vector direction approaches the Earth-Mars direction.

b) By tracking the spacecraft in orbit for an extended period, the direction of the Earth-Mars line changes (due to annual motion). Therefore the rotation axis about which the ephemeris ambiguity exists changes. Assuming the orbit is fixed, this change will enable the ambiguity to be resolved. The chief limitation of this method is that unknown perturbations of the orbit will also affect the apparent orbit as tracked by DSIF, and it will be difficult to separate the changes due to orbit perturbation from changes due to relative change in the position of the Earth.

But if it is assumed that DSIF tracking will determine the ephemeris of the spacecraft's orbit about Mars, then the direction from the spacecraft to Mars is known at all times, and requirement b) is met. It should be noted that for DSIF determination of the spacecraft orbit, at best a period of days or weeks ensues before the determination is refined to a satisfactory accuracy. Therefore, for part of the mission, this determination is satisfactory only for data reconstruction, and not for real time commanded operations.

In the second approach to measuring the direction from the spacecraft to Mars, on-board components are used entirely, with no recourse to DSIF tracking. In this method, the attitude control system assures that the spacecraft orientation conforms to some set of fixed (non-Martian) references. The pointing of the mapping package at Mars by a closed loop system employing Mars horizon sensing elements then ascertains the direction of Mars relative to the spacecraft axes. The problem is that to establish spacecraft attitude and the relative direction of Mars requires 5 degrees of freedom, but the Case 1) design of a rigid spacecraft body and a single mapping package gimbal constitutes only 4 degrees of freedom. The added degree of freedom may be supplied by one of these methods:

a) Employing a Canopus (or other reference star) tracker which has a gimbal enabling it to maintain lock on the star, regardless of the spacecraft roll attitude. Measurement of this gimbal angle will establish the spacecraft attitude. For versatility, this method requires a Canopus sensor which can "look" in any (roll) direction relative to the spacecraft. This may pose geometrical design problems, as centerline space is at a premium. But other approaches may solve these problems.

b) The spacecraft attitude in roll may be measured by means of the gimbaled Earth-pointing antenna if the communication subsystem includes a tracking loop which is closed through sensing by the antenna feed of offsets in the direction of propagation of rf signals from Earth transmitters. (Although we have now added the two gimbals of the antenna, the effective increase in the number of degrees of freedom is only one, because the fixed Sun-spacecraft-Earth angle makes one gimbal redundant.)

The above discussion reviews possible methods of attaining spacecraft attitude control while in orbit about Mars. In addition to the obvious reasons for attaining this attitude control, it should be noted that the requirements of mapping the surface of Mars, with probable particular interest devoted to high resolution coverage of specific areas, impose severe requirements on the ability to point the mapping package in specific, predetermined directions, and on the ability to know, for purposes of mapping coverage reconstruction, just what part of Mars' surface was photographed.

### 5.3.5 On-Board Computation

An area of decision remains to be made concerning the desirability of incorporating on board the spacecraft a computer capable of more capability than relaying commands, programming sequences of commanded events, and performing pre-programmed diagnostic checks and corrective procedures in the event of some malfunction. In any case, these programming (or "sequencing") functions must be accommodated, and in considerably greater degree of complexity and detail than, say, in the present Mariner IV Mars probe.

But an additional computing capability can be considered, for such purposes as:

- performing calculations (for example, the solution of spherical triangles) for interpreting pointing and attitude instructions,
- performing calculations for biased aiming or scanning of the mapping package, e.g., away from the center of Mars, and
- on-board data reduction, interpretation, and compression, to conserve in the number of bits transmitted to Earth.

The advantages of the on-board computing capability are:

- With some data processing on the spacecraft, more efficient use is made of the available downlink communication data rate.
- With commanded spacecraft attitude and pointing programs interpreted by an on-board computer, uplink data requirements and on-board data storage capacity may both be reduced.
- Real-time operations in conjunction with the taking of data (for example, image motion compensation) are made simpler and more feasible.
- It saves a time interval of 15 to 25 minutes round trip communication time between Mars and Earth where it replaces such an information loop. For some events, this is time well saved.

The disadvantages are:

- The reliability of on-board hardware is reduced, unless realistic failure-mode implementation is assured.

- o In event of failure of some component (not necessarily in the computer), the use of on-board data interpretation and processing makes more difficult the process of locating the trouble and the reconstructing the raw data.
- o It requires space, power and weight.

5.3.7 Weights

Since the weight tables of Section 5.2.2 add up to a total which exceeds the launch vehicle capability for all of these missions, it is obvious that a weight tradeoff must be made. In addition, the recommendations for a consistent redundancy policy to be formulated out of the reliability assessment will influence the weights of the various subsystems.

The most likely resolution of the weight limitations during unfavorable years is to reduce or eliminate the lander package. It is unlikely that the basic bus design will be altered from mission to mission.

5.3.8 Reliability

Projecting or extrapolating integrated circuit failure rates from past history to the 1969 - 1971 era makes the probability of mission succes 0.4 to 0.5. The use of redundancy in a few areas where it obviously makes sense shows prospects of raising the probability of mission success (still excluding experiment performance) to 0.7 or 0.8.

5.4 Subsystem Design (Spacecraft Bus)

5.4.1 Communications

Communication systems were considered for both configurations. The assumptions made for initial design considerations are presented for the solar powered orbiter. Subsequently, differences for the other configurations are considered.

Solar Powered Orbiter - The "Bus" must communicate with the DSIF tracking stations on earth and with the "Lander." The principal task will be the transmission to earth of the large quantities of Mars mapping data acquired by the orbiter, however, the "Bus" may also be required as a backup system to relay relatively large quantities of "Lander" data to earth.

The earth-bus link was assumed to employ the DSIF transmitting with an 85 foot antenna at a frequency of 2115 mc with a power output of 100 kw. Receipt of commands by an omni-directional antenna on the "Bus" is required since the high gain antenna will not be properly oriented during some portions of the mission.

The "Bus" was assumed to transmit to the DSIF 210 foot antenna at a frequency of 2295 mc. The data acquisition capability will be extremely high and consequently it was assumed that the bus-to-earth communication system should have as much capacity as could be reliably achieved within the spacecraft weight and volume limitations. A 100 watt transmitter power output was assumed to be available and the largest conventional antenna (12 ft dia.) compatible with the solar powered spacecraft configuration was specified for this configuration.

The "Lander" will transmit data to the "Bus" during re-entry and landing. After landing and orienting, the "Lander" was assumed to transmit data directly to earth with the "Bus" serving as a backup relay if required. For initial considerations, a "Lander" bit rate of 200 bits per second at a range of $2 \times 10^4$ km was assumed. A 400 mc frequency was specified for this link since the noise temperature for solid state receivers is near minimum at this frequency and a reasonable "Bus" antenna gain could be provided within "Bus" space limitations.

The "Bus" also serves as a backup relay for transmitting earth commands to the "Lander". A bit rate of 1 PPS and a frequency of 370 mc were assumed for this backup relay link.

A block diagram for the "Bus" communication system is shown on Figure 5.4 and the performance of the various communication links is summarized on Table 5.5. The components are listed on Table 5.6 together with their estimated weight and power requirements.

The earth-bus link provides for a command data rate of 1 PPS with the "Bus" receiving on an omni-directional antenna out to a range of about $2.5 \times 10^8$ km. While this will provide for considerable time after arrival at Mars, it may be desirable to continue operations as long as the spacecraft is alive which could be at ranges of $4 \times 10^8$ km. Some increase in range should be obtainable through the omni-directional antenna since all the negative margins will not

Table 5.5.

PRELIMINARY ESTIMATE OF COMMUNICATION SYSTEM PERFORMANCE FOR SOLAR POWERED BUS (ORBITER)

| | COMMUNICATION LINK | | | |
|---|---|---|---|---|
| | Bus - Earth | Earth - Bus | Bus - Lander | Lander - Bus |
| Frequency, MC | 2,295 | 2,115 | 370 | 400 |
| Transmitter Power, Watts | 100 | $100 \times 10^3$ | .8 | 8 |
| Transmitting Antenna | | | | |
| Type | Cassegrain | Cassegrain | Helix | Turnstile |
| Size | 12 ft.(dia) | 85 ft.(Dia) | - | - |
| Gain, db | 35.8 | 51 | 10 | +1.7 to -1.5 |
| Pointing Accuracy Req.,Deg. | $\mp .8$ | - | - | - |
| Pattern Coverage, Deg. | - | - | $\mp 32$ | $\mp 120$ |
| Receiving Antenna | | | | |
| Type | Cassegrain | Turnstile | Turnstile | Helix |
| Size | 210 ft.(dia) | - | - | - |
| Gain, db | 61 | +1.7 to -1.0 | +1.7 to -1.0 | 10 |
| Pattern Coverage, Deg. | - | $\mp 120$ | $\mp 120$ | 63 |
| Telemetry Input Capacity | | 128 Commands | 128 Commands | |
| Engr. Instr. | 56 Analog | - | - | 56 Analog |
| | 64 Bi-level | - | - | 64 Bi-level |
| Exper. - Scientific | 6 Digital | - | - | 6 Digital |
| Reference Range, Meters | $2.46 \times 10^8$ | $2.46 \times 10^8$ | $2 \times 10^4$ | $2 \times 10^4$ |
| Bit Rate at Ref. Range PPS | approx 40,000 | 1 | 1 | 200 |
| Range - 2 Des. Bit Rate, Meters | - | - | - | $1.4 \times 10^4$ |
| Range - $\frac{1}{2}$ Des. Bit Rate, Meters | $3.48 \times 10^8$ | - | - | - |
| *Range - $\frac{1}{4}$ Des. Bit Rate, Meters | $4.92 \times 10^8$ | - | - | - |
| Data Perf. Margin, db | 5.8 | 3.1 | 14.3 | 7.4 |
| Perf. Margin Req'd., db (Approx. Sum. of Neg. Tolerences) | 5.5 | 3.0 | 6.5 | 6.6 |

* Max Earth Mars Range = $4.0 \times 10^8$ KM
Earth Mars Range at Time of 1971 Encounter Less than $2 \times 10^8$ KM for 30 Day Launch Window

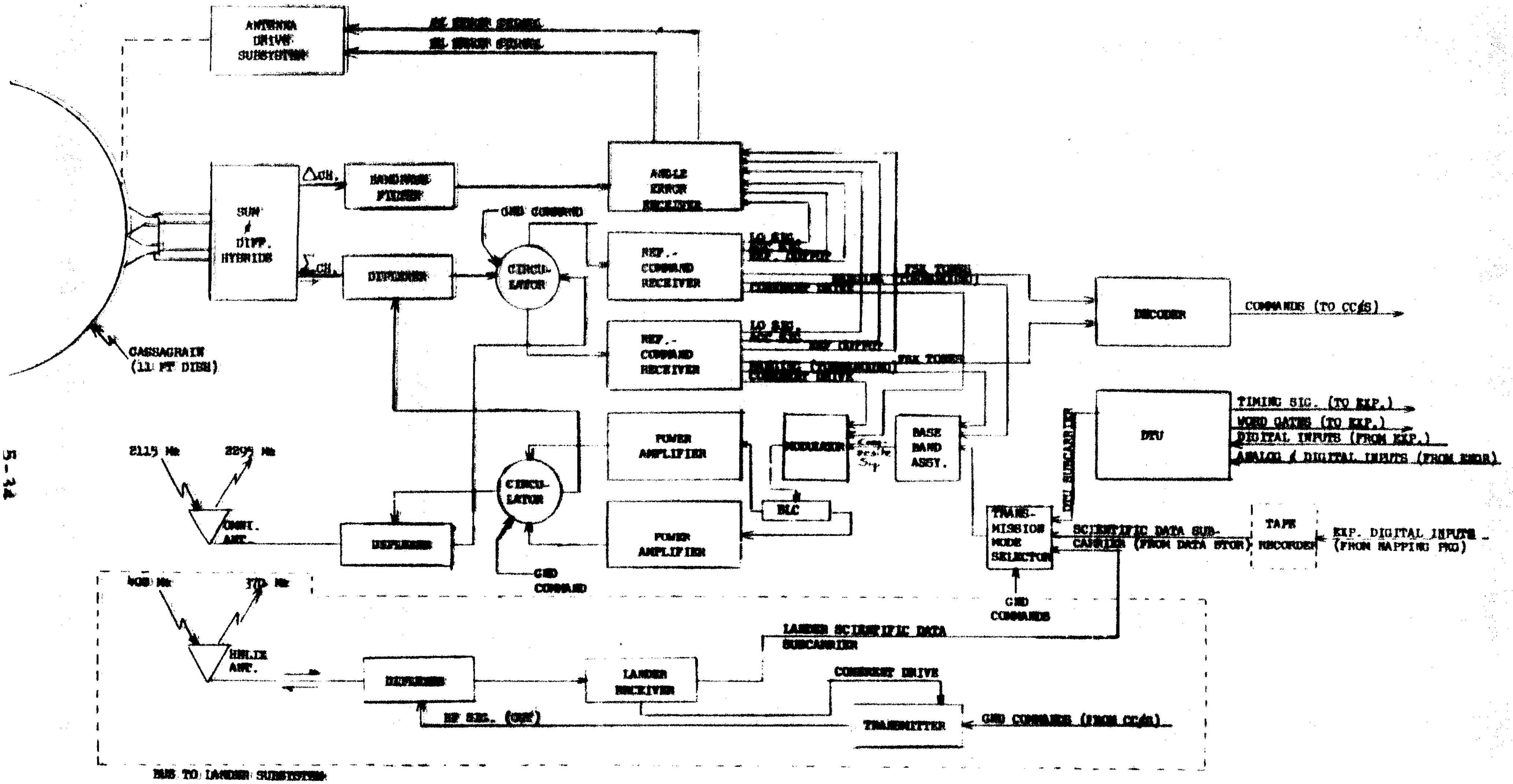



Figure 5.4. Preliminary Communications Subsystem for Mars 69-71 Bus
(With Automatic Track)

Table 5.6. Preliminary Communication Power, Weight and Volume Breakdown for Solar Powered Spacecraft Bus (Orbiter)

| Unit | Power (watts) | Weight (lbs) | Volume ($in^3$) |
|---|---|---|---|
| **Bus/Earth Equipment W/O Automatic Track** | | | |
| Receiver, Command (2 each) | 2.6 | 7.0 | 216.0 |
| Decoder, Command | 0.4 | 3.5 | 113.0 |
| Digital Telemetry | 1.2 | 5.2 | 50.0 |
| Modulator | 1.5 | 1.4 | 50.7 |
| Baseband Assy. & Mode Selector | .04 | 1.0 | 26.0 |
| Branch Line Coupler | -- | 0.3 | 4.3 |
| Power Amplifier, (2 each) | 256 (1 only) | 10.8 | 125.8 |
| Antenna Subsystem | | | |
| Diplexer (2 each) | -- | 1.6 | 95.2 |
| Circulator (2 each) | .03 | 0.8 | 1.0 |
| Antenna (Omni) | -- | 1.0 | |
| Antenna (Directional) | | | |
| Feed | -- | 2.0 | |
| Dish - 12 ft. | -- | 40.0 | |
| Antenna Drive | 10 (avg) 28 (peak) | 7.3 | |
| TOTAL | 271.8 (avg) | 81.9 | 681.0 |
| **Additional Equipment for Automatic Track** | | | |
| Receiver, Error Angle | 0.3 | 2.0 | 54.0 |
| Sum & Diff. Hybrids | -- | 1.3 | 9.0 |
| Bandpass Filter | -- | .3 | 11.0 |
| Feed (Horn) | -- | 6.6 | |
| TOTAL | 0.3 | *10.2 | 74.0 |
| **Bus/Lander Equipment** | | | |
| Receiver | 1.2 | 2.0 | 60.0 |
| Transmitter | 5.4 | 2.5 | 56.0 |
| Diplexer | -- | 1.1 | 47.0 |
| Antenna (Helix) | -- | 2.4 | |
| TOTAL | 6.6 | 8.0 | 163.0 |

*The total difference in weight for a self-tracking system would be 8.2 lbs, because the 2 lbs. for the single feed would be deleted.

be additive, however, at extreme ranges, transmission to the spacecraft will have to utilize the high gain antenna which will probably have to be continually earth oriented for extended range operations. Command bits are sent as FSK with 140 cps and 240 cps tones representing the data. Synchronism is obtained by starting a sync bit clock in the "Bus". Pseudo random noise (PRN) coding for obtaining ranging information is accomodated within the command and telemetry links. The "Bus" contains a wideband turnaround channel which reamplifies the PRN code after demodulation in the "Bus" receiver and transmits it back to earth.

The bus-earth link is shown on Figure 5. 4 with redundant 100 watt power amplifiers. The telemetry data is placed on a square wave subcarrier similar to the Pioneer system. A demodulator and bit synchronizer for this type of system has been built for the Pioneer program. Bit rates of 4096 PPS may be achieved at distances exceeding the maximum earth-Mars separation distance. The 12 foot high-gain antenna must be earth oriented within relatively narrow limits. One means of achieving this which could serve as a primary or a backup mode is to employ a self tracking antenna subsystem. The use of a simultaneous lobing system to achieve error signals as illustrated in Figure 5. 5 has been considered to provide self-tracking for the high gain antenna. A TRW "wabble" drive is proposed for positioning the 12 foot "Bus" antenna.

The "Bus" antenna employed to receive from and transmit to the "Lander" is a helical antenna 9-1/2 in. in diameter and 20 in. long with a flat ground plane 28 in. in diameter. It is mounted on the instrument package, which looks at Mars.

RTG Powered Orbiter Solar Powered Flyby - These configurations differ from the solar powered orbiter configuration in the power amplifier power output and in the high-gain antenna diameter. The characteristics of the bus-earth communication link for these configurations is summarized on Table 5. 7.

The use of a 36 foot antenna for the RTG powered orbiter results in a narrow beam width (about 0.8 degree at the 3 db points). While dynamic studies on "Bus" - antenna orientation have not been carried out, it is believed that a self tracking antenna subsystem will provide adequate "Bus" - antenna orientation.

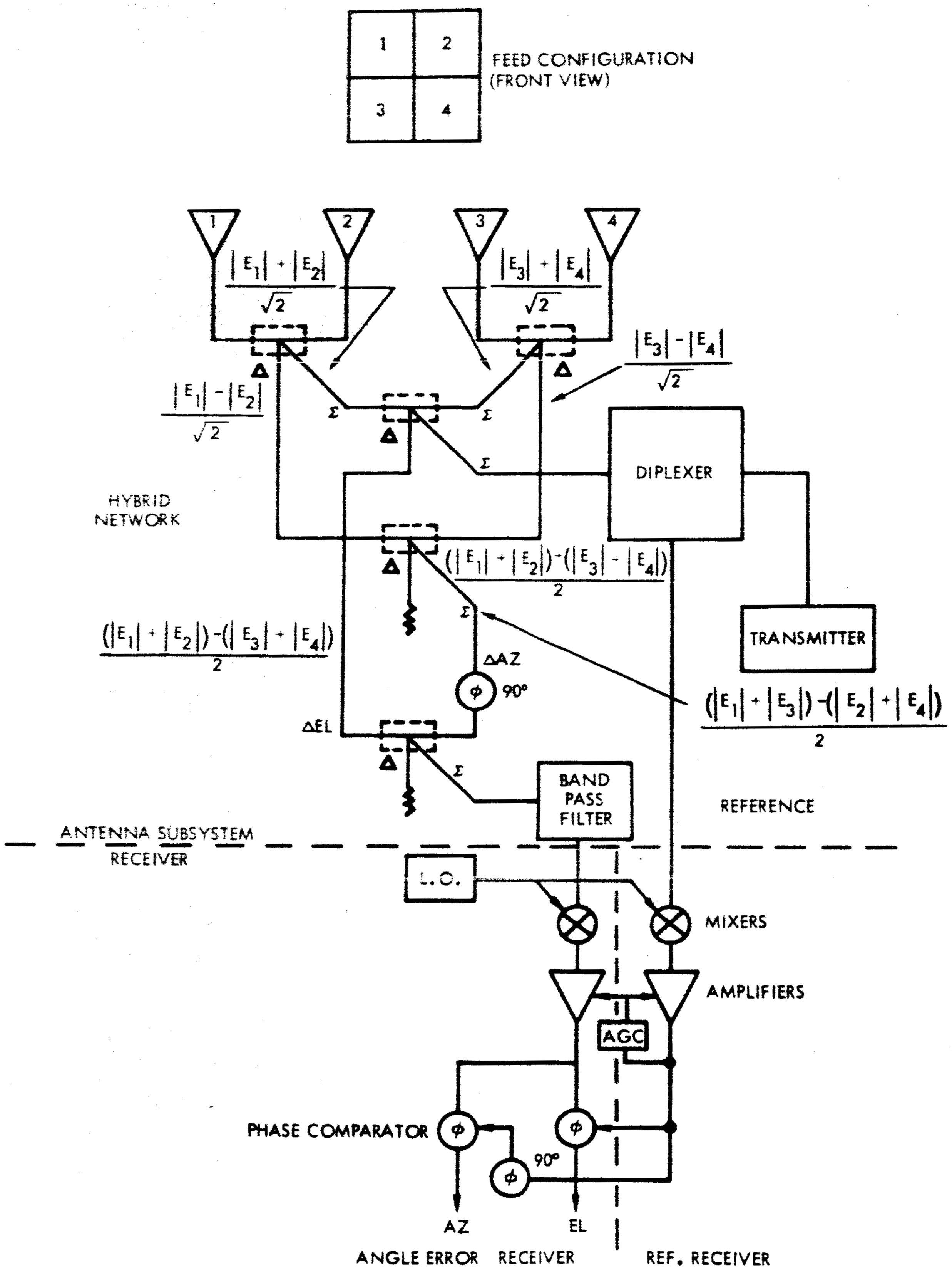



Figure 5.5. Block Diagram - Self-Tracking Antenna Error Channels

Table 5.7. Bus to Earth Communication Link Performance For Spacecraft (Orbiter) with RTG Power Supply and For Solar Powered Flyby Spacecraft

| | Spacecraft with RTG Power Supply |
|---|---|
| Frequency, MC | 2,295 |
| Transmitter Power, Watts | 20 |
| Transmitting Antenna | |
| Type | Sunflower Cassegrain |
| Size, Ft. | 36 |
| Gain, db | 45.5 |
| Pointing Accuracy Req., deg. | ± .35 |
| Approx. Weight, Incl. Feed, lbs. | 300 |
| Receiving Antenna | |
| Type | Cassegrain |
| Size | 210 ft (dia) |
| Gain, db | 61 |
| Telemetry Input Capacity | |
| Engr. - Instr. | 56 Analog<br>64 Bi-level |
| Exper. - Scientific | 6 Digital |
| Ref. Range, km | $2.46 \times 10^8$ |
| Bit Rate at Ref. Range, PPS | Approx. 80,000 |
| Range - 2 x Des. Bit Rate, km | - |
| Range - 1/2 Des. Bit Rate, km | $3.48 \times 10^8$ |
| Range - 1/4 Des. Bit Rate, km | $4.92 \times 10^8$ |
| Data Perf. Margin DB | +6.8 |
| Perf. Margin Required (Approx. Sum. of Neg. Tolerances) | +6.5 |

5.4.2 Power

The initial design considerations are presented for the solar powered orbiter; subsequently, differences for the other configurations are considered.

Solar Powered "Bus" (Orbiter) - The power system must provide a reliable source of power during the long trip time to Mars and for several months after arrival at Mars. The sun - mars distance at the time of arrival varies from near minimum (1.38 AU) for a 1969 mission to near maximum (1.67 AU) for a 1975 mission. Since it is desirable to gather experimental data as long as the spacecraft is operating, a design range of 1.67 AU was selected for initial design considerations. The orbit about Mars is also important for design considerations since the solar system must be designed to charge the batteries which are utilized when the bus is eclipsed. For initial design considerations an eccentric orbit (20,000 km by 2,000 km) was assumed which has an orbital period of 14.46 hours. The maximum occult period was assumed which is 2.26 hours when the orbit is in the sun - mars plane.

Solar cell considerations led to the selection of N on P cells since the degradation of these cells from solar flares is significantly less than for P on N cells. Computations indicated that the degradation due to solar flares would be less than 5% over a period of one year.

The conditioned power requirements assumed for initial design considerations are shown on Table 5.8.

