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A VENUS LANDER PROBE FOR MANNED FLYBY MISSIONS

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ABSTRACT

The Venus lander probe conceptual design described in this paper is part of a larger study of manned flyby missions to Mars and Venus. The probe is one of a group of instrumented unmanned vehicles which might be deployed from the manned flyby spacecraft shortly before planetary encounter. These probes would provide in situ and long term planetary measurements which cannot be made from the manned flyby spacecraft. The conceptual design was developed to determine how a Venus lander probe might operate on a manned flyby mission, what data it could contribute, and approximately how much it would weigh. Like the study of the manned planetary flyby missions, the study of this probe is in the preliminary stages.

The probe design includes a landing capsule which makes measurements on the surface of Venus and an entry spacecraft which transports the landing capsule through space and protects it during the high speed atmospheric entry. After entry, the capsule is separated from the entry spacecraft, and is fin stabilized during its fall to the surface. It lands on a crushable impact limiter which covers the lower half of the capsule. Protected by thermal insulation and cooled by melting ice, the capsule survives for about one hour on the surface and transmits data directly to the manned vehicle. The weight of the landing capsule is approximately 150 pounds. The weight of the entire probe is approximately 900 pounds.

A VENUS LANDER PROBE FOR MANNED FLYBY MISSIONS

INTRODUCTION

Manned planetary flyby missions have been considered as one of several possible means of exploring Mars and Venus. This class of mission represents the easiest type of manned planetary mission and is a natural starting point for developing a manned planetary flight capability. Single, dual, and triple planet flyby opportunities have been identified with the triple planet opportunity affording two Venus passages. The manned vehicle follows an essentially ballistic or free-return trajectory flying by Venus, then Mars, then Venus again, and returning to Earth about two years after launch.

During each of the three planetary encounters, the crew performs extensive remote sensing experiments, and much of the crew's time between planetary encounters is devoted to space science and astronomy experiments, particularly those which take advantage of the spacecraft's trajectory around the sun. However, the most important means of collecting scientific data at the planets is the use of unmanned spacecraft, called probes, which are deployed from the manned vehicle shortly before encounter with the planet.

Several types of probes could be carried on a single manned flyby mission. A typical probe complement might include orbiting probes and landing probes for both Mars and Venus. Probes equipped with balloons to float in the dense atmosphere of Venus have also been considered. A soft landing Mars probe which carries a rocket capable of returning a sample of the surface material to the manned flyby vehicle seems to be another possibility.

Each of the probes would carry several experiments appropriate to its objectives and capabilities. The orbiting probes would carry principally remote sensing experiments while the other probes would make primarily in situ measurements. The data would be telemetered to the manned vehicle where it would be processed and relayed to Earth. Some long life probes such as the orbiters would also have the capability of transmitting directly to Earth at a lower data rate. The probes would rely on the manned vehicle for navigation and guidance during their deployment, and for real time control of the experiments during the first few days of operation.

The probes are especially important to the manned flyby missions because they extend the duration of the planetary experiments beyond the few hours of encounter and because they can carry experiments where manned vehicles cannot go - either because of planetary quarantine restrictions or because of the hostility of the environment. One of the most forbidding environments faced by any of the probes is the Venus surface environment. This paper describes the conceptual design of a probe to conduct experiments in that environment.

The purpose of this conceptual design is to aid the study of manned planetary flyby missions by showing how a Venus lander probe might operate on such a mission, approximately how much it would weigh, and what measurements it would make. It represents a part of a much larger effort to understand the capabilities and limitations of the manned flyby mode of exploring Mars and Venus, and, like the larger effort, is still in the early stages of investigation.

OBJECTIVE OF THE VENUS LANDER PROBE

The objective of the Venus lander probe is to make scientific observations of the surface and near surface environment of Venus and to determine the feasibility and usefulness of surface landing vehicles in the scientific exploration of Venus. If this mode of operation proves successful, the lander probe would act as a precursor to more elaborate landing probes such as a long life surface geophysics laboratory probe. The lander probe should demonstrate the capability to make a successful landing on Venus and to erect and deploy experiments in a useful manner. It should determine which of the various types of terrain discerned by remote sensors, such as radar, is more suitable for landing so the subsequent landers can be targeted to favorable areas. It should also return basic data about the surface topography, bearing strength, and atmospheric conditions to aid in the design of subsequent landing vehicles.

MISSION PROFILE

Several lander probes, enclosed in individual sterilization canisters, are stored in the probe hangar of the manned vehicle. Deployment of these probes takes place one to three days prior to Venus encounter. Deployment consists of ejecting the probe and sterilization canister from the probe hangar, separating the probe from the canister, and establishing communications between the manned vehicle and the probe.

The probes are then commanded to the proper attitude for an injection propulsion maneuver. This maneuver is designed to stagger the arrival times of the probes at the planet. The first probe will intercept the planet as early as three hours before the periapsis passage of the manned vehicle. This staggered early arrival time is needed since the beam width of the receiving antenna on the manned vehicle is too narrow to receive data from all the probes simultaneously. A sterilizable liquid propellant rocket shown in Figure 1 is used for the injection and subsequent midcourse maneuvers.

During cruise the probe is attitude stabilized using the sun as one reference direction as indicated in Figure 2. Roll control is maintained using an Earth, Venus, or stellar reference. During the propulsion maneuvers and prior to atmospheric entry the vehicle is stabilized by an inertial reference unit.

Just before entry the landing capsule is activated and its operational status is monitored and transmitted to the manned vehicle by the entry spacecraft. All the electrical subsystems in the landing capsule including the experiments operate continuously from this time until the batteries are exhausted. However, since the capsule is contained within the metallic aeroshell of the entry spacecraft, its transmissions cannot be monitored directly by the manned vehicle before entry. After the capsule operation has been verified by the entry spacecraft, the spaceflight subsystems are jettisoned in preparation for entry. These subsystems are mounted on a single support ring to facilitate jettisoning.

The probe enters the atmosphere, becomes aerodynamically stabilized, and decelerates. As terminal descent velocity is approached, the landing capsule is separated from the entry spacecraft by ejection downward through the nose. Aerodynamic forces aid the separation since the ballistic coefficient of the capsule is about three times as great as that of the aeroshell. As soon as the capsule clears the aeroshell, its radio transmissions can be received by the manned vehicle.

