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REPORT TO THE
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

WORKING GROUP ON LUNAR AND PLANETARY SURFACES

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II. 4

PRELIMINARY STUDY OF AN UNMANNED LUNAR SOFT LANDING VEHICLE

(Scientific Application)

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By

Development Operations Division
ARMY BALLISTIC MISSILE AGENCY
ARMY ORDNANCE MISSILE COMMAND
Redstone Arsenal, Alabama

1 May 1959

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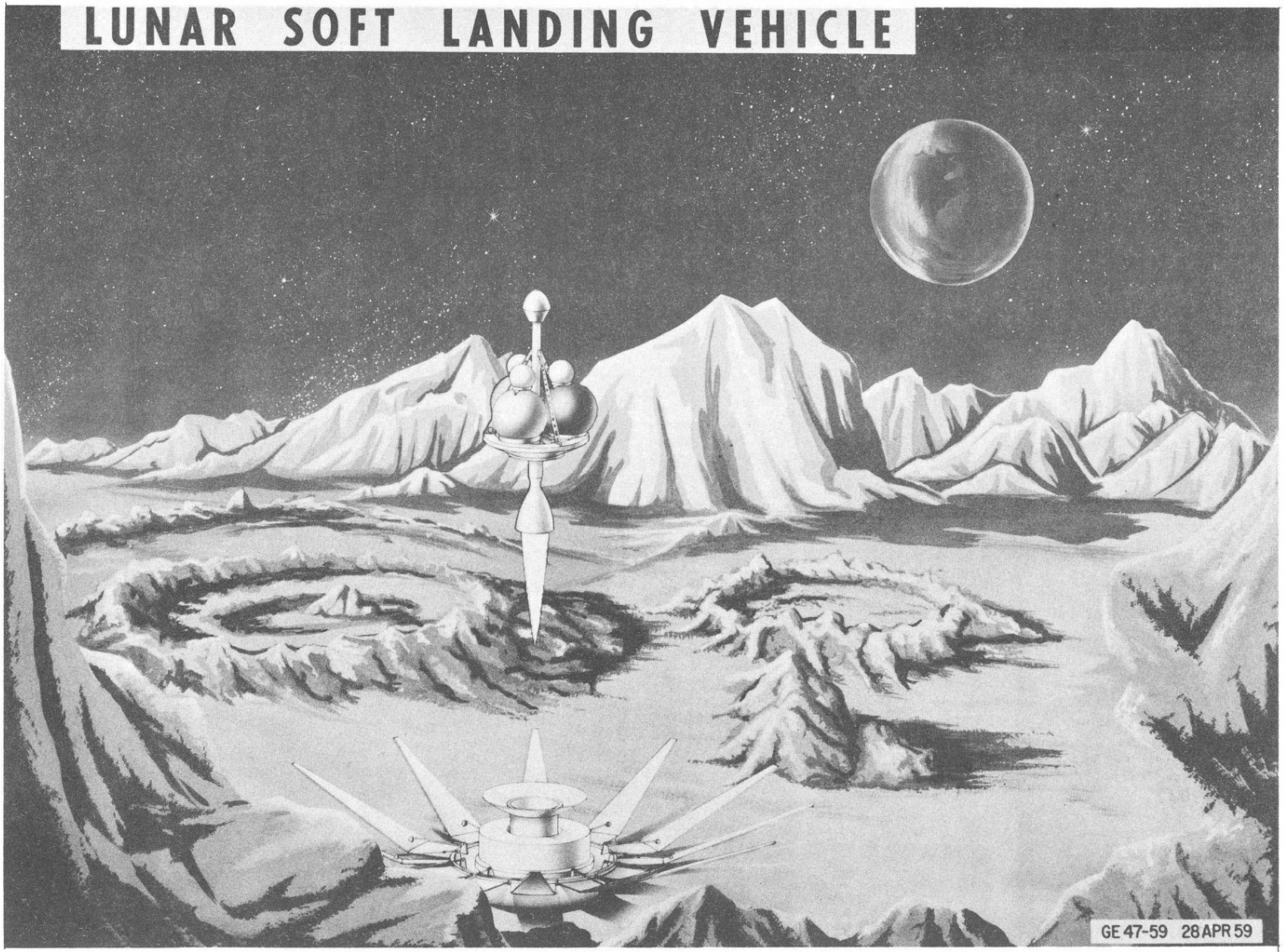
LUNAR SOFT LANDING VEHICLE
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Chapter I

INTRODUCTION

A. BACKGROUND

(U) One of the most challenging objectives of the National Space Program is the scientific exploration of the Moon. The initial step toward this objective was taken with the successful launching of Pioneer IV as a Lunar area probe on 3 March 1959.

(U) The Working Group on Lunar and Planetary Surfaces of the National Aeronautics and Space Administration is studying a continuing Lunar scientific exploration program consisting of the following sequence of steps:

1. An instrumented hard landing in which no attempt will be made to cancel impact velocity.

2. An instrumented Lunar satellite from which observation of the Lunar surface and investigation of the near-Lunar environment will be performed.

3. An instrumented rough landing in which retro-rockets will provide for partial cancellation of impact velocity resulting in a possible residual velocity of the order of 200 feet per second.

4. An instrumented soft landing in which essentially all of the impact velocity will be cancelled. A capability for return of samples from the Lunar surface to Earth may be included.

(U) Initial emphasis is being placed on the establishment of projects to accomplish the first three steps since these objectives are attainable within approximately two years.

(U) The soft landing mission is estimated to require approximately three to four years for accomplishment and is currently being given secondary emphasis. However, in recognition of the requirement for early initiation of planning toward the soft landing objective, the NASA Working Group has expressed an interest in obtaining preliminary data in this area from a number of scientific and development groups. Since the SATURN vehicle represents the earliest means for the soft landing of significantly large scientific payloads, particular interest has been shown by NASA personnel in the current thinking of the Army Ballistic Missile Agency development team of the Army Ordnance Missile Command.

B. SUMMARY

(U) This report presents the results of a preliminary study by members of the ABMA Development Operations Division of a Lunar soft landing vehicle based on SATURN as a carrier and is furnished for use by the NASA Working Group in planning their Lunar exploration program.

(S) This preliminary study indicates that it is fully feasible to achieve Lunar soft landings with gross payloads of the order of 2000 pounds within approximately four years, using the SATURN vehicle. Payloads of this size will allow for the incorporation of sophisticated guidance, control, and communications equipment plus several hundred pounds of active scientific payload.

Such payloads will provide, for the first time, a capability to perform a wide variety of Lunar surface experiments from one vehicle, thereby assuring accurate correlation of related scientific data.

(C) It also appears feasible to provide a return-to-Earth capability for small samples of Lunar material or recorded data.

(U) It is emphasized that this study is preliminary in nature and that a firm design for a soft landing vehicle will require a more detailed and intensive study, which should be undertaken at an early date.

Chapter II

RESEARCH OBJECTIVES

(U) An instrumented soft landing vehicle, which would essentially cancel out all of the impact velocity and possess a capacity for return of samples from the Lunar surface to Earth, will have as its broad objectives the following:

1. To provide information concerning the Moon's physical condition.

2. To provide criteria which indirectly will add to our knowledge of the Earth.

3. To perfect equipment and techniques for further Lunar probes.

(U) The SATURN booster combined with suitable upper stages will make possible the landing of a generous amount of scientific instrumentation on the Lunar surface. Although too much complexity in the scientific payload should be avoided, it will neither be necessary nor warranted to reduce the number of measurements to an absolute minimum. New techniques in flight mechanics and design which are established by the Lunar satellites and hard and rough landing missions will aid in the final detail design of both the landing vehicles and the associated instrumentation discussed in this report.

(U) It is realistic to plan for a series of Lunar unmanned soft landings. Repetition of flights will permit design improvements, will verify data obtained from previous flights, will extend the measuring program, and will assure that the broad objectives will be met. The first of the series of flights will be aimed primarily at establishing the techniques of Lunar approach and safe landing with relatively straight forward payloads. Since these flights must necessarily be based on the use of hardware either now in use or in later stages of development, pinpoint accuracy in landing and very soft landings are not probable.

(U) All vehicles of this series may have the same basic design. Attitude control and landing point accuracy will improve as the program evolves. It is likely that all SATURN-type vehicles will be equipped with a number of components which will constitute their standard equipment. In addition to the standard equipment, instruments will be selected for each vehicle in such a way that a comprehensive scientific program results. In this selection, particular attention will be paid to the compatibility of experiments and to the potential benefit each experiment can draw from the preceeding flights.

(U) Each landing vehicle of the SATURN-type would be equipped with certain standard components such as the following:

Antenna
Transmitter
Command Receiver
Programmer
Tape Recorder
TV System
Temperature Protection System
Power Supply

(U) If feasible, the payload package should have the capability of assuming a pre-determined direction after landing. This would greatly facilitate most of the experiments, and it would assure the proper functioning of the transmission system. The sensor for this motion, and possibly even the prime moving force, could be taken from a pendulum.

(U) In the latter flights, complex payloads and special experiments can probably be placed with almost spot landing accuracy at desired locations because of increases in overall system reliability.

Chapter III
CARRIER VEHICLE

A. CONFIGURATION

(S) The carrier vehicle used as the basis for this study is made up of a SATURN booster equipped with a modified ICBM vehicle as a second stage and the CENTAUR vehicle as the third or injection stage.

(S) The SATURN booster is being developed by the Army Ordnance Missile Command for the Advanced Research Projects Agency under ARPA Order 14-59. Its propulsion system consists of a cluster of 8 JUPITER-type engines of 188,000 pounds thrust each, providing a total booster thrust of 1.5 million pounds.

(S) The AOMC (ABMA) has recently completed at the request of ARPA a SATURN systems study as a basis for the selection of upper stages to provide an integrated multistage vehicle system. While a final decision on the upper stage configuration has not yet been made by ARPA, indications are that the second stage will be either a modified ATLAS of modified TITAN with 200 to 400 thousand pounds thrust and that the third stage will be a CENTAUR package of 30,000 pounds thrust. The general configuration of the 3-stage SATURN vehicle is shown in Figure 1.

B. PERFORMANCE

(S) Detailed weights, dimensions, and performance data for the SATURN vehicle cannot be firmly established until a decision on upper stages is made by ARPA. However, a set of characteristic data on the 3-stage vehicle were assumed for this preliminary study and are considered representative of the performance capability which can be achieved. These detailed weight and performance assumptions are contained in Appendix A.

(S) The salient SATURN performance parameter of interest in the study of Lunar landing vehicles is the cutoff weight of the 3rd stage at its escape-velocity injection into the Earth-to-Moon trajectory. This on-the-way weight of approximately 14,400 pounds includes that of the 3rd stage propulsion system and tankage. It is also assumed that the SATURN inertial guidance system is carried on top of the 3rd stage and is included in the on-the-way weight. This will allow for accurate injection phase guidance through 3rd stage cutoff and, as discussed in Chapter IV, will provide a basis for mid-course guidance corrections. A propellant reserve is also assumed so that the 3rd stage CENTAUR propulsion system may be re-ignited for the performance of mid-course path corrections. A summary of the weights involved in the 3rd stage at injection cutoff are presented below. A more detailed breakout is contained in Appendix A.