A block diagram for the solar powered bus power system is presented on Figure 5.8. Silver cadmium batteries were assumed since magnetic requirements may rule out the use of nickel cadmium batteries. Solar power system design parameters are summarized in Table 5.9 and a weight estimate is presented on Table 5.10. Redundancy was assumed for the batteries, battery charger and power conditioning equipment.

RTG Powered "Bus" (Orbiter) - The RTG considered were Snap 19 units and optimized units. Optimized units could provide power system weights comparable with solar systems, however, they may not be available. Consequently the use of Snap 19 units, which results in power system weights significantly higher than that for solar powered systems, was assumed. Estimated power requirements for this configuration are shown on Table 5.8. A block diagram for this configuration is presented on Figure 5.9 and a weight estimate is summarized on Table 5.10.

Table 5.8. Assumed Spacecraft Power Requirements at Mars

| | Power Requirements - Watts | |
|---|---|---|
| | Solar Powered Bus (Orbiter) | RTG Powered Bus (Orbiter) |
| Communications | 300 | 100 |
| Thermal Control | 20 | 20 |
| Attitude Control | 30 | 30 |
| Central Computer and Sequencer | 15 | 15 |
| On Board Experiments | 100 | 100 |
| TOTAL | 465 | 265 |
| TOTAL ASSUMED FOR DESIGN CONSIDERATIONS | 500 | 300 |

**Features:**

1. Two SNAP 19's in series, 4 strings of RTG's in parallel
2. Low raw power bus voltage
3. Low voltage batteries and charge controls
4. Redundant batteries
5. Redundant "dummy load" or shunt regulators to limit bus voltage
6. Redundant low voltage power conversion

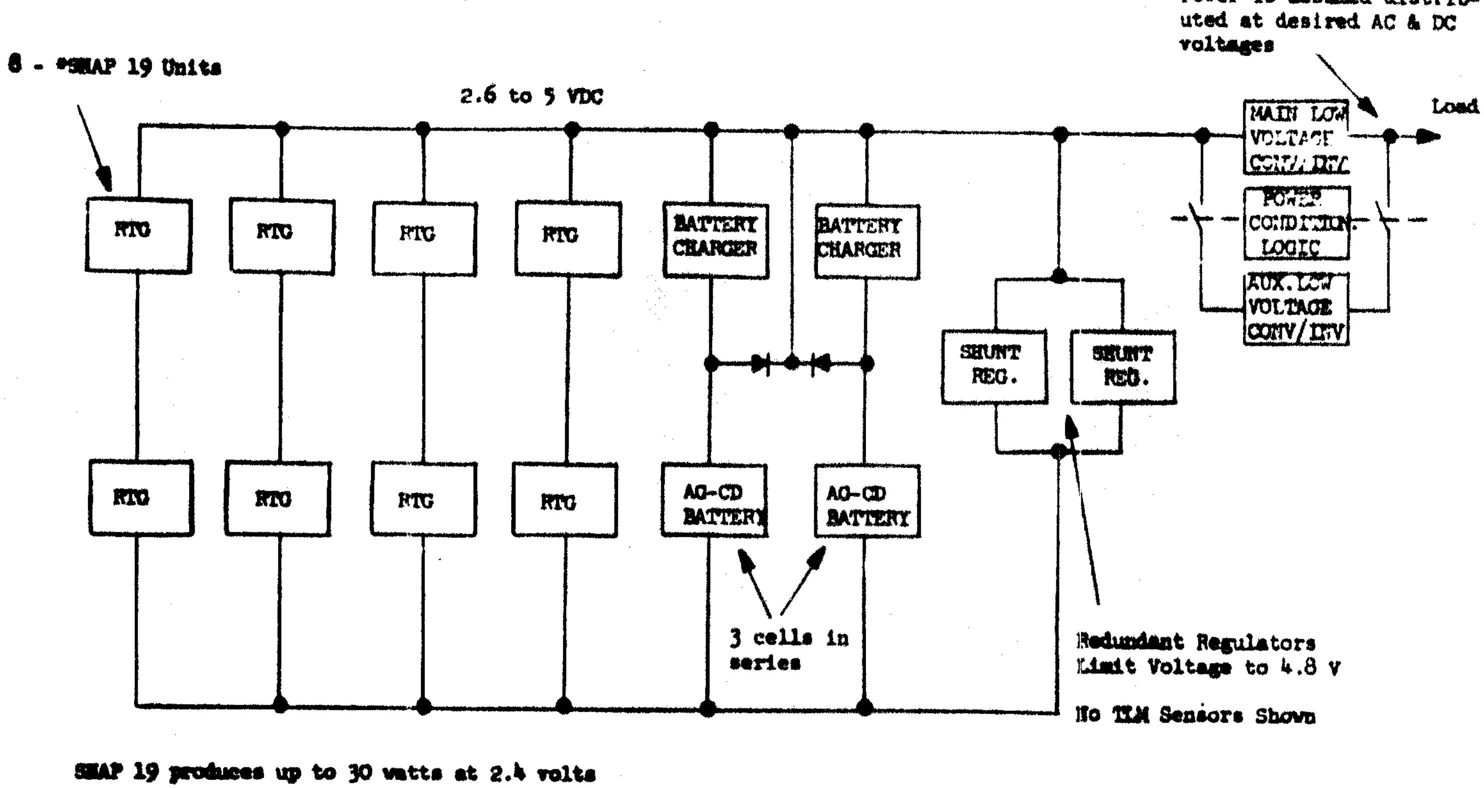



Figure 5.8 Solar Powered Bus (Orbiter Block Diagram

Table 5.9 Solar System Design Parameters

1. Solar Array Characteristics
(Evaluated for 2300 km circular orbit)

| | | | |
|---|---|---|---|
| Distance from Sun, AU | 1.38 | 1.54 | 1.67 |
| Solar intensity, $mw/cm^2$ | 73.5 | 59.0 | 50.1 |
| Subsolar temperature, $^oC$ | 16 | 4 | -2 |
| Terminator temperature, $^oC$ | -1 to -107 | -13 to -107 | -23 to -107 |
| Solar array power output, $watts/ft^2$ | | | |
| Subsolar | 5.23 | 4.25 | 3.62 |
| Terminator | 5.60 | 4.50 | 3.74 |

2. Solar array area and battery power requirements per 100 watts of conditioned power.

$$A = \frac{100}{E.W} + \frac{100}{E.W.E_c E_b} \quad \frac{Te}{T_s} = \frac{133}{W} + \frac{198}{W} \quad \frac{Te}{T_s}$$

Where A = array area in square feet

E. = power conditioning efficiency for load power (assumed = 0.75)

$E_c$ = power conditioning efficiency for battery charge (assumed = 0.90)

$E_b$ = battery charge efficiency (assumed = .75 for silver cad batteries)

$T_e/T_s$ = ratio of occult time to time in Sun

$$B = \frac{100 \ Te}{E. \ E_b D}$$

Where B = battery power required per orbit, watt hours

Te is the occult time per orbit

D is the depth of discharge (assumed = .8 for eccentric orbit 2000 km x 20,000 km and = .5 for 2000 km circular orbit)

3. **Solar System Weights**

**Solar array - 1.13 $lbs/ft^2$**

**Batteries - 16 watt hours per pound**

**Battery charging equipment - 3.33 pounds/100 watts**

**Solar array control - 2 pounds/100 watts**

**Load conditioning equipment - 10 pounds/100 watts**

Table 5.10. Power System Weights and Volumes

| | Solar Orbiter | RTG Orbiter |
|---|---|---|
| Assumed conditioned power requirements, Watts | 500 | 300 |
| Assumed Design Range, AU | 1.67 | |
| Solar Array area required, $ft^2$ | 230 | |
| Snap 19 Weight, lbs. | - | 465 |
| Solar Array Weight, lbs. | 260 | |
| Battery Weight, lbs. | 119.5 | 9 |
| Battery Volume, $ft^3$ | .98 | .06 |
| Battery Charging Equipment Weight, lbs. (100% redundant) | 33.3 | 6 |
| Battery Charging Equipment Volume, $ft^2$ (100 % redundant) | .7 | .12 |
| Source Control Weight, lbs. | 10 | - |
| Source Control Volume, $ft.^3$ | .20 | - |
| Load Conditioning Equipment Weight, lbs. (100% redundant) | 100 | 60 |
| Load Conditioning Equipment Volume, $ft^3$ (100% redundant) | .92 | .55 |
| **Total System Weight | 521 | 540 |
| *Total System Volume, $Ft^3$ | 2.77 | .73 |
| **Total System Weight (100% Redundant Batteries), lb. | 640 | 549 |

*Plus RTG units or Solar array.
**Not including harnessing and cabling.

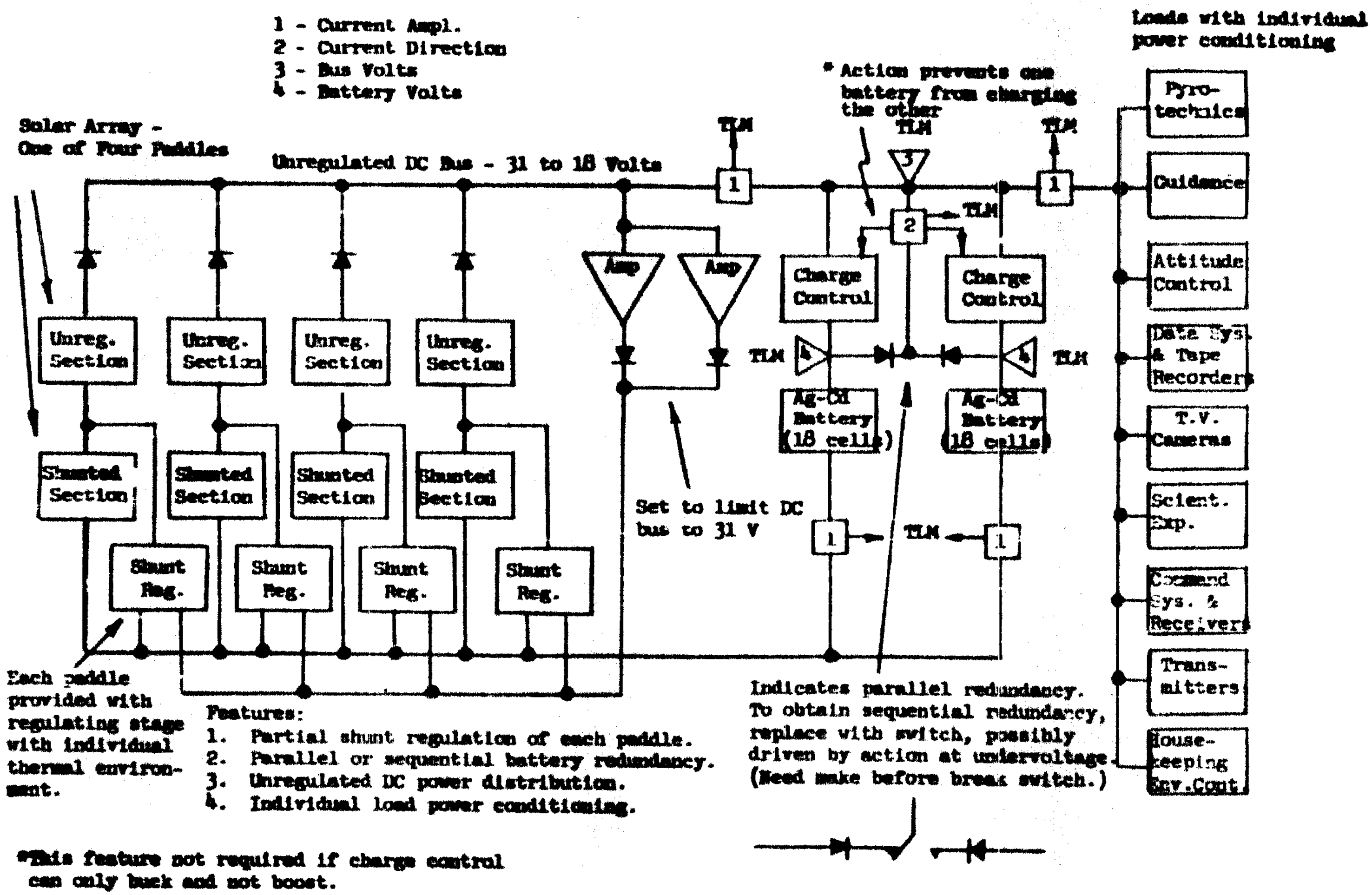



Figure 5.9. RTG Powered Bus (Orbiter) Block Diagram

### 5.4.3 Propulsion

The assumptions made for initial design considerations are presented for the solar powered "Bus".

Solar Powered "Bus" (Orbiter) - The propulsion system must provide thrust for midcourse maneuvers of the "Bus" - "Lander" configuration and thrust for de-boost for the "Bus" configuration into Mars orbit.

A single engine propulsion system using storable liquid propellants and a low thrust level of 600 pounds was specified for the "Bus" to perform both the midcourse and de-boost maneuvers. The single storable liquid engine was selected over a storable liquid midcourse engine plus a solid de-boost engine since the solid arrangement was heavier and would not provide as much mission flexibility (off loading of fuel for payload for some missions). The low thrust of 600 pounds was selected for initial considerations because

a) it is sufficiently high so that a negligible effect on de-boost efficiency results
b) propulsion system weight and engine dimensions decrease with decreasing thrust
c) structural weight (extended items) decreases with decreasing thrust
d) the thrust level is compatible with reasonable ablative nozzle engine burning times (1000 - 1500 seconds)
e) a low thrust level is desirable for midcourse maneuvers.

The propellant selected for initial considerations was 75% $N_2H_4$ and 25% MMH as fuel and $N_2O_4$ as oxidizer. The MMH was added to the fuel to depress the freezing level of the fuel mixture to that of the oxidizer. The amount of propellant capacity selected for the initial considerations was to be sufficient to execute a midcourse correction $\Delta V$ of 300 ft/second with 8000 pound bus-lander and to place a 4000 pound bus less midcourse fuel into a 2000 km circular orbit ( $\Delta V$ = 2.20 km/sec for impulsive maneuver) when approaching Mars with a $V_\infty$ value of 3.05 km/sec. This amounts to a usable propellant capacity of approximately 2,270 pounds at a specific impulse of 294 seconds.

A block diagram for the preliminary propulsion system design is shown on Figure 5.10. Series valve assemblies are used for propellant shut off valves (both explosive and engine shut off). A cross link is incorporated to permit either fuel or oxidizer pressurization subsystems to provide pressure to both tanks in case of a gross malfunction of one of the pressurization subsystems. The propulsion system characteristics are summarized in Table 5.11. The weight estimate shown is based on empirical data for similar propulsion units.

RTS Powered "Bus" (Orbiter) - Assuming that the lander and bus weights for this configuration are approximately the same as those for the solar powered bus, the considerations given above are applicable to this configuration. This system would utilize the same components as the solar powered bus. The difference is in the configuration arrangement.

### 5.4.4 Central Computer and Sequencer

The computation and sequence operations for the Mars mission have not been fully established. It is possible that a programmer alone may be adequate with computations done on earth and transmitted to the spacecraft. For initial considerations, however, it has been assumed that some computational capability will be required on the spacecraft in view of the long times required for transmission of data between the earth and the spacecraft, and the desire to achieve flexible spacecraft operations. The functions which the CC & S is assumed to perform includes:

1) Receiving commands from the decoder, verifying the commands, and translation of the commands into computational and programming activities on board the spacecraft.
2) Receiving information from operational sensors and the translation of this information into proper housekeeping and operating commands.
3) Translating experimental and engineering data into telemetry format with earth control of the telemetry format and data transmission modes.
4) Providing timing signals for controlling events and indexing data.
5) Performing calculations for interpreting pointing and attitude instructions for trajectory control or for aiming the mapping package.
6) Performing malfunction detection and executing recovery routines.

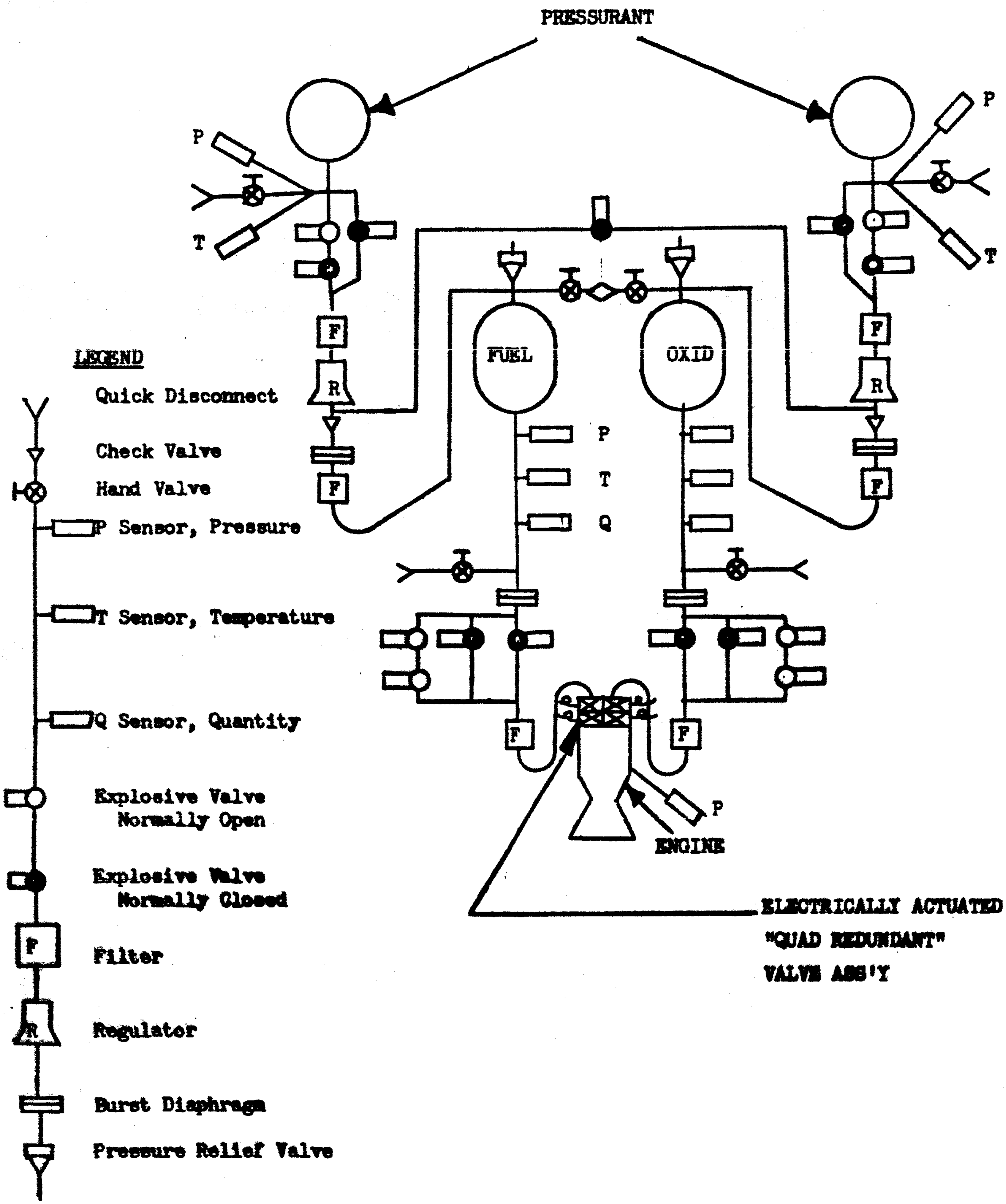



Figure 5.10. Sample Design Bipropellant System Schematic

Table 5.11. Propulsion System Characteristics, Solar Powered "Bus"

| | |
|---|---|
| Thrust | 600 lbs. |
| Propellants | |
| Fuel | 75% $N_2H_4$, 25% MMH |
| Oxidizer | $N_2O_4$ |
| Mixture Ratio | 1.418 |
| Chamber Pressure | 100 PSI |
| Nozzle expansion ratio | 40 |
| Specific Impulse | 294 seconds |
| Usable propellant, lbs. | 2270 |
| Total Engine Length | 24 in. |
| Nozzle Diameter | 13.2 in. |
| Approximate Total Propulsion System Weight (Dry) | 325 lbs. |

**Approximate Power Requirements**

**Explosive valves (7), momentary 5 amps at 28V each**

**Solenoid valves (2) engine operation 4 amps at 28V each**

It was assumed that the prime requisite for the mission was to achieve a "minimum success" type of mission. Thus since the state-of-the-art permits the design of extremely reliable programmers and less reliable computers, the programmer would be designed to provide for mission success with reduced flexibility even if the computer failed. The programmer would detect spacecraft errors and take pre-determined corrective actions.

A preliminary estimate of the characteristics of a CC & S with the capabilities described above, was prepared for initial design considerations. A preliminary block diagram of the system is shown in Figure 6-11.

Command verification is required. One way to accomplish this is to receive command signals from earth and transmit these signals to earth for verification prior to execution. However, since the two way transmission time is long (about 22 minutes at $2 \times 10^8$ km ), other means of achieving command verification should be studied. The programmer would be receptive to earth control of telemetry format and telemetry transmission modes.

Timing signals for timing, controlling events and for indexing data will be generated by a master clock. Since accuracies of less than 1 part in $10^6$ should be adequate, no timing problems are anticipated.

Magnetic circuits, which are extremely reliable, were assumed for the programmer. Integrated circuits were assumed for the computer since integrated circuits lend themselves to computer operation while magnetic circuits do not. While reliability data on integrated circuits are not available, the use of integrated circuits should result in an order of magnitude increase in reliability over the use of discrete circuits.

A preliminary estimate of the CC & S weights, volumes, power requirements, and parts count is given below:

a) Approximate weights and volume:

| | Weight, lbs. | Volume, cu. in. |
|---|---|---|
| Programmer | 7 | 400 |
| Computer | 15-20 | 800 |
| Power Supply | 4 | 500 |
| Miscellaneous | 5 | |

b) Approximate power requirements:
Programmer operations only - 10 watts
Short term (with computer operations) - 40 watts

c) Approximate parts count:
Programmer - 4000 discrete parts
Computer - 1000 discrete parts + 200 integrated circuits

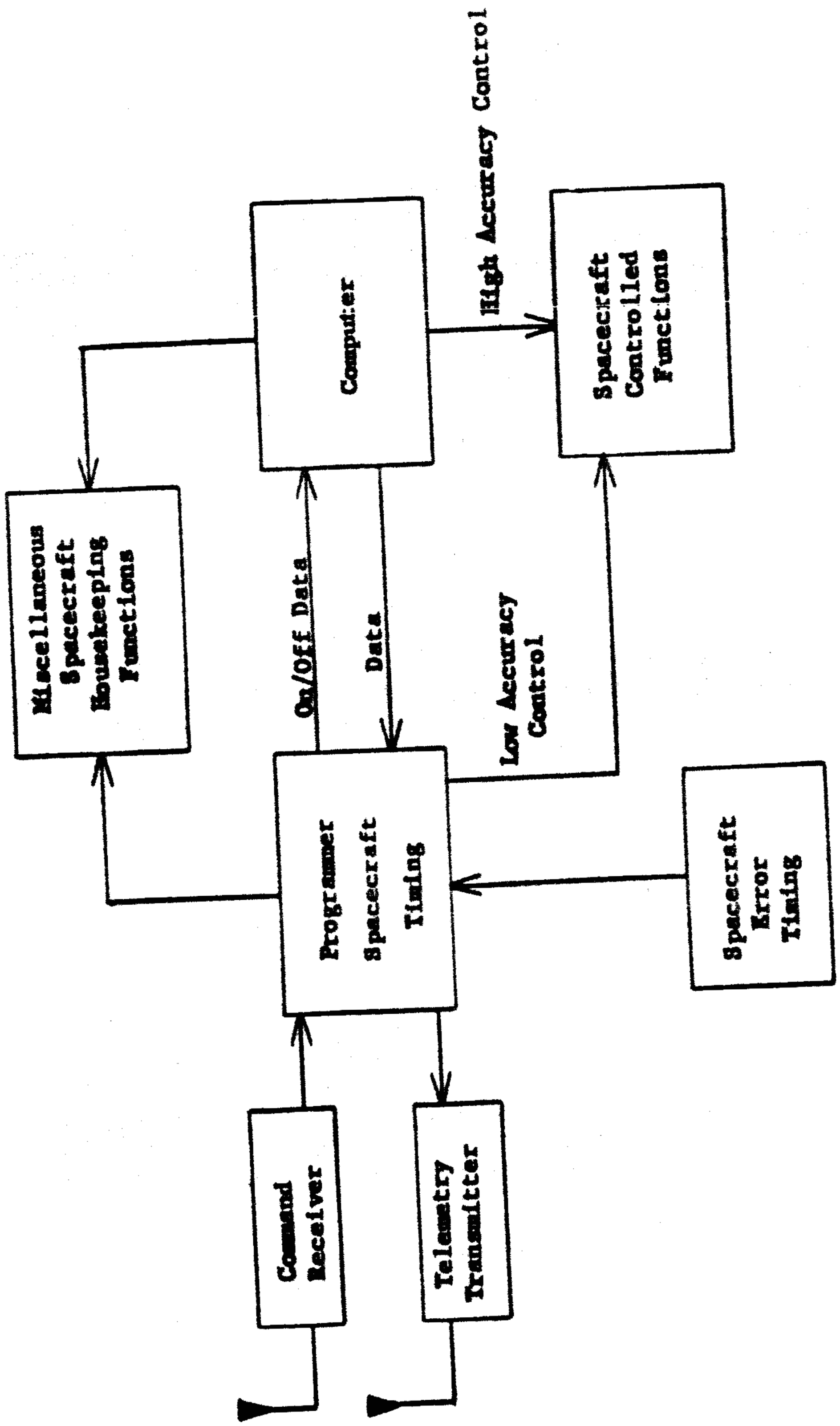



Figure 5.11. Block Diagram - Computer Subsystem

5.4.5 Structure

The basic bus structure subsystem includes the composite of load carrying members providing mounting points or surfaces for all of the other defined subsystem components, i.e., power, propulsion, attitude control, thermal control, communications and data, mapping package and other experiments. It also includes the supporting structure for the lander package together with its sterilization container, the structural interstage supporting the spacecraft and all separation devices for the bus and the lander.