The fin stabilized capsule accelerates to terminal velocity. It impacts the surface at a velocity of about 150 fps in the minimum density model atmosphere of Reference 1 (5000 mb). The impact velocity based on the surface conditions measured by the Russian Venus IV probe is about 75 fps. A crushable steel honeycomb impact limiter is used to absorb the energy of impact, thereby limiting the deceleration and rebound. The spring loaded legs are unlatched on impact and fold outward to erect and stabilize the capsule, lifting the impact limiter off the surface as shown in Figure 3. This action also uncovers the window of the panoramic television camera and folds the fins down to reduce the overturning moment of surface winds. The manned vehicle monitors the data transmission during capsule descent, landing, and surface operations.

The operating lifetime of the Venus landing capsule is limited by the time the high gain antenna on the manned vehicle is available to receive data. The capsule communications subsystem is sized to transmit the total data return (about 10^7 bits) in one hour starting when the manned vehicle is still three hours from periapsis passage.

During this time the manned vehicle might also be receiving data from another probe which is within the beam of the receiving antenna but transmitting on a different frequency. After this hour, the manned vehicle would monitor other probes with the high gain antenna. Because the range to the manned vehicle is reduced, a Venus lander probe could transmit its data in about half an hour starting two hours before periapsis. The useful life of the probe is therefore one hour or less and is ended when the approaching manned vehicle switches its receiving antenna to a different probe. The capsule battery and thermal control subsystems are sized to last two hours.

PAYLOAD SUBSYSTEM

The payload subsystem for the Venus lander probe consists of instruments to measure the surface topography, bearing strength, and atmospheric conditions. Table I lists the instruments and gives their estimated weight, power consumption, and data output during the nominal one hour lifetime. The instruments are designed to survive a 3000 g shock from 150 fps as are all the landing capsule subsystems.

A television camera would be used to observe the local terrain, degree of surface illumination, and penetration of the soil penetrometers. A mechanically scanned television camera, known as a facsimile camera, was selected for this application because of the convenient panoramic format. It makes a panoramic scan in one hour and has a vertical field of view of about 50° as shown in Figure 4. The generated data rate is compatible with the data transmission rate of the communications system, thereby making television data storage unnecessary. Facsimile television cameras have survived 3200 g shock tests from 180 fps[2].

The camera has a tubular shape because it is essentially a narrow field telescope. A solid state photosensor located at the focus generates the video signal. A small mirror within the camera housing is moved to scan the field of view vertically through 50° at a rate of about one scan per second. The camera itself is mounted on a turntable which rotates about one revolution per hour thereby providing the horizontal scan. The resulting panoramic picture has a resolution of about 5 milliradians, i.e., black and white line pairs 5 mm wide can be resolved at 1 meter distance. The turntable has room for as many as six facsimile cameras each about 1 inch in diameter and 10 inches long. Only one camera is used in this conceptual design because of data communications limitations, but additional cameras might be added for multispectral or full color photographs if the communications capability were increased. A continuous ring-shaped window affords an unbroken view of the surrounding terrain.

The camera is equipped with an illuminator which is turned on automatically if the natural illumination is too low. The illuminator projects a narrow beam of light which is mechanically synchronized with the camera scanning beam. By illuminating only the part of the scene being observed, the illuminator power requirements are significantly reduced compared with omnidirectional illumination. Since the camera instantaneous field of view is about 1.9 milliradians wide, a 2 mr wide illuminator beam would be adequate. The theoretical power saving compared with an omni-directional illuminator is:

$$\frac{4\pi}{(0.002)^2} = \pi \times 10^6$$

Assuming only 10% of the light output of the light source can be focused into the beam, this still provides a substantial ($\pi \times 10^5$) reduction in illuminator power.

The required illuminator power level can be calculated for a non-absorbing or non-scattering atmosphere assuming it is desired that the highlights (normal surfaces) of a dark subject (reflectivity = 0.05) be detected at 100 feet from a camera without natural illumination. The threshold luminance detected by a facsimile camera is about 12 ft-lamberts[3]; hence an illumination of $\frac{12}{.05} = 240$ foot candles is needed at 100 feet from the illuminator. This requires $4\pi(100)^2(240) = 3 \times 10^7$ lumens from an omni-directional source. At 20 lumens per watt from a tungsten filament light bulb, this requires 1.5 million watts. For the synchronous illuminator it requires $(1.5\times10^6) \div \pi\times10^5 = 5$ watts of electrical power. This power level is similar to that required by the camera. The facsimile camera is a proven device having been used successfully on Luna IX; the synchronous illuminator for the facsimile camera is currently only a concept.

A differential pressure anemometer is used to measure the wind speed and direction at intervals of about 1 second throughout the mission. It consists of a series of pressure taps equally spaced around the cylindrical stem which projects up from the camera housing. Adjacent pressure taps are connected across differential pressure transducers which are located in a compartment below the camera housing as indicated in Figure 4. These transducers are read simultaneously to provide a measure of the instantaneous pressure distribution around the cylindrical anemometer stem.

The free stream dynamic pressure (q_∞) and the direction of the wind can be found by comparing the measured pressure distribution with the known pressure distribution caused by flow around a cylinder as shown in Figure 5, reproduced from Reference 4. The maximum pressure occurs on the upstream ($\phi=0^o$) side of the anemometer. The wind speed (V_∞) can be calculated from the definition of dynamic pressure:

$$q_\infty = 1/2\ \rho\ V_\infty{}^2$$

The density of the atmosphere (ρ) can be calculated from static pressure and temperature measurements which are made by the landing capsule.

Figure 5 indicates that two pressure distributions around the cylindrical stem are possible depending on the flow Reynolds number. At Reynolds numbers below about 3×10^5, the flow is subcritical; above this value, a transition to supercritical flow occurs.(4) Since a Reynolds number of 3×10^5 corresponds to wind velocities of about 5 meters per second on the surface of Venus based on the Russian data, such a flow transition is possible. This complicates the interpretation of the pressure distribution data in terms of q_∞ since it is first necessary to find out whether the flow is subcritical or supercritical. The number of simultaneous differential pressure measurements needed to reliably determine wind velocity to any given degree of accuracy under these conditions has not been established; six equally spaced pressure taps and five differential pressure measurements were assumed to be adequate based on a brief inspection of Figure 5. Additional pressure taps would entail only very small weight penalties, and the data penalty is negligible compared with the facsimile data.