Component	Weight (lb.)
Guidance System and Instrument Compartment	2000
Engine	1127
Tankage and residuals	1588
Propellant reserve (for mid-course corrections)	1150
Landing Vehicle separation device	100
Landing Vehicle gross (at ignition)	8400
TOTAL (at 3rd stage injection)	14365

TYPICAL SATURN CARRIER VEHICLE

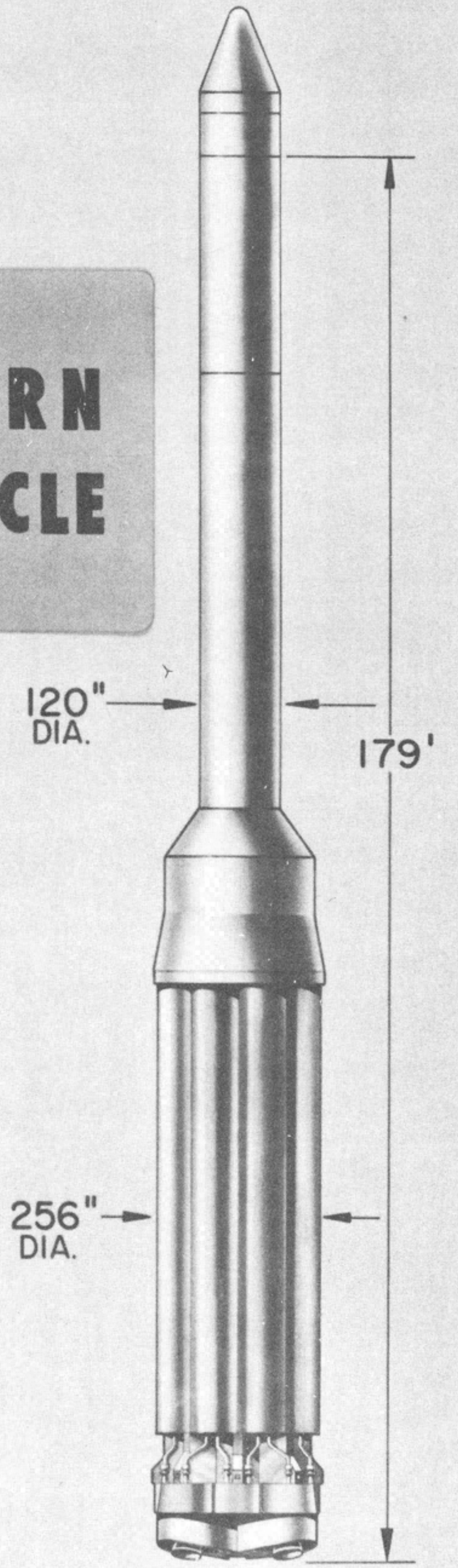

FIG. 1

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(S) The 100 pound separation device will be used to separate the final landing vehicle (or 4th stage) from the engine, tankage, and guidance system and instrument compartment of the CENTAUR 3rd stage after mid-course corrections have been completed.

(S) It is noted that a gross weight of approximately 8400 pounds is thus available for the provision of a Lunar landing vehicle including the necessary propulsion, terminal guidance, and communications equipment required for a soft landing, plus instrumentation for the performance of scientific experiments on the Lunar surface. The landing vehicle is discussed in Chapter IV.

C. SCHEDULE

(S) The current SATURN program calls for a full scale captive test in December of 1959, followed by four flight tests during the period October 1960 to September 1961. The first two flights will be tests of the booster only and the remaining two vehicles will incorporate second stages and will have an orbital capability. It is expected that the forthcoming ARPA decision on upper stages will result in an expansion of the flight test program to provide a reliable 3-stage carrier vehicle for operational use by 1962-63.

Chapter IV

LANDING VEHICLE

A. SYSTEM OBJECTIVES

(U) The final objective of a vehicular system for lunar soft landings will be to place an instrumented scientific payload on the lunar surface with essentially zero terminal velocity. Further, the payload should be landed in a specific area (with an accuracy of perhaps better than one mile) which is favorable for accomplishment of research objectives and which has certain preferred characteristics such as a surface smoothness and tolerable initial or average temperature. A highly efficient communications system must be included to transmit observed data back to Earth for evaluation. In addition, the return of specimens and/or data via a return vehicle is a desired ultimate objective.

(S) These ultimate objectives are not fully attainable by the time the basic SATURN carrier vehicle offers a significant soft landing capability, for reasons of technological state of the art and system reliability. It is believed therefore that the objectives for initial soft landing attempts must be simplified considerably. For example, the objective of the first vehicle may well be limited to a landing anywhere on the Moon's surface with a terminal velocity of the order of 20 meters per second. No return capability for lunar surface samples would be included.

(U) Subsequent flights would incorporate more sophisticated components and techniques leading up to the accomplishment of the final objectives outlined above.

(S) It thus appears that the soft lunar landing mission can best be accomplished by the establishment of a 6 to 12-vehicle program with flights beginning in late 1962 or early 1963, to allow the ultimate objective to be attained in a series of phased steps.

B. BASIC APPROACHES

(S) Two basic approaches to the attainment of a soft landing on the Moon were considered in this preliminary study. The first of these is a direct approach in which the vehicle proceeds directly from the surface of the Earth to the surface of the Moon. This approach would involve the incorporation of a fourth propulsive stage in the 8500-pound gross payload of the SATURN third stage to provide for braking of the estimated arrival velocity (2750 meters per second) at the lunar surface.

(S) The second approach, an indirect one, involves re-ignition of the SATURN third stage engine (CENTAUR) after arrival in the lunar vicinity in order to establish the entire third stage in a satellite orbit around the Moon. From this orbiting platform, one or more landing vehicles could then be sent to the lunar surface.

(U) The direct approach offers the advantage of greater overall system simplicity and, in particular, involves less stringent terminal control problems because of the almost vertical descent.

(U) The indirect approach also has certain advantages. For example, time would be available for a thorough study, through on-satellite sensors, of lunar surface details at close range prior to release of the landing vehicles in order to select a favorable landing area. Further, the satellite proper could act as a relay station for transmission of data from the lunar surface to Earth.

(U) Consideration was also given to a Moon-to-Earth return vehicle for each of the approaches discussed above. Return of lunar surface samples or recorder scientific data for examination and evaluation on Earth would represent an invaluable step in lunar exploration.

(U) Both the direct and indirect approaches warrant further detailed study with a view toward utilizing one or both methods in a lunar soft landing program.

C. LANDING VEHICLE CONFIGURATIONS

(S) A number of landing vehicle configurations have been investigated with respect to their performance capability. The more promising of these -- three for the direct approach case and one for the indirect approach -- are described below. The basic parameter which was used for the study of these configurations is the gross payload capability of the SATURN third stage of 8400 pounds as discussed in Chapter III B. Estimated weight and propulsion data for the configurations studied are contained in Appendix B. Each of these configurations would include a terminal guidance and control system as discussed in Chapter III D.

(U) It is emphasized that there are numerous other approaches to landing vehicle design which must be evaluated before a final selection of propulsion and configuration is made. Since the earliest available propulsion system may give low performance when compared to subsequently available systems, the soft landing program may well include more than one landing vehicle configuration.

1. Direct-Approach Types

a. Landing Vehicle with 6K Storable Liquid Propulsion System

(C) Figure 2 shows a landing vehicle configuration based on the JPL N_2O_4/N_2H_4 engine of 6000 pounds thrust. The payload is doughnut-shaped (Figure 3) and is located to the rear of the landing vehicle surrounding the rocket engine. Shortly before landing, the propulsion system and tankage is released from the remainder of the vehicle while the engine is still burning. The payload then proceeds to an impact in the landing area. The nose section remains attached to the payload and is opened into a star-shaped pattern to serve as a crush device to brake the lateral motion of the payload at touchdown. Air bags, which are inflated prior to impact, serve as shock absorbers for the residual vertical speed of impact. The engine and tankage, relieved of the weight of the remainder of the vehicle, will rise and follow a trajectory which will carry it away from the payload. Thus the lunar surface at the payload landing area will not be contaminated by the propulsion system.

(U) This configuration requires that the entire landing vehicle be rotated 180° (in pitch or yaw) after separation from the SATURN third stage and prior to ignition of the 6K engine.

(S) An active scientific payload of approximately 583 pounds can be landed with this configuration. (Table III, Appendix B)

(U) An alternate configuration for the 6K JPL engine is shown in Figure 4. In this case the propulsion system continues to a landing and the payload is ejected prior to impact.

b. Landing Vehicle with 60K Propulsion System

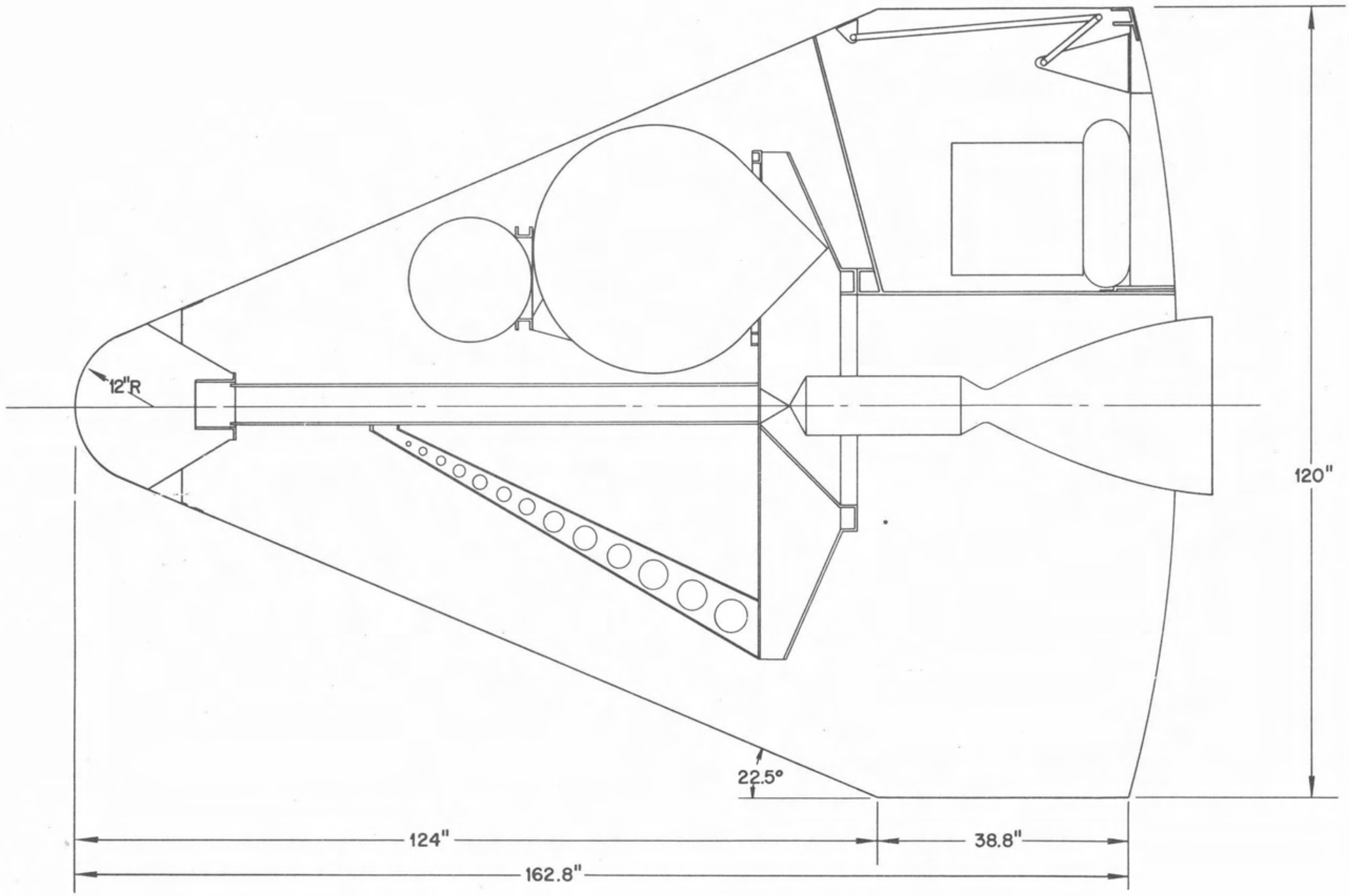
(S) Figure 5 shows a landing vehicle based on a scaled PERSHING solid propellant motor of 60,000 pounds thrust. The orientation of the motor is such that no turning of the landing vehicle after separation from the SATURN third stage is required.

(U) The rocket motor will be burned until impact. Shortly before impact, the spherical payload (Figure 6) will be ejected and carried away from the landing site to reduce fire and contamination hazards. A shock-absorbing spherical bag will be inflated to cushion the impact of the payload. The spherical shape will result in a random orientation of the payload after landing, unless means are provided for post-impact orientation.

(S) An approximate weight of 457 pounds of active payload may be landed with this configuration (Table III, Appendix B).

