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A LUNAR EXPLORATION PROGRAM BASED UPON SATURN - BOOSTED SYSTEMS

1 FEBRUARY 1960

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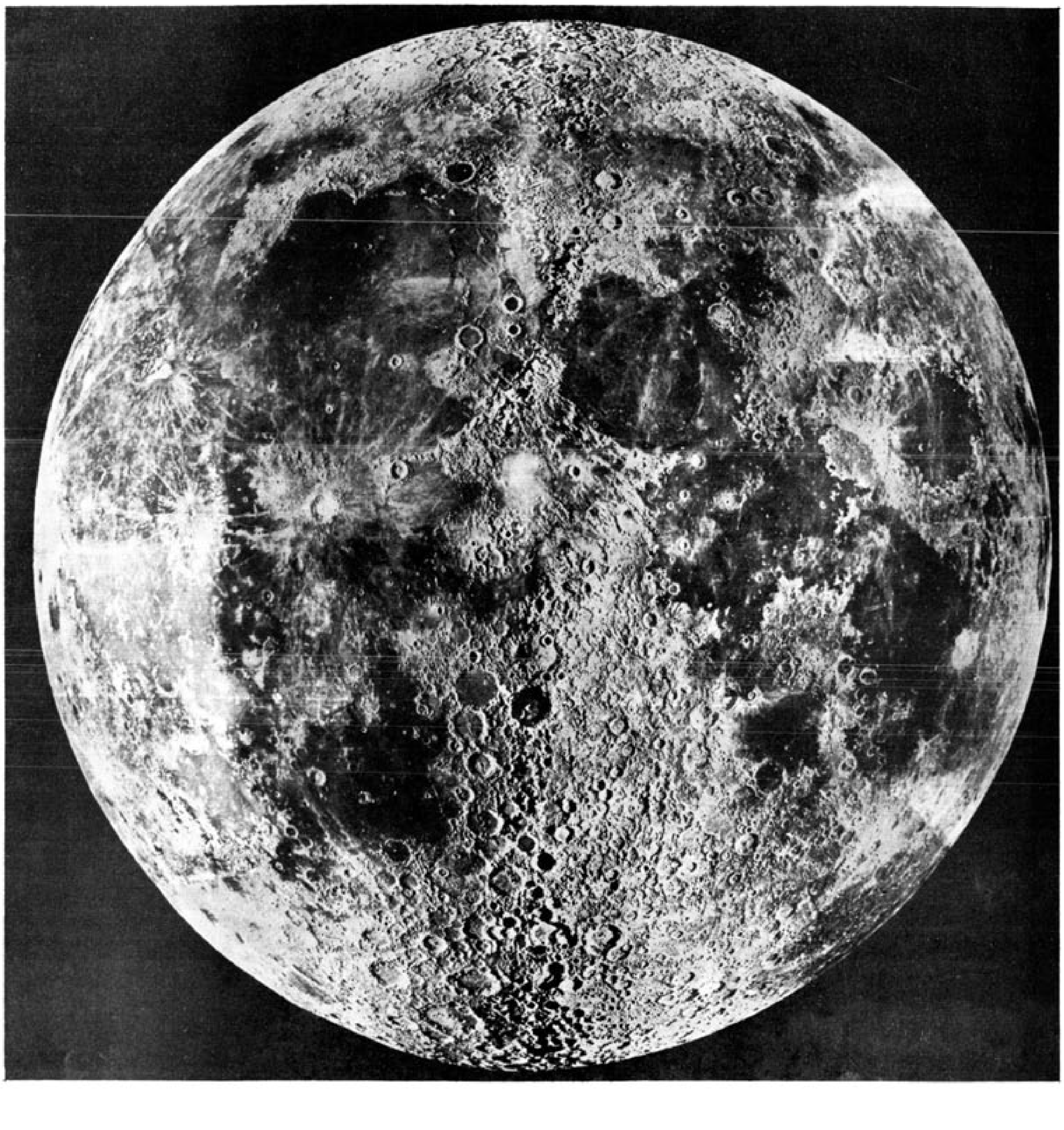

1 February 1960 Report No. DV-TR-2-60

A LUNAR EXPLORATION PROGRAM
BASED UPON SATURN-BOOSTED SYSTEMS

FOREWORD

This is the final technical report on the Lunar Soft Landing Study, accomplished by the Development Operations Division, Army Ballistic Missile Agency, Army Ordnance Missile Command, for the National Aeronautics and Space Administration under NASA Request HS-219. The report covers accomplishments during the period 1 July 1959 through 31 December 1959.

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CLASSIFICATION CHANGE

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Prepared by

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PREFACE

The National Aeronautics and Space Administration (NASA), in a meeting at the Jet Propulsion Laboratory (JPL) on 5 February 1959, established a Working Group on Lunar Exploration. Members of NASA, JPL, the Army Ballistic Missile Agency (ABMA), California Institute of Technology, and the University of California participated in the meeting. The Working Group was assigned the responsibility of a lunar exploration program, which was outlined in the following phases:

Circumlunar vehicles, unmanned and manned

Hard lunar impacts

Close lunar satellites

Soft lunar landings (instrumented)

Preliminary studies showed that the SATURN booster, with an ICBM as second stage and a CENTAUR as third stage, would provide a capable carrier for manned lunar circumnavigation vehicles and for instrumented packages of about one ton to land softly on the lunar surface.

On 1 May 1959, ABMA submitted to NASA a report entitled "Preliminary Study of an Unmanned Lunar Soft Landing Vehicle." Subsequent to this report, NASA Order HS-219 was issued to ABMA on 18 June 1959 requesting a study of a lunar exploration program based on the SATURN vehicle. The study was to cover soft landings on the moon for a stationary payload package and a package with roving capability, and a manned circumlunar flight with subsequent recovery. The study was originally scheduled for completion on 1 January 1960, but was informally extended to 1 February 1960. A preliminary report was issued on 1 October 1959.

The present report presents ABMA's accomplishments in the study. It discusses all the subjects agreed upon by NASA and ABMA, including a recently assigned section on manned lunar landings. Many problems still exist, and further effort is necessary, as described in Chapter VII of this report. All of the offices and laboratories of ABMA contributed to the report, especially the Aeroballistics Laboratory, the Guidance and Control Laboratory, the Research Projects Laboratory, the Structures and Mechanics Laboratory, and the Systems Support and Equipment Laboratory. The study reflects a state of the SATURN Vehicle Project as it existed in early December, 1959.

A large number of studies and reports by other organizations have been evaluated by members of ABMA during the past years. Some of the ideas and facts presented 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 bibliography of publications which are of interest in lunar studies is also included.

One valuable source of information was JPL Report 30-1, "Exploration of the Moon, the Planets and Interplanetary Space," edited by A. R. Hibbs, which presents a schedule of lunar missions designed to fit into the National Space Vehicle Program. The three rocket systems selected to carry out the lunar missions were the ATLAS VEGA,[1] the ADVANCED ATLAS VEGA[1] (with $H_2 + O_2$ second stage), and the SATURN. While the VEGA systems would have carried instrumented packages into lunar orbits and to the lunar surface with "hard" 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. The present study conforms to the missions proposed for the SATURN 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 in 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,

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[1]The VEGA Project was cancelled in December, 1959. Instead of the VEGA, lunar projects will be carried out by the ATLAS AGENA and the ATLAS CENTAUR before the SATURN becomes available.

numerous contacts were made with other organizations and individuals, and problems of lunar observations were discussed at length. The scientific measuring program and analysis of the design and operation of the instruments were established as thoroughly as possible within the time and funding limitations of the study.

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INTRODUCTION

This report has been written in a manner which will permit the reader with limited time to select the subjects most interesting to him without loss of continuity.

Chapter I describes the philosophy of the scientific exploration program, the SATURN booster system chosen for this study, and the terminology used for the payloads.

Chapter II deals with the transportation of the payload from earth to the moon. It describes the various phases of the guidance scheme, tracking requirements, trajectory and orbit requirements, and the lunar approach scheme.

Chapter III describes the construction and operating details of the stationary lunar payload, roving lunar payload and the manned circumnavigation payload. It also describes the problems encountered in designing lunar payloads to maintain proper operating temperatures, and considerations regarding engineering materials on the moon.

Chapter IV covers the scientific exploration program for the stationary and roving payloads on the lunar surface, and describes in detail each piece of scientific instrumentation, and the integrated program for operation of all instruments.

Chapter V describes the manned circumlunar flight, as well as a program for a manned lunar landing which can be accomplished by an orbital refueling or re-assembly maneuver.

Chapter VI provides a very detailed account of the methods for supplying power requirements while in flight, for the lunar surface stationary and roving payloads, and for the circumlunar payload.

Chapter VII describes how the program may be implemented, the over-all program requirements, the testing and training requirements, the method of supplying ground support, along with a SATURN lunar flight schedule. A table of organization is proposed for accomplishing the program, and a time schedule is shown for the over-all program.

This report is merely a summary of the principal efforts of the Development Operations Division of the Army Ballistic Missile Agency in the realm of lunar exploration. Numerous reports have been published within ABMA covering in more detail the items discussed herein and are included in the bibliography.

CHAPTER I

SCOPE OF PROJECT

I.1 OBJECTIVES OF PROGRAM

The program of SATURN flights considered here has two broad objectives: the continuation on an expanded scale of the exploration of the moon begun with other vehicles, and the development of technology for later manned lunar operations. These objectives, of course, are closely related. The relationship will become even closer as time passes, technology improves, and knowledge increases, until finally a manned laboratory is established on the moon for execution of scientific experiments.

The experimental program of lunar exploration described in Chapter IV is expected to follow a number of previous lunar investigations carried by vehicles less powerful and less sophisticated than those of the SATURN class. Hence, any group of experiments considered at present are still tentative, and may possibly change upon 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 possible atmosphere, the nature and magnitude of its fields, and its content of organic material, if any.

Landings of a stationary instrumentation packet and 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 at least one period of lunar daylight.

The scientific program proposed for the manned circumlunar flights 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 (C)

Because the SATURN vehicle system is the first with the capability of placing a non-marginal payload on the lunar surface, or of returning a man to earth after a circumlunar flight, it appears to be the logical follow-up program for the ICBM-boosted vehicles.

The SATURN booster as the basic unit of the SATURN vehicle system is now in development; its first full scale static firing test is expected in the near future.

Various configurations and types of upper stages have been studied for use in the system but a final choice has not been made. Performance and other considerations of these different versions of this system are discussed in great detail in some of the reference material.

The SATURN B-1, a four-stage vehicle using conventional propellants in Stages I and II and high energy liquid hydrogen and liquid oxygen in Stages III and IV, is considered the carrier vehicle for the purposes of this study. The 1.15 million-pound vehicle has a lift-off thrust of about 1.5 million pounds and an over-all length of slightly more than 200 feet. Approximately 16,440 pounds are boosted to slightly below escape velocity. Of this figure, 10,150 pounds, or about 60 per cent, are available for use in the mission. The remaining 6,290 pounds comprise the fourth stage engines and structure, residual fuel, guidance and control equipment and a device for separating the fourth stage from the useful payload.

Although it is recognized that a final decision on the choice of the upper stages of the SATURN system may change the quoted performance, the necessity for establishing a performance capability to obtain an integrated lunar mission must be accepted. It is not expected that a change to the SATURN C-2 configuration, a possible choice over the B-1, will vary the basic mission systems such as injection, lunar approach, roving vehicle, etc. Therefore, the use of the B-1 performance is believed justifiable for this study. The C-2 version, which features H_2 and O_2 engines in the second, third, and fourth stages, will be capable of injecting into a lunar trajectory approximately twice as much usable payload as the B-1 version. Because of the fixed weights of the engine, guidance and control equipment, attitude control equipment, etc., the weight available for the scientific payload in the lunar landing package would be approximately eight times greater using the C-2 version.

The brief description of the four-stage B-1 version of the SATURN given in the ensuing paragraphs is applicable to either the soft landing or the circumlunar missions. The B-1 configuration, with a typical payload attached, is shown in Figure I.1.

The SATURN booster stage with eight JUPITER-type engines develops a nominal thrust of 1.5 million pounds. A propellant loading of about 585,000 pounds of liquid oxygen and Rocket Propellant 1 (LOX/RP1) is assumed for this stage. The clustered tank construction is 21 feet in diameter and approximately 80 feet long. Since recovery of this stage is considered technically feasible and economically desirable, a parachute and retro-rocket recovery system will be employed.

The second stage of the carrier vehicle will consist of a single tank with four 220,000-pound thrust (220 K) (vacuum) clustered engines. Here again LOX and RP1 are the propellants used. The tank diameter is 220 inches and the over-all stage length is about 46 feet. Separation of the burned second stage will occur immediately after burnout of the propellants as it does in Stage I. A propellant loading of about 350,000 pounds is used for this stage.

For the third stage, approximately 72,000 pounds of liquid hydrogen and liquid oxygen ($H_2 + O_2$) high energy propellants are carried in this modified CENTAUR tankage. Use of four uprated (20 K) engines instead of the normal two 15 K CENTAUR engines gives this stage a thrust of 80,000 pounds. The tankage diameter of 220 inches and the length of 40 feet reflect the additional propellants carried in this stage. Again, separation will occur immediately after burnout without a coasting period for either of the missions to be performed. In each case, only a minimum of guidance, controls, and instrumentation are separated with each stage since the main instrumentation compartment is carried above the fourth (injection) stage.

The standard (25,000-lb propellant) CENTAUR stage was taken for the fourth (injection) stage of the SATURN carrier vehicle system. For purposes of this report, the two 15K engines were not considered to be uprated to 20K as were those for the third stage. Separation of this stage from the payload will not occur until the vernier corrections have been introduced, possibly several hours after injection. A portion of the instrumentation and guidance and control package will be designed to be retained by the payload package to provide power and some sub-assemblies which may be common to both the injection phase and later to the mid-course and landing phases.

I.3 PAYLOAD TERMINOLOGY AND WEIGHTS (U)

The useful payload is, of course, divided differently for the lunar landing and the circumnavigational missions. For the soft lunar landings, the landing vehicle (Figure I.2) consists of a braking stage rocket of about 20K thrust, guidance and attitude control system, propellant to accomplish the terminal approach, and the soft landing package. The soft landing package will be separated from the remainder of the landing vehicle prior to a final free fall to the lunar surface. The soft landing package consists of devices to cushion impact, some guidance and control equipment, attitude control equipment, and the payload itself. The payload consists of the necessary structure, power supplies, 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.

A hypergolic storable propellant combination has been chosen for the braking stage. Using this low impulse (300 I_{sp}) propellant, it is possible to place a 2125-pound soft landing package on the moon. If early flights by the ICBM-boosted vehicles show that the use of high energy fuels is possible in the lunar vicinity, then a sizable increase in payload weight will be realized.

For the circumnavigational mission, the circumlunar vehicle (Figure I.3) includes the injection guidance equipment (which is not separated as in a soft landing), a 20 K thrust engine, hypergolic storable fuel for trajectory maneuvers and attitude control, and the manned capsule. Approximately 7600 pounds are available for the manned capsule, including devices for atmospheric re-entry protection and recovery. Again, if earlier flights show that it is possible to use high energy propellants

FIG. I.2
LUNAR LANDING VEHICLE
FOR STATIONARY PACKET

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11 JAN 60

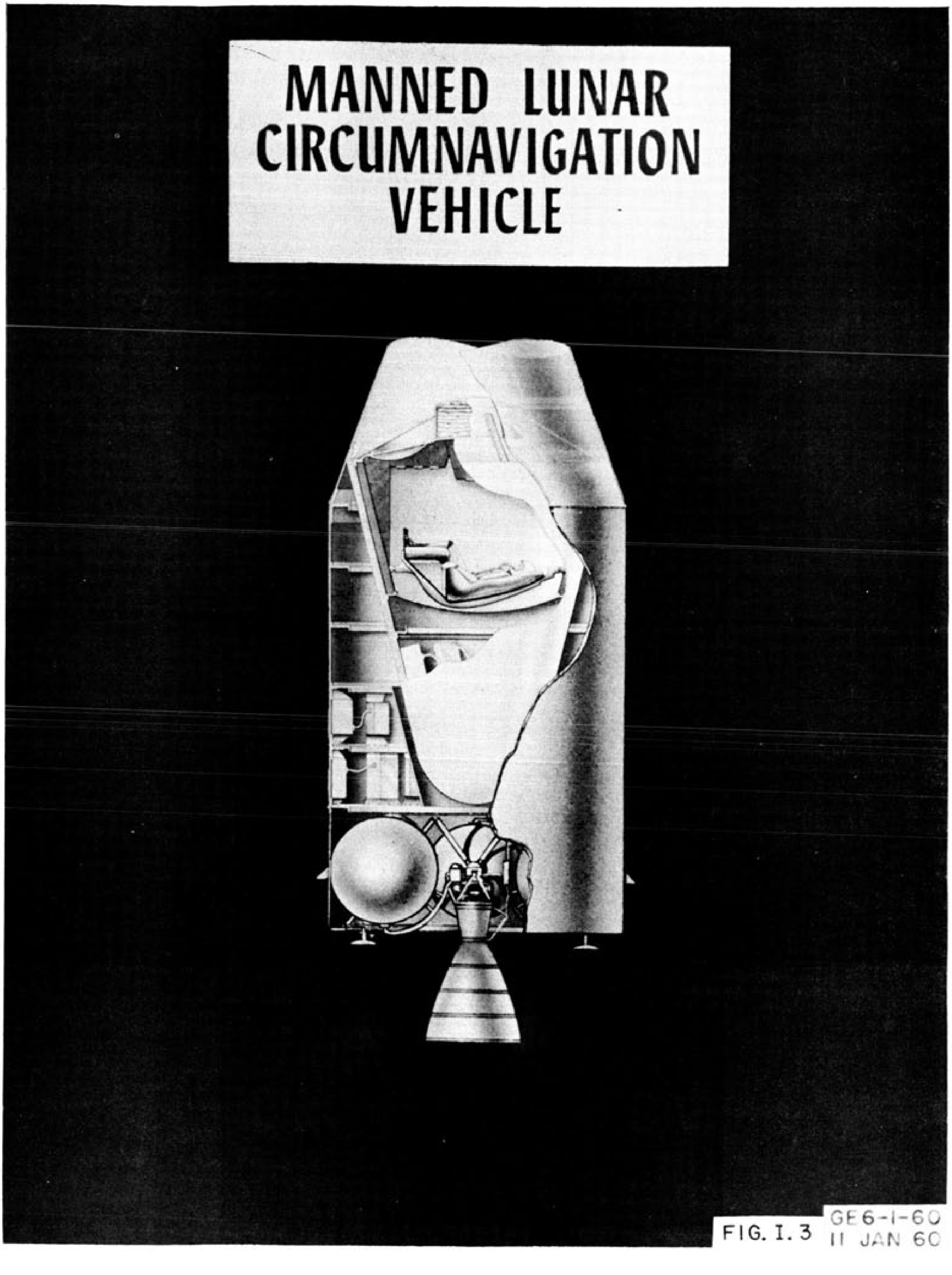

FIG. I. 3 GE6-1-60 11 JAN 60

such as liquid hydrogen and liquid oxygen, a sizable increase in the manned capsule weight may be expected.

The SATURN B-1 vehicle does not have sufficient payload capability for a direct manned landing on the lunar surface with subsequent return to earth. It is possible, however, to achieve the manned landing mission by introducing an orbital refueling or re-assembly maneuver and this program is discussed later in Chapter V.

1.4 OBJECTIVES OF FLIGHTS

The flights described in this program will furnish scientific data on the moon and its environment as well as valuable performance data on the SATURN system. The objectives of each payload will be discussed in turn.

Circumlunar Flights - An unmanned circumlunar flight will be the first flight to be accomplished in the lunar exploration program. Later flights will carry primates, and these will be followed by manned flights when reliability has been proven. The circumlunar flights have the following objectives:

(1) To accomplish a specified lunar scientific observation program. It is expected that the actual viewing of the moon, including the side away from the earth, will be of foremost importance. Also, to provide photographic and television views of the lunar surface, for possible use in selecting future landing sites.

(2) To test the complete vehicle and life support equipment, and to prove the atmospheric re-entry and recovery systems.

(3) To determine the effects of prolonged space flight and space environment upon human passengers.

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

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

Soft Lunar Landing Stationary Packet - The second flight in the lunar exploration program will place a stationary packet softly on the moon. It has the following objectives:

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

(2) To provide a television view of the lunar terrain, during approach and while on the surface, giving possibly the first U. S. close-up pictures of lunar selenological (topographic) conditions for scientific uses.

(3) To provide actual approach and touchdown experience for later flights.

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

(5) To provide the first flight test for operation of the braking stage rocket in a lunar environment.

Soft Lunar Landing Roving Vehicle - The flights following the stationary packet soft landing are assugned 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 in Chapter IV.

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

(3) To provide the first actual test of the television and command system for guiding the vehicle during its travel on the lunar surface, and for testing the drive mechanism under lunar conditions.

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

Manned Landing on Moon - These flights are not scheduled for a definite time in conjunction with the others, but may be expected to be accomplished when refueling and maneuvering techniques described in Chapter V are perfected. The objectives are:

(1) Primarily, to prove the system for transporting man from earth to the moon and back, and man's capability to do useful work on the moon.

(2) Scientific exploration of the moon in more detail than is possible by unmanned vehicles, or in special fields of work where robots are less suited.

(3) To obtain information on the feasibility of a sustained or periodically operating lunar observatory.

CHAPTER II

GUIDANCE, CONTROL, AND FLIGHT MECHANICS

II.1 GUIDANCE AND CONTROL REQUIREMENTS (C)

The objective of the guidance and control scheme is to inject a large lunar vehicle into a trajectory which places the vehicle within a small specified region of space near the lunar surface. Moreover, for a soft lunar landing, the vehicle must enter the specified region on a trajectory which very nearly coincides with the lunar vertical at impact. For lunar circumnavigation, similar severe restrictions may be placed upon the terminal portion of the trajectory. Terminal guidance is then initiated when the vehicle enters the specified region.

The flight path of a lunar vehicle is customarily divided into three phases for convenience in appraising the functions which must be performed by the guidance and control system. The flight path of the vehicle is diagrammed in Figure II.1. The first phase is referred to as the injection phase, the second is the post injection phase, and the third is the terminal phase.

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, mainly for two reasons: first, the low altitude of the vehicle during this phase and the geographical position (in the middle of the Atlantic Ocean) makes radio tracking untenable; second, inertial equipment such as is used in the JUPITER missile is entirely satisfactory for this phase provided later trajectory corrections can be made from ground-based radio trackers.