Atmospheric pressure is measured by a high pressure ($\sim$ 1000 psi) transducer connected to one of the anemometer lines. Since the inside of the capsule is evacuated, this gauge would read absolute pressure. Temperature sensors on the landing capsule exterior measure the ambient temperature, and additional sensors are used to monitor the temperature of certain internal parts. Thermocouples might be used for the temperature sensors since melting ice will be available in the thermal control subsystem to stabilize the reference junction temperature.

An impact accelerometer measures the deceleration time history of the impact as the capsule lands on the surface. From this information and a knowledge of the impact velocity and physical properties of the vehicle, rough information about the surface dynamic bearing strength can be deduced. The accelerometer output is sampled several times during impact, digitized and stored for transmission to the manned vehicle.

The inclinometer measures the attitude of the capsule with respect to the gravitational vector. Since the output of this instrument is sampled several times during the one hour surface lifetime, it gives information about the stability of the vehicle. This information, when combined with the surface wind data, may give some indication about the stability of the surface. The instrument consists of an insulating shell partly filled with mercury. Electrical contacts on the inside of the shell sense the position of the mercury.

The soil penetrometers are four springloaded plungers attached to the stabilizing legs. A combination of different plunger areas and spring rates could be used to gain information about the surface static bearing strength. The plungers are released shortly after landing and the depth of penetration is indicated on graduated rods visible in the television picture. Release might be effected by a fusable element designed to melt after a short exposure to the surface thermal environment. The soil penetrometers are the only instrumentation exposed to the surface environment; all other instruments are within the cool, low pressure environment of the capsule.

CAPSULE OPERATIONAL SUPPORT SUBSYSTEMS

Nearly all of the 150 pound landing capsule weight consists of operational support subsystems needed to power, protect, and handle the data from the 8 pounds of instruments in the payload subsystem. Table II shows the estimated weights and power requirements of the landing capsule subsystems. All of these subsystems are designed to survive a 3000 g landing shock, and like all of the probe subsystems, are heat sterilizable.

The data handling subsystem consists of signal conditioning circuits, analog-to-digital converters, data storage and a sequencer to control the order of data transmission. Since nearly all the data is produced by the television camera, data from other instruments is stored until it can be transmitted during the return scan period of the television sequence. All the data stored during the approximately 1 second television scan period is cleared from storage during the return scan period. Since the return scan period lasts only a few hundredths of a second and the communications subsystem transmits data at about 2500 bits per second, the total data stored in each period is only about 100 bits. An additional 200 bits of storage is needed for the accelerometer data, which is collected during impact.

The communications subsystem consists of an S-band transmitter radiating 7 watts through a wide beam antenna. A traveling wave tube is used as the final amplifier because of its high efficiency. This reduces the load on the power subsystem and the thermal control subsystem. Traveling wave tubes with efficiencies of 33% and capable of withstanding 3,000 g shock loads are under development.[5] A cup antenna mounted at the top of the capsule radiates power with an effective power gain of at least 0 db for angles within 54° of the capsule axis.[6] A rigid coaxial transmission line connects the transmitter to the antenna.

The capsule transmitter was sized to transmit 2500 bits per second assuming a 30 foot diameter receiving antenna on the manned flyby vehicle and an average flyby velocity of 10.5 km per second. This velocity results in a maximum transmission range of 113,000 kilometers three hours before periapsis. At this range the communications system has a margin of about 4 db when the capsule is tipped 54° away from the line of sight to the manned vehicle. The transmitting antenna provides zero gain at this angle; it provides about 6 db gain on axis.[6] Negligible loss through the atmosphere was assumed for the 2300 MC transmission frequency.

The power subsystem consists of two silver-zinc batteries of 120 watt hours capacity each and an inverter which provides alternating current for the camera motor. Either battery could power the capsule for about two hours. The redundant battery was considered in the landing capsule weight estimate since shock-resistant sterilizable silver-zinc batteries have not been developed to a high state of reliability.

The structure subsystem consists of a pressure vessel, an inner shell, and an internal support structure. The pressure vessel protects the internal parts from the high surface atmospheric pressure on Venus and is made of stainless steel with a cylindrical glass window for the television experiment. It is by far the heaviest component of the landing capsule, and, since most of its weight is in the bulbous lower end, it contributes to the aerodynamic and surface stability of the vehicle. The inner shell forms a space about 1" wide for thermal insulation inside the pressure vessel. It is also made of stainless steel and is connected to the pressure vessel only by a one inch diameter stainless steel tube at the top. This tube serves to position the inner shell within the pressure vessel, as a spindle for the TV turntable, and as a conduit for the lead-in wires, tubes, and cables. Asbestos spacers also help to position the inner shell within the pressure vessel, and bumpers are provided to limit the relative displacement of the inner shell and the pressure vessel during the landing impact. The inner shell contains internal support structure which forms compartments for the various internal components. Some of these compartments, such as the battery compartment, are pressurized; otherwise the inside of the pressure vessel is evacuated to reduce convective heat transfer into the capsule.

The thermal control subsystem uses melting ice to stabilize the temperature inside the capsule and thermal insulation to reduce the heat flow into the capsule. The insulation is evacuated, laminated, reflecting foil "superinsulation" which almost completely surrounds the internal parts of the capsule, as shown in Figure 4. Assuming an outer wall temperature of 800° F and an inner wall temperature of 32° F, the effective thermal conductivity of this material is about 2×10^{-4} Btu-ft/hr-ft^2 °F.(7) The area of the capsule is less than 10 square feet; therefore, a 1 inch layer of this insulation allows only about 18 Btu/hr to enter the capsule.

The heat entering the capsule through the insulation is small compared with the internally generated heat. The heat generated in the capsule is 55 watts minus 7 watts radiated = 48 watts or 164 Btu/hr. If the additional heat leaks due to the lead-in wires, tubes, etc. can be controlled, very little ice is needed since ice absorbs 144 Btu per pound in melting.