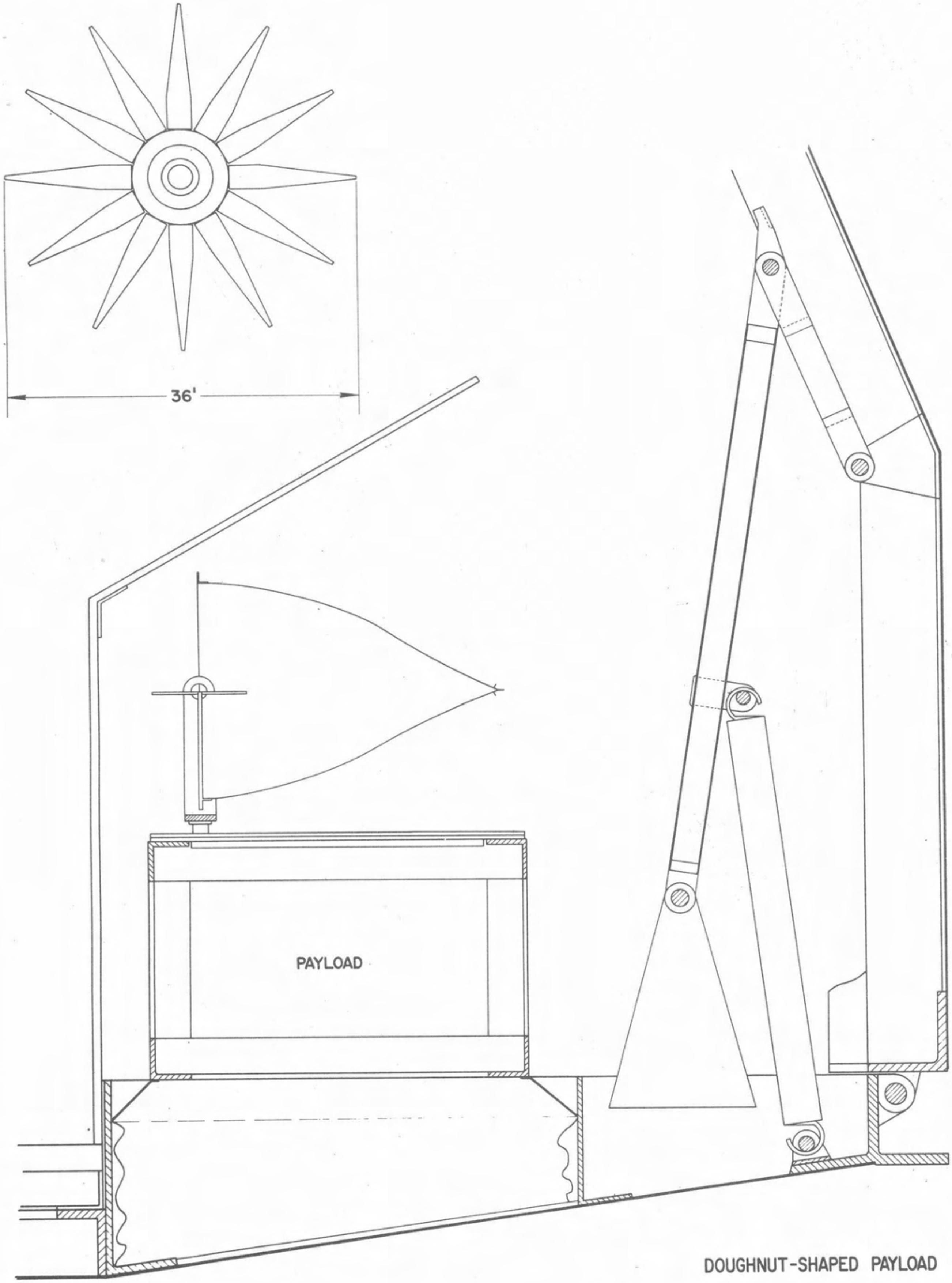
c. Landing Vehicle with 15K O_2/H_2 Propulsion System

(S) Figure 7 shows a landing vehicle based on the Pratt and Whitney O_2/H_2 rocket engine of 15,000 pounds thrust (one engine of the two-engine CENTAUR propulsion system).



