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ARMY BALLISTIC MISSILE AGENCY

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ABMA Form 485, Rev 1 May 59 Previous edition is obsolete.

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1 October 1959 Report No. DV-TR-3-59

LUNAR EXPLORATION
WITH
SATURN - BOOSTED SYSTEMS (U)

Downgraded to Confidential by authority of Automatic, time-phased downgrading stamp. 2/6/66

A Preliminary Study Report

by the

Development Operations Division
Army Ballistic Missile Agency

[redacted]

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PREFACE

The National Aeronautics and Space Administration, in a meeting at the Jet Propulsion Laboratory on 5 February 1959, established a Working Group on Lunar Exploration. Members of NASA, ABMA, Cal Tech, JPL, and the University of California participated in the meeting. The responsibility of the Working Group was the accomplishment of a lunar surface exploration project. The following phases of the project were outlined:

Circumlunar probes, manned and unmanned

Hard landings

Close lunar satellites

Soft landings (instrumented)

It was found that the SATURN booster, in combination with an ICBM as second stage and a CENTAUR as third stage, would provide a proper carrier for manned lunar circumnavigation and for instrumented packages landing softly on the lunar surface. ABMA then prepared a report entitled "Preliminary Study of an Unmanned Lunar Soft Landing Vehicle" (1 May 1959). Subsequent to this report, the National Aeronautics and Space Administration requested the Army Ballistic Missile Agency by NASA Order Hs-219 of 18 June 1959 to carry out a study of a lunar exploration project based on the SATURN vehicle. The project should include the soft landing on the moon of a stationary package and of a package with roving capability, and the circumnavigation of the moon by a manned vehicle with subsequent recovery. The assignment asked for an interim technical report by October, and for the final report on 1 January 1960.

This interim technical report presents ABMA's accomplishments thus far. It discusses all the subjects agreed upon by NASA and ABMA. In some cases, component designs are still too preliminary to allow a reasonable decision regarding their weight, power requirements, and even principle of operation. In such cases, the preliminary character of the design is pointed out. At some places, two alternate designs are described, which are presently under continued study.

During the next three months of the study, design and performance figures of the SATURN vehicle will be more firmly established; the final approach guidance system will be refined; details of design and operation of the instruments for lunar exploration will be further analyzed; technical features of the roving vehicle will be determined; and the sequence of scientific observations will be worked out in more detail. It is expected that many of these decisions will be made in mutual discussions with NASA and members of the Lunar Working Group after review of this report.

A large number of studies and reports by other organizations were read and evaluated by members of ABMA during the past years. Some of the ideas and facts presented in these reports have found application in this study. However, some of the schemes, particularly in the guidance area, did not appear too promising for this project, even though they may find very useful application in other projects. A list of references which are of interest in connection with lunar studies will be included in the final report.

Reference was made to Jet Propulsion Laboratory Report No. 30-1, "Exploration of the Moon, the Planets and Interplanetary Space," edited by A. R. Hibbs. This report presents a schedule of lunar missions which is tailored to fit into the National Space Vehicle Program. The three rocket systems selected to carry out the lunar missions are the ATLAS VEGA, the ADVANCED ATLAS VEGA (with O_2 + H_2 second stage), and the SATURN. While the VEGA systems would carry instrumented packages into lunar orbits and to the lunar surface with "rough" landings, the SATURN would be used for manned lunar circumnavigation with return to earth, and for the "soft" landing of instrumented packages with considerable payload. This study conforms to the schedule presented in the JPL report. However, the guidance scheme for approach and landing was established under the assumption that no information about structural features of the lunar surface will be available to the landing vehicle besides what is known today. Also, it was assumed that no radio beacon, dropped previously in a rough landing, will exist on the surface of the moon to assist the approach of the soft landing vehicle.

Particular care was exercised in the present study to treat the problems of scientific instrumentation as broadly as possible. Besides studying a great many reports on measurements useful in the exploration of the lunar surface, numerous contacts were made with other organizations and individuals, and problems of lunar observations were discussed at length. Although the principal features of the measuring program are well established by now, refinements of this program, as well as a more detailed analysis of the design and operation of the instruments, will take place before the final report is written.

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CHAPTER I

(S) SCOPE OF PROJECT

I.1 SCIENTIFIC PROGRAM

The experimental program of lunar exploration described in Chapter IV is expected to follow a number of previous lunar investigations carried by vehicles less sophisticated than those of the SATURN class. Hence, any group of experiments considered at present are highly tentative, pending results from earlier experiments.

Inclusion of a particular instrument in the soft lunar landing packages is justified only if its nature requires a soft landing, if it is too heavy for less capable carriers, if its power demands are large, or if it must be transported over the lunar surface. Other experiments can be conducted more economically with smaller vehicles.

Emphasis has been placed on those experiments which investigate the structure and selenologic history of the moon, its atmosphere, the nature and magnitude of its fields, and its content of organic material, if any.

Two landings of a stationary instrumentation packet and two landings of a roving vehicle are proposed. In both types of mission, a moon - earth telemetry system will be established having considerable data transmission capability.

The operational lifetime of the stationary packets has been planned for approximately two lunar days and nights. The roving vehicles will operate over one period of lunar daylight.

The scientific program for the two manned circumlunar flights proposed will emphasize utilization of the human ability to observe at close range and comprehend unexpected or unusual properties of the moon's surface. Experiments requiring intelligent decisions may also utilize the human capability. The experience gained in a manned flight will be useful in preparations for later manned landings.

I.2 DESCRIPTION OF THE VEHICLE

Investigation has shown that the SATURN carrier vehicle will be the first United States vehicle with the capability of accomplishing the missions required under this program. Some of the carriers based on ICBM boosters will be able to land a payload softly on the lunar surface, however, such a payload will be measured in pounds.

The carrier vehicle is a three-stage vehicle using conventional propellants in the first two stages, and high energy liquid hydrogen and liquid oxygen in the third stage. It will have an overall length of 200 feet, a thrust at lift-off of 1.5 million pounds, and a weight at take-off of about 1.1 million pounds.

Performance-wise the SATURN will place a payload of approximately 7500 pounds on the way to the moon, exclusive of burned-out third stage tankage, engines, fuel residuals, and injection guidance and instrumentation.

I.3 PAYLOAD TERMINOLOGY AND WEIGHTS

The 7500-pound landing vehicle consists of a braking stage rocket, propellant to accomplish the terminal approach, and the soft landing package. The soft landing package consists of devices to cushion impact, and the payload itself. The payload consists of the necessary structure, power supplies, guidance and control equipment, cooling system, communications equipment, and scientific instrumentation. The payload which remains stationary after landing will be termed the stationary packet. The payload which travels on the lunar surface after landing will be termed the roving vehicle.

Use of the JPL developed 6000-pound thrust N_2O_4/N_2H_4 rocket engine for the braking stage, with 5080 pounds of propellants for terminal approach, permits a 1750-pound landing package to be placed softly on the moon. The Pratt and Whitney O_2/H_2 rocket engine, having 15K to 20K pounds thrust, could also be used. In this case 2350 pounds would be available for the landing package.

The manned circumlunar vehicle consists of the third stage rocket (which is not separated from payload as in a soft landing), fuel for trajectory maneuvers, and the manned capsule. Approximately 6400 pounds will be available for the manned capsule, including devices for atmospheric re-entry protection and recovery.

CHAPTER II

(S) CARRIER VEHICLE AND GUIDANCE FOR LUNAR FLIGHT

II.1 VEHICLE BACKGROUND

In this formulation of a soft lunar landing project, the SATURN system is used as the carrier vehicle. A vehicle of the SATURN capabilities is required to land a significant payload of instrumentation softly on the moon, or to provide a manned circumlunar flight. The SATURN is now under development for the Advanced Research Projects Agency (ARPA) of the Department of Defense (DOD). The SATURN system has great versatility and is adaptable to many diverse missions. This vehicle is one of a series of vehicles planned for the national space program and will eventually become a work horse of the space flight programs.

II.2 LAUNCH VEHICLE

The launch vehicle is a SATURN booster with a modified TITAN ICBM second stage and a modified CENTAUR rocket for the third stage (Figure II.1). There is a study underway regarding the use of the SATURN with various programs and, until the results are known, the dimensions of the upper stages of the launch vehicle will not be finalized. For the purpose of this report it has been assumed that such upper-stage studies will result in establishing the three-stage SATURN system in the configuration and with the performance as described below.

This study has considered only upper stages which are modifications of existing vehicles. A study of the vehicle requirements for optimum staging ratios, which might require new vehicles, is being made. If such optimum upper stages should become available, the payload weights for scientific instrumentation will be raised significantly.

<u>II.2.1 Boost or First Stage</u>. The SATURN booster is a multi-engined, clustered-tank vehicle, 21 feet in diameter and approximately 80 feet long. Its eight JUPITER-type engines develop a nominal thrust of 1.5 million pounds. The eight propellant tanks contain 750,000 pounds of LOX-RP1 which is consumed in slightly over two minutes.

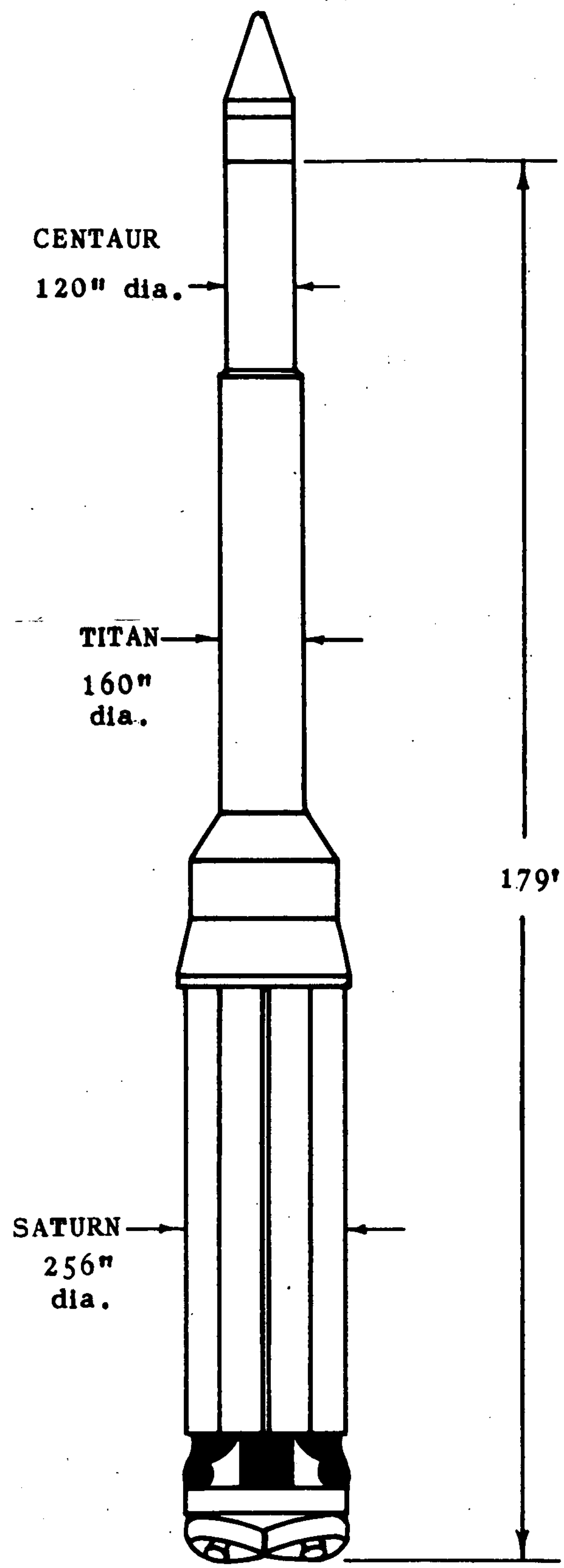


FIGURE II.1 - TYPICAL SATURN CARRIER VEHICLE

Because of the relatively low velocity of the launch vehicle at first stage burnout and separation, a parachute and retro-rocket recovery system will be employed with this stage to cut program costs.

II.2.2 Second Stage. The second stage of the carrier vehicle is a modified TITAN ICBM. This stage has a diameter of 160 inches and a length of about 60 feet. The 220,000 pounds of LOX-RPl propellants are consumed by the two Aerojet 188K engines in about 200 seconds burning time.

Separation of the second stage, like the first stage, occurs immediately after burnout without a coasting period. In each case only a minimum of guidance, controls and instrumentation are separated with each stage since the main guidance compartment will be kept with the third stage.

II.2.3 Third Stage. An increase in tankage volume and strength is incorporated into the CENTAUR package for the third stage of the carrier vehicle. This increase in propellants doubles the 20,000 pounds of liquid hydrogen - liquid oxygen of the standard CENTAUR. No change in the two 15K engines is anticipated except that the thrust level will be increased to 20K each if the engines can be improved prior to scheduled firing dates. The third stage has a length of about 45 feet and a diameter of 120 inches.

Included in the CENTAUR package is an attitude control and vernier system which has four 50-pound thrust nozzles. Consideration is currently being given to the addition of four nozzles for application in launching a satellite into an orbit with a period of 24 hours. If the study proves feasible, these additional nozzles may be incorporated in the lunar missions for possible vernier and mid-course corrections. Under this system the third stage would not be separated from the payload until after the mid-course corrections are made.

A second system is being studied which calls for separation of third stage and payload prior to mid-course corrections. In this case, the mid-course corrections and attitude control of the payload are accomplished by the Naval Ordnance Test Station (NOTS) variable thrust IRFNA - UDMH engine, which is used as the payload attitude control until lunar touch-down, and for hovering if required in the terminal approach schemes.

The above description of the booster, the second and the third stages applies to both the lunar soft landing vehicle and the lunar circumnavigation mission. Because of the very low velocity requirements, on the order of 100 meters per second, needed to accomplish the circumlunar maneuver, it appears that either the present CENTAUR 50-pound engine system or the NOTS variable thrust system will be capable of accomplishing the necessary velocity changes.

The SATURN has an escape payload capability of approximately 7500 pounds. This figure does not include the empty third stage tankage, fuel residuals, engines and injection guidance which, when totaled amount to 14,000 pounds on the way to the moon.

II.3 GUIDANCE SCHEME FOR INJECTING A LANDING VEHICLE INTO A LUNAR TRAJECTORY

The scheme described is capable of performing all guidance and control functions necessary to establish a vehicle in a trajectory which terminates in a selected area of the lunar surface. Upon the approach to the lunar surface, a terminal guidance system becomes operative to allow the vehicle to land softly on a preselected area of the moon.

The scheme was conceived for use with boosters such as SATURN. As the SATURN is a cluster of eight engines, a reasonable scheme should be able to cope with the failure of one engine. This consideration combined with the stringent accuracy required for a lunar mission indicates the need for a relatively sophisticated large-capacity on-board computer.

Since the stabilized platform (ST-90) now employed in the JUPITER missile is of established accuracy and reliability, it is very desirable to develop a scheme which could use this platform. The described scheme can be employed with any combination of boosters which can provide the required performance, such as the SATURN-TITAN-CENTAUR combination.

<u>II.3.1 Flight Mechanics</u>. The objective of the guidance and control system is to inject a soft lunar landing vehicle of approximately 7500 pounds into a trajectory which places the vehicle within a specific region near the lunar surface. Terminal guidance is then initiated when the vehicle enters the specified region.

It is customary to divide a lunar flight into three phases. The first phase is referred to as the injection phase, the second is the post-injection phase, and the third is the terminal phase (Figure II.2). Only the first and second phases are considered in this section.

The injection phase includes the propelled flight from the launch site to the point of injection into the lunar trajectory. The guidance during this phase is entirely inertial. Injection into the lunar trajectory occurs approximately 1000 seconds after launch, and the injection point lies approximately 45° from the launch site, assumed to be Cape Canaveral. A vernier engine will probably be used to smooth out the CENTAUR cutoff errors. At the point of injection into a typical lunar trajectory, the vehicle has an altitude of 520 km and a speed of 10,635 m/sec. The velocity is directed 10° above the local horizontal, the lead angle (see Figure II.2) is 111.58°, and the ensuing transit time from this point to the lunar surface is 58.4 hours. For the hypothetical case without terminal braking, the vehicle would impact vertically on the moon with 2722 m/sec speed. As observed on the moon there would be no lateral velocity component.

The post-injection phase consists of the flight from the injection phase to the terminal phase. The early part of the post-injection phase is often referred to as the vernier phase, and the portion is usually called the mid-course phase. The accuracy of the injection phase obtainable with the guidance system is not sufficient to obtain impact in a fifty-mile square on the moon. Hence, a post-injection correction is desirable in order to conserve energy and to relax the demands on terminal guidance. Information obtained from ground tracking data will be used to determine the fine correction of the trajectory. Several existing types of tracking systems appear to be suited for this purpose.

II.3.2 Guidance and Control Scheme

II.3.2.1 Injection Guidance. The guidance scheme uses an inertial measuring unit consisting of three orthogonally oriented accelerometers mounted on a stabilized platform, and an air-borne navigation computer which computes control and guidance commands. The stabilized platform will be used as an attitude reference throughout the injection phase, and may be used during the mid-course and terminal phases.

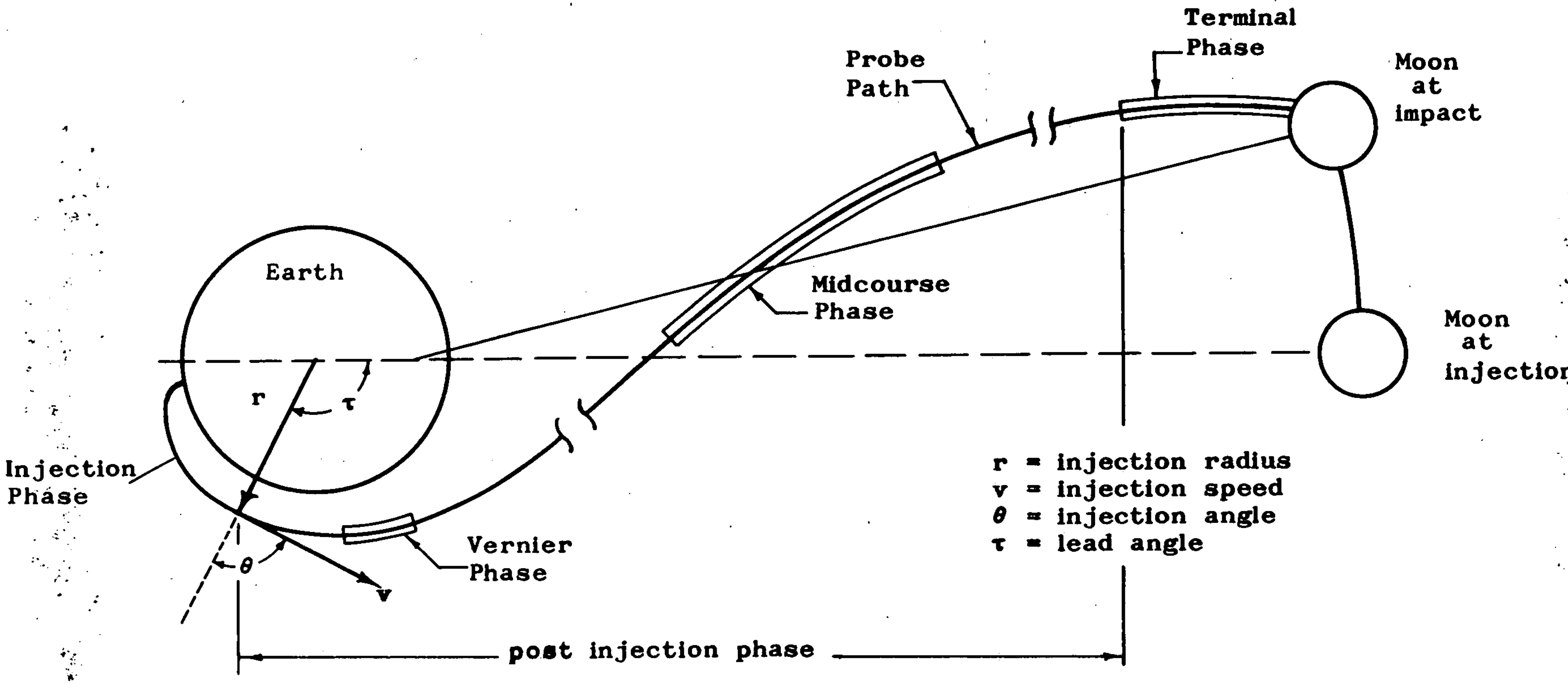


FIGURE II.2 - GEOMETRY OF A LUNAR TRAJECTORY

Delta-Minimum guidance is used throughout the injection phase. "Delta-Minimum" refers to the system's principal feature, that of correcting deviations from the standard flight path as soon as they occur. In order to determine the deviations, a standard flight path, referenced to a Cartesian coordinate system which is space-fixed in direction and moving with the standard missile in a standard gravitational field, is programmed into the missile as a function of time. Measurements of the perturbed flight are made for comparison with the standard so that the deviations become available. Since the perturbed missile will follow a path different from that of the standard, it will experience different gravity accelerations, and its time history in the gravitational field will also be different. Errors due to gravity are a function of the displacement deviations and the time elapsed in the gravitational field. In order that the path deviations be accurately known for corrective purposes, these errors are continuously computed and employed to correct the measured deviations. The coordinate system is oriented initially so that the ξ-axis is tangent to the reference trajectory at some point (e.g., burnout of the first stage), the η-axis is perpendicular to the ξ-axis, and ζ-axis is normal to the flight plane. When the guidance mission with the initial coordinate orientation has been fulfilled, the coordinate system is rotated about the ζ-axis so that the ξ-axis is tangent to the reference trajectory at some later point, e.g., burnout of the second stage. Since the Delta-Minimum guidance used in this scheme guides to an η-reference throughout the flight, the process of rotation of the ξ-axis and η-axis around the ζ-axis may be employed several times to minimize the deviations in η-displacement and η-velocity component.

After burnout of the CENTAUR there will be errors due to the impulse dispersions of the engine. Normally, at this point, there are position, velocity, and time errors. Since the vernier stage cannot eliminate all of these errors, a new trajectory must be selected in order to impact in the chosen lunar area. Hence, it seems desirable that the vernier engine be used to obtain conditions expressed by a cutoff equation.

<u>II.3.2.2 The Vernier Correction</u>. Superimposed upon the known deviations from standard conditions, there will be unknown errors at vernier engine cutoff due to the guidance scheme and the hardware errors. A preliminary error analysis indicates that these unknown errors are sufficiently large that the vehicle may fail to come into the designated fifty-mile square on the lunar surface. Therefore, it is clear that

corrections must be made during the post-injection phase, or the terminal navigation scheme will be subjected to severe demands. Energy considerations indicate that the corrections should be made as soon as possible after vernier engine cutoff.

Preliminary studies show that several available radio tracking methods can provide data which are sufficiently accurate to make post-injection corrections. At present it seems desirable that these corrections be made during the vernier phase, perhaps one hour after injection. The necessary instructions for the corrective maneuver to be executed by the missile will at least consist of the following five pieces of information: alignment angle with x-axis; alignment angle with y-axis; alignment angle with z-axis; setting of integrating accelerometer for proper corrective velocity increment; and time of ignition. The stabilized platform will be used to provide the x,y,z-attitude reference during this correction. There will be errors due to gyro drift and the tracking data. The effects of these errors depend upon the magnitude of the corrections to be made. A mid-course correction may or may not be necessary, but it seems advisable to plan for such corrections.

II.4 TRAJECTORY AND ACCURACY REQUIREMENTS FOR INJECTION

A tentative trajectory for the lunar landing probe has been prepared. The predominant consideration is maximum payload capability. Pertinent considerations determine a velocity level somewhat above the minimum possible to reach the moon. They also require a low declination of the moon for a favorable nearly eastward injection azimuth. The pitch angle of velocity at injection is optimized predominantly from considerations concerning the propelled flight phase.

Further requirements necessitate compromises which cause slight deviations from the performance optimum. Thus, the travel time of the total flight was arranged to accommodate observations of the lunar landing from the United States. Trigonometric relationships between injection point and moon (at encounter) call for certain combinations of azimuth, pitch angle and injection velocity.

The nominal trajectory calculated (based on the Jacobian Model) for the lunar inclination of 23.6 degrees (1964), has the following characteristics:

Declination of moon	-23.6 deg
Injection velocity	10,635 m/sec
Injection altitude	520 km
Injection pitch angle (against local vertical)	80 deg
Transit time	58.4 hrs

This trajectory is presented geometrically in Figures II.3 and II.4 in a space-fixed and a rotating coordinate system, which are simplified by neglecting the third coordinate.

Figure II.5 shows the velocity history referred to the coordinate system rotating with the moon about the barycenter. In the nominal ballistic trajectory, the vehicle would impact on the moon perpendicularly to the surface with a velocity of 2722 m/sec.

The motion of the projection on the earth surface as a function of time is graphically shown on Figure II.6.

The impact point of this trajectory is about 38 degrees toward the moon's leading edge from the moon - earth line as measured at the center of the moon, and about 3 degrees below the lunar plane of motion. This is plotted on Figure II.7.