Vastly differing structural approaches between the two launched spacecraft were followed primarily because of the fact that one spacecraft uses solar power and has a gimballed 12-ft diameter antenna while the other uses RTG power and has a non-gimballed 36-ft diameter antenna which before deployment is stowed in a folded-petal configuration. The former also is structurally supported off the lander vehicle, which must provide load paths to the Centaur upper stage. The latter employs a supporting truss structure which bypasses the lander vehicle.

5.4.6 Thermal Control

The spacecraft thermal control system must contend with a wide variety of environmental factors. The primary external factor is the decrease of the solar flux to 40% of its initial value over the duration of the mission. The spacecraft-induced environment is comprised of many factors: the distribution and duty-cycling of electrical power, the effects of propulsion engine firings, the thermal flux from external elements such as solar arrays or RTG units, the attitude maneuvers of the spacecraft, the presence of the lander, etc.

In the face of this environment, the general electronic equipment must be maintained approximately between the limits of 20°F and 100°F. Special requirements are imposed by batteries, engine propellants, experimental sensors, optical systems, solar arrays, etc.

Solar Powered Configuration - The overall thermal design selected is to house the equipment in a well insulated enclosure with louver controlled radiating surfaces which normally face away from the sun. The inner surface of the radiating panels will serve as the mounting panel for the most of the equipment, with the panel serving as a fin to distribute heat from the higher powered components. Low powered components need not be well coupled to the radiating panels.

The total radiating area, which may be made up of several separate radiating surfaces, is sized for the maximum power condition, including heat inputs to the radiators from external elements such as solar arrays. The radiators could be located on the vertical sides, where they are partially blocked by the solar arrays, or on the aft surface, where they are partially blocked by the lander sterilization container. The number of bus compartments should be kept to a minimum so as to minimize the surface area for heat loss through insulation and minimize the need to balance dissipated power among several compartments.

RTG Powered Configuration - The general design criteria are the same as for the solar powered configuration. Even though essentially constant power is available from the RTG supply, there will probably be a need for active radiator control (e. g. , by louvers), partly because of local duty cycle operation and partly because of changing inputs from engine firings, attitude maneuvers, etc. The bus configuration using RTG units, employs radiators facing slightly sunward, but they are adequately shielded from the sun by the large antenna. The equipment compartments must be well isolated thermally from the hot RTG units.

The Optical Systems - The fixed telescope in the main compartment introduces stringent thermal requirements: temperature gradients in the structure holding the optical elements must be held to a low level; and temperature gradients in the optical elements must be minimized, especially non-symmetrical ones. These requirements lead to a design in which the entire optical system is well insulated from space and from the bus by multilayer radiation insulation around the structure and by low conductance pa ths at the structural mounting points. Thermostatically controlled heaters may be used along the structure to minimize the structual gradients. Recently developed TRW "heat pipes" provide a possible alternate means of making the structure uniform. An insulated thermal door covers the opening to complete the insulation of the system, and is opened only as needed for picture taking. The temperature level of the optical system can be maintained by fixing its temperature such that the (small) net heat flux is outward, and making up the loss by well distributed electrical heaters.

The optical system of the mapping package imposes the same general requirements which can be met by the same design approach as for the fixed telescope. The smaller size of the optics in the mapping package tends to reduce the severity of the problem. All sources of heat should be kept out of the insulated region of the optical systems, so any optical system electronics, other experiments or subsystem components (e. g. , the attitude control system's Mars sensor) in the mapping package must be well insulated from the optical system. It may be desirable to physically divide the mapping package into two packages, mechanically tied together, for this reason.

Other Thermal Considerations - The solar array, in order to survive the potentially long eclipses at Mars, must have a relatively low back surface emissivity to reduce rate of temperature decay in eclipse. This will increase the operating temperature and reduce power output near the earth.

The propulsion system engine must be relatively well insulated from the spacecraft to minimize the heating of spacecraft components during and shortly after an engine firing. Heating by plume impingement can either be eliminated by geometrical arrangement or by use of an insulating shield on the impinged surfaces. Control of the propellant temperature level, as well as that of other engine components during the transit periods, will require insulation to minimize heat loss to space, with any heat loss made up electrically. Alternatively, absorption of solar energy, even though time-varying, might be used to make up the heat loss since the engine is normally exposed to the sun.

Experiments tend to have somewhat greater thermal problems than typical spacecraft electronic components because, (1) the experiment sensor is often relatively more temperature sensitive, (2) certain experiments require open ports, and (3) they are often located external to the main spacecraft compartments, thus requiring their own thermal control system. This may be achieved by use of insulation, a fixed radiating area and thermostatically-controlled heaters.

Certain subsystem components, such as the Mars sensor in the mapping package, have characteristics similar to some experiments, as described above, and their thermal treatment can be similar.

Gimbals, hinges, and moving cable bundles may have to be insulated and provided with make up heat to maintain them at suitable operating temperatures.

Lander Interface - The lander is arranged in the sterilization container with the heat shield toward the bus and the lander RTG units facing aft. This eases the heat removal problem for the lander (the aft half of the sterilization container radiates away the waste heat from the RTGs with an unobstructed view of space) and at the same time minimizes the thermal effect of the lander on the bus. Additional insulation could be provided between the bus and lander if necessary. Retention of the top half of the sterilization container will minimize any change in the thermal environment due to lander ejection.

### 5.4.7 Attitude Control

The Attitude Control Subsystem includes the reference sensors, the computer and generating the control torques. The sequencer will provide the command inputs for maneuvers.

The baseline system requires tracking on the sun, on Canopus and employing gyros for preserving the reference during maneuvers. The secondary configuration which would require pointing at Earth probably will depend on tracking the transmission from Earth using a self tracking antenna. An Earth sensor for operating near the Sun line would be an extremely difficult development problem.

Available sensors will provide accuracies of 0.1 degree. The OGO sun sensor achieves this performance with a $17^{\circ}$ field of view. The Canopus tracker developed by JPL is achieving this accuracy with a $4^{\circ} \times 11^{\circ}$ field of view. It appears practical to obtain 0.1 mr precision by reducing the FOV to about $1^{\circ}$ (or using a two-stage sensor). Gyros are capable of 8 mr/hr. These would achieve 0.1 mr for about 40 seconds or so during maneuver.

The baseline system contemplates a roll maneuver for taking pictures. The system can remain locked on the sun, but a fixed Canopus tracker would break lock. Two schemes for controlling the roll maneuver have been considered. One consists of tracking Mars with a planet sensor and controlling roll to keep pointing at the planet center. The other consists of using the gyros to measure roll angles and operating with roll commands transmitted from earth and stored in the sequencer.

Observation of Mars might be accomplished by locking on the sun for longitudinal orientation and by employing a Mars sensor in the articulated TV package to lock on Mars for roll control. This would normally provide for viewing Mars along the intersection of the orbit plane with Mars. Other view angles could be achieved by programming off-set look angles for the Mars sensor (which may be simpler than providing an additional degree of freedom for the entire articulated package). This would not work in the region where the axis of the articulated package is parallel to the Mars - sun line where gyros would be required to keep the articulated package on the proper side of the spacecraft-mars line.

Depending on gyros to point at Mars will require low drift gyros and short observation times. It will be necessary to check the performance of this system by monitoring the position of the optical axis by some means such as wide angle photographs of the planet, horizon scanners, etc.

In any event, relatively precise pointing is required in order to permit mosaicing pictures together. If one adopts the criterion that the optical axis must point at the selected spot on the surface within 10% of the FOV ($3\sigma$), then the curves of Figure 5.12 shows the angular precision required. These curves are based on the requirements that contiguous coverage (10% precision) is required in each mode. It is obvious that one must either use better gyros or provide for continuous tracking of Canopus in order to take advantage of a zoom lens with TV. Alternatively, one could provide for an additional degree of freedom on the articulated TV package.

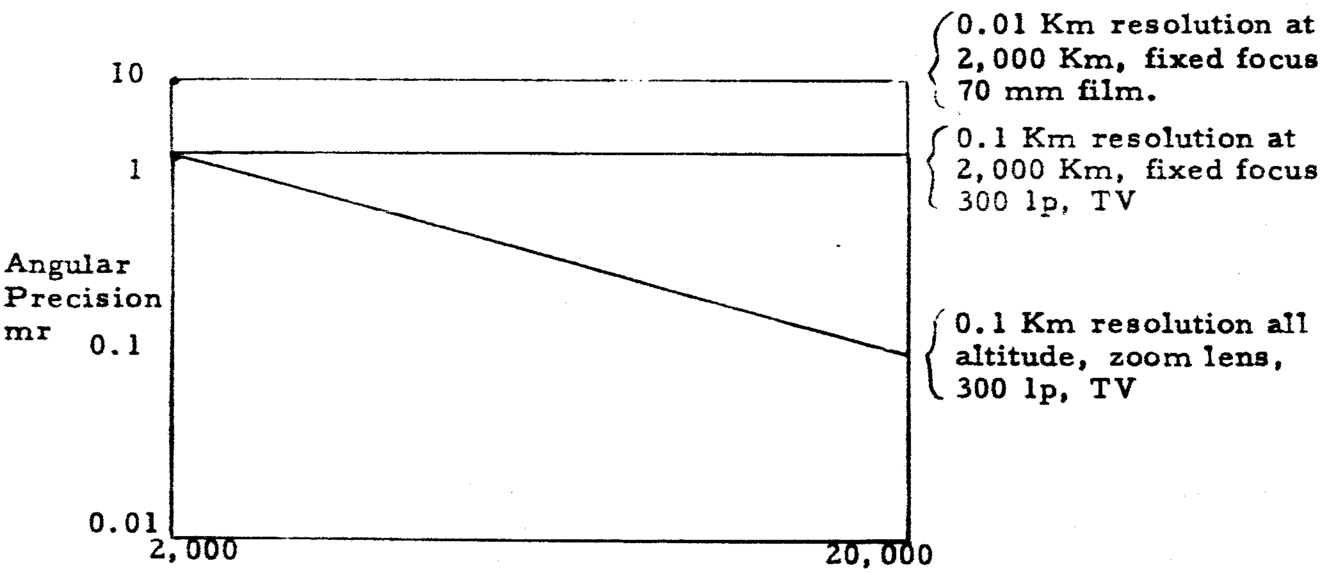



Figure 5.12. Angular Resolution Requirements

The error budget for maneuvers including allowable time (away from lock) has not yet been calculated. It may prove practical to monitor other stars seen by the Canopus tracker during roll to correct gyro drift.

By depending on the sequencer and executing the attitude changes one axis at a time, it appears practical to use analog integration to keep track of attitude during maneuvers. Although the drift error budget has not been calculated, it appears that analog integrators can be made good enough to work with the best gyros.

Control torques will be generated by gas jets, thrust vector control, and, possibly, by reaction wheels and solar trim panels. Maneuvering of the spacecraft for boosting will require large gas jets. It is practical to consider saving gas through the use of solar trim panels during cruise, but these are not part of the baseline configuration. Even the large gas jets cannot cope with expected engine thrust misalignment and/or shifts of c.g.. Consequently, means for directing the thrust will be included. Electric actuators driving a gimballed chamber and nozzle are the baseline. However, jet vanes may be preferred. Control of the orbital body rates to $10^{-4}$ rad/sec, or less, during picture taking will very likely require reaction wheels. Multi-stage gas jets will also be considered.

### 5.4.8 Imaging Systems

The baseline imaging system has been selected to provide the capability to perform a variety of jobs on command. The continuous operation of all the sensors would provide for more data than can reasonably be expected to be transmitted and so considerable ground control is assumed.

The assumed reliability doctrine which has been followed for the planet-oriented sensors (in order of priority) is

a) Maximize probability of obtaining overall coverage of the planet with visible pictures (3 colors) at a ground resolution of 1 km, or less.

b) Maximize probability of obtaining coarse resolution IR and UV spectrophotometric measurements over broad (known) regions of the surface and atmosphere.

c) Maximize probability of obtaining fine resolution IR and UV spectrophotometric measurements over known regions of the surface and atmosphere.

d) Maximize the probability of obtaining high resolution (order of 0.01 km) coverage of selected regions of the planet surface.

e) Maximize the probability of obtaining data from scanning IR radiometer, microwave radiometer, polarimeter, etc.

Since the system is bandwidth limited, the problem of designing the imaging subsystem becomes one of providing the flexibility to permit best utilizing the available bits. It is likely that the definition of "best utilization" will change as time goes on. For example, on the first shot, overall photo-TV coverage to provide some aerodidic control and to sort out regions of interest, together with spectrometric investigations of atmospheric composition, will be most important. On later flights, analysis of specific areas including high resolution photography, measurements of spectral emissivity, polarization of light, etc., will be most important. These arguments indicate the need for some potential for changing functions between missions. This time will be very short, of the order of one year, so the flexibility must exist in the system from the beginning.

### 5.4.8.1 Picture Taking Subsystem

The baseline configuration contains the elements to provide this flexibility. An articulated package low/intermediate resolution picture taking and IR and UV spectrophotometry. A few small photometers or polarimeters may also be included. A large fixed package will provide high resolution photography.

Articulated Package - The articulated package for the baseline design is 18" x 42" x 44", weighs about 240 lbs. and contains

a) Three TV cameras with 38 mm dia optics of various focal lengths which will achieve 1 km resolution or better.

b) One 3-way optical system 24" f/4 lens providing
   (1) $5^{\circ}$ on-axis FOV for 70 mm film on TV with 0.1 km resolution or better.
   (2) Off-axis (approx. $5^{\circ}$) FOV for an IR spectrophotometer.
   (3) Off-axis (approx. $5^{\circ}$) FOV for an UV spectrophotometer.

c) Associated electronics and (for film version) developer and readout.

Fixed Telescope - The fixed telescope is approximately 28 in. overall diameter and 60 in. long exclusive of film transport, development and readout. The clear aperture is 24 in. and focal length 85 in. The total weight including film mechanisms is 400 lb. This system will achieve around 0.01 - 0.005 km at 2,000 km altitude over a 7-1/2 in. x 7-1/2 in. picture ($5^{\circ}$ FOV).

### 5.4.8.2 Mars Sensors

Image motion compensation (IMC) may be required. 10% of the value in each of two axes is achievable with the sensor on LOPP. On the other hand, it may be simpler to compute the expected image motion and make an open-loop correction. IMC is not required on the TV systems for 1 km resolution. It may not be required on the 0.1 km resolution system at all, and will only be required on one axis if the spacecraft orientation and Articulated Package angle are properly chosen.

In order to compute V/h for use in open-loop IMC, it will be necessary to know V, h, and the angles between the orbit and the line to the center of Mars. Furthermore, as was mentioned in Section 5.4.7, it will be necessary to know the direction of the optical axis to within 0.1 mr or so in order to control TV coverage. These factors impose stringent requirements on the determination of the orbit parameters (instantaneous spacecraft position and velocity relative to Mars in the sun-Canopus coordinate system).

It has been shown that all orbital parameters can be computed except $\Omega$, the longitude of the ascending mode. By observing the planet during terminal maneuvering it will be possible to establish a position accuracy at least as good as the lander entry corridor. If this is $\pm$ 500 km, it appears that $\Omega$ can be inferred to an accuracy of $\pm 2^{\circ}$. Continuous monitoring of the output data may result in refining the data. However, due to the potential of error due to gyro drift and the communication time lag, it appears necessary to measure the relative position of the planet center, or to make measurement from which the location of the planet center can be calculated on earth.

The possibilities for direct measurement are horizon scanners, total planet sensors (such as RES), and wide angle pictures showing enough of the planet disc to permit determining the relative position of the planet sensor.

Horizon scanners which are presently available offer precision of about 0.25 degree, or 5 milliradians. Possible future development such as the TRW $CO_2$ horizon scanner appear capable of around 1 milliradian in near earth orbit. If similar conditions exist on Mars, 0.1 mr precision may be possible. However, successful application of this device depends on having considerably more knowledge of Mars than is currently available.

Taking pictures with a wide-angle lens so as to include the entire planet disc in the field of view appears capable of providing a precision of about 1 milliradian. This accuracy is adequate for 1 km resolution TV or 0.01 km resolution film, but not for 0.1 km TV.

Taking pictures in several small fields of view which include some of the limb and some stars is potentially capable of an accuracy of 0.1 mr or better. Such a device will be most practical if it turns out that an accurate planet reference measurement once an orbit is adequate. If measurements must be made for any position of the terminator, the accuracy will fall off to about 1 mr (unless more complex optics can adjust for a partially illuminated disc.).

5.4.8.3 Frame Indexing and Matching

It is assumed that an indexing accuracy of 10% of the width of the picture is adequate to provide assurance of contiguous coverage without undue overlap for straight picture making. The implications of this assumption are summarized in Figure 5.12. If stereo coverage is required, this precision might not be adequate, but the waste appears tolerable - even for stereo requirements.

As the spacecraft orbits Mars, the relative orientation of the picture format changes with respect to the surface. Additional overlaps of pictures will be required in the regions of high angular change. This will not reduce the pointing accuracy required, but will require additional pictures in these regions. An alternative technique is to refrain from taking pictures at these times, and obtain coverage on subsequent orbits. This latter technique is most attractive when film is used because the large field of view provided by film will render easier the job of properly relating the disconnected strips. For an all TV system, with frames 300 resolution elements on a side, the reconstruction will be more difficult.

With the possible exception of a scanning radiometer, the scientific data will be at such low spatial resolution that indexing and matching will more or less amount to identifying the coordinates of the optical axis within a half a frame width. Scanning radiometer data will probably be useful only in those parts of the orbit in which the scan axis is more than 45 degrees from the ground track, and the precision suitable for visible pictures will be more than adequate for identifying this data.

### 5.4.8.4 Data Handling

TV output data can be digitized and recorded or recorded in analog form and digitized on readout. Although the rates of input bandwidth to output bandwidth will not be appreciably altered by the latter technique, the upper frequency limit of the recorded can be significantly reduced (a factor of 5 or so). Analysis of data has been based on a 5 bit quantizing level. Methods of reducing the number of bits per resolution element appear to offer savings in telemetry capacity, but no quantitative results are available at present. Each TV picture will contain approximately $10^5$ resolution elements ($5 \times 10^5$ bits if quantized). Space-qualified recorders with 1000:1 input/output bandwidth ratios are available with a capacity of $10^8$ bits, or space for 200 TV pictures. At 1 km ground resolution (300 km x 300 km frame), this provides enough storage for three TV cameras for one orbit. At 2,000 bits/sec, the telemetry system can readout $10^6$ bits in one orbit. (At 10,000 bits/sec - $5 \times 10^6$ bits/orbit). Consequently, continuous photography of the ground with TV is not practical - even at 1 km ground resolution.

Film output data can be readout at a speed compatible with the telemetry data rate. No intermediate storage (except a one word buffer) is required. In addition, the coverage provided by a 70 mm film frame (at 150 lp/mm readout) will represent $5 \times 10^8$ bits in one frame. This indicates the practicability of taking only a few pictures. It is this reasoning which lead to the idea of a fixed telescope capable of taking a small number of high resolution pictures.

In this regard, the 7-1/2 in. frame size available in the 24 in. aperture telescope is capable (at 2,000 km) of 0.01 km ground resolution. At this resolution, the capacity of one frame is about $4 \times 10^9$ bits. Providing the capability of reading out at 1 km as well as 0.01 km will permit reducing the "quick look" capacity to about $4 \times 10^5$ bits which can be transmitted in one orbit for rapid interpretation. Interesting areas can then be examined in greater detail.

### 5.4.9 Sterilization

Spacecraft sterilization procedures are discussed in Appendix A.

# 6. ATLAS/CENTAUR SPACECRAFT CONCEPTS

The spacecraft designs considered is Section 5 required Saturn 1B/Centaur launch systems becuase the heavy landers (4,000 lbs) increased the spacecraft weights to approximately 8,000 lbs for 2,000 x 20,000 km orbits about Mars. While detailed studies of lander systems of this size have not been made, it appears that smaller landers may be adequate to obtain necessary "engineering" data for the manned Mars missions. A survivable lander is mandatory for the precursor missions, but an elaborate surface "laboratory" is not essential for present purposes.

An attractive class of precursor spacecraft can be launched with Atlas/ Centaur vehicles, which appears adequate for an orbiter or flyby/lander mission in the favorable years. An Atlas/Centaur plus kick stage (7,000 lbs) is significantly more useful because an orbiter plus lander can be launched in 1971 and 1973, and an orbiter plus small entry capsule (non-survivable) in 1975. The availability of the high performance ($H_2$-$F_2$) kick stage is uncertain at present; consideration should be given to use of existing stages for this purpose.

## 6.1 Assumptions

It was assumed that a minimum-weight survivable lander capable of accomodating 50 lbs of experiments on the surface of Mars was required, plus an orbiter with approximately 80 lbs of experiments, including a medium resolution TV system. It was found that a spacecraft of the above scientific experiment payload capacity could be launched with the Atlas/Centaur plus kick stage in 1971 and 1973; an orbiter plus small entry capsule could be accomodated in 1975. A science payload of this magnitude can accomplish all high-priority experiments stipulated for the precursor missions, although photo mapping from the 2,000 x 20,000 km orbit can be accomplished with marginal resolution only. An alternate mode is to replace the 700-lb lander with a high resolution mapping package (Mars Orbiter Photographer).