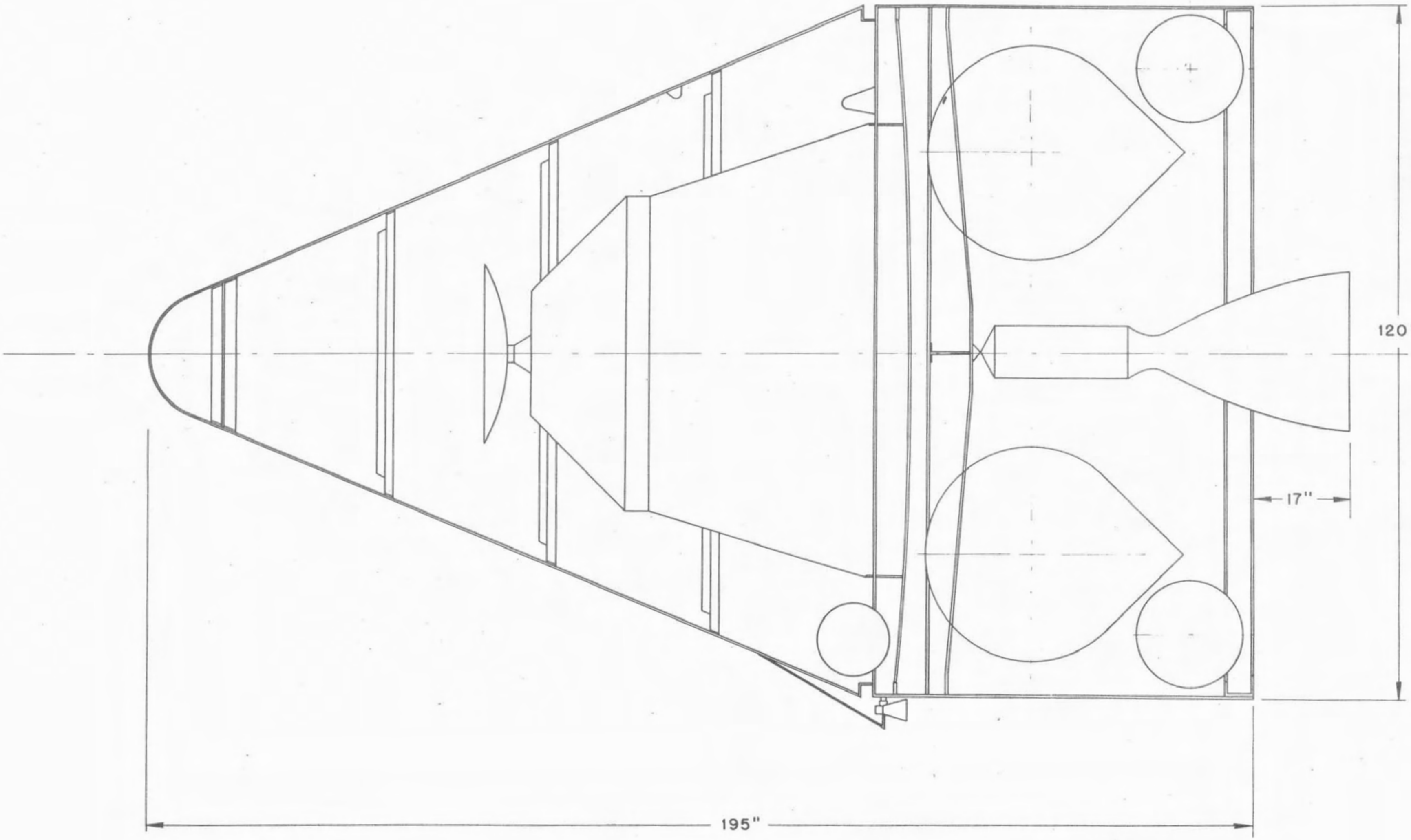
LANDING VEHICLE WITH 6K STORABLE LIQUID PROPULSION SYSTEM

FIG. 2



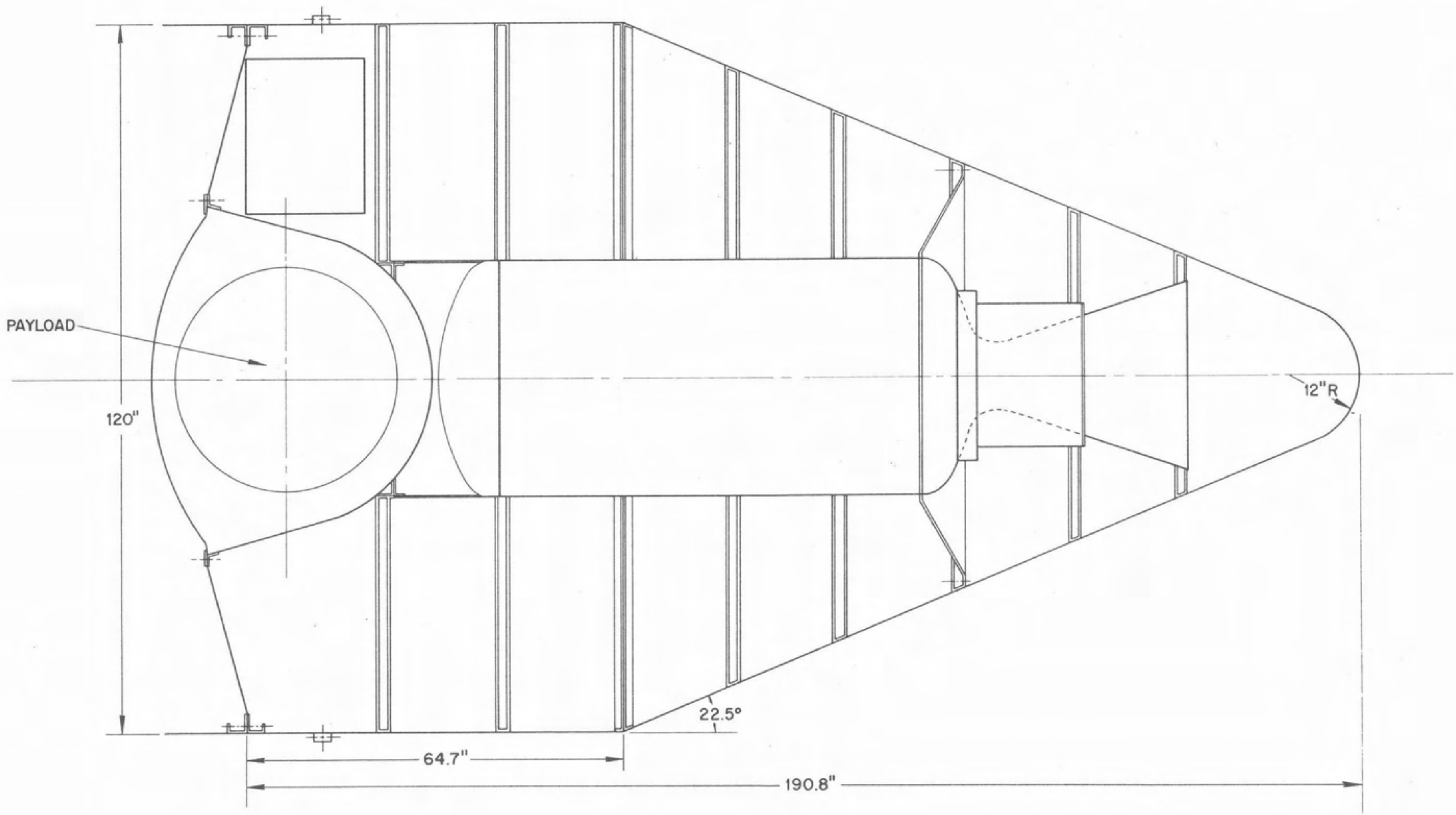
DOUGHNUT-SHAPED PAYLOAD

FIG. 3



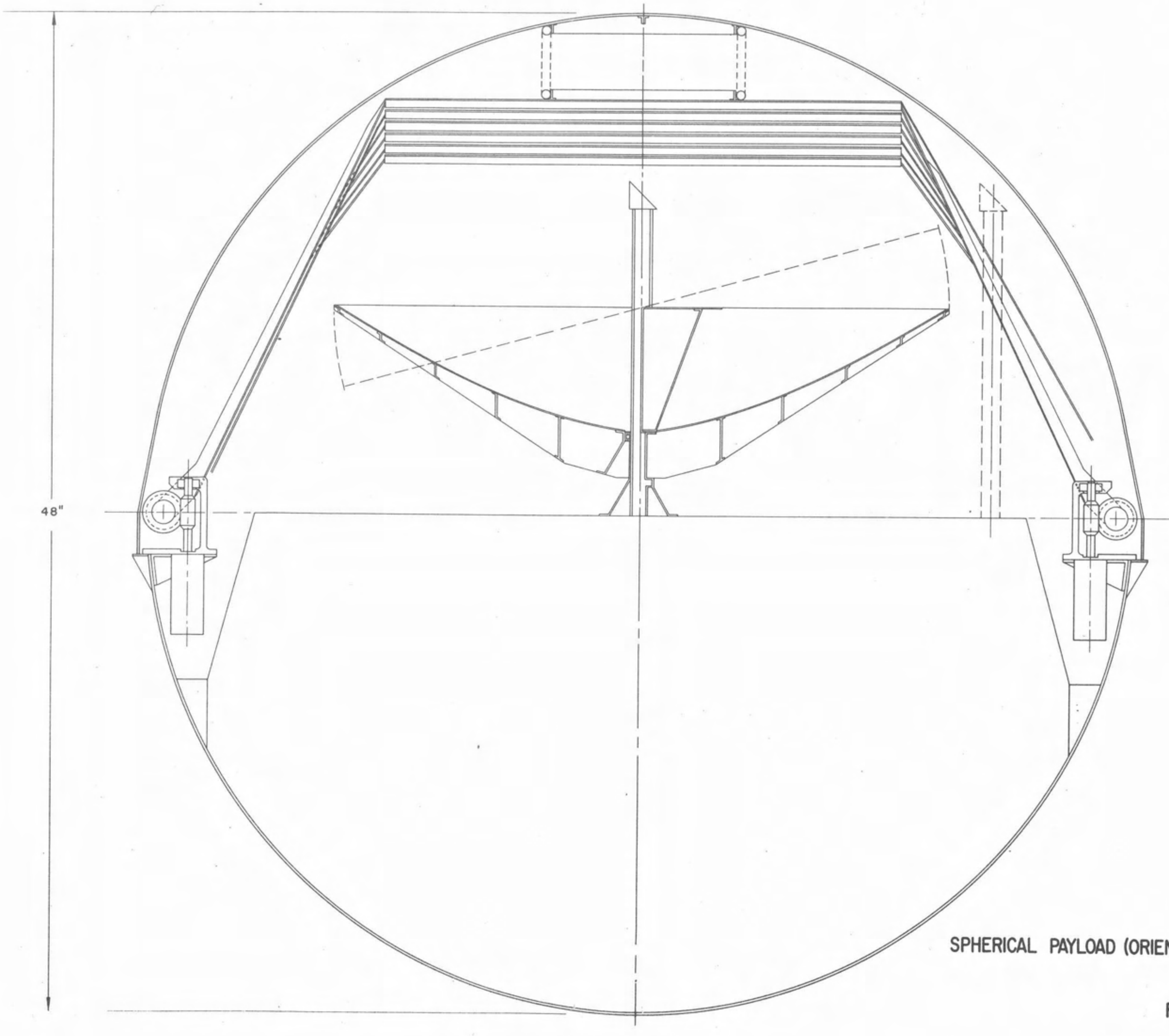
LANDING VEHICLE WITH 6K STORABLE LIQUID PROPULSION SYSTEM (ALTERNATE VERSION)

FIG. 4



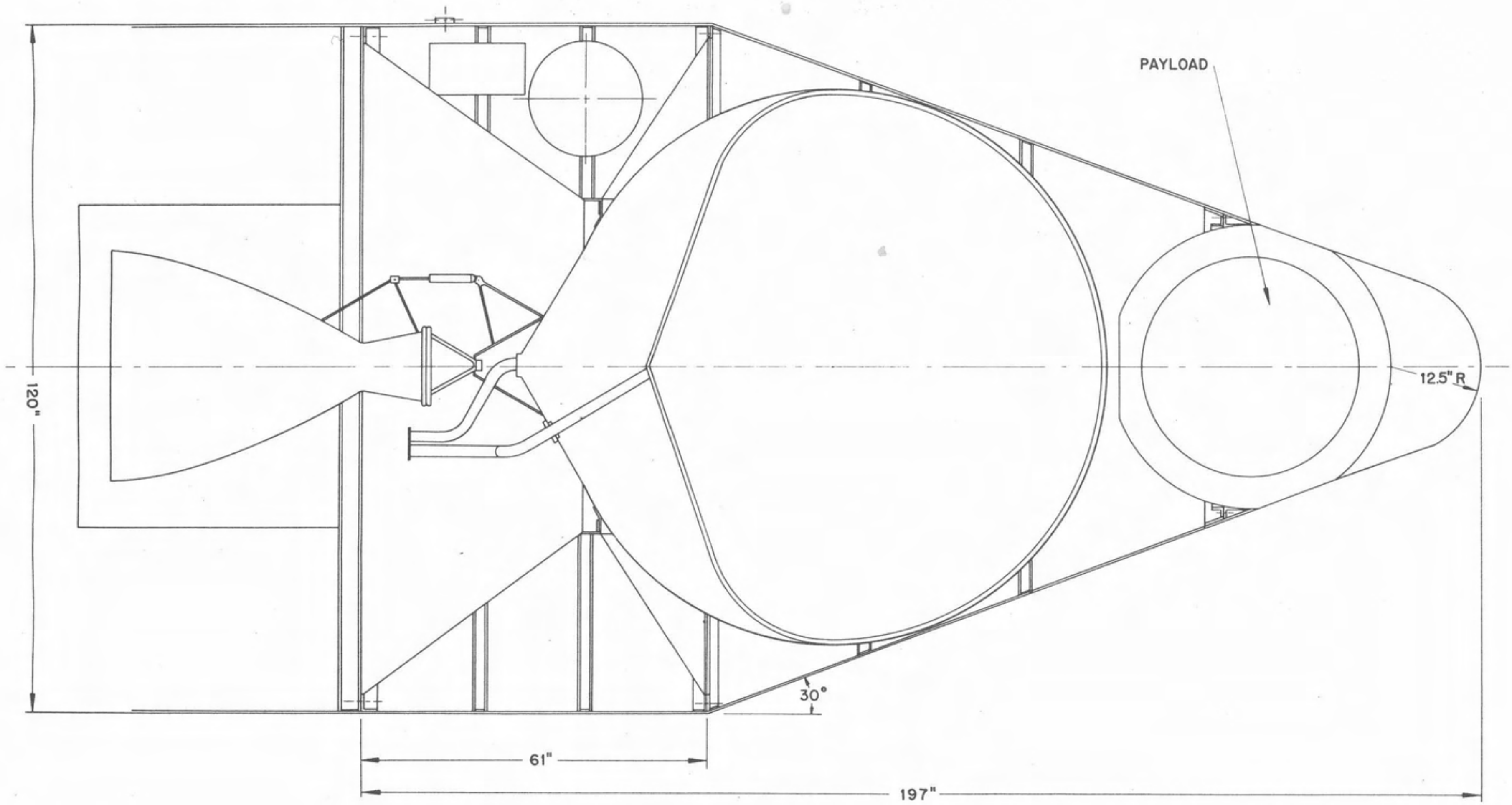
LANDING VEHICLE WITH 60K SOLID PROPULSION SYSTEM

FIG. 5



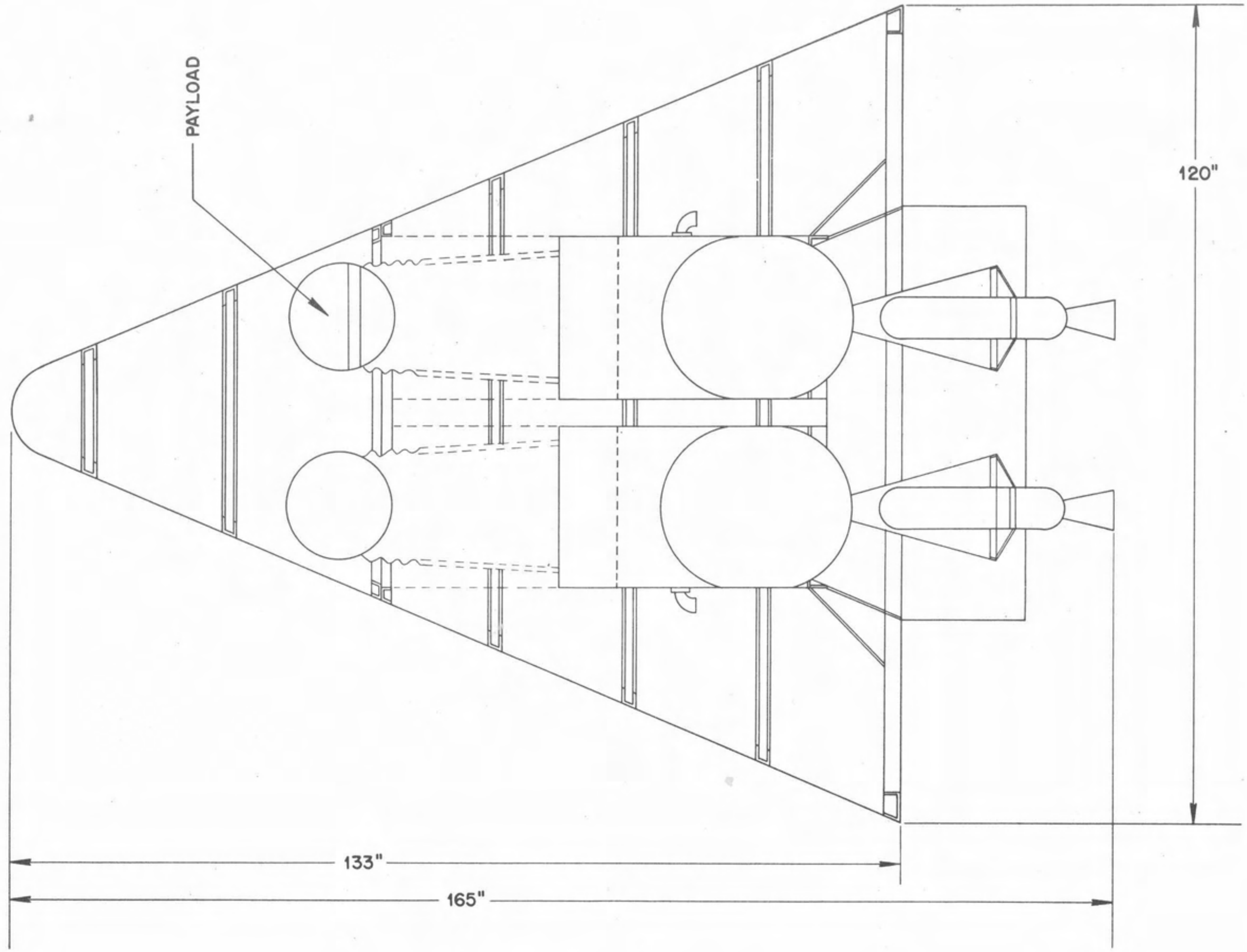
SPHERICAL PAYLOAD (ORIENTED)

FIG. 6



LANDING VEHICLE WITH 15K O_2/H_2 PROPULSION SYSTEM

FIG. 7



LUNAR SATELLITE WITH TWO LANDING VEHICLES

FIG. 8

(U) The vehicle configuration and operational sequence is essentially the same as for the solid propellant version discussed in b. on the preceeding page.

(S) While this vehicle is highly advantageous from the performance standpoint (approximately 1227 pounds of active payload landed on the Moon), it is expected to require more development time than versions a. and b. on the preceeding pages.

(C) As shown in Table IV of Appendix B, this configuration has sufficient payload capability to permit the incorporation of a return vehicle to deliver approximately one pound of material back to Earth.

2. Indirect Approach Type

(S) Figure 8 shows a lunar satellite which will be placed in orbit by the third stage of the SATURN vehicle. As shown in Tables IIa through III of Appendix B, this approach will allow the retention of up to several hundred pounds of active payload on the satellite proper and the landing of one, two or three small landing vehicles launched from the satellite. Each landing vehicle would incorporate a solid propellant kick motor to decelerate the vehicle to below orbital speed, and a larger solid propellant motor for final braking and landing.