Injection into a typical lunar landing trajectory occurs approximately 865 seconds after launch. The injection point lies approximately 40 degrees from the launch site which is assumed to be at Cape Canaveral, Florida. At the point of injection into a typical lunar trajectory, the vehicle has an altitude of approximately 300 kilometers and a speed of 10,808 meters per second. The velocity is directed 5 degrees above the local horizontal, the lead angle (see Figure II.1) is 125 degrees. These injection conditions give a transit time to the

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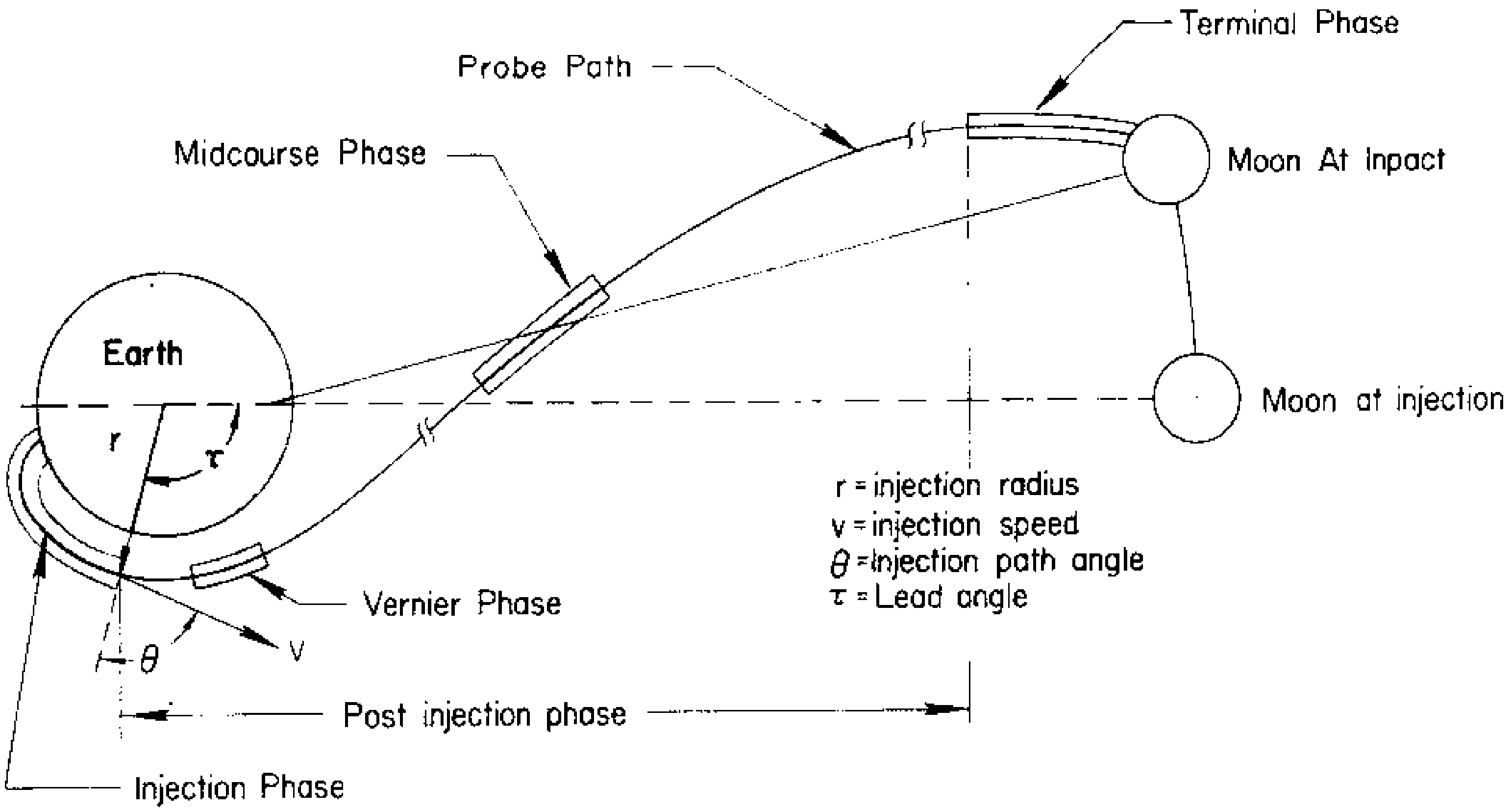

FIGURE II.1 GEOMETRY OF A LUNAR TRAJECTORY

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lunar surface of 57.3 hours. If no terminal braking were applied near the lunar surface, the vehicle would impact vertically on the moon with a speed of 2727 meters per second. Control of the vehicle during the injection phase will be accomplished by swiveling the main engines of the booster. At the end of the fourth stage burning period when vernier thrust is being applied to compensate for the thrust decay of the main CENTAUR engines, control will be accomplished by means of thrust units mounted on the fourth stage frame.

The post-injection phase consists of the flight from the injection to the terminal phase. The portion of the flight during the early part of the post-injection phase is referred to as the vernier phase, and the portion of the flight intermediate to the injection and terminal phases is called the mid-course phase.

The stringent accuracy requirements of lunar trajectories indicate that the all-inertial injection is not sufficiently accurate to inject the vehicle into a lunar trajectory which leads into a selected small region of space near the moon. Hence, provisions must be made for post-injection corrective maneuvers in order to conserve energy and to relax the demands on the terminal guidance. A study of the problem of post-injection correction indicates that ground-based radio tracking stations can supply trajectory data that are sufficiently accurate for use in a post-injection corrective maneuver. Energy considerations emphasize that any correction should be made soon after injection. At present, it appears that ground tracking data could be available for a vernier correction as early as two hours after injection. The proper point for making a post-injection vernier maneuver depends upon several factors: first, the magnitudes of the corrections are functions of time; second, the stabilized platform used for post-ignition correction drifts away from its proper orientation and the effects of the orientation are introduced into the trajectory by the corrective maneuver; third, the quality of the estimates of the trajectory parameters is dependent upon the tracking scheme and the tracking time. An analysis of the problem indicates that the vernier correction should be made at two to ten hours after injection, depending upon the final selection of tracking equipment. The use of existing tracking nets would place the correction nearer the ten-hour point on the trajectory, whereas specially constructed lunar tracking nets would enable the correction to be made nearer the two-hour point.

The desired terminal conditions of a lunar trajectory for some missions are sufficiently demanding that the vernier correction alone may not provide a satisfactory trajectory correction. For example, such a desired terminal condition may be vertical impact on a given small area on the moon. For a vertical lunar impact, one corrective maneuver during the vernier phase may not be sufficient to satisfactorily compensate for errors made during the inertial injection phase if the time elapsed from injection to the first correction is relatively long. This time period is dependent upon the performance and design of the tracking system. Hence, a second correction to be effected at some point during the mid-course phase may be indicated by the tracking data. Therefore, it is necessary that a post-injection guidance scheme be employed which provides for a vernier corrective maneuver with the option of a subsequent mid-course maneuver.

The detailed mathematical aspects of a guidance scheme designed to make two such corrective maneuvers in an optimum manner have not been explored. However, it is anticipated that if the correction can be made soon enough after injection or if a more elegant terminal guidance scheme evolves, then a single vernier correction will probably become sufficient. At the present it seems advisable to concentrate on one vernier correction with provisions for a second fine correction in the event that the first maneuver does not provide a satisfactory trajectory.

Attitude control of the vehicle during the post-injection phase is required so that corrective impulses may be applied and to maintain the vehicle antenna toward earth to facilitate tracking and communications. Attitude control will be accomplished with small thrust units. The reference for attitude control during that phase will be optical sensors.

II.2 INJECTION GUIDANCE SCHEME (C)

Guidance intelligence is obtained from a stabilized platform upon which are mounted three mutually orthogonal accelerometers. The outputs of the accelerometers are used to compute the guidance and thrust cutoff signals. The stabilized platform is operative from launch until after final inertial injection and is space-fixed in direction at launch. It maintains this space-fixed direction throughout the injection phase.

In order that deviations from the desired flight path may be determined, the accelerometer outputs corresponding to the prescribed standard flight path are programmed into the guidance computer. Measurements of the perturbed flight are made so that deviations become available. Since the actual vehicle will probably follow a perturbed path different from the standard, it will experience different gravitational accelerations. The errors resulting from these different gravitational accelerations are a function of displacement deviations and flight time deviations. Consequently, in order that the perturbation errors be accurately known for corrective action, the errors resulting from different gravitational accelerations are continuously computed from launch to injection into the lunar trajectory 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 burnout of the first stage, the η axis is in the flight plane and perpendicular to the ξ axis, and the ζ axis is normal to the flight plane (see Figure II.2).

At burnout of the first stage, the guidance coordinate system (hereafter referred to as the coordinate system) is rotated about the ζ axis so that the ξ axis will be tangent to the reference trajectory at burnout of the second stage. This transformation consists of mathematically resolving the accelerometer output into the components representing the desired coordinate orientation. This, of course, does not disturb the original space-fixed position of the stabilized platform. The ξ and η deviations from the reference trajectory existing at burnout of the first stage are then expressed in the new coordinate system. Since the missile will be guided to an η reference throughout the flight, the deviations in η displacement and velocity should be small. A diagram of a typical trajectory showing points where basic coordinate transformations are effected is shown in Figure II.2.

At second stage burnout, the coordinate system is again rotated through an angle about the ζ axis such that the ξ axis will be tangent to the reference trajectory at a point where injection into the lunar trajectory occurs. As before, the ξ and η deviations are expressed in the new coordinate system. Thus, this coordinate system serves for both the third and fourth booster stages. Just before injection velocity is obtained, the fourth stage (CENTAUR) is cut off and vernier thrust is initiated to effect the final injection. Vernier thrust is terminated when a solution to a specified cutoff equation is

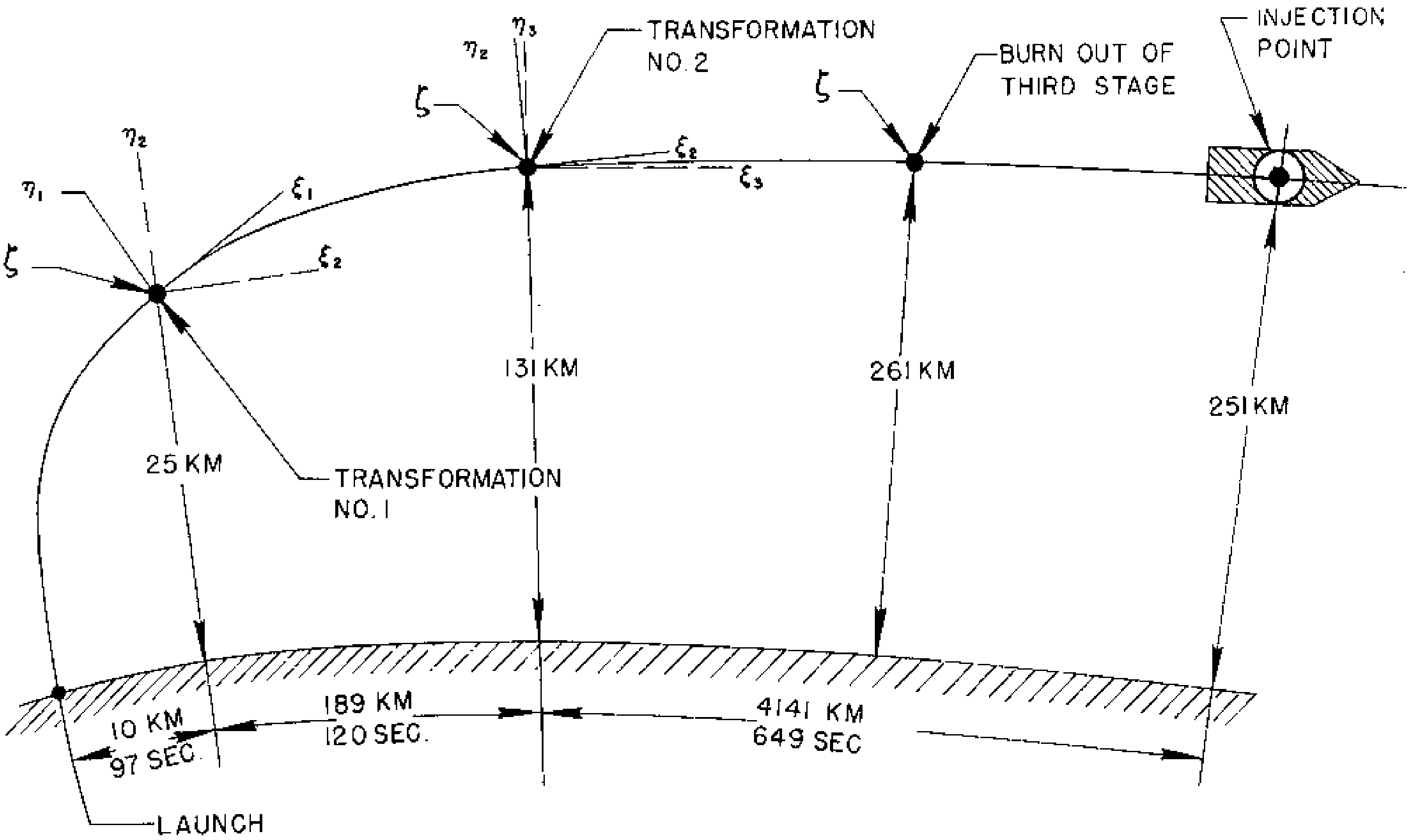

FIGURE II. 2

COORDINATE TRANSFORMATION FOR BASIC GUIDANCE SCHEME

obtained. The main stage cutoff signal for the CENTAUR is also supplied by the solution of a cutoff equation.

A basic guidance scheme which will provide accurate injection into the desired lunar trajectory has been described. Any such scheme conceived for use with boosters such as the SATURN should be capable of successful guidance under the conditions resulting from a failure of a single engine of the SATURN cluster. The basic scheme described can be made sufficiently flexible to cope with such a condition through the process of optimizing the standard trajectory.

A modification in the basic scheme would result in a final guidance scheme, capable of guiding the vehicle successfully even when one of the SATURN engines fails after liftoff. It will consist of the basic scheme described plus additional coordinate transformations occurring during burning of the first and second stages. A preliminary investigation indicates that these additional coordinate transformations will occur approximately at the times indicated in Figure II.3. Further study will be required to fix their precise number and location. In addition to the extra coordinate transformations, the orientation of the coordinate system at each point of transformation will be made optimum for guidance under standard and engine-out conditions. By properly selecting the number of coordinate transformations and coordinate orientations for the standard programs, accurate injection into the lunar trajectory will occur automatically under standard burning conditions or with a failure, after liftoff, of one engine of the SATURN booster.

It should be emphasized that the coordinate transformations are effected by mathematical computation in the digital type guidance computer and not by torqueing the platform. The platform retains its original space-fixed direction from launch until after injection.

Signals for attitude control and stabilization during the propelled flight phases will be derived from the stabilized platform, vehicle-mounted accelerometers, and rate gyros. A block diagram of the guidance and control system for the inertial injection is shown in Figure II.4. Control of the boosters during the powered phases will be accomplished by swiveling the main engines. Vernier thrust for the inertial injection will be provided by low thrust nozzles affixed to the fourth stage. After separation of the fourth stage, which occurs immediately after the injection phase, additional nozzles

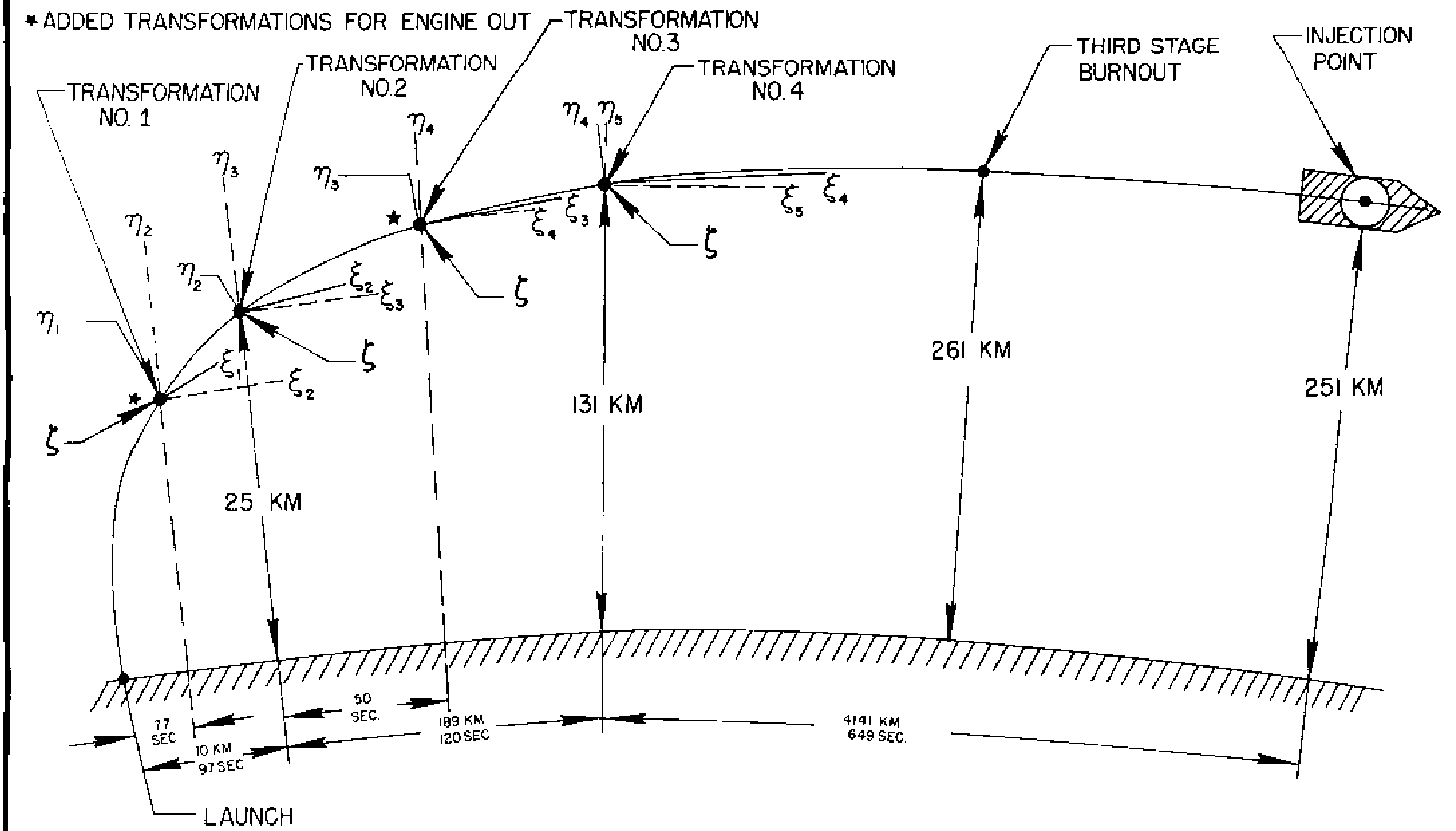

FIGURE II . 3

COORDINATE TRANSFORMATION FOR FINAL GUIDANCE SCHEME

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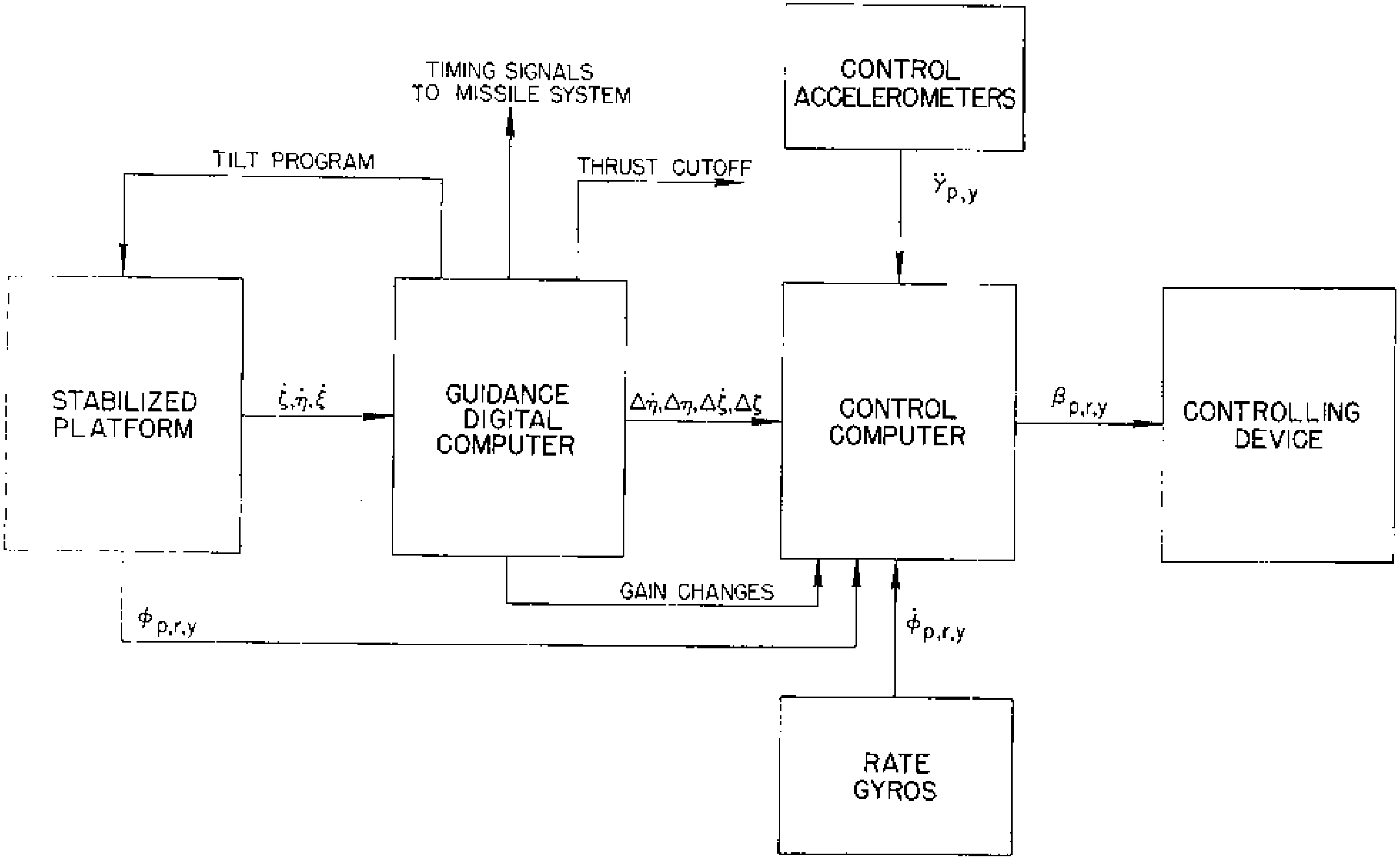

GUIDANCE AND CONTROL SCHEME

FIGURE II.4

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mounted on the payload stage will direct thrust for corrections during the post-injection phase and provide attitude control.

Several reaction type systems were considered for these applications. Among them were pressurized gas systems and hydrogen peroxide systems. It was found that due to high container weights and leakage rates the pressurized systems imposed a severe weight penalty for such a long flight time. Hydrogen peroxide systems were discarded because of their limited total impulse capability and their low specific impulse when compared to the more energetic propellants. Thrust systems of the type considered (i.e., hydrazine and nitrogen tetroxide systems) are well within the state of the art.

It is planned that the control system for the payload stage will consist of thrust units which will be designed to control the vehicle attitude in accordance with information received from optical sensors. Additional low thrust units will be attached to the payload for attitude control during the last few seconds before contact with the lunar surface.

Vernier thrust for the payload section will be accomplished with four 100-pound thrust units.

It is proposed to employ the gyro stabilized platform of the JUPITER missile for the inertial injection phase. This platform is shown in Figure II.5. The accuracy and reliability of this platform has been firmly established through numerous flight tests in the JUPITER missile.

The platform consists essentially of three integrating gyro accelerometers and three air bearing stabilization gyros mounted on the inner gimbal of a three-gimbal axes configuration. Adequate gimbal clearance ($\pm 15^\circ$ in roll and yaw, and 200° in pitch) is provided for this portion of the total flight. A pitch tilt program of the desired angular velocity and magnitude may be applied to the platform if desired. The platform capsule is rigidly mounted to the missile frame; it weighs 115 pounds and occupies 2.5 cubic feet. The platform and associated guidance and control equipment, but not the digital guidance computer, are located in an instrument compartment immediately above the fourth stage. The digital guidance computer is located in the payload stage and is used for guidance functions throughout the lunar flight.

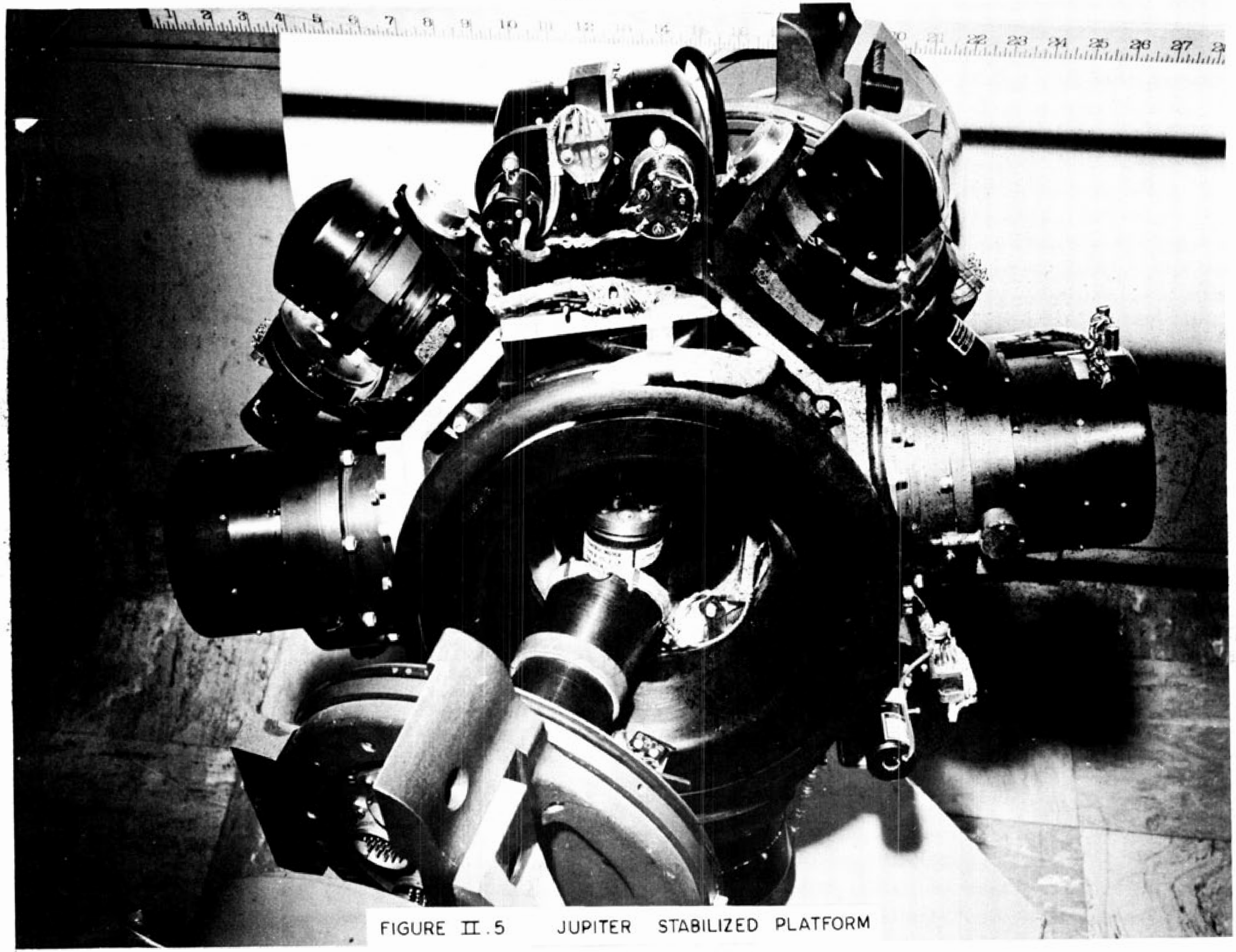


FIGURE II.5 JUPITER STABILIZED PLATFORM

The platform also furnishes the control signals for attitude control of the vehicle (sensitivity: 0.02 volts for 0.01 degrees). Special torque input loops are provided to achieve alignment and azimuth orientation of the stabilized platform before launching.