One coaxial cable, 6 anemometer pressure tubes, and several thermocouple wires must be connected from the inside of the capsule to the outside across the insulation barrier. In addition, several electrical wires leading in from an umbilical plug surrounding the cup antenna are desirable for battery charging and testing. If connected directly across the 1" thermal insulation layer, these lead-in connections would conduct a considerable amount of heat into the capsule. This heat leak is reduced by making the lead-in connection long; they extend more than a foot between the hot outer wall at the top of the anemometer stem and the cool interior of the capsule. This length reduces the heat conducted through the lead-in connections to about 20 Btu/hr. The 1" diameter stainless steel tube, which serves as a conduit for the lead-in connections and as a structural member, represents a direct metallic connection between the inside and outside of the capsule, but, because of its length, it conducts only 30 Btu/hr into the capsule. The asbestos spacers used to center the inner shell in the pressure vessel conduct about 50 Btu/hr into the capsule.

The gap in the thermal insulation at the television camera window allows about 60 Btu/hr to be radiated into the interior of the capsule. The total heat load on the thermal control subsystem, summarized in Table III, is 342 Btu/hr. This requires 2.4 pounds of ice per hour; 5 pounds of ice are provided. The thermal insulation and asbestos spacers weigh about 4 pounds, bringing the thermal control subsystem weight to a total of 9 pounds.

Ice was selected as a coolant because it absorbs a reasonably large amount of heat in melting, melts at a convenient temperature, and does not generate high pressures at sterilization temperatures. The capsule would be assembled with five pounds of water sealed into a compartment which is large enough to allow the water to freeze without bursting. The entire probe could then be heat sterilized in its sterilization canister and subsequently cooled sufficiently to freeze the water. The vapor pressure of water at a sterilization temperature of 135° C is only about 45 pounds per square inch absolute. The landing capsule would be left full of air during the heating and cooling process so as to facilitate heat transfer into and out of the capsule. The capsule would be evacuated at some later time, possibly by venting it to space after deployment. Leaving the capsule full of air until after deployment would facilitate heat transfer from the capsule during storage on the manned vehicle.

The stabilization subsystem performs the dual functions of stabilizing the landing capsule during its descent and erecting and stabilizing the capsule on the surface. It consists of four hinged stabilizing legs equipped with sheet metal fins. With the legs latched in the "descent" position as shown in Figure 4, the fins cause the landing capsule to fall in a stable attitude so that it will land on the crushable impact limiter. The legs are unlatched on impact and are deployed by spring driven actuators as indicated in Figure 4. The legs are long enough to lift the impact limiter off a level surface; therefore, footpads are provided to reduce the surface bearing stress. Static stability on a slope of 45° and the ability to resist the overturning moment of 20 meter per second winds on a level surface were estimated for the configuration shown. The footpads float to provide some stability on a liquid surface.

The impact limiter subsystem limits the landing deceleration to less than 3000 g. It consists of an approximately hemispherical shell of maraging steel honeycomb about 28" in outside diameter and 6" thick. The use of this material as an impact limiter has been studied experimentally[8] and found to be well suited to that purpose because of its ability to crush to a small fraction of its original thickness while absorbing the impact energy with very little rebound. Hexagonal steel honeycomb with an average crushing strength of about 900 pounds per square inch would crush to a depth of about 4 3/4 inches with a peak deceleration of about 2000 g if the capsule impacted an unyielding flat surface at a speed of 150 fps. This impact speed occurs only in the minimum atmospheric density model of Reference 1.

The subsystem conceptual designs described in this and the following sections have been investigated only far enough to obtain reasonable assurance that such a design is possible and to arrive at an approximate estimate of the weight. No tradeoff analyses were made either for a single subsystem or for the entire probe.

ENTRY SPACECRAFT SUBSYSTEMS

The entry spacecraft carries and protects the landing capsule during the atmospheric entry and the space flight phases of the mission. The subsystems which comprise the entry spacecraft are shown in Table IV. Figure 1 shows the general arrangement of the landing capsule within the entry spacecraft. Only the entry subsystems consisting of the landing capsule, the capsule ejector, and the aeroshell survive the atmospheric phase. These subsystems total about 370 pounds and represent the entry weight of the probe. Approximately 286 pounds of space flight subsystems are jettisoned just before atmospheric entry. The rocket engine propellants and the sterilization canister, which is jettisoned during deployment from the manned vehicle, bring the gross probe weight to about 900 pounds, as shown in Table V.

The capsule ejection subsystem includes spring loaded capsule ejection mechanisms, linear shaped charges to cut the connecting structure and to open the nose of the aeroshell, and the controls which initiate the separation maneuver. The 50 pound weight allowance includes the structural members which connect the landing capsule to the aeroshell. It also includes the umbilical connection to the landing capsule.

The aeroshell subsystem is based on the design of an aeroshell for a small Venus entry probe given in Reference 9. It consists of a 35° half-angle spherecone shaped vehicle fabricated of aluminum with a phenolic nylon ablative heat shield. It is capable of vertical entry into the Venusian atmosphere at about 40,000 fps.

The referenced design has a weight of 47 pounds for a 55 pound payload; this was scaled up to 170 pounds for the 200 pound payload of the Venus lander probe aeroshell. The diameter was increased from 30" to 57" to keep the ballistic coefficient at 0.8 slugs/ft^2.

The space flight subsystems give the lander probe space flight capabilities similar to a small three-axis stabilized spacecraft such as the Mariner or Lunar Orbiter. The weight estimates for these subsystems were based on the detailed weight statement for a proposed Mars and Venus orbiter(10) and represent technology of the mid-1960's. All of these subsystems are attached to a ring-shaped support structure so that they can be separated from the aeroshell prior to entry.

A telecommunications subsystem which allows the probe to be tracked from the manned vehicle while providing low (~100 BPS) data rate command and telemetry channels is used. It consists of a transponder, a command detector, a telemetry encoder, and an omni-directional antenna. The omni-antenna is deployed on a folding boom.

The electronics subsystem includes the autopilot electronics, the command and data handling electronics, the spacecraft timer, programmer, and switching equipment. This subsystem is housed within the temperature controlled support ring. The weight shown in Table IV includes some redundant circuits.

The attitude control subsystem includes the inertial, stellar, and solar attitude sensors and the cold nitrogen attitude control propulsion equipment. The stellar and solar attitude sensors are used during the cruise mode of space flight; the inertial sensors are used during the deployment and propulsion maneuvers and just prior to entry into the atmosphere of Venus.