3. Return Vehicle

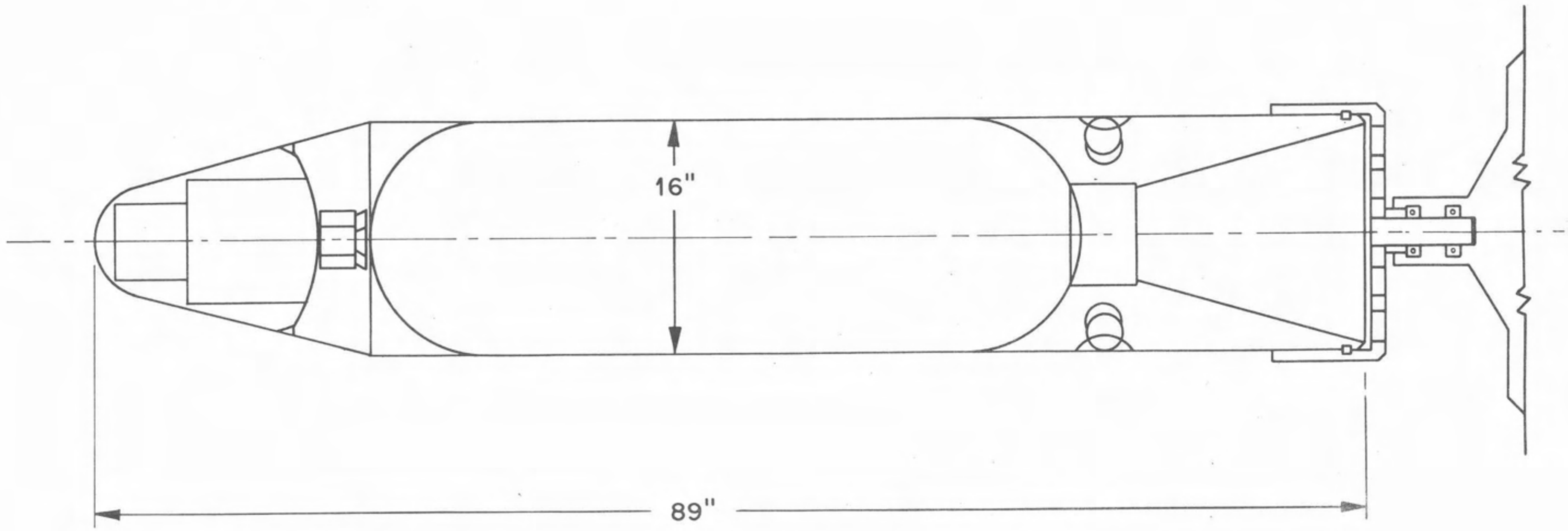
(C) Figure 9 shows a return vehicle which can be included to provide for the return of approximately one pound of material from the lunar satellite or lunar surface (configuration C only) to Earth. It would have a solid propellant rocket motor with launcher and ignition device, vernier correction motors, Earth-atmosphere re-entry heat protection and recovery equipment. Tables IVa and b of Appendix B show weight and performance data on the return vehicle when launched from the lunar surface and lunar satellite, respectively.

D. GUIDANCE AND CONTROL

(U) Regardless of the basic approach used or the landing vehicle configuration chosen, the guidance and control problem for a soft lunar landing vehicle is essentially the same. This problem, and means for its solution, are discussed below in three major phases -- initial injection guidance, mid-course guidance, and terminal guidance. Although the three phases are discussed separately, some of the actual guidance and control equipment may be used in more than one phase.

1. Injection Guidance

(S) For the powered ascent up to injection into the free-flight trajectory to the Moon a SATURN-ICBM-CENTAUR combination will be used to establish the correct velocity vector at third stage cutoff. The design values quoted in this report are based on firing due east from AMR taking full advantage of Earth rotation. Cutoff altitude is assumed to be low so as to reduce gravity losses. The injection angle is assumed to be close to 90° for the same reason. Local escape speed may be assumed as initial velocity. The latter would result in about 51 hours flight time to the Moon.



RETURN VEHICLE

FIG. 9

(S) It is assumed that no cutoff computations will be made for the first and second stages and that fuel depletion burn-out will be employed (allowance for mixture-ratio deviations is assumed). Inertial guidance hardware will be used to achieve computation of an accurate cutoff condition for the CENTAUR third stage. Slant-altitude, cross-range, and slant-range guidance will be employed for cutoff. Corrections in slant-altitude guidance during third stage burning will be made as a result of disturbances during first and second stage burning. The slant-altitude correction will reduce the injection angle error. The slant-range velocity presetting will be altered so that the velocity at injection will correspond to the injection angle and missile position for that particular flight even though the injection angle and position may be different from the standard. The necessary computation will be made by missile-borne equipment solving the cutoff equation.

(S) The all-inertial guidance scheme will employ the JUPITER-tested ST-90 stabilized platform without modification. The three perpendicularly mounted accelerometers measure the cross-range acceleration perpendicular to the flight plane, and the slant-range and slant-altitude accelerations in the flight plane.

2. <u>Mid-Course Guidance</u>

(C) The duration of the flight Earth-to-Moon is assumed to be 51 hours. During this time some controls in the vehicle must work, e.g. temperature control of propellants, whether they are liquid or solid, and in some cases attitude control with respect to the sun. In this phase -- the mid-course -- between injection and initiation of terminal guidance, corrections probably will have to be applied to ease the stringent injection accuracy and to facilitate close enough setting for the spot landing approach. On the other hand, feasibility studies have been undertaken at ABMA which indicate that the inertial injection guidance can be refined so that even without mid-course correction the lateral displacement at the instant of activating descent control can be kept to less than one Moon radius. It will be worthwhile to further investigate this possibility for elimination of the mid-course guidance phase. In any event, only minor mid-course corrections will be required.

(C) If mid-course guidance proves to be necessary, Earth bound radar stations or a missile-borne optical tracker will be used for lateral angle or position control. The stabilized platform in the missile will be operating with sufficient freedom from drift so that it may be used for attitude reference. Re-ignition of the CENTAUR third stage or auxiliary jets will provide the necessary corrective forces.

(C) Three, or preferably four, tracking stations about 4000 miles apart would be capable of determining the missile velocity magnitude to within $\pm$9m/s and the velocity angle to within $\pm$.5 degrees at a range of 100,000 miles. Ground equipment will compare these measured values with stored ideal trajectory values. Because of the length of time available, smoothing of the data can be optimized.

(U) Preference is generally given to a ground-based guidance system because of the lower weight which the missile has to carry. This would be compensated to some extent if the terminal guidance and its sensors could also be used for the sensing of mid-course errors. Work has been going on to evaluate missile-borne mid-course tracking systems with a view toward weight reduction. An advantage of a missile-borne tracker is that it provides a means for monitoring the attitude reference information from the stabilized platform.

(C) In the case of an Earth located tracking network, mid-course guidance would be effected early during flight when platform drift is still small. Following such flight path correction no control over platform drift would be considered until initiation of the terminal phase. However, it will be necessary to control the missile in attitude to keep it within the angular freedom of the stabilized platform. This requirement imposes no additional problem since such control is required prior to ignition of the terminal stage engine anyway. An attitude control system for this purpose has been studied and is discussed in Appendix C. Conventional air jet control as used on current ABMA missiles would be completely adequate, weight requirements being quite reasonable. Attitude oscillation periods in the order of 20 to 30 minutes can be attained with on-off thrust levels of only five pounds, as simulation studies show.

(U) As discussed in III C above, it may be necessary to effect a 180-degree pitch or yaw turn of the landing vehicle prior to ignition of the terminal propulsion stage. This can be easily achieved with the same attitude control actuators.

3. Terminal Guidance

(U) The combined effect of initial and mid-course (if required) guidance will be the delivery of the landing vehicle to the vicinity of the Moon with sufficient accuracy to hit the Moon in a free fall. It is then the task of the terminal guidance system to control the deceleration of the vehicle to either (a) brake the approximately 2750 m/sec. vertical free-fall speed to zero at the lunar surface, while exercising fine control over lateral speed and displacement as well (direct approach), or (b) place the vehicle in a satellite orbit about the Moon (indirect approach), after which a landing vehicle or vehicles, containing their own terminal guidance systems, are dispatched to the lunar surface. In the latter case, the terminal guidance requirements of the final landing vehicle will be essentially the same as for the direct approach except that lesser vertical and greater lateral velocity components must be cancelled.

(C) In general, the most appropriate terminal guidance system appears to be one of the collision or proportional navigation type, in which the basic guidance intelligence is derived from the angular velocity of the line of sight between vehicle and target. Such a system will require as its major elements an inexpensive form of inertial reference system and a sensory element which will maintain the line of sight to the target or landing area. The complexity of this sensory element and thus the entire terminal guidance system is determined by the nature of the "target".

(C) If the target or landing area is chosen to be the entire lunar disc, the guidance task is relatively simple. If, however, as will ultimately be the case, a landing in a specific preselected area on the Moon is desired, the problem is considerably more complex.

(C) The solution to the specified-area landing problem may be found in the use of map matching techniques now under development for the less exacting requirements of surface-to-surface missiles. Such map matching could be either automatic or controlled by an observer on Earth receiving TV pictures of the area viewed by a camera on the landing vehicle. A technique which appears promising is that of altitude-triggered area matching, in which a discrete number of pictures, representing views of the desired landing area, are carried on board the vehicle. As each reference altitude is reached, the guidance system compares the actual area viewed by its optics with the desired view recorded on the reference picture and effects a control maneuver as required. A possible solution of this type, making use of an observer on Earth, is described in Appendix D. Further study of this technique is required in order to determine the number of such monitoring steps required and to investigate means of providing the "close-look" reference pictures of lunar areas.

(U) Another parameter which effects the complexity of the terminal guidance system is the degree of softness desired for the landing. Ideally, both the vertical and lateral differential speeds should be zero at touchdown. Although the vertical component can be controlled through a radar-altimeter type distance measurement, the precise control of the lateral component is a more serious problem because of the difficulties of measuring lateral velocity. A considerable simplification of the guidance system is possible if small residual speeds of the order of 20 meters per second are allowed.

(U) Thus, while it is necessary to keep the ultimate objective in mind -- that of zero velocity landing in a pre-selected spot -- it appears most advantageous to begin the soft landing program with a basically simple system which can be refined by addition of equipment as the project progresses.

(U) One relatively simple system has been investigated which would permit landing in the general area of the center of the illuminated (visible or infrared) lunar disc with an impact speed of 20 meters per second or less. This system is described below.

(C) The terminal guidance will begin at an altitude somewhat above 400 kilometers from the lunar surface. A radar altimeter, energized by a signal from a tracking station, will supply a signal to initiate the terminal guidance phase.

(C) At approximately 430 kilometers, the altimeter will cause the missile attitude to become closely supervised by the stabilized platform. Thus, the longitudinal axis of the missile will be directed approximately along the vertical to the lunar surface. The velocity at this point will be in the neighborhood of 2800 meters per second.

(C) As soon as the attitude is established by the platform, a shroud covering a horizon seeker will be blown off, permitting the instrument to become operative. The horizon seeker will operate from the total infrared radiation from the lunar surface and will position the missile along the vertical more accurately than the platform. Three optical scanners mounted 120 degrees apart and positioned at an angle of 45 degrees from the lunar end of the missile axis will scan the lunar surface.

(C) When the horizon seeker has had time to stabilize, attitude control will be transferred from the ST-90 to the horizon seeker. After the horizon seeker has assumed control of the missile, the stabilized platform will be separated and permitted to fall to the lunar surface.

(C) Shortly after separation of the stabilized platform, (at about 340km altitude) the engine (assumed here to be the 6K JPL engine) will be ignited to begin reducing the missile velocity. After a period of initial deceleration, and when the missile has descended to an altitude of approximately 120 kilometers, the radar altimeter, being oriented along the local vertical (as established by the horizon seeker), will provide information on radial distance and radial speed. The three optical scanners will sense sufficient lunar surface details so that the lateral component of velocity can be measured. The thrust vector will then be under the control of the lateral velocity sensor, the radar altimeter, and the horizon seeker. The thrust will be directed so as to reduce the velocity vector to zero at an altitude of approximately 10 kilometers. At this time the engine will be cut off and the horizon seeker will become inoperative. However, a missile mounted gyroscope will provide an attitude reference for the remainder of the descent.

(C) The missile will free-fall from approximately 10 kilometers until a velocity is attained such that the engine will ignite and reduce the velocity to zero at less than 100 meters above the lunar surface. This will insure an impact velocity less than 20 meters per second.

(C) The total duration of the terminal phase will be approximately 300 seconds. This can be reduced if an engine (e.g. two 6K engines) larger than 6K JPL engine is used for the landing; also, active payload will be increased.

E. POSSIBLE EXPERIMENTS AND PAYLOAD SENSORS

(U) Scientific experiments recommended by the NASA Working Group on Lunar and Planetary Surfaces fall into two categories. The first refers to effects which should be measured over as long a period of time as possible. Instruments used in this category of experiments require a solar power supply, with storage batteries to carry on during the long lunar night. Measurements in this group are:

1. Radiation

a. Cosmic radiation study using the Moon as a shield between the vehicle and the Sun and with the vehicle in sunlight.

b. Incoming radiation of lower energy.