Appendix A describes the guidance computer system for the inertial injection phase of the flight.

After termination of the vernier thrust of the fourth stage, the fourth stage and the instrument compartment which in part contains the stabilized platform and associated equipment except the digital guidance computer are separated from the payload stage.

II.3 POST INJECTION GUIDANCE SCHEME (C)

During the injection phase the ST-90 stabilized platform with its accelerometers will be used as the inertial measuring unit in the guidance system. The power requirements of this platform are large enough to render its use undesirable over long periods. Furthermore, the gimbal limits of this platform restrict the direction in which corrective velocity increments may be conveniently made.

Since maneuvers during the post-injection phase may require 360 degrees of platform freedom as well as relatively long periods of operation, it is proposed that the ST-300 stabilized platform be used for this portion of the flight. This unit is a four-gimbal platform with 360 degrees of freedom which is light in weight and economical in terms of power consumption. Although this platform is not as accurate as the ST-90, it is sufficiently accurate for the intended application. The platform is stabilized in three axes and contains three mutually orthogonally mounted accelerometers and other necessary equipment.

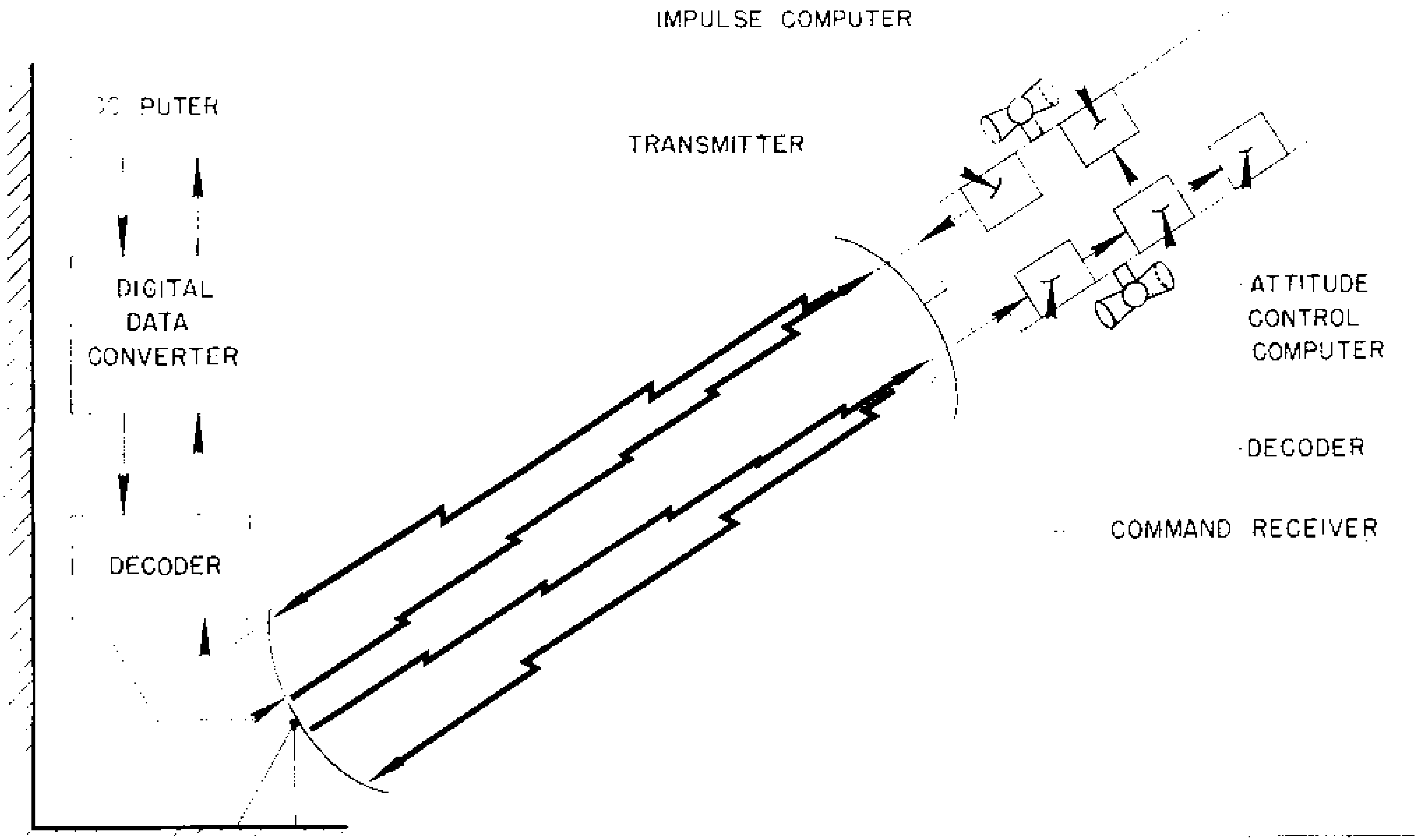
The ST-300 will be erected prior to launch and will become space-fixed at lift-off just as the ST-90 platform. However, no output from the ST-300 will be used until after termination of the injection phase. After injection is accomplished, the fourth stage and the instrument compartment, containing the ST-90 and other injection guidance and control equipment, will be separated from the lunar landing vehicle. Contained within the lunar landing vehicle will be the ST-300, the digital guidance computer, required power supplies, and other necessary guidance and control equipment.

At separation of the second stage, shrouds will open to expose the antenna and optical sensing devices (horizon seeker and sun seeker). After separation of the fourth stage, the vehicle is oriented by means of the small thrust units, according to a programmed attitude so that the directional antenna is properly oriented for tracking and communication purposes. A directional antenna, although requiring continuous attitude control throughout the total flight, is more economical in power consumption than an omnidirectional antenna. Moreover, a directional antenna is required in the terminal guidance system and for communication purposes after landing on the moon.

As previously pointed out, ground tracking stations may require from two to ten hours in order to accumulate and process the tracking data. If the vernier correction is not given within approximately five hours, the horizon seeker and sun seeker will begin supervising the ST-300 to preclude excessive drift of the platform.

After ground tracking has produced the necessary data, a correction command will be transmitted from the ground station to the vehicle. This command will be received by a command receiver aboard the vehicle. The command will then be decoded to provide the inputs to the control computer which causes the vehicle to assume the proper attitude and the vernier engines to become operative for application of the required impulse. This process is illustrated in Figure II.6. The command decoder also supplies presetting to the impulse computer for the subsequent solution of an appropriate cutoff equation. Signals from the ST-300 and associated accelerometers will also be transferred to the impulse computer.

The cutoff equation to be solved by the impulse computer requires special consideration. Because of injection scheme errors, hardware errors, and the flight history of the vehicle, there will be position, velocity, and time errors at the point of injection. Lunar trajectories are sensitive to one or more of these variables and it is not likely that the standard trajectory will remain useful as an absolute reference. That is, it may be necessary to alter the trajectory from the standard in order to come within the specified region near the moon. Thus, the vernier correction may be based on the conditions expressed by a cutoff equation formulated to cause the vehicle to follow a path slightly different from the original standard. The vernier maneuver would alter the velocity vector so that the new trajectory intersects the original standard trajectory



RADIO GUIDANCE SCHEME

FIGURE II.6

in the prescribed region near the lunar surface where terminal guidance is initiated.

For the simple geometry described by Figure II. 1, a first order cutoff equation may be written, for discussion purposes, in the form:

$$A\,\Delta v + B\Delta\theta = C\,\Delta r + D\Delta\tau + E\,\Delta t$$

where A, B, C, D, and E are constants for a given case, and Δv, $\Delta\theta$, Δr, $\Delta\tau$, and Δt are deviations from the standard case in speed, path angle, radius, lead angle, and time, respectively. This equation may be used to alter speed and path angle in such a way as to eliminate errors in position and time at the prescribed region near the lunar surface. However, such an equation places no restrictions on the velocity vector in the prescribed region. If the vernier correction is not made very early after injection, a second correction may be required in order to cause the velocity to conform to selected conditions. It should be noted that the above equation indicates that Δv could be restricted to positive values, but in doing so $\Delta\theta$ may become excessively large. On the other hand, A and B can be so chosen that the effects of hardware errors introduced by the corrective maneuver are minimized.

After this vernier corrective maneuver has been completed, the ST-300 and various other equipment will be made inoperative in order to conserve electrical power. The horizon seeker and sun seeker will remain operative in order to provide an attitude reference. The attitude is maintained to within a specified deadband throughout the post-injection phase in order to keep the antenna in the desired direction. A communication and tracking link is maintained between the ground stations and the vehicle throughout the post-injection phase for monitoring purposes as well as for possible initiation of a mid-course correction.

Any mid-course correction indicated by ground tracking information will follow the pattern of the vernier correction. However, in order to make a mid-course correction the ST-300 must be erected and oriented by the sun seeker and horizon seeker.

As the vehicle approaches the moon, the horizon seeker, functioning also as a rough altimeter, will provide a command to erect the ST-300, according to the information from the sun seeker and horizon seeker. The ST-300 then provides the vehicle with an initial attitude reference for entering the terminal phase. The terminal guidance is described in Section II.6 of this chapter.

II.4 TRACKING (U)

Post-injection or mid-course maneuvers will be applied to the lunar vehicle according to the results of tracking of the early free flight. In view of the required accuracy in the flight parameters, the time period of tracking will be important. In order to perform the desired maneuver as early as possible, it is desirable that the tracking stations acquire the vehicle as soon after completion of powered flight as possible, and continue tracking as long as possible. The acquisition problem of high precision tracking instruments also makes this desirable.

The location of any tracking stations will be determined primarily by these considerations, by logistical problems, and by consideration of expense and multi-use of stations. A discussion of possible station locations is made in Appendix B.

II.4.1 Feasibility for Use of Existing Stations. Three different possibilities for solving the tracking problem have been considered. The first one is a Doppler and Range Tracking system (DORA) particularly designed to meet the requirements of lunar or similar space missions. By suitable choice of station locations (probably near the east coast of Africa), fastest orbit determination could be reached. The second possibility is the utilization of existing tracking stations of the NASA/JPL Deep Space Net. A third possibility can be obtained by combining the two.

The Dora Tracking System - The proposed DORA system is a precision continuous wave (CW) Doppler system with additional arrangements to obtain unambiguous range data. Position of the vehicle is computed from range information provided by each site of a three-station system with the stations spaced to give a wide base line. Time and phase reference between stations is maintained by atomic frequency standards in conjunction with long-term phase comparison by low frequency transmission or other methods suitable to a particular application. A preliminary description of the system is given in Appendix C.

The flexibility, minimum cost and growth potential of the DORA system makes it attractive. So far as possible, it is proposed to make use of existing tracking facilities where they are suitably located. The base lines of the stations are to be in the order of 100 to 800 miles. The electronic equipment could be portable. The most favorable station locations must be determined after the final trajectories of the lunar missions are fixed.

It appears that with an optimized DORA station system, the minimum tracking duration for the case of soft lunar landing would be in the order of two hours after launch. Studies are continuing to determine optimum tracking conditions for the lunar circumnavigation.

The DORA ground equipment could also be used separately as a single tracking station, providing unambiguous range data in addition to the range rate information that is obtained with a normal Doppler system.

NASA/JPL Deep Space Net - A tracking study conducted by the Jet Propulsion Laboratory (JPL) indicates that the NASA Deep Space Net would also offer a good possibility of tracking the lunar probes, using only existing stations with installations planned for the near future.

With some anticipations (e.g., ranging capacity in Johannesburg) JPL came to the conclusion that a tracking duration of about three hours would permit orbit determination with an accuracy corresponding to lateral touchdown tolerances of 10 kilometers on the lunar surface.

From a purely technical standpoint, this multi-purpose net does not represent the optimum solution for tracking of lunar probes. However, the utilization of existing stations, tracking instrumentation, communication links and computing facilities would insure highest economy.

Possible Combination of DORA and Deep Space Net - Perhaps a favorable compromise can be made between technical optimization and economic limitations by utilizing DORA modified tracking equipment at existing station installations of the Deep Space Net. In the event of this combination, further studies will have to be made.

II.4.2 Supplementary Optical Tracking. Radio tracking measurements could be complemented by optical measurements to increase the over-all accuracy. An applicable system would be the "Cat Eye" light amplifier. This is an image orthicon tube with special intensifier sections capable of amplications in the order of 10^9 or better.

Continuous optical tracking may be impossible from a single station because of cloud coverage. However, due to lower cost, several stations could be used to assure adequate coverage.

The light amplifier should not be considered as a substitute for radio, but rather as a complementary device. The two instruments together will give information and make tracking data available which neither instrument alone can produce.

II.5 FLIGHT MECHANICS OF THE TRAJECTORY FOR LUNAR LANDING

II.5.1 General Considerations and Requirements. The nature of the flight mechanical problem in flights to destinations whose motions do not follow that of any surface point of the earth, is, first, establishing the family of feasible solutions insofar as these connect launching-point and destination, and second, choosing the optimum solution within this family. The "optimum" is defined as the engineering optimum, considering a multitude of rather unrelated viewpoints. Since the feasibility area is changing, e.g., with the changing position and orientation of moon and earth, and the true optimization is dependent on hardware and other layouts, any final solution can be accomplished only at a date relatively near to the scheduled firing date.

Present information, therefore, is restricted to enumerating the essential problems with contingent implications and presenting one solution of the over-all problem, which may bear some features of the final flight path.

For the selection of the orbital characteristics, the problem of choosing the energy level (or injection velocity) is first noticed. On the lower end of the energy scale there are savings in propellants for injection and braking; on the higher energy level: (1) time will be gained and by that eventually power sources may be smaller in weight; (2) correction maneuvers will be smaller, since a high energy flight is less sensitive to certain types of injection errors.

There is the problem related to the most favorable launch dates. Laying the orbit as far coplanar to the lunar orbit as possible reduces the accuracy requirement of some parameters, with a consequential reduction in correction impulses. This, however, limits the dates of launching. The second requirement, that of landing in the desired area at dawn, may, however, be in conflict with the former requirement.

Also, to accomplish landing in a pre-selected area on the moon, one may be required to depart from otherwise desirable conditions of injection parameters as, for example, eastward launching. Next, there may be restrictions of the orbit laying to assure favorable tracking situations that are basic for the planned mid-course corrections.

Range safety restrictions on the propelled flight phase give limiting conditions on azimuth.

Aerodynamic heating viewpoints, as well as those of structural load and control modes during the high dynamic pressure flight, impose further restrictions on the freedom of the trajectory design problem.

This list, which does not cover all possible requirements, may be sufficient to indicate the complexity of the problem and the futility of trying to solve the problem at the present status.

When in the following a nominal trajectory for the lunar landing is described, some possible requirements are deliberately disregarded.

The earth - moon model underlying the study is that of the two celestial bodies revolving in circular orbits about their barycenter. The earth - moon distance is 385,080 kilometers (or 239,277) statute miles) from center to center. Other gravitational influences, as from sun or earth's oblateness, are disregarded.

The moon's orbit is assumed to be at an inclination of 25 degrees to the equator.

It is further assumed that the time of travel of the vehicle does not influence the optimization problem. Therefore, the injection into the orbit may be governed by the propellant savings (or payload gains) accomplished with a very low injection velocity.

The next section discusses the characteristics of the propelled path, and the following describes the orbit to the moon.

<u>II.5.2 Analysis of the Powered Phase of the SATURN B-1 Vehicle</u> (U). A four-stage missile is assumed for the powered phase of the configuration. It is known as the SATURN B-1 at ABMA.

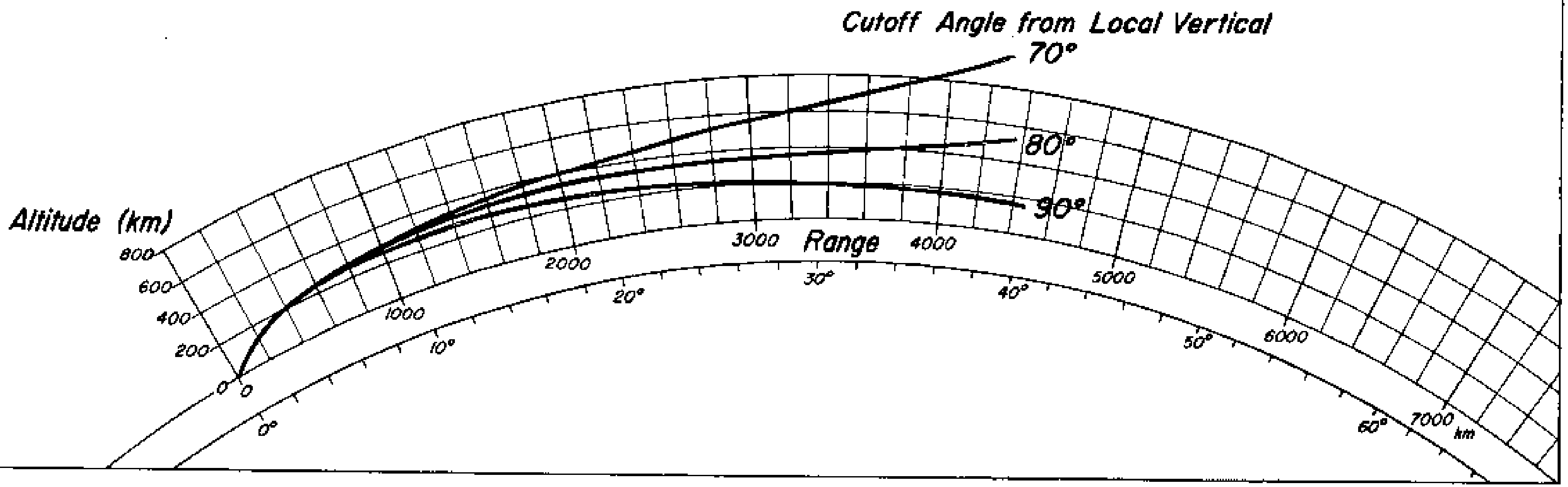
For the general optimization problem, information is needed about the payload capability of the missile for a variety of launching azimuth directions combined with varied path angles (in pitch direction). Out of this double-parameter family, a trajectory is to be chosen that combines with the celestial matching problem in the most favorable way.

For each of three typical launch-azimuth conditions, three trajectories of differing steepness were established. They resulted, at the termination of the fourth stage, in the path angles of 70, 80, and 90 degrees from the local vertical. Consideration was given to the aerodynamic heating viewpoint by raising the path into sufficient altitudes before certain velocities were reached. Also the load and control problem was minimized by requiring zero-angle of attack in the region of high dynamic pressure. Otherwise, optimal tilting for maximum payload was followed. This is achieved by applying methods of calculus of variations. When the conditions of the three terminal path angles are imposed, altitude is a resultant parameter. The conditions are to be achieved in conjunction with velocity attainment, e.g., the local escape speed or some other specified speed level. A restraint of minimum altitude of 130 kilometers was additionally imposed for the high speed stages, mainly to avoid excessive aerodynamic heating. The trajectory pattern resulting from this study is geometrically presented on Figure II.7. This family is coordinated to the launch azimuth of 90 degrees from north, but changes little for the two other chosen azimuth values of 70 and 110 degrees.

It is of interest to notice that the injection altitude for the 90° pitch case, which is the case of maximum payload capability, is at the imposed minimum limit of 130 kilometers, while for 70° pitch angle the orbit injection occurs near 900 kilometers altitude.

FIG. II.7 FLIGHT GEOMETRY OF PROPELLED PHASE OF SATURN B-I FOR TRAJECTORIES OF DIFFERENT STEEPNESS

Launching: East from AFMTC



Also remarkable is the range coverage of nearly 40 degrees earth's central angle, which is rather independent of the injection path angle.

For raising the injection velocity, 400 pounds of payload capacity must be converted into additional propellants to step the speed up by 100 m/s. However, a further payload loss ensues, since a higher arrival speed at the moon must be eliminated by the braking engine.

For the tracking problem as well as for the principal lunar rendezvous problem, the paths of the trajectories in their ground projections are relevant. These plottings are shown on Figure II.8. It may be mentioned that these trajectories are calculated on the rotating earth and that a lateral path guidance was imposed, which keeps the lateral velocity constant as far as this is accomplished by inertial measuring instruments.

A point to mention is the change in azimuth the trajectories experience between launch and injection. The following table shows the relationships:

Azimuth at Launch	Azimuth at Injection
70°	97°
90°	111°
110°	124°

<u>II.5.3 Description of the Trajectory and Orbit Chosen for the Lunar Landing Mission</u> (U). The pattern of injection possibilities, as discussed in the previous section, is a basic step toward the solution of the lunar rendezvous problem. A thorough discussion of this problem is retained for a later report. The method applied, however, may be briefly outlined.

A useful tool for a preliminary survey of the orbits is the central field assumption (zero mass of moon), for which all orbits develop within great circle planes, through the center of the earth. Projection of the vehicle orbits toward a celestial sphere coaxial with the earth shows the latitudes of the terminal orbit points. Also, the great circles show the inclination toward the equator, from which the approximate relationship to an inclination of the moon's orbit can be

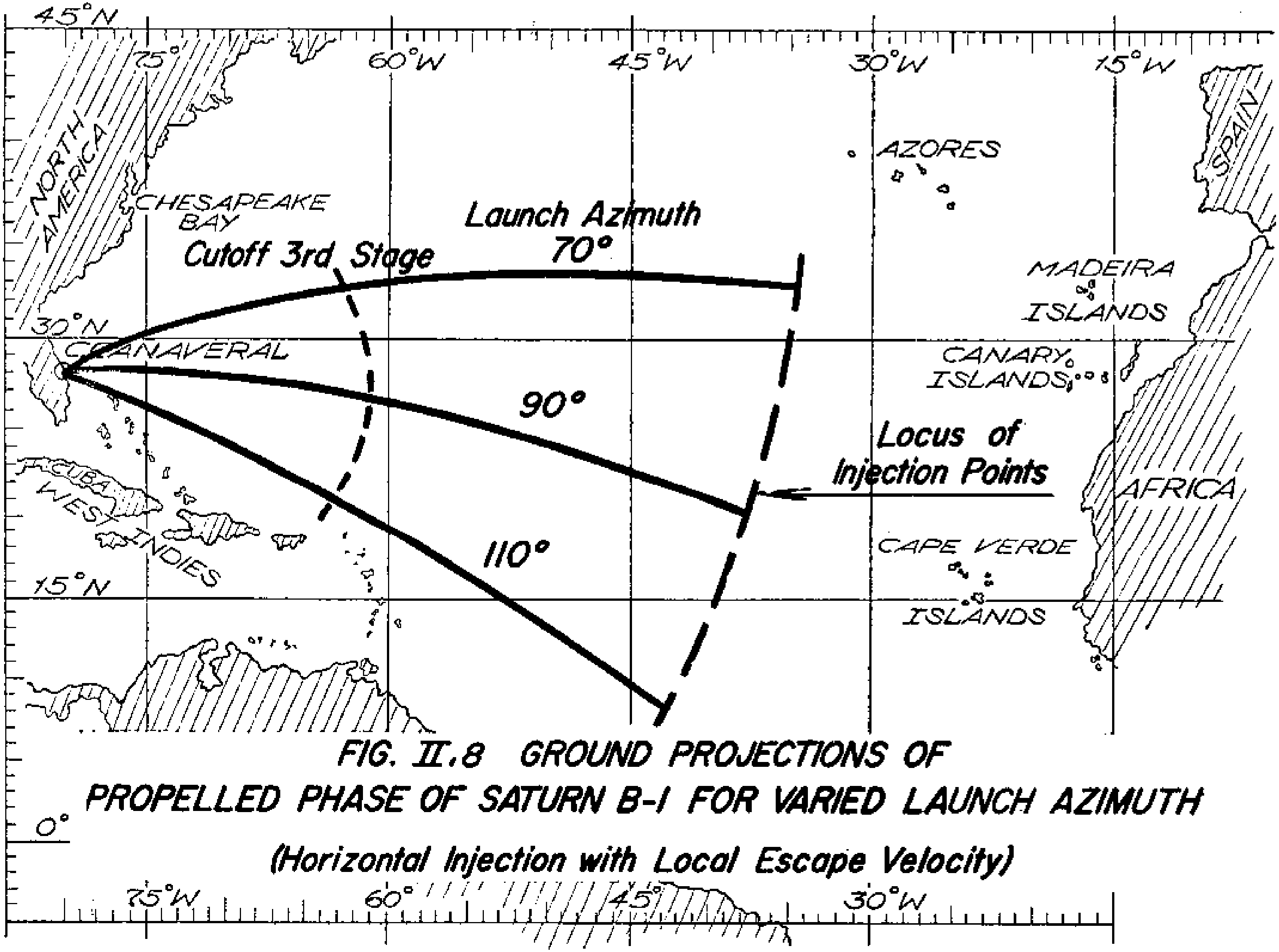


FIG. II.8 GROUND PROJECTIONS OF PROPELLED PHASE OF SATURN B-I FOR VARIED LAUNCH AZIMUTH

(Horizontal Injection with Local Escape Velocity) derived. Considerations of this type lead to the first approximation of the solution. After this, computation is applied to the true celestial model, and trial and error methods establish the final solution.