This figure also exhibits the effects of injection errors on the impact placement. To assure impact on the moon, the restrictions on the injection accuracy are of the following magnitude:

Velocity error	= ± 4.5 m/sec
Pitch angle error	= ± 0.24 deg
Azimuth error	= ± 0.85 deg
Injection time error	= ± 108 sec
(or moon's lead angle error	= ± 0.45 deg)

To insure an impact within an area of 25 miles radius about the selected point, the allowed injection errors reduce

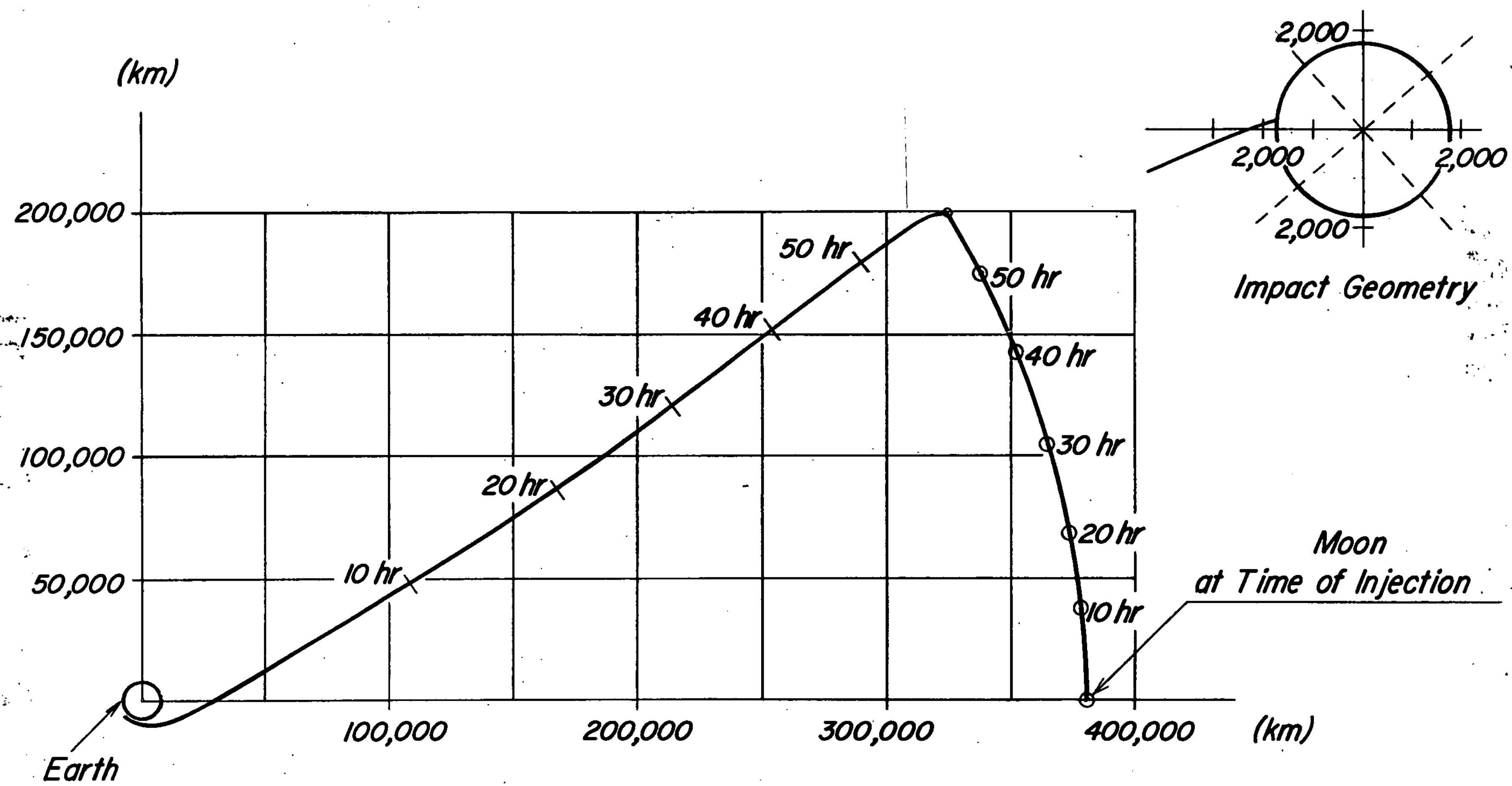


FIG. II.3 LUNAR PROBE TRAJECTORY IN SPACE FIXED COORDINATES
(Slightly Distorted by Neglecting the Third Coordinate)

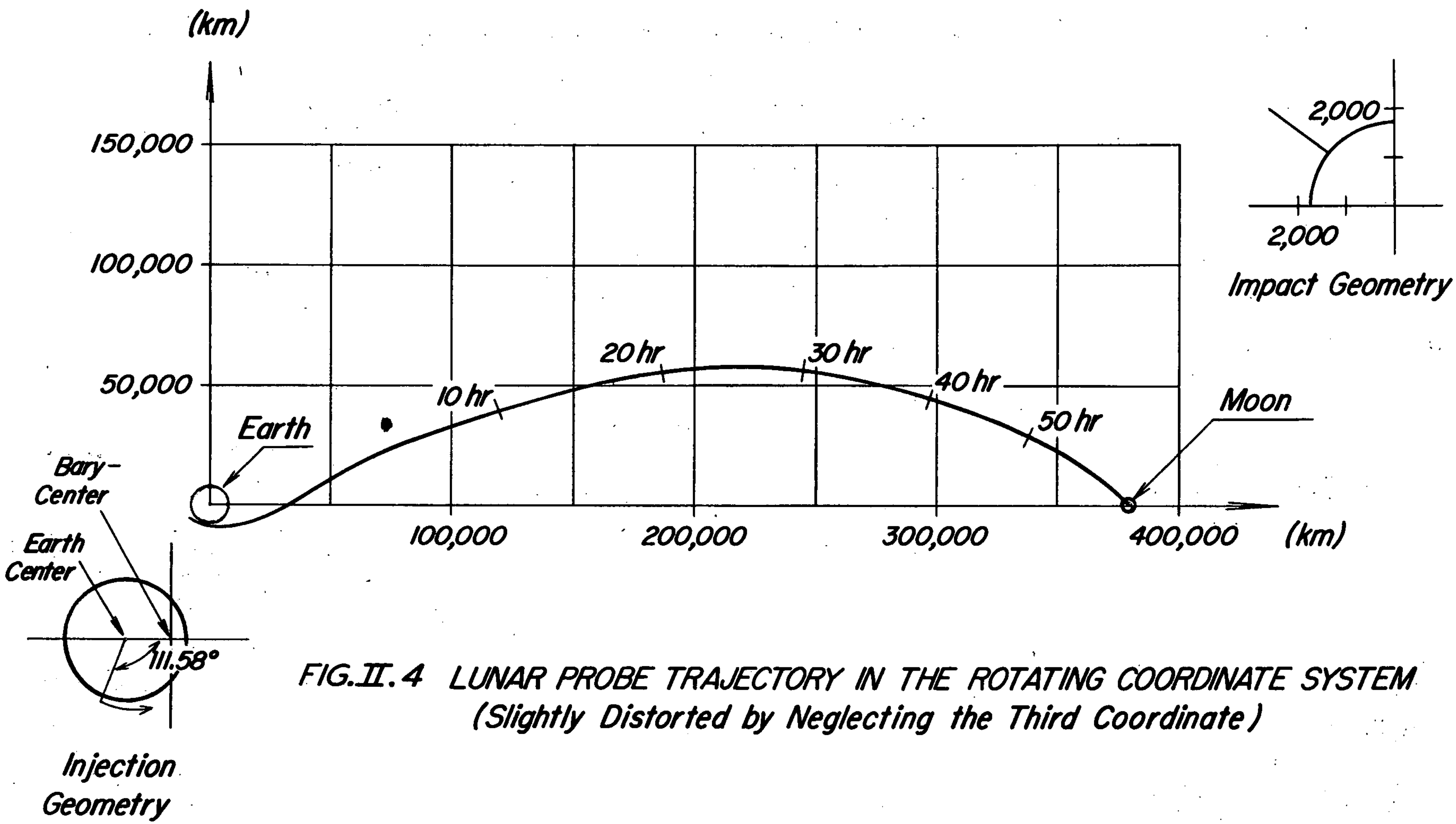


FIG. II. 4 LUNAR PROBE TRAJECTORY IN THE ROTATING COORDINATE SYSTEM (Slightly Distorted by Neglecting the Third Coordinate)

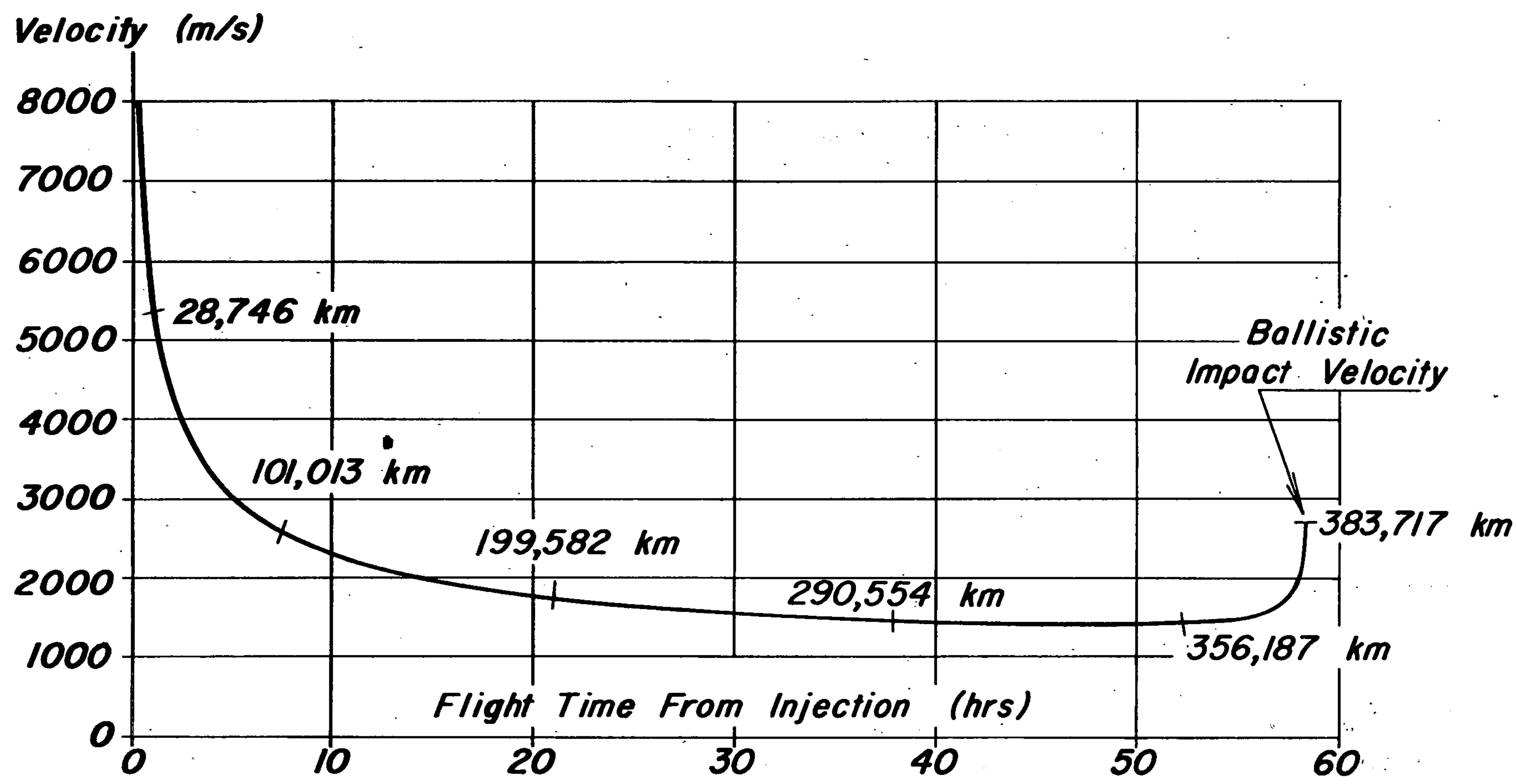
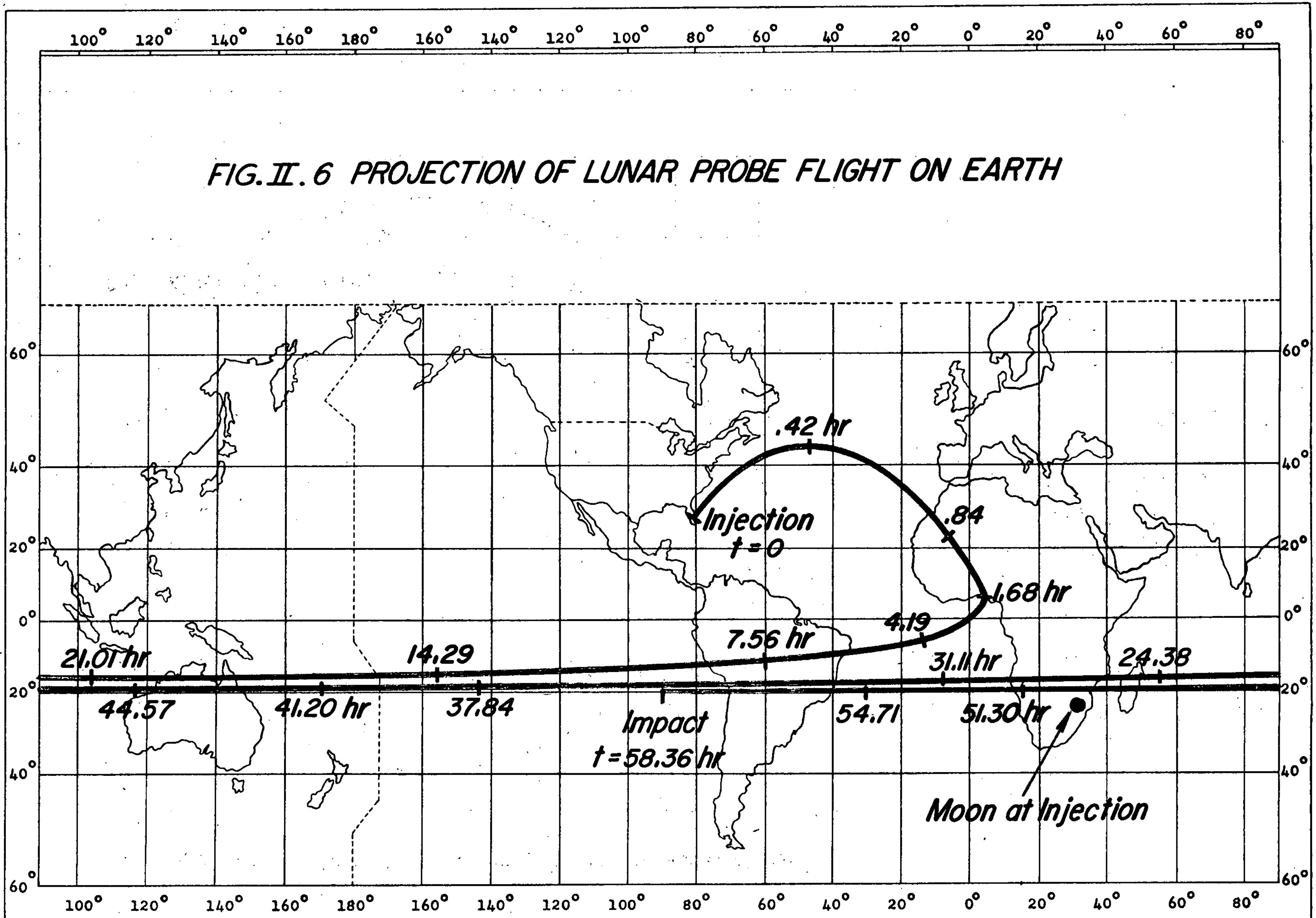


FIG. II.5 VELOCITY MEASURED WITH RESPECT TO ROTATING SYSTEM

FIG. II. 6 PROJECTION OF LUNAR PROBE FLIGHT ON EARTH



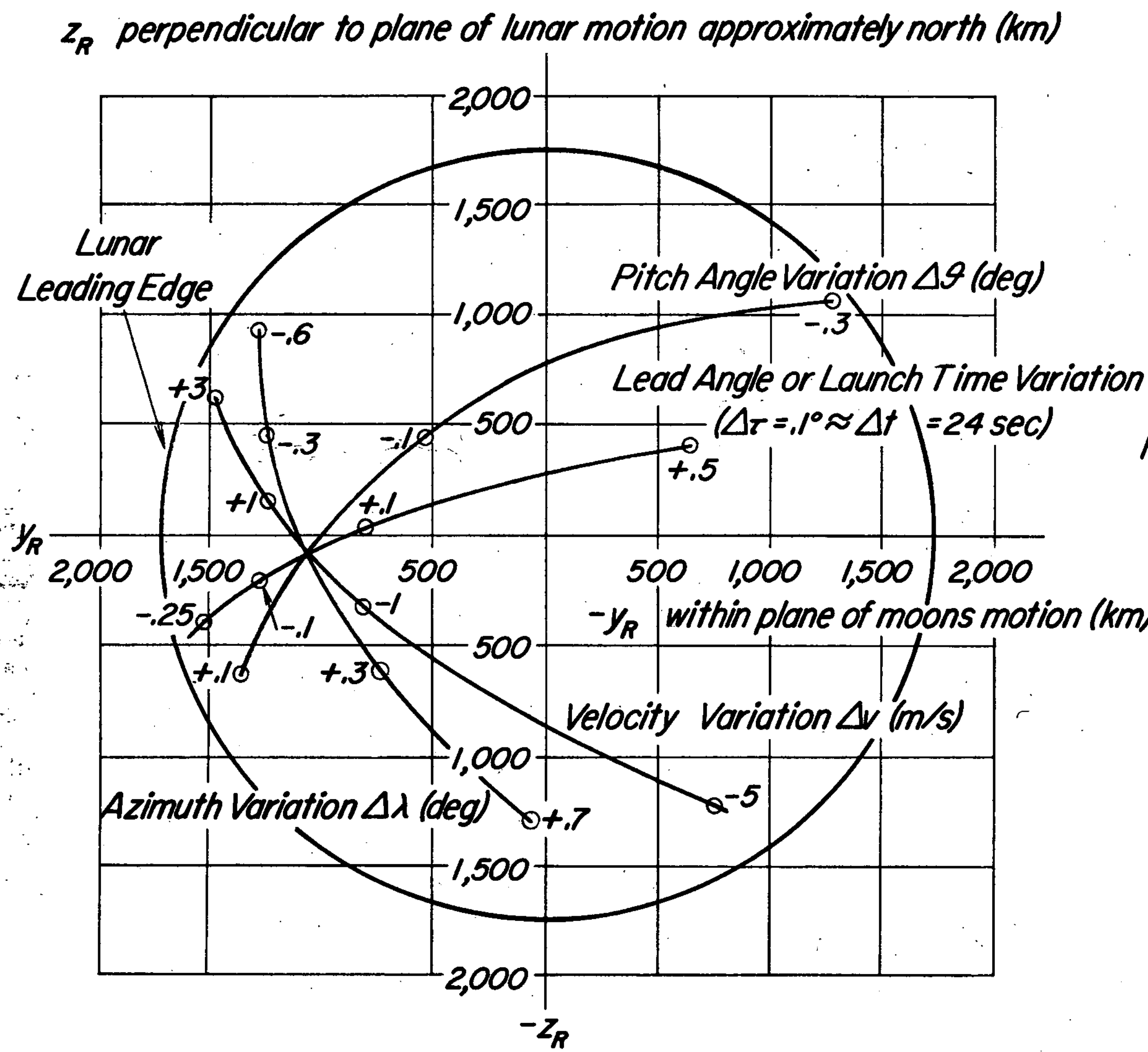


FIG. II. 7 IMPACT POINT VARIATION AS A FUNCTION OF INJECTION CONDITIONS to the following magnitudes:

Velocity error	= ± .13 m/sec
Pitch angle error	= ± .0058 deg
Azimuth error	= ± .022 deg
Injection time error	= ± 3.3 sec
(or lead angle error	+ ± .014 deg)

Any combination of errors reduce the individual allowances listed.

II.5 TERMINAL GUIDANCE SCHEME FOR SOFT LUNAR LANDING

A terminal guidance system employing transmission of television pictures from an on-board camera to a station on the earth presently appears to be the most favorable scheme. From the observation of successive pictures, the point at which the vehicle would touch down if no corrective maneuvers were introduced can be predicted. Correction maneuvers can be commanded from the ground and the resulting touch-down point can again be predicted from continuing television pictures. This procedure may be repeated a number of times if necessary.

The system is capable of first making coarse corrections of the impact point as great as 50 miles and fine corrections up to 100 meters immediately before landing.

The analysis of this terminal guidance scheme is based upon the following suppositions:

(1) The nominal trajectory results in a perpendicular impact.

(2) The injection guidance (improved if necessary by mid-course guidance) is sufficiently accurate to insure ballistic impact within a 100-mile radius of nominal impact.

(3) A braking engine will have about 20K thrust, re-ignition capability and ± 10% controllable thrust level. From the terminal guidance point of view, this is a more desirable thrust than the 6K.

(4) A nozzle system is available for attitude control during thrustless periods.

(5) A command receiver is available as part of the payload.

(6) A television system is available, with the transmitter a part of the payload.

The components of the terminal guidance system are:

(1) Television camera with two different focal lengths.

(2) Inertial platform with one accelerometer, drift rate less than 1^{o}/hr.

(3) Radar altimeter, pencil beam type, for range measurements.

(4) Control system with autopilot.

(5) Infrared horizon sensor

The television camera, altimeter antenna, and communication antenna are not steerable but ridigly aligned with the longitudinal axis of the vehicle as shown in Figure II.8.

The greatest altitude above the moon from which television pictures may be taken having quality sufficient for guidance purposes is limited by the accuracy of the vehicle attitude control and the resolution of the television system. This altitude will be at least some hundred miles, but less than 2000 miles. The distance from this altitude to the surface would be traversed by an unretarded vehicle in a relatively short time. Because of the high approach velocity of about 2700 m/sec, the unretarded vehicle would travel almost two miles during the travel time of a television signal to the earth.

An approach scheme that uses minimum propellants would apply maximum thrust and bring the vehicle to a stop near the surface of the moon. However, this places severe requirements on the guidance instrumentation and allows only a short time to select a landing site.

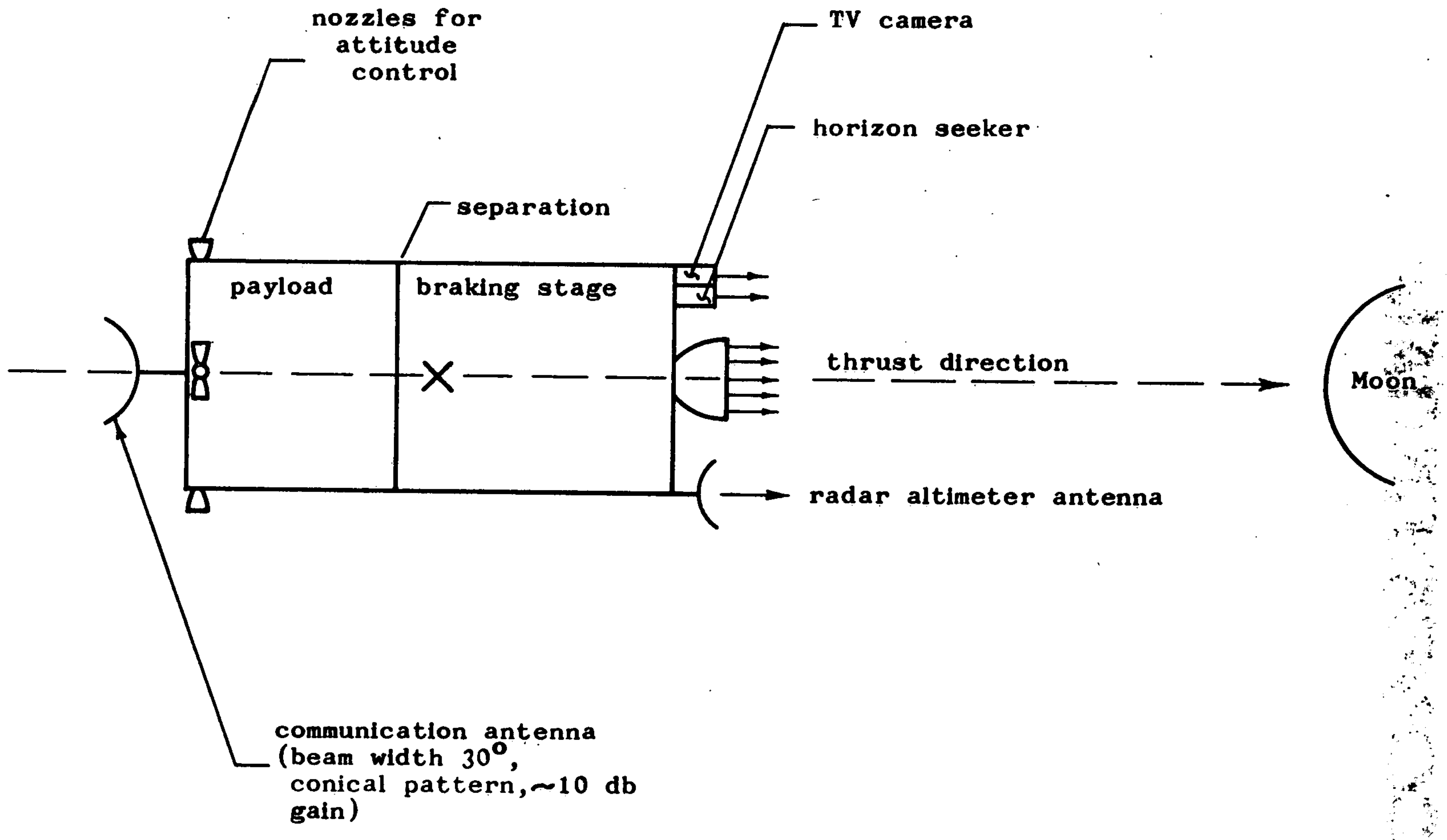


FIGURE II.8

[redacted]

The particular terminal guidance scheme sketched in this section yields a comparatively long descent time. This allows adequate time for transmission of television pictures to earth, for their analysis, and for command signals to the vehicle. While the scheme may not be optimal from the point of view of propellant economy, it imposes realistic demands upon guidance instrumentation. Further analysis of instrumentation techniques should result in a refined descent program using less propellants than the program discussed in the following. Such studies are continuing.