Compared with most planetary spacecraft, the power requirements for the entry spacecraft are small. This is because of the low data rate of the communications subsystem, the almost negligible power drawn by the payload, and the simple mission profile with short off-sun times. A conventional solar cell and battery subsystem is used. The solar cells are mounted on the top of the structural support ring. The weight shown in Table IV includes the battery and its charge controller and regulator.

The weight estimates for the support ring structure and the assembly and integration subsystem were based on the dry weight of the spacecraft (total weight less propellant weight). According to Reference 10, the structure represents about 18% of the dry spacecraft weight and the assembly and integration subsystem is about 8%. The weight allowance for the support ring structure on the Venus lander probe was estimated to be 12% of total spacecraft weight because of the structural contribution of the aeroshell and the absence of solar panel support structure and actuators. The resulting 78 pound weight estimate is slightly greater than the weight of the Mariner IV structure.[11] The assembly and integration subsystem, which includes wiring, hardware, and thermal control surfaces, is about 8% of the 656 pound spacecraft dry weight.

The propulsion subsystem was sized to provide 500 meters per second of velocity change using a sterilizable propellant with a delivered specific impulse of 300 seconds. The propellant mass fraction was assumed as 0.85; the propellant mass fraction in Reference 9 is 0.87 for a somewhat larger propulsion subsystem. The 500 meters per second total velocity change includes 435 meters per second for injection and 65 meters per second for midcourse maneuvers. The injection velocity change provides about one hour of early arrival for each day of launch before periapsis passage of the manned vehicle. Therefore, to achieve three hours early arrival, the Venus lander probe would be launched toward Venus three days before the manned vehicle encounters the planet.

The sterilization canister is a lightweight hermetically sealed shell which completely encloses the Venus lander probe. Its purpose is to prevent biological contamination of the sterilized spacecraft, and it is not removed until the probe is clear of the manned vehicle. It consists of two truncated right circular cones with a short cylindrical mating section as indicated in Figure 1. The total area is about 75 square feet. Based on the 1.6 pounds per square foot estimated for the sterilization canister of the Voyager landing capsule,[12] the Venus lander probe sterilization canister would weigh about 120 pounds.

Many of the subsystem weight estimates for the Venus lander probe were based on existing hardware; therefore, they are based on a technology which predates the Venus lander probe by at least 10 years. No weight reduction due to technology advancement was assumed. However, no weight allowance was made for the sterilization requirement other than the weight of the sterilization canister. The extent to which the sterilization requirement will offset the weight reduction afforded by advanced technology is difficult to predict. It was assumed that sterilized spacecraft of the late 1970's will weigh about the same as comparable unsterilized spacecraft of the mid 1960's.

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Attachments
  Figures 1-5
  Tables I-IV

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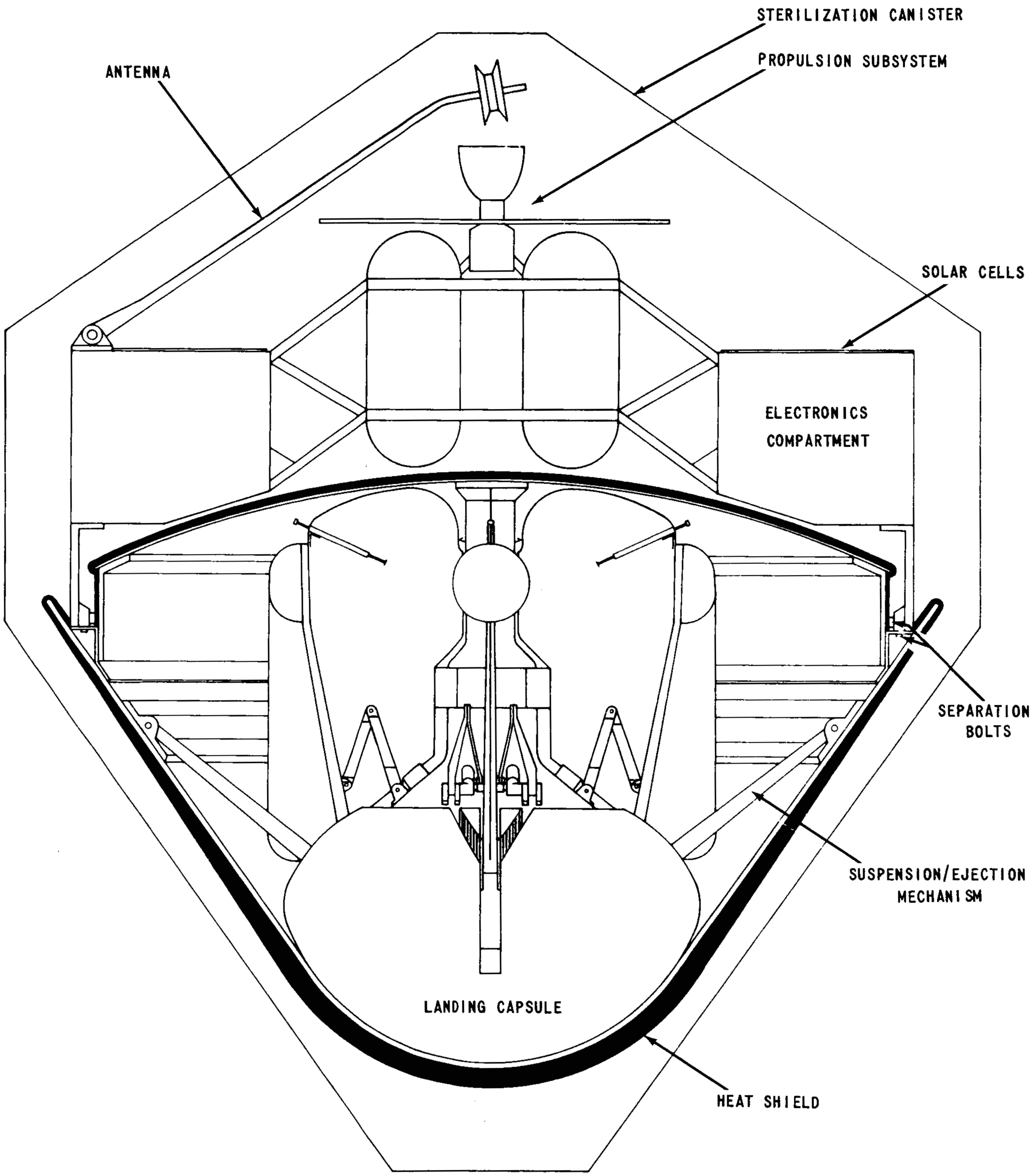