This procedure has been followed in establishing the nominal orbit as described in the following. The trajectory of the powered phase is launched under the azimuth of 110 degrees east from north at the Air Force Missile Test Center.

Trajectory data for the most important flight parameters are listed in Table II.1. The data does not represent the exact trajectory for the listed injection condtions, but it is close enough to the exact trajectory that the values are representative for this.

The injection conditions of this orbit are as follows:

Path Angle (from local vertical)	85.0°
Azimuth (east from north)	123.62°
Latitude (north)	9.6°
Longitude (west)	44.0°
Altitude	305 km
Velocity (in rotating coord. system)	10,808 m/s
Velocity (in space-fixed coord. system)	10,832.8 m/s
Lunar Declination	-25.0°
*Lunar Lead Angle	125.91°

The orbit projection onto the celestial sphere is shown in Mercator projection on Figure II.9. The vehicle covers a central angle of more than 195° . The path goes through a latitude minimum of minus 34 degrees at the first hour of orbital flight and raises before the rendezvous to the desired latitude of minus 21.3 degrees. Also, the projection of the moon's orbit is plotted on the graph.

The vehicle path in its projection on the rotating earth is depicted on Figure II.10. Time after injection is labeled to

*The lead angle is measured by the projection onto the equator of the angle between injection point and moon, measured at earth center.

TABLE II.1

POWERED FLIGHT DATA OF SATURN B-1 VERSION FOR A TRAJECTORY APPROXIMATING THE POWERED PHASE FOR THE LUNAR PROBE ORBIT AS DESCRIBED

Time (sec)	Ground Distance (km)	Altitude (km)	Earth Fixed Velocity (m/s)	Longitudinal Inertial Accel. (m/s^2)	Path Angle against local vertical (deg)	Angle of Attack ** (deg)	Gyro Attitude against local vertical at launch (deg)
0	0	0	0	9.80	0	0	0
14	.02	.31	47.57	13.68	3.59	2.16	1.11
26	.08	1.19	99.75	14.59	5.21	1.48	3.87
50	.75	5.14	244.82	16.83	13.31	0	13.54
74	3.34	12.82	439.81	18.73	23.31	0	23.65
* 97	9.89	24.68	773.60	27.03	33.58	- .97	35.04
111	16.88	34.10	913.68	18.92	39.58	- 2.23	42.40
159	63.08	72.72	1707.49	27.57	58.29	- 4.65	64.11
207	165.58	119.99	3243.26	48.56	70.20	- 3.03	75.49
* 217.2	198.58	131.53	3751.71	57.93	71.77	- 2.70	77.06
231.2	247.96	147.38	3807.95	6.92	73.22	- 3.39	79.69
327.2	616.86	229.16	4344.48	8.25	81.76	- .06	88.54
423.2	1051.13	268.34	5147.77	10.22	87.59	1.15	97.39
519.2	1573.41	272.25	6260.73	13.41	91.02	- 1.73	108.73
* 575.98	1936.90	261.06	7116.31	16.45	92.11	- 3.94	115.49
604.0	2131.45	253.04	7353.42	8.35	92.28	- 2.54	116.11
700.0	2852.79	224.94	8277.72	10.37	91.60	1.49	118.20
796.0	3672.03	218.43	9422.58	13.67	88.89	4.38	120.29
865.85	4340.59	250.89	10475.74	17.80	85.52	5.73	121.82

* Cut-off   ** Measured Positive Nose Down

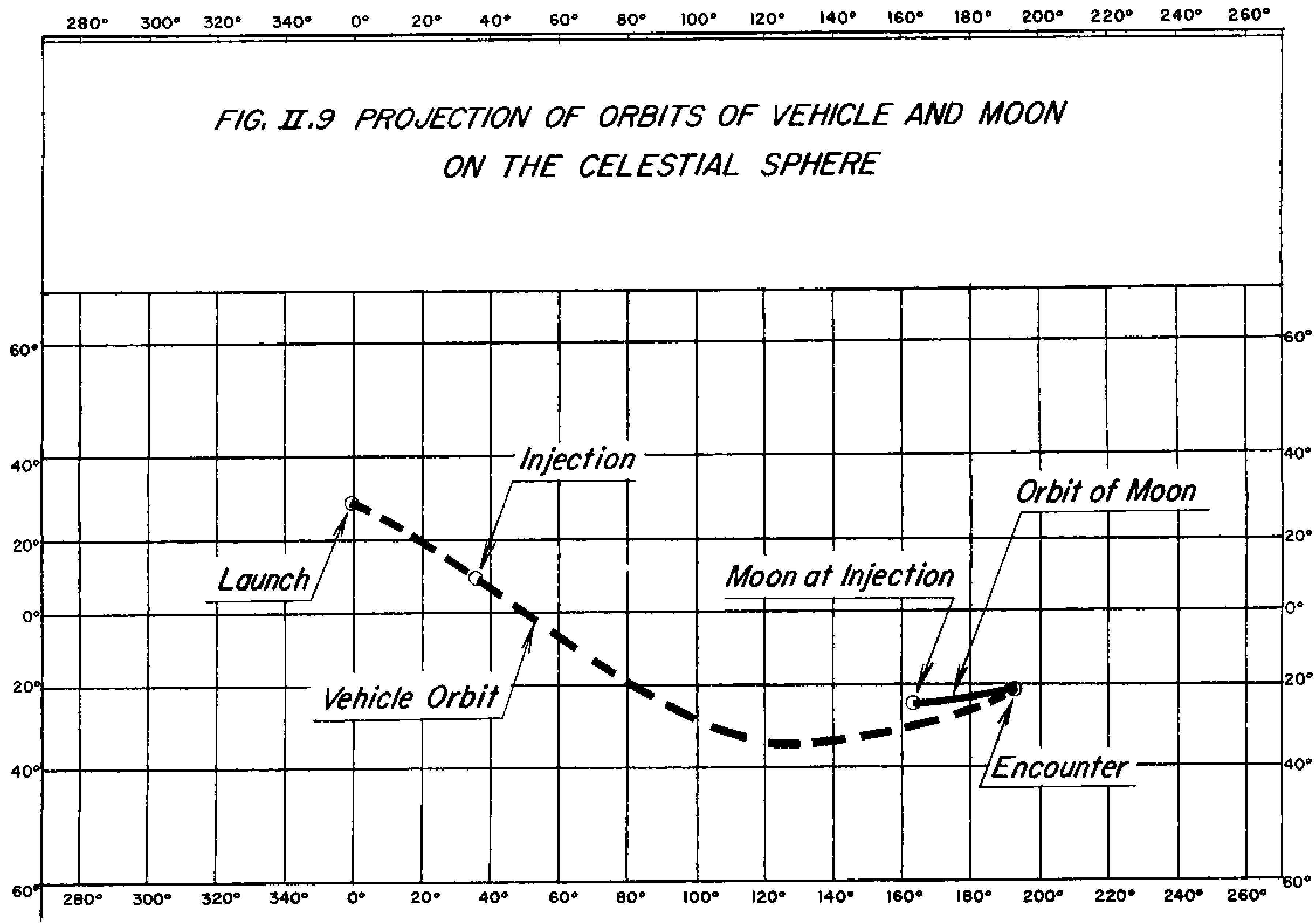
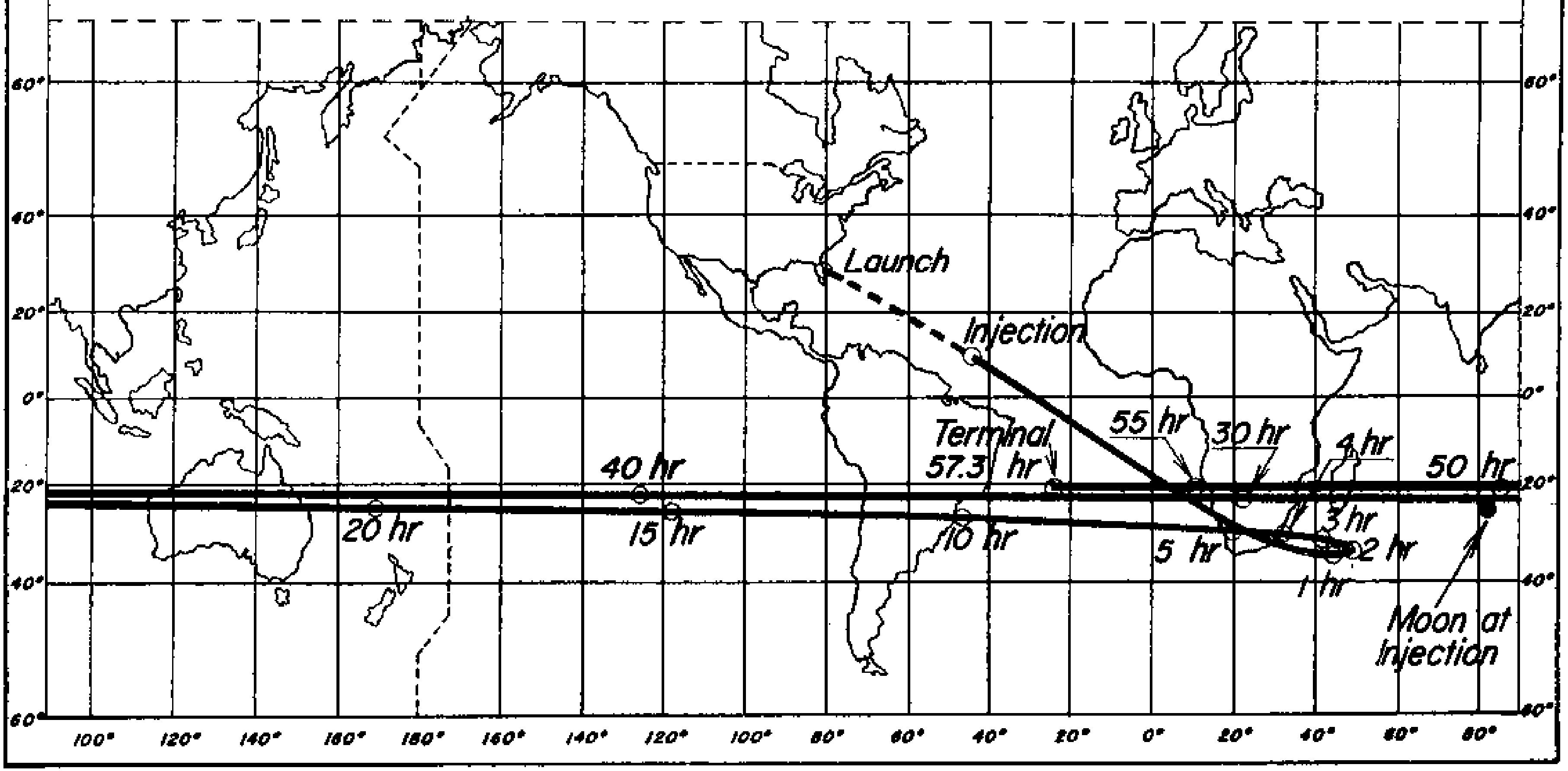


FIG. II.9 PROJECTION OF ORBITS OF VEHICLE AND MOON ON THE CELESTIAL SPHERE

FIG. II.10 PROJECTION OF THE VEHICLE ORBIT ON THE ROTATING EARTH

Figures Inscribed Refer to Time After Injection



the curve. From this picture, suitable tracking stations can be picked as function of travel time. It shows in particular the favorable location of South Africa for tracking the vehicle from the first half hour to the sixth hour of injection.

The geometry of the orbit between earth and moon is illustrated by Figures II.11 and II.12. The first shows, in the form of a three-walled box, the three reference coordinate planes which are (Plane A) the moon's motion plane, (Plane B) the plane perpendicular to it, going through earth and moon at encounter, and (Plane C) the plane perpendicular to both former planes, going through the earth's center. The orbit is shown within this reference frame in a sketch-like fashion.

Figure II.12 shows the projection of the orbit on the three coordinate planes, with time inscribed in hours from injection.

The data for the rendezvous conditions, ignoring the braking history, are listed as follows:

Time from injection	57.3 hrs
Velocity (perpendicular ballistic impact)	2726.9 m/s
Impact locations (measured with respect to moon's center)	
(angle between projection of radius vector in the lunar plane of motion and the earth-moon line)	36.1°
(angle between radius vector and lunar plane of motion)	1.1°

For convenient reference, a plotting of the velocity as function of time is included (Figure II.13) referencing velocity to the coordinate system that rotates with the earth - moon line. The minimum speed is reached near the 46th hour of the flight with about 1450 m/s velocity. The ballistic impact would occur after the 57th hour with about 2700 m/s velocity.

The impact on the moon is calculated to be perpendicular to the moon's surface, as observed from the moon. The impact location, if measured from the center of the moon, is at 36 degrees from the earth-moon line toward the leading edge, and at one degree above the plane of the moon's motion.

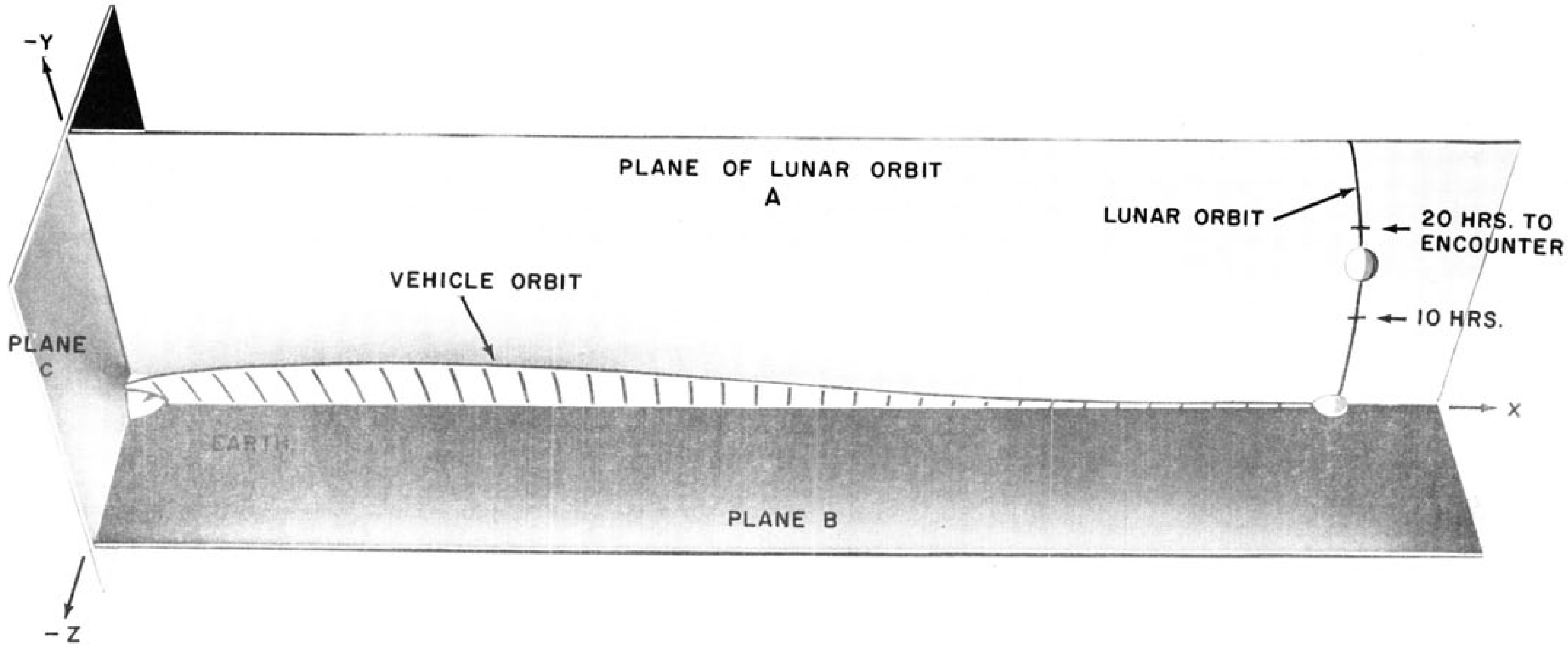


FIG. II.11 GEOMETRY OF VEHICLE'S PATH TO MOON

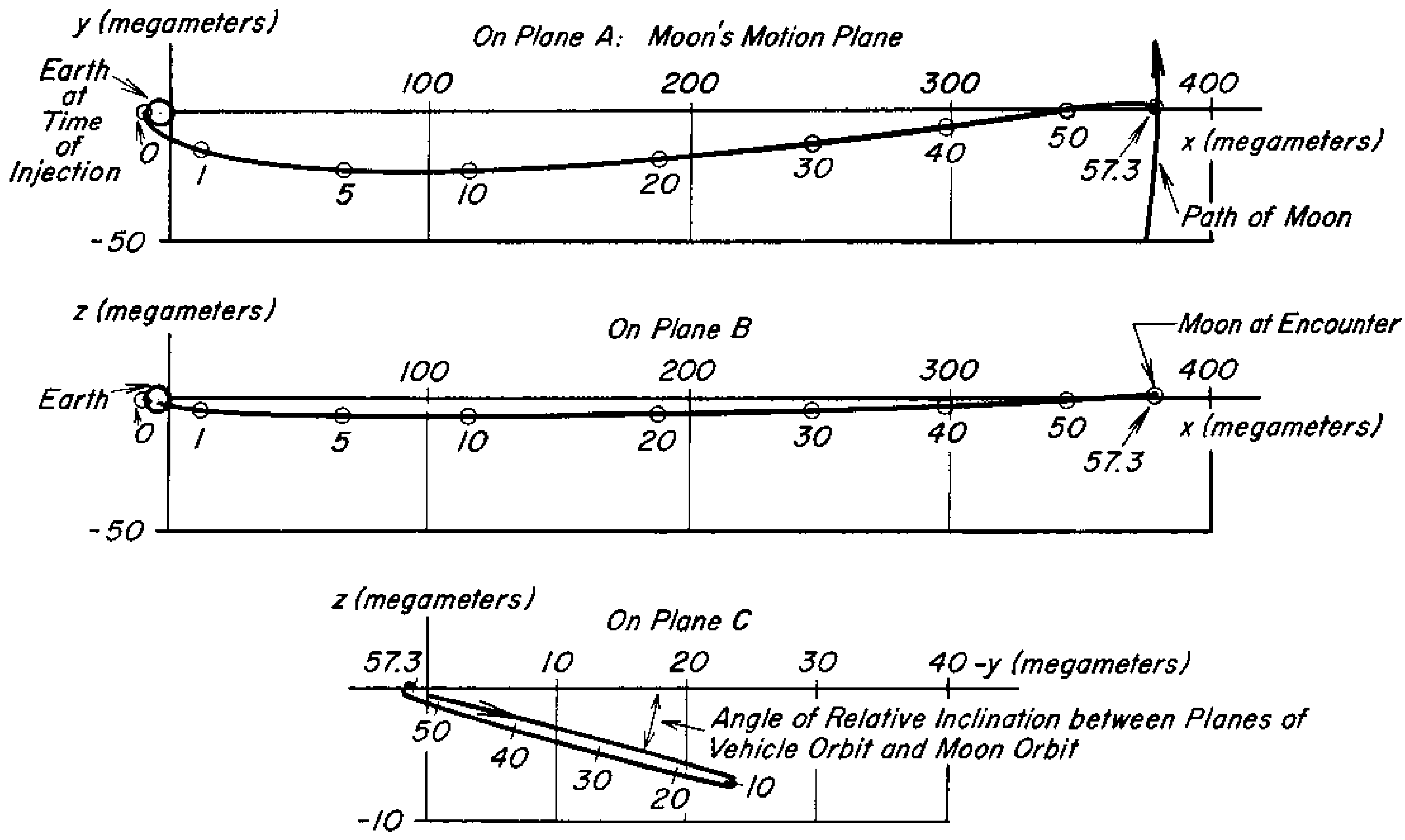


FIG. II.12 PROJECTION OF THE VEHICLE ORBIT ON THE THREE SPACE-FIXED PLANES OF PRECEDING FIGURE

Figures Labeled Refer to Time after Injection in Hr

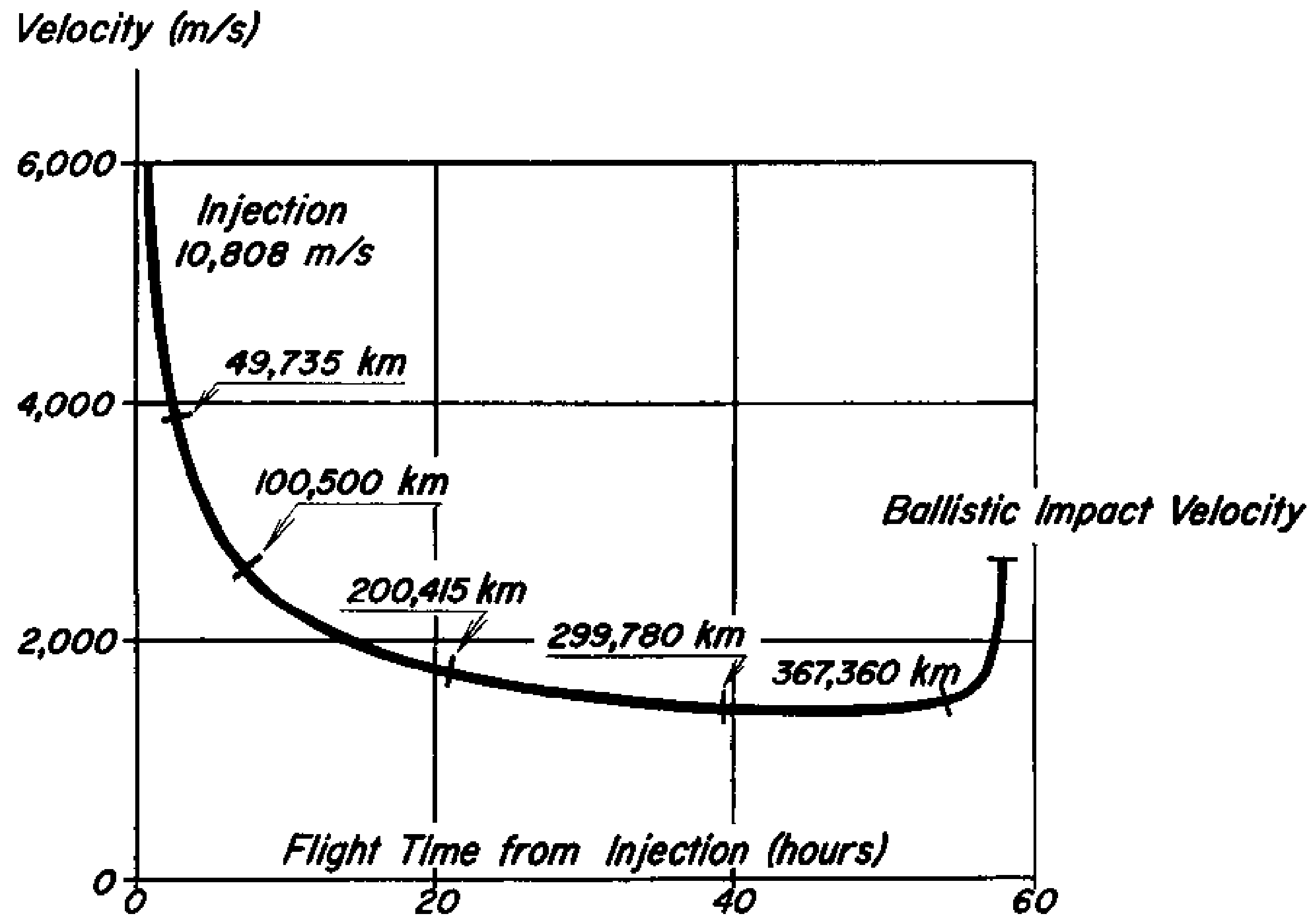


FIG. II.13 VELOCITY MEASURED WITH RESPECT TO ROTATING SYSTEM
Inscriptions Refer to Distance of Vehicle from Earth's Center

The location is shown on Figure II.14.