The ground rules and assumptions for the design of the Atlas/Centaur class launch systems are given as follows:

| | Orbiter | Lander |
|---|---|---|
| Bit Rate | 3000 bits/sec | 100 bit/sec* |
| Transmitter | 10 watts | 7 watts |
| Power Source | Battery & Solar Array** | RTG |
| Antenna | Omni to Lander<br>9 ft Dish to Earth | Approx. 3 db |
| Frequency | 2300 mc | 200 mc |
| Science Weight | 80 lbs | 50 lbs |
| Stabilization & Control | 3 axis (Earth Oriented) | Spin |
| Life Time After Lander Impact on Mars | 90 days | 30 days |
| Separation | Explosive Bolts | Explosive Bolts |

* After Lander impact on Mars, 600 bits/sec during descent

** Solar Array Area = 100 $ft^2$

Further assumptions for the orbiter propulsion system are:

**Orbiter Vehicle**

Retropropulsion

| | |
|---|---|
| Solid | Aluminized Rubber-base |
| Thrust (Average) | 2500 lbs. |
| Propellant Density | 0.061 $lbs/in^3$ |
| $I_{sp}$ | 295 sec. |
| $\Delta V$ (for apoapsis = 20,000 km) | 1.64 km/sec |

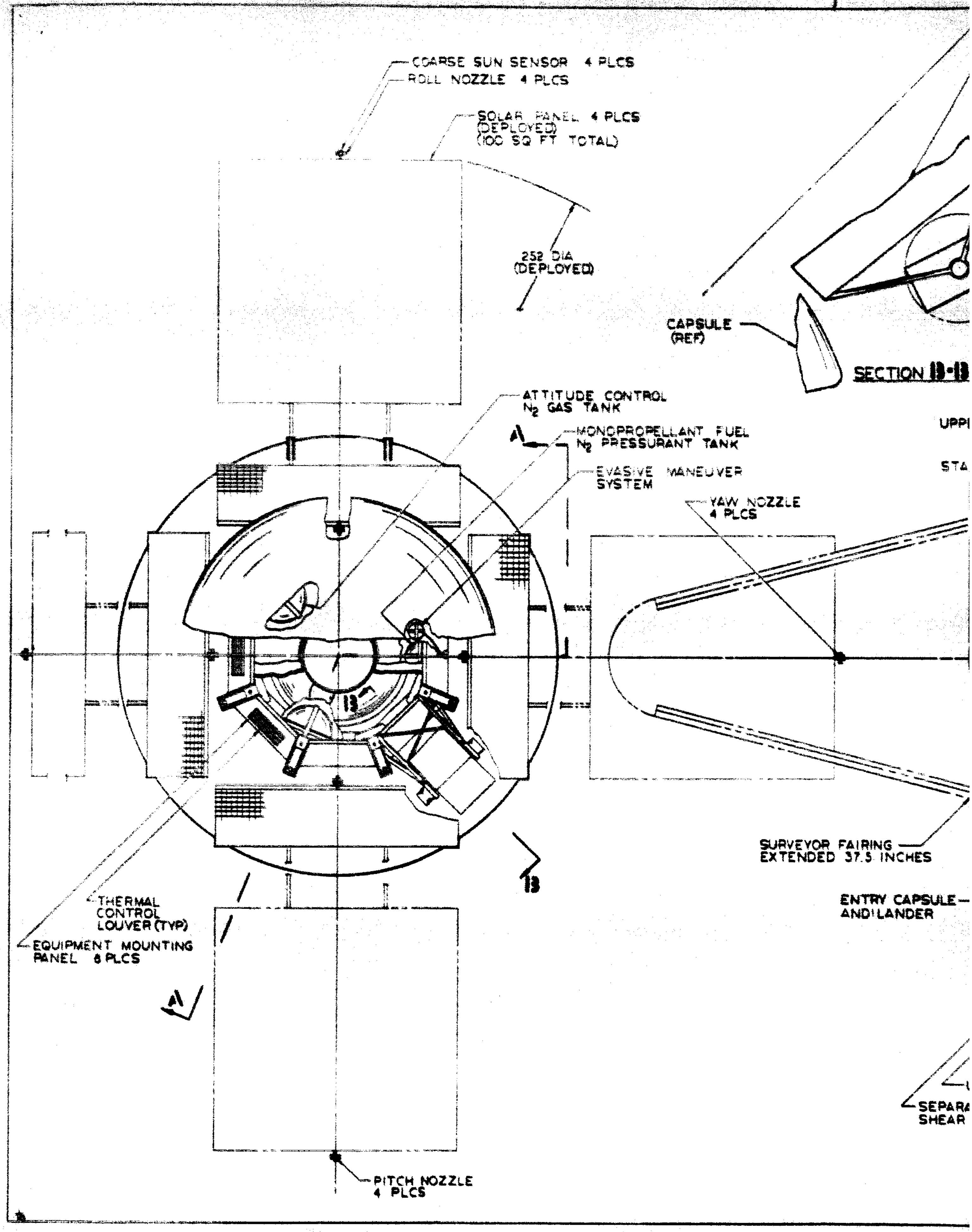
Midcourse Propulsion

| | |
|---|---|
| Monopropellant | Hydrazine |
| Thrust | 50 lbs |
| Propellant Density | 62.6 $lbs/ft^3$ |
| $I_{sp}$ | 230 sec |
| $\Delta V$ | 75 m/sec |
| Tank Pressure | 250 psi |
| Pressurization Gas | Nitrogen |

A basic consideration in the development of the configuration of the bus/orbiter was the choice of the bus-antenna orientation mode. Two techniques were studied: 1) articulated parabolic antennas with sun-oriented spacecraft, and 2) body-fixed parabolic antennas with earth-oriented spacecraft. It was found that body-fixed antennas are larger for a given bit rate and power level because the solar arrays can be oriented away from the sun by angles of 45 degrees, and the loss in power must be compensated by a larger antenna diameter. The articulated antennas are substantially heavier, however because of their driving mechanisms. For most missions the worst-case design requirements, which involve both sun distance and sun angle, result in comparably sized solar arrays for a given power requirement. It was decided therefore, that on the basis of a higher bit rate capability with relatively low-power transmitters, a large body-fixed earth-oriented antenna and an earth oriented solar array are superior.

Both solid and bi-propellant liquid retropropulsion systems were investigated. Liquid systems have an advantage in that a single engine can be used for both retro and midcourse propulsion and eliminate (in the configurations studied) the separation joint which jettisons the portion of the lander sterilization container immediately adjacent to the bus. However, the weight, size and complexity of the liquid system tankage, plumbing and ullage control systems appear to outweigh its advantages for this mission. Additionally, better midcourse correction control is obtainable from a small liquid monopropellant thruster. Therefore, solid propellant motors were selected for retropropulsion and liquid monopropellant systems for midcourse correction propulsion. It appears feasible to off load the solid retro motor to convert from an orbiter to a flyby vehicle, if desired.

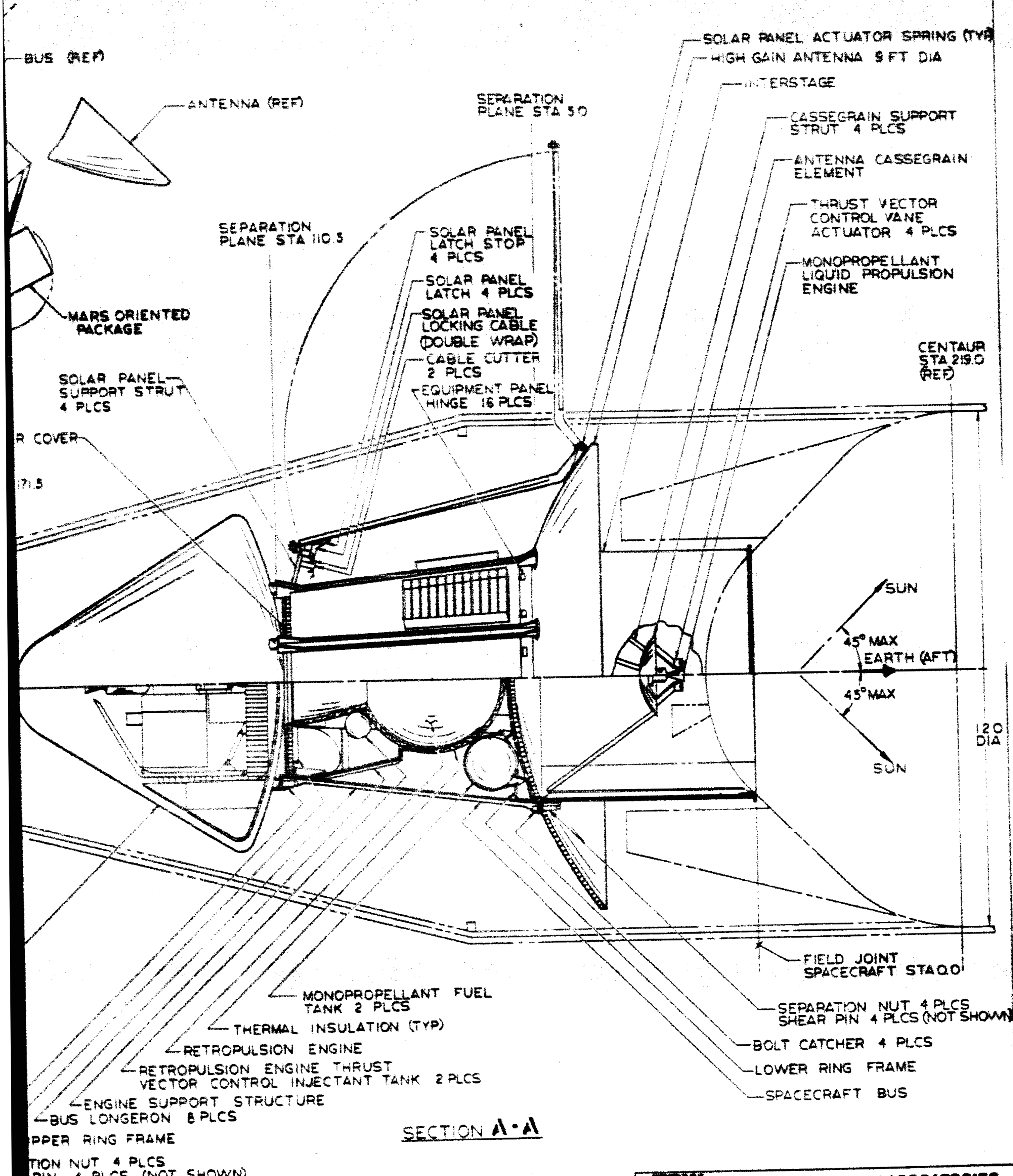
Three basic spacecraft design concepts were developed based upon the subsystem concepts described above. Figure 6.1 depicts a spacecraft system incorporating a Mars orbiting bus and a survivable lander. Figure 6.2 shows a spacecraft concept incorporating a "flyby" bus and survivable lander, and Figure 6.3 illustrates a configuration based on an orbiting bus and a non-surviving entry capsule; Figure 6.4 shows the spacecraft of Figure 6.1 installed on an Atlas/Centaur booster.



COARSE SUN SENSOR 4 PLCS
ROLL NOZZLE 4 PLCS
SOLAR PANEL 4 PLCS (DEPLOYED) (100 SQ FT TOTAL)
252 DIA (DEPLOYED)
CAPSULE (REF)
SECTION B-B
ATTITUDE CONTROL $N_2$ GAS TANK
MONOPROPELLANT FUEL $N_2$ PRESSURANT TANK
EVASIVE MANEUVER SYSTEM
YAW NOZZLE 4 PLCS
A
A
B
B
SURVEYOR FAIRING EXTENDED 37.5 INCHES
ENTRY CAPSULE AND LANDER
THERMAL CONTROL LOUVER (TYP)
EQUIPMENT MOUNTING PANEL 8 PLCS
PITCH NOZZLE 4 PLCS


Figure 6.1

2

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| | |
|---|---|
| MARS ENVIRONMENT STUDY | 20 SEPT 1965 |
| INBOARD PROFILE, BUS, ENTRY CAPSULE & LANDER | PD87-06 |

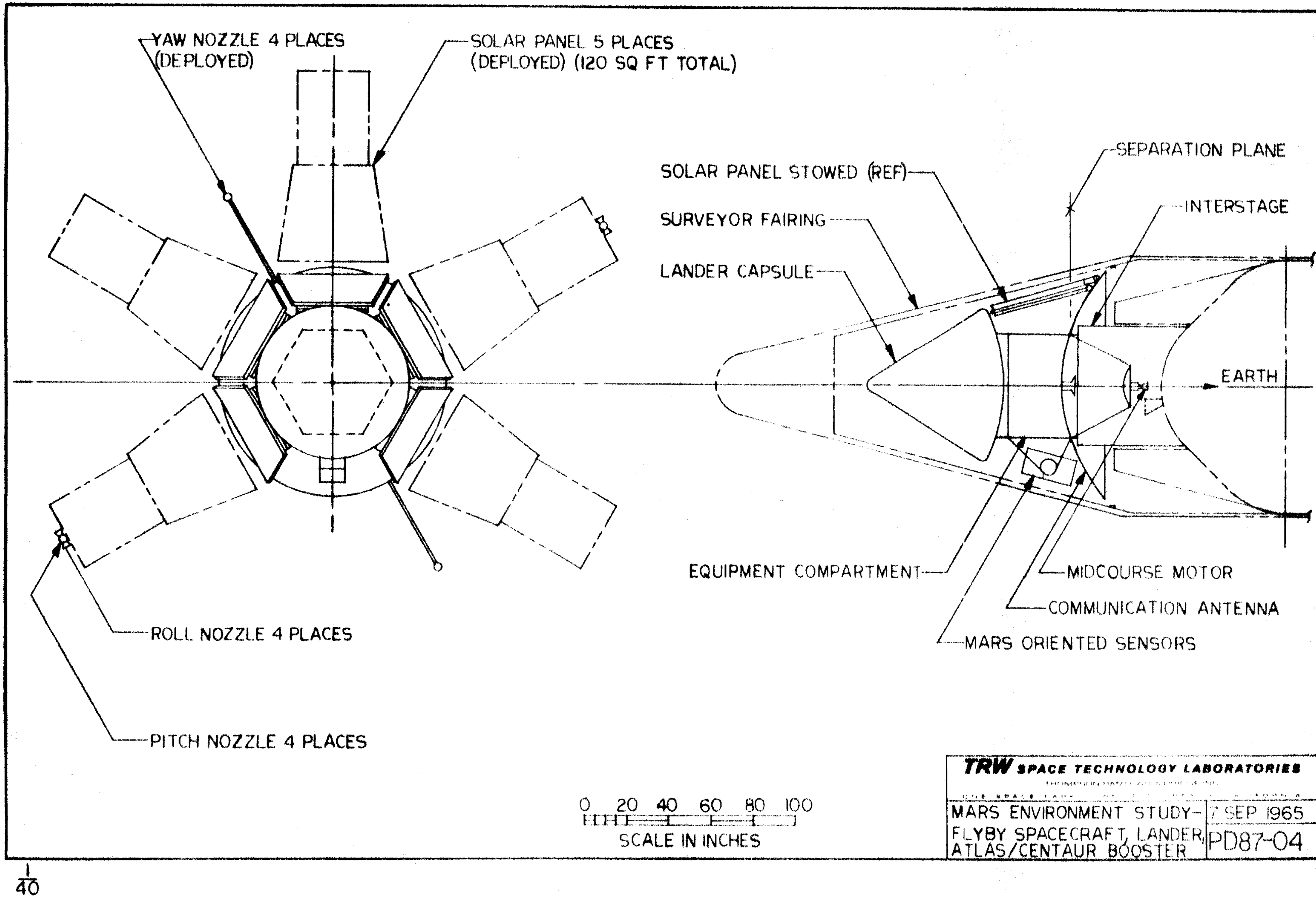
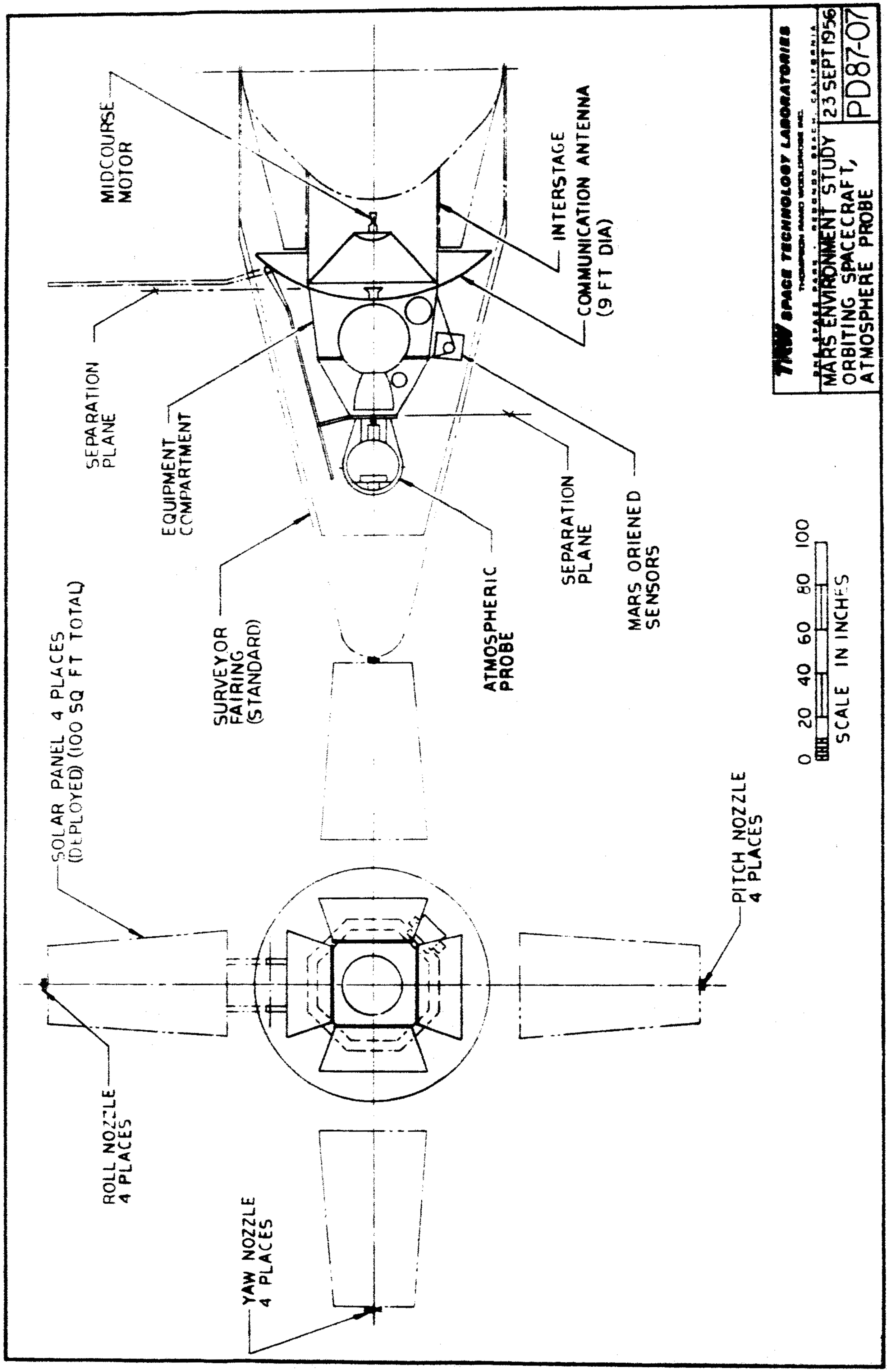
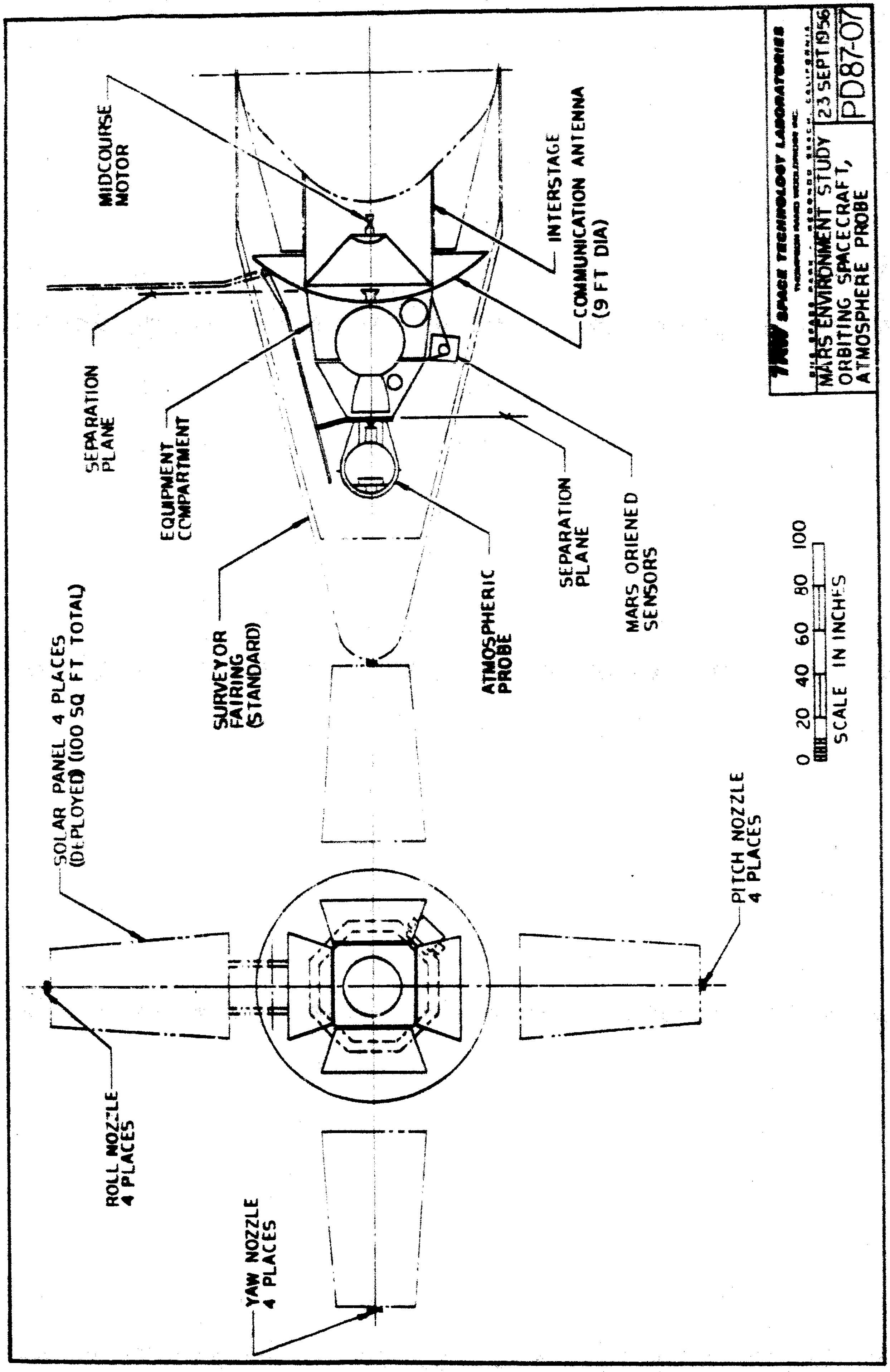
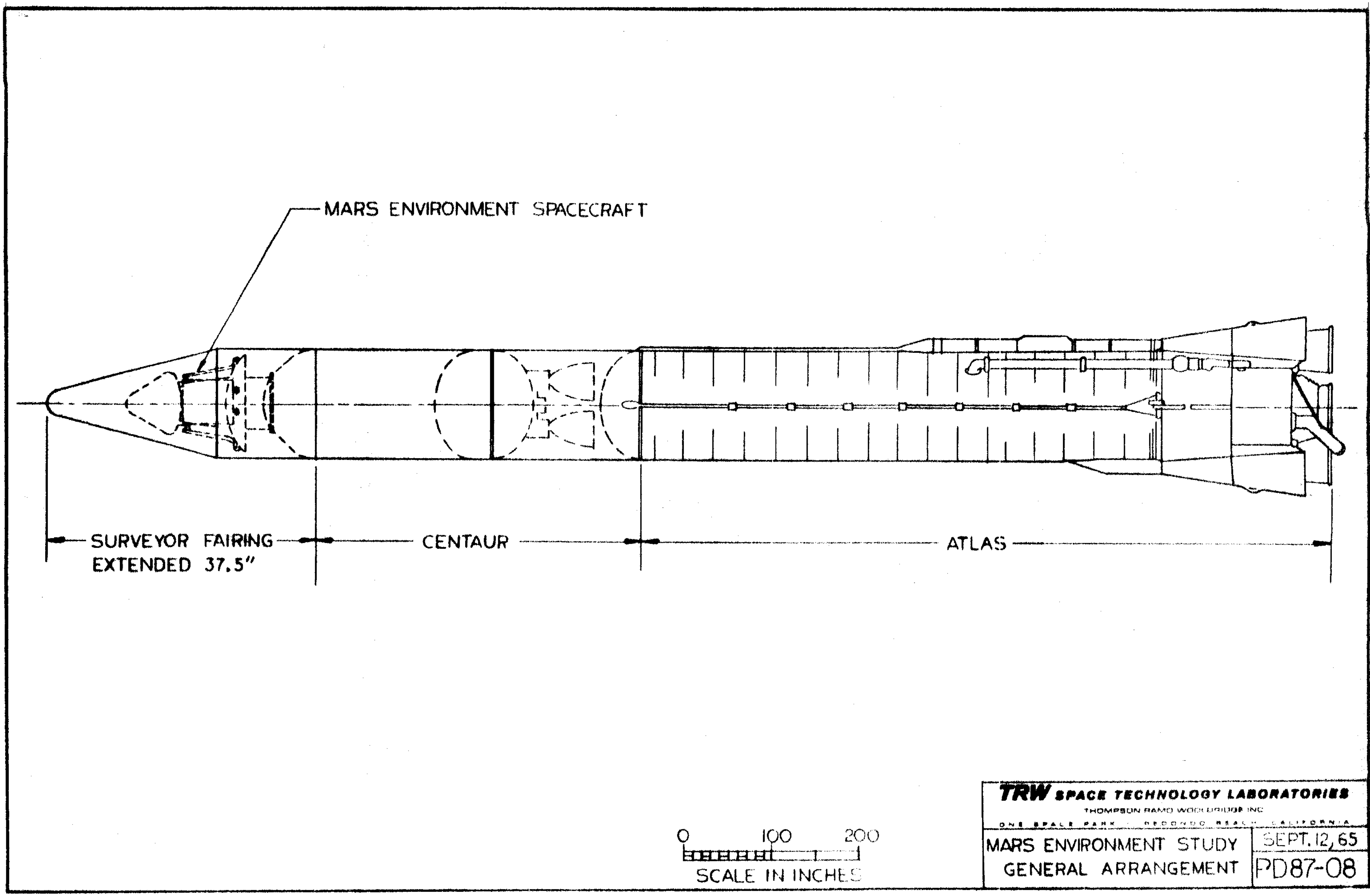