The proposed scheme employs two complete stops, prior to the final landing stop, at altitudes of 100 miles and 1200 meters, each one of the stops being followed by a free fall due to the moon's gravity. Error analyses indicate that the insertion of definite stops rather than continuous retardation allows larger errors in approach velocity, altimeter measurements, and retro-thrust application. During the free fall periods, television observations and altimeter measurements may be made without the possible difficulty of viewing the moon through the rocket exhaust.

The lunar approach sequence is illustrated in Figures II.9 and II.10. Phase 1 extends from 2000 - 3000 miles to 160 miles distance from the moon. At an altitude of about 2000 - 3000 miles, the horizon sensor is actuated by a program timer. It establishes the direction of the local vertical with an accuracy of 0.1°. Simultaneously, the inertial platform is activated. The vehicle is aligned with the local vertical as shown in Figure II.9. For the subsequent television observations, the roll attitude could remain random, but an active roll orientation (e.g., with a sun seeker) would shorten the time necessary for evaluation of possible landing sites. The television camera is then activated and takes pictures at regular time intervals (shutter speed 0.5 milliseconds, maximum frame rate 1 per second). Simultaneously, altimeter readings are transmitted to the earth.

The center of each picture corresponds to the instantaneous subvehicle point on the moon's surface. During the available observation time of about 15 minutes, these points are immediately plotted on a large-scale moon map, located on earth as shown in Figure II.11. From successive plots, the lateral component of the approach velocity is determined and an impact point is predicted. In the event that the topography of the indicated impact area seems unsuitable for landing, the desired lateral correction is determined in magnitude and direction. From this, the necessary change in attitude relative to local vertical is calculated which will produce the lateral velocity to

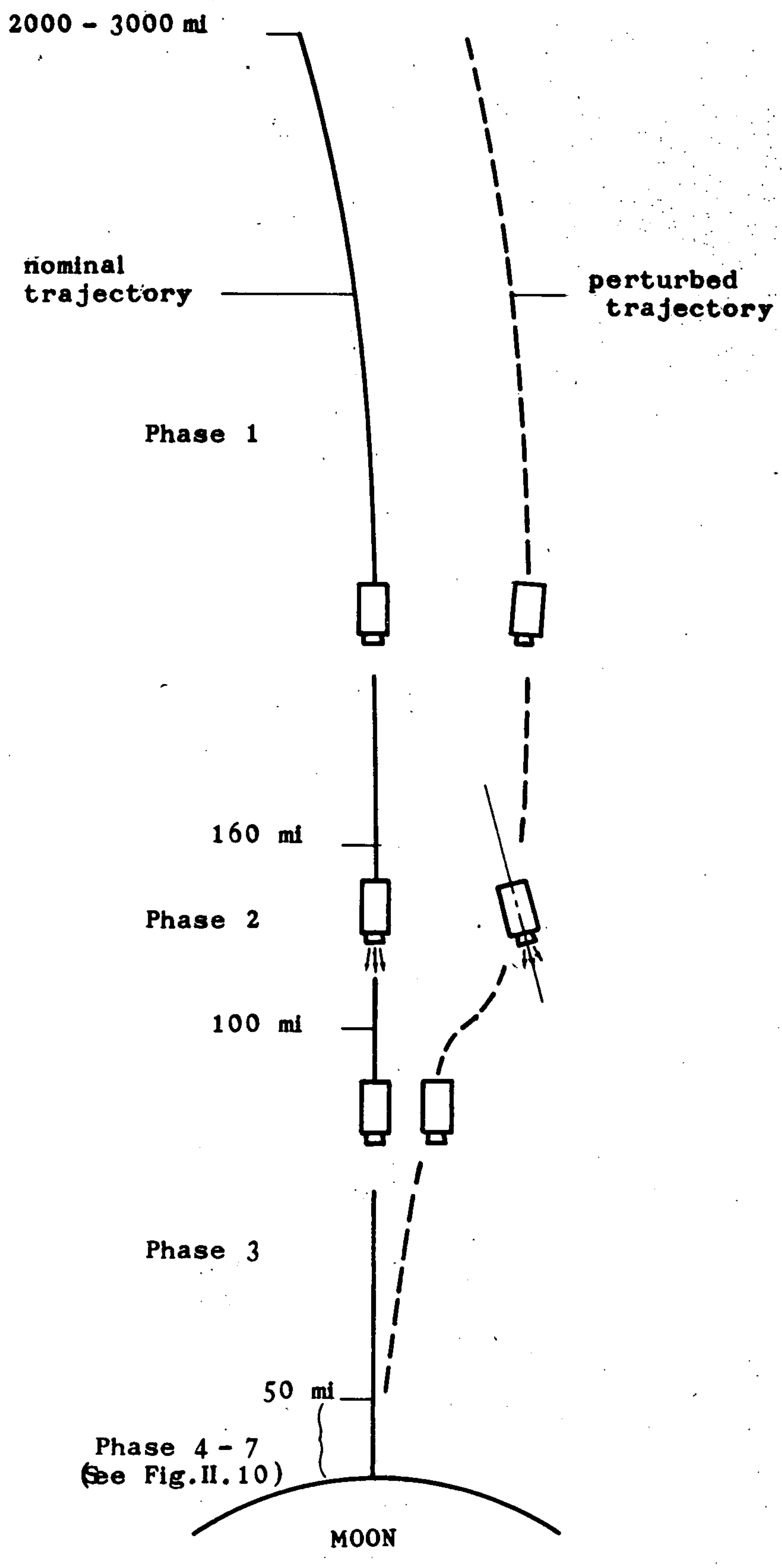

2000 - 3000 mi
nominal
trajectory
perturbed
trajectory
Phase 1
160 mi
Phase 2
100 mi
Phase 3
50 mi
Phase 4 - 7
(See Fig.II.10)
MOON

FIGURE II.9

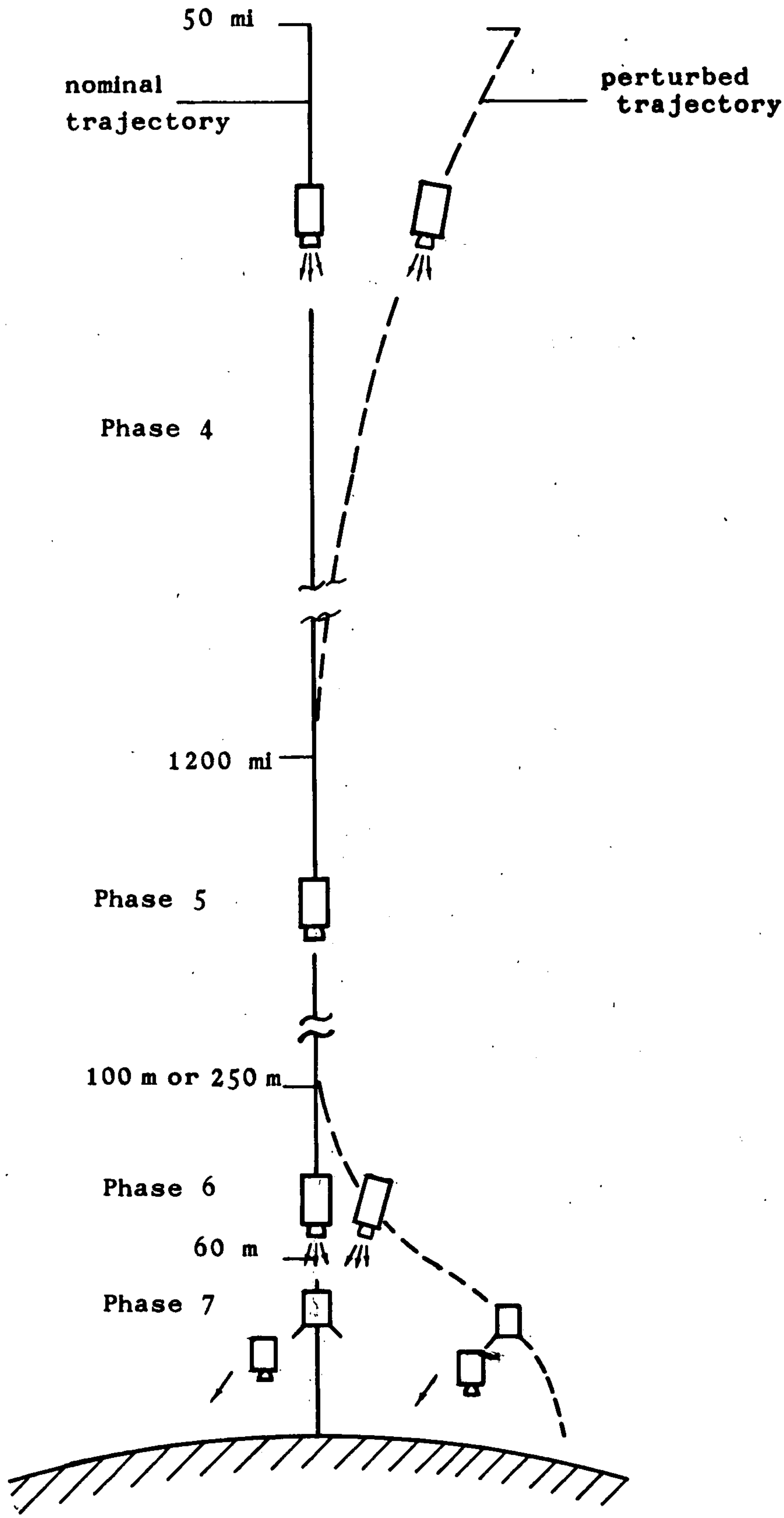

50 mi
nominal trajectory
perturbed trajectory
Phase 4
1200 mi
Phase 5
100 m or 250 m
Phase 6
60 m
Phase 7

FIGURE II.10

[redacted]

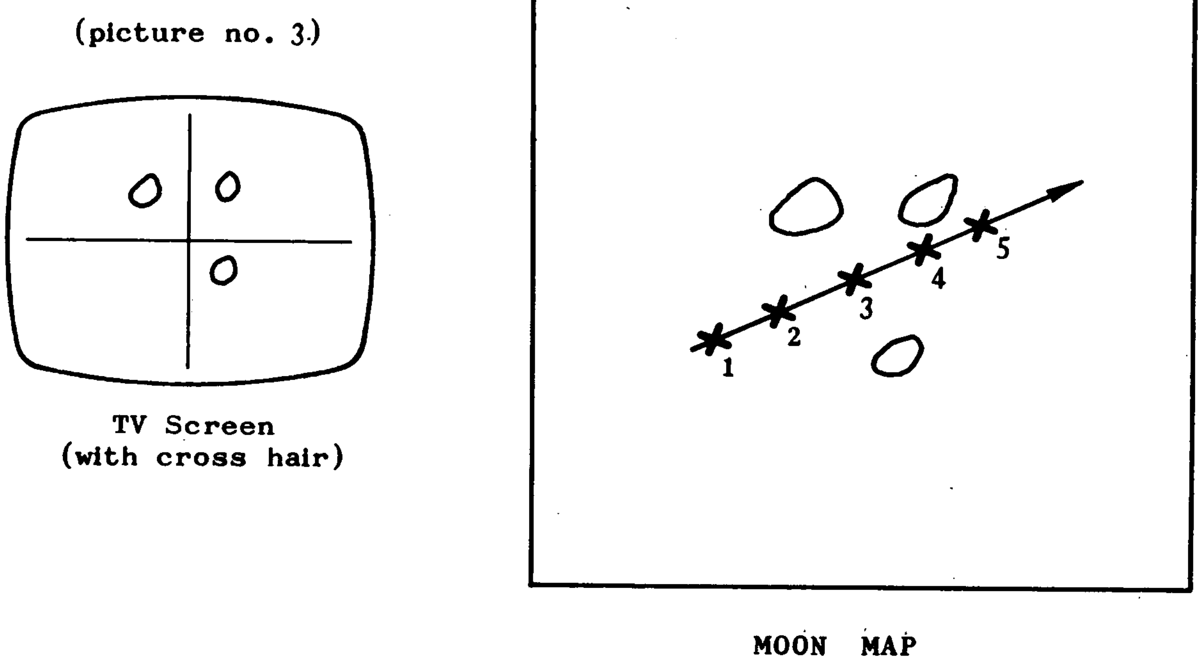


FIGURE II.11

[redacted]

give the desired correction during the subsequent thrust period. This angular change is transmitted to the vehicle by the command link and stored there until needed.

While approaching the 160 miles altitude, the horizon sensor information is smoothed and transferred to the inertial platform which becomes the new attitude reference for the remainder of the flight.

Phase 2 extends from about 160 miles to about 100 miles altitude. At an altitude of about 160 miles, the exact altitude depending on engine characteristics, the braking engine is ignited by a signal from the radar altimeter. Using the nominal approach trajectory, a thrust program was precalculated to obtain zero velocity at exactly 100 miles altitude. Thrust application according to this program is controlled by the accelerometer. Attitude reference is the inertial platform. At the end of this phase, the velocity of the vehicle is nearly zero at a point about 100 miles above the surface.

Phase 3 extends from about 100 miles to about 4 miles altitude. This is a free fall period which lasts for 7½ minutes. The end velocity is about 670 m/sec. During this phase, a second series of television pictures allow the exact determination of the accidental or intentional lateral velocity component remaining after Phase 2. A new predicted impact point is obtained. The necessary thrust program for desired compensation during the next thrust application is calculated, transmitted to the vehicle and stored.

Phase 4 extends from 4 miles to about 1200 meters altitude. The braking engine is started again by a signal from the radar altimeter at an altitude of 4 miles. From the nominal final velocity of Phase 3, a thrust program was precalculated to obtain zero velocity at an altitude of 1200 meters. Inertial control of thrust application takes place as described under Phase 2.

Phase 5 extends from nominally 1200 meters to 250 or 100 meters altitude, depending on the type of engine used in Phase 6. This second free fall period has a duration of 33 or 39 seconds, respectively. It provides a thrustless period for altimeter measurements, television observation, and for the selection of the final landing spot. Reference for the vehicle attitude is the inertial platform. A third series of pictures is taken. If due to the observed topography a lateral correction seems necessary, the corresponding change in attitude for the last thrust application is estimated, transmitted to the vehicle, and stored. The nominal end velocity of this phase is 57 m/sec for free fall to 250 meters altitude or 60 m/sec for free fall to 100 meters altitude.

Phase 6 extends from 250 or 100 meters to 60 meters altitude. The details of this phase depend on whether the thrust of the braking engine can be fully throttled or only controlled up to ± 10%. If the braking engine cannot be fully throttled, an altimeter signal ignites the engine at an altitude of 100 meters. Full stop is achieved at about 60 meters altitude through a programmed thrust application. If the braking engine can be fully throttled, the thrust level for the braking process is arbitrarily reduced to about 30% for minimization of contamination of the landing area. The altimeter ignites the braking rocket at an altitude of 250 meters. Full stop is achieved at about 60 meters altitude.

In both cases, the thrust program determined during Phase 5 is applied. A lateral correction of about 100 meters is possible with an attitude inclined less than 10^{o} to the vertical.

Phase 7 extends from 60 meters altitude to the surface of the moon. During this final free fall phase, separation of the braking stage from the payload is activated by the radar altimeter at an altitude of about 40 meters. Delay of this separation allows for thrust decay. The payload continues a ballistic path while the braking stage receives an additional lateral kick by release of the tank pressure to one side.

The permissible vertical impact velocity, corresponding to a free fall from about 100 meters, is about 17-20 m/sec. Since the vertical free fall velocity from 60 meters is about 14 m/sec, the maximum permissible velocity includes some allowance for residual velocities remaining from the preceding phase.

The attitude of the vehicle and the landing package (after separation) is controlled by the inertial platform.

As previously indicated, the disadvantage of the proposed scheme is that additional energy is needed for execution of the stops, mainly for the high altitude stop at 100 miles. This results in an an additional consumption of propellants of some 600 pounds compared with the ideal case of a uniformly decelerated trajectory with maximum thrust. However, some weight is saved in the guidance components for the proposed scheme compared to the more sophisticated components that would be necessary for the minimum propellant scheme.

CHAPTER III

LANDING AND CIRCUMLUNAR PACKAGES

(S) III.1 DESIGN PHILOSOPHY

The design of the landing package for the stationary packet should, if possible, be similar to the landing package of the roving vehicle in order to gain landing experience. The lessons from the first soft landings may well be applied with profit to later landings. Consequently, the final design of the landing accessories for the stationary packet and roving vehicle package will be nearly the same. The final design of the manned circumnavigation package will be based upon the unique requirements of that flight.

As discussed in Chapter II, the SATURN vehicle system is capable of injecting approximately 14,000 pounds into a lunar trajectory. The empty third stage weighs approximately 6,500 pounds, and the useable payload weight is approximately 7,500 pounds.

The proposed braking stage considered in this report is the JPL variable thrust engine with a nominal thrust of 6,000 pounds, using hydrazine - nitrogen tetroxide, or a similar hypergolic storable propellant combination. The propellant is pressure-fed by stored gas, probably helium at 3000 - 5000 psi. Using the JPL 6K engine, the weight division for the landing vehicle is as follows:

Braking stage hardware	670 lbs
Braking stage propellant	5080 "
Landing package	1750 "
Total	7500 lbs

The selection of the JPL 6K engine is definitely a conservative approach. If the Pratt and Whitney O_2/H_2 rocket engine of 15K pounds thrust (one engine of the two-engine CENTAUR propulsion system) were chosen, a weight of 2350 pounds would be available for the landing vehicle, as contrasted to 1750 pounds using the JPL 6K engine. These landing vehicle weights are based upon the three-stop terminal guidance scheme described in Chapter II.

The Pratt and Whitney engine can be up-rated to 20K thrust. This value has arbitrarily been used in discussions of the terminal guidance scheme in Chapter II.

After burnout, the braking stage is separated from the landing package. To prevent the braking stage and the landing package from falling on the same spot, several schemes could be employed. Under investigation are venting the propellant tanks of the braking stage on one side to give it lateral velocity; or displacing the payload laterally by a small rocket. The control system is housed in the landing package. Stored gas or propellant is used with nozzles on the circumference of the package to control roll, pitch, and yaw.

Two alternate approaches are described for the design of the soft landing packages. The first gives prime consideration to the stationary packet, which is modified appropriately for the roving vehicle. Conversely, the second gives prime consideration to the roving vehicle, which is modified as required for a stationary packet. This dual approach insures adequate consideration of requirements for both missions. Both interim designs, in their present form, represent compromises between the requirements of the stationary packet and the roving vehicle. Upon completion of this study, it is anticipated that the best points of each will be integrated into a single final design.

The requirements imposed upon the lunar roving vehicle are as follows:

(1) Withstand free fall onto the lunar surface from a height of 300 feet ($\sim$100 m) without exceeding 20 G (earth) deceleration. This requirement also exists for the stationary packet.

(2) Operate under lunar environmental conditions as a mobile vehicle for one full lunar day (daylight).

(3) The roving vehicle must have a range of about 50 miles, be able to negotiate a slope of 15°, and pass over boulders 1 foot in diameter, and avoid larger boulders.

(4) Due to uncertain surface conditions, it should be able to travel on thick layers of dust.

(5) Lunar contamination should be kept to a minimum.

The trajectory for the lunar circumnavigation flight has not been finally determined in detail, and consequently the thrust and weight requirements for the maneuvering stage have not been determined. At this time it may be tentatively assumed that a weight of approximately 6,400 pounds, exclusive of the maneuvering stage, will be available for the circumlunar payload.

(S) III.2 DESIGN OF LANDING PACKAGE, FIRST ALTERNATIVE

This alternative gives primary consideration to the stationary packet, which is modified appropriately for the roving vehicle.

The preliminary design of the landing package with the stationary packet is shown in Figure III.1. The total weight is approximately 1,750 pounds, distributed as follows:

Stationary packet structure	250 lbs	
Device for separation from braking stage	50	
Solar cells	50	(100 W, daytime)
Mercury batteries	225	(20 W, 2 lunar nights)
Structure for solar cell bank	50	
Shock absorber & mounting	250	
Guidance & control equipment	350	
Cooling system	175	
Weight available for scientific instrumentation	350	
Total Weight	1750 lbs	

Shock absorption during landing is provided by an inflatable bag beneath the payload packet. The bag is constructed of flexible side walls, and a stiff metal bottom, to prevent punctures. Eight lightweight metal arms are attached to the structure, which extend in a manner resembling spider legs. The extended arms provide a large landing base, and prevent toppling of the landing package if the landing has lateral motion, or occurs on a slope. The outer section of the arms are crushable in order to brake lateral motion. Upon landing, the inflated bag releases gas at a rate to minimize instantaneous deceleration, and to prevent rebounding. The stationary packet does not

[redacted]

FIG. III. 1

separate from the landing package, but merely comes to rest on top of the deflated bag.

The selection of communication antennas depends upon the terminal guidance scheme. If a dish antenna is required, it can be placed on an inner section of the opened arms. Dipole antennas can be placed at any location on the packet shell.

The stationary packet itself consists of a cone-shaped container housing the sampling drill at the center, and related instruments around it. The remainder of the scientific instrumentation, as well as the guidance, control and communications equipment, is also within the container. An active cooling system is used, and will be described later in more detail.

Electric power is generated during the lunar day by a solar cell bank with an output of 100 watts. The bank is gimbal mounted and follows the sun by means of two electric motors actuated by a light-sensing device. At the end of a lunar day, the bank is reoriented toward the rising sun for operation during the next lunar day. During two lunar nights, non-chargeable mercury batteries supply 20 watts of continuous power for operation and heating of essential instruments, and for heating the battery container. The batteries also provide the power for reorientation of the solar cell bank.

The stationary packet can be modified for use as a roving vehicle. Figure III.2 shows the modification. Shock absorption during the landing is accomplished in the manner previously described. The solar cell power supply is similar to that of the stationary packet, but an additional power supply is required for the vehicle drive motors. An open cycle turbine-driven electric generator system, utilizing storable propellants, is used for power generation. The open cycle system has the disadvantage that its exhaust may contaminate the lunar environment, but it has a weight advantage over a closed cycle system, which requires a condenser and heat radiator. Two large inflated wheels, each driven by an electric motor, are folded into the landing package during flight. They are extended and locked into place shortly before landing. Two arms, extended perpendicularly to the axle, transfer to the ground the reaction resulting from the wheel torque during travel. The vehicle can be moved forward or backward by reversing the motor rotation, and can be turned by application of dissimilar torques to the wheels.

[redacted]

FIG. III. 2

The roving vehicle landing package weighs 1750 pounds, divided as follows:

Roving vehicle structure	250 lbs
Device for separation from braking stage	50
Solar cells	50
Gas turbine and fuel	135
Structure for solar cell bank	50
Shock absorber and mounting	250
Guidance and control equipment	350
Cooling system	175
Driving mechanism, tires, wheels and other roving accessories	220
Scientific instrumentation	220
Total Weight	1750 lbs

(S) III.3 DESIGN OF LANDING PACKAGE, SECOND ALTERNATIVE

This alternative is based primarily upon the requirements for a roving vehicle, and is modified for use as a stationary packet. This roving vehicle preliminary design is shown in Figure III.3.

The vehicle consists basically of two 16-foot diameter balloon tires connected by an axle about 20 feet long. The payload package is suspended from the axle between the two tires. Drive torque reaction will be transmitted to the lunar surface by a torque arm trailing the vehicle, extending from the payload package. The vehicle has two torque arms, one for forward drive and one for reverse. They are designed such that the one not used for torque reaction is inclined forward 30° from the vertical and houses a forward scanning television camera. The arm which is touching the ground houses an identical television camera to be used for very close observation of the lunar surface. Upon reversing the drive torque the raised arm will come down and its camera will examine the lunar surface. A wheel is located on the end of each torque arm to reduce friction during motion and absorb the shock when lowered.