This figure exhibits also the pattern of impact point shifts for erroneous injection conditions. To assure impact somewhere on the moon, the restrictions on the injection accuracy are of the following magnitude (provided that no further corrections take place):

Path Angle	$\pm .26°$
Azimuth	$\pm .9°$
Latitude	$\pm .35°$
Lead Angle	$\pm .5°$
Altitude	$\pm$ 10 km
Velocity	$\pm$ 8 m/s

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

	Path Angle	$\pm .0057°$
	Azimuth	$\pm .025°$
	Latitude	$\pm .0096°$
	Lead Angle	$\pm .012°$
or	Lead Time	± 2.88 secs
	Altitude	$\pm .24$ km
	Velocity	$\pm .19$ m/s

Any combination of errors reduces the individual allowances listed.

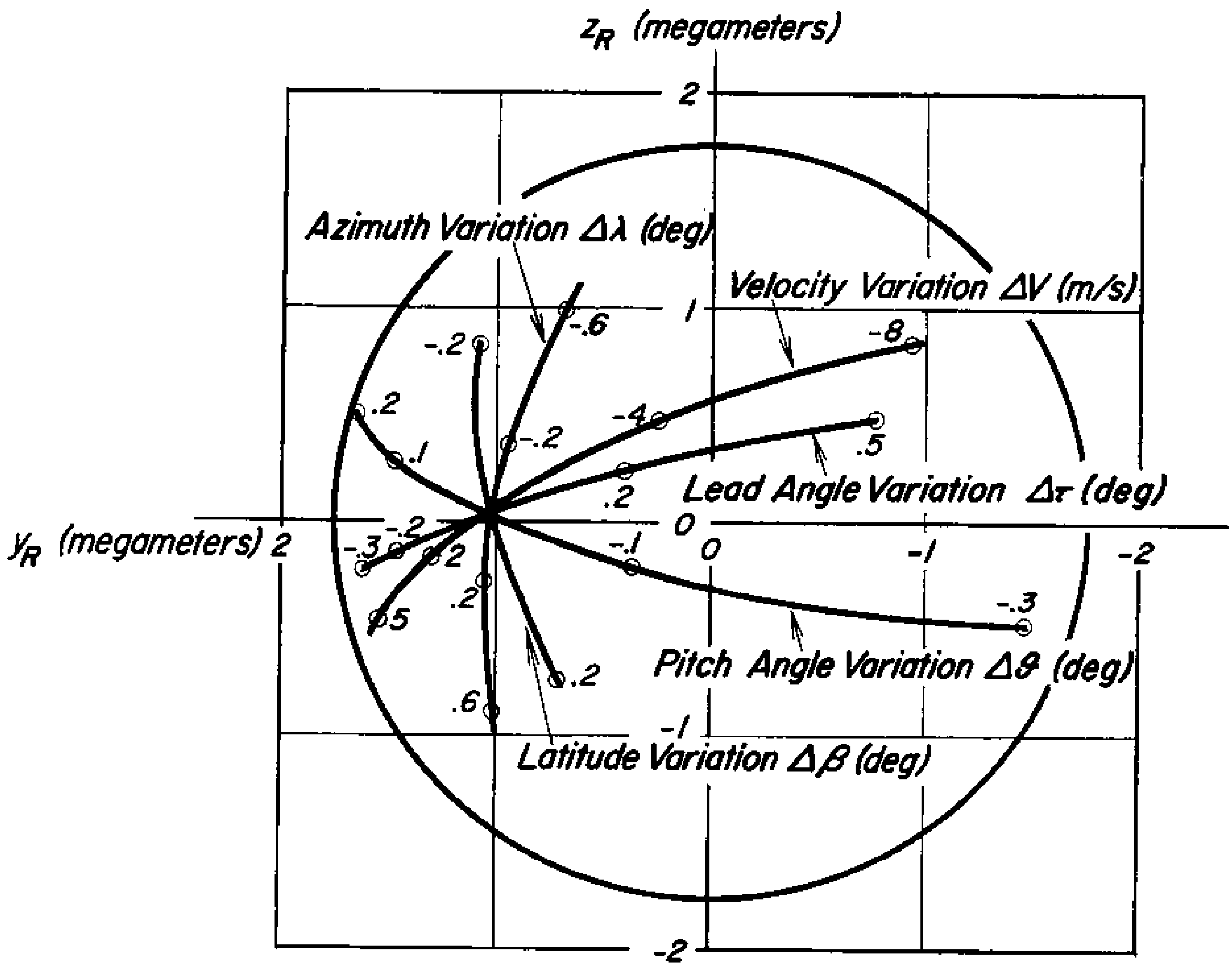


FIG. II.14 IMPACT PATTERN FOR INDEPENDENT INJECTION PARAMETER VARIATIONS

II.6 TERMINAL GUIDANCE SCHEME FOR SOFT LUNAR LANDING (C)

II.6.1 Introduction. A TV camera with adequate transmission to earth is considered to be of primary importance for the lunar exploration by softly landed vehicles. Also, television can be used advantageously within a guidance system and compares favorably with other possible schemes. It thus seems logical to concentrate on a terminal guidance system that takes advantage of the TV equipment on-board. This appears all the more appropriate since considerable experience with a TV system capable of taking and transmitting pictures of the earth's surface from a descending TV capsule already exists. This particular development, which was conducted under ABMA's technical supervision during the past two years, has advanced to the point that flight tests with the REDSTONE missiles are scheduled for the period January through May 1960.

The general landing scheme includes a series of distinct steps above the lunar surface (see Figure II.15). There will be three power-on periods during each of which the descending velocity will be reduced to a very small fraction of the initial velocity. During the intermediate power-off periods, while the vehicle slowly picks up velocity again, guidance signals from the earth will be applied. These signals will be given by a human operator who uses TV picutres from the landing vehicle, and properly prepared computers, to arrive at the required guidance signals. The inertial system monitors the vehicle during the powered periods when it is assumed that TV and radio measurements cannot be made because of rocket flame interference. It has not been determined whether radio measurements can be made satisfactorily through the exhaust flame. More research work has to be done in this field. The conservative approach, that it will be impractical to use radio through the rocket flame, is taken in this study.

Breaking up the approach phase into several powered and unpowered periods does not appear at first glance to be the most economical way of landing as far as fuel consumption is concerned. A single deceleration period ending with zero velocity on the ground would be expected to be the least energy-consuming descent. However, there are three distinct advantages to the multi-step scheme:

(1) Certain errors reduce themselves automatically in case a program of alternate braking and free fall periods is chosen. This effect is independent of

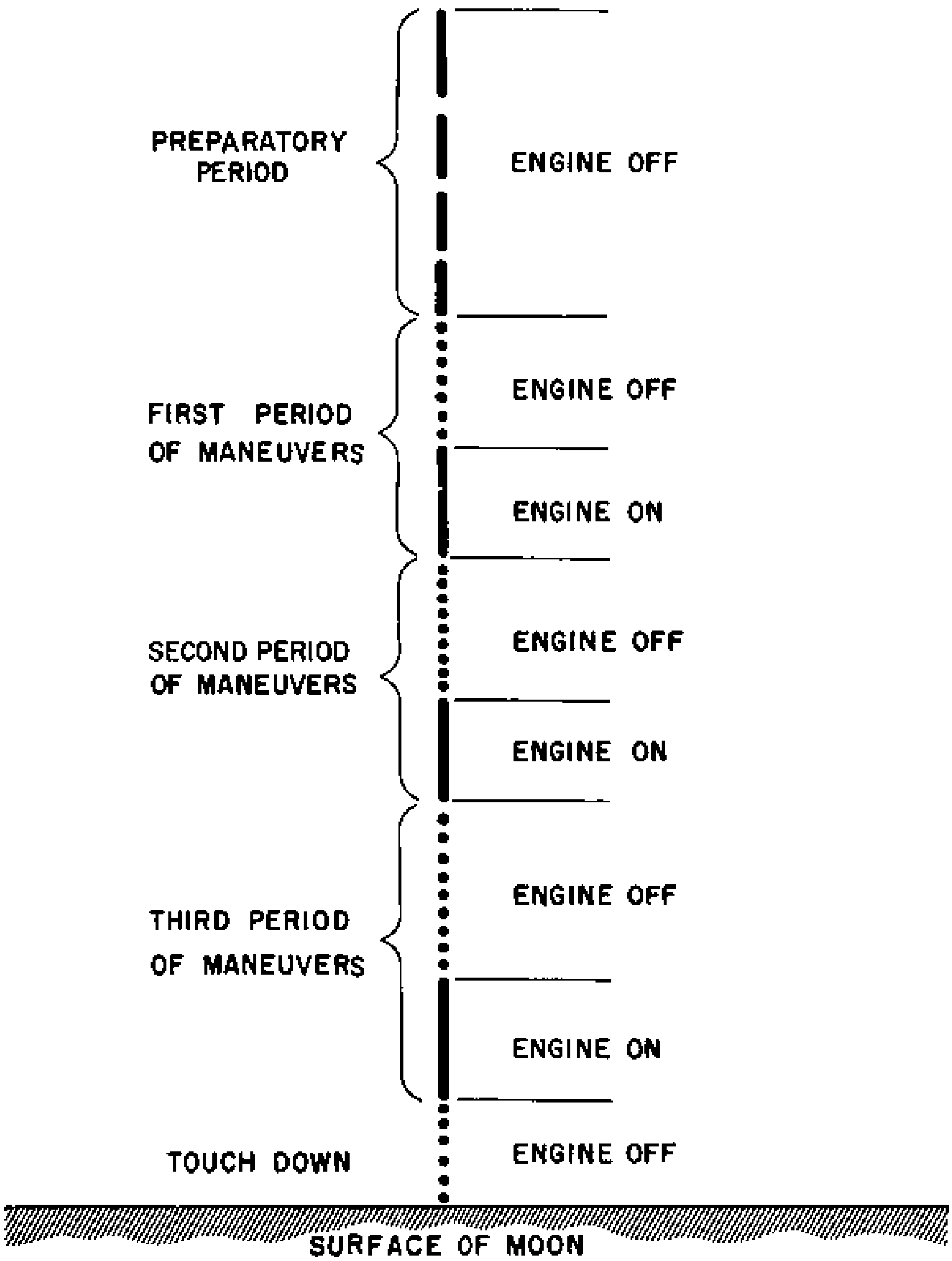


FIG. II. 15
TERMINAL GUIDANCE SEQUENCE OF MANEUVERS

any measurements taken at any time.

(2) During the free fall period, a readjustment of the initial parameters is made by increasingly refined measurements. In other words, the starting conditions of the guidance and control system are re-established several times, thereby, making successive improvements in over-all accuracy.

(3) Any necessary lateral maneuvers with respect to the surface of the moon may be accomplished over a longer time period since the free fall periods prolong the total landing procedure.

If large lateral displacement maneuvers are required this latter advantage results in an actual savings in fuel in contrast to the expected fuel penalty for the multi-step system. The first two advantages listed under (1) and (2) lead to a very simple scheme for fulfilling the location requirements of the instrument carrier landings. The simplicity is in the guidance hardware. It requires only equipment that either is available today, or is in the developmental stage with good promise of being available within the next 24 months. Except for the guidance and control equipment necessary for attitude control and for lateral maneuvers, radio altimeter and an inertial platform is all that is essential to guide the vertical descent in this scheme. A preset program is triggered by the altimeter and monitored by the inertial platform. This "minimum effort scheme" could even be accomplished with one constant-thrust engine with conventional thrust level control. An engine like the fourth stage $H_2 + O_2$ engine could probably be used. Appendix D explains the argument in favor of this simple scheme and also cites its limitations and its adaptability to future more sophisticated schemes for extremely soft landings.

The ultimate goal in this scheme is to perfect the landing system to the point where the ratio of the intermediate free fall periods to the powered periods is optimized with respect to fuel consumption and desirable lateral maneuverability. Furthermore, it is part of the final goal to incorporate more of the functions which are now proposed for performance on earth, into the vehicle itself. Thus, the vehicle is intended to eventually become completely self-contained and land on the moon with a closed-loop optical inertial guidance and control system without input from earth. It is felt that the proposed TV-inertial guidance with radio equipment for position determination is well suited for this step-by-step development program.

Preliminary investigations into the final version of this scheme are encouraging. However, more detailed knowledge of the fine structure of the lunar surface is necessary before the merits of a fully automatic landing scheme can be appraised.

II.6.2 Equipment. The analysis of the terminal guidance scheme is based upon the following assumptions concerning trajectory, injection guidance, vehicle design and availability of equipment:

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

(2) Landing anywhere within an area of 40 km radius from the nominal point of ballistic impact is satisfactory.

(3) The injection guidance, improved by a vernier and possibly mid-course correction, is sufficiently accurate to insure ballistic impact within a 40 km radius of nominal impact.

(4) Coarse correction in the lateral direction, with respect to the moon's surface, need not exceed 20 km. A scrutiny of lunar maps shows that a displacement of 20 km within practically any reasonable landing area of 40 km radius will change the topography decisively. In other words, a displacement of 20 km will practically always bring the landing point away from a crater or mountain range into a more level terrain.

(5) Fine correction in the lateral direction need not exceed 100 meters.

(6) The braking engine will have a thrust of 20 K. Investigations so far indicate that a constant thrust, re-ignitable engine will be satisfactory. It is very likely that an $H_2 + O_2$ engine of the type used in the fourth stage will be a proper choice.

(7) A system of small nozzles with 90 per cent controllable thrust is available for attitude control of the vehicle.

(8) A command receiver with appropriate body-fixed antenna is available.

(9) A television system is available insuring uninterrupted TV link to earth once the vehicle approaches the surface of the moon to within 1500 kilometers.

In addition to the above equipment the terminal guidance system uses the following components:

(1) Inertial platform ST-300 with three mutually orthogonal accelerometers.

(2) Two radio altimeters, one for long range, up to 300 or 400 km and one for short range, 15 km and down. There is promise that both altimeters can be combined into one electronic package.

(3) Infrared horizon seeker of 0.1 degree accuracy for establishing the local vertical.

(4) Sun seeker.

(5) Guidance computer.

(6) Control computer.

(7) A picture-evaluating unit of the image correlator type with automatic scale adjustment of pictures (stationed on earth).

(8) Earth-bound computer for prediction of impact point and for calculation of guidance signals and of the necessary change in control parameters for arriving at a new landing site (stationed on earth).

(9) Picture display unit (stationed on earth).

II.6.3 General Sequence of Maneuvers for Descent. The general sequence of maneuvers is sketched in Figure II.16. There are several preparatory steps, followed by essentially one type of maneuver which repeats itself three times. The description to follow will be confined to outlining the maneuver in one axis only. The actual maneuver is two-dimensional.

Figure II.16

SEQUENCE OF MANEUVERS FOR DESCENT

		Altitude Velocity	Time to Impact
		7000 km	60 min
Preparatory Steps	Stabilization of roll position (sun seeker, attitude control nozzles) First alignment, longitudinal axis parallel to local vertical (horizon seeker, attitude control nozzles) Test of television system and command link Altitude measurement, h, at one minute intervals (horizon seeker)		
		1500 km	12 min
First Period	Second alignment, longitudinal axis parallel to local vertical (horizon seeker, attitude control nozzles) Inertial platform uncaged Operation of television system (10 frames per minute) Determination of average lateral velocity, $\dot{x}$ (image correlator) Altitude measurement, h (horizon seeker) Determination of velocity vector ($\dot{x}$ and known total velocity) Determination of landing coordinated (computer on earth) Determination of first set of initial conditions, $\Delta\dot{x}_0$ (computer on earth)		
		273 km 2500 m/s	286 sec
	First power-on maneuver - coarse lateral correction (by command, based upon $\Delta\dot{x}_0$) Monitoring by inertial platform Ignition based upon altitude (altimeter or horizon seeker) Cut-off based upon inertial measurement of vertical velocity change, $\Delta\dot{y}$		70 sec
		151 km 800 m/s	216 sec

(cont'd)

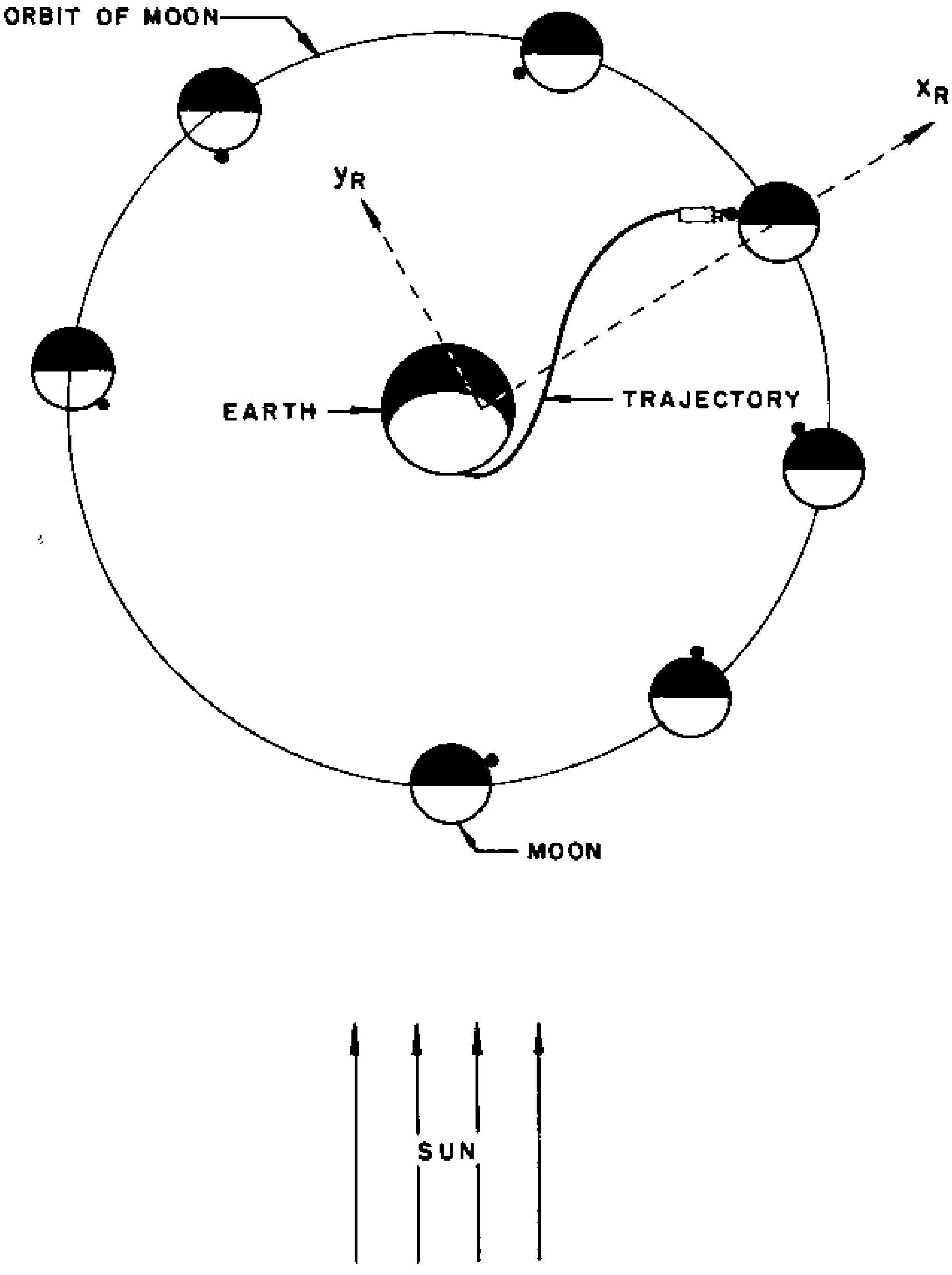
Second Period	First free fall - coarse lateral displacement build-up Operation of television system (10 frames/min) Determination of average lateral velocity, $\dot{x}$ (image correlator) Altitude measurement, h (altimeter) Determination of landing coordinates (computer on earth)		147 sec
		15.8 km 1040 m/s	69 sec
	Second power-on maneuver - elimination of lateral velocity except for fine correction Monitoring by inertial platform Ignition based upon altitude (altimeter) Cut-off based upon inertial measurement of vertical velocity change, $\Delta\dot{y}$		25 sec
		1800 m 25 m/s	44 sec
Third Period	Second free fall - fine lateral displacement build-up Operation of television system (10 frames/min) Determination of average lateral velocity, $\dot{x}$ (image correlator) Altitude measurement, h (altimeter) Determination of landing coordinates (computer on earth) Determination of third set of initial conditions, $\dot{x} = \Delta\dot{x}_0$, x_0, h (computer on earth)		31 sec
		226 m 76 m/s	13 sec
	Third power-on maneuver - final braking to $\dot{x} \approx 0$, $\dot{y} = 0$, over landing spot Final lateral correction by x_0 command Monitoring by inertial platform Ignition based upon altitude (altimeter) Cut-off based upon inertial measurement of vertical velocity change, $\Delta\dot{y}$		4 sec
		60 m 1 m/s	9 sec
Touchdown	Third free fall Separation of engine and payload package Touchdown on lunar surface, 14 m/sec		
		0 14 m/s	0 Landing

Preparatory Period - First among the preparatory steps is the refined stabilization of the roll position. Establishing the roll position is not an absolute must but a desirable redundancy over the bare minimum requirements for the soft landing. The stabilization takes place well in advance of the actual maneuvers at about 7,500 km altitude or 65 minutes before impact on the surface of the moon. The roll position is sensed by a sun seeker. A system of eight small attitude control nozzles serves as the actuator for moving the vehicle until it is locked onto the sun. The general direction of the sun will be in the direction of the negative Y-axis, due to the fact that the vehicle should land on the moon at the beginning of a lunar day (see Figure II.17). A command signal from earth or a timer may initiate the roll stabilization. The roll orientation will be allowed a backlash of about $\pm$ 4 degrees. This is sufficient for carrying out the next step and will save fuel by having the attitude control nozzles work only intermittently.

The second of the preparatory steps is a warm-up and check-out of the terminal guidance equipment. This also takes place well in advance of the actual maneuvers at about 7,000 km altitude or 60 minutes before impact on the surface of the moon. The horizon seeker is energized. It is body-fixed and has the reference axis of its sweep angle parallel to the longidutinal axis of the vehicle. In conjunction with the attitude control nozzle system, it will line up the longitudinal axis of the vehicle with the local vertical of the moon (see Figures II.18 and 19). The vehicle will then have the required attitude for establishing the TV link to earth. The TV transmission will be tested by starting up the camera and TV transmitting facilities on-board and the image correlator and picture display unit on earth. The initiating signal will be given through the command link from earth, thus testing the command link.

The test program will provide assurance that the equipment on both ends is working properly. It will also familiarize the human operator on earth with the actual working conditions. This is of importance since he must later, by evidence of the TV picture, make the decision whether the landing site should be changed. The test will give the operator a feel for the equipment for 5 or 10 minutes with 10 TV pictures per minute.