YAW NOZZLE 4 PLACES
(DEPLOYED)
SOLAR PANEL 5 PLACES
(DEPLOYED) (120 SQ FT TOTAL)
SEPARATION PLANE
SOLAR PANEL STOWED (REF)
INTERSTAGE
SURVEYOR FAIRING
LANDER CAPSULE
EARTH
EQUIPMENT COMPARTMENT
MIDCOURSE MOTOR
COMMUNICATION ANTENNA
MARS ORIENTED SENSORS
ROLL NOZZLE 4 PLACES
PITCH NOZZLE 4 PLACES
0 20 40 60 80 100
SCALE IN INCHES
TRW SPACE TECHNOLOGY LABORATORIES
MARS ENVIRONMENT STUDY-
FLYBY SPACECRAFT, LANDER,
ATLAS/CENTAUR BOOSTER
7 SEP 1965
PD87-04
1/40

Figure 6. 2


MIDCOURSE MOTOR
INTERSTAGE
COMMUNICATION ANTENNA (9 FT DIA)
SEPARATION PLANE
EQUIPMENT COMPARTMENT
SEPARATION PLANE
MARS ORIENED SENSORS
SURVEYOR FAIRING (STANDARD)
ATMOSPHERIC PROBE
SOLAR PANEL 4 PLACES (DEPLOYED) (100 SQ FT TOTAL)
ROLL NOZZLE 4 PLACES
YAW NOZZLE 4 PLACES
PITCH NOZZLE 4 PLACES
0 20 40 60 80 100
SCALE IN INCHES
TRW SPACE TECHNOLOGY LABORATORIES
MARS ENVIRONMENT STUDY
ORBITING SPACECRAFT, ATMOSPHERE PROBE
23 SEPT 1956
PD87-07


MIDCOURSE MOTOR
INTERSTAGE
COMMUNICATION ANTENNA (9 FT DIA)
SEPARATION PLANE
EQUIPMENT COMPARTMENT
SEPARATION PLANE
MARS ORIENED SENSORS
SURVEYOR FAIRING (STANDARD)
ATMOSPHERIC PROBE
SOLAR PANEL 4 PLACES (DEPLOYED) (100 SQ FT TOTAL)
0 20 40 60 80 100
SCALE IN INCHES
PITCH NOZZLE 4 PLACES
ROLL NOZZLE 4 PLACES
YAW NOZZLE 4 PLACES
TRW SPACE TECHNOLOGY LABORATORIES
MARS ENVIRONMENT STUDY
ORBITING SPACECRAFT, ATMOSPHERE PROBE
23 SEPT 1956
PD87-07


MARS ENVIRONMENT SPACECRAFT
SURVEYOR FAIRING
EXTENDED 37.5"
CENTAUR
ATLAS
0
100
200
SCALE IN INCHES
TRW SPACE TECHNOLOGY LABORATORIES
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GENERAL ARRANGEMENT
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Figure 6.4

## 6.2 Bus/Orbiter Design Concept

The bus of Figure 6.1 is 60.5 inches long and forms an eight-sided truncated pyramid. It is approximately 60 inches in diameter at the aft end and 46 inches in diameter at the forward end.

The use of eight facets in the bus provides the following advantages:

1) Boost loads are kept at low levels and are spread uniformly about the structure by incorporating eight longerons at the corners of the bus.
2) A bus cross section of eight facets approximates a circular cross section and provides maximum clearance between the facets and the retropropulsion engine, enhancing equipment installation.
3) Sufficient equipment mounting surface (including provision for growth) is available on five of the bus facets so that the three facets immediately adjacent to the Mars oriented package (MOP) can be left blank. This allows the electronic equipment to balance the weight and the relatively long moment arm of the MOP.
4) The eight-sided bus provides optimum support for the quadrature geometry of the solar panels and attitude control nozzles as well as support for the MOP.

The forward and aft ends of the bus longerons are interconnected by ring frames. The lower ring frame helps resist the small kick load developed by the change of load path direction.

Trapezoidal shaped flat honeycomb sandwich panels form the sides of the bus compartment and are bolted to the longerons and upper and lower ring frames. The interior surfaces of these panels are used to mount the bus equipment and act as thermal heat sinks for the equipment.

Hinges are provided at the aft edges of the equipment mounting panels which permit them to rotate down when the panel attach bolts are removed, giving Access to the bus interior and to the panel mounting equipment for installation, test and maintenance is thereby provided without the necessity of removing the panel from the bus or disconnecting equipment unnecessarily.

The use of a single bus compartment for all the interiod equipment minimizes structural weight, maximizes thermal coupling, including the tankage, permits optimum cable routing and minimizes the number of connections made within the bus. The design permits the location of particular items of equipment on the various panels to take into account the thermal and power distribution requirements, the mass property distribution of the overall spacecraft and the desired close association of highly interconnected components.

The exterior surfaces of the equipment mounting panels are furnished with active thermal control louvers which allow the thermally radiating positions of the panels to view and radiate to deep space when cooling is required.

All exterior non-louvered portions of the equipment mounting panels and all other exterior surfaces of the bus are thermally insulated from the external environment.

A thermally insulated upper cover of honeycomb sandwich material closes off the forward end of the bus. The aft end of the bus is covered by the parabolic antenna, which is also fabricated from honeycomb sandwich material. In the area of the bus, the interior surface of the antenna is thermally insulated.

The parabolic antenna, upper cover and the equipment mounting panels, all of which are fabricated from honeycomb sandwich materials as noted, provide the micrometeorite protection of the bus interior.

A monopropellant liquid hydrazine propulsion system is used for midcourse corrections. A 50-pound thrust engine is mounted, on centerline, to the cassegrain element of the parabolid antenna at the extreme aft end of the spacecraft. The engine uses four electrically-driven jet vanes for thrust vector control. The hydrazine is stored in a single spherical tank mounted to the forward face of the parabolic antenna; a gaseous nitrogen propellant pressurant tank is also mounted to the parabolid antenna.

The retropropulsion system for deboost at Mars incorporates a 2500-pound thrust solid motor located on the bus centerline with the nozzle pointing forward. The engine is supported by a semimonocoque truncated cone structure. The aft end of the cone forms a ring which provides uniform support to the engine housing, and the upper end of the cone is scalloped in eight places with the scallop points picking up fittings attached to the corner longerons and to the upper ring frame. Liquid freon stored in two spherical tanks attached to the engine, injection is used for thrust vector control. To protect the engine grain from vacuum effects during transit, the nozzle is sealed until the engine fires. As the engine is aft-pointing, it is necessary to jettison the lower portion of the entry capsule sterilization container before the engine fires.

Communication between earth and the spacecraft is obtained with the aid of a body-fixed, earth-oriented, nine-foot diameter parabolic antenna. The antenna utilizes a horn feed located at the apex of the dish and a cassegrain reflector which is supported by a quadripod whose struts attach to fittings mounted on the parabolic antenna dish. The honeycomb sandwich structure of the antenna dish is reinforced in order to support the propellant tankage and the solar panels. A communications power budget and equipments are shown in Tables 6.1 and 6.2. A low gain omni-antenna is mounted to a solar panel and provides "on stand" and "near earth" communications when the spacecraft cannot view the earth.

The lander-to-spacecraft antenna (not shown) is also mounted to a solar panel. The antenna has a 120-degree total view angle and has essentially the same longitudinal field of view as does the MOP.

Power for the bus is obtained from a 100 sq. ft. solar array supported by four 5 x 5 ft. panels. The panels are symmetrically located about the principal axes of the spacecraft and attach, through hinged joints, to the outer periphery of the parabolic antenna.

In the stowed (launch) position the panels are folded back against bus mounted support struts and are held in that position by a latch and double-wrapped cable system. When panel deployment is desired, a signal actuates redundant cable cutters which sever the cables.

The solar panels have been located far aft on the spacecraft to make effective use of the maximum diameter of the fairing, and to minimize the shadowing of the panels by spacecraft component during normal cruise flight.

The orientation of the array, with the cells fixed to the aft surfaces, minimizes the plume impingement of the solid retro motor on the solar cells. The monopropellant liquid engine's plume contains no carbon or metallic components and is of low density, therefore, no adverse effects on the solar

Table 6.1 Orbiter-to-DSIF Communication Power Budget

Frequency: 2295 mc

Modulation: 3K bps coherent PSK on 6000 cps sq. wave subcarrier

| Parameter | Value |
|---|---|
| Total Transmitter Power (10 w) | 40.0 dbm |
| Circuit Loss (including diplexer) | 1.6 db |
| Orbiter Antenna Gain (9 ft dish) | 33.2 db |
| Orbiter Antenna Pointing Loss | 1.0 db |
| Space Loss ($R = 2.5 \times 10^8$ KM) | 267.6 db |
| Polarization Loss | 0.1 db |
| DSIF Antenna Gain (210 ft dish) | 61.0 db |
| DSIF Circuit Loss | 0.2 db |
| Net Transmission Loss | 176.3 db |
| Total Received Power | -136.3 dbm |
| Receiver Noise Spectral Density ($T_S = 25^\circ K$) | -184.8 dbm/cps |
| **Carrier Performance** | |
| Carrier Modulation Loss (Mod. Index = 1.1 radians) | 6.9 db |
| Received Carrier Power | -143.2 dbm |
| Carrier Loop Noise BW ($2\,B_{L0} = 12$ cps) | 10.8 db |
| Threshold SNR in $2\,B_{L0}$ | 6.0 db |
| Threshold Carrier Power | -168.0 dbm |
| Performance Margin | +24.8 db |
| **Data Performance** | |
| Data Modulation Loss | 1.0 db |
| Received Data Power | -137.3 dbm |
| Data Noise Bandwidth (2048 bps) | 33.2 db |
| Threshold SNR in Data Bandwidth ($P_e^b = 5 \times 10^{-3}$) | 7.3 db |
| Threshold Data Power | -142.7 dbm |
| Performance Margin | +7 db |

Table 6.2 Orbiter Communication Equipments

| | Wt (lbs) | Power (watts) |
|---|---|---|
| S-Band 10 watt Transmitter (TWT, dc-dc converter) | 5.0 | 40.0 |
| Beacon Solid State 1/2 watt Transmitter (136 mc) | 2.0 | 1.5 |
| Lander Telemetry Receiver (200 mc) | 2.0 | 1.2 |
| Diplexer | 1.5 | - |
| Telemetry Unit (including 25,000 bit core memory) | 5.0 | 1.2 |
| S-Band Command Receiver | 3.5 | 0.4 |
| Command Decoder | 3.5 | 0.4 |
| Programmer-Sequences | | |
| Tape Recorder | | |
| Diplexer | 1.5 | |

array from plume contamination or from plume thermal radiation are anticipated. The solar panels also provide the structural support of the attitude control coarse sun sensors and the roll, pitch and yaw nozzles.

The spacecraft is fully attitude controlled. The attitude control system uses a single high pressure gaseous nitrogen tank installed on the parabolic antenna. Two sets (six per system) of attitude control thrusting nozzles are arranged in redundant pairs about each control axis (yaw, pitch and roll) of the spacecraft. To achieve maximum moment arms about the spacecraft cg, the nozzles have been installed at the ends of the solar panels. For system simplicity, the pitch and yaw moments are equal, the line lengths between all valves and nozzles are the same in order to equalize the like losses relative to each control axis, and control gas impingement on the spacecraft has been minimized. Also, the attitude control gas lines have been thermally coupled to the warm solar arrays in order to heat the control gas and increase the system efficiency.

Two sets of sun sensors (four elements) have been located immediately outboard of the attitude control nozzles for unimpeded $4\pi$ steradian look angle.

In addition to the complement of scientific experiments body-fixed to the bus, provisions have been made for experiments which require pointing. A fully environmental controlled container is located outboard of the bus as shown in Section B-B of Figure 6.1. The container designated as the Mars Oriented Package (MOP) has a single-axis gimbal which provides a longitudinal look angle approaching $180^{\circ}$. This capability coupled with spacecraft roll enables the MOP to be directed at any portion of the Martian surface or atmosphere. Currently, the MOP accommodates a TV camera system and an IR radiometer experiment.

After launch of the Mars entry capsule from the bus, but before the capsule's injection motor fires, it is necessary to maneuver the bus out of the path of the entry capsule. This evasive maneuver is accomplished by a simple gaseous nitrogen blow down system. A small nitrogen tank is mounted to the parabolic antenna and on signal the gas is expelled through a nozzle located on the longitudinal cg of the bus. This imparts a lateral velocity to the bus so that after a suitable time delay, the capsule may be injected into its Martian trajectory.

Figure 6.2 shows a concept for a Mars flyby bus and survivable lander.

The communication antenna, the MOP and the midcourse correction propulsion system are similar to those described previously. The bus is six-faceted and is sufficiently short that a standard Surveyor fairing may be used. Power is provided by 120 sq. ft. of solar array equally divided between five folding panels as shown. Folding the panels provides adequate solar array area without lengthening the standard fairing.

The roll and pitch attitude control nozzles are supported by the solar array but the yaw nozzles are mounted on deployable booms to provide large moment arms.

The configuration shown in Figure 6.3 is similar to the orbiter/lander in Figure 6.1 except that the survivable lander has been replaced by the atmospheric probe noted earlier. Some minor changes were made to the bus to accommodate this change.

The lander systems are described in the following sections. A weight statement is given in Section 6.5.

## 6.3 Lander Design Concept

Following is a design concept for a survivable lander capable of supporting a scientific payload of 50 lbs for 30 days after touchdown. The lander system weight is approximately 700 lbs, and is designed for launch with a bus/orbiter from an Atlas/Centaur plus kick stage. The lander described herein is designed for the NASA Model 3 (mb) atmosphere.

### 6.3.1 Lander Mission Sequence and Terminology

The lander mission is divided into five phases, as follows:

Transit Phase - The lander is "packaged" inside a sterilization container during Earth to Mars transit. This phase terminates at the time of sterilization container is opened.

Separation and Injection Phase - Approximately 48 hours prior to entry into the Mars atmosphere, the separation sequence occurs, controlled by the spacecraft programmer after receipt of initial command from DSIF. The spacecraft is orientated to the proper entry capsule separation attitude, the aft portion of the sterilization container is ejected, and the entry capsule is subsequently separated from the spacecraft.

The entry capsule is spun up immediately following separation in order to maintain the proper orientation for injection and to provide thrust vector control during burning of the injection rocket motor. The entry capsule is then allowed to coast until it is well clear of the spacecraft, after which the injection rocket motor is fired by command from the lander programmer to place the entry capsule on an impact trajectory to Mars. The injection motor is separated from the entry capsule, terminating the separation and injection phase.

Coast Phase - Following burnout of the injection rocket motor, the entry capsule coasts for approximately 48 hours before entering the Mars atmosphere. Immediately prior to atmosphere entry, the capsule is despun to a small non-zero spin rate in order to provide rapid convergence of angle-of-attack during entry. This event terminates the coast phase.

Entry Phase - The entry phase encompasses all events from initial penetration of the Mars atmosphere until the lander comes to rest on the planet surface. The major events are listed below:

1) Atmospheric entry measurements begin
2) Radio blackout begins
3) Radio blackout ends
4) Supersonic parachute is deployed
5) Entry capsule heat shield is jettisoned and subsonic parachute is deployed in reefed condition
6) Subsonic parachute is disreefed
7) Descent TV pictures are taken and stored aboard the lander
8) Impact occurs and subsonic parachute is jettisoned

These events are depicted in Figure 6.5, which shows altitudes and times for a typical mission.

Lander Deployment and Surface Operations - After the lander has come to rest on the planet surface, lander erector vanes are deployed to place the lander in an upright position. The surface experiment sequence then begins, during which stored onboard the lander and transmitted to the orbiting spacecraft periodically for relay to Earth.

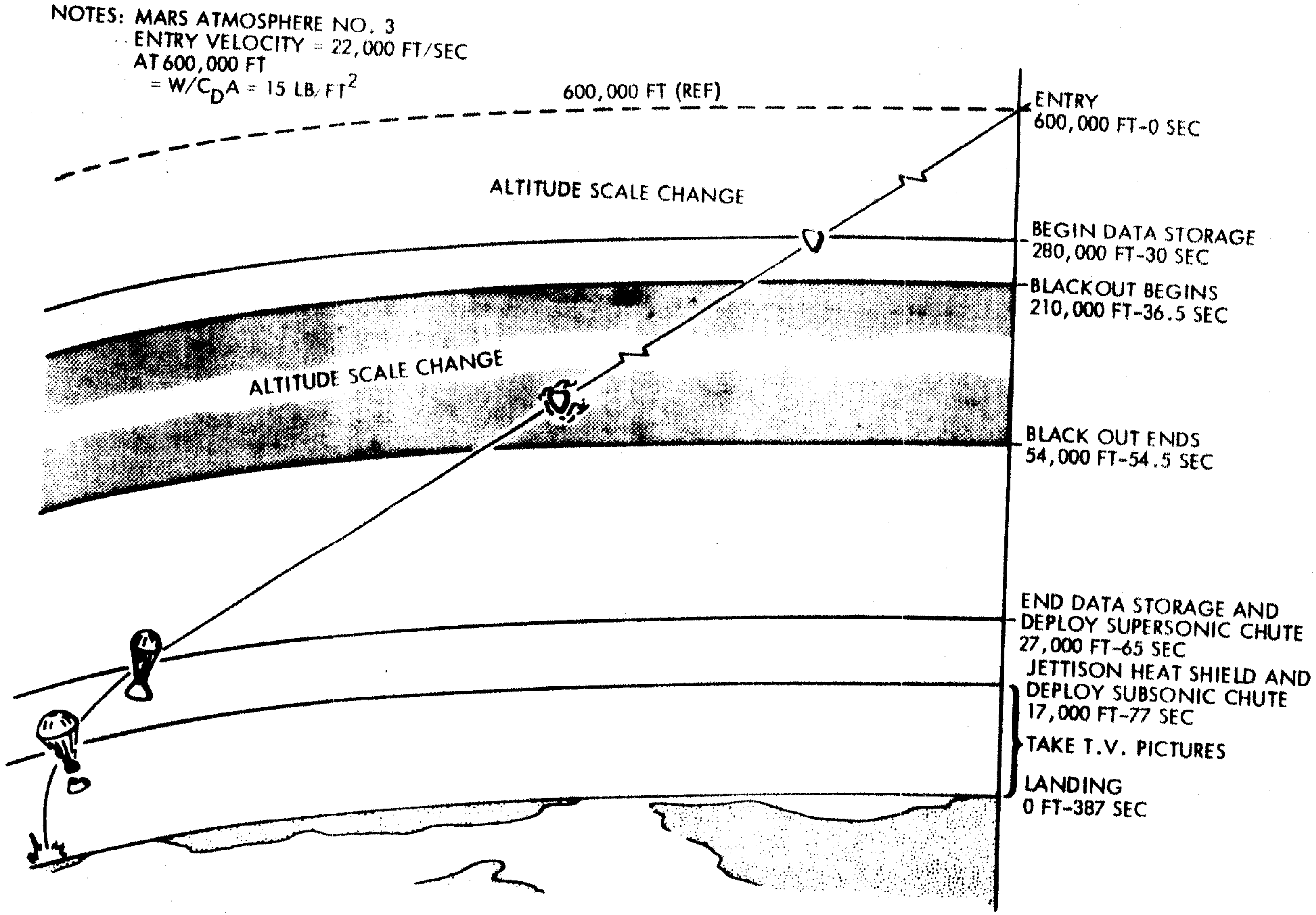



Figure 6.5 Lander Sequence of Events

### 6.3.2 Entry Capsule Injection and Stability Considerations

To minimize entry capsule injection propellant weight, the entry capsule separates from the spacecraft well before Mars encounter. Separation occurs early enough that the injection $\Delta V$ is very small compared with $V_\infty$ (Mars) of the approach asymptote, such that the entry capsule and spacecraft approach asymptotes are nearly parallel. There are several reasons for not injecting too early, however:

1) The spacecraft trajectory must be accurately established following final midcourse correction by DSIF tracking. These trajectory measurements are used to establish the time of separation for a fixed injection $\Delta V$ to obtain accurate Mars entry.
2) The Mars entry angle dispersion is partially dependent upon the angle, $\theta$, between spacecraft $V_\infty$ (Mars) and the entry capsule $\Delta V$ vector. The effect of $\Delta V$ angle injection error upon entry angle dispersion grows sharply with small values of $\theta$ associated with early separation. This factor becomes particularly important as the ballistic coefficient of the entry capsule increases and the allowable entry corridor becomes narrower. Thus, as $\theta$ is made small, a premium is placed upon small tipoff errors during entry capsule separation from the spacecraft and upon very accurate spin stabilization following separation to provide tolerable $\Delta V$ angle injection errors.

Separation-to-impact times which are odd multiples of 12 hours should be avoided to ensure the same ground station contact (ideally Goldstone) for the spacecraft deboost maneuver as for the separation maneuver. A separation to impact time of 48 hours was selected for purposes of this study.

The $\Delta V$ angle injection error discussed above is influenced by the characteristics of the injection rocket motor and the spin rate of the entry capsule which provides thrust vector control during rocket burning. $\Delta V$ angular dispersion decreases with increasing rocket burn time and increasing spin rate. From earlier studies, a spin rate of 40 to 60 rpm and a burn time greater than 5 seconds appear desirable.

### 6.3.3 Entry Phase Aerodynamic and Thermodynamic Considerations

The limits of the Mars entry corridor are established by two constraints: On the "undershoot" side, the entry capsule must slow down sufficiently to permit a soft landing with the aid of terminal deceleration devices such as parachutes; on the "overshoot" side, the entry capsule must dip deeply enough into the atmosphere to prevent "skip-out". Considering only the NASA Model 3 atmosphere (10 mb surface pressure), the "overshoot" limit is relatively independent of the entry capsule ballistic coefficient and the limit entry angle* is 13 degrees. The "undershoot" limit is strongly dependent upon the entry capsule ballistic coefficient and the type of terminal deceleration devices used.

For reasons discussed in the following section, a two-stage parachute system was selected to provide a terminal velocity of 50 ft/sec. For such a system to operate properly, aerodynamic drag must slow the entry capsule to a velocity of Mach 4, the maximum velocity at which a supersonic parachute may be deployed, at an altitude of approximately 20,000 feet. The ballistic coefficient of the reference entry capsule design (Figure 6.6) is approximately 15 lbs/ft.$^2$ which yields a maximum entry angle limit of about 35 degrees. A typical entry sequence of events is shown in Figure 6.5 for an entry angle of 30 degrees.