[redacted]

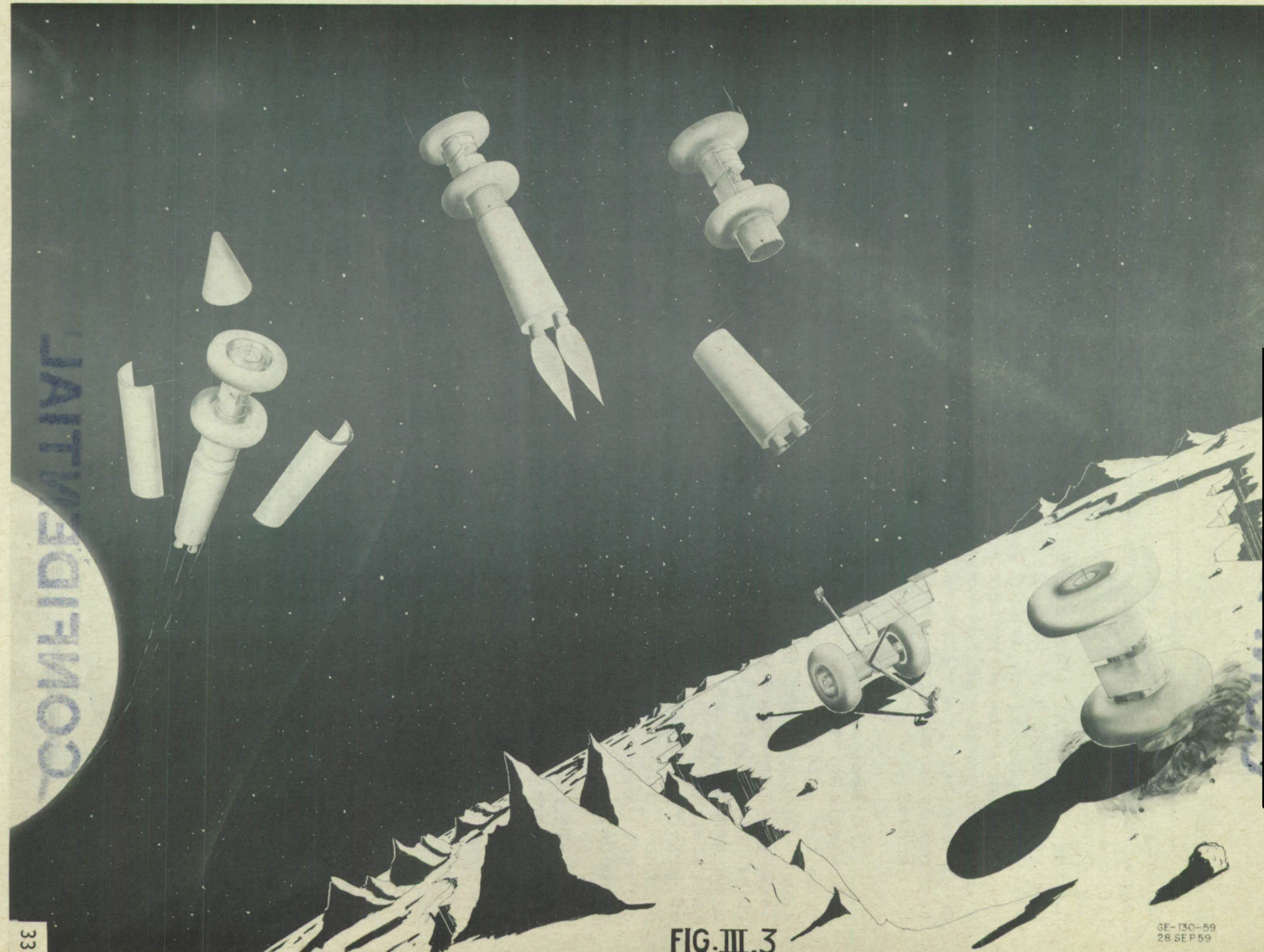


FIG. III. 3

To assure that the vehicle survives the initial lunar surface impact, a crush bag of collapsible material is provided. It is designed to give as near constant deceleration as possible. The large balloon tires filled with a small amount of air before launching expand after the vehicle leaves the earth's atmosphere to 0.5 psi pressure, thus providing a secondary impact area in case the vehicle does not land directly on the crush bag, and when the vehicle topples after impact into driving position. To withstand initial impact, the major components of the vehicle are flexible. A crush bag made of honeycomb material or plastic foam probably gives the best deceleration characteristics. A gas-inflated bag may result in objectionable rebound. A study is now being made to determine which of these designs will be most suitable.

The body of the vehicle is free to rotate with respect to the wheels so that if the body has a center of gravity offset from the axle, it will always come to rest with the same orientation. During flight and landing, the torque arms and solar cell bank are retracted into the payload package. After landing a timer will initiate the release mechanism, extending the torque arms and solar cell bank. This simple scheme is possible because the position of the package is known due to the offset center of gravity. Some instrumentation components are located on one of the torque arms for convenient lunar surface contact. The correct torque arm must be down before these experiments can be made.

Vehicle drive can be attained in many ways: a power plant with direct mechanical drive, a turbogenerator with one drive motor, or a turbogenerator with two drive motors.

If a direct mechanical drive is used, a central power plant driving rotating axles is probably better than one power plant for each wheel. Steering of the vehicle necessitates the use of a split drive axle with some provision to drive either side independently or together. A tank-type drive would probably be an acceptable choice.

Using a turbogenerator with one electric drive motor would present the same basic steering problems as the straight mechanical drive. If the motors were driven from batteries and the turbogenerator used to keep them charged, then an underrated turbogenerator could be used. The same batteries could be used for both electronic equipment and locomotion. A study to find the best weight distribution between fuel, turbogenerator, drive motors, and storage is presently underway. The turbogenerator has the additional asset that it can charge storage batteries for any purpose. A turbogenerator with two drive motors may offer the best type of drive. The motors and a planetary reduction gear would be housed inside the axle at the wheel hub. Steering of the vehicles could be achieved by using servotachometers attached to each motor. When both tachometers are turning at the same speed, their electrical outputs will be equal and cancel. With different speeds, the different outputs will, when subtracted, produce a net output which operates a control mechanism. To maneuver the vehicle signals are fed to the control mechanism.

The tires will present several design problems due to air leakage, ultraviolet effects, abrasion, tear resistance, heat absorption, out-gassing, micrometeorite damage, sensitivity to sharp rocks, need for tire treads, and landing effects.

The roving vehicle landing package weighs approximately 1750 pounds, distributed as follows:

Device for separation from braking stage	50 lbs
Solar cells	50
Gas turbine and fuel	135
Structure for solar cell bank	50
Crush bag and mounting	235
Guidance & control equipment	350
Cooling system	175
Tires	100
Wheels	50
Axle and instrument housing	150
Electric motors	30
Planetary gears	30
Torque arms	50
Mounting for electronic equipment	50
Scientific instrumentation	245
Total Weight	1750 lbs

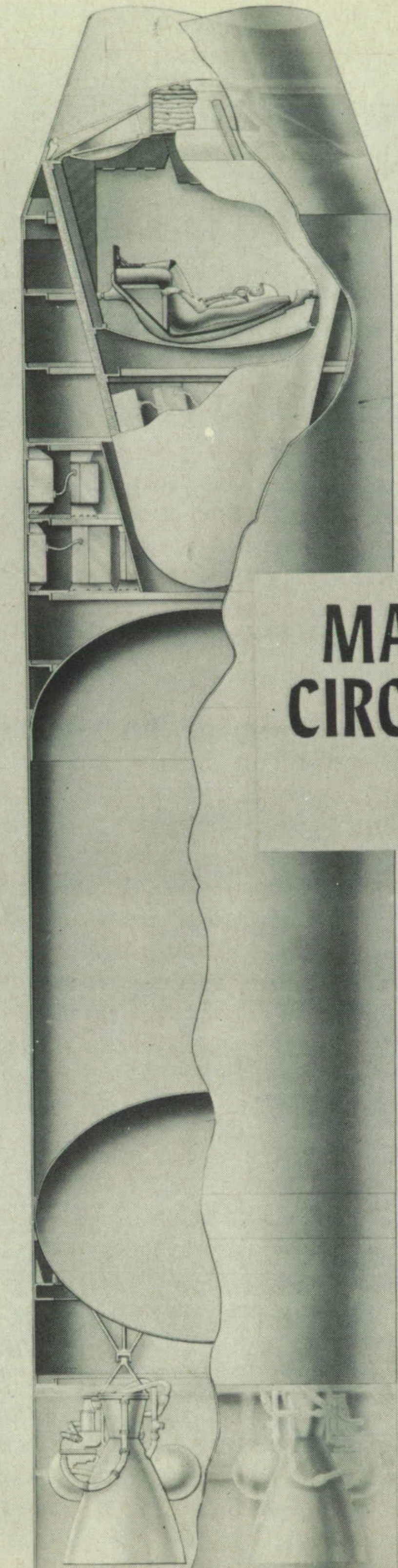
The roving vehicle can be modified for use as a stationary packet by elimination of the drive mechanism and its power supply, by modification of the torque arms for use as mountings for the television camera and other instruments, and by modification of the tires. After landing, the packet rights itself, the body assumes the correct position with respect to the vertical, the arms are extended, and the experimental program begins.

(S) III.4 MANNED LUNAR CIRCUMNAVIGATION VEHICLE

The preliminary design for the manned lunar circumnavigation vehicle is shown in Figure III.4. The vehicle consists of the empty third stage of the launch vehicle, the manned capsule, and an aerodynamic shroud. The total weight injected into a lunar trajectory is approximately 14,000 pounds, consisting of 6500 pounds for the empty third stage, and 7500 pounds payload. As mentioned in Section III, approximately 6400 pounds is expected to be available for the manned capsule. This weight is distributed as follows:

Structure	1510 lbs
Re-entry heating protection	2500
Guidance, control and communication equipment	600
Electrical network	260
Recovery system	300
Passengers, scientific equipment and environmental control	1230
Total Weight	6400 lbs

The manned capsule has a configuration similar to the reentry nose cone of the JUPITER ballistic missile. Control surfaces provide capability for maneuvering during re-entry. The capsule can house two men, their artificial environment, food and water supply, instruments for observations, survival kit, and the communication and guidance equipment. Parachute recovery gear is attached to the capsule. The electrical power for this relatively short mission will be supplied by silver oxide-zinc batteries, since this system weighs less than others if only short operating times are required.



MANNED LUNAR CIRCUMNAVIGATION VEHICLE

FIG. III.4

GE 133-59
30 SEP 59

III.5 THERMAL PROBLEMS

The sources of thermal energy for an object on or near the lunar surface during the lunar day are:

(1) Direct solar radiation

(2) Infrared radiation from the lunar surface

(3) Moon albedo

(4) Energy from on-board mechanical, electrical and electronic devices

(5) Energy from on-board exothermic chemical reactions.

Means for dissipating excess thermal energy from the object are:

(1) Infrared radiation to the lunar surface and/or to space

(2) Absorption of heat by materials undergoing a thermodynamic change of phase

(3) Endothermic chemical reactions.

The temperature of the moon at the subsolar point has been measured by observers. Their results vary between 81 and 134°C.

A flat plate that has an emissivity equal to 1 at all wavelengths on the sun-exposed side and is adiabatic on the opposite side will have an equilibrium temperature of 123°C. A small object near the surface of the moon receives infrared radiation from all sides. For a sphere the flux of this radiation is twice the solar constant.

A flat plate that is suspended over the lunar surface at the subsolar point has an equilibrium temperature less than 123°C, if it is reflective on the moon side, and if it is partially reflective to the solar spectrum but black at long wavelengths on the sun side. Such a plate might be used for a radiator.

On the other hand, if the plate is almost black to the solar spectrum and has an emissivity less than 1 at long wavelengths on the sun side, then its equilibrium temperature will be higher than 123°C. This case will be represented by a solar cell bank, which will assume 190° or more if no precautions are taken as described in Chapter V.

During the lunar night, the equilibrium temperature of the moon has been deduced from experiment and theory to be from -121°C to -153° C. For vehicles primarily dependent upon power from solar cells during the lunar day, night operation must be severely curtailed. The problem then is to provide thermal energy to instruments to prevent low temperature damage, in contrast to the problem of removing excess thermal energy during the lunar day.

Moon albedo has been deduced to be about 7.3% of the sunlight or about 87.6 Kcal/hr m. In all data that follow, the albedo has been considered negligible, but the moon temperature has been taken as 123°C. For a detailed analysis of the thermal problems for the final design, it will be necessary to include the moon's albedo, and its emissivity for infrared radiation which is less than 1. However, the more accurate results obtainable by this analysis will not change considerably the approaches and design concepts discussed in this study on the basis of the stated simplifying assumptions.

Dissipation of electrical energy into thermal energy, called internal generation, is a significant factor in the temperature control of the vehicles considered here. During the lunar night, internal generation represents the only source of energy available to maintain the temperature.

Exothermic chemical reactions become important when chemical sources are used for power, such as the use of an H O system for locomotive power of the roving vehicle. Thermodynamic change of phase becomes important if gases from chemical reactions must be condensed in a semi-closed process, or if vaporization and condensation is used in a closed cycle power generation system. Change of phase could also be used to maintain devices at constant desired temperatures, but from the standpoint of payload limitations, materials in large quantity could not be transported to the moon for the purposes of thermal control. Endothermic chemical reactions for thermal control are in the same category.

III.5.1 Stationary Packet. The stationary packet contains instruments which must be kept within allowable operating temperature limits. In addition, the power supply equipment, solar cell deck and batteries function only within a limited temperature range. The exact limits for each instrument have not yet been exactly defined. However, for this initial temperature control study the upper limit has been considered to be about 60 or 70°C, and the lower limit about 0 or −10°C. The battery operating range corresponds approximately to the instrument operating range.

In the following discussion the backside of the solar decks is assumed to be highly reflective in the infrared. If naked solar cells are used without a coating, then their maximum equilibrium temperature will be about 190°C. If the cells are coated with a material transmissive in the solar spectrum and with a high infrared emissivity, then their maximum equilibrium temperature will be about 110°C. If the cells are coated with a material that reflects three-eighths of the unconvertible solar energy, transmits the remainder, and has a high infrared emissivity, then their maximum temperature will be about 55°C. However, this condition is very optimistic and requires coating materials that are not yet known.

Another way to reduce the solar cell temperatures is to include some radiative area on the cell side of the decks. This area would be covered with a material that is reflective in the solar spectrum, has a high infrared emissivity, and would always be oriented away from the moon's surface. The ratio of the radiative area to solar cell area, A_R/A_S, can be calculated as a function of deck temperature and various emissivities. For example, uncoated cells would require a ratio of 2.1 for a temperature of 60°C. Cells coated for high infrared emissivity but no special reflection characteristics would require a ratio of 1.3 for a temprature of 60°C. Cells using the very optimistic coating previously described would not require a radiator and would have a temperature of about 55°C.

Adding radiator area adds weight to the solar decks. Increasing temperature of the cells decreases efficiency and, hence, adds weight. Adding radiator area decreases temperature. Obviously an optimum radiator area and solar deck temperature exists which results in minimum weight.

During the lunar night the solar decks will be at equilibrium with the moon surface (−121 to −153°C) unless energy is generated within the payload and conducted to the decks. The solar decks must necessarily be designed for the sun period to radiate as much thermal energy as possible. Therefore, they would also radiate energy at a high rate during the night. It does not appear practical to design elaborate multiple shielding to automatically enclose the decks at sundown and thus conserve energy during the night. Neither chemical energy nor battery-supplied electrical heaters are feasible for heating the cells, due to the excessive weight required. It appears that the solar cells cannot be protected during the lunar night with the size payload under consideration. Heating of cells is a doubtful possibility due to power limitations. Every attempt should be made to design solar cells to withstand a night temperature of about $-150^{\circ}C$.

It is clear from the introductory remarks that the main body containing the instruments must be cooled by a radiator if the packet is to operate at the subsolar point. The radiator surface must be parallel to the moon's tangent plane at the subsolar point because its surface must possess a high infrared emissivity. In addition, it must be shielded from reflections from the main body and the solar decks. The external surface of the main body must be highly reflective to infrared radiation from the lunar surface and should be in the shadow of the solar decks when the sun is overhead.

Alone, the main body would have a high equilibrium temperature, and the radiator alone would have a low equilibrium temperature. Therefore, adequate transfer of energy from the main body to the radiator must be assured. This can be accomplished by a closed fluid circulation system consisting of tubing and a small rotary type pump (gear, cam, or screw). Since the main shell assumes a uniform temperature and completely encloses the instrument compartment, the instruments are in equilibrium with the shell. Under operation, the radiator temperature will be almost the same as the shell temperature. To maintain the packet at $60^{\circ}C$ with 100 watts internal generation requires about 0.6 m^2 of radiator area with a surface designed for a high heat rejection rate.

During the lunar night the radiator-circulation system is inactive. Since only a few instruments will operate during this period, the battery power for these instruments will be considerably less than the solar power generated during the sun period. Consequently, the internal heat generation will be much less, as will be the energy available for temperature control. Assuming no protection other than the low infrared emissivity of the main body surface, an average instrument temperature of $0^{\circ}C$ would require about 60 watts. The weight of batteries necessary to provide this internal generation is prohibitive.

One recourse is to enclose each instrument or group of instruments in small vessels with muptiple walls (Dewar). This scheme appears feasible for protection of the operating instruments. Each instrument must be examined individually when its temperature limitations are better defined. Those instruments that do not operate during the lunar night need not be heated to their operating temperature, but they should be controlled within broader limits that represent the range outside of which mechanical damage would occur.

III.5.2 Roving Vehicle. Many of the principles discussed in connection with the stationary vehicle apply equally well to the roving vehicle. There are factors that must be taken into account on the roving vehicle that were not considered in the discussion about the stationary vehicle, namely:

(1) The instrument compartment is not necessarily shaded from direct sunlight at the subsolar point.

(2) A fuel tank is necessary for storage of fuel used to generate locomotive power.

(3) If a closed thermodynamic cycle must be used for the locomotive power to avoid possible moon contamination, a condenser will be required.

(4) The vehicle traverses uneven terrain so that the problem of minimizing the radiator's view of the moon's surface is more difficult.

Considerable research and development must be accomplished to solve the thermal problems of the roving vehicle.

III.5.3 Landing Location. During the lunar day, considerable relief would be obtained if the vehicles were not landed within the moon's equatorial region, but within the northern or southern temperate zone. Every attempt will be be made to design the payloads for operation under extreme thermal conditions, but it should be noted that proper choice of the landing location would make some of the temperature problems less severe.

III.6 ENGINEERING MATERIALS FOR USE ON THE MOON

This section is intended to introduce lunar material problems and to provide preliminary recommendations for materials subjected to the following environment:

(1) Estimated vacuum below 10^{-7} mm Hg

(2) Two-week temperature cycles between approximately + 125°C and −150°C.

(3) High intensity radiation spectrum from the near UV to the far IR. The intensity of the energetic particle radiation is not known exactly, but is expected to be high.

In the expected temperature range mentioned above, many metals and alloys can be used as structural materials, especially the following alloys: stainless steels, types 202,302,304,321,347 17-7 PH and 15-7 Mo PH. Titanium alloys are suitable, particularly B 120 VCA and 6 Al -4V. For further weight savings, many aluminum alloys should be considered to be very practical because of their light weight and ability to be surface treated by anodizing. This is essential for heat balancing of the structure because anodized surfaces can be colored (dyed) with many inorganic pigments which are believed to be radiation resistant.

In the expected temperature range, many ceramics (including metal oxides, silicates and cermets such as borides, carbides, nitrides, silicides, etc.) are suitable. Preference should be given to materials with low coefficients of thermal expansion and resistivity to high energy radiation. Aluminum oxide, magnesium oxide, zirconium oxide (stabilized), porcelain (glazed for radiation shielding purposes, for example, with a $CdO-PbO-B_2O_3$ glass) and other ceramics are suitable materials. Beryllium oxide is very suitable for applications where high specific heat is advantageous. For best transmission of the solar radiation spectrum, quartz glass is recommended. Only radiation resistant glasses can be used for optical purposes such as windows, lenses, filters, etc. Some such glasses are already commercially available.

Because of the expected ultra-high vacuum on the moon, foamed ceramics and ceramic powder bodies in combination with solar radiation shields will be most suitable for thermal insulation. For example, multilayer materials in which good solar radiation reflectors (such as aluminum or silver) alternate with foamed or powdered ceramics or fibrous glasses will give top performance.

Due to the environmental conditions such as ultra-high vacuum, high energy radiation (particle and solar), and the relatively wide range of temperature, most organic dielectric materials deteriorate sooner or later. Such deterioration is, in the simplest cases, vaporization of plasticizer or vaporization of the material itself, loosening of linkage and weakening of bonding forces. In the more complicated cases, radiation intensifies the vacuum effects. At low temperatures these effects are somewhat retarded, however, most plastics become very brittle. Even for short-time application, as considered in this study, the selection of organic materials is very small, and these must be selected carefully for the specific engineering application. Mylar has good low temperature flexibility and vacuum stability. No information is available on its particle radiation stability. Other organic materials particularly for the upper temperature range with excellent vacuum stability are silicone rubbers and teflon plastics. The life expectancy of these materials can be increased by reinforcing them with radiation absorbent fiberglass. In areas shielded against solar and particle radiation, the deterioration of these materials can be retarded considerably. For long service life under space conditions, organic dielectrics may be replaced by inorganic insulating materials. Instead of copper, aluminum conductors can be used whose surface can be anodized for insulation. Such anodized aluminum conductors will give trouble-free service for an indefinite time, however, flexibility of such conductors is somewhat limited.

Extensive studies are underway concerning lubrication in vacuum or under space environment. So far, only meager results are available. Organic lubricants are suitable only for a limited time due to the vaporization of these compounds, particularly in the upper temperature range. Encapsuling bearings and other moving parts in pressurized areas is an unsatisfactory solution because of design problems. A more promising approach is the use of solid inorganic lubricants. An understanding of low temperature performance of solid lubricants, however, requires more investigation. Questions related to the general problem of wear, fretting and dry friction are at the present time in the focus of interest and additional studies are underway.

To our present knowledge, materials for solar cells are not subject to deterioration in ultra-high vacuum. Their behavior in the high energy particle radiation environment of space has not been studied so far. Some detrimental changes may be expected. Electrical energy storage at low temperatures is not difficult, however, discharge of suitable electric batteries may be a serious problem below $-20^{\circ}C$. Well insulated, pressurized areas may be necessary in order to keep batteries under suitable temperature which can be provided by a slow catalytic combustion of hydrogen and oxygen, generated in times of power surplus by high pressure electrolysis.

The foregoing discussion for applicable materials is of a very general nature. Detailed answers will be provided as soon as more specific requirements become known.

CHAPTER IV

LUNAR EXPLORATION

IV.1 PRELIMINARY CONSIDERATIONS CONCERNING LUNAR EXPLORATION

Prior to landing a large payload of the SATURN class softly on the moon, much will have been learned about the moon from experiments transported by less sophisticated vehicles. The experiments considered here belong to some later generation of experiments.

These criteria suggest certain priority policies with respect to the SATURN payload. Out of the many possible experiments, preference should be given first to those requiring the deposit of sizable packages gently on the moon, and to those demanding transportation of the instruments over the lunar surface. Highest priority should be reserved for investigations of the structure of the moon, its atmosphere, its fields, and other characteristics. A lower priority is appropriate for experiments measuring the environment near the moon, such as meteorite or primary cosmic ray fluxes, which can be determined without a costly soft landing on the moon itself. A still lower priority would seem to apply to those experiments using the moon only as a convenient platform for observation of the earth, the sun, or astrophysical phenomena. The latter undertakings may, of course, be eminently justified in later phases of space operations after the properties of the moon have been adequately determined.

Early lunar probes and impact vehicles, both from the USA and USSR, have been carefully sterilized to reduce the danger that the lunar surface will be contaminated with living organisms from the earth. Presumably this troublesome procedure will have to be continued until appropriate studies have been made on the lunar surface to determine what organic materials, if any, are naturally present. Since a soft landing probably presents the first opportunity to undertake such studies, the inclusion of such experiments is of great importance.

Many experiments to yield information about the moon have been suggested by scientists in the journals or verbally. The NASA Lunar Working Group has met repeatedly to evaluate these suggestions and to generate still further possibilities. The experiments proposed for the SATURN soft lunar landing missions have been selected on the basis of the findings of this group and on the basis of the priority philosophy expounded in the previous paragraph.

These proposed experiments must be recognized, at best, to be representative of the experiments which will finally be carried to a soft lunar landing by vehicles of the SATURN class. Experience with satellites, as with other programs, has shown that results of the first generation of experiments dictate major changes in the nature and the details of experiments in later generations. Contrasted with this is the long time interval required to design, prepare, and test an experiment for lunar investigation. The reconciliation of these situations in a rapidly moving lunar exploration program will demand the utmost ingenuity from the scientists involved.

A problem common to all the measurements proposed is the interference caused by the presence of the vehicle. This difficulty is familiar to every experimental worker. The problem may be reduced if certain precautions are taken. Thus, for example, a landing scheme should be adopted which lands the vehicle in a location having minimum contamination from the jet of the deceleration rocket. Outgassing and biological contamination may be reduced by careful preparation of the vehicle and its payload, as described in the section on the mass-ion spectrometer.

Lunar environmental conditions which may degrade, or at least influence the operation of instruments designed for use on the earth are: lunar surface temperatures, high vacuum, micrometeorite impact, and unfiltered cosmic and solar radiation. All of these must be taken into consideration in the payload development program.