After the testing is completed the TV equipment will go on a stand-by for the rest of the preparatory period. The horizon seeker will stay energized and with the attitude control nozzles will keep the vehicle lined up to within 4 degrees of the local vertical. Furthermore, during the entire



SCHEMATIC OF TRAJECTORY FOR LANDING
WITH LUNAR DAWN
FIG.II. 17

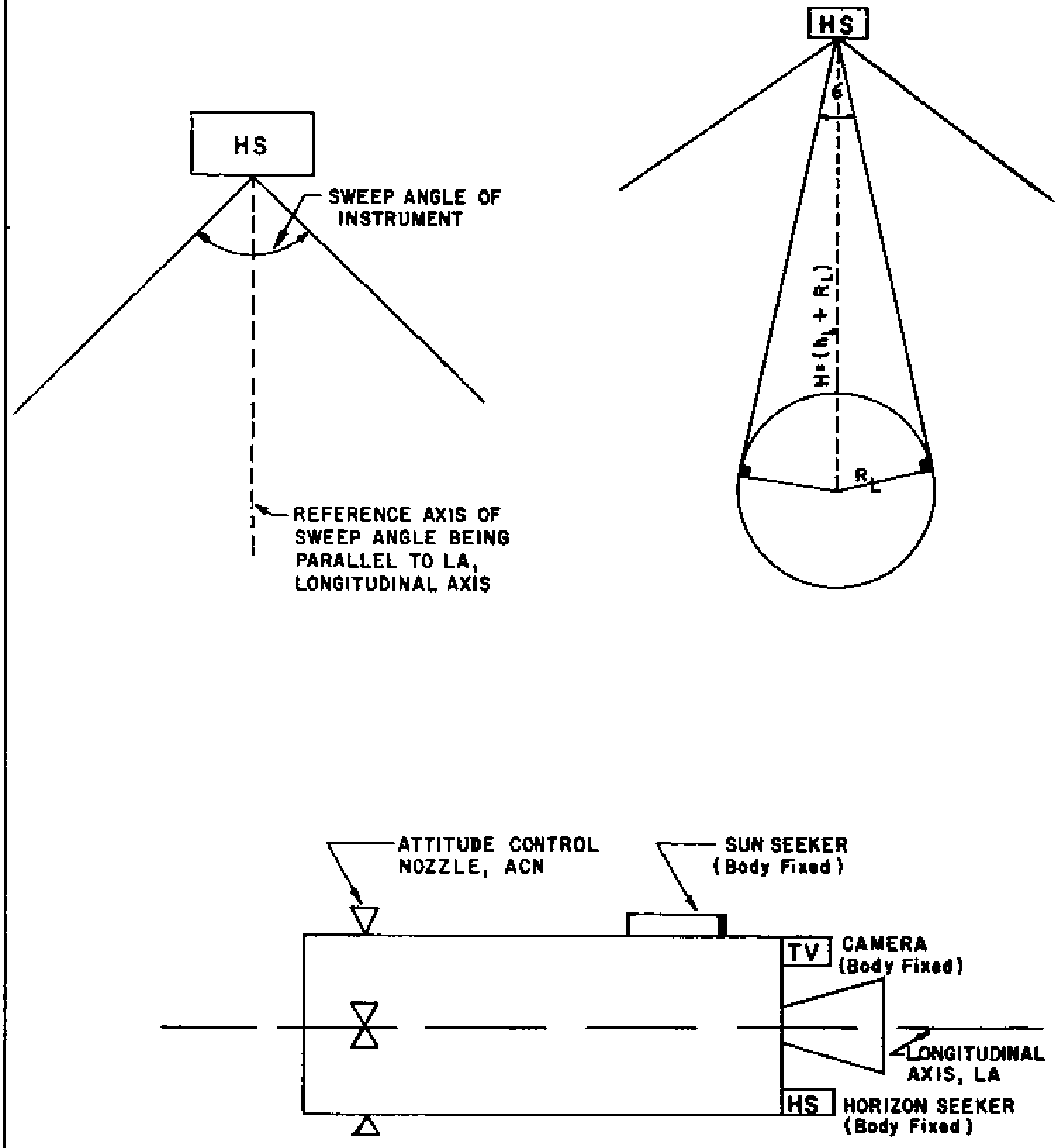


FIG. II. 18

SCHEMATIC OF ARRANGEMENT OF SENSORS

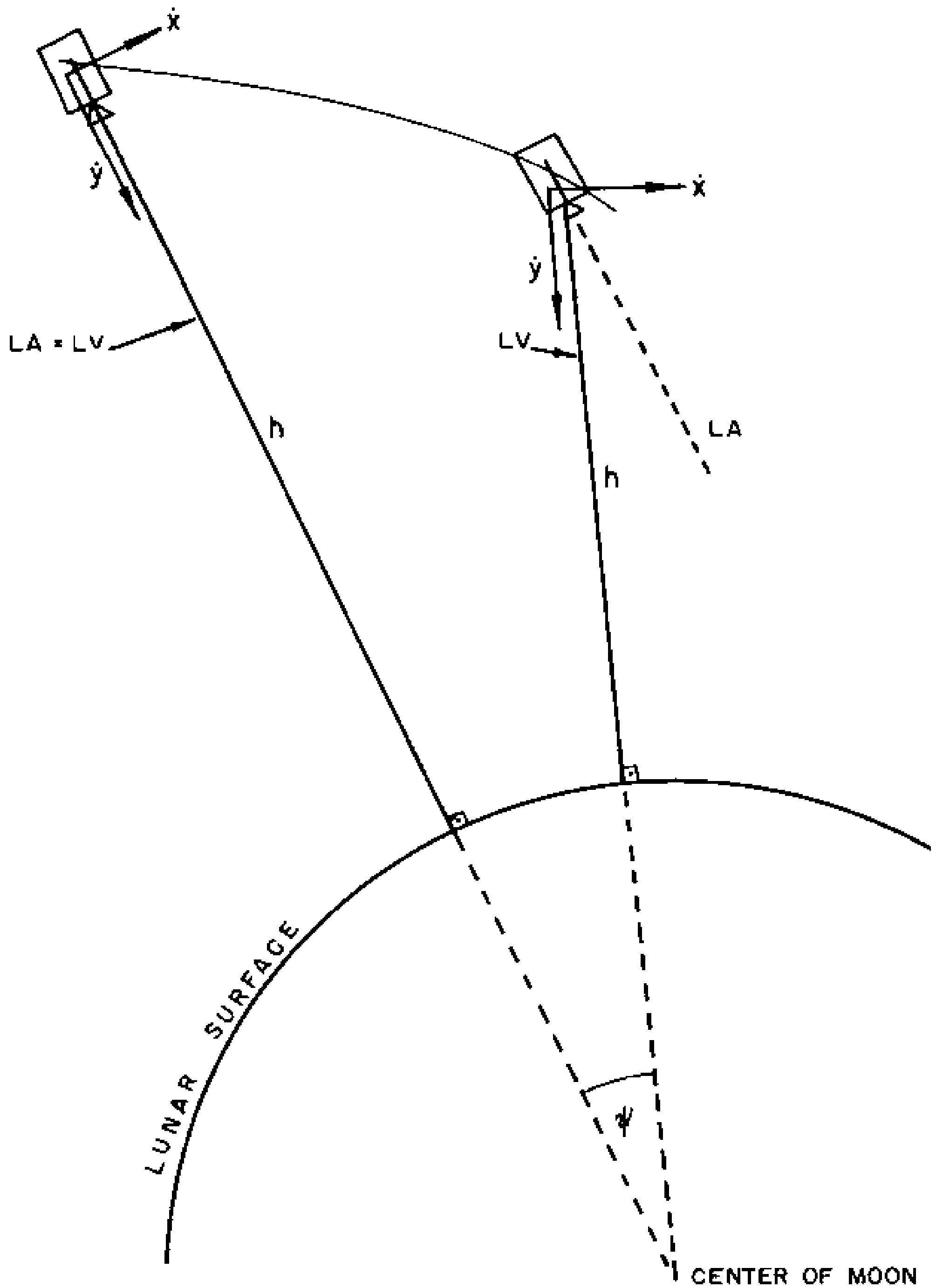


FIG. II. 19
GEOMETRIC RELATIONS BETWEEN VEHICLE AND MOON

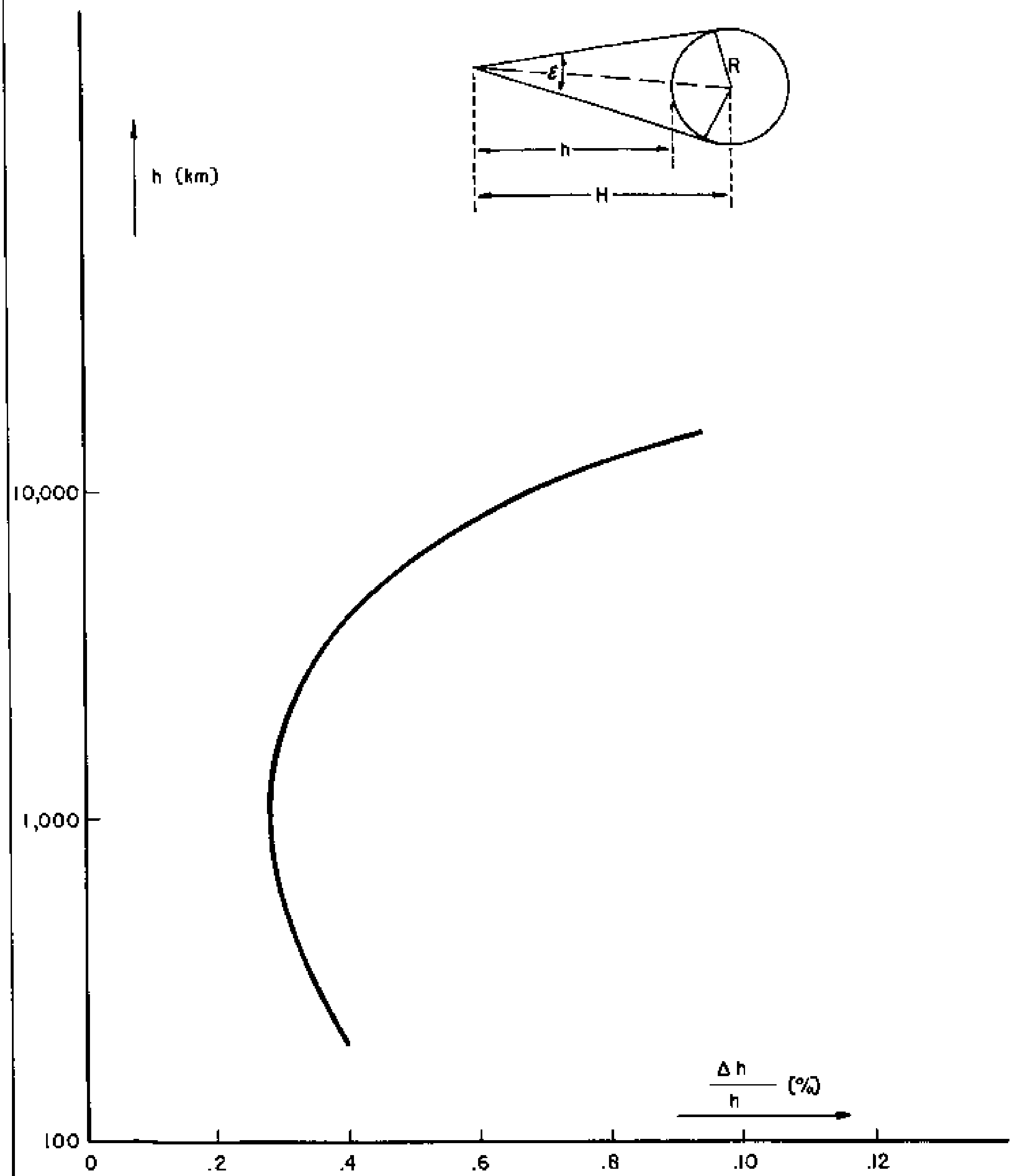
preparatory period the horizon seeker will obtain readings of the height of the vehicle above the lunar surface at regular intervals of about 30 seconds for transmission to earth. The accuracy of determining the altitude with a horizon seeker at the altitudes under consideration here is better than one per cent (see Figures II.18,19,20,21 and Appendix E).

As a result of the preparatory steps the proper functioning of the critically important TV link and command link between vehicle and earth will be ascertained. Also, the human operator on earth will have a chance to get a feel for the type of TV picture he will receive. Furthermore, the altitude of the vehicle above the lunar surface will be known.

If, however, these tests should prove that the television and command link do not work satisfactorily, then an emergency landing scheme will be initiated. This emergency descent deviates from the standard terminal approach. It does not guarantee a landing as softly as planned but it still provides a good chance for landing the vehicle safely. The essential difference is that all equipment designed to carry out the lateral maneuvers is deactivated, and the landing is performed by a system based only on inertial platform, altimeter, and computed velocity data, without further reference to signals from the earthbound station.

Maneuvers - Having gone through the preparatory period, the actual landing maneuvers will follow. There is essentially only one type of maneuver which is repeated three times. It consists of two parts: the first part is a period of measuring and decision making during free fall with the engine off; the second part is a period when the engine is on and the aforemade decisions are executed. The sequence of events during the first period will be the following (see Figures II.16 and 22):

At a height of 1500 km, or about 12 minutes before touch down, the sun seeker and attitude control nozzle system will establish the roll position accurately to within 0.1 degree. Following immediately, the horizon seeker and attitude control nozzle system will, by command from earth, line up the longitudinal axis of the vehicle with the local vertical to within an accuracy of 0.1 degree whereupon the inertial platform will be uncaged; it thereafter provides a space-fixed reference which is maintained to within 0.01 degree throughout the remaining landing operation.



HORIZON SEEKER
ACCURACY OF ALTITUDE MEASUREMENTS BASED ON ± 0.1° ACCURACY OF SENSOR

FIG. II.20

~~CONFIDENTIAL~~

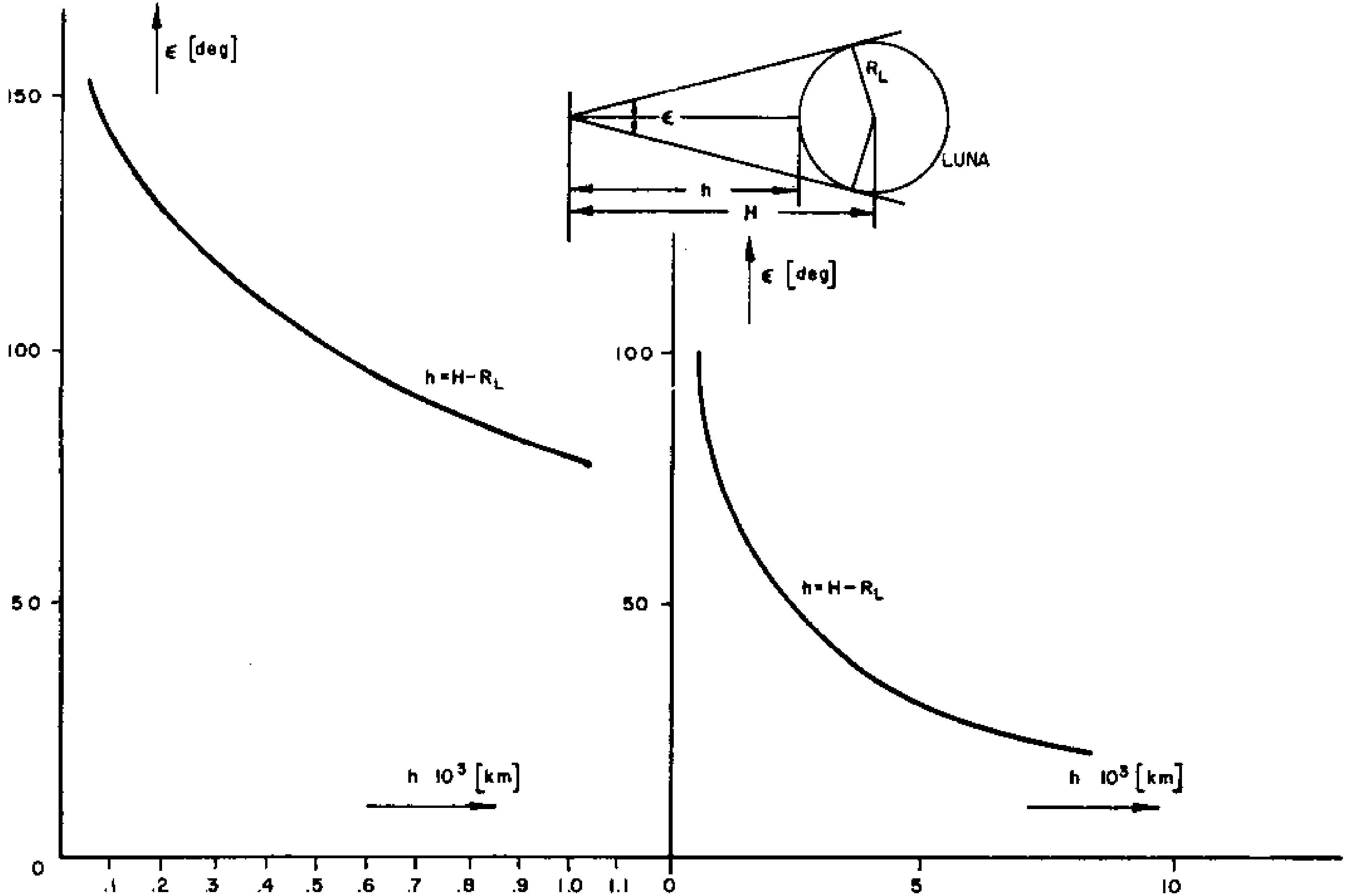


FIGURE II. 21
ANGLE SUBTENDED BY THE MOON AS A FUNCTION OF ALTITUDE

~~CONFIDENTIAL~~

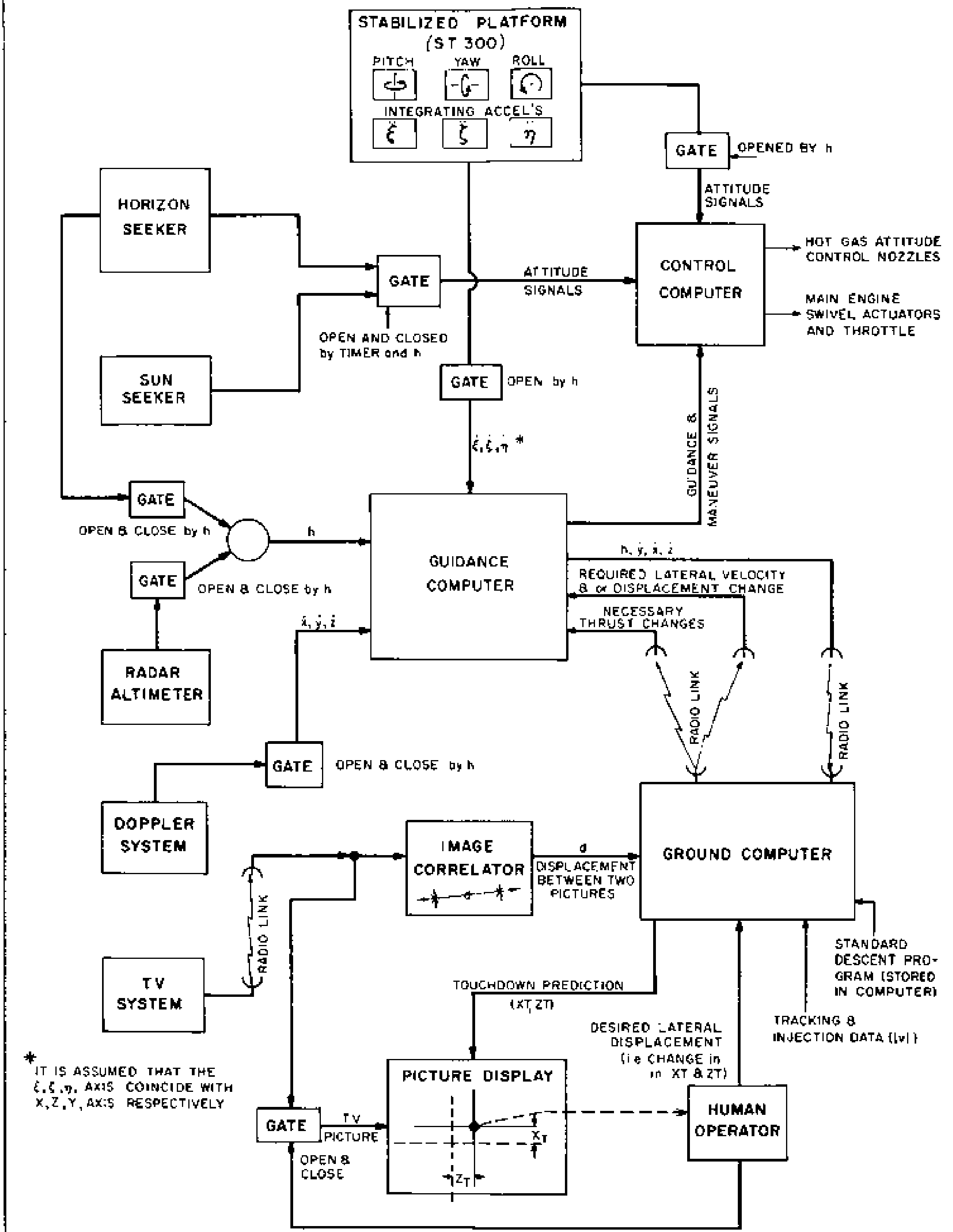


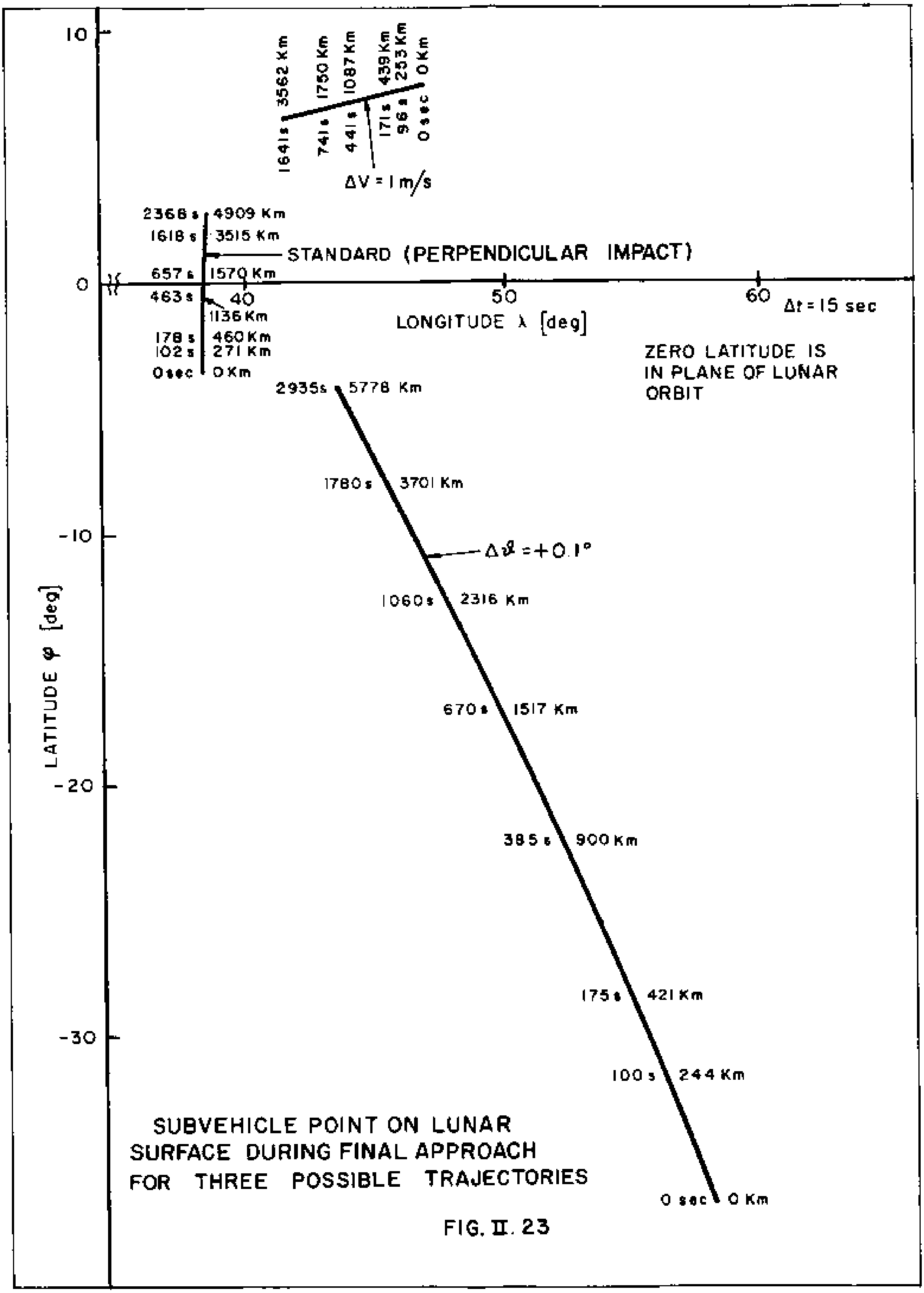
FIG. II. 22
TERMINAL GUIDANCE BLOCK DIAGRAM

The TV system will start taking pictures at the rate of 10 frames per minute at 1 millisecond shutter speed. The earth station will thus receive TV pictures at 6-second intervals and with about 1 second delay. The image correlator on earth together with a digital computer on earth will determine from two consecutive pictures and the altitude information the average lateral velocity, $\dot{x}$, of the vehicle with respect to the surface of the moon. The accuracy with which $\dot{x}$ can be measured depends on altitude. At 1500 km it is in the order of 20 m/s; at 20 km it is about 2 m/s. From an altitude of 20 km on down, velocity measurement with the Doppler method becomes feasible and is a possible alternative for the image correlation method.