2. Meteorite Impact

a. Erosion by micrometeorites of a strip gauge mounted on the vehicle.

b. Impact on the vehicle measured by internal detectors, and seismic detectors to record hits in the vicinity of the vehicle.

c. Electrically insulated laminated strip mounted on the vehicle to study penetration of micrometeorites.

3. Temperature

a. Surface temperature in full sunlight and in the shade of the vehicle.

b. Vehicle component temperature.

c. Thermal conductivity of surface material.

d. Sub-surface temperatures.

4. Detection of Moon tremors over long periods with recording seismic equipment to determine if the Moon is quiescent or whether its heat engine is operating.

5. Short and long term variations in the magnetic field. If it is determined that a magnetic field exists, and whether or not the Moon has a core by seismic methods, data may be obtained to clarify the relation of the field with a fluid core.

6. Measurements of tidal effects caused by the earth. Indirect evidence is hinted through changing fault patterns over the Moon's surface.

7. Simultaneous radio transmission at the proper frequencies to Earth ground stations for ionospheric study.

(U) The second category of experiments contains measurements which can be obtained with success over short periods of time. Instruments used for these measurements will be powered by short life batteries.

(U) These measurements are:

1. <u>Measurements Relating to Landing Technique</u>

a. Deceleration and impact rate of initial soft landing vehicles.

b. Hardness of surface at landing site.

c. Photographs of Moon during approach.

2. High Resolution TV Scanning of Surroundings and Near Surface

a. Study of surface forms found in the landing area.

b. Determine roughness of landing surface and local change in character of the surface materials.

c. Aid in the identification of the immediate area of sampling.

3. Geophysical Investigation of the Moon's Surface and Sub-Surface

a. Change in the character of lunar material from the surface to the sub-surface with seismic apparatus using ejected detectors.

b. Radiation methods to investigate physical characteristics and changes in sub-surface material.

c. Conductivity of lunar surface and sub-surface material of resistivity methods.

d. Occurence and concentration of local magnetic materials by magnetic separator working in conjunction with sampling apparatus.

e. Thermal gradient of lunar sub-surface to the surface under conditions of sunlight and dark.

4. Analysis of Surface and Sub-Surface Material

a. Determine elements present in powdered material by x-ray fluorescence analysis.

b. Determine elements present by spark-gap spectrographic analysis.

c. Selected chemical test on crushed or powdered lunar material detectable by instrument analysis.

d. Neutron absorption measurement to determine hydrogen content in sub-surface material.

5. Analysis of Atmosphere (if detectable)

a. Attempt to gauge pressure with modified "Omegatron" high vacuum gauge.

b. Composition of near surface atmospheric particles if measurement "A" is successful.

(U) The two groups of measurements indicate that two separate sources of power supply may be required. One power source will consist of primary batteries and works only as long as the payload container can be kept at a reasonably high temperature, (i.e., the first lunar day after landing). The second type is made up of solar cells and storage batteries. Stored energy from the batteries charged during the lunar day will provide power for long duration measurements, the transmitter, and heating elements which serve to keep working compartments from freezing. The best obtainable temperature insulation is a necessity for long duration measurements.

(U) A feature of this lunar landing vehicle is the probability that some data will be obtained from the landing even if it is not completely successful. This is made possible by allowing components of the scientific payload to operate independently possibly to the extent of providing redundant transmission and antenna equipment.

F. COMMUNICATIONS

(U) An important requirement for the successful accomplishment of a lunar soft landing and the return of scientific data therefrom is a communications system. The communications system required for such a mission can be provided on a time scale which is compatible with the expected carrier and landing vehicle schedules and would be composed of both Earth and vehicle based equipment to perform the following major functions.

1. Telemetry of vehicle performance parameters throughout the flight and of scientific data from payload sensors after arrival at the Moon.

2. Tracking of the vehicle from launch to lunar arrival.

3. Command control of vehicle as required for mid-course and terminal guidance.

(U) A discussion of communications problems and possible methods for their solution is contained in Appendix E.

Appendix A
SATURN VEHICLE
Estimated Weight and Propulsion Data

STAGE	I	II	III
Type	SATURN	ICBM	CENTAUR
Propellant	LOX/RP I	LOX/RP I	O_2/H_2
Thrust (lb)	1,504,000	400,000	30,000
I_{sp}(sec)	256.7	308	412
Flow Rate (lb/sec)	5,858.98	1,298.70	72.82
Exit Area $(in)^2$	8 x 1,630	-	-
Missile Diameter (in)	256"	120"	120"
$W_{11,15}$, Payload, lb.	297,200	39,215	8,500*
W_{16}, Guid. Compartment, lb.	500		500
W_2, Guid. & Control, lb.	1,100	500	1,500
W_3, Fuselage, lb.	45,000	5,378	1,178
W_4, Propulsion, lb.	22,400	4,967	1,127
W_5, Recovery Eq., lb.	6,000	-	-
W_6, Trapped Prop., lb.	15,500	1,740	150
W_7, Usable Residuals, lb.	7,500	2,400	260
Prop. Reserve(Midcourse)	-	-	1,150
W_8, Prop. Consumption, lb.	750,000	243,000	24,850
W_s, 16, Structure Wt., lb.	75,000	10,845	4,305
W_n, 16, Struc, Net Wt., lb.	98,000	14,985	5,865
W_a, 16, Stage Wt., lb.	848,000	257,985	30,715
W_o, Liftoff Wt., lb.	1,145,200	297,200	39,215
W_o, Cutoff Wt., lb.	395,200	54,200	14,365

*Estimated (includes 100# Separation device)

Appendix B
POSSIBLE LANDING VEHICLE CONFIGURATIONS
Estimated Weight and Propulsion Data

TABLE I
Direct Approach Landing Vehicles

Configuration Series		A	B	C
Engine		JPL	Solid	1 P&W
Propellant		N_2O_4/N_2H_4	Solid	O_2/H_2
I_{sp} (sec)		300	258	412
Thrust (1000 lb)		6K	50K	15K
Burning Time (sec)		278	24	121
Deceleration (g)	min.	0.71	5.95	1.78
	max.	2.11	18.51	3.75
Slant Firing Range (Km)		340	27	150
Weights (lb)				
Ignition		8400	8400	8400
Cutoff		2838	2701	4002
Propellants		5692	5813	4792
Fuselage		475	830	317
Engine		400	-	564
*Gross Payload		1833	1757	2727

*See TABLE III for breakdown of Gross Payloads

TABLE IIa

Lunar Satellite (Indirect Approach)

Weights (lb):

At CENTAUR Ignition	8500
Fuel Used	3327
2% Evaporation	66
Fuselage	170
Heat Insulation	100
Re-ignition equipment	50
*Gross Satellite Payload	4787

*See TABLES II b to d for further breakdown of Satellite Payload

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TABLE IIb

Breakdown of Lunar Satellite Gross Payload

Weights (lb):

Structure	400
Guidance (over and above 2000 lb in 3rd stage proper)	200
Power supply	500
Altitude control	200
Antenna	50
Transmitter	50
*Special payload	3387
	4787 lb

*For breakdown see TABLES II c and d

TABLE IIc

Breakdown of Satellite "Special Payload"

	D	E	F
Configuration Series	D	E	F
No. of Landing Vehicles	1	2	3
Weights (lb)			
On-satellite Research Instruments	87	400	37
On-satellite Comm. Package	200	200	200
Launcher for Landing Vehicles	100	187	150
*Gross weight of Landing Vehicles	3000	2 x 1300	3 x 1000
Total	3387	3387	3387

*See TABLE II d for breakdown

TABLE IId

Breakdown of Landing Vehicle Weights

Configuration D (one Landing Vehicle):	
Retro-rocket (for removal from orbit)	124 lb.
Brake rocket (for landing)	1782 lb.
*Gross Payload	1094 lb.
Total	3000 lb.
Configuration E (Each of 2 landing vehicles):	
Retro-rocket	58 lb.
Brake rocket	780 lb.
*Gross Payload	462 lb.
Total	1300 lb.
Configuration F (Each of 3 landing vehicles):	
Retro-rocket	50 lb.
Brake rocket	593 lb.
*Gross Payload	357 lb.
Total	1000 lb.

*For breakdown see TABLE III

TABLE III

Summary of Payload Capabilities

	JPL	Solid	P&W	Satell, 1 Land	2 Land	3 Land
Configuration Series	A	B	C	D	E	F
Landing Vehicle Weights (lb):						
Power Supply	100	100	150	100	2 x 50	3 x 50
Container	180	180	270	110	2 x 50	3 x 40
Term. Guidance	400	400	400	100	2 x 100	3 x 100
Altit. Control	200	200	200	100	2 x 50	3 x 40
Shock Absorbers	100	100	150	70	2 x 35	3 x 30
Eject. Syst.	50	50	80	50	2 x 15	3 x 10
Comm.	100	100	100	70	2 x 45	3 x 40
Hovering	120	170	150	30	2 x 15	3 x 10
Landing Vehicle Active Payload	583	457	1227	464	2 x 102	3 x 37
TOTAL	1833	1757	2727	1094	2 x 462	3 x 357

TABLE IV

Return Vehicle Possibilities

a. From Lunar Ground

Effective Payload	1 lb	
Structure	14	
Heat Shield	50	
Engine	50	
Recovery Gear	10	
Transmitter & Power Supply	10	
Command Receiver	5	
Vernier Connection Motors	20	
Gross Payload	160	
Fuel	446	
Fuselage	54	
Total Vehicle:	660	
Launcher, igniter, etc.	40	Remaining on Moon
Special electronics Lander	200	Remaining on Moon
Effective Payload required on Moon:	900 lb.	

Capability: only Configuration C

TABLE IV (Cont'd)

b. From Lunar Satellite

Active Payload	1 lb	
Gross Payload (Same as a above)	160 lb	
Fuel	82	
Fuselage	11	
Total Vehicle	253	
Launcher	20	Remaining in satellite
Special electronics	100	Remaining in satellite
Effective payload required in Lunar satellite	373 lb	

Capability: Configuration D and reduction of landing vehicle size

Configuration E and reduction of satellite active payload to 27 lb.

Configuration F - Not feasible to have 3 landing vehicles plus return vehicle.

Appendix C

MID-COURSE AND TERMINAL CONTROL SYSTEM FOR A SOFT LUNAR LANDING

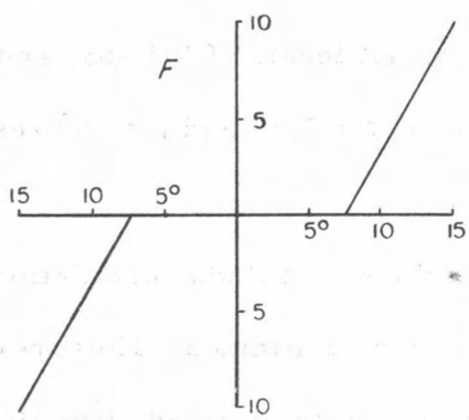
(U) For a soft Lunar landing mission it is desirable to maintain a space-fixed reference through the mid-course phase in order to properly orient the missile if guidance corrections are necessary during this phase. In case mid-course guidance is unnecessary, the space-fixed reference system is still desirable for use in the approach phase. Although such a reference cannot be used directly to indicate the direction of vertical approach to the Moon, it can greatly reduce the region which a scanning device would have to cover in locating the Lunar vertical.

(S) The body to be controlled was assumed to weigh 8500 pounds, which could be boosted by a SATURN-ICBM-CENTAUR combination. The shape was assumed to be conical, with a 120" base. From this data, the moments of inertia in pitch-yaw and roll were estimated to be 413 $m/kg/sec^2$ and 275 $m/kg/sec^2$, respectively.

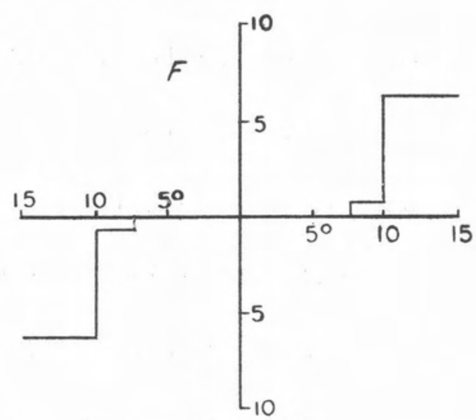
(U) Control of the vehicle attitude to contain it within the platform freedom during mid-course by conventional air jet control methods has been investigated. Under the assumptions made for this investigation, jet attitude control was found to be completely adequate, with weight requirements being quite reasonable.