It seems that of all the environmental influences experienced by scientific apparatus aboard a soft landing vehicle, that imposed by the temperature on the electronic circuitry is the most serious. In particular, transistorized circuits and photomultiplier tubes have narrow operating temperature ranges. On the other hand, the following instruments are quite insensitive for the lunar day-night temperature range: Geiger-Mueller counters, magnetometers (the proton precession and helium vapor magnetometer may require some intermittent temperature control), and mass-ion spectrometers of the conventional magnetic-electrical field separation type.

IV.2 LANDING SITES

The proposed program of lunar exploration includes two soft landings of a staionary packet, two soft landings of a roving vehicle, and two circumlunar flights. The landing sites selected for the first four missions are intimately related to the experiments to be performed.

The nominal earth - moon trajectory discussed in Section IV.4 terminates at a point between Kepler and Landsberg in a vertical impact on the moon's surface (Figure IV.1). From the flight mechanics point of view, this is the easiest point for landing. It also corresponds to the greatest probability of success. Although the area does not provide striking topographic or structural features such as the "Straight Wall" or the "Alpine Valley," it does provide a region covered by rays from the crater Copernicus, a mare floor, numerous craters and a good position for a gravity oriented antenna aboard the vehicle as well as for the gravimeter. This area is an appropriate location to undertake the landing of the stationary packages.

The roving vehicle will be designed for a range of about fifty miles. The vehicle would be best utilized by landing it within roving distance of several strikingly different features of lunar topography. Examples have been offered of landing sites which are desirable from this point of view. One such spot might be in the Mare Nubium near the Straight Wall. Another might be in the Mare Imbrium near the Alps Mountains. A landing in the crater Alphonsus might answer questions about this area. Drawing upon experience gained from landing the stationary vehicles, landing the roving vehicles at points offering features of particular interest should be practical. Operation of the roving vehicle at a latitude away from the latitude of the subsolar point has the added advantage of making the vehicle temperature control problem less severe.

IV.3 PROGRAM OF SCIENTIFIC INVESTIGATIONS

A diverse group of instruments are included in the stationary packet and the roving vehicle. They have a common objective, namely, investigating the material and processes which have been, and are, responsible for the properties of the moon.

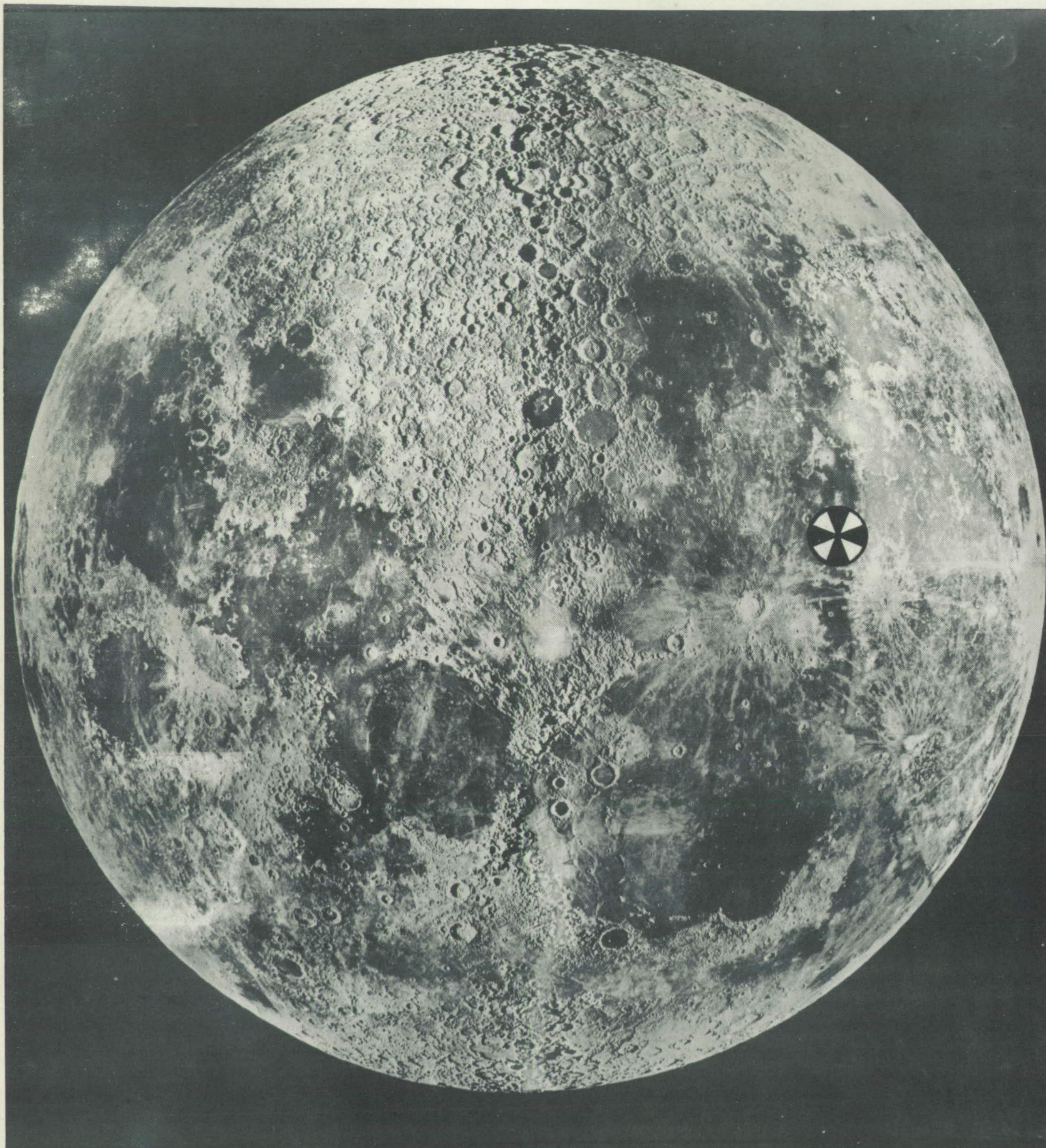


FIG. IV.1

The scientific program of the proposed soft lunar landings include investigations of the structure and selenologic history of the moon, its atmosphere, the nature and magnitude of its fields, and its content of organic material, if any. The results of these investigations will answer many of the questions raised by scientists. They also bear directly or indirectly on our understanding of the earth and the solar system. Likewise, data obtained through these scientific studies contribute environmental information essential to human occupation of the moon.

Because of the high payload capability of a SATURN class vehicle, a great variety of instruments in both the stationary packet and roving vehicle may be devoted to the first area of study, the structure and selenologic history of the moon. The proposed assembly of apparatus includes: 1) a television reconnaissance system; 2) gravimeter; 3) seismic apparatus; 4) thermal measurement devices; 5) special sample collecting equipment; 7) X-ray fluorescence and diffraction equipment; and 8) a gamma ray scintillation spectrometer. The investigations facilitated by these instruments range from a study of the moon's history, as evidenced by its local relief, to a study of the state of the moon's interior by its response to the influence of solar gravity. Instruments will be included to determine the chemical composition of the exogeneous material in its surface as well as that of the primary material of the moon.

Instruments designed for the measurement of atmospheric phenomena, i.e., electron and ion density, neutral particle density, and identification of constituents, are carried both in the stationary packet and mobile vehicle. Not only do these measurements determine the atmospheric environment on the moon, but they also have implications concerning the processes associated with the formation and evolution of this semi-planetary body. Instruments used primarily for lunar atmosphere detection and identification are the mass-ion spectrometer, an ion gauge and plasma probe.

Measurements of the lunar magnetic and electric fields are also included in the instrumentation packages. Knowledge of the magnetic field contributes to an understanding of the moon's origin, and also complements the radiation data obtained. The electric fields are likewise intimately related to the atmospheric study. In the case of the lunar gravimeter, measurements of the gravitational field are utilized to determine the state of the moon's interior.

While the origin of the solar system is not definitely understood, it is possible that the moon may have been surrounded by an atmosphere at some time during its evolution. Organic molecules may have been produced when energy was absorbed by the lunar atmosphere, and may have drifted to the surface. The moon could also be the landing place for the hypothetical cosmobiota, micro-organisms in space. Instruments to prove or disprove these hypotheses would be quite valuable, and are proposed.

Each of the proposed experiments is discussed in detail in the subsequent paragraphs. Their characteristics are summarized in Table IV.1.

IV.3.1 Television System. As a versatile, multipurpose tool for the lunar scientific mission, a television system may be used to observe the lunar surface and instrument operation, and to monitor the operation of the roving vehicle.

A television system with a linear resolution of one part in 10^3 can provide high resolution pictures of the lunar surface within a few miles of the vehicle. Surface characteristics important to a study of lunar relief and its causes, such as facture patterns, roughness, extrusive and small topographic features, may be obtained over the lunar day. By utilizing the varying angles of solar illumination on the surface, even small displacements may be seen. Fracture patterns, noted from the high resolution descent phase pictures, can be examined at close range. Extrusives, such as dikes or lava flows, may or may not be detected depending on the difference in their color, texture, or relief from that of the host material.

It may be assumed that television will be utilized in lunar vehicles prior to the SATURN flight. However, the high payload capacity of the SATURN vehicle allows the use of this device as a multipurpose instrument capable of providing optical control of the scientific mission.

A television system with extremely high resolution may find application in the actual experimental work being performed aboard the vehicle. Transmission of spectrographic or diffractometer images, and the microscopic examination of lunar samples under different conditions can be a valuable task performed by special cameras within the instrument package.

The television camera utilized in the terminal guidance phase of the lunar landing will be separated and discarded with the braking stage. The scientific payload will contain a variety of special purpose cameras and associated circuitry. Assuming a weight of approximately 10 pounds for a scanning camera capable of variable focus (Zoomar lens), and 3 pounds each for the auxiliary cameras plus circuitry and structure, the television package for a stationary vehicle should weigh about 35 pounds. Power will be around 20 watts unless a high rate of transmission is required.

In the stationary package, the camera for scanning the lunar surface may be mounted internally and the image reflected into the vehicle via rotating mirrors, prisms, etc. One camera should be placed in the instrument package to observe the sampling and testing operation. A light source must be provided for such a monitor. With special light source and suitable optics, a visual examination of the lunar samples may be made either in combination with fluorescense apparatus, or simply by observing the character of the lunar material at low powers of magnification.

The state of the art in special purpose television is quite advanced; adopting such a system to the lunar vehicle involves primarily design considerations.

IV.3.2 Sample Collector. The sample collector must obtain samples of the lunar surface and subsurface material without excessive mixing. These samples must be presented in a suitable form to the analyzers. Reliable operation must be insured.

The lunar surface composition is unknown, but it may reasonably be assumed that the top few centimeters of the sample will be mixed with cosmic dust accumulated over selenologic time. Another assumption, the logic of which depends on the concept adopted for the origin of lunar surface features, is that fragments produced by collision of foreign bodies may have accumulated to a depth of many meters over portions of the surface. In the course of selenologic time, with the absence of appreciable atmosphere, this debris, except for the uppermost portion, may have bonded together to form a conglomerate of mixed composition.

It is desirable for the sampler to keep the surface material separate from the subsurface material, and to provide samples obtained from known depths, if changes in the vertical section of the moon's outer portion are to be detected. Although it is not certain that any changes will occur within the short depth attainable with the small apparatus in the lunar payload, perhaps some indication may be obtained concerning the nature of the indigeneous lunar material.

A number of methods may be used to obtain a sample of the lunar surface. These include scoop and clamshell devices, core drills, drills with gas or liquid lift, brushes, and magnetic or electrostatic means. Both the core drill and the drill with gas or liquid lift involve problems of operating reliability and sample mixing that have not been resolved as yet.

One method may be recommended which obtains a sample from a selected depth and which prepares the sample in suitable form for analysis. This method uses a cutting bit with a mechanical lift. Preliminary studies indicate that a tungsten carbide bit with a slightly concave cutting head and a sample chamber mounted atop the bit allow the recovery of very small samples without mixing or loss of the sample. The drill stem can be retracted to lift the sample either to the test apparatus or to reject the material at the surface. Estimated dimensions and power requirements are given below. With optimum materials and configuration, the requirements should be lowered considerably.

Estimated weight	60 lbs
Power requirement	200 watts
Size, retracted	Cylinder, 6" dia. x 24 to 36"

The device is shown in Figures IV.2, IV.3, and IV.4.

In a retracted position, the drill and case are housed in a vertically mounted cylinder located in the base of the stationary packet or roving vehicle. In drilling, both the case and the drill are lowered to the lunar surface, the drill case resting firmly against the surface. In recovery, the drill stem is retracted, and the drill bit rises with the drill case. The base of the cylindrical housing may contain a magnetic separator as well as a sample collector and container to carry the sample to the instrument sensors.

With the solar cell output combined with a battery supply, a total of five hours of drilling will be available to the stationary package. In the roving vehicle, the sampling time will be determined by the earth-based operation, monitoring the program via television. Since the sampler will operate from the auxiliary power source of the roving vehicle, a number of holes may be drilled along the vehicle path by using only a few pounds of fuel. It appears that the lifetime of the apparatus will depend more on the bit wear than on the power available.

Successful operation in high vacuum environment will require some outgassing lubricant for the sampler surfaces under load. As in the other mechanical apparatus used in the lunar environment, care must be taken to utilize materials tested and proven under vacuum conditions and to test the actual apparatus in vacuum.

An automatic drilling system having the ability to recover hard rock samples under conditions as extreme as those encountered on the moon has not been developed by industry. A modification of existing drill components and considerable testing will be necessary to develop a workable apparatus for lunar sampling.

Some of the disadvantages of alternative methods for obtaining samples are given briefly as follows:

<u>Scoop</u>: The compactness and hardness of the lunar surface and subsurface material is unknown, but may approximate that of igenous rocks. The scoop method is thought to be undesirable except for surface detritis, because its application is limited to soft materials.

<u>Clamshell</u>: Although the clamshell has been used extensively for the sampling of surface materials, the method is not usable for indurated or hard materials.

<u>Core drill</u>: Core drilling has been widely used to obtain samples in sequence as they are found in the rock system. Recovery of the cored sample is quite poor wherever the sampled section varies considerably in density or where fractures or crevices exist. In ordinary practice, the core bit and barrel are lubricated in the sample hole to prevent seizing and wear, and the sample is examined for contamination from smearing of the drilled material along the core.

PRELIMINARY DRAWINGS OF A PORTION OF THE SAMPLING APPARATUS

(Figures IV.2, IV.3, and IV.4)

a - support

b - drive motors

c - lift motor

d - outer drill stem

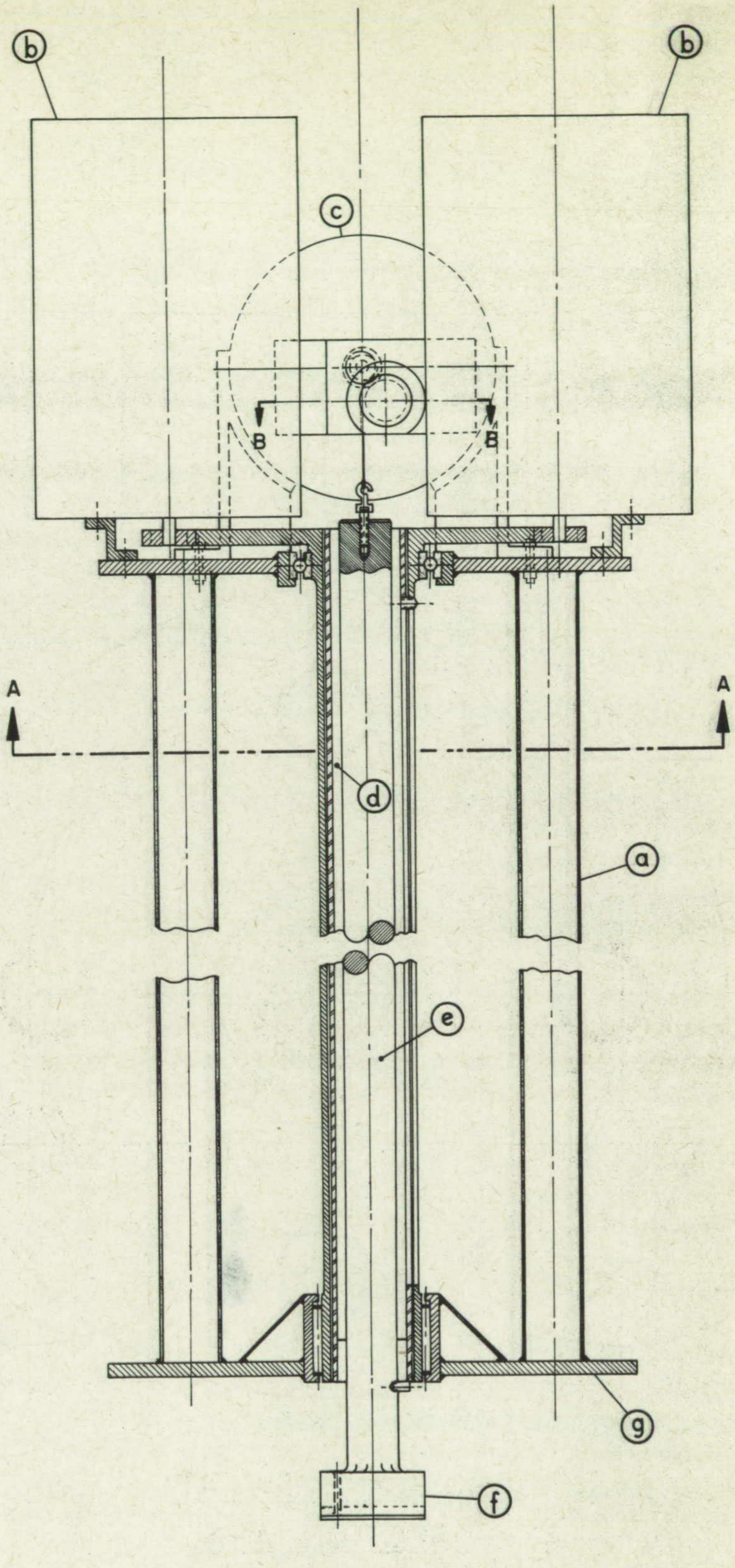
e - inner drill stem

f - cutting heat (collector not shown)

g - base

The preliminary concepts of a drill sampler for the lunar scientific package are shown in the following three figures.

Omitted from these drawings are the sample collector, thermal probe, sample handling mechanism, torque, temperature and depth monitoring apparatus.

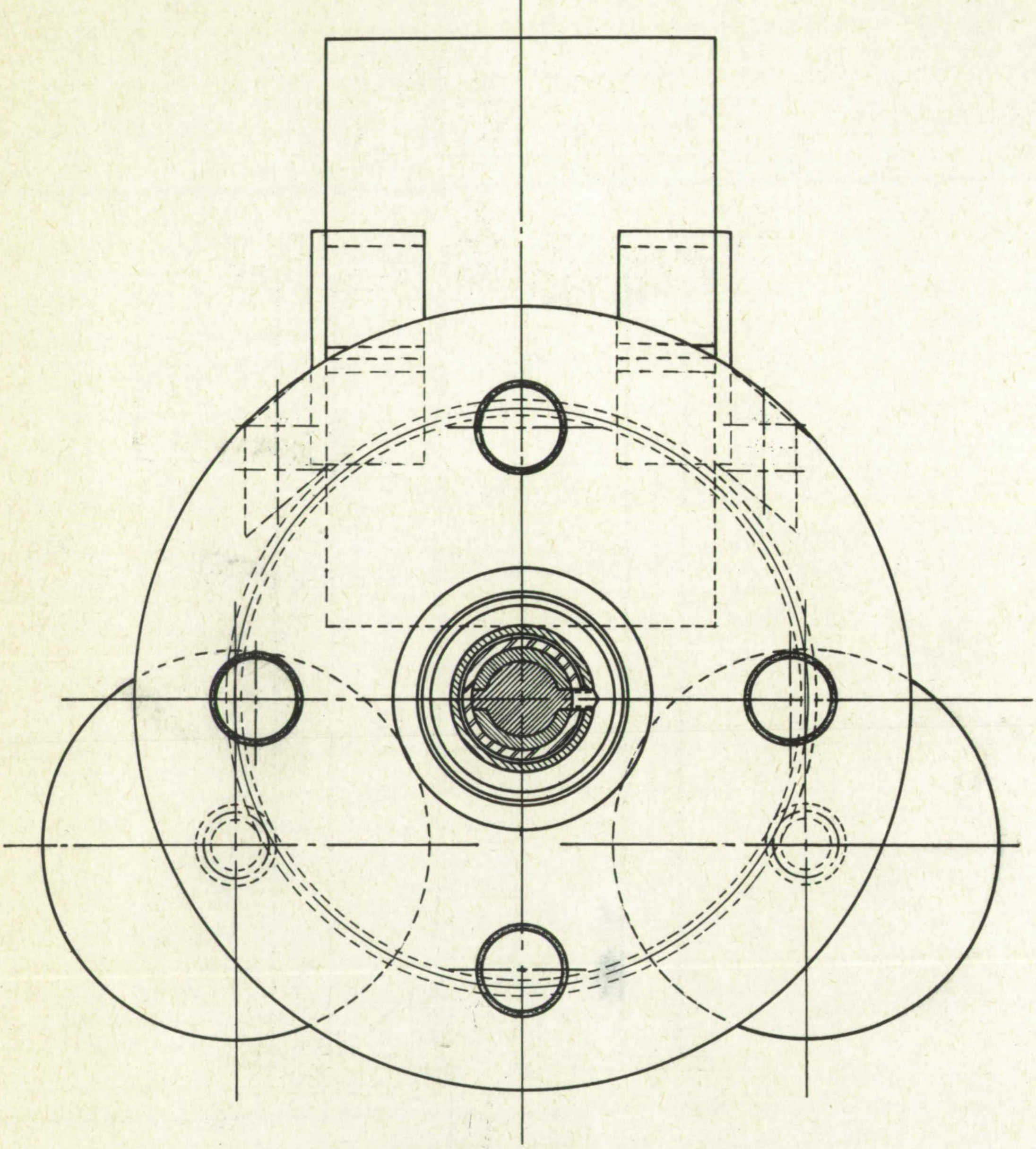


ELEVATION SECTION C–C

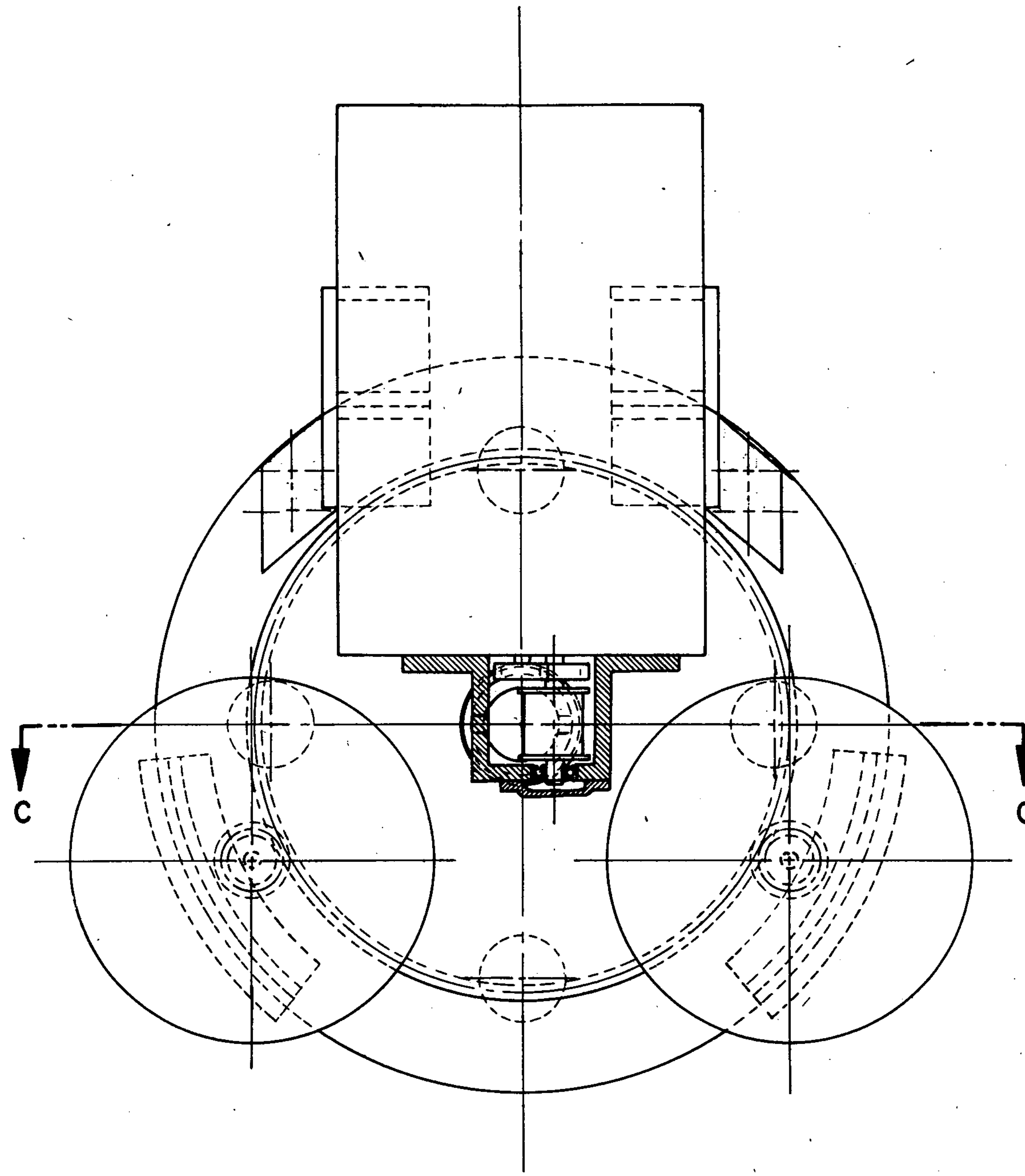
FIG. IV. 2

DRILLING APPARATUS FOR SAMPLER

(COLLECTOR NOT SHOWN)



BOTTOM VIEW AT
SECTION A-A
FIG. IV.3



TOP VIEW WITH SECTION B-B

FIG. IV. 4

IV.3.3 Fluorescence Spectrometer and Diffractometer. Near the base of the sampling apparatus an X-ray fluorescence apparatus is mounted to analyze the material obtained from the lunar surface and subsurface. The sample must be prepared in powder form and transported from the drill to the analysis apparatus by appropriate mechanical devices not illustrated in this preliminary document.