If the surveying and measuring functions are separated by using two cameras the accuracy in determining $\dot{x}$ is increased considerably. Additional information can be gained from the surveying camera by increasing its angle of vision up to a thousand kilometer base line on the lunar surface. The advantage of the survey is that the human operator on earth can estimate the trajectory actually flown. He could do this by comparing the landscape under the vehicle with prepared charts of the landscape as seen from a given trajectory and from a given height above the lunar surface. Figure II.23 demonstrates how the projection of the approach trajectory onto the lunar surface is related to the actual trajectory. For the sake of illustration, extremely perturbed trajectories are pictured in Figure II.23. Velocity measurements of 5 per cent accuracy at an altitude of about 1500 km may not be sufficient to distinguish between trajectories, since the drastic changes in velocity occur rather late, and only for greatly perturbed trajectories (see Figure II.24). However, the location over the lunar surface is always notably different even for relatively small deviations from the standard trajectory.

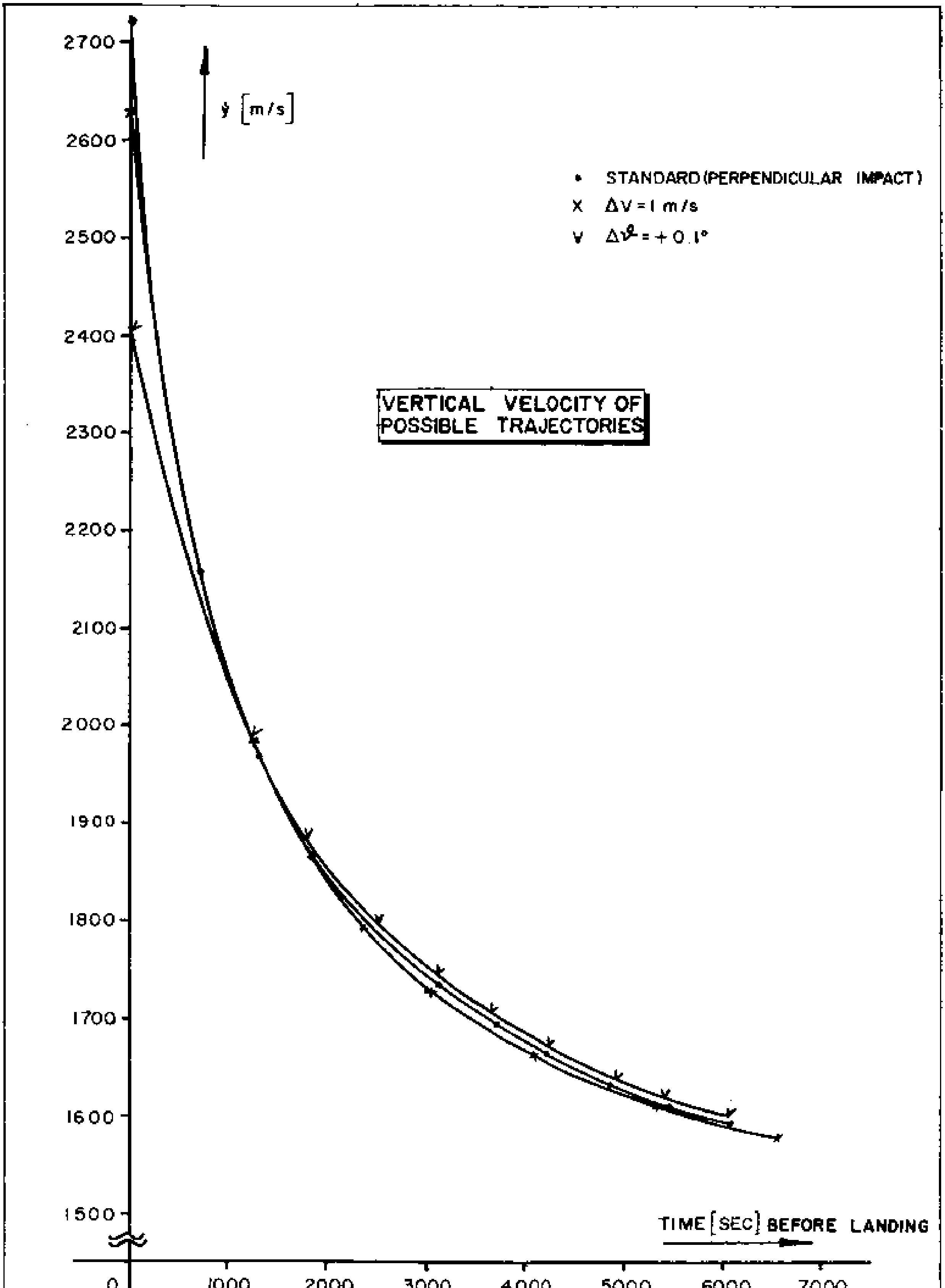
On the other hand, a narrow angle TV camera that could give $\dot{x}$ with higher accuracy will not show enough characteristic features of the surface to enable the human operator to recognize readily the particular part of the lunar surface. Extensive studies are continuing to weigh the pros and cons of the wide angle and narrow angle TV optics.

After the TV system has delivered the first two or three pictures, calculation of the landing coordinates then becomes possible. This involves computation of the trajectory from position and velocity data. The position data will be



SUBVEHICLE POINT ON LUNAR SURFACE DURING FINAL APPROACH FOR THREE POSSIBLE TRAJECTORIES

FIG. II. 23



$\dot{y}$ [m/s]
2700
2600
2500
2400
2300
2200
2100
2000
1900
1800
1700
1600
1500
• STANDARD (PERPENDICULAR IMPACT)
X ΔV = 1 m/s
V Δϑ = + 0.1°
VERTICAL VELOCITY OF POSSIBLE TRAJECTORIES
TIME [SEC] BEFORE LANDING
0
1000
2000
3000
4000
5000
6000
7000

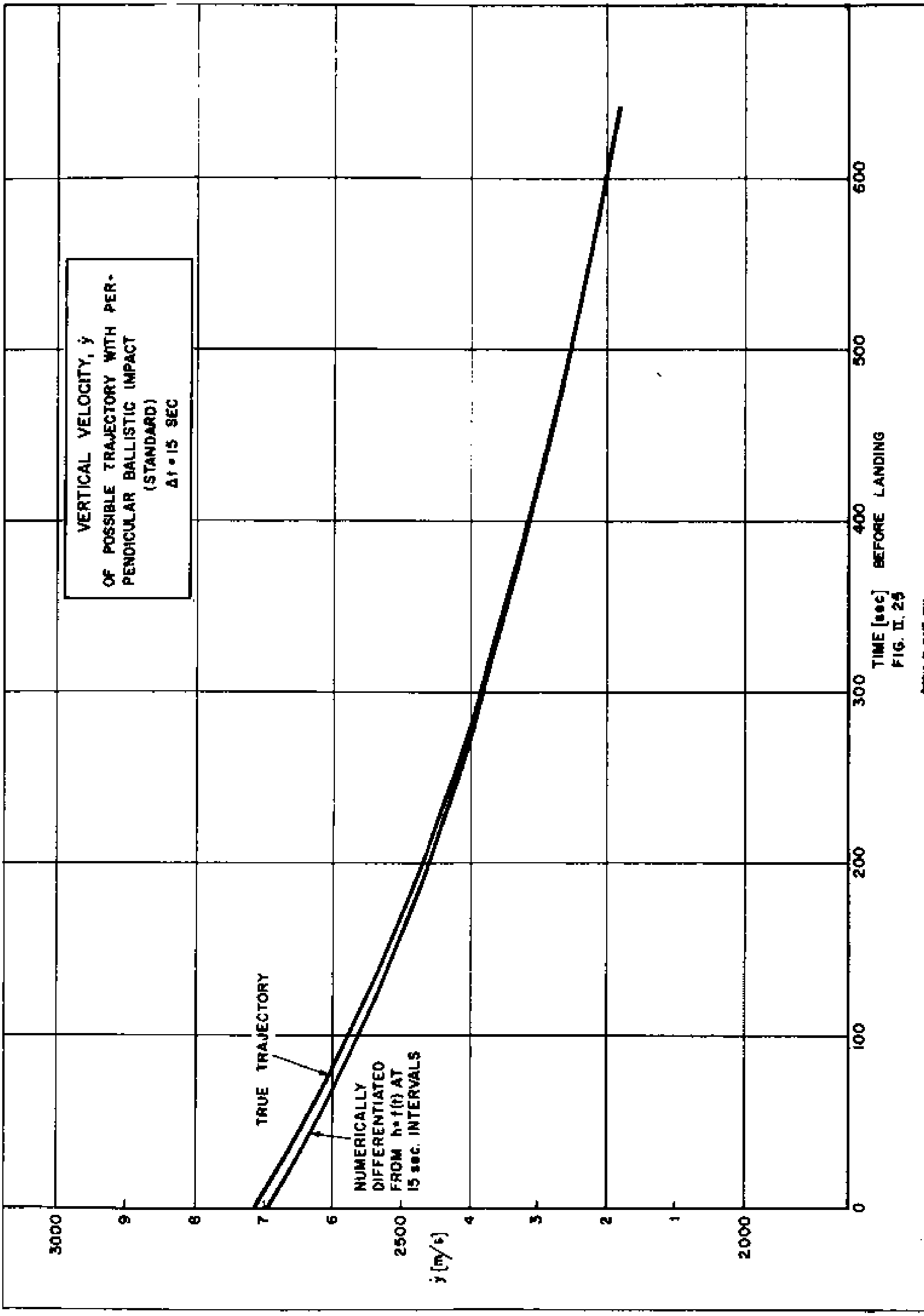
FIG. II. 24

available in the form of altitude and subvehicle point, the first being supplied by the horizon seeker or radio altimeter, the second taken from the TV picture. Velocity data are available in the form of $\dot{x}$ and the magnitude of the total velocity, the latter computed from injection and tracking data on earth. It is permissible to rely on the magnitude of the velocity vector as computed from data on earth, since this quantity will change very little for even strongly perturbed trajectories. As a rule of thumb, one may allow a 3 m/s difference in velocity at the moon for every 1 m/s deviation from the nominal value at injection (on earth). A total of 8 m/s error at injection, however, could already result in a trajectory that misses the moon all together.

In case of emergency, the vertical velocity, $\dot{y}$, could be derived by analog differentiating techniques presently being developed at ABMA. They would make use of the data on altitude being continuously secured by the horizon seeker during the preceding 45 minutes. Numerical differentiation of altitude, with time intervals of about 15 seconds, would yield a very good average vertical velocity, provided that no error in altitude information exists. This is shown in Figures II.25,26, and 27 where the true vertical velocity, $\dot{y}$, and the $\dot{y}$ obtained by numerical differentiation, are plotted against time for a standard trajectory with perpendicular ballistic impact and two perturbed trajectories. The differentiating interval is 15 seconds. Any errors in altitude, however, have a very strong effect on $\dot{y}$ as outlined in Appendix F. One may expect an over-all accuracy of 5 percent by this numerical method. The analog differentiating techniques might be able to avoid these coarse mistakes. The velocity data obtained by differentiating will serve to back up the scheme and take over if the velocity components cannot be determined in the normal way.

With lateral velocity $\dot{x}$, vertical velocity, $\dot{y}$, and height, h , being available with accuracies satisfactory for the first phase of approach, a computer on earth will calculate the trajectory and predict the coordinates of the lunar landing site.

The information is then fed into a picture display unit and is visible to the human operator either by moving the picture under a well-marked transparent grid, or by moving the grid over the picture, while the center of the grid marks the landing spot.



VERTICAL VELOCITY, ẏ
OF POSSIBLE TRAJECTORY WITH PER-
PENDICULAR BALLISTIC IMPACT
(STANDARD)
Δt = 15 SEC
TRUE TRAJECTORY
NUMERICALLY
DIFFERENTIATED
FROM h = f(t) AT
15 sec. INTERVALS
ẏ [m/s]
3000
2500
2000
9
8
7
6
5
4
3
2
1
0
100
200
300
400
500
600
TIME [sec] BEFORE LANDING

FIG. II. 25

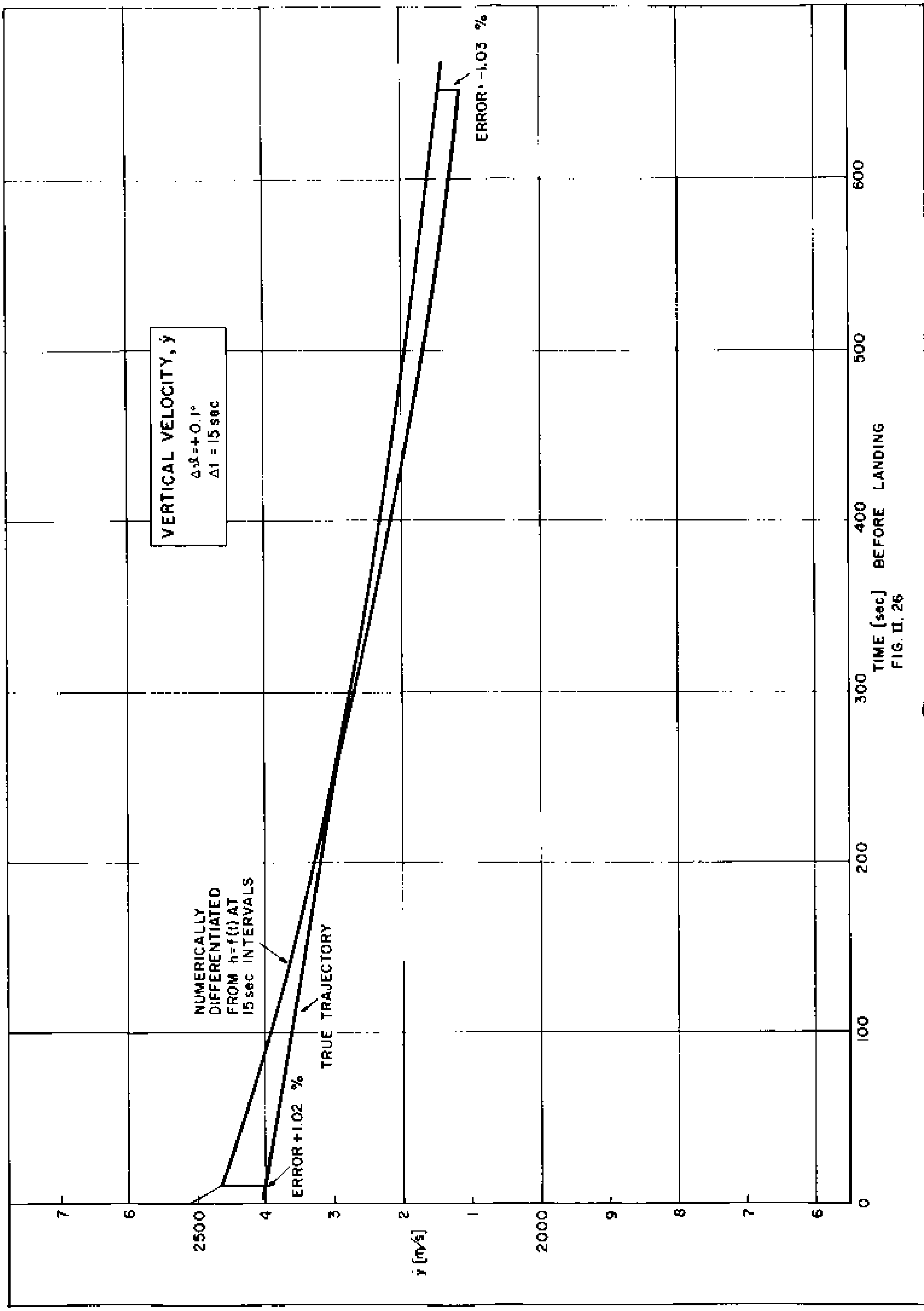

VERTICAL VELOCITY, ẏ
Δϑ = +0.1°
Δt = 15 sec
NUMERICALLY DIFFERENTIATED FROM h=f(t) AT 15 sec INTERVALS
TRUE TRAJECTORY
ERROR +1.02 %
ERROR -1.03 %
ẏ [m/s]
2500
2000
TIME [sec] BEFORE LANDING
0
100
200
300
400
500
600

FIG. II. 26

[redacted]

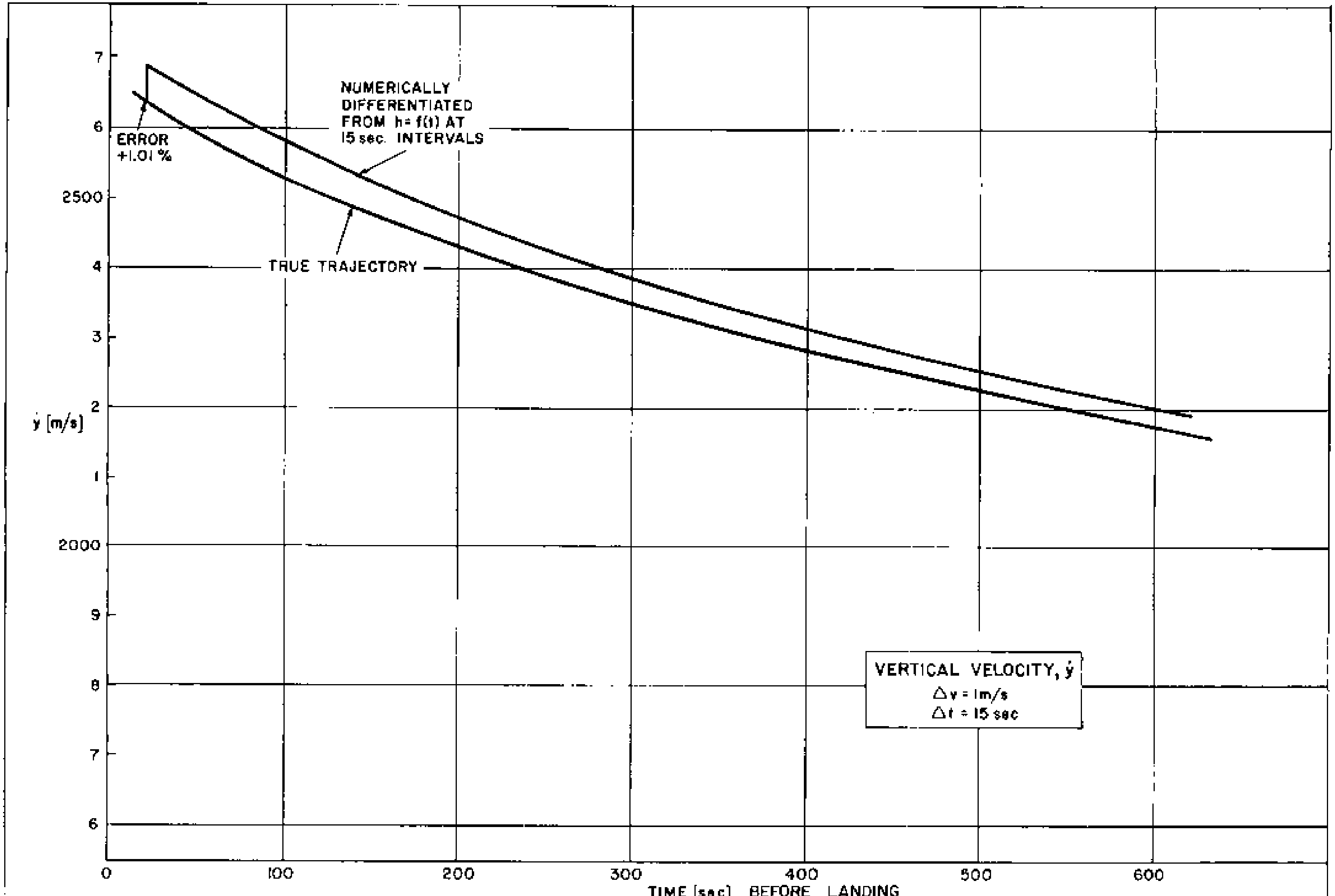


FIG. II 27

[redacted]

The first picture will also be fed through a gate into the display unit as a reference picture. The gate will shut off following pictures but may be opened again by the human operator when a new reference becomes desirable.

During the six-second interval between the second and third picture, the process of determining the coordinates of the landing spot will be run for the first time. As more pictures arrive on earth, the errors inherent in the measuring techniques and the processing will produce variations in the landing coordinates. Thus, a smoothing process suggests itself which will be left to the judgment of the human operator.

When the operator thinks the pictorial information on the landing spot is sufficient, he will decide whether a correction should be executed in the powered period to follow. If a correction is advisable he will determine the total coarse displacement to be made, and an earth-bound computer will calculate a lateral velocity change, $\Delta \dot{x}_o$, that will later carry the vehicle to the new position during these three periods: the first powered period, when the $\Delta\dot{x}$ builds up; the time of the first free fall period; and the time of the second powered period, when $\dot{x}$ is reduced to near zero again (see Fig. II.28).

The control equation for the swivel angle, β , of the engine against the longitudinal axis is expressed by (see Appendix G):

$$\beta = a_o\phi + a_1 \dot{\phi} + e_1 (\Delta \dot{x} - \Delta \dot{x}_o)$$

where ϕ is the angle between the longitudinal axis, and the local vertical. The earth-bound computer will have to select $\Delta \dot{x}_o$ such that the area under the velocity curve of Fig. II.28 amounts to the desired coarse displacement.

The initial tilting of the vehicle may either be incorporated into the control of the main engine or could be done by the attitude control system before the main engine is ignited. This latter method would increase the maneuverability if the burning time of the main engine should be comparable to the time constant of its control loop.

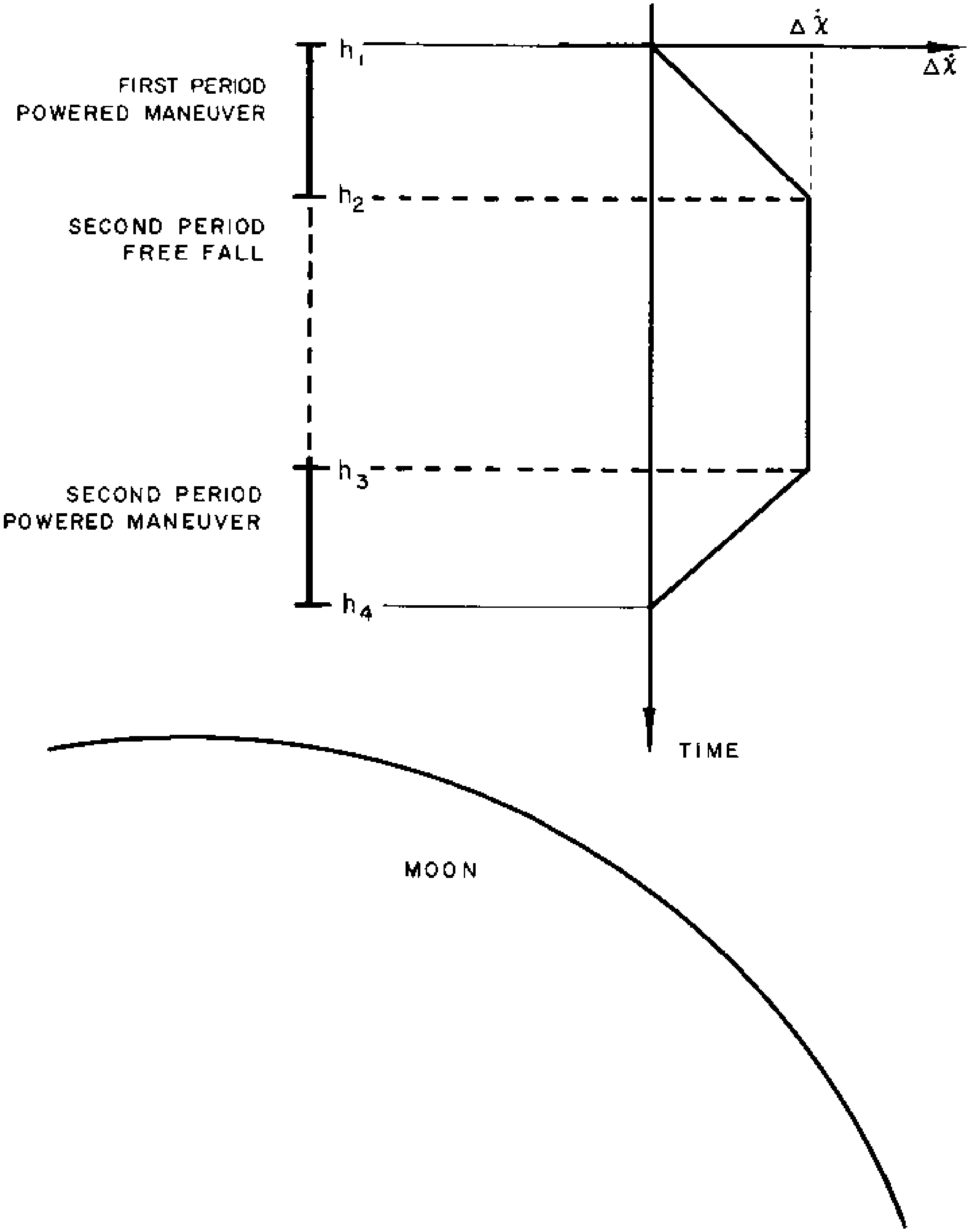


FIG. II. 28
TERMINAL GUIDANCE LATERAL VELOCITY MANEUVERS

The lengths of the powered periods and the free fall are standardized from general considerations of fuel availability, error minimization, total energy to be dissipated, and time necessary to take measurements between powered periods. Studies are under way to determine an optimum program. Typical values are shown in Figure II.16.