(C) Since accurate attitude control during the mid-course is unnecessary, the body can be allowed to drift within limits determined by the angular freedom of the stabilized platform. Assuming a platform freedom of $\pm 15^{\circ}$, the free drifting region can be set at 7.5°, or even higher. The upper limit is chosen somewhat below the platform freedom to take care of transient overshoots of the attitude, as well as gain inaccuracies.

(C) Either a proportional nozzle system with a superimposed deadband or a 2-step on-off nozzle system can be used. The desired static control force characteristics of each pair would be as shown below:



PROPORTIONAL



ON—OFF

(C) The nozzle pairs would be located at four stations spaced 90° apart. Available proportional nozzles are subject to leakage when in the zero position. The leakage rate is approximately 30 times that of an equivalent on-off nozzle. To prevent the objectionable loss of air a solenoid valve would be used in series with the nozzle, thereby removing pressure from the nozzle except when control torques are demanded.

(C) If an on-off system is used, two force levels are necessary, since the force of the nozzle which controls the drifting region must be extremely small in order to keep the drift velocity and subsequent air consumption reasonable. The larger nozzle is necessary in order to overcome transient disturbances (e.g., at separation) and force the attitude oscillation back into the low-velocity condition. The relative triggering levels of the two nozzles must be set so that the impulse of the low-force nozzle can compensate for the quasi-steady state velocity imposed by the high force nozzle after transient disturbances are damped out. Assuming thrust levels of 0.5 lbs. and 5 lbs. the triggering levels could be set at $\phi = 7.5^{\circ}$ and $\phi = 10^{\circ}$ respectively, for the body assumed.

(C) With either of the systems mentioned, attitude oscillation periods in the order of 20 to 30 minutes can be attained. These periods have been determined by extrapolation of results obtained from simulation studies where portions of the actual control system hardware were incorporated. Specific information on weight and power estimates is shown on the following page, assuming a mid-course flight time of 50 hours.

On-Off Nozzle System

5 pound nozzles	16.0 lbs.
0.5 pound nozzles	10.5 lbs.
Regulators, lines, etc.	17.5 lbs.
Air and container	15.0 lbs.
Computer	14.0 lbs.
TOTAL	73.0 lbs.
Electrical energy required	85 watt hours

Proportional Nozzle System

Nozzles	9.0 lbs.
Regulator, lines, solenoid valves, etc.	27.0 lbs.
Air and container	10.0 lbs.
Computer	12.0 lbs.
TOTAL	58.0 lbs.
Electrical energy required	70.0 watt hours

(U) The torques available from the air nozzle system will provide rapid orientation of the vehicle to any angle which might be required upon entry into the terminal phase before firing of the retro-engine. Possible thrust misalignments of the retro-engine, however, will not allow the air nozzle system to be used for pitch-yaw attitude orientation while the decelerating thrust is being applied. Since roll torques will be quite small even during the deceleration phase, roll control can be provided by the air nozzles to impact. The additional air supply requires for the terminal roll control would weigh approximately 0.5 pounds.

(S) It is assumed that a 6K JPL engine will be used to decelerate the vehicle upon approach to the Moon. A nominal firing time of approximately 270 seconds is adequate to produce the required deceleration. This time may be extended somewhat if the engine is throttled to facilitate accurate control of the landing velocity or to provide a "hovering" capability. Therefore, a total time of 300 seconds has been assumed for the terminal phase.

(U) Orientation of the thrust vector to minimize vertical and horizontal velocities can be best obtained by direct manipulation of the retro-engine thrust vector. The thrust vector direction would be controlled by swiveling the engine, thereby providing the required orientation in pitch and yaw. Swiveling of the engine by both electrical and hydraulic means has been investigated.

(U) Electrical system would use two 60 watt DC motors driven directly from power transistors.

(C) The weights and power requirements are:

Electrical Actuators.......................6 lbs.

Computer..............................20 lbs.

TOTAL..................26 lbs.

Electrical energy required.............5 watt hours

(C) The primary fault of the hydraulic system is the leakage in the servo valve, which represents an inefficient expenditure of energy; with presently available valves, having leakage rates of 0.1 in 3 sec., the power consumption of the system is considerably more than that of the electrical system. Except for the power supply, the hydraulic system weight is approximately the same as the electrical system:

Hydraulic actuators	5 lbs.
Pump	4 lbs.
Reservoir, lines, etc.	6 lbs.
Computer	10 lbs.
TOTAL	25 lbs.
Electrical energy required	20 watt hours

(U) Considering the electrical energy requirements, the hydraulic system does not look as attractive as the electrical.

(C) Servo valves, having leakage rates of 0.02 in 3 sec., appear quite feasible; based on a proposal presently in hand, they can be developed in one year or less. Utilizing these values, a pre-charged accumulator, can be used to furnish the pressurized hydraulic fluid, thereby eliminating the pump and its power supply. The total oil volume required would be 25 cubic inches.

The weights on the following page would result:

Hydraulic actuators	5 lbs.
Accumulator	9 lbs.
Lines, solenoids, etc.	4 lbs.
Computer	10 lbs.
TOTAL	28 lbs.
Electrical energy requires	0.5 watt hours

(U) If a pump is used, instead of an accumulator with the lower leakage value, the weights will be similar to those shown for the present value-pump system. The difference would be a considerable saving in electrical energy, since the total required would be only 4 watt hours.

(C) In summary, a proportional nozzle system with a superimposed deadband is the best solution for the mid-course control requirements. In the landing phase the same nozzles would be used for control about the roll axis. Pitch and yaw can best be controlled by swiveling hardware the total weight of the mid-course and terminal system, exclusive of power supply, would be less than 100 pounds. The electrical energy required would be approximately 75 watt hours.

APPENDIX D

TERMINAL GUIDANCE BASED ON AREA MAP COMPARISON

(U) The measurement of the lateral velocity of a landing vehicle relative to the lunar surface represents a problem which must be solved before a fully automatic map-matching terminal guidance procedure can be developed.

(U) The limited knowledge which currently exists relative to the geological composition of the moon will not support an assumption that a radar picture would suffice as a basis for such velocity measurement. However, the concept described below does offer promise as an early solution to the problem.

(U) Figure A shows the landing area geometry at a given instant of time during the landing phase when the vehicle is at an altitude h above the lunar surface. Area A' B' C' D' represents a pre-selected landing area on the lunar surface for which a map has been prepared and is in the hands of an observer on earth. A larger area ABCD, which includes A' B' C' D', is assumed to be in the field of veiw of the lunar landing vehicle's optical system during its descent to the lunar surface. It is further assumed that the viewed area is illuminated by either (a) the sun or (b) artificial light from the vehicle or a flare. (In the case of artificial illumination, the optical system will use

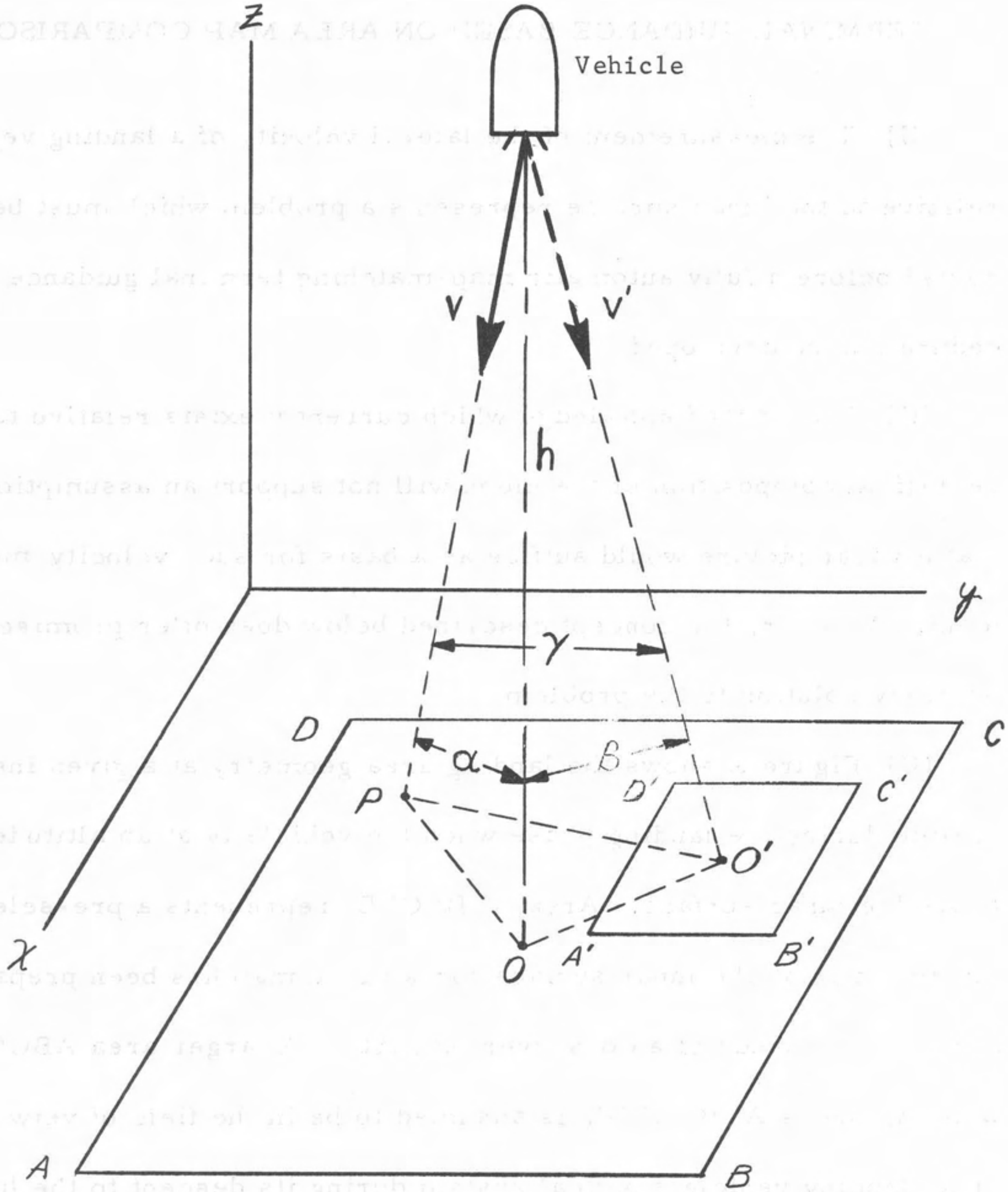


Figure A
LANDING AREA GEOMETRY

sensors of the "cat-eye" type.)

(C) The attitude control system of the vehicle will keep the optical axis aligned with the local vertical so that point 0, representing the center of the viewed area, will be at the point of intersection of the local vertical with the lunar surface. Point P represents the intersection of the vehicle's instantaneous velocity vector with the lunar surface, and is thus the point at which the vehicle will land if it continues on its present course. Point 0', however, which is the center of the Preselected landing area, is the point at which a landing is desired. The terminal guidance process must then determine the error in the present vehicle velocity vector (V) and effect corrections in velocity direction (to V') and magnitude to that point P will coincide with point 0' and so that the velocity magnitude is reduced to essentially zero at touchdown. For the guidance concept being discussed here, it is assumed that all guidance processes for the controlled reduction of velocity magnitude are performed within the vehicle, based on intelligence provided by a radar altimeter. The basic decisions regarding control of the velocity direction, however, will be made by the observer on earth.

(C) The earth-based observer is provided, via a communication link from the vehicle, with a continuing series of pictures of the viewed area (ABCD) as the vehicle descends. He is also provided with information as to the instantaneous value of altitude (h) corresponding to each picture (from the radar altimeter). From this information, together with his previously prepared map, he can reconstruct the entire landing area geometry shown in Figure A. Point 0 is the center of the picture being viewed. The coordinates of Point 0' are known since 0' is a pre-selected point. The position of point P is determined by observing the relative motion of surface details as successive pictures are viewed. Since the missile is moving toward point P, terrain features near P will appear to remain stationary on the viewing screen as the vehicle descends. All other terrain features in the viewed area, however, will appear to move away from P as the vehicle descends to lower altitudes and area ABCD becomes smaller. Thus the observer can determine the coordinates of point P by noting the coordinates of the apparent stationary point.