Although an X-ray fluorescence instrument has not been built for lunar application, a few brief studies have been made by industry concerning the probable characteristics of the apparatus for this application. One such study indicates the following:[1]

Weight: 40 lbs

Power requirement: 30 watts - $2\frac{1}{2}$ watts for tube

Telemetry bandwidth required: 5 to 10 Kc

Volume: 4" x 12" x 15" in instrument case

Sampling time: 1 hour per analysis

Another study[2] indicates that the operating temperature range for this equipment is between -20^{o} to $+60^{o}C$. Certain laboratory instruments have been tested at 20 g's for 5 to 10 minutes without apparent damage.

Where an auxiliary power supply is available, as in the roving vehicle, a combined X-ray diffractometer and fluorescence spectrometer may be used to obtain much more information from the lunar sample than is possible with the fluorescence apparatus alone.

The function of the X-ray diffractometer is to perform a chemical and crystallographic analysis of the material of the moon's surface or crust. This is accomplished by obtaining the usual Debye-Scherrer powder patterns of the sample from the drill sampler. The function of the fluorescence spectrometer

[1] North American Phillips Co.-private communication to W. Cunningham, NASA (1959).

[2] Applied Research Labs - private communication to W. Cunningham, NASA (1959).

is to identify chemical elements in the sample by observing their fluorescence spectra induced by proper wavelength incident radiation. Both functions, diffraction and fluorescence spectrometry, are performed by a single instrument represented in principle by Figure IV.5. Both the diffraction testing and the fluorescence spectrometry are nondestructive tests and produce no vaporization of the sample, thus avoiding any trace of contamination of the moon's "atmosphere" or of the instrumentation in the lunar vehicle.

The instrument shown in Figure IV.5 operates as a diffractometer during about one-half its duty cycle. In this case the tantalum filter M is in the incident beam R_1 which gives approximately the monochromatic incident x-radiation (tungsten K-alpha) required for the Debye-Scherrer analysis of sample S . In the other one-half of the duty cycle the device operates as an x-ray fluorescence spectrometer in which case the tantalum filter M is out of the incident beam. The heterogeneous radiation is incident in beam R_1 upon the sample S , resulting in fluorescent radiation, beam R_2 , which is analyzed by the crystal A automatically placed in the proper Bragg angle position during some part of its rotation.

A high atomic number metal, tungsten with Z=74, is used as target material in order to obtain sufficiently energetic X-ray photons (that is, sufficiently short wavelengths) to excite fluorescence in as many elements as possible. In principle the excitation of tungsten by 70-kilovolt electrons in the x-ray tube makes it possible to induce fluorescence in those elements which have an atomic number less than that for tungsten. The short wavelength limit for x-rays produced by 70-kilovolt electrons is 12400/70000 = 0.18 A. Since the tungsten K-alpha$_1$ radiation is at 0.209 A and the tungsten L-alpha$_1$ radiation is at 1.47 A, both wavelengths will be present with good intensity.

The rotating analyzing crystal A of Figure IV.5 is a single crystal so that the interplaner distance must be properly chosen to secure analysis of the fluorescent radiation incident upon it. For a complete analysis of sample S, several crystals at least must be used. Hence, A must be a combination of crystals so arranged that all crystals are bathed in the fluorescent radiation or else several of the fluorescence spectrometer units (of which one is schematically represented in Figure IV.5) must be placed around the rotating sample S. It is probably necessary, for example, to use a crystal of gypsum with crystallographic spacing 7.6 A to measure any aluminum K-alpha fluorescence radiation (8.3 A).

X-RAY DIFFRACTOMETER AND FLUORESCENCE SPECTROMETER

(Figure IV.5)

F, filament of x-ray tube; T, tungsten target of x-ray tube; R_1, x-ray beam from tungsten target T incident upon powder sample S; S, powder sample of lunar crust material prepared from drill (S is represented as a cylindrical sample continuously rotated, about 1 rpm, in a cylindrical geometry Debye-Scherrer x-ray diffractometer, using the back-reflection method with diffracted beam represented by R_2; a flat sample may be used instead of cylindrical sample); C_1, circle of travel for G-M or scintillation detector for diffracted x-rays; M, slowly rotating disc (1 revolution per several minutes) which is a combined tantalum filter for x-rays and an incident beam monitor (zinc sulfide coating which fluoresces due to incident beam, with fluorescent output dependent upon incident beam intensity). The tantalum filter acts as a monochromator for x-ray diffractometer but is out of incident beam for each half revolution of M to permit heterogeneous radiation from target T to be incident upon sample S for use of instrument as fluorescence spectrometer, fluorescent radiation from sample S being represented by beam R_3; A, rotating (about 1 rpm) analyzing single crystal with rotation axis perpendicular to rotation axis of S; C_2, circle of travel for G-M or scintillation detector for fluorescent radiation which is Bragg-reflected as beam R_4, the glancing angle theta and the reflection angle twice theta are automatically coupled in the theta - twice theta relationship; P, phototube monitor viewing zinc sulfide screen on M; D, lead shield; B, Soller slit.

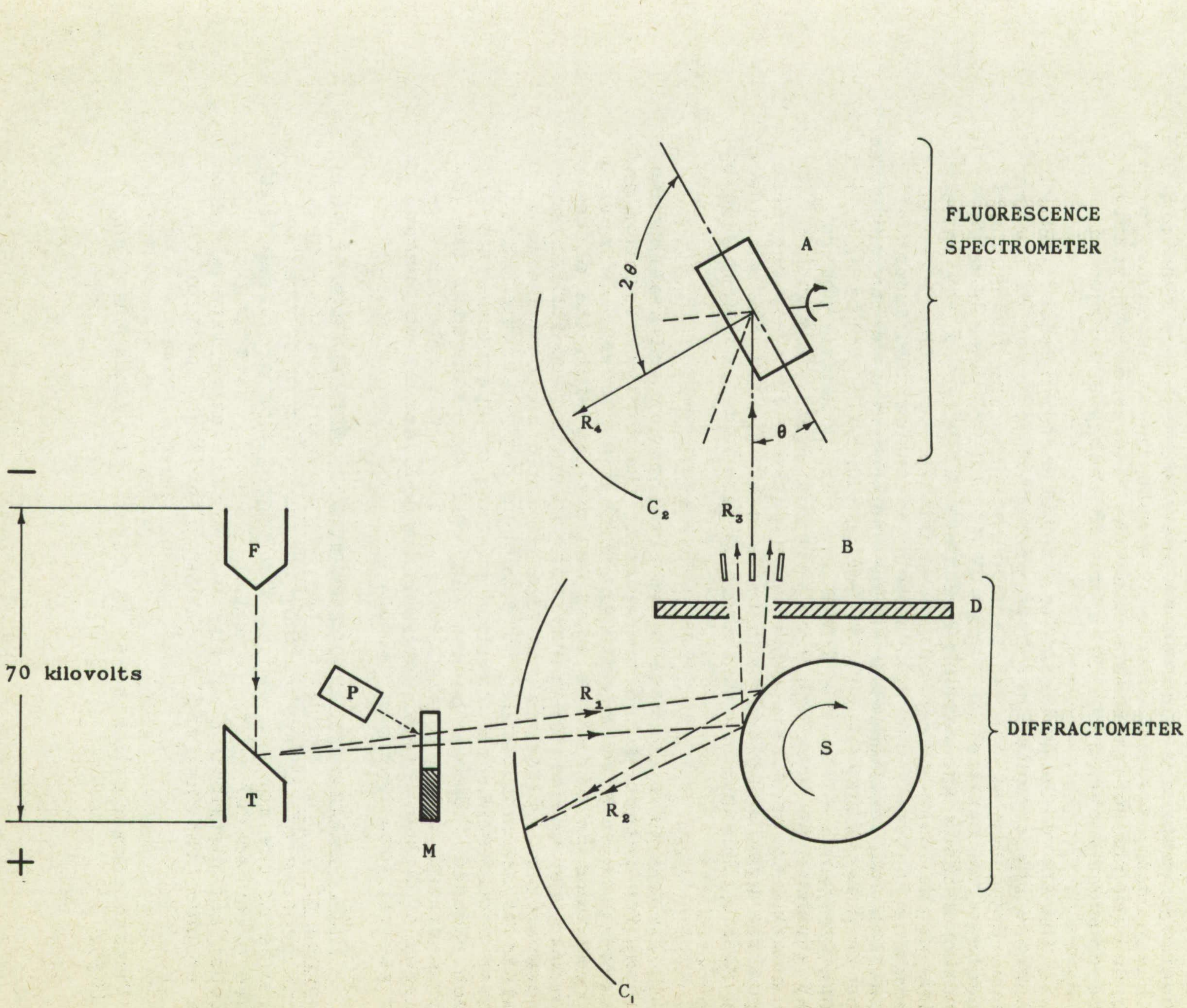


FIGURE IV.5 SCHEMATIC DIAGRAM OF ESSENTIAL FEATURES OF X-RAY DIFFRACTOMETER AND FLUORESCENCE SPECTROMETER

The power requirements for the 70-kilovolt tungsten target x-ray tube are not small. Assuming 10 ma of target current, about 750 watts are required to operate the x-ray tube and instrumentation; assuming that a 20-minute run is necessary during which both the diffractometer and the fluorescence spectrometer are completely cycled, 0.25 kw-hrs of energy are required.

The above discussion of the x-ray diffractometer and fluorescence spectrometer has been presented in terms of conventional detection of the diffraction patterns and fluorescence spectra (see caption for Figure IV.5). Other more novel detection methods are possible including: 1) Direct x-ray excitation of the image screen of a television pickup tube. The determining factor is the resolution. A considerable amount of research and development would be required in order to determine the final feasibility of this detection scheme; 2) Use of a light amplifier panel. This is a 2 x 2 inch, approximately square, array sensitive to both light and x-rays. Again, in principle, it is possible to present a diffraction pattern or x-ray spectrum upon the light panel and then to transmit the information directly by radio. This possible detection is also under consideration by RCA[3].

The use of a complete image presentation and transmission scheme for detecting and transmitting the x-ray diffraction and fluorescence spectra is quite attractive in concept. If such a scheme could be devised it would multiply many-fold the effectiveness and use of the x-ray diffractometer and fluorescence spectrometer. Patterns and spectra could be produced in a matter of seconds instead of something like one-third hour. Power requirements per complete pattern or spectrum would be much reduced. Simple changes only would be required in the instrumentation shown by Figure IV.5. Thus, the rotation frequency of M and crystal analyzer A would have to be speeded to several rotations per second.

<u>IV.3.4 Gamma-ray Spectrometer for Determination of the Amounts of U, Th, and K^{40} on the Moon</u>. An interesting experiment because of its applicability to the question of the heat balance of the moon as well as the segregation processes that have occurred during its formation, is that concerning the amount and distribution of the moon's radioactive materials such as uranium, thorium, and potassium. The major part of the

[3] Private communication to A. H. Weber, St. Louis University, 1959.

radioactively generated heat of the earth is produced by the abundant radioactive energy available from the decay of uranium, potassium, and thorium. There is apparently a considerable concentration of these materials near the earth's surface, brought about, perhaps, by the segregation processes attending the earth's evolution. That there is a concentration of the radioactive elements is evidenced by the fact that if their near-surface density were to be continued uniformly through the globe there would be more heat produced than has been observed.[4]

The direct radiometric assay of uranium, thorium, and potassium in a sample has been possible for several years through the use of a gamma-ray scintillation spectrometer.[5] The apparatus consists essentially of a source container, shield, absorber, photomultiplier tubes and pulse height analyzer and counter.

A thallium-activated sodium iodide crystal is used to convert the absorbed energy of the gamma rays emitted by the sample into light, which can be detected by the photomultiplier tubes. Either of two methods may be used in the spectral analysis. First, a count may be made of the total number of gamma-rays having energies above each of several energy levels. Counts per energy band could be calculated on the earth. Second, multi-energy channel analysis in the lunar vehicle and transmission of the counts per unit time per channel to the earth could be made. The key energy lines are 1.76 Mev from Bi^{214}, a daughter product of U^{238}, 2.62 Mev from Tl^{208}, a daughter product of Th^{232}, and 1.46 Mev from K^{40}.

In attempting to obtain a direct assay of the concentrations of uranium, thorium, and potassium, enough source material should be obtained to assure sufficient counts for analysis, while avoiding a surplus which will swamp the photopeaks from Compton scattered radiation within the source. In an automatic system, the sample will be passed through an annular space surrounding the crystal.

The pertinent data for the gamma-ray scintillation spectrometer are:[6]

[4] Jakosky, J.J.; Exploration Geophysics, Times-Mirror Press, Los Angeles, Calif.

[5] Bulletin of the Geological Society of America, Vol. 67, pp. 405-12, April 1956 (Hurley).

[6] Based on communication from Texas Instrument Co. to J. Bensko, ABMA, 1959.

Sodium iodide crystal	1 - 3 lbs
Circuitry	6 lbs
Sample unit	6 lbs
Total Weight (approx.)	15 lbs
Volume	1 cu ft

IV.3.5 Temperature Measurements on the Moon. Existing temperature measurements of the moon have been made from the earth by astronomical observations. These measurements were limited to the surface of the moon; theoretical deductions have been made regarding temperatures below the surface. It appears to be necessary to directly measure temperatures at various distances below the surface.

Measurements should be performed by boring holes into the ground to various depths and placing temperature sensors into them. This experiment should be carried by the stationary packet as well as the roving vehicle.

In each location six measurements should be planned. These may be made at the following depths: one surface measurement and five measurements at, say, 2 cm, 4 cm, 8 cm, 16 cm, and 32 cm depth.

In each location the sensor has to be protected from direct radiation. For the surface measurement the protection should consist of a hemispherical shield, the internal surface of which is black with respect to infrared radiation, and the external surface of which reflects 7% solar radiation. The sensor will be placed on the surface of the moon in the center of the hemisphere.

The hole should be covered by a shield having high reflectivity on both the internal and external surfaces. An approximation to such a surface would be a shield which is covered internally by gold or silver and externally by Rokide A.

The shields described apply to exposed holes. In the event that the holes are located underneath the vehicle, all shields should possess surfaces which are highly reflective in the infrared region.

IV.3.6 Lunar Gravimeter. The lunar gravimeter experiment uses a modified "Earth Tide" gravimeter to obtain some measure of the moon's elasticity. The "Earth Tide" gravimeter, as used at permanent sites on earth, weighs approximately 250 pounds. It has a sensitivity of one-half microgal. This is a "zero length" spring type gravimeter mounted on a stable platform (concrete pier) to eliminate the effects of wind and other disturbances. Temperature stability is quite critical, and the laboratory model is kept to within 0.01°C with a mercury thermostat and electric heating units.

According to a manufacturer,[7] the basic instrument weight is about 30 pounds excluding electronics and temperature control apparatus. For the lunar instrument, a lever micrometer and capacitor would be used rather than a photocell to detect the beam deflection. Likewise, replacement of the mercury thermostat is presently being considered. A second model under study by the same manufacturer which weighs 7 pounds may have lunar applications.

If the lunar gravimeter can be made to operate at 60°C (± 0.1°C) during the daylight period and 0°C (± 0.1°C) during the nighttime period without damaging temperature hysteresis effects, a considerable advantage may be gained. Research will be necessary with different spring materials or mechanical arrangements in order to solve this problem.

While the combined effect of the sun and the moon on earth is some 300 microgals, the sun alone accounts for approximately 100 microgals. A sensitivity of 1.0 microgal on the moon would be desirable to detect the crustal response to the solar field.

To summarize, it seems that this instrument can be made to have a final weight of around 50 pounds, and be able to perform satisfactorily in the lunar payloads. This weight reduction, of course, would be possible only after a rigorous research program in materials and thermal control.

IV.3.7 Seismic Investigations. Measurements with seismic apparatus will give information concerning the subsurface character of the moon; make possible the monitoring of meteorite

[7] Le Coste & Romberg Co.; private communication of Mr. Le Coste to J. Bensko, ABMA, September 1959.

impacts; and may give some indication of the effects of sudden temperature changes on the moon's surface. A series of seismic measurements constitutes one of the most versatile experiments that can be conducted with a single apparatus in a lunar vehicle.

In addition to the investigations outlined above concerning the moon's physical characteristics and environment, seismic apparatus may also be used as an important accessory in the sampling program to determine homogeneity of the vertical section.

The moon has been regarded by most workers as being a cold body without the ability to undergo major structural change. Apparently during the previous history of the moon heating has occurred, enough, at least, to bring about large-scale surface displacements. Several hundred earthquakes per month, mostly small ones, can be recorded with a seismograph on the earth's surface. A soft landing lunar vehicle equipped with a recording seismometer should, in a period of several lunar days, determine whether or not seismic activity exists there. Since our present knowledge indicates that a perfectly cold earth would not experience major earthquakes, a lack of seismic activity on the moon would indicate that the radioactive heat being generated at depth is not of sufficient magnitude to cause a melting or plastic flow of sizable portions of the moon's interior.

The existence of a fluid core in the moon is a question of great interest in connection with the study of the moon's magnetic field. Data obtained from the use of seismic techniques in conjunction with magnetometer surveys on the moon may give indirect information about the relationship between the earth's magnetic field with a fluid core.

A currently accepted theory on the formation of a crustal zone on earth relates it to a phase change. The same phase conditions on earth should not exist on the moon since it has a different gravitational attraction, density, and thermal history. If a crust is found on the moon through the use of seismic apparatus, this discovery may help to suggest a new theory concerning the formation of the earth's crust.

Investigations of near subsurface structural conditions on the moon will produce a store of knowledge applicable to manned exploration of the moon. The existence of large underground crevices or discontinuities may be found with a modification of a seismic apparatus already developed by a major company.

A variety of seismometer designs based on different principles are found in the literature. The sensing element of a seismometer presently being studied by Lamont Geological Observatory for NASA can be made as small as one cubic inch. The measuring and recording apparatus can be placed in a container 8 inches in diameter and 5 inches in height. Power requirement is about 2 watts. For a study of subsurface conditions in the sampling area, a four-amplifier system with power supply and geophones weighing around 7 or 8 pounds could be used, requiring 2.4 watts maximum power. The amplifier and power supply volume would be approximately 6" x 6" x 12". Measuring depths of 200 - 400 feet could be made with two or three microphone stations placed by the roving vehicle in line over 300 to 500 feet. Microphones would weigh 3 to 4 pounds.

Information received in the microphones would be in the 100 cps to 500 cps frequency range. The information could be stored on tape or telemetered directly.

The energy source could be a shaped charge utilizing fast explosives which will eliminate the need for burying the charge. Measurement time would be on the order of 0.1 to 0.2 seconds.

The same instrument could be used to record moon tremors over long periods of time.

IV.3.8 Mass-Ion Spectrometer. In considering the problem of analyzing any tenuous lunar atmosphere, it is clear that the outgassing of the lunar instrument package presents a problem. Since the instruments will be operating in extremely low pressure environments, any steady and continuous outgassing of the materials would create artificial atmospheres, thus rendering the measurements made with instruments such as mass-ion spectrometers and pressure gauges quite fallacious. Hence, the following procedures are recommended:

(1) All instrumentation, including the landing vehicle in its entirety, should be vacuum conditioned in earth laboratories before installation in the launch vehicle. If all parts are carefully cleaned and then heated with infrared lamps in high vacuum for periods of several days, they will be denuded of practically all adsorbed and absorbed gases and vapors, including water vapor. Following such treatment the various parts of the vehicle and its instruments may be exposed to atmospheric pressure without too much re-adsorption or re-absorption, provided water vapor is excluded as much as possible. Metal parts which have been outgassed in vacuum may be handled with clean, lint-free

gloves, without serious contamination. It follows, of course, that such vacuum outgassing treatment should precede as closely as possible the actual launching operation.

(2) During the vacuum heating and conditioning procedure, laboratory tests can be made with the instruments undergoing the conditioning. Thus the pressure gauge, for example, can be studied under heat cycling equivalent to the lunar day-night temperature cycle and background correction curves may be determined.

(3) Finally, it may be necessary to study the outgassing of the lunar instrument package when on the moon, since it may be impossible to eliminate occluded gases sufficiently beforehand. The high vacuum and high (lunar-daylight) temperature of the moon form an ideal outgassing environment and it should be possible to follow the progress of outgassing with the payload instruments.

A mass-ion spectrometer is proposed that will 1) measure atomic and ionic content of extremely rarified gases such as may be expected near the lunar surface by a charge-accumulation pulse-counting technique rather than by current measurement; 2) measure both atoms and ions in the same instrument using an on-off voltage pulse to energize or de-energize an auxiliary electron gun (for "on", atoms and ions are measured; for "off", only ions are measured); 3) measure both positive and negative ions in the same instrument using an alternating retarding potential on the collecting electrodes, alternate positive and negative pulses yielding pulses of negative and positive ions respectively; 4) use a compact and lightweight permanent magnet to produce magnetic dispersion of the ions and so differentiate the various chemical elements or compounds in the gaseous environment; 5) use an electric field to eliminate the incoming velocity of the atoms or ions; and 6) use a standard ion gauge to monitor the pulse-counting mass-ion spectrometer.

A Redhead gauge is recommended because of low power demand, high sensitivity, no inherent limitation on minimum pressure to be measured (less than 10^{-11} mm Hg), and its quick cleanup of residual gases. The Redhead gauge should be provided with a simple electrostatic ion trap to remove ions from the beam being sampled.

On the lunar surface, one of the mass-ion spectrometers should be arranged also as a probe projecting from the main body of the lunar vehicle in order to minimize background due to gases from the vehicle.

Figure IV.6 illustrates schematically the essential features of the proposed pulse-counting mass-ion spectrometer employing charge-accumulation pulse-counting technique. The notation used is as follows: v_o, entering speed of atom or ion; C, collimating slits; G, electron gun (10 ma, 100 volts) for on-off operation to record both atoms and ions; A, accelerating electrode (alternating potential for both positive and negative ions); B, magnetic dispersion and E, electrostatic dispersion with both flux densities perpendicular to plane of page; D, detector consisting of array of small charge-accumulation plates $Q_1 \ldots Q_n$ connected to R-C circuits which may be discharged after variable and adjustable time of charge accumulation by electronically closing switch S which causes discharge of capacitor and in turn creates a cathode-to-grid potential in turn causing a pulse in triode finally yielding an amplified pulse which can be measured and counted. All switching of voltages is done electronically.

The pertinent data for the mass-ion spectrometer are:

Weight: 8 pounds including mass-ion spectrometer without power supply.

Volume: About 45 cubic inches for the mass-ion spectrometer without power supply.

Power: About 20 watts during operation

Sampling frequency: On command.

The pertinent data for the Redhead ion-gauge monitor are:[8]

Weight: About 8 pounds including all electronics.

Range: 10^{-5} to less than 10^{-11} mm Hg in decades.

Volume: About 20 cubic inches.

Power: Less than 3 watts on 10% duty cycle.

Acceleration resistance: Tested under 100 g longitudinally, 60 g transversely

[8] N.R.C. Equipment Corp., Newton, Mass.; private communication to A. H. Weber, St. Louis University, August 1959.