For purposes of determining spacecraft guidance and attitude control requirements, and entry capsule injection thrust vector requirements, it is convenient to define the entry corridor depth as the difference in periapses altitudes for the vacuum trajectories corresponding to the limiting entry angles. The variation of this corridor depth with entry capsule ballistic coefficient is shown in Figure 6.7 which illustrates one reason for minimizing the entry capsule ballistic coefficient, i.e. less stringent accuracy requirements during transit, separation, and injection maneuvers. Additionally, the steeper entry angles permitted with small ballistic coefficients provide a shorter time of flight from entry to impact resulting in reduced heat shield and communications subsystem requirements. Since the heat shield functions primarily as an insulator, shorter flight times mean less total heating and less heat shield thickness required.

*The entry angle is defined as the angle between the entry capsule flight path (trajectory) and the local horizontal at 600,000 ft above the Mars surface.

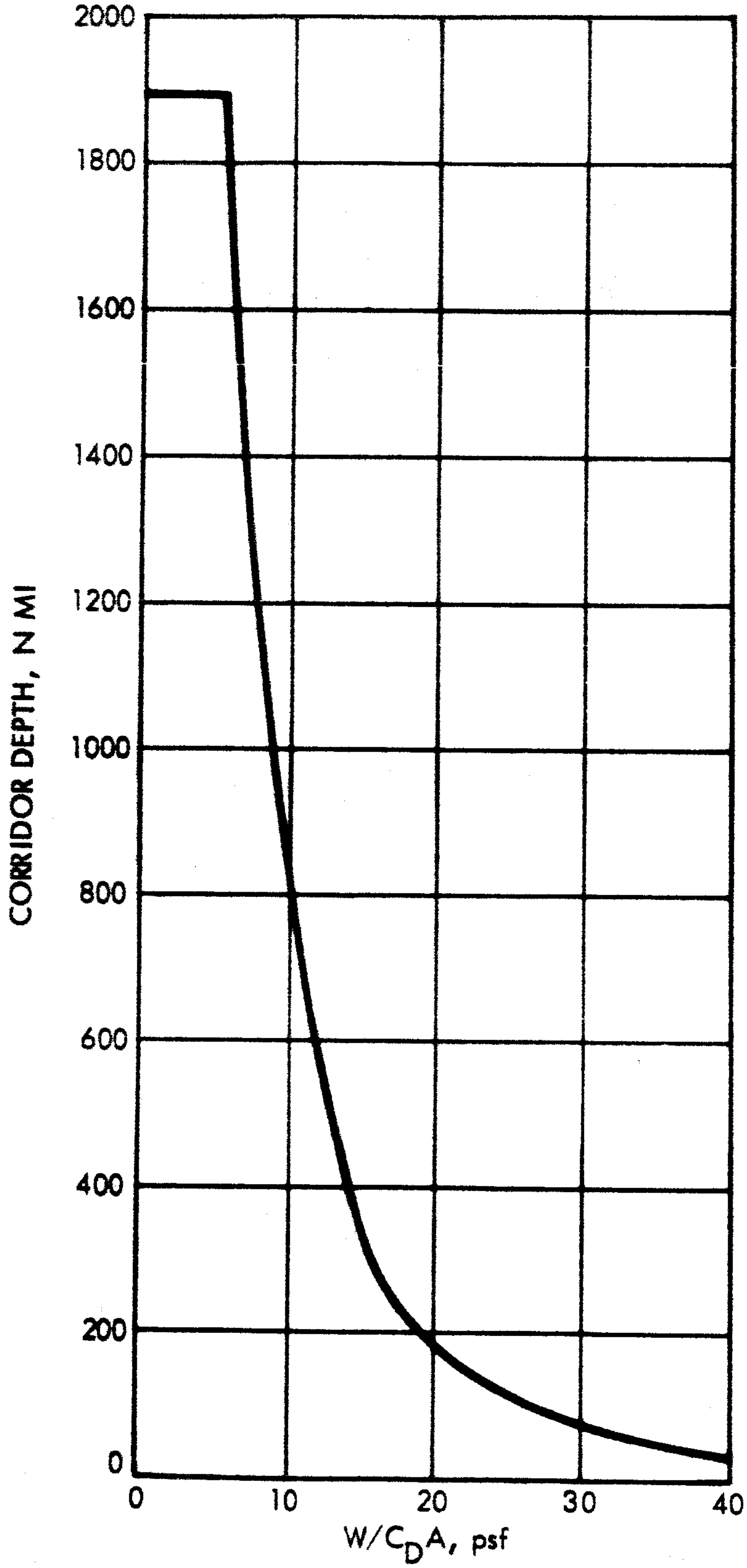



Figure 6.7 Corridor Depth as a Function of Ballistic Coefficient

The communications advantage from shorter entry flight times relates to the maximum communications distance between the entry capsule and the spacecraft.

In the event of lander failure during landing or failure to deploy for surface operations, all data stored aboard the lander during entry would be lost. This data would be especially critical for failure analysis if failure should occur. Consequently, it is desirable to transmit all entry and landing data to the spacecraft prior to firing the spacecraft deboost engine and possible temporary disruption of communications. To accomplish this, the entry capsule must "lead" the spacecraft by a distance which increases as the entry time increases. Minimizing the entry time minimizes this distance and reduces the lander telemetry transmitter power required for a given bit rate. Electrical power requirements and power supply weight are thus reduced contributing to an even lower entry capsule ballistic coefficient.

### 6.3.4 Terminal Deceleration and Landing

A detailed tradeoff study of terminal deceleration and landing techniques was not attempted. However, the following factors were considered in the selection of a two-stage parachute subsystem with impact absorbing material for landing:

Use of a single subsonic parachute requires a small entry angle (estimated to be approximately 25 degrees) in order to provide sufficient time between the Mach = 0.9 point (the highest velocity at which the subsonic parachute may be deployed) and impact for the deployment, disreefing, and inflation cycle. Such an entry angle provides an entry corridor depth of only about 120 nmi for an entry capsule ballistic coefficient of 15 $lbs/ft^2$. In addition, total entry heating is expected to increase substantially resulting in a heat shield weight increase exceeding the weight saved by eliminating the supersonic parachute.

Use of a single supersonic parachute of the Hyperflo type to provide a terminal velocity of less than 100 ft/sec (compatible with simplified landing shock absorption techniques and lander component design loads) is expected to result in an excessive weight penalty. Earlier studies of a 3500-pound lander resulted in a required parachute weight of 2000 pounds.

A combination of supersonic and subsonic parachutes appears to be the best approach. The supersonic parachute size is dictated by the entry capsule diameter. The sizing of the subsonic parachute involves a tradeoff between subsonic parachute weight and the weight of an impact shock attenuating subsystem to maintain a specified shock load to lander components for a given terminal velocity. The subsonic parachute weight decreases approximately as the inverse square of the terminal velocity in the range of 0-100 ft/sec under consideration. However, the weight of crushable material or other shock attenuating subsystem required to limit component shock loads to approximately 100 earth g's (compatible with entry design loads) increases with increasing terminal velocity. A terminal velocity of 50 ft/sec was selected as a reasonable compromise.

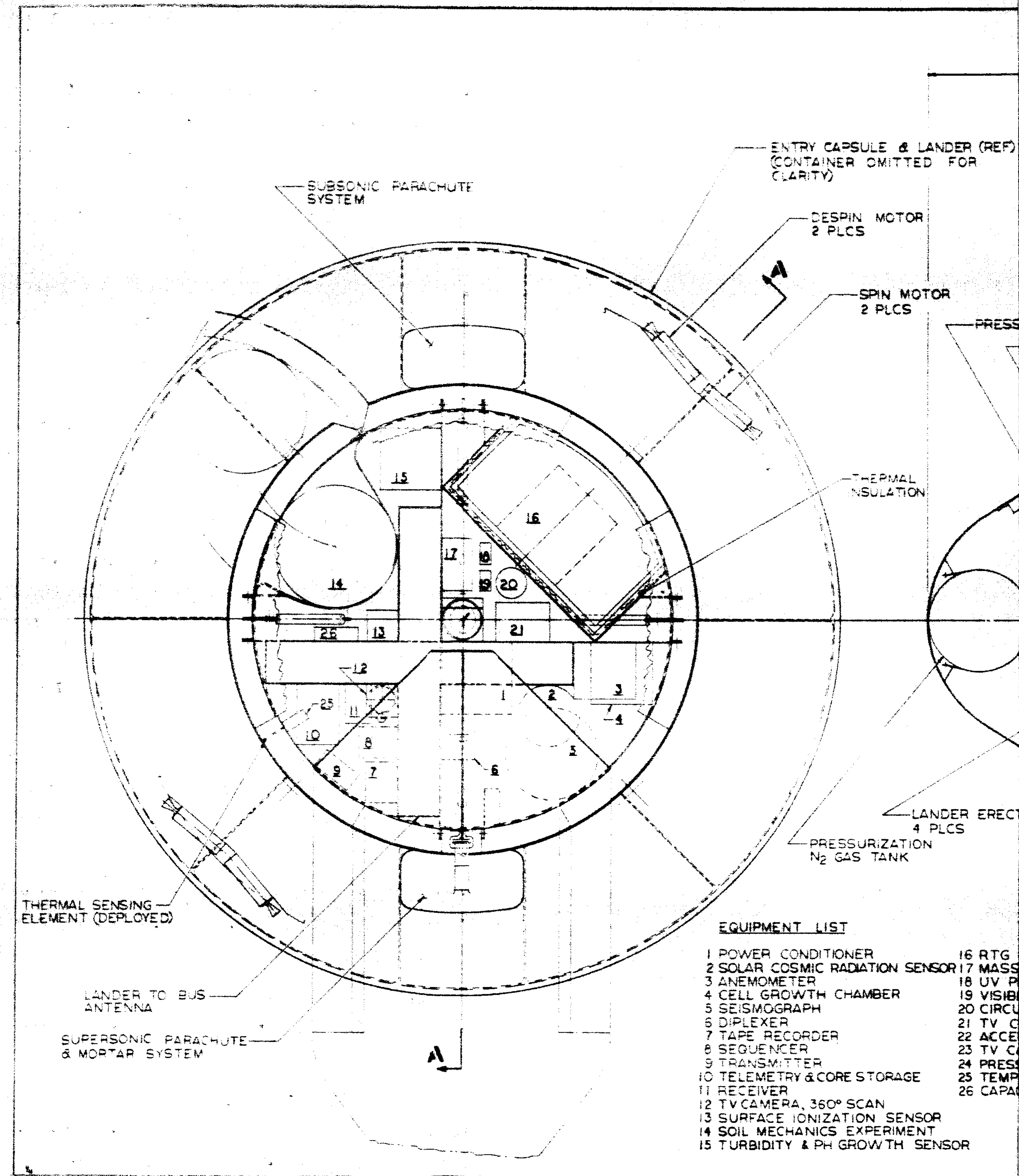
Extended arms with shock absorbers were considered as an alternate to the honeycomb crushable material shown in the reference design (Figure 6.6) but these were considered to be inadequate in the event surface winds and high lateral velocities are encountered during landing.

Solid propellant retro-rockets were considered as an alternate to the subsonic parachute for terminal deceleration. The retro-rocket approach deserves further study, but it is not obvious that it offers greater reliability or less weight for the lander. Unanswered questions involve the provisions of a positive altitude reference independent of lateral drift for initiating the retrofire, the means of accommodating unequal thrust from multiple retro-rockets, and the relative effect of Mars atmospheric density uncertainty upon the performance of the retro-rocket or parachute subsystems.

### 6.3.5 Configuration

The lander is a thin guage aluminum cylindrical container 45 inches in diameter and 32 inches in height with four shear webs radial displaced at 90° intervals running the length of the cylinder (see Figure 6.6). The injection motor, RTG unit, and all experiments are mounted on these shear webs and container walls.

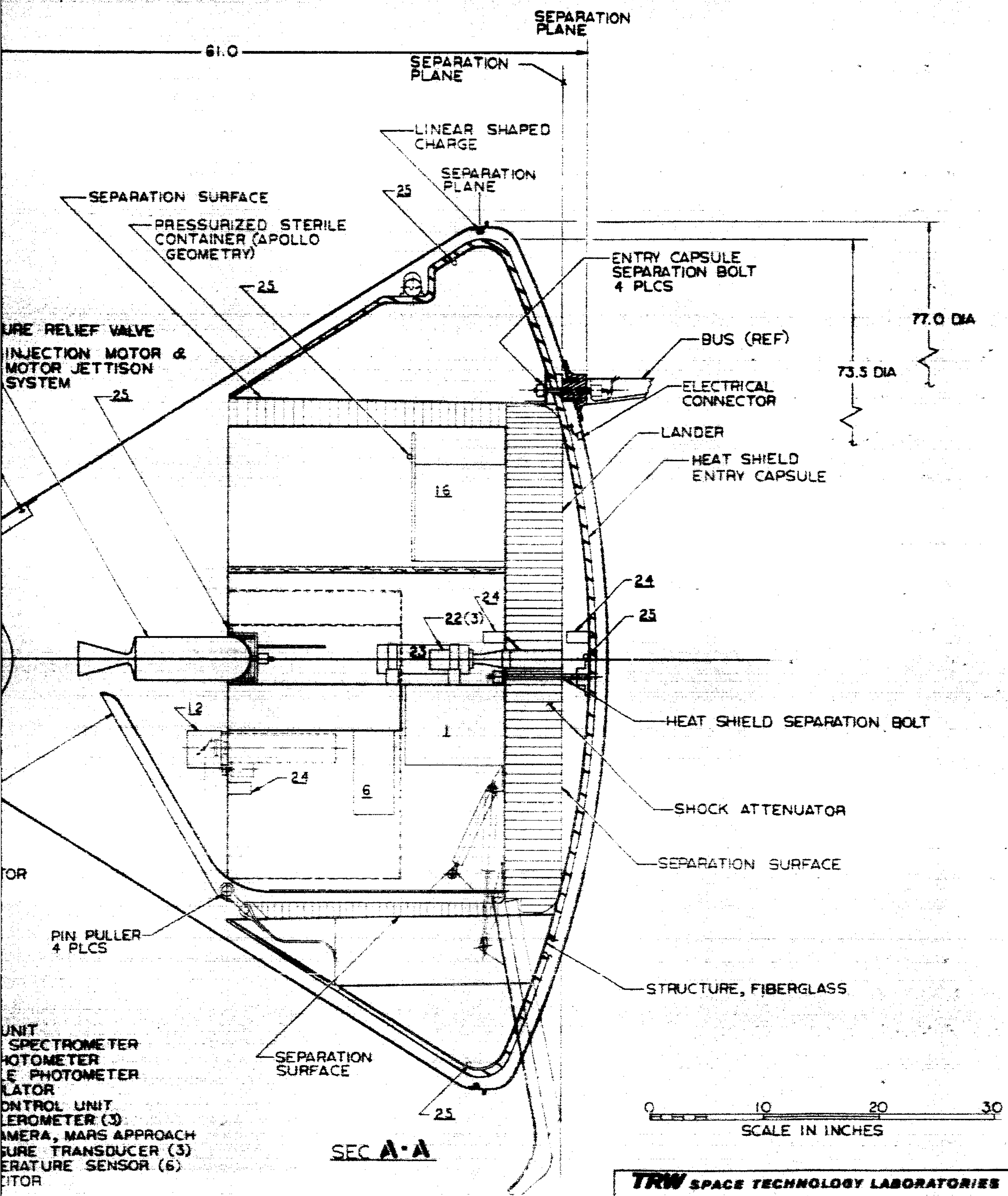
The lander is protected for post-impact survival by aluminum honeycomb shock attenuator panels. These panels are bonded to the lander on the cylindrical walls and on the impact side of the lander. The payload was assumed to be capable of withstanding a vertical impact of 100 earth g's.



ENTRY CAPSULE & LANDER (REF) (CONTAINER OMITTED FOR CLARITY)
SUBSONIC PARACHUTE SYSTEM
DESPIN MOTOR 2 PLCS
SPIN MOTOR 2 PLCS
A
THERMAL INSULATION
LANDER ERECT
4 PLCS
PRESSURIZATION N2 GAS TANK
THERMAL SENSING ELEMENT (DEPLOYED)
LANDER TO BUS ANTENNA
SUPERSONIC PARACHUTE & MORTAR SYSTEM
A
EQUIPMENT LIST
1 POWER CONDITIONER
2 SOLAR COSMIC RADIATION SENSOR
3 ANEMOMETER
4 CELL GROWTH CHAMBER
5 SEISMOGRAPH
6 DIPLEXER
7 TAPE RECORDER
8 SEQUENCER
9 TRANSMITTER
10 TELEMETRY & CORE STORAGE
11 RECEIVER
12 TV CAMERA, 360° SCAN
13 SURFACE IONIZATION SENSOR
14 SOIL MECHANICS EXPERIMENT
15 TURBIDITY & PH GROWTH SENSOR
16 RTG


**Figure 6. 6**

2

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| | |
|---|---|
| MARS ENVIRONMENT STUDY-<br>ENTRY CAPSULE & LANDER | 20 SEPT 1965 |
| | PD87-05 |

The recovery system was designed to survive tumbling and to have the capability to right itself after coming to rest. The lander erector mechanisms consists of four petal-shaped assemblies. Each assembly is hinged to the lander cylindrical container in a manner to provide protection for the antenna and equipment when they are in a closed position. Each segment is individually operated by a gas actuator that is triggered after landing.

The parachute system consists of a supersonic and a subsonic chute, packaged on the exterior of the lander container $180^{\circ}$ apart and on the interior of the heat shield for reentry protection. The supersonic parachute extracts and deploys the subsonic parachute from its canister.

At the time of the subsonic parachute deployment a signal is generated to fire the heat shield separation explosive bolt and thus jettison the base structure and heat shield. Release of the heat shield reduces the weight on the parachute and enhances its characteristics.

As the lander impacts on the planet surface the main chute bridle is released by a signal to the pin pullers. This permits the lander to tumble free of the parachute and thus prevent entanglement and subsequent failure of the erection mechanism. It is also possible to eliminate one gore from the subsonic parachute so that the parachute will tend to deploy itself to one side of the impact area.

The Apollo shaped heat shield is 73.5 inches in diameter and 32 inches in height. It consists of a fiberglass substructure with thermal protection of phenolic nylon which has a maximum thickness of one-half inch on the lower surface and a taper of 1/2 to 1/4 inch on the sides.

Separation of the lander from the heat shield is provided by a single heat shield separation explosive bolt. The heat shield and substructure is attached to the bus by four explosive bolts.

Two spin and two despin solid rockets are located around the periphery of the entry capsule. Indentations are provided on the aft side of the heat shield for rocket installations.

The sterile container is an Apollo-shaped 0.020 guage aluminum container 77 inches in diameter and 62 inches in length. It is constructed in two parts which are joined by welding flanges at the maximum diameter. Four fittings are provided at aft end for structural attachment.

The container prevents contamination of the sterile probe during prelaunch and launch operations. It is equipped with a check valve to permit outflow of gas as it is heated during the sterilization cycle and an onboard gas supply to maintain a slight positive internal pressure during all operations, thereby preventing inflow of contaminated air. Maintenance of a slight internal pressure also provides pressure stabilization and permits a minimum weight design. A requirement for venting the shroud during ascent is necessary as well as means for complete evacuation in space to prevent large decompression problems.

### 6.3.6 Power and Communications Subsystems

A brief description of the power and communications subsystems is given below.

#### 6.3.6.1 Power

Electrical power for the lander is furnished by a radioisotope thermoelectric generator (RTG) with a rated output of 25 watts. The RTG was selected over the alternate concept of a chemical battery and solar cells primarily on the basis of mission reliability. The RTG is less susceptible to degradation or total failure resulting from landing shock loads and unknowns concerning the Mars surface environment, e.g. dust storms. Although an RTG unit does not presently exist for planetary entry missions, it is reasonably expected that such a unit could be developed to meet hardware schedules for a 1971 mission.

The power requirements for the experiments are shown in Table 6.3 and for other lander components in Table 6.4.

Power requirements during a typical twenty-four hour period of operation on the Martian surface are shown in Figure 6.8. From Table 6.4, it is obvious that all experiments must be turned off during operation of the lander-to-orbiter data link. All experiments operate as programmed during other times. Note that experiments 6, 16, 30 and 31 are not programmed to operate simultaneously with the TV experiment (No. 25) in order to have sufficient power for the TV camera, tape recorder and associated electronics.

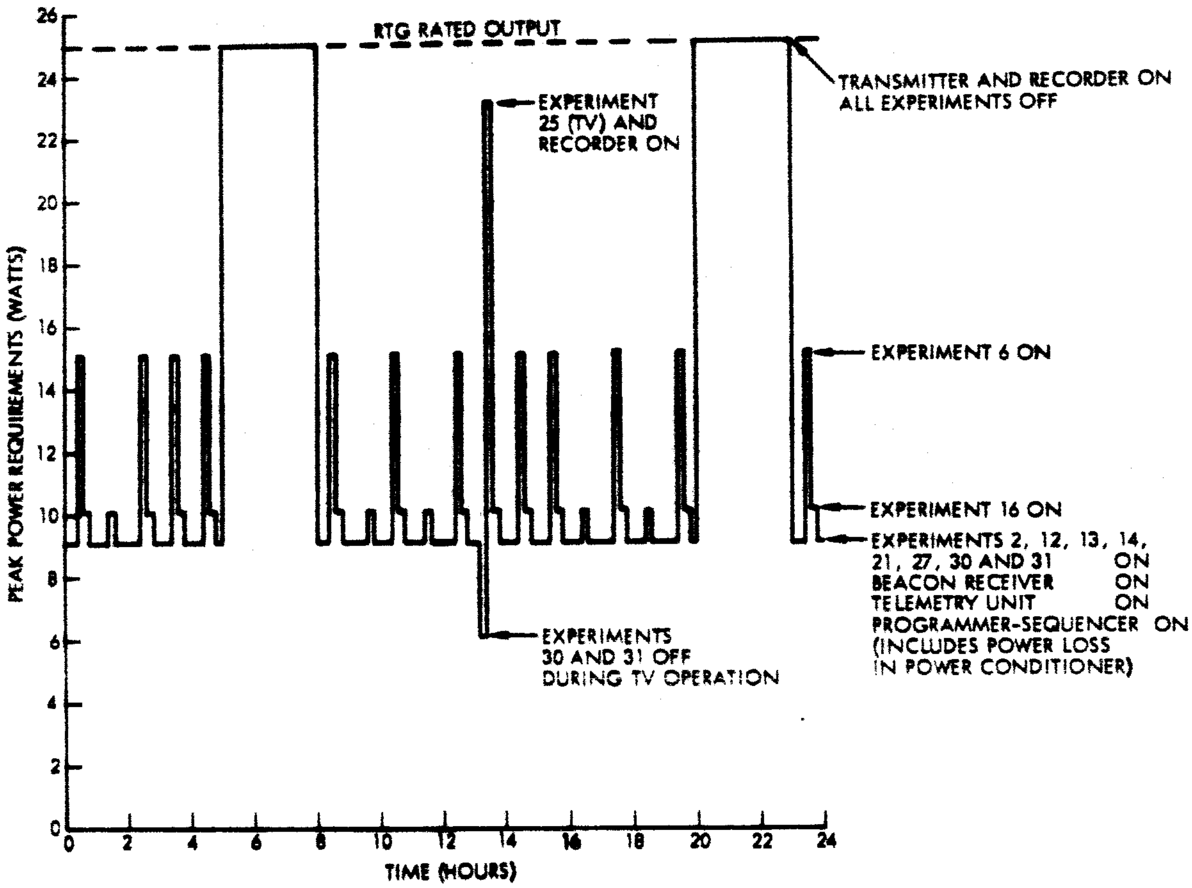


Figure 6. 8 Power Profile During Typical Day of Mars Surface Operation

## Table 6.3 Mars Entry and Surface Experiments Power Requirements

| Experiment | Description[1] | Weight (lbs) | Power (Watts) | Duty Cycle | Data Bits Each Sample | Data Bits Each 24 hrs | Data Bits Stores During Entry |
|---|---|---|---|---|---|---|---|
| 1 | Accelerometers | 1.0 | 2.0 | Continuous during entry | 24 | N/A | |
| 2 | Pressure and Temperature | 0.5 | 0.7 | Continuous during entry | 25 | 1,000 | |
| 6 | Mass Spectrometer | 6.0 | 6.0 | 12 Samples/day | 1100 | 13.200 | |
| 12 | Solar Radiation | 1.3 | 0.1 | Programmed - 12 hrs/day | 20/sec | $8.6x10^5$ | |
| 13 | Light Intensity | 0.5 | 0.1 | 1 Sample/hour | 50 | 1,200 | |
| 14 | UV Intensity | 0.5 | 0.1 | 1 Sample/min | 50 | 72,000 | |
| 16 | Surface Ionization | 1.5 | 1.0 | 1 Sample/hour | 100 | 2,400 | |
| 21 | Surface Winds | 1.0 | 0.1 | 12 Samples/hour | 9 | 2,600 | |
| 25 | TV Pictures | 17.0 | 10.0 | 1 Frame/day | $8x10^5$ | $8x10^5$ | $8x10^6$ |
| 26 | Soil Mechanics | 13.0 | 3.0 | One time only | 1000 | N/A | |
| 27 | Seismic Activity | 8.0 | 1.0 | Continuous operation; threshold initiated transmission | 20/sec | 5,000 | |
| 30 | Cell Growth | 4.0 | 2.0 | 45 Samples/day | 100 | 4,500 | |
| 31 | Turbidity and pH | 4.0 | 1.0 | 2 Samples/hour | 7 | 350 | |
| | All Experiments | | | | | $1.76x10^6$ | |
| | Engineering Data | | | | | $0.01x10^6$ | |
| | Total Data Bits each 24 hours | | | | | $1.77x10^6$ | |

[1] See Section 3, Volume 2 for a description of experiments.