(C) From the above information, the observer can determine the distances PO and OO' and, knowing h, can compute the angles α, β, and γ. He is then in a position to send corrective commands to the landing vehicle for a velocity correction so that the new velocity vector V' is established.

(C) The time delay represented by the observer's computation process will, of course, result in an error in the new velocity vector. However, the process will result in a movement of P closer to O'. By successive repetition of the processes of observation, calculation, and correction during the landing phase, the observer can cause the vehicle to land in the vicinity of point O', e. g. within the preselected area A' B' C' D'.

Appendix E

COMMUNICATION SYSTEMS FOR LUNAR SOFT LANDING

A. MEASURING INSTRUMENTATION

(U) Typical measurements to be considered might include magnetic field studies; Lunar surface features, including surface hardness and penetrability, surface dust, photo reconnaissance, composition of surface material, fluorescence of surface, electrical conductivity, and thermal conductivity; seimographic studies, Lunar atmosphere studies; gravity measurements; cosmic radiation measurements; temperature measurements; meteorite and micrometeorite studies; and electron density. It is expected that any measuring program will undergo considerable revisions especially after hard landings, rough landings and Lunar satellites are successfully accomplished.

(U) Preliminary estimates of weights and power requirements for measuring instruments can be made on the basis of the 5-experiment JUNO II IGY satellite which ABMA is developing for NASA.

(U) The average weight per measurement for this satellite is close to two pounds, not including power supplies or telemetering equipment. The average power required to operate the equipment was 80 milliwatts per experiment. In a Lunar probe the complexity of some of the experiments can be expected to increase the average weight considerably. In addition the g-forces will necessitate more rugged construction. An estimate of 10 pounds per measurement seems reasonable. This would include end-organs or sensors as well as electronic adaption to the telemetering system, but would not include power supplies or telemetering. Due to the possible need for more mechanical emplacement of the end organs, the existence of electromagnetic components, and the general desire for reliability, the average power consumption may be expected to increase to an estimated 500 milliwatts per measurement.

(U) The number of experiments to be carried is estimated to be between 30 and 40. For 35 measurements the total weight might be about 350 pounds, and the total power about 18 watts. Some measurements might involve equipment unable to function in the cold environment present during the two-week period between last quarter and first quarter, so that additional electrical power may be necessary for heaters. It is hoped that techniques can be resolved so that all measurements can be time shared, none requiring an entire straight channel. Some storage system may be necessary to provide instrumentation coverage during the Lunar night.

B. TELEMETRY INSTRUMENTATION

(U) A multiplexer designed to handle 40 measurements would consume approximately 150 mw of power and the encoding would require an additional 100 mw. The space required would be approximately 24 cu in.

For transmission of telemetry only and RF power of 100 mv would suffice.

(U) The band width required for telemetering would be very low except for seismic and cosmic measurements. These probably could be reduced by a conditioning process on these two measurements. The cosmic ray counts could be integrated over a period of time; exactly what method could be used with seismic measurement is uncertain with the present state of knowledge of this measurement. At any rate it is clear that telemetering could be sampled, coded and transmitted over any practical BW required for transmission of fixed photo data.

(C) Based on these assumptions, the RF link may have the following characteristics:

1. 3 KC RF BW
2. 1 db. receiver NF
3. Transmitting antenna gain: 0 db
4. Receiving antenna gain: 46 db
   (Goldstone antenna at 1000 mc)
5. A S/N ratio and fading allowance of 12 db
6. A transmitting power of two watts

(U) This link would be capable of transmitting a frame of photo data each four minutes.

C. TRACKING

(U) For tracking the Earth-Lunar trajectory may be divided into five phases: (a) Initial Phase - from launch to injection of vehicle into its Lunar trajectory; (b) Post Injection - from injection to mid-course; (c) Mid-course - extending from approximately 80,000 to 160,000 miles; (d) Terminal Approach; (e) Landing.

(U) For the initial phase (1), the UHF dovap (UDOP) and safety devices presently used is considered most suitable. After study of various feasible systems, the optimum for the trajectory phase (b), injection through mid-course, appears to be a doppler system in which methods for range unambigiously, and for correcting ionospheric effects are incorporated. In addition to being a minimum cost system, it offers maximum growth possibilities. For the case of the Earth-Moon trajectory, an Earth based system is considered sufficient. The system employs sets of ground stations with three or more stations in each set. Each ground station is equipped with a standard frequency transmitter (1×10^{-9} or better) and a receiving system. Two phase related frequencies transmitted from one ground station, are received on the vehicle and are retransmitted at a frequency, off-set, but phase related to the received frequency. These frequencies are received at each ground station and the phase compared with that station's frequency standard. The rate of change of phase gives the algerbraic sum of velocity components as seen by the transmitting and receiving station. The receiver located at the transmitter gives the velocity component as seen by that station only. Velocity components as seen by each of the other receiving stations can be obtained by subtracting that seen by the transmitting station from the Sun as read at each of the other receiver stations. The range is obtained by timing the interval between phase coincidence between the two reference frequencies and a phase coincidence between the returned waves from the vehicle.

(U) The ground frequency standards are maintained in a known phase relation by a reference transmission from the master station which may take either the form of a phase locked low frequency transmission or by coded transmission of frequency count at over a fixed interval of time. Velocity and range information can be processed at a central point, preferably, the transmitting station. Real time data processing is not necessary but continuous data processing with a minimum lag (5 to 10 min) accurately related to time, is feasible and desirable.

(C) Baselines for the system are 4,000 to 6,000 miles. For maximum accuracy stations should be located so that important parts of the trajectory are viewed from the center of the station system at not greater than 30° from vertical with the Earth's surface. The ground system must be regarded as a short baseline system. Ranges at mid-course are of 40 to 50 times the baseline resulting in errors of range and velocity components which are tangential to the Earth's surface of approximately 40 to 50 times (at mid-course) the basic system errors.

Estimated Errors:

Basic velocity error - 0.3 meters/sec

Basic range error - 3.0 meters

Overall Errors (average)	Mid-course	At Moon
Velocity, Vertical	0.3 m/sec	0.3 m/sec
Velocity, Tangential	15 m/sec	36 m/sec
Range, Vertical	3.0 meters	3.0 m/sec
Range, Tangential	150 meters	360 m/sec

(C) Advantage may be taken of the relatively straight path of the trajectory and small change in velocity at mid-course to determine the vector velocity to a reasonably high degree of accuracy by measuring the deviation from the trajectory over a period of time. The estimated accuracy in velocity magnitude is within $\pm$ 0.6 m/sec and velocity angle within 2×10^{-4} radians over a period of 10 minutes and average velocity of 3000 m/sec.

Tracking Transmitter and Receiver

Est. Wt (lbs)	Volume (cu. in)	Power Req (watts)	Power Rad (watts)	Ant Gain (db)	Ant Beam (width)	Rec Sen Dbm
15		20	2	17	25°	-96

(S) A radio altimeter is to be employed during the approach to landing to determine altitude above the Moon's surface. Planned additional development on an altimeter presently used in the Jupiter nose cone can be used up to 320 km with an accuracy in altitude of $\pm$ 50 ft. Employment of the altimeter at a greater altitude (up to 350 km) can be provided by reducing the receiver band width to increase its usable range at reduced accuracy ($\pm$ 1% of range). Estimated weight is 40 lbs with power consumption of eighty watts.

D. PHOTODATA TRANSMISSION

(U) Visual information may be of value during landing operation and after landing. Present video equipment available that is suitable for transmission of photo data has a maximum of 600 line resolution and a maximum rate of 30 pictures per second which gives a maximum information rate of 10.8 Mc/sec. The band width required for this maximum rate of transmission is 21.2 mc. phase modulated. Very close to proportional reduction in information rate can be allowed by reducing the number of picture per unit time. With maximum resolution, the total number of bits of information per frame is 36,000 (100 km altitude or distance) this is a resolution of 10 meters over each one of 6 km (3.60).

(U) Direct transmission of photo data (TV) during landing requires prohibitive power because of the high information rate. To reduce the rate to a reasonable value requires that a picture be stored and scanned at a slower rate. The information rate required to obtain one frame per min. at maximum resolution is 6 kc. A reduction in resolution to 300 lines gives a reduction in information rate to approximately 2.5 kc.

(U) After landing, since no motion is involved, the scanning rate may be adjusted to provide an information rate that give maximum efficiency in equipment. (See end of integrated system for power required).

E. COMMAND AND FUNCTION CONTROL SUBSYSTEM

(U) The function control is included in the RF system to program the instrumentation and RF functions of the vehicle. A command receiver is included to over ride the function control if needed and to initiate other functions from the ground. The function unit also supplies a stable frequency for the vehicle transmitters. Typical uses of the command and function subsystem are to provide guidance correction, program the radar altimeter, change power level of tracking and data transmitter, etc.

(U) The command receiver and decoder must have a sufficient number of channels to command the number of functions required, and be simple and extremely reliable. The sensitivity of such a receiver is estimated to be approximately -125 dbm. The tracking receiver can also function as a command backup. Because of particular requirements of the package landed on the Moon, a separate command system may be needed. The optimum frequency for that use could be different and must be decided as the requirement is developed. It is proposed that the command receiver be time coded.

F. ANTENNAS, FREQUENCY

(U) The antenna problem is intimately tied to vehicle structure design. Structural problems are reduced by using a minimum number of antennas, but places heavier design burden on RF systems. Ideal radiation from vehicle is isotropic, but for some uses impractical because of extreme power requirements. VHF frequencies appear best for data transmission because of greater circuit efficiencies and higher effective aperture of minimum gain antennas. Mid-course tracking must be in the lower VHF range to minimize propagation problems. Reasonable directivity in tracking antennas is no great problem because the vehicle velocity vector relative to Earth's center does not wary over wide angles. If missile attitude is changed for any purpose, tracking antennas must be programmed or commanded to correspond. Optimum frequencies must be decided by balancing complexity of antenna systems (missile structure) against circuit efficiency functions and ground capability. The use of inflatable or extendable antennas appears to probably be necessary in order to achieve maximum utilization of RF functions, (radar altimeter and tracking antennas for example).

G. INTEGRATION OF SYSTEM

(U) For functional purposes, the aim is for the RF system to consist of subsystems arranged to give maximum reliability and flexibility with a minimum duplication of equipment. For example, the data transmitter used after landing, or a special transmitter for short time high information rate could employ the power stages of the radio altimeter. Subassemblies of the radar altimeter may also be used in the command or tracking system. Detailed engineering of the interrogated system must be worked out by matching the functional requirements vs time, power segments, etc. as they are developed.

(U) The radiated power required from the vehicle is governed principally by the capability of the ground facilities, interference level from cosmic radiation (Sun mostly) and band widths required. Information rate of the usual telemeter information is estimated to be approximately 300 cycles per second based on 40 information channels. Time sharing and communication of channels is assumed to give approximately eight subcarriers with average band width requirements of approximately 100 ($\pm$ 50) cycles per subcarrier.

(C) The total band width required would be approximately 6 kc modulation of a common transmitter with the photo data transmitter offers a saving in equipment and power. Several methods of allowing the common transmission of the telemeter and photo data transmitter is possible. One is by placing the photo data on a subcarrier of approximately 500 kc which would allow transmission of photo data at the rate required if used on the landing approach (100 or 800 kc) or with reduced band width (between 1 and 10 kc) after landing. Photo data could be removed by calling off the subcarrier without disclosing the telemeter data.

(C) On the basis of present and immediate future development in receiver design, the use of parametric and low temperature devices, noise figures of 1 db or less is expected. However, it is not believed safe to consider noise figures of less than 2 db. Noise contributions from celestial bodies are well below this amount of noise. The maximum usable sensitivity of a receiving system is -140 dbm for 6 kc band width, -128 dbm for 10 kc and -127 for 600 kc band width. Additional gain in signal is obtained in the telemeter by virtue of band restriction in the subcarrier discriminators.

(C) With ground facilities limited to 40 db antenna gain in antennas and with a whole antenna gain estimated at 2 db, the resulting power requirements are:

Information Rate	Radiated Power	B. W.
3 Kc (TM)	.300 MW	6 Kc
6 Kc (photodata)	600 MW	12 Kc
10 Kc	1.2 watts	20 Kc
30 Kc	3.0 watts	60 Kc
300 Kc	30 watts	600 Kc