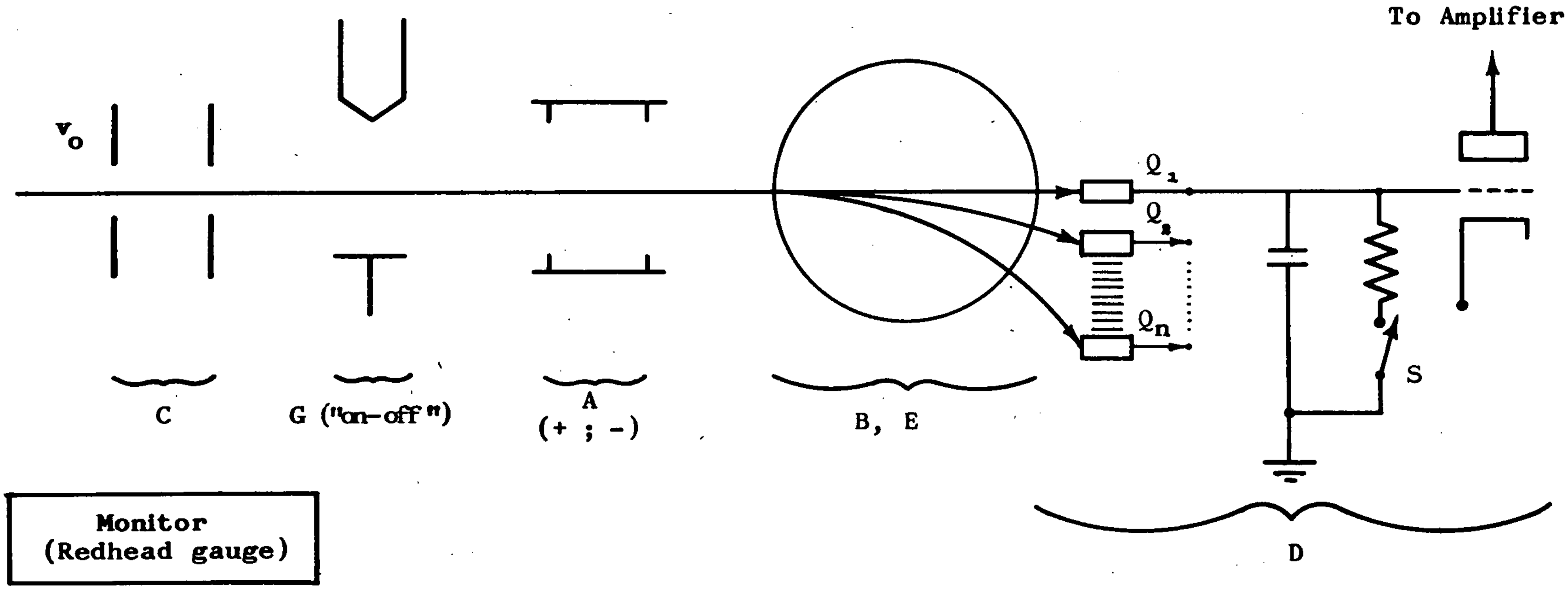


FIGURE IV.6 PROPOSED MASS-ION SPECTROMETER WITH CHARGE INTEGRATION FOR EXTREMELY LOW INTENSITIES

IV.3.9 Plasma Probe Experiment. The purpose of the plasma probe experiment is to measure directly the plasma densities and motions in the solar system as functions of position and time.[9] The apparatus is essentially a Faraday cage with the charge collector shielded by four grids. A diagram of the probe is given in Figure IV.7.

The purpose of the collector and the electrometer circuit is to measure the current of positive ions. The grid system prevents the entry of positive ions with energies below a threshold provided by the system, and it minimizes the effects which might interfere with a measurement of the positive ion current. In particular, G_1 with its negative potential relative to the collector prevents the departure from the collector of electrons produced by photoelectric emission and secondary emission by ion collection, and also prevents the arrival at the collector of electrons from the outside. The migration of electrons to or from the collector would be damaging because the measurement of ion currents is desired. The third grid G_3, because of a positive square-wave imposed on it, repels ions which do not have sufficient energy to overcome the positive potential presented by G_3. Thus, the potential applied to G_3 modulates the collector current, and the degree of modulation depends on the number of ions above the threshold energy.

The plasma probe, because it accepts ions from a limited direction, may be used under some conditions to study the mass motion of the plasma. A survey of more than one direction can be achieved either by changing the direction in which one device faces, or by providing more than one probe.

The power supply for the probe electronics will require an input of 1.0 amp at 12.0 volts. The weight of one system, exclusive of the source power and the telemetry, will be approximately 2 pounds. The volume is approximately a right circular cylinder roughly 6 inches in diameter and 4 inches in depth. The signal output will be 0 = 5.0 VDC at 200 ohms.

IV.3.10 Magnetometers. The moon's magnetic field, if it is measurable, is likely to be very weak. Therefore, a sensitive, lightweight instrument will be necessary. Even the influence of the stationary packet or roving vehicle which carries the

[9] M.I.T. Informal report on Rossi experiment to R. Shelton, ABMA, September 1959.

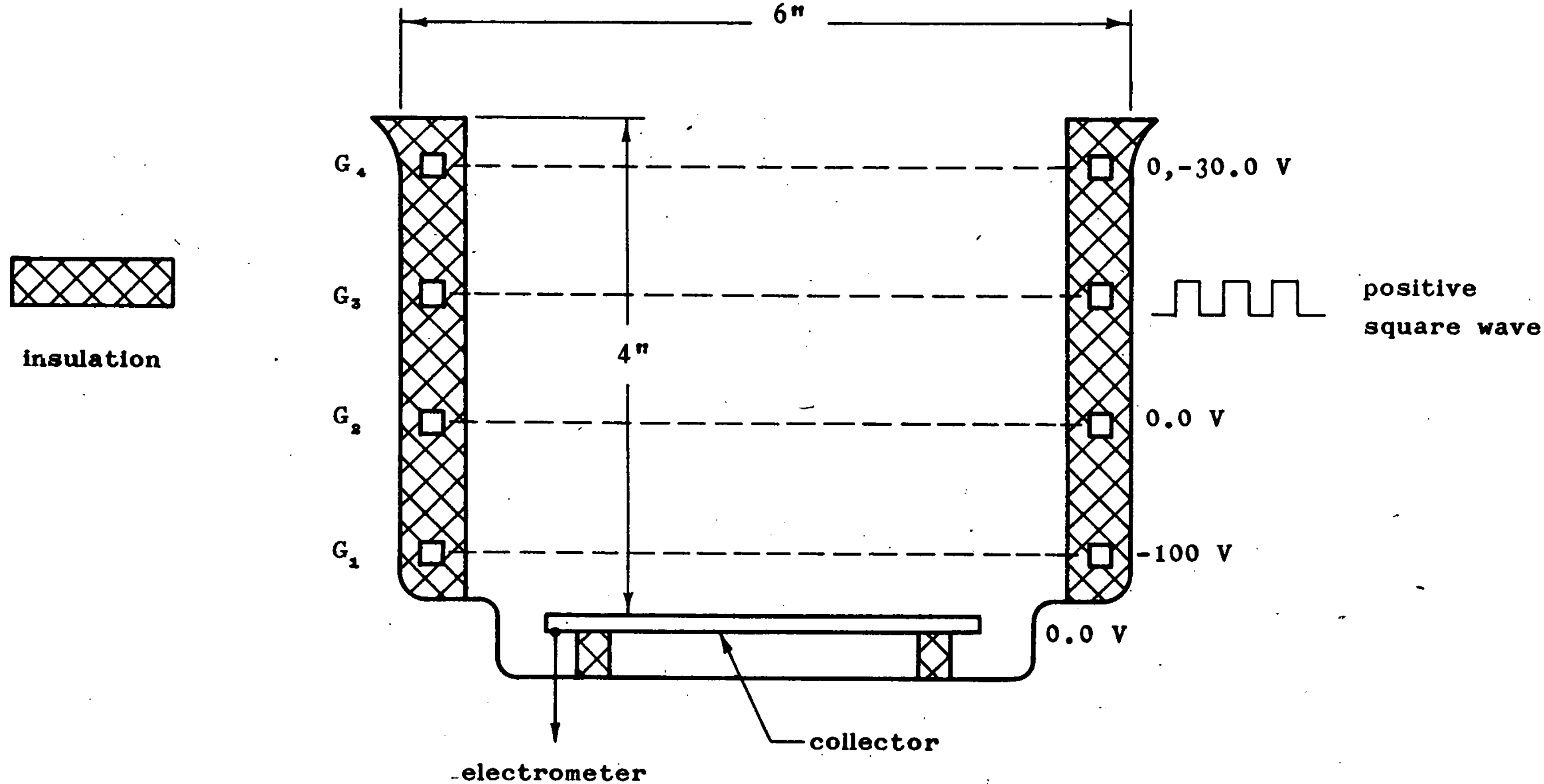


FIGURE IV.7 DIAGRAM OF THE PLASMA PROBE

instrument is likely to disrupt the measurement. For this reason, care must be taken to isolate the sensors from the effects of the vehicle components. Several instruments appear competitive for the scalar measurement. These are the proton precession magnetometer, the rubidium vapor magnetometer, and the helium vapor magnetometer. For the magnetic field direction measurements a simple fluxgate instrument is recommended. The rubidium vapor magnetometer is already being considered at NASA laboratories.

The metastable helium magnetometer is a rugged lightweight and accurate magnetometer with high sensitivity. The device consists of an infrared source provided by a helium discharge tube, a second helium discharge tube through which the infrared radiation from the first tube is passed, and an infrared detector. A radio frequency sweep signal is imposed on the second tube by means of a coil. The sweep frequency at which the infrared absorption is a maximum is a function of the external magnetic field (2.8 megacycles per orested).

The high operating frequency of 2.8 mc per orested (28 cycles per gamma) is particularly favorable in measuring extremely low fields likely to be encountered on the moon.

An estimate of the payload weight of a helium vapor magnetometer is as follows:[10]

Magnetometer head	1 lb
Electronic equipment	1 lb
Digitizer and storage	1 lb
Probe shell	1 lb
Total weight	4 lb

Power requirements will be about 4 watts, divided as follows:

Magnetometer helium lamp	2 watts
Magnetometer electronics	1
Digitizer and storage	1
Total	4 watts

[10] Texas Instrument Co. - Proposal to NASA, 24 March 1959.

IV.3.11 Measurement of the Electric Field on and Near the Moon.[11] One of the physical quantities that may be of importance on and near the moon is the electric field. If the surrounding medium of the moon is not electrically conductive, a conventional electric field meter could be used. The field meter consists of two electrodes which are electrically connected through a parallel RC network and are mechanically maneuverable so that one electrode is alternately exposed to and shielded from an external electric field by the second electrode. The latter is generally grounded. The external field terminates on the fixed electrode and charges it by induction. As the shielding electrode interrupts the field, the induced charges flow to ground via the RC network. Depending on the time constant of the RC network and frequency of interruption of the field, either the maximum charge on the ungrounded electrode or its rate of change is directly proportional to the unknown field.

The typical input impedance of conventional field meters in the voltage mode exceeds 10 ohms, and in the current mode is of the order of one megohm. The resistivity of tropospheric air, in which such meters are often employed, is very large by comparison. The resistivity in the ionized layer of the upper atmosphere and perhaps of the moon are orders of magnitude smaller than the typical minimum input impedance of the field meter. Thus, the external conductivity would short-circuit the meter.

If there are ionized layers around the moon, a conventional instrument cannot be used. If two mutually insulated conductors separated by a considerable distance d, are placed in a conducting medium, each will assume the potential of its immediate surroundings. If a field E exists in the medium, the potential difference V between the two probes in principle will be $V = Ed$. Because a high input impedance is necessary for this measurement, an electrometer tube should be used as the input to an amplifier. Effects of the geometry of the isolated conductors and the vehicle must be included in the analysis of the data.

The effective area of the probes would have to be worked out before any weight values could be given. If transistors are used for outer stages, the packages can be kept relatively small, about 12 cubic inches, with a weight of about 8 ounces, and one watt of power.

[11] ITEK Corp. (22 Oct 58); Cornell Aeronautical Laboratory, Inc. (Oct 1958) - proposals to ABMA.

TABLE IV.1 - ESTIMATED CHARACTERISTICS OF SOFT LANDING PACKAGE INSTRUMENTATION

INSTRUMENT	WEIGHT - lb Vehicle		SIZE OR VOLUME	OPERATING POWER Vehicle		LIFE-TIME	REMARKS
	Stat.	Rov.		Stat.	Rov.		
TV system	35	35	cyl 10" dia x 8"	20	20	Lunar day	By command
X-ray spectrometer	40	—	4" x 12" x 15"	30	—	"	"
Combination diffractometer & spectrometer	—	50	4" x 12" x 15"	—	*750	"	"
Drill sampler	60	60	200 cu in	200	*1200	"	"
Gravimeter	50	—	18" x 15" x 15"	6	—	2 cycles	Continuous
Seismograph	8	8	cyl 8" dia x 5" + extension	2	2	"	"
Thermal probes	1	1	—	1	1	"	"
Mass & ion spectrometer & pressure monitor	16	16	65 cu in	23	23	Lunar day	By command
Plasma probe	2	2	6" x 4"	12	12	Lunar day part night	"
Magnetometer	4	4	Sensors 2"x2"	4	4	Lunar day	"
Electric field	0.5	—	12 cu in	1	1	L.d.pt.night	"
γ-ray spectrometer	15	15	cyl 5" x 6"	2	2	Lunar day	"
Biological experiment	5	5	40 cu in	2	2	"	"

* if auxiliary power is used

IV.3.12 Biological Measurements. A device is presently being designed[12] which will inoculate a number of selected sterile culture media with material from the lunar surface and subsurface. Telemetered changes in acidity and turbidity are employed to detect the growth of micro-organisms in the culture.

An apparatus can be built which will collect a small sample of surface material, heat it, and determine the emission of spectral lines indicative of carbon compounds. This device can also be adapted for the examination of subsurface samples if it is combined with the sample collector.

Existing automatic devices for the chromatographic analysis of mixtures, although bulky, could be miniaturized and adapted to the analysis of soluble components of lunar samples. The development of such an apparatus will be of equal importance in the selenochemical and organic studies.

IV.4 OPERATIONAL REQUIREMENTS FOR INSTRUMENTS IN LANDING MISSION

A landing at the selected site at the beginning of a lunar day is desirable. This will give the maximum operating time for apparatus powered by a solar cell supply. A severely limited number of functions may be maintained during the lunar night. This, however, requires storage batteries and active heating of components. Some instruments inoperative during the lunar night can be reactivated after sunrise.

The instruments aboard the stationary packet may be divided into three separate packages according to their required operating lives. Certain instruments, accorded a high priority and requiring a long lifetime, must be protected during the lunar night to assure their operation during the night and the subsequent lunar day. These instruments are the gravimeter, the recording seismograph, and the thermal experiment. These constitute the first package.

Instruments concerned with the analysis of the lunar crust will be required to operate only during the lunar day. These instruments, constituting the second group, are the sampling apparatus, the x-ray diffractometer, the x-ray flourescence

[12]Wolf Visniac, Yale University (NASA Contract No. NSG 19-59).

apparatus and the television system.

A third group of instruments aboard the stationary packet are: the mass and ion spectrometer, the plasma probe, the particle detectors, and the magnetometer.

The first of these instrument groups may have special thermal protection. Instruments in the other two categories will not be protected during the lunar night. The solar cell bank will be arranged so that the power necessary to supply the first instrument group will be available even if part of the solar cell bank is destroyed. This arrangement is discussed in Chapter V.

Instruments in category two require a high operating power but will be used only for very short periods of time. Instruments in the third category are used for intermittent periods.

Operation of instruments carried by the roving vehicle will be limited to one lunar daylight period. Power will be supplied by solar cells. A large battery power supply cannot be carried on the roving vehicle due to weight requirements of the locomotion mechanisms.

In addition to the above "scientific" measurements, there will doubtless be a number of vehicle-monitoring measurements. Some of these, notably temperature, might economically be combined with one or more of the scientific experiments. Such a program might include the following:

Temperature (10 measurements)

Pressure (3 measurements)

Attitude (2 measurements)

RPM (1 measurement) (mobile vehicle only)

Voltage (2 measurements)

The total weight of the equipment for these measurements would be about 5 pounds and require about 2 watts of power.

It is expected that all measurements can be transmitted on a time-sharing basis, and that a high sampling rate can be avoided.

As the moon offers a static environment from several viewpoints, it is primarily necessary to worry about those few measurements concerned with non-static conditions. The events causing changes are:

Annual revolution of the earth-moon system around the sun

Revolution of the moon around the earth

Seismic disturbances within the moon

Cosmic radiation

Fluctuations in the magnetic field

Probably the most noteworthy event to occur regularly on the lunar surface is the transition from lunar day to night, and vice versa. There will also be gravitational effects. The time required for these effects can be estimated by reference to the lunar rotational speed about its axis; 360 degrees during 28 earth days or about 0.54 degree per hour. This means that approximately one hour would be required for occultation of the sun by the lunar horizon. It is indicated then that a sampling rate of one per hour would be adequate for thermal and instantaneous particle rate instruments. Gravitational effects would pass through extremes every 14 earth days. It can be seen that sampling rates for these measurements might well be much slower than one per hour.

Seismic disturbances, meteoric impacts and cosmic radiations represent events occurring at irregular intervals. There are several choices when considering instrumentation requirements:

Forego gathering of complete information

Integrate data electronically and lose occurence times

Monitor continuously

Utilize storage and playback

Continuous monitoring must be ruled out at present if excessive bandwidth is to be avoided. Storage and playback presents mechanical and electrical problems. Although such devices have been successfully used in satellites, a desire for the ultimate in reliability might exclude them here. If the first possibility is chosen, no further discussion is needed. If the second is chosen, special instrumentation must be devised. In any event, a time-shared measurement is strongly recommended.

IV. 5 SOFT LUNAR LANDING TELEMETRY SYSTEM

In view of the very great similarity of the proposed measuring programs of the stationary experiment and the roving experiment, it is proposed to use identical systems for the two experiments and to modify a given system to accommodate the variation in the measuring requirement.

The following assumptions were used in reaching preliminary specifications concerning the telemetry system:

(1) The measuring program shall require

  (a) one continuous channel with a response of 20 cps

  (b) one continuous channel with a response of 3 cps

  (c) seven sampled channels each with 3 cps response

(2) The television picture will be transmitted once each 30 seconds.

(3) The television picture transmission may be interrupted once each 15th to 25th frame to allow transmission of telemeter data via the television transmitter.

(4) The television video bandwidth will be the equivalent of 4 Mc for instantaneous transmission.

(5) The receiver shall have a noise figure of 2 db.

(6) The receiver input signal-to-noise ratio shall be 12 db.

(7) A radio carrier frequency of 1250 Mc shall be used.

(8) The antenna on-board the moon shall have a gain at this frequency of 13.5 db.

(9) The earth antenna shall have a gain at this frequency of 43 db. (Theoretical gain = 47 db)

IV.5.1 Modulation System. Various modulation systems were examined. Of these, PCM-FM appears to have the greatest advantage for telemetry data, other than the television. For television, an FM transmission is used.

For the PCM-FM system, the following will be required:

(1)	1 multiplexer	1/4 lb	50 mw	10 cu in
(2)	1 encoder	1/4 lb	100 mw	10 cu in
(3)	1 memory device	$1\frac{1}{2}$ lb	150 mw	100 cu in
(4)	15 Kc video bandwidth, 15 Kc RF bandwidth at 2 w radiated			
(5)	1 transmitter	7 lb	38 w	150 cu in
	TOTALS	9 lb	38.3 w	270 cu in

The system would give approximately 2% accuracy.

IV.6 ANTENNAS FOR THE STATIONARY PACKET AND ROVING VEHICLE

IV.6.1 Antenna for Stationary Packet. If a directional antenna is used on the stationary packet, it has to be directed only once towards the earth, provided the beamwidth is broad enough to cover the movement of the landing site (e.g., libration $\pm$ 7°) and the movement of the observation station (earth rotation). An antenna beamwidth of about 20 - 25° would satisfy this condition, assuming an antenna gain of about 15 - 17 db. The antenna could be pointed toward the earth with a simple two-axis servo drive actuated on command from the earth. The command link could be established prior to the antenna erection with a omnidirectional auxiliary antenna.

The diameter of a parabolic antenna would be smaller than 3 feet for 1000 Mc and smaller than $1\frac{1}{2}$ feet for 2000 Mc.

IV.6.2 Antenna for Roving Vehicle. The results of a preliminary evaluation of the antenna required on the roving vehicle showed that for the transmission of a modest measuring program, a radiated transmitter power of 10 to 60 watts is necessary. This figure is based on the following assumptions: omnidirectional antenna on the moon; 60-foot dish antenna on the earth; and carrier frequency 1000 to 2000 Mc/sec.

The corresponding operating power for the transmitter would be some 30 to 200 watts, which is high compared with the total power capability of the vehicle. For this reason, it may be desirable to reduce the necessary radiated power by the introduction of a vehicle antenna having a higher gain. On the other hand, a high gain antenna is inherently highly directive. A complicated scanning and tracking device would be necessary to direct the antenna toward the earth station after each movement of the vehicle. The weight and power requirements for this tracking could easily surpass the savings in transmitter requirements, and thus defeat the advantages of the high gain antenna.

For this reason a compromise is proposed. A vehicle antenna with a moderate gain and a broad pattern is aligned with the local vertical on the moon by a pendulum mechanism. The antenna pattern has a conical shape. The vertex angle of the cone is determined by the lunar latitude and longitude of the region of operation. The main lobe is always pointing to the earth, independent of the vehicle attitude or heading. This is illustrated in Figure IV.8. The beamwidth of the main lobe has to be such that the changes in the angle between the local vertical and the earth direction caused by impact tolerances, the libration of the moon, and the vehicle motion are covered. A wide range of compromises between gain and beamwidth for the main lobe is possible. A typical antenna gain of perhaps 10 db may be obtained by this scheme, and the transmitter power may accordingly be reduced by a factor of 10.

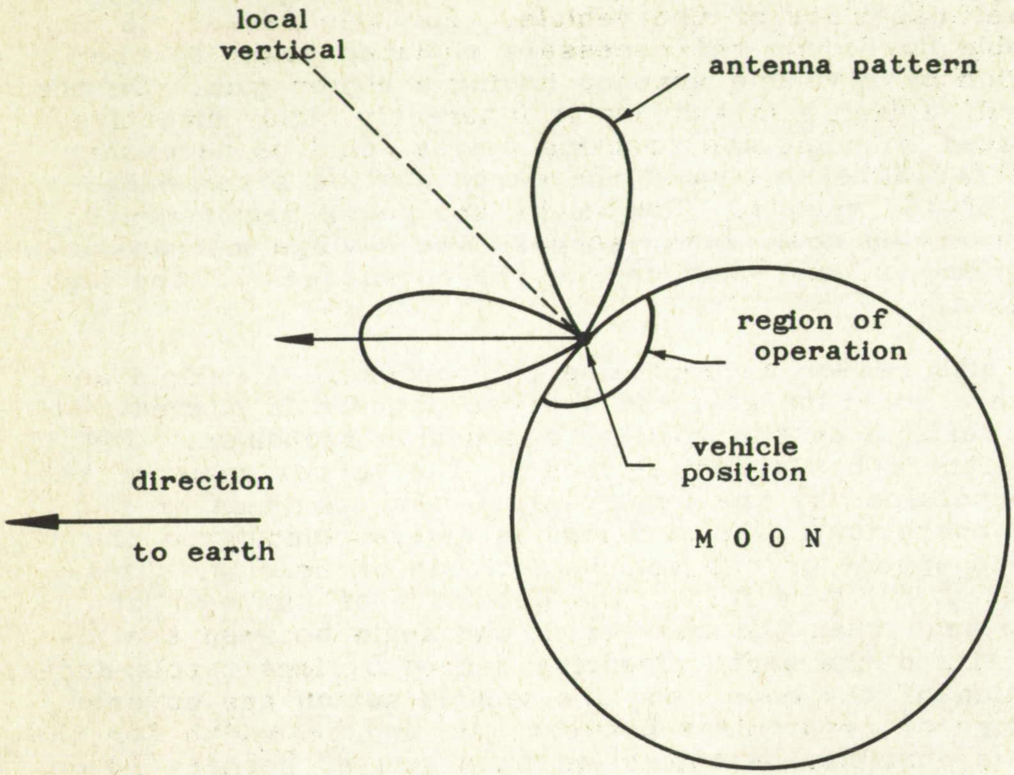


FIGURE IV.8 SKETCH OF THE ANTENNA PATTERN FROM A STATIONARY VEHICLE ON THE LANDING SITE

CHAPTER V

POWER SUPPLIES

V.1 POWER SUPPLIES FOR LUNAR OPERATION

Ideally, power supplies for lunar operation should have long operating lifetimes, high power-to-weight ratio, insensitivity to extreme temperature fluctuations, high reliability, and operating characteristics which avoid contamination of surroundings. However, from a practical standpoint, all of these ideal attributes cannot be obtained in the near future.

Nuclear power supplies probably cannot be considered for use on lunar payloads because of possible contamination of the surface with radioactive material. This reduces the choice of a power supply to systems that convert solar energy into electrical energy or that carry their own energy source.

Chemical batteries must be rejected due to the prohibitive weights required for the operating time and power level considered in this study.

Fuel cells have been considered and, based on an estimated output of 500 whr/lb, look very attractive. However, development of a reliable unit that would operate in the lunar environment appears to be some years away. Although the Army and Air Force are supporting research and development of fuel cells, it is not deemed feasible at present that plans for a lunar vehicle power supply be based on the fuel cell.

A hot-gas-driven turbogenerator to supply electric power for instruments has been considered and investigations are underway to determine when a reliable unit will be available to operate for two months in a vacuum. As with the fuel cell, the products of the chemical reaction must be expelled and would cause some contamination of the lunar surface.

Silicon solar cells appear to offer the best solution although there are severe temperature problems with this as with any other system. Satisfactory operation probably can be obtained during the lunar day but the problem of the cells surviving the $-150^{\circ}C$ temperature of the night has not been resolved.