Table 6.4 Lander Component Subsystem Power Requirements

| Component | Watts |
|---|---|
| Transmitter (7 watts output, 50% efficiency) | 14.0 |
| Tape Recorder | 7.0 |
| Telemetry Unit | 1.0 |
| Programmer-Sequencer | 1.0 |
| Beacon Receiver | 1.2 |
| Power Conditioner (full load) | 0.8 |
| | 25.0 |

Power requirements during the separation and injection, coast, and entry phases of the mission will exceed the capability of the RTG unit for short periods when pyrotechnic devices are initiated. Capacitors charged by the RTG during periods of reduced load are used to provide surge power for the pyrotechnic devices.

### 6.3.6.2 Lander-to-Orbiter Communications

During a typical 24 hour period of operation on the Martian surface, $1.77 \times 10^6$ bits of data are expected to be generated (Table 6.3). These data will be stored on tape aboard the lander and periodically transmitted to the orbiter for relay to Earth. The communications time between the lander and orbiter will vary from orbit-to-orbit depending upon the position of the landing site on Mars with respect to the elliptical orbit of the spacecraft. A beacon transmitter in the orbiter and a beacon receiver in the lander are provided to initiate and terminate transmissions of data. Variations in communication time from day-to-day, and thus variations in the total amount of data which can be transmitted during each 24-hour period, can be accommodated by programming the operation of experiments 12 and 25 (which generate 95% of the total data) to coincide with periods of maximum communication time. Additional flexibility is provided by the tape recorder which has a capacity of nearly $10^8$ bits.

Assuming six hours of communication time during a typical 24-hour period (two orbits of the spacecraft) and a lander-to-orbiter bit rate of 100 bits/sec, $2.16 \times 10^6$ bits can be transmitted. This compares with a bit generation rate of $1.77 \times 10^6$ during the same time period. Transmission of TV pictures taken and stored during terminal descent may be interspersed with TV pictures taken after landing.

Power budgets for the lander-to-orbiter and orbiter-to-lander, communication links are presented in Tables 6.5 and 6.6. Additional circuit loss of 0.6 db for both orbiter and lander should be added to allow for diplexing the data and beacon links through a common antenna (NOTE: The orbiter-to-DSIF link is not affected). The beacon frequency should be increased to 180 mc to permit diplexing.

The telemetry link from the lander to the orbiter will be at VHF. Two alternate modulation techniques are shown in the power budget. The coherent PSK system is more efficient but requires acquisition of the lander signal with its attendant problems in the orbiter receiver for each Mars orbital revolution. A simpler scheme which requires no signal acquisition by using a wideband IF in the receiver is noncoherent FSK using demodulation by the two filter-comparison method. This latter scheme is the one preferred even though, as the power budget shows, the performance margin is smaller.

A one-half watt VHF beacon transmitter in the orbiter and an acquisition receiver in the lander are provided to let the lander know when communication with the orbiter is possible each revolution about Mars. The acquisition for this link is simpler due to the absence of modulation on the beacon carrier. When the beacon receiver indicates signal presence, the lander transmitter is turned on for transmission of telemetry to the orbiter. When the orbiter goes out of sight of the lander, the lander transmitter is turned off. The beacon link will be on 136 mc and have its own omnidirectional antennas. Adding the beacon capability adds 2.0 lbs and 1.2 watts power consumption to the lander plus the 136 mc antenna weight. In the orbiter, 2.0 lbs and 1.5 watts power consumption are added plus the weight of the 136 mc antenna.

A list of lander communications equipments is given in Table 6.7.

Table 6.5 Lander-to-Orbiter Communications Power Budget

Frequency: 200 mc

| Parameter | Value |
|---|---|
| Total Transmitter Power (7watts) | +38.5 dbm |
| Circuit Loss | 1.6 db (including diplexer for beacon) |
| Lander Antenna Gain | 0.0 db |
| Space Loss (R=20,000 KM) | 164.4 db |
| Polarization Loss (circular-to-circular) | 1.0 db |
| Orbiter Antenna Gain | 0.0 db |
| Orbiter Circuit Loss | 1.6 db (including diplexer for beacon) |
| Net Transmission Loss | 168.6 db |
| Total Received Power | -130.1 dbm |
| Receiver Noise Spectral Density ($T_S$=600°K) | -170.8 dbm/cps |
| COHERENT PSK ON 800 CPS SQ. WAVE SUBCARRIER | |
| **Carrier Performance** | |
| Carrier Modulation Loss (Mod. Index = 0.68 radians) | 2.2 db |
| Received Carrier Power | -132.3 dbm |
| Carrier Loop Noise BW* ($2\ B_{L0}$ = 200 cps) | 23.0 db |
| Threshold SNR in $2\ B_{L0}$ | 6.0 db |
| Threshold Carrier Power | -141.8 dbm |
| Performance Margin | 9.5 db |
| **Data Performance** | |
| Data Modulation Loss | 4.0 db |
| Received Data Power | -134.1 dbm |
| Data Noise Bandwidth (100 bps) | 20.0 db |
| Threshold SNR in Data Bandwidth ($P_e^b = 5 \times 10^{-3}$) | 7.3 db |
| Threshold Data Power | -143.5 dbm |
| Performance Margin | 9.4 db |
| NONCOHERENT FSK | |
| Filter Noise BM (25 kc) | 44.0 db |
| Input SNR at Envelope Detector | -2.1 db |
| Threshold SNR at Envelope Detector ($P_e^b = 5 \times 10^{-3}$) | -8.2 db |
| Performance Margin | 4.9 db |

*Requires approximately 7-8 sec for acquisition

Table 6.6 Orbiter-to-Lander Beacon Power Budget

Frequency: 136 mc

Modulation: CW

| Parameter | Value |
|---|---|
| Total Transmitter Power (1/2 watt) | 27.0 dbm |
| Circuit Loss | 1.6 db (including diplexer) |
| Orbiter Antenna Gain | 0.0 db |
| Space Loss (R = 20,000 km) | 161.1 db |
| Polarization Loss (circular-to-circular) | 1.0 db |
| Lander Antenna Gain | 0.0 db |
| Lander Circuit Loss | 1.6 db (including diplexer) |
| Net Transmission Loss | 165.3 db |
| Total Received Beacon Power | 138.3 dbm |
| Receiver Noise Spectral Density ($T_S$=600°K) | -170.8 dbm/cps |
| Beacon Loop Noise BW* (2 $B_{L0}$ = 100 cps) | 20.0 db |
| Threshold SNR in 2 $B_{L0}$ | 6.0 db |
| Threshold Beacon Power | -144.8 dbm |
| Performance Margin | 6.5 db |

*Requires acquisition time of about 30 seconds

Table 6.7 Lander Communications Equipment List

| | Wt (lbs) | Power (watts) |
|---|---|---|
| Solid State 7 watt Transmitter (200 mc) | 2.0 | 25.0 |
| Beacon Receiver (136 mc) | 2.0 | 1.2 |
| Telemetry Unit (including 25,000 bit core memory) | 4.0 | 1.0 |
| Programmer-Sequencer | 7.5 | |
| Tape Recorder | 7.0 | |
| Diplexer | 1.5 | |

## 6.4 Weight Summaries

The weights in Tables 6.8 and 6.9 represent a typical orbiter-lander for an Atlas/Centaur plus kick stage launch system. The total spacecraft separated weight is 2560 pounds including a 10 percent weight contingency. (See Figures 6.1 and 6.5).

The retropropulsion system contains enough propellant to place the orbiter into an elliptical orbit of approximately 2,000 by 20,000 km which satisfies quarantine requirements.

A discussion of the orbiter subsystem weights follows:

### Mechanical and Pyrotechnics

The weights are for an explosive bolt type of separation system including bolts and nuts, cartridges, pin pullers, bolt catchers, and miscellaneous brackets. The solar array mechanism includes the hinges, springs, etc., for the deployment and latching of the solar paddles.

### Spacecraft Structure

The external panel weights are based on honeycomb construction at 1 $lb/ft^2$. Four of these panels contain rails and fasteners for equipment mounting provisions.

### Thermal Control

Thermal control for the spacecraft includes both active louvers and passive insulation. The louvers are located on the four equipment panels and weigh 0.56 lb/sq ft. The insulation is located on the solid and on the internal areas of the spacecraft.

### Electrical Power

Solar array weights are based on 1 lb/sq ft including structure. The batteries and regulators, power conversion, power control unit, and shunt element assembly weights are estimated to be 71.7 pounds.

### Electrical Distribution

Cabling and connector weights are based on empirical data considering the amount of equipment requiring power and electrical connection, the spacecraft geometry, and the packaging factor used.

Stabilization and Control

The stabilization and control weights are estimated to be 71. 4 pounds which includes 6 pounds of nitrogen gas.

Propulsion System

The retropropulsion system is a spherical solid with a partially submerged nozzle. The propellant is aluminized rubber-base with a density of 0. 061 lb/cu in and an $I_{sp}$ equal to 295 seconds. The thrust is assumed to be 2500 pounds. The $\Delta V$ allowed for the 2, 000 x 20, 000 km orbit is 1. 64 km/sec. A liquid injection thrust vector control unit is included for control during retro-firing.

The midcourse correction propulsion unit is a monopropellant (hydrazine) system with a thrust of 50 pounds. The propellant required is based on a $\Delta V$ equal to 75 m/sec with an $I_{sp}$ = 230 seconds. The propellant density is 0. 0362 lb/cu in and the tank pressure is 250 psi. Nitrogen is used for pressurization stored at 3000 psi.

Science and Science Supports

The science and science support weights are estimated at 80. 8 pounds and 26. 2 pounds, respectively.

The entry capsule-lander weights are based on the configuration shown in Figure 6. 6 and are discussed below:

Structure and Sterilization Canister

The heat shield is phenolic nylon with a thickness of 1/2 inch on the lower surface and a taper from 1/2 to 1/4 inch on the sides. Lander impact absorbing material is honeycomb at 2. 5 lb/cu ft. The inner faces are aluminum with a gage of 0. 032 and the outer faces are 0. 010 fiberglass. The sterilization canister is 0. 020 aluminum with a shaped charge for ejecting clear of the capsule.

Electrical Power and Integration

Electrical power is furnished by an RTG with an estimated weight equal to 30 pounds. Cabling and connector weights are based on an empirical data.

Experiments

The experiment package weight is estimated to be 56.8 pounds.

Parachute System

Both a supersonic and subsonic chute is assumed with an estimated weight of 146 pounds total. These weights are based on a terminal velocity of 50 ft/sec.

Spin/Despin Rockets

Four solid rockets are located around the periphery of the entry capsule, each weighing 1.1 pounds total.

Thermal Control, Injection Motor and Separation

These subsystem weights are estimated to be 6, 10, and 3 pounds, respectively.

A weight contingency of 10 percent has been added to both the orbiter and lander weights. This contingency reflects the overall level of confidence of the weight estimates and is consistent with the current level of design. The contingency allows for uncertainties in weight estimation techniques and slight modifications of the design. It also allows for normal weight growth during design completion and the development phase of the spacecraft.

A weight history is given for both vehicles in Table 6.10.

Table 6.8 Orbiter Spacecraft Weight Statement

| | Weight, lbs |
|---|---|
| Mechanical and Pryotechnics | 22.2 |
| Launch Vehicle Separation | 6.6 |
| Capsule Jettison | 5.0 |
| Solar Array Mechanism | 9.2 |
| Attachment and Misc. | 1.4 |
| Spacecraft Structure | 128.4 |
| Equipment Panels | 73.5 |
| Frame-Work | 20.0 |
| Equipment Mounting Provisions | 15.0 |
| High Gain Antenna Supports | 6.8 |
| Low Gain Antenna Supports | 0.5 |
| Lander Antenna Supports | 0.6 |
| Stabilization and Control Supports | 0.8 |
| Attachment and Misc. | 6.2 |
| Lander Support Structure | 5.0 |
| Thermal Control | 27.7 |
| S/C Insulation | 12.0 |
| Solid Motor Insulation | 9.0 |
| Louvers | 5.7 |
| Heaters and Thermostats | 1.0 |
| Telecommunications | 123.2 |
| S-Band Xmtr (10w) (2) | 10.0 |
| Beacon Solid State Xmtr (1w) (136 mc) | 2.0 |
| S-Band Receiver (2) | 7.0 |
| Command Decoder (2) | 7.0 |
| Lander Receiver (200 mc) | 2.0 |
| Diplexer | 1.5 |
| DTU (incl 25,000 bit core memory) | 5.0 |
| Sequencer and Power Supply | 22.6 |
| Low Gain Antenna (9 ft dish) | 63.6 |
| Lander Antenna (136 mc) | 1.5 |

Table 6.8 Orbiter Spacecraft Weight Statement

| | |
|---|---|
| **Electrical Power** | **171.7** |
| Solar Array | 100.0 |
| Batteries | 40.0 |
| Power Conversion | 10.0 |
| Battery Regulator | 10.4 |
| Power Control Unit | 6.3 |
| Shunt Element Assembly | 5.0 |
| **Electrical Distribution** | **47.0** |
| Cabling and Connectors | 35.0 |
| J-Boxes | 10.0 |
| Umbilicals | 2.0 |
| **Stabilization and Control** | **71.4** |
| Control Electronics Assembly | 13.0 |
| Gyros and Electronics | 10.0 |
| Coarse Sun Sensor | 2.0 |
| Fine Sun Sensor | 0.6 |
| Canopus Sensor + Elec. | 11.0 |
| Gas Vessel + Transducers | 9.0 |
| $N_2$ Gas | 6.0 |
| Pressure Reg. + Transducers | 3.0 |
| Valves + Plumbing Set | 6.5 |
| Earth Detector and Sensors | 10.3 |
| **Propulsion System** | **212.0** |
| Retropropulsion | |
| Solid Motor Inert Weight | 138.0 |
| Solid Motor Support | 10.0 |
| LITVC Injectors<br>LITVC Fluid + Tankage | 25.0 |
| Midcourse Propulsion | 37.0 |
| Evasive Maneuver Propulsion | 2.0 |

Table 6.8 Orbiter Spacecraft Weight Statement

| | |
|---|---|
| Science Support | 26.2 |
| MOP Gimbal, Drive, etc. | 5.2 |
| MOP Cable Wrap-up | 1.0 |
| MOP Structure and Thermal Control | 12.0 |
| Science Thermal Control | 1.5 |
| Magnetometer Support | 3.0 |
| Science Cabling and Connectors | 2.0 |
| Attach and Misc. | 1.5 |
| Science Experiments | 80.8 |
| Particle Flux (high energy) | 10.0 |
| Particle Flux | 2.5 |
| Ion Chamber | 1.3 |
| Trapped Radiation Detector | 4.0 |
| Magnetometer | 5.0 |
| Meteoroid Environment | 5.0 |
| Micrometeoroid Environment | 8.0 |
| TV | 17.0 |
| UV Spectrometer | 22.0 |
| Ionosphere Experiment | 3.0 |
| IR Radiometer | 3.0 |
| Contingency 10% | 91.1 |
| Mars Orbiter Weight | 1001.7 |
| Midcourse Correction Propellant | 64.0 |
| Retropropellant | 785.0 |
| Capsule, Lander & Sterilization Canister | 708.8 |
| TOTAL SPACECRAFT WEIGHT | 2559.5 |

Table 6.9 Lander Weight Statement

| | Weight, lbs |
|---|---|
| Electrical Power and Integration | 51.5 |
| RTG | 30.0 |
| Power Conditioning | 10.0 |
| Cabling and Connectors | 7.0 |
| J-Box | 2.0 |
| Miscellaneous | 2.5 |
| Communications | 33.0 |
| Transmitter (2) | 4.0 |
| Beacon Receiver (2) | 4.0 |
| Telemetry Unit (incl. 25,000 bit core memory) | 5.0 |
| Tape Recorder | 7.0 |
| Sequencer and Power Supply | 7.5 |
| Circulator | 1.0 |
| Antenna | 4.5 |
| Experiments | 56.8 |
| TV | 17.0 |
| Mass Spectrometer | 6.0 |
| Turbidity and PH | 4.0 |
| Cell Growth | 4.0 |
| Solar Cosmic | 1.3 |
| Anemometer | 1.0 |
| UV Detector | 0.5 |
| Surface Ion | 1.5 |
| Surface Properties | 13.0 |
| Seismometer | 8.0 |
| Visible Intensity | 0.5 |

Table 6.9 Lander Weight Statement

| | |
|---|---|
| Structure | 298.8 |
| Heat Shield | 216.6 |
| Impact Absorbing Material and Bonding | 29.3 |
| Outer Face Material - Fiberglass | 4.1 |
| Inner Face Material - Aluminum | 14.1 |
| Stabilizing Legs and Actuation | 8.0 |
| Inner Support Structure | 6.0 |
| RTG Support | 4.0 |
| Attach and Miscellaneous | 16.7 |
| Sterilization Canister | 34.8 |
| Structure | 31.3 |
| Shaped Charge | 1.5 |
| Miscellaneous | 2.0 |
| Parachute System | 146.0 |
| Subsonic Chute | 120.0 |
| Supersonic Chute | 25.0 |
| Parachute Containers (2) | 1.0 |
| Spin/Despin Rockets (4) | 4.4 |
| Thermal Control | 6.0 |
| Injection Motor | 10.0 |
| Separation | 3.0 |
| Contingency (10%) | 64.4 |
| TOTAL WEIGHT | 708.7 |

Table 6.10 Weight History

## Orbiter

| | |
|---|---|
| TOTAL SEPARATED ORBITER WEIGHT | 2560 |
| Midcourse Propellant | -64 |
| Spacecraft Prior to Lander Separation | 2496 |
| Capsule and Lander | -674 |
| Sterilization Canister | -35 |
| Spacecraft Prior to Retro Firing | 1787 |
| Retropropellant | -785 |
| Orbiter Weight | 1002 |

## Lander

| | |
|---|---|
| TOTAL LANDER AND STERILIZATION CANISTER WEIGHT | 708.8 |
| Upper Canister Jettison | -20.3 |
| Capsule and Lander Prior to Separation From Spacecraft | 688.5 |
| Lander Separation | -2.0 |
| Lower Canister Jettison | -14.5 |
| Capsule and Lander After Separation From Spacecraft | 672.0 |
| Spin-up Propellant | -0.7 |
| Injection Propellant | -8.2 |
| Despin Propellant | -0.7 |
| Capsule and Lander After Re-entry | 662.4 |
| Heat Shield Jettisoned (includes ablated material during re-entry) | -216.6 |
| Supersonic Chute Jettison | -25.0 |
| Lander Just Prior to Mars Impact | 420.8 |
| Subsonic Chute Jettison | 120.0 |
| Lander @ Mars Impact | 300.8 |

# 7. MISSION PERFORMANCE COMPARISON

A summary of mission payload performance is shown in Table 7.1, comparing the launch vehicle capabilities with mission requirements for orbit (or flyby), entry capsule and survivable lander missions. The data give a general indication of mission effectiveness for the various classes of booster systems.

Four categories of payloads are shown, based on the list of experiments given in Section 3 (Table 3.1):

1. Orbiter (or flyby)
2. Orbiter Mapper
3. Entry Capsule
4. Lander (survivable)

The weights quoted for each of the payloads include all priority experiments required to evaluate the critical elements in the Martian and cismartian environment. The orbiter mapper requires a heavy telescope to achieve adequate resolution from the high orbit altitudes (2,000 to 20,000 km) to which the spacecraft is restricted because of lifetime requirements (50 years), arising from noncontamination constraints. Conventional TV systems without telescopes have resolutions of the order of 1 km which is not adequate for landing site selection or for general purpose detailed reconnaissance. The telescope considered here would improve the resolution to about 0.005 km which could be improved somewhat if orbital altitudes could be reduced based on a better resolution of density levels in the upper atmosphere.

In general, the usefulness of a single lander mission is open to question from the standpoint of landing site selection. Surface property measurements at a single point may not be representative of general conditions within a proposed exploration traverse area and detailed television coverage of one landing site may have limited applicability to other sites under consideration. The only practical means of "extrapolating" data is by TV scan around the lander base, or by deployment of a miniature surface rover, properly instrumented to measure surface properties over a fairly extensive area. However, these modes at best yield limited data, and it seems assured that heavy reliance will be placed on extensive mapping (general and detailed) from a Mars orbiter photographer. It is recommended that a mission be devoted to the mapping function.

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Table 7.1 Mission Payload Performance Comparison

| PAYLOADS | Spacecraft Weight (lbs) | Atlas Centaur 1971 (1650 lbs) | Atlas Centaur 1973 (1200 lbs) | Atlas Centaur 1975 (560 lbs) | Atlas SLV (Advn)-Centaur 1971 (2250 lbs) | Atlas SLV (Advn)-Centaur 1973 (1700 lbs) | Atlas SLV (Advn)-Centaur 1975 (910 lbs) | Atlas Centaur + Kick Stage 1971 (3650 lbs) | Atlas Centaur + Kick Stage 1973 (3200 lbs) | Atlas Centaur + Kick Stage 1975 (2080 lbs) (Orbiter) | Saturn IB-Centaur 1971 (9900 lbs) | Saturn IB-Centaur 1973 (8600 lbs) | Saturn IB-Centaur 1975 (6600 lbs) |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| ORBITER (FLYBY)<br>Experiments: (50 - 80 lbs) Particle and magnetic fields, meteoroid environment, spectroscopy, TV (low resolution). | 1850 (Flyby: 500 - 1000) | FB | FB | FB | OR | FB | FB | OR | OR | OR | OR | OR | OR |
| ORBITER MAPPER<br>Experiments: (400 lbs) Same as above plus high resolution photo coverage. | 22; | -- | -- | -- | OR-M | -- | -- | OR-M | OR-M | -- | OR-M | OR-M | OR-M |
| ENTRY CAPSULE<br>Experiments: (18 lbs) Atmosphere pressure, temp, density, TV | 137 | FB+EC | FB+EC | -- | OR+EC | FB+EC | FB+EC | OR+EC (OR-M+EC) | OR+EC (OR-M+EC) | OR+EC | OR+EC (OR-M+EC) | OR+EC (OR-M+EC) | OR+EC (OR-M+EC) |
| LANDER<br>Experiments: (56 lbs) Solar radiation, spectroscopy, winds, surface properties, TV, life detectors, seismometer. | 709 | -- | -- | -- | FB+L | FB+L | -- | OR+L (OR-M+L) | OR+L (OR-M+L) | FB+L | OR+L (OR-M+L) | OR+L (OR-M+L) | OR+L (OR-M+L) |

A comparison of scientific payloads with launch vehicle capabilities is given in Table 7.1. Payload weights are tabulated for each of the four modes described above, and an estimate of the weight of the spacecraft necessary to support each payload. These weights are then matched with the payload capabilities of the various launch systems for the mission opportunities of 1971, 1973, and 1975. A summary comparison is made of missions obtainable by each launch system for each mission opportunity.