FIGURE 1 - VENUS LANDER PROBE GENERAL ARRANGEMENT

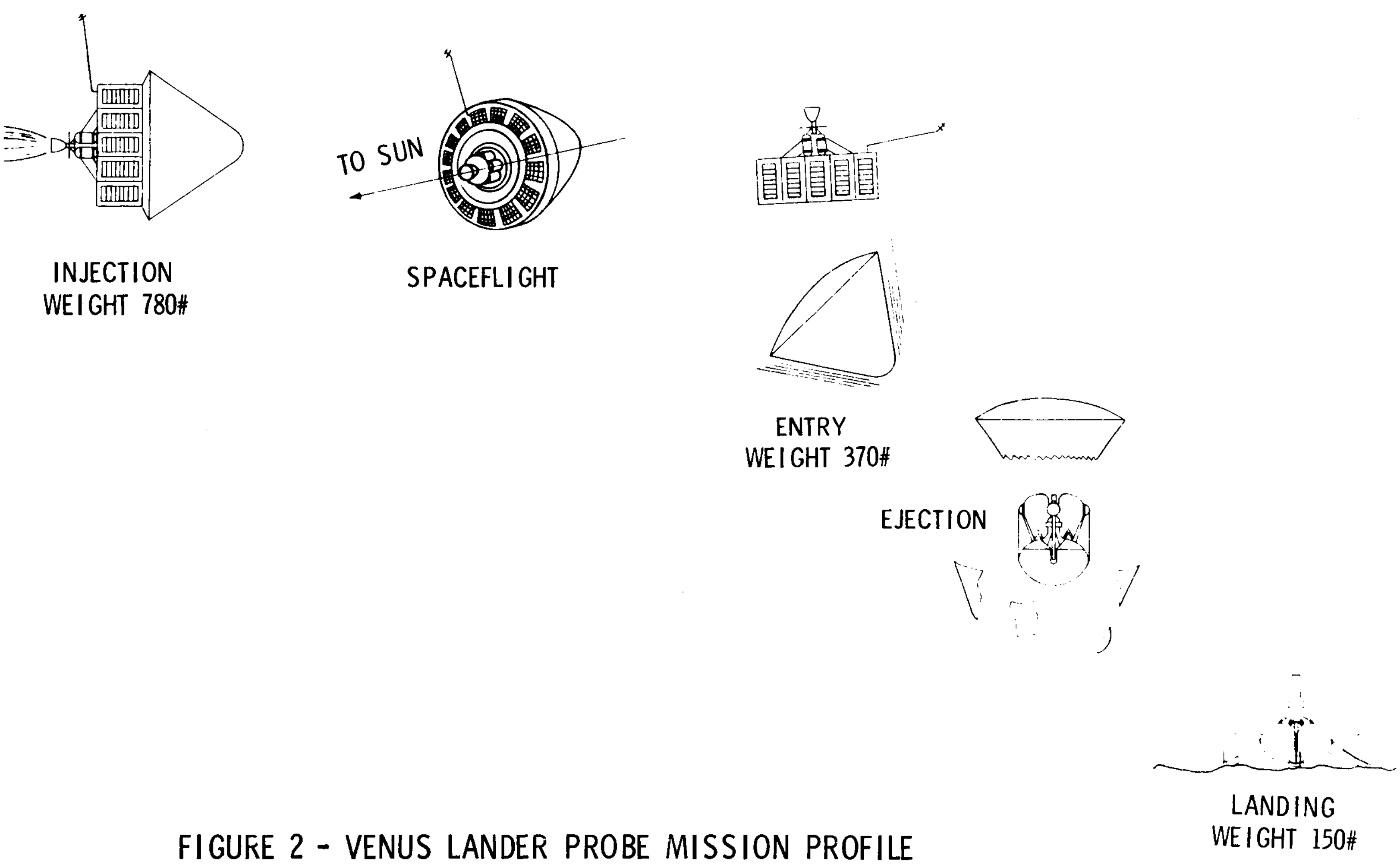

FIGURE 2 - VENUS LANDER PROBE MISSION PROFILE

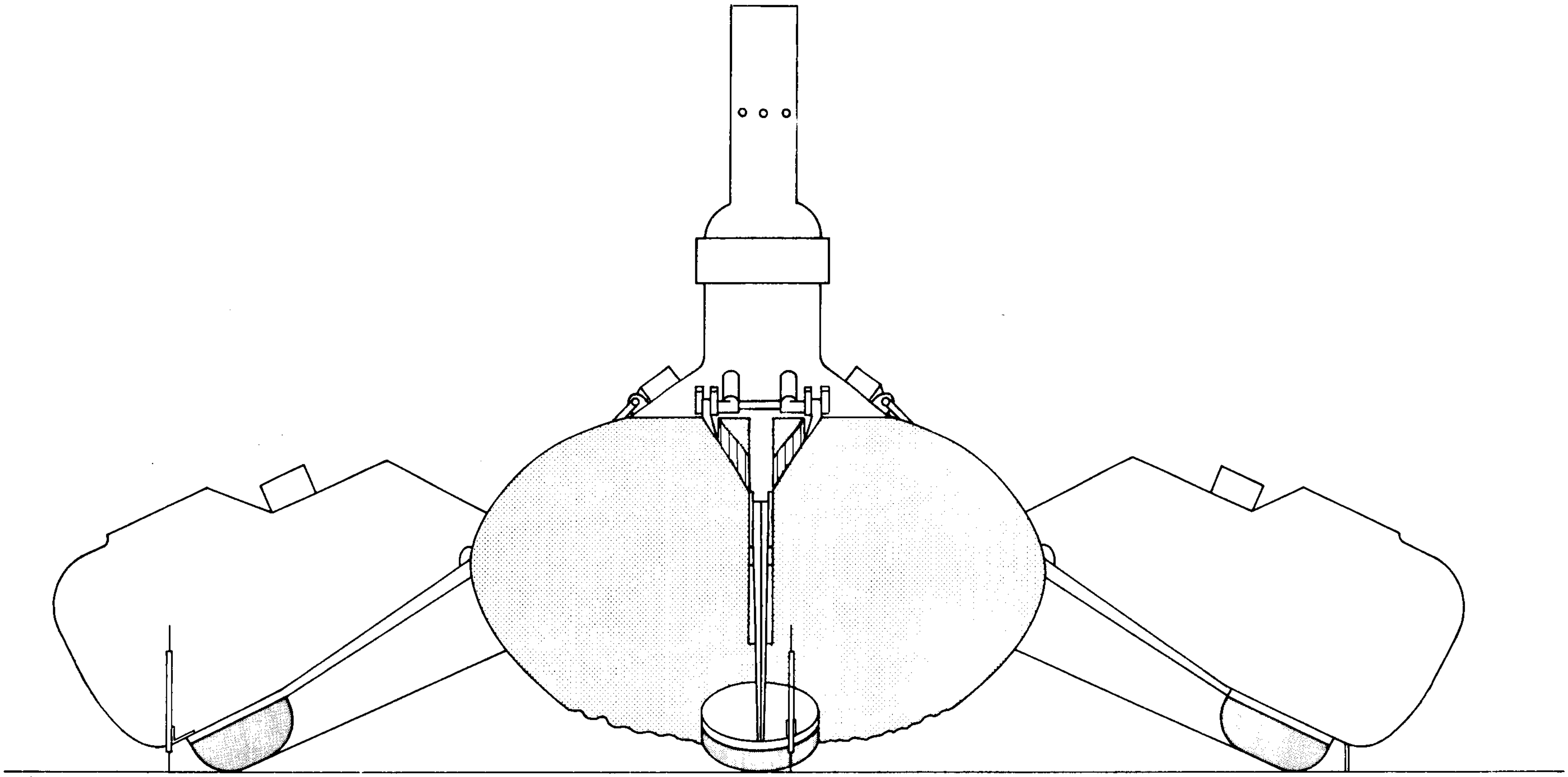

FIGURE 3 - LANDING CAPSULE IN LANDED CONFIGURATION

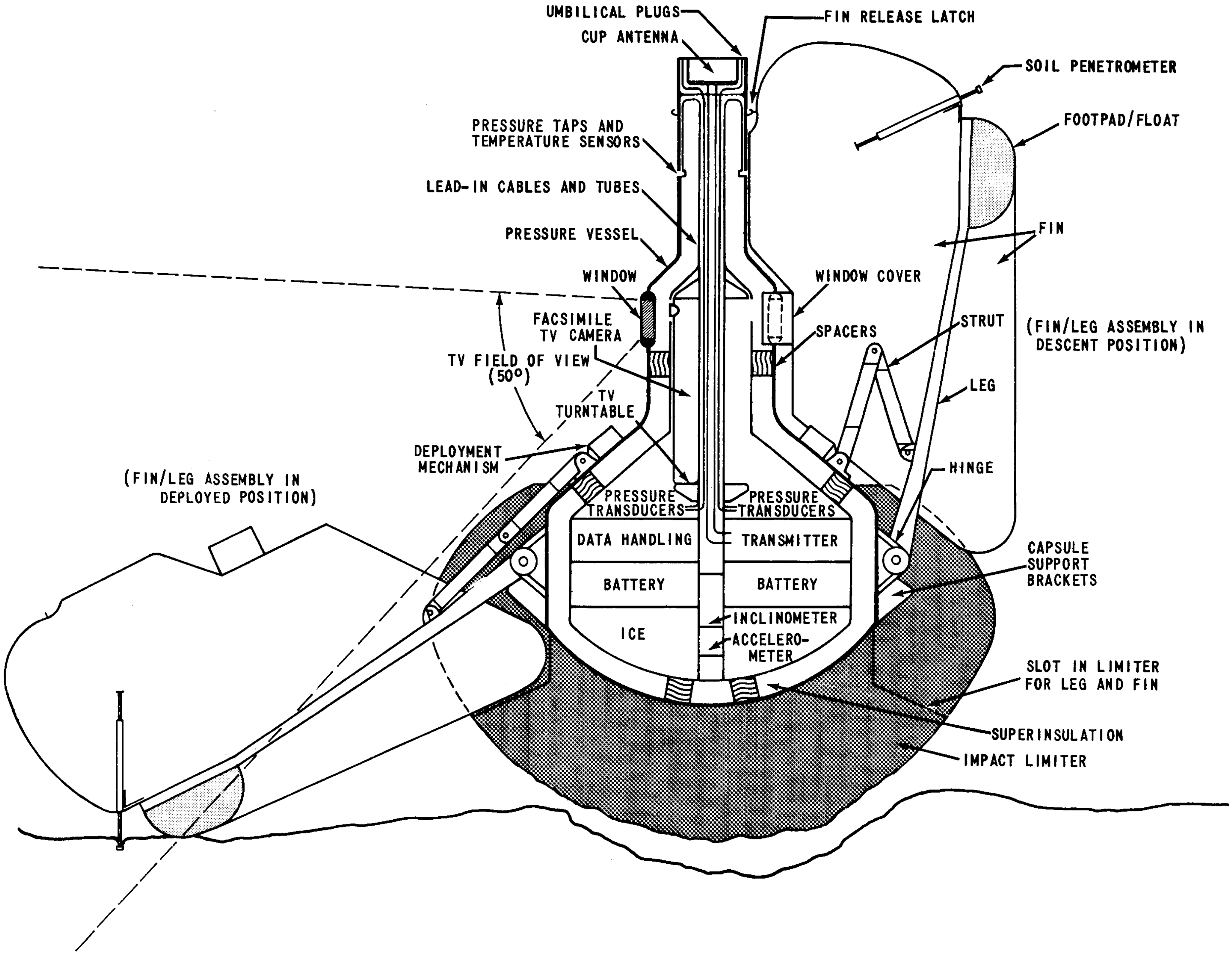

FIGURE 4 - LANDING CAPSULE INTERNAL ARRANGEMENT

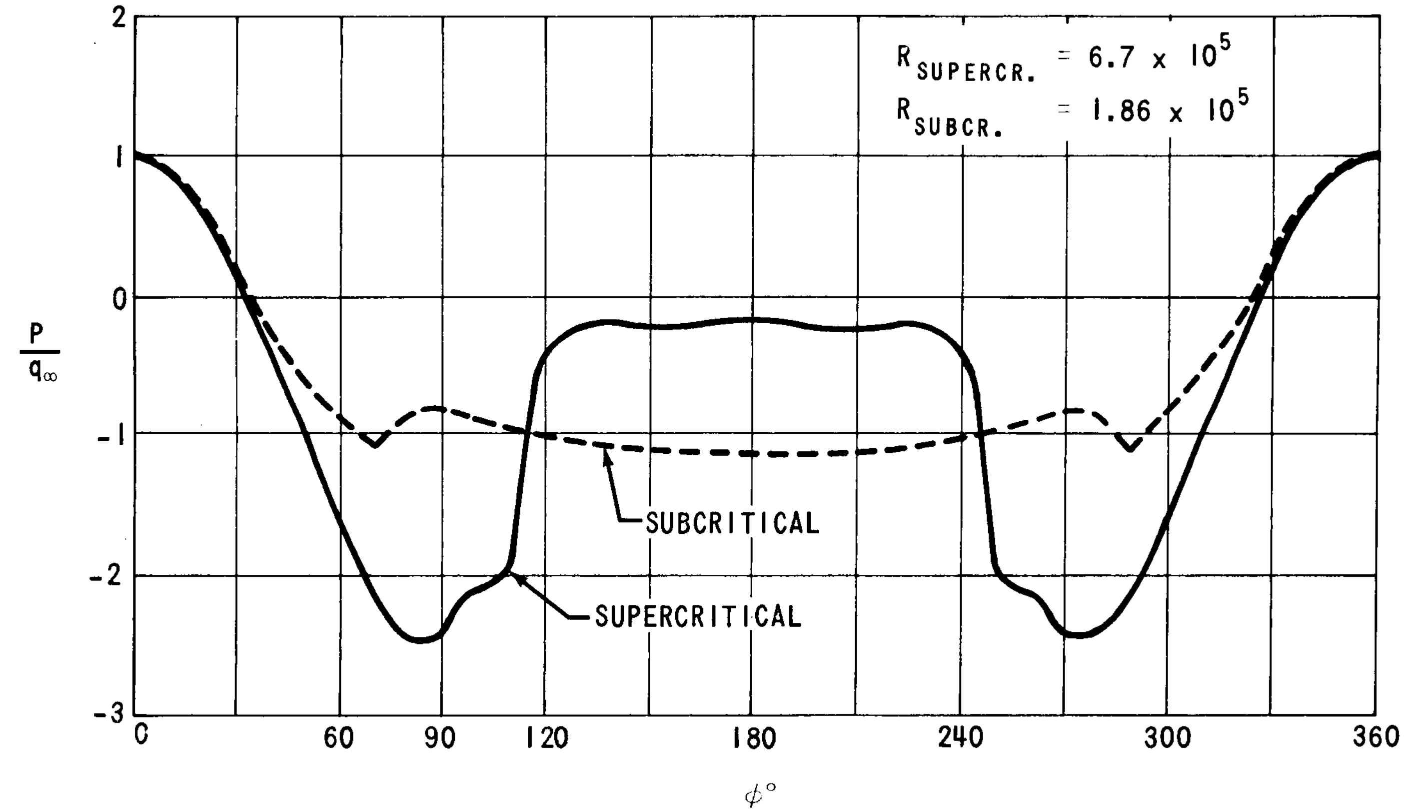

FIGURE 5 - PRESSURE DISTRIBUTION AROUND A CIRCULAR CYLINDER

TABLE I - PAYLOAD SUBSYSTEM

INSTRUMENT	WEIGHT (LBS.)	POWER (WATTS)	TOTAL DATA (BITS)
FACSIMILE TELEVISION CAMERA	0.8	4	9×10^6
SYNCHRONOUS ILLUMINATOR	---	5	---