Some preliminary tests have been performed by this agency in an effort to determine the effects of very low temperatures. Shingled cells mounted on an aluminum plate were kept at $-50^{\circ}C$ for two hours in a temperature chamber with no noticeable effect. Another sample was reduced to $-150^{\circ}C \pm 10^{\circ}C$ by suspending it above a pool of liquid nitrogen in a Dewar flask. This temperature did not damage the cells. However, immersing the cell patch in the liquid nitrogen (about $-190^{\circ}C$) did cause the cells to crack and reduced the power output of the patch by about 50%. It is planned to continue the investigation of the effects of low temperature on solar cells.

Solar cells have been used successfully on VANGUARD I and EXPLORER VI and would appear to be a good choice for a lunar vehicle except for the limitation mentioned above.

During the lunar day it is expected that the equilibrium temperature of the cells can be maintained from 100 to $110^{\circ}C$ by using an optical coating on the cells to increase the emissivity. A solar cell of 9% efficiency at room temperature will convert about 5.4% of the incident solar energy at $105^{\circ}C$. Diode losses and mismatch of cells would reduce this to 4 or 4.5% efficiency.

Protection of the solar cells from micrometeorite erosion can probably be provided by placing a thin glass directly over each cell. This slide may be optically coated to aid in temperature control of the cells. There remains the possibility of solar cell damage from meteorites so large that glass covers would offer no protection. If any cell in a series group is destroyed then the circuit is opened and the output of all cells in the series is reduced to zero. This loss can be overcome only by overdesigning the power supply. The probability of this occurring is not well known but it is proposed to add about 25% more solar cell area to compensate for this possible loss.

V.2 STATIONARY PACKET POWER SUPPLY

Preliminary power requirements of the stationary packet indicate that about 50 watts are required continuously with intermittent loads of 200 watts, 120 watts, 30 watts, 20 watts, 15 watts, and some smaller loads. These large loads would not occur simultaneously.

The 200 watts are required for driving the drill which should be capable of operating for about 5 hours. It is proposed to connect mercury primary batteries in parallel with the solar cells to meet this power requirement. Battery weight for this drill requirement is about 125 pounds.

The 120 watt load has a duration of only 20 milliseconds and this surge could also be supplied by the batteries. The mercury batteries should have the additional requirement to be of great enough capacity to permit operation of part of the equipment during the lunar night.

The next highest intermittent load is 30 watts and it is proposed that the solar bank be large enough to provide this power plus the 50 watts continuous load. This would give a total solar power requirement of 80 watts. Allowing excess area for possible meteorite damage would increase the solar cell capacity to about 100 watts.

Allowing for about 20% of the active solar cell area for spacing between the cell strips, the total solar bank area required is 24 square feet. The weight of the cells, cement and mounting tray for this area is approximately 50 pounds. This does not include the weight of the supporting structure.

In order to get reasonable efficiency from mercury batteries, the temperature should be controlled between 10 and 100°C. Based on these temperature limits, the power required to operate the drill for 5 hours and provide 20 watts of power continuously during one lunar night will require about 225 pounds of mercury batteries. Such a power supply would permit operation of all instruments during the first lunar day and 20 watts would be available during the night for heating and operating a limited number of instruments; equipment that survived the night could then be operated from solar cells the following day.

If it is determined that the solar cells cannot withstand the low temperature of the night and revive to provide power during the second lunar day, consideration should be given to the operation of the instruments from mercury batteries at a high power level for 25% or 50% of the night.

V.3 ROVING VEHICLE POWER SUPPLY

Instrumentation for the roving vehicle is virtually the same as that of the stationary packet except for the omission of the gravimeter. The power requirement of the gravimeter is small, and therefore the electric power supply for the instrumentation should be practically the same for both vehicles.

In addition to the power required for instrumentation, it is estimated that about 3 hp will be required to move the vehicle at a speed of 3 to 5 miles per hour over terrain specified in III.3. It is desirable to move the vehicle about 50 miles, requiring a minimum of 10 hours of operation.

It appears that the most practical method of providing this power would be from a hot gas turbine using a storable liquid fuel such as hydrazine. Assuming a weight of 4 pounds per hp-hr, the power supply for this requirement would be 120 pounds.

For the roving vehicle, a minimum total of 10 feet of drilling is desirable. Assuming that a 1200-watt drill can be used, it is estimated that about 2 hours of drilling will be required. It is proposed that the hot gas turbine also be used to drive the drill. This additional requirement on the turbine amounts to 3.2 hp-hrs or about 12.8 pounds of additional weight.

Thus, the total additional weight for the roving vehicle would be approximately 135 pounds.

CHAPTER VI

PROJECT IMPLEMENTATION

(S) VI.1 VEHICLE DESIGN, DEVELOPMENT, AND TESTING

The SATURN program has the first static test scheduled for early in calendar year 1960 with the booster flight test program scheduled to start early in 1961. Later flights will include the upper stages and flight tests of the complete vehicle.

Based on this flight test program and other anticipated programs, a tentative schedule has been established for the SATURN vehicle program. This schedule indicates that vehicles for the soft lunar landing missions will be available in the first half of calendar year 1963. With the presently planned missions and facilities for the SATURN vehicle, the initial rate of launching for the soft lunar landing mission will be one every four months, with a build-up to one every other month.

(S) VI.2 INSTRUMENTATION RESEARCH AND DEVELOPMENT SCHEDULE

For almost all of the proposed scientific instrumentation, NASA has either already funded its development at an appropriate laboratory or is contemplating doing so soon. If these development programs are pursued vigorously, the equipment should be ready in time for the SATURN soft lunar landing experiments.

Experiment designers should expect to deliver detailed specifications to the payload design group about two years before execution of the lunar mission.

(S) VI.3 GROUND SUPPORT AND LAUNCHING FACILITIES

Ground support facilities for the SATURN are now undergoing development. The ground support program for lunar landing and circumnavigation vehicles will be similar to that for the standard SATURN, except for some modifications to handle the guidance system, communications systems, and the final rocket stage. Of course, additional requirements will also arise for the manned circumlunar vehicle.

It is planned that the SATURN booster will be transported from Redstone Arsenal, Alabama to Cape Canaveral by barge. Figure VI.1 shows the loading, transport, and unloading of the booster. The booster will be transported overland as shown in Figure VI.2. At the launch site, the booster and upper stages will be assembled as shown in Figure VI.3.

It is also planned that the SATURN booster be recovered, reconditioned, and used again. After burnout and separation, the booster fall will be retarded by parachute. After a water impact, it will float until recovered. Figure VI.4 depicts the water recovery of the booster.

The final report of this study will contain more detailed data on the facilities and procedures for ground support and launching.

(S) VI.4 DROP TEST AND TRAINING PROGRAM

Inclusion of an electro-mechanical lunar landing simulator as well as drop tests and a thorough training program is expected to greatly increase the chances of successful mission accomplishment.

To be included will be a test of a dropped vehicle which is monitored by TV and controlled in much the same manner as in the terminal guidance phase of the lunar approach. Also, a vertical lift aircraft whose descent velocities will approximate those of the last few thousand feet near the moon can be used.

In the event that sufficient information is available from the re-entry programs to properly design the return vehicle for the circumnavigation missions, no drop or re-entry tests need be conducted for this phase of the program. However, if the proper information is not available, it is proposed that a minimum of two NIKE-CAJUN type re-entry tests be conducted along with aircraft drop tests of the re-entry vehicle.

VI.5 OBJECTIVES OF FLIGHTS

The objectives of the landing packages and the circumlunar vehicle will be discussed in turn.

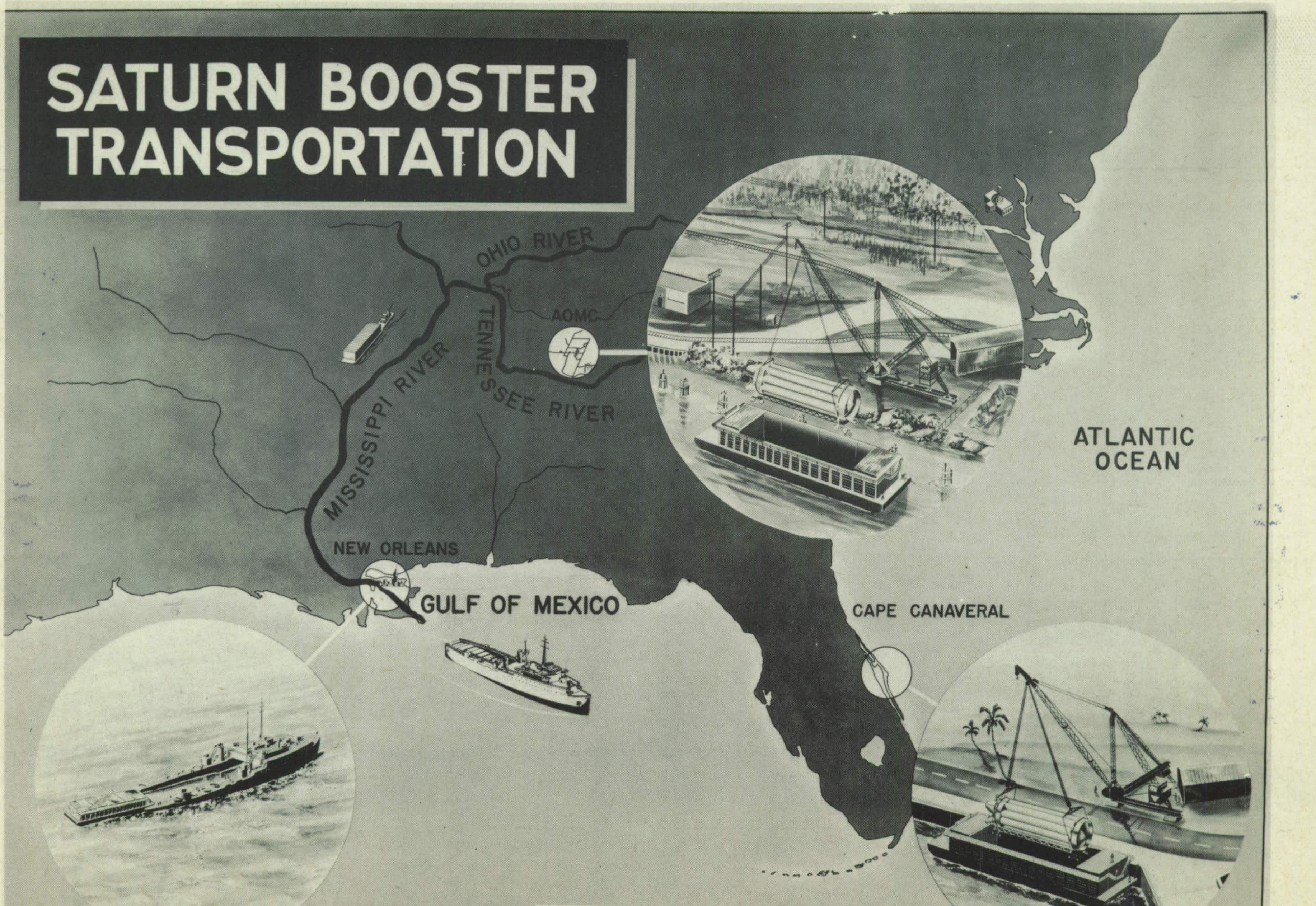

SATURN BOOSTER
TRANSPORTATION
OHIO RIVER
AOMC
TENNESSEE RIVER
MISSISSIPPI RIVER
NEW ORLEANS
GULF OF MEXICO
ATLANTIC
OCEAN
CAPE CANAVERAL

FIG. VI.1

SATURN TRANSPORTER AND TUG

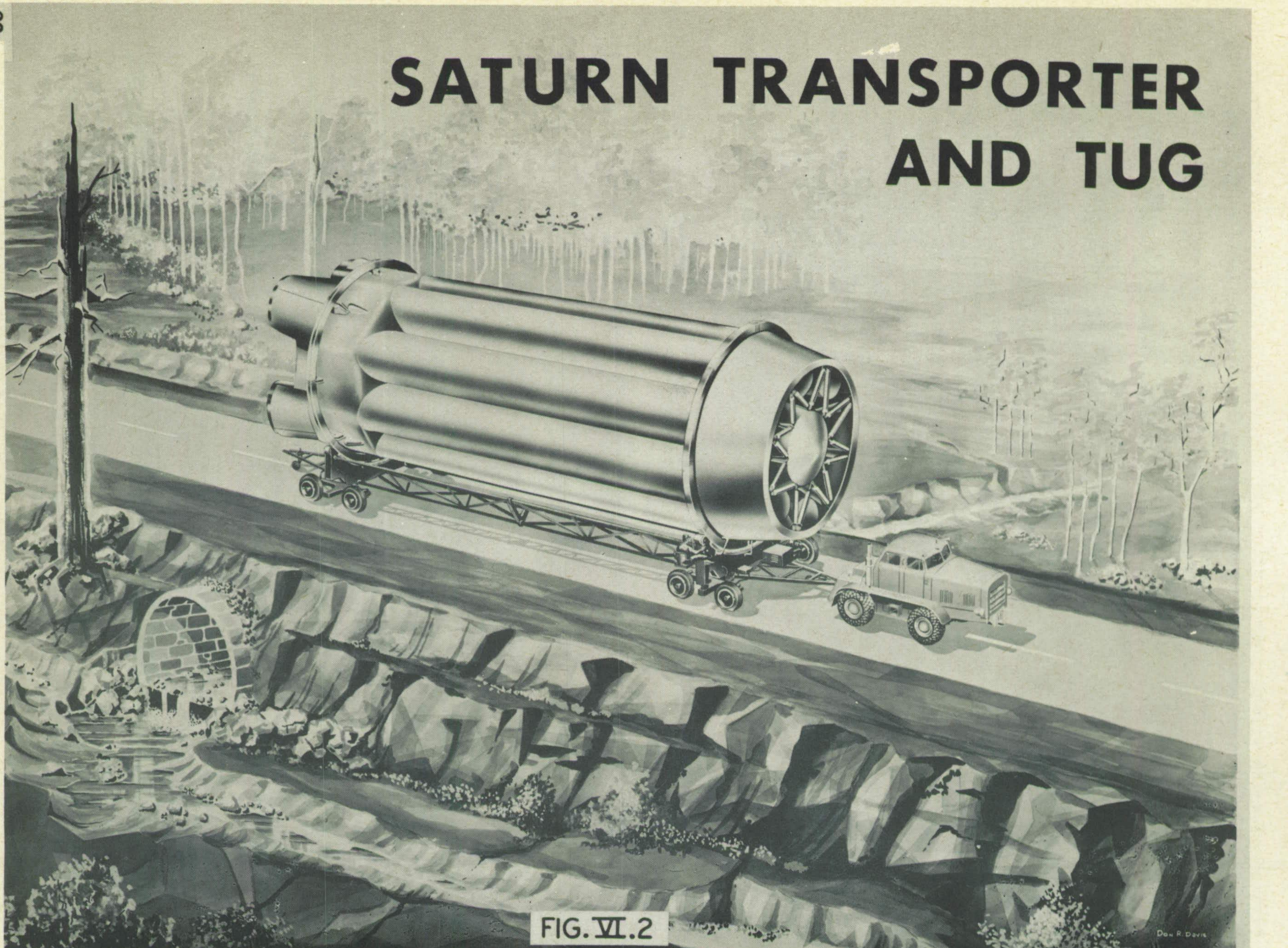
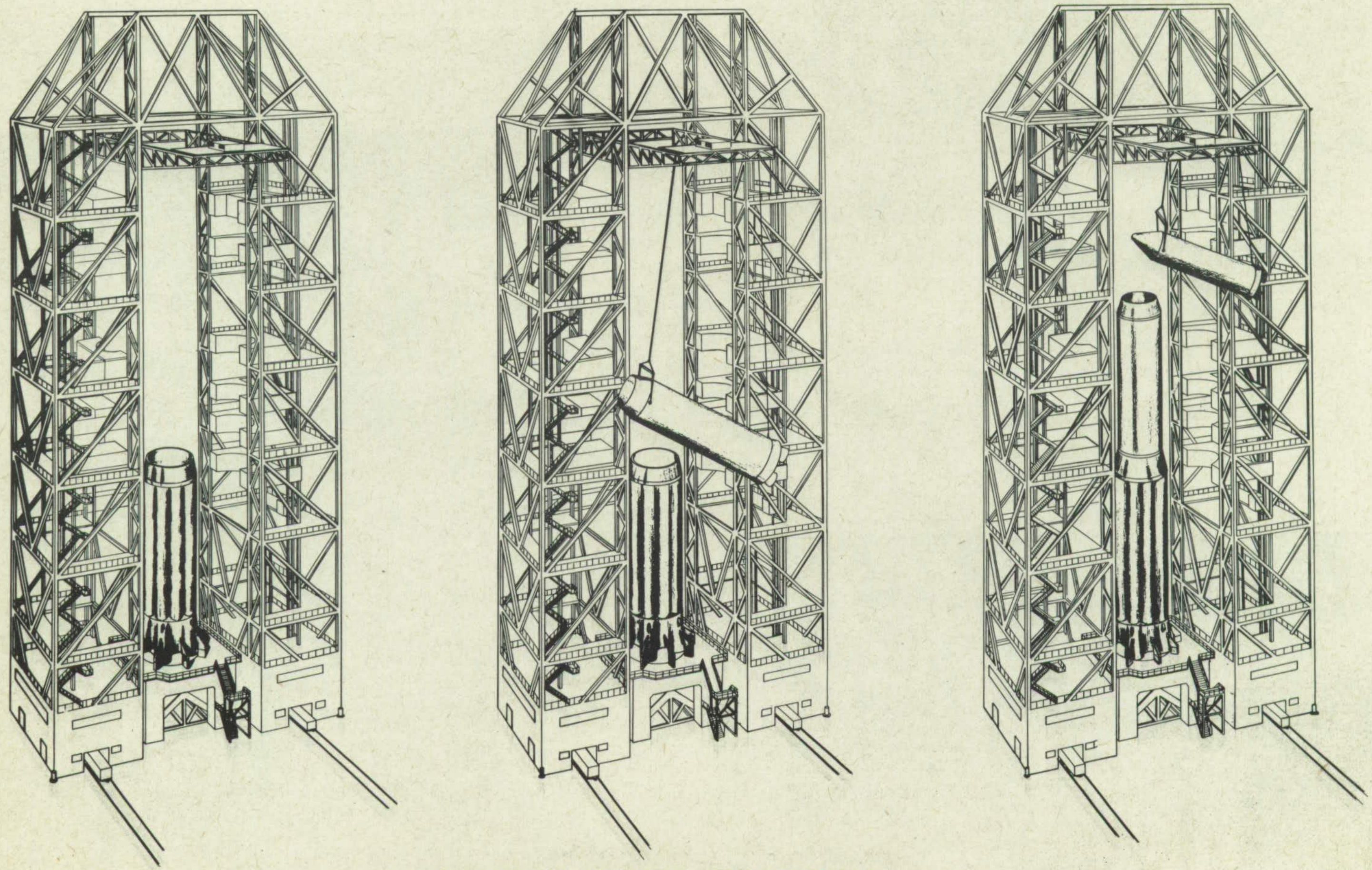


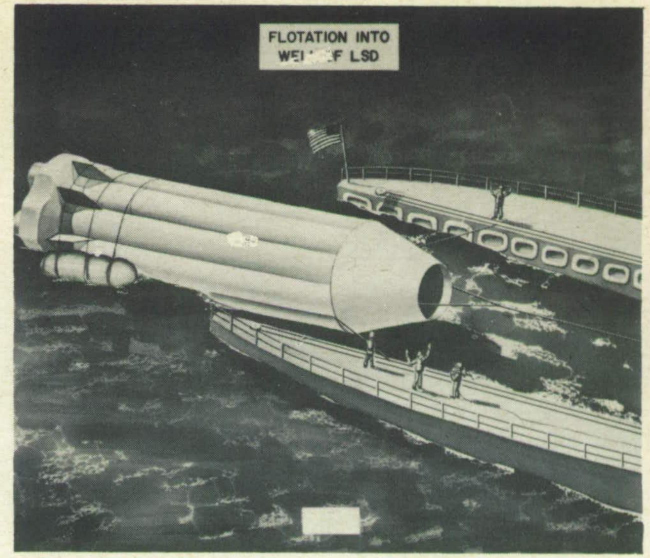
FIG. VI.2



SATURN STAGE ASSEMBLY

FIG. VI.3


DAMAGE CONTROL
SURVEILLANCE


FLOTATION INTO
WEL F LSD

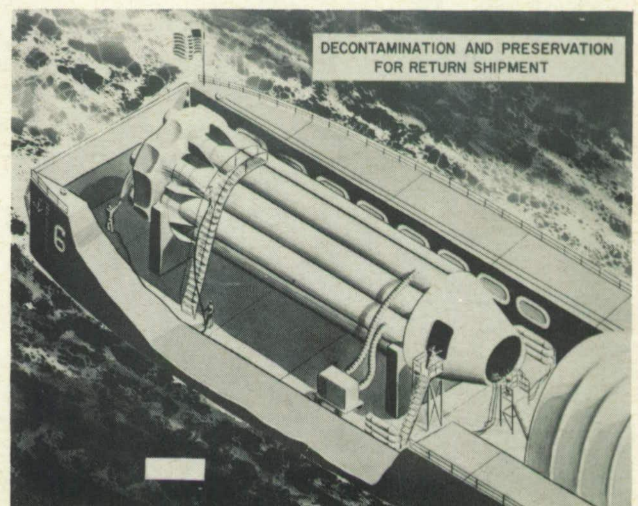

DECONTAMINATION AND PRESERVATION
FOR RETURN SHIPMENT

FIG. VI.4

VI.5.1 Soft Lunar Landing Stationary Packet. These two flights will be the first to be accomplished in the series and have the following objectives:

(1) Primarily, to accomplish the lunar scientific exploration and experimentation program, as discussed previously in Chapter IV.

(2) To provide a television view of the lunar terrain, during approach and while on the surface, giving possible the first U.S.A. real close-up picture of lunar selenological (topographic) conditions.

(3) To provide high altitude television views of the lunar surface during approach, to be used as a basis for selection of future landing sites.

(4) To provide actual impact experience for later flights.

(5) To provide the first full-scale flight test for the lunar guidance, tracking, and communication systems.

(6) To provide the first test for operation of vehicle components under lunar environmental conditions. Components include solar cell power supply, thermal control system, programming device, command receiver, etc.

(7) To provide the first actual flight test for operation of the braking stage rocket.

(8) To provide the first test of the SATURN booster vehicle system for injecting a payload into a lunar trajectory.

VI.5.2 Soft Lunar Landing Roving Vehicle. The third and fourth vehicles are assigned to the roving vehicle mission and have the following objectives:

(1) Primarily, to accomplish the lunar scientific and experimentation program, in more than a single location, as discussed previously in Chapter IV.

(2) To provide the first actual test for driving the roving vehicle in a lunar environment.

(3) To provide a test for the operation of vehicle components with attitude control (solar cell bank, antennae, radiator) under roving conditions, with changing terrain.

(4) To provide the first actual test of the TV and command system for guiding the vehicle during its travel on the lunar surface.

(5) To provide the first opportunity to view (via TV) a changing lunar terrain from the surface.

VI.5.3 Manned Circumlunar Flight and Return to Earth. The last two vehicles in the series have the following objectives:

(1) Primarily, to accomplish the specified lunar scientific and experimentation program. It is expected that the actual viewing of the side of the moon away from earth by a human will be of foremost importance.

(2) To provide the first full-scale flight test for the lunar circumnavigation guidance, tracking, and communications systems.

(3) To provide the first U.S.A. lunar vicinity flight to determine the effects of the flight upon human passengers.

(4) To provide a flight test of the system for approaching the earth, entering the atmosphere, and for impact and recovery of the manned payload package.

VI.6 LAUNCHING TIME CONSIDERATIONS

There are two conditions to be considered in choosing the initial values of the launching parameters:

(1) According to mechanical conditions the firing time has to be chosen so that the moon is at its lowest declination (about -23.6°) or at least close to this minimum. The minimum in declination occurs in the interval of the anomalistic month, which varies slowly, and secularly, in time, and is approximately 27.5 days. The minima themselves vary in amount, periodically, and slowly, secularly. During one year, the periodic variation amounts to $\pm .5^{\circ}$. The average rate of change

of declination around the instant of minimum declination is about 0.5^{o} per day both before and after the instant of minimum.

(2) The landing should take place on the terminator, at sunrise for the landing spot, in order to provide the longest working time using the solar cell power power supply, which is half a moon day, or 14 earth days. The terminator passes the given landing spot every synodic month, i.e., every 29.5 days and moves with constant speed, roughly 10 mph, along the moon equator.

If both requirements 1 and 2 should be exactly satisfied, favorable instants for launching occur every 4 or 5 years. Since this is not a feasible interval, one must be content in satisfying either one or the other conditions only approximately. Consideration has been given to changes in launching parameters due to delay in firing for one or two days. If a one-day delay is allowed, concerning the requirement in 1, a preliminary consideration has shown that favorable instants would occur about once a year. If two days of delay are allowed, the two adjacent minima are also favorable. If, also, one or two days of condition 2 could be waived, then about 6 to 7 minima during the year could be considered as favorable.

Lunar probes circumnavigating the moon may require illuminated lunar back side. Combining this condition with condition 1 and replacing condition 2, one faces the same conditions and intervals of favorability as combining 1 and 2.

Exact figures about the location of the terminator, the instant of minimum declination, and of the effect of libration can be given as soon as the ephemeris of the year of launching is published. Only then will it be possible to select the optimum firing time.

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