The results indicate that the Atlas Centaur system is capable of supporting only flyby missions with or without nonsurvivable entry capsules. The launch system is restricted to Mariner 4 types of spacecraft in the unfavorable 1975 mission opportunity, and is not capable of supporting orbiter systems with even moderate television resolution capabilities, nor can it support a flyby system plus a survivable lander. Hence, the Atlas Centaur launch vehicle system is of marginal effectiveness for the mission requirements considered herein.

A highly useful launch system is the Atlas Centaur plus a high energy kick stage, a. described in Section 6 of the present report. The launch system is capable of injecting approximately 3600 pounds of payload weight to Mars in the favorable 1971 mission opportunity, and of launching approximately 2,080 pounds to Mars in the unfavorable 1975 mission opportunity. This payload capability is sufficient for launching an orbiter plus lander with a high resolution mapping telescope in the favorable mission opportunities of 1971 and 1973. In the unfavorable opportunity of 1975 the launch system is capable of injecting an orbiter spacecraft with possibly a minimum weight nonsurvivable entry capsule only, or medium resolution photo mapping package. This latter spacecraft is of marginal utility in 1975 if the prior missions of 1971 and 1973 have been accomplished satisfactorily. In general, the Atlas Centaur plus high energy kick stage appears to be a highly atractive and useful launch system for accomplishing the mission goals stated herein.

The Saturn IB Centaur launch system is capable of injecting orbiter plus lander with high resolution mapping packages to Mars in all mission opportunities including 1971, 1973, and 1975. In fact, additional weight capability is available for more sophisticated lander systems, or for multiple lander deployments. The relative effectiveness of the Saturn IB Centaur systems, however, is reduced by the greatly increased cost of the launch system compared to the Atlas based boost systems.

It is recommended that the Atlas Centaur plus kick stage launch system be utilized for the favorable mission opportunities of 1971 and 1973, and that consideration be given to the use of Saturn IB Centaur launch system for the unfavorable opportunity of 1975. The requirement for the 1975 mission, however, will be based upon the results of the earlier missions.

# 8. CONCLUSIONS

The following conclusions were reached from the material presented herein.

1. The evaluation of the martian environment requires monitoring of the interplanetary fields and particles, measurement of the atmosphere of Mars, measurement of the surface properties in potential landing sites, and requires a capability to perform general mapping augmented by detailed photo analysis over areas of interest on the surface (the photo reconnaissance can be augmented with TV based on lander systems). Hence, a lander is mandatory plus orbiter photo mappers of high resolution for evaluation of the Mars environment.

2. Preliminary designs indicate that an orbiter bus weighs approximately 1,850 pounds, a survivable lander 700 pounds, and a photo mapper package for use aboard orbiters approximately 400 pounds. Eighty pounds of experiments are included or can be supported on the bus/orbiter, and 56 pounds on the landers; these scientific payloads include all priority experiments as defined during prior phases of the study.

3. Preliminary designs indicate that a highly effective spacecraft system can be launched by an Atlas Centaur plus kick stage booster. It is possible using this launch system to inject an orbiter plus lander plus high resolution mapper packages in 1971 and 1973; however, in 1975 the payload capability is sufficient to launch an orbiter only, with intermediate resolution mapper capability. The Saturn IB Centaur, can launch the orbiter-mapper plus lander in all years and may be required in 1975 depending upon the results of the earlier missions.

4. The orbiter bus design selected for use with the Atlas Centaur plus kick stage boost system uses an Earth pointing mode with body fixed antenna and solar arrays, the latter having somewhat reduced output because of the orientation mode throughout the duration of the mission. A communication capability of 4,000 bits per second is available in the orbiter bus system with a 10 watt transmitter and a 9-foot diameter antenna. A solid retro propulsion system was used to place the spacecraft in a 2,000 by 20,000 km orbit about Mars, which was

selected to give a 50 year lifetime, based on noncontamination constraints. A monopropellant midcourse and thrust vector control system was used. It was found that the radioisotope thermal electric power generators were not available in adequate quantities for the orbiter/bus power system; the use of solar cells was indicated for this reason. The total weight of the orbiter bus system is 1,350 pounds including retro for the orbiting maneuver. Approximately 30 pounds of experiments are supported by the orbiter/bus system.

5. The lander, which was designed for the Model 3 (10 millibar) atmosphere, uses an Apollo-shaped shield, two parachutes (a supersonic and subsonic unit), and is designed for impact and lateral velocities of 50 fps, with 100 earth g's impact. Selective elements within the lander system may require additional shock attenuation. It was found that the drift velocities cannot be removed without the use of elaborate sensing equipment and velocity removable techniques. The lander is capable of self-righting after touchdown and subsequent tumbling. Approximately 55 pounds of experiments are supported by the lander, with data being relayed back to the orbiter by a 200 mc communication subsystem. A television system is incorporated for mapping during subsonic descent through the atmosphere, and for photoanalysis of the surrounding areas after touchdown and subsequent activation of the lander system. The television pictures acquired during descent are stored on a tape recorder for subsequent transmission to the orbiter spacecraft.

# APPENDIX A. STERILIZATION

## A.1 Introduction

The following presents a discussion of sterilization procedures for the Mars lander and bus vehicles and reflects the impact that sterilization will have upon:

- Lander subsystem design
- The lander/bus interface
- Spacecraft handling and launch sequence
- Lander separation sequence upon approaching Mars

The first section discusses the sterilization requirement as well as the processes and problems associated with sterilizing the lander vehicle. Specific operational problems are discussed in the second section and design constraints are suggested which will minimize the imporatance of factors about which there is scientific uncertainty. The last section presents an operations plan which outlines the sterilization procedures to be executed during the various operational phases of the Mariner mission.

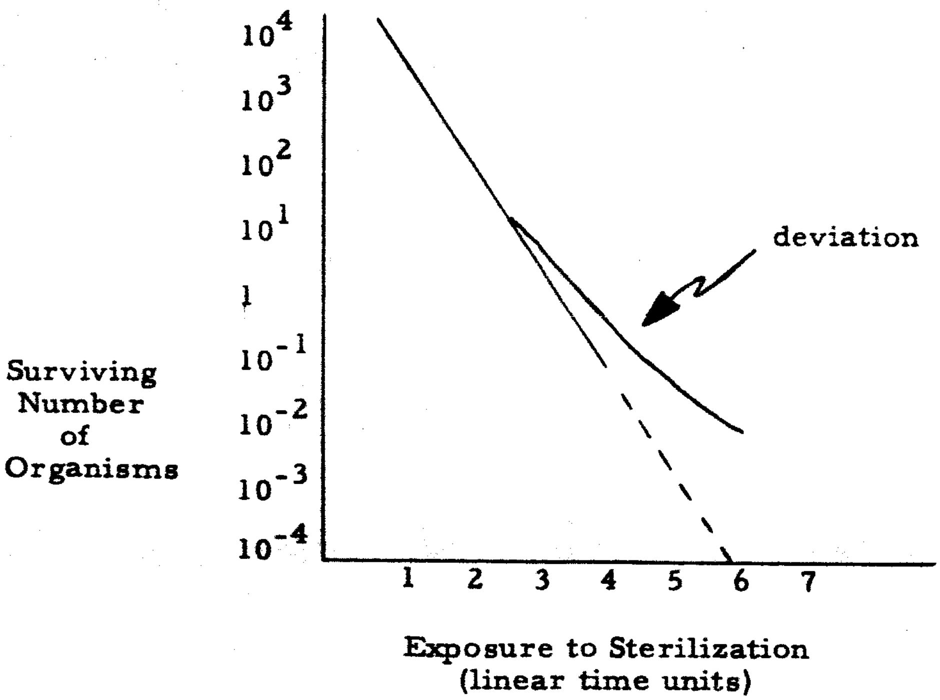
### A.1.1 Sterilization Requirements

As background for understanding the impact of the lander sterilization requirement on lander/bus integration, a discussion of procedures for achieving a sterile lander is presented below.

Laboratory studies have indicated a high probability that microorganisms of earth origin will survive on Mars. Spacecraft sterilization requirements for Mars flights thus are more severe than for flights to the Moon or Venus. A per flight goal for the probability of Mars contamination by an unmanned vehicle is placed at $10^{-4}$. This means that any vehicle designed to enter the Mars atmosphere must be sterilized by a procedure which will reduce to $10^{-4}$ the probability of a living organism remaining either on the surface or in the interior of the vehicle. The requirement also means that adequate procedures must be implemented to retain this probability through launch, flight, and until the vehicle has landed on Mars.

Consideration must be given to sterilization of the bus as well as the lander vehicle. The bus used either for flyby or Mars orbit may enter the Mars atmosphere because of guidance errors and orbit decay. The overall probability of bus contamination and entry into the Mars atmosphere thus must be compatible with the contamination goal previously indicated.

To estimate the probability of contamination an experimentally derived kill curve is used. If the surviving number of organisms of a homogeneous population is plotted versus time of exposure to a sterilization process, a logarithmic relationship generally results as indicated below.



Deviation from a logarithmic relationship generally results when the original population is not homogeneous but contains resistant strains. If the logarithmic relationship is extrapolated below unity, the resulting relationship can be used as an estimate of the probability that a single viable microorganism remains. Thus using $10^9$ as an estimate of spacecraft contamination and using the contamination probability goal of $10^{-4}$ an exposure period to a given sterilization process can be determined as that period required to obtain a population decrease of $10^{-3}$.

A. 1. 2 Sterilization Processes

Heat and radiation are the only two methods of sterilization which can achieve internal as well as external sterility. Of the two, heat is by far the simpler method and is less damaging to materials. Exposure of a heterogeneous population of microorganisms to 135°C for 24 hours will result in a decrease in viable population of $10^{13}$. The $10^{-4}$ probability therefore can be obtained with heat if:

- The original population of viable microorganisms does not exceed $10^9$.
- All parts of the lander and its sterilization shroud are maintained at a temperature of 135°C for 24 hours.

To prevent the size of the initial microorganism population inside the sterilization shroud from exceeding $10^9$ involves controlling the source of microorganisms as well as employing cleaning procedures to reduce contamination to an acceptable level. Many parts of the lander will have to be assembled and handled in a clean room to reduce contamination due to fallout of microbe-bearing particles in the air. The number of people and the nature of the activity in the area also will have to be controlled to reduce the generation of viable particles.

Handling of parts must be controlled to prevent unnecessary handling and to minimize contamination when handling is required. Parts must be transported and stored in closed containers to avoid fallout contamination. Finally, a program of systematic monitoring must be implemented to determine the degree of contamination that exists on each part or assembly. If the contamination level exceeds predetermined goals, then additional decontamination/sterilization must be initiated.

Once the lander has been entirely assembled it must be sealed into a sterilization shroud and the entire assembly subjected to dry heat sterilization. Continuous monitoring will be necessary to make certain that the temperature does not drop below 135°C on any part of the craft during the 24 hour dry heat exposure period.

After heating, the sterilization shroud cannot be opened again until some time after launch when all sources of contamination have been eliminated. Opening the shroud for any reason for any period of time would seriously jeopardize the craft's sterility, and would make resterilization necessary.

## A. 2 Problem Areas

The requirement to produce a sterile Mars lander and to maintain a sterile system up to the point of Mars entry generates problems of both a design and an operational nature. Several potential problems areas are isolated for discussion in the following sections.

### A. 2. 1 Contamination of the Lander from the Bus Vehicle

The possibility of contamination of the sterile lander by organisms transferred from the nonsterile bus exists as soon as the sterile shroud surrounding the lander is opened for either of the following reasons:

- The sterile shroud might be opened following injection into a trans-Mars trajectory to facilitate energy rejection from lander systems.

- The sterile shroud must be opened to separate the lander from the shroud before lander retrograde.

Cross contamination occurs when microorganism bearing materials liberated from the nonsterile bus impinge upon the lander. Material will flow continually from the surface of the spacecraft due to outgassing, evaporation and sublimation of materials. Operation of attitude control jets and midcourse rockets also are potential sources of contamination. These materials move outward following trajectories which reflect particle velocities, their physical characteristics and the resulting forces. This flux gives rise to a very low density "cloud" of material which is roughly cenetered about the spacecraft.

Two exposure conditions must be analyzed:

- The lander attached to the bus with the sterile shroud open but shielded from direct "view" of the bus by a portion of the sterile shroud which is left in place

- The exposed lander propelled into its descent trajectory through the "cloud" of material emanating from the bus.

The first exposure condition could be coupled with long exposure periods if the sterile shroud were opened to facilitate energy rejection from the lander RTG. The second situation will always occur since the sterile shroud must be jettisoned before making the trajectory change which will impact the lander on Mars.

Data are being collected which will be used to estimate the probability of lander contamination during each exposure condition. A preliminary survey has not produced sufficient information to justify an elaborate analytical approach. A rough calculation suggests that there is high probability that the lander will impact material which has originated from the bus during lander deboost and descent. Whether contamination occurs depends upon the presence of viable organisms in the intercepted material. Because of the sterilizing effect of ultraviolet radiation and hard vacuum the mortality rate of microorganisms may be sufficiently high to make the probability of cross contamination acceptable low. Further analysis will be made of this problem. It is probable, however, that a definitive solution must be sought experimentally.

It is noted that the expanding "cloud" of material originating from the bus will also impact the Mars atmosphere. Mars contamination by this process therefore also would appear to be possible. If cross contamination by this process proves to be a serious consideration, then the probability of Mars contamination by vehicles which already have been launched may justify reconsideration of the basic sterilization requirement.

In the absence of definitive information, an operations plan must be developed which will minimize the probability of lander contamination both during trans-Mars flight and during the lander separation sequence. The following thus is proposed.

- The lander is to remain sealed within the sterilization shroud until initiation of the lander separation sequence near Mars.
- When a portion of the sterilization shroud is opened during the separation sequence to permit lander checkout and separation, no portion of the sterilized lander should "see" an unsterilized surface, i.e., straight line trajectories between sterile and nonsterile surfaces are to be avoided.
- The bus vehicle and the exterior of the sterilization shroud are to be surface sterilized prior to launch. The bus vehicle attitude control jets and propellant are to be heat sterilized.
- Surface coatings used on the bus vehicle should be of a self-sterilizing type.

### A. 2. 2 Temperature Control During Sterilization

As noted previously, sterilization of the lander requires that all parts of the lander vehicle and sterilization shroud be maintained at 135°C for 24 hours. It is to be emphasized that all lander systems must be a assembled, checked out and sealed into the sterilization shroud prior to dry heat sterilization. The presence of RTG power generating equipment within the sealed shroud makes control of temperatures during sterilization a significant problem. The lander structure as well as the structure which connects it to the sterilization shroud must be designed to provide heat transfer paths so that the entire assembly may be heated to 135°C without creating local "hot spots" and without requiring an excessively long heating period. Energy generated by the RTG's can be used to facilitate heating for sterilization. However, the thermal design for sterilization must be compatible with thermal control during all other phases of ground and flight operation.

As presently conceived the RTG's will be operated during sterilization at a temperature high enough so that their energy may be transferred out of the sterilization shroud by radiation to the shroud itself. The sterilization shroud temperature, of course, will be equal to or exceed 135°C during this period. Because of the high RTG temperatures which result, special care must be exercised to prevent overheating adjacent equipment. An analog simulation of the heat transfer problem will be used for system design. However, an actual "sterilization" vehicle will be required for accurate determination of the system's time-temperature histories which will be used to specify the details of the sterilization cycle and the required temperature instrumentation.

### A. 2. 3 Control of the Pressure Within the Sterilization Shroud

A negative pressure difference between the inside and outside of the sterilization shroud must be avoided to prevent contamination of the sterile lander by leakage of contaminated air into the shroud. Further, differential pressures across the sterilization shroud must be kept small to avoid complication of the sterilization shroud structure. The fact that a sealed shroud heated to 135°C will give rise to a 5.7 psia pressure differential and that launch will impose a full atmosphere pressure differential on the shroud leads one to consider an active system for controlling the pressure within the sterilization shroud. This system must be installed on the sterilization shroud at the time the shroud is sealed and must provide a controlled differential pressure across the shroud during sterilization, ground handling and launch. During sterilization the system must bleed air from the shroud heating, withstand temperatures required during sterilization and finally supply sterile gas to the interior of the shroud to maintain a positive differential pressure during cooling. The system further must supply sterile gas to the interior of the shroud to compensate for leakage while the spacecraft is on the ground, but must provide for controlled pressure bleed during launch.

### A. 2. 4 Selection of Temperature Resistant Materials

Lander components must be capable of undergoing exposure to a temperature of 135°C for 24 hours without serious degradation of their performance and reliability. Extensive testing has been done by various companies to identify satisfactory mechanical and electrical components. In general, components are available with which to execute lander design and construction. One noticeable exception, however, exists. Batteries

The aerodynamic shroud is next installed around the spacecraft and the forward section of the shroud is purged with ETO to surface sterilize exposed components of the bus vehicle. A seal placed around the outer rim of the sterilization shroud serves to isolate the forward section of the aerodynamic shroud from the aft section which contains the Centaur stage. Consideration must be given to the fire and explosion hazard of handling ETO above LOX filled stages. Surface sterilization of the bus is executed just prior to launch with a final purge of the ETO from the aerodynamic shroud with sterile gas continuing until the time of liftoff.

To implement these sterilization procedures requires launch site facilities, including mobile ETO sterilizing equipment, heat sterilizing equipment, instrumentation to monitor ETO concentrations, relative humidities, temperatures, etc., and bacteriological laboratories for biological control testing. When doing gaseous sterilization, biological controls, such as the use of paper strips impregnated with spores ("spore-strips"), may be required. These paper strips may be placed downstream of the sterilizing shroud where they are easily retrievable for sterility testing in addition to gas analysis and humidity analysis. Ground mobile cooling equipment is also required for heat rejection from spacecraft systems prior to launch.

Continuous monitoring and recording of sterilization procedures will be necessary until the moment of launch. This will require training of technicians and engineers who will execute and supervise the sterilization procedures.

Operations on both the bus and lander at the launch site will be accomplished under clean room conditions. Operators will wear head and shoe covers, clean room coats, sterilized gloves, and surgical type face masks.

capable of withstanding dry heat sterilization are not available. This places serious design restrictions upon the lander power system, since peak power and emergency power loads must be handled by the primary RTG system.

### A. 3 Sterilization Procedures Plan

Figure A. 1 presents the major sterilization operations and relates them to other prelaunch and flight operations. Basic to this plan is the assumption that cross contamination is possible during the lander separation sequence and hence that portions of the bus must be sterilized. The impact of this assumption is apparent from the sterilization procedures plan.

#### A. 3. 1 Prelaunch Sequence

Handling of lander components will be controlled to minimize the degree of contamination of the assembled lander vehicle. Periodic decontamination of components may be necessary to meet the Lander contamination criterion of less than $10^9$ organisms. Following lander checkout and lander/bus interface tests the lander vehicle is installed within the sterilization shroud. Any tests or calibration to be performed following this step must be performed without direct access to lander components. The lander system then is heat sterilized followed by performance verification tests to insure system integrity.

The bus vehicle attitude control reaction jets and propulsion systems are separately manufactured and sterilized with dry heat. These subsystems are integrated into the bus just prior to mating with the lander assembly. Exposed surfaces of the bus will be decontaminated with ETO (12 percent ethylene oxide and 88 percent Freon-12) prior to mating with the heat sterilized encapsulated lander. Following mating and interface checkout the spacecraft will be installed on the Centaur stage and boost vehicle.

### A. 3. 2 Flight Sequence

Sterile gases will be vented from the sterilization shroud and aerodynamic shroud during ascent. Following booster burnout, the aerodynamic shroud will be jettisoned (~300,000 feet) and the Centaur stage fired to inject the payload into Earth orbit. A second Centaur firing subsequently injects the payload into a trans-Mars trajectory. The surface of the bus is exposed as soon as the aerodynamic shroud is jettisoned, and hence the surface of the bus can become contaminated by Centaur operation. The degree of bus contamination however should be acceptably low.

The lander is maintained within its sterile shroud during flight to Mars and is opened only at that point required for lander checkout and separation.

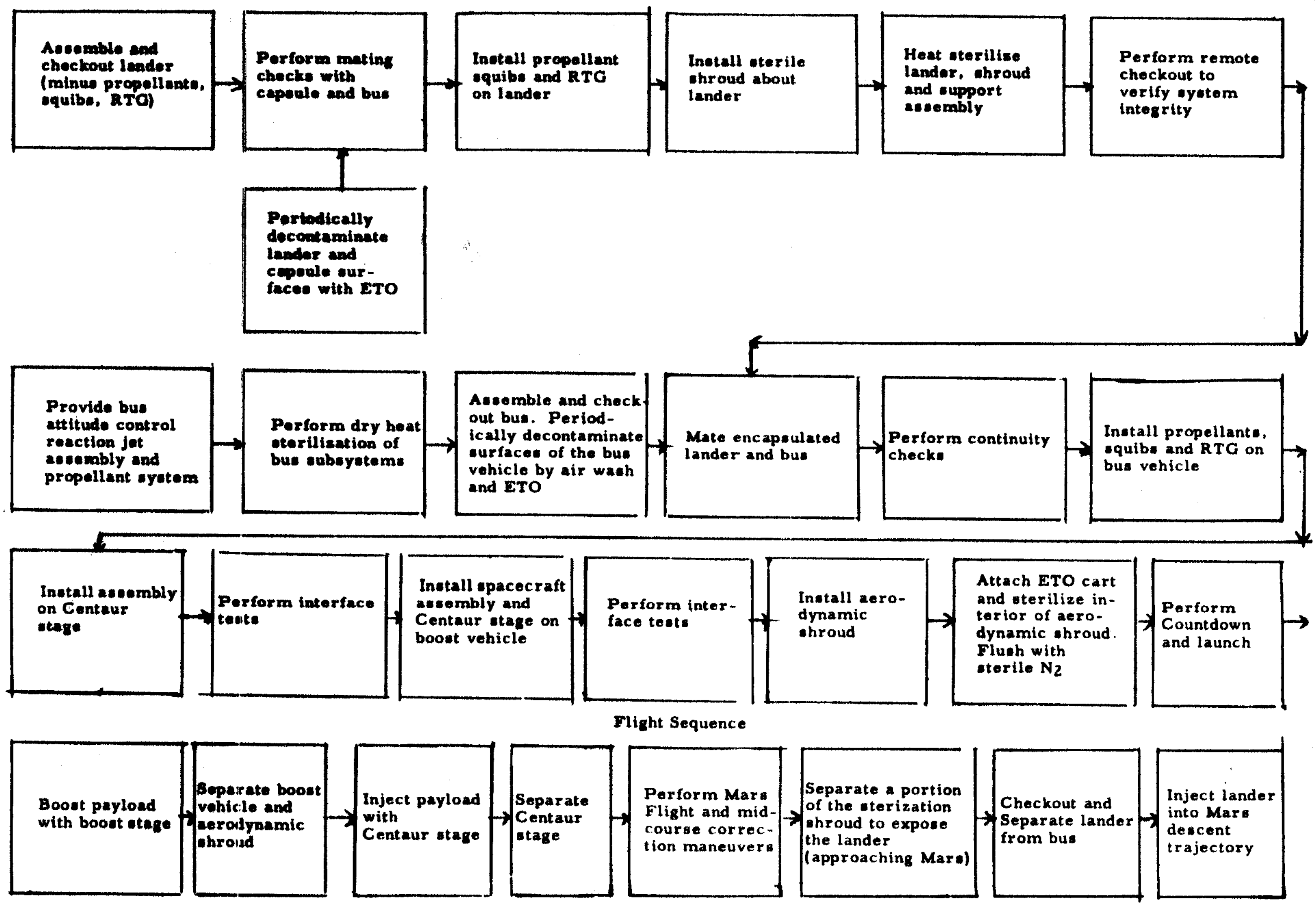



Figure A.1 Sterilization Procedures Plan