TELEVISION CAMERA TURNTABLE	2	---	---
ANEMOMETER	1-1/2	1/2	100,000
STATIC PRESSURE TRANSDUCERS	1/4	1/10	1,000
THERMOCOUPLES	1	---	1,000
IMPACT ACCELEROMETER	1/4	1/10	200
INCLINOMETER	1/4	1/10	1,000
SOIL PENETROMETERS	2	---	---
TOTAL	~ 8	~ 10	$\sim 10^7$

TABLE II - LANDING CAPSULE SUBSYSTEMS

SUBSYSTEM	WEIGHT (LBS.)	POWER (WATTS)
PAYLOAD	8	10
DATA HANDLING	3	3
COMMUNICATIONS	7	42
POWER	8	
STRUCTURE	54	
TEMPERATURE CONTROL	9	
STABILIZING LEGS AND FINS	41	
IMPACT LIMITER	20	
TOTAL	150	55

TABLE III - HEAT LOAD ON THERMAL CONTROL SUBSYSTEM

INSULATION		18 BTU/HR
LEAD-IN WIRES, TUBES, ETC.		20 BTU/HR
STRUCTURAL CONNECTION		30 BTU/HR
SPACERS		50 BTU/HR
RADIATION AT WINDOW		60 BTU/HR
INTERNAL HEAT GENERATION		164 BTU/HR
	TOTAL	342 BTU/HR

TABLE IV - ENTRY SPACECRAFT SUBSYSTEM

		WEIGHT (LBS.)
ENTRY SUBSYSTEMS		370
LANDING CAPSULE	150	
CAPSULE EJECTION MECHANISM	50	
AEROSHELL	170	
SPACE FLIGHT SUBSYSTEMS		286
TELECOMMUNICATIONS	40	
COMMAND AND DATA HANDLING	32	
ATTITUDE CONTROL	32	
POWER	30	
STRUCTURE	78	
ASSEMBLY AND INTEGRATION	52	
PROPULSION INERTS	22	
PROBE DRY WEIGHT		656
PROPELLANTS		124
STERILIZATION CANISTER		120
PROBE GROSS WEIGHT		900

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