Ample time, about 10 minutes, is set aside for the first measuring and calculating period, mainly in order to allow the human operator to make his decision under a minimum of psychological strain.

There follows for the first time a period with the engine on. It begins by ignition of the braking engine. The moment of ignition is determined by altitude. Cut-off is given when the desired vertical velocity change, $\Delta\dot{y}_0$, is reached.

A typical example of the descent program is shown in Figure II.29. It is an extract from analogue computer studies on the subject. These studies are not completed yet. It is one of the key areas where work must continue in order to optimize the scheme and analyze the effects of perturbations.

Figure II.29 is indicative of what can be accomplished, particularly with respect to changes in lateral velocity and achieving lateral displacements. It will be noticed that a total lateral velocity change of $\Delta\dot{x}$ = 112 m/s is made. The strong effect of the free fall in the second period on the lateral displacement, x , is very obvious. At the end of the first braking period, only 7 km displacement have accumulated, but at the end of the free fall it is 26 km. The run given in Figure II.29 was made with the swivel angle β limited to 4° corresponding to a maximum angle of the vehicle axis, ϕ , equal to 12° . There are indications, resulting from the computer studies, that with the general vehicle dimensions given in this report a maximum $\Delta\dot{x}$ of 350 to 380 m/s can be achieved during the first powered maneuver and about 135 m/s in the second powered maneuver. Appendix G deals with the equations of motion that were used for these computer studies.

II.6.4 Characteristics of the Three Periods. Although the three periods basically do not differ with respect to the type of measurements taken and the method of executing commands, they nevertheless do have some differences. The primary functions of the periods differ, and so do the details of their instrumentatation used for measuring.

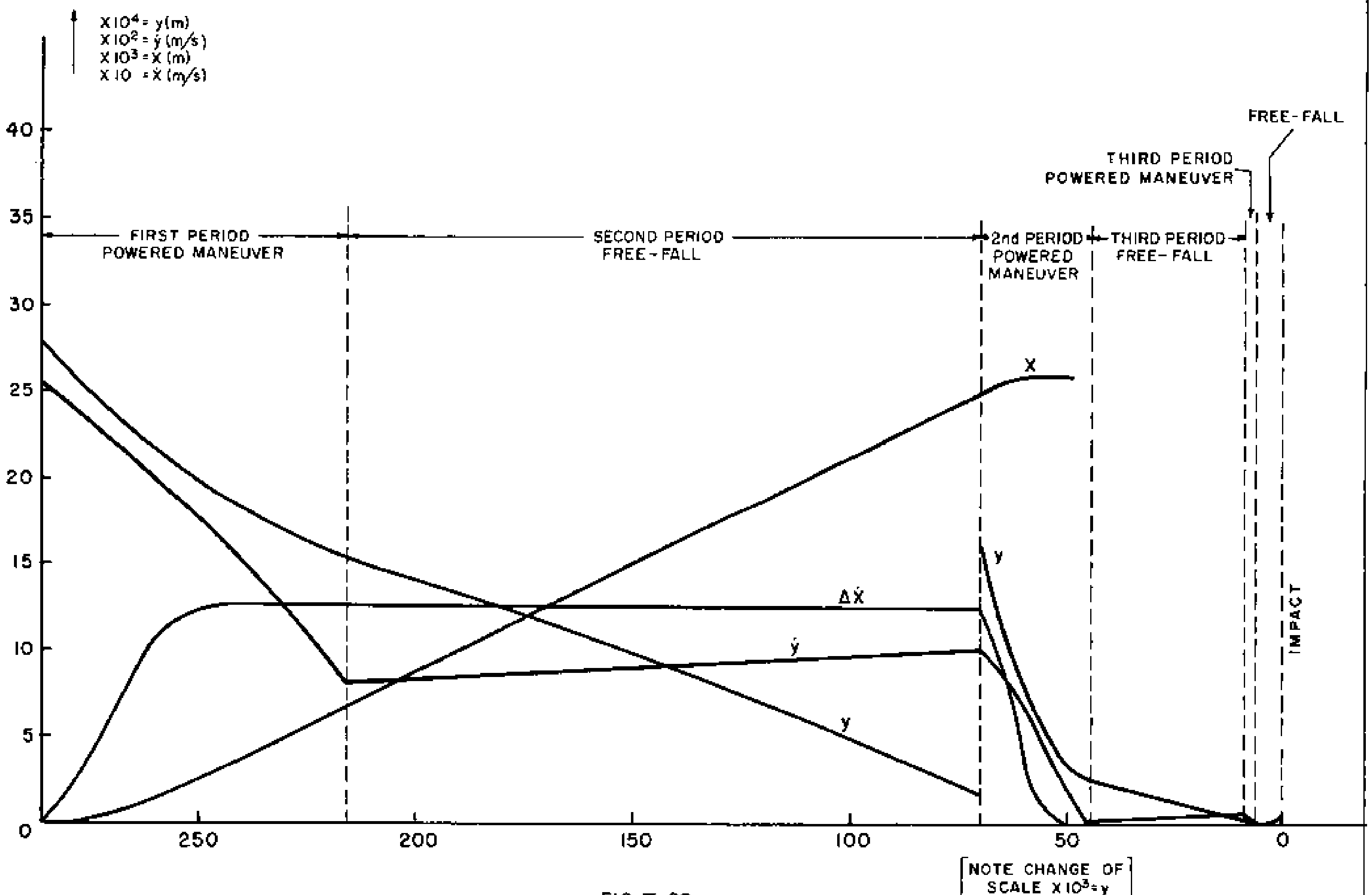


FIG. II. 29
TERMINAL GUIDANCE DESCENT CALCULATED BY ANALOG COMPUTER

Here are some differences in primary functions:

First Period: The primary functions are to reduce drastically the vertical velocity, and to execute the coarse lateral displacement as dictated by the human operator.

Second Period: The primary function is to reduce the lateral velocity to essentially zero.

Third Period: The primary functions are to reduce the velocity vector to near zero and to execute the fine lateral correction as dictated by the human operator.

The equipment for performing the measurements during the three periods differs in several respects. During the first period, the altitude is measured by the horizon seeker. During the second period, a radio altimeter will measure altitude and, pending further investigations, velocity may be measured by a Doppler-type velocity meter, thus relieving the TV from the measuring function, and the guidance computer from calculating vertical velocity from altitude data. This does not mean that the TV measurements and the guidance computer output of vertical velocity would be thrown away; they are kept as redundant data to be drawn on at any moment in case of emergency. The same holds true during the preparatory steps for the data available from injection and radio tracking on earth. The knowledge of the trajectory as computed from these data is a back-up to the measurements taken by the vehicle, and will be used in case of emergency, i.e., if equipment fails.

During the third period of descent, the command for the fine lateral displacement may be given in a different manner. The displacement command, x_o , need not be transformed by the earth-bound computer into a velocity command, $\Delta\dot{x}_o$, and then introduced into the control equation. Rather the command, x_o , may be introduced directly into the controls. This results in the condition of zero lateral velocity at the end of the powered maneuver in contrast to the other type of maneuver which leaves a lateral velocity at the end of the powered period for the desired purpose of building up lateral displacement during the subsequent free fall. The control equation for this period then is

$$\beta = a_o \phi + a_1 \phi + e_o (x - x_o) + e_1 (\Delta \dot{x})$$

This control is not used for the coarse correction during the first period since it would not be able to take advantage of the free fall during the second period to build up the lateral displacement. It would demand a drastically higher fuel consumption for the lateral correction. This extra fuel is negligible during the third period when only small corrections of less than 100 meters are considered.

One other noteworthy point is this: It is planned that high altitude measurements will be made by the horizon seeker. A pulse type radio altimeter would probably be more accurate than the horizon seeker, but power and weight requirements for such an altimeter conflict with the available weight and power. If current investigations of the design of a high altitude pulse type radio altimeter should prove that it can be built within the power and weight restrictions imposed by the vehicle, then it will replace the horizon seeker at about 300 or 400 km. The horizon seeker would remain for setting up the platform with reference to the local vertical, but it would give altitude data only during the early preparatory steps.

II.7 CIRCUMLUNAR ORBIT AND RE-ENTRY (U)

II.7.1 Energy-Level Consideration. A fundamental problem in the design of the orbit is the choice of the energy-level under which the orbiter is to be injected into its circumlunar orbit.

Shorter flight times from the earth to the proximity of the moon can be obtained by higher injection velocity. To achieve a shorter total flight time from injection to return, however, a braking impulse must be applied shortly before reaching the moon. The trade-off to this twice-increased propellant requirement is: (1) the possible reduction in energy requirement for sustaining life of men and machines, and (2) possible propellant savings for correction maneuvers, brought about by reduced sensitivity of the orbital trajectory to some injection errors. The choice of the injection velocity may also be influenced by reliability viewpoints. With a high energy trajectory, inaccurate operation of the braking rockets would render the safe return more difficult than with a low-energy trajectory.

Before the technology of inhabited space capsules has progressed far enough to furnish data and trends, it is safest to plan on trajectories of the near lowest energy level. This is the guideline for the following trajectory description.

II.7.2 Celestial Model and Injection Conditions. The vehicle orbit will be assumed to be coplanar with the moon's travel. This condition will be closely approached for launchings from the Air Force Missile Test Center after 1965. The exact coplanarity assumption also aids greatly in a clear understanding of principal relationships between injection errors and resulting flight mechanical effects.

The earth - moon model used is that by Jacobi, assuming circular orbits of the celestial bodies about their barycenter. Other celestial bodies are assumed to be of negligible influence. The distance between the centers of earth and moon is taken as 385,080 kilometers (239,277 statute miles).

Injection altitude and path angle are chosen somewhat arbitrarily at 200 km and 60 degrees from the local vertical. The final selection of these parameters is very much dependent on the characteristics of the ascending flight. The ascent phase is shaped in its geometry for minimum propellant consumption and other considerations, such as aerodynamic heating and structural load.

The velocity level for injection is then essentially determined for optimum mission accomplishment. Observation time on the reverse side of the moon is extended with increasing injection velocity, but distance of closest approach to the moon is also increased. Compromising, a distance of approximately one moon radius from the surface of the moon was chosen. This results in a total of three hours of observations, defined as the period for which the vehicle's distance from earth is larger than that of the moon's center from earth.

Since there are two parameters of injection at our disposition, viz., velocity and lunar position (expressed in lunar lead angle ahead of the vehicle at injection time, as measured at the earth's center), the condition of favorable return geometry can be imposed. This may be specified by the path angle of re-entry into the earth atmosphere at a given altitude.

II.7.3 Description of the Nominal Orbit. The vehicle enters its orbit with the conditions as follows:

Altitude - 200 km

Path Angle from Local Vertical - 60 degs

Velocity - 10,856.4 m/sec

Lunar Lead Angle - 70 degs

Azimuth - In plane of moon's orbit

The orbit geometry is shown on Figures II.30 and 31 in space-fixed and earth - moon oriented coordinate systems. Inserts on these figures show enlarged sections of the orbit near the moon.

Since the return-leg is chosen such that departure and return legs entwine the globe, the orbit is to follow generally a figure eight shape, with almost symmetrical legs. The closeest approach to the moon occurs above the center of the moon's backside, on the axis earth - moon. This is best shown on the insert on Figure II.31. The geometry of the close-in phases near the earth is depicted in enlarged scale on Figure II.32, using the coordinate reference that rotates with the celestial model.

The point of periselenum is characterized by the following flights data:

Position - At axis earth - moon

Altitude above moon's surface - 1788 km or 1.03 lunar radius

Time after injection - 77.5 hours or 3.23 days

Velocity with respect to rotating system - 1942 m/s

Velocity with respect to space-fixed system - 935 m/s

For the description of the earth approach phase, the altitude of 120 km is taken as reference. The following flight status ensues:

Altitude above surface of earth - 120 km

Velocity (in the rotating system) - 10,924 m/s

Path Angle (from local vertical) - 97 degs

Time from injection - 154.1 hours or 6.42 days

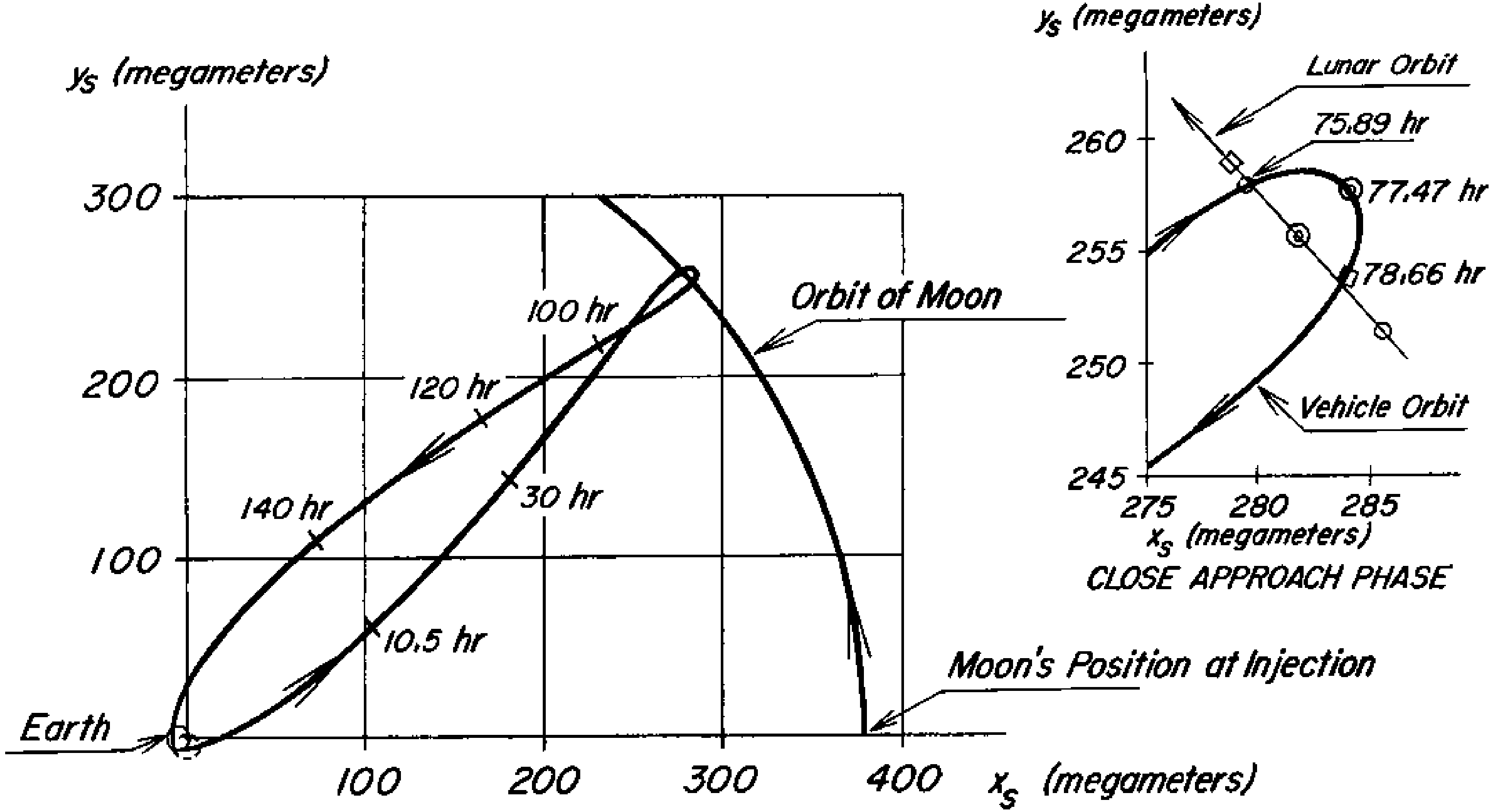


FIG. II.30 FLIGHT GEOMETRY OF CIRCUM-LUNAR ORBIT IN THE SPACE-FIXED COORDINATE SYSTEM

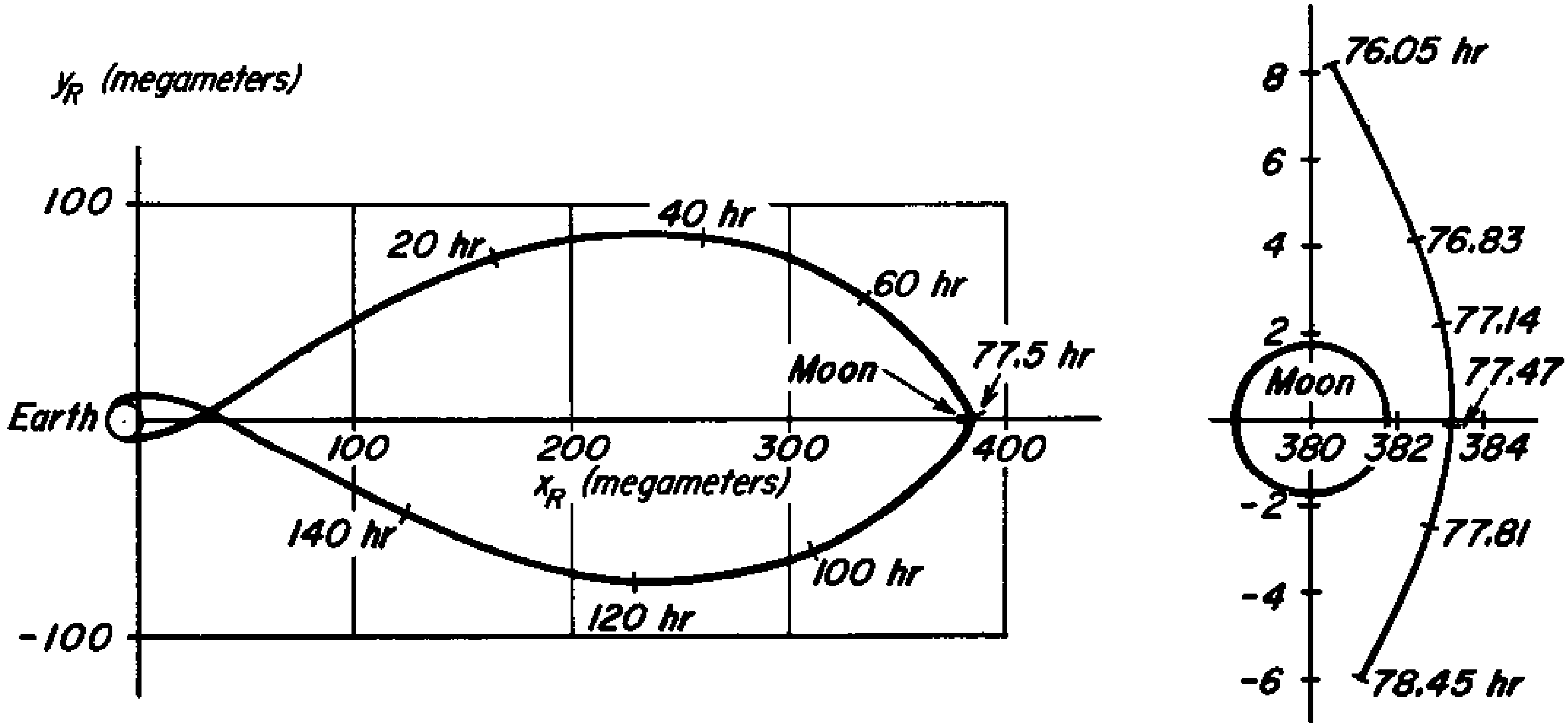


FIG. II.31 GEOMETRY OF A LUNAR CIRCUMNAVIGATING TRAJECTORY IN THE ROTATING COORDINATE SYSTEM

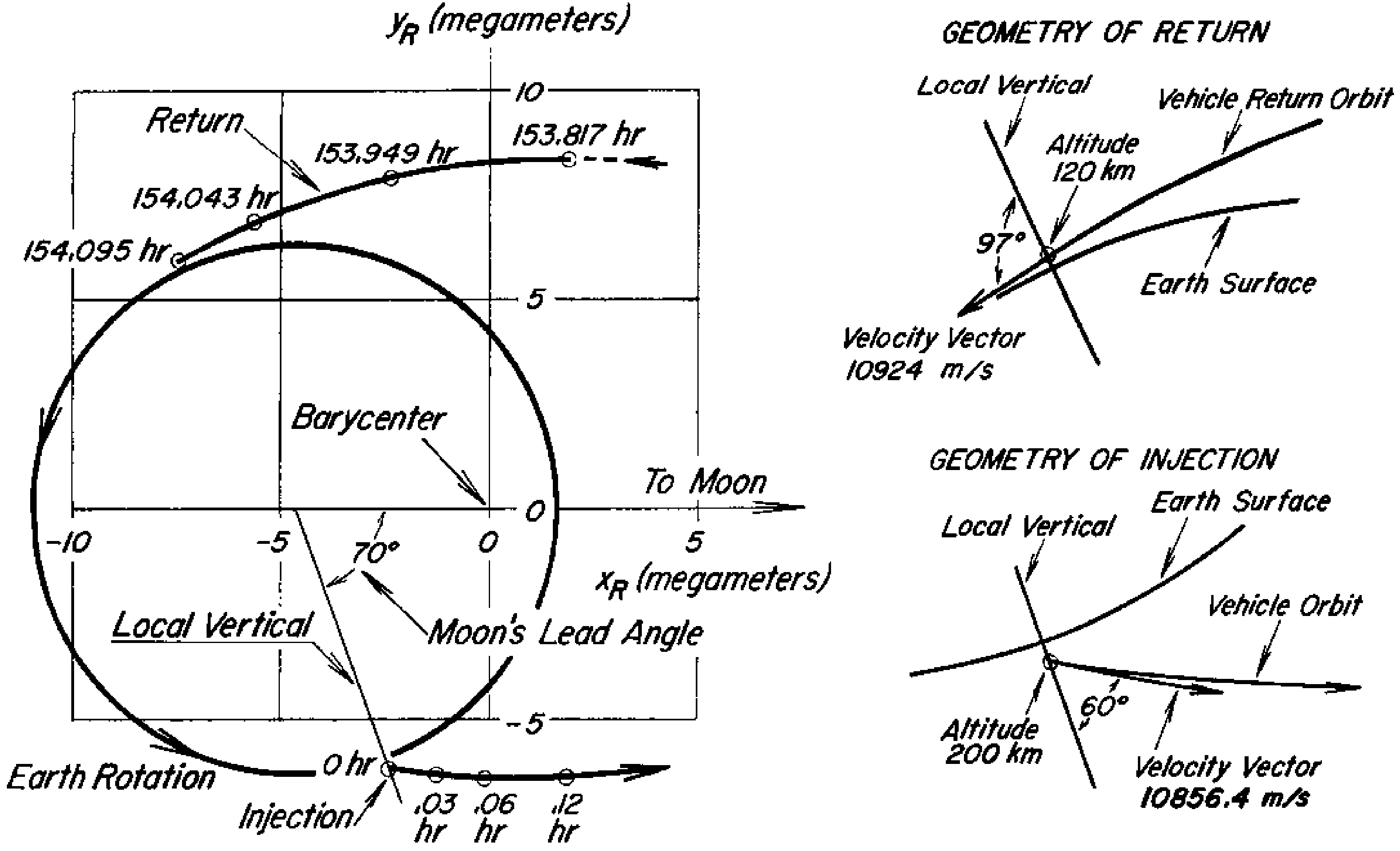


FIG. II.32 DEPARTURE AND RETURN PHASES OF CIRCUM-LUNAR ORBIT IN EARTH-MOON-ORIENTED COORDINATE SYSTEM

The path angle was chosen for optimum single pass re-entry, as will be discussed in one of the following sections.

For information, the velocity is plotted over the whole course (Figure II.33) as a function of orbit time. Reference is taken to both the space-fixed and the rotating coordinate system. The smallest velocity level of 1090 m/s is reached near the 60th hour of flight. At the periselenum point, the curve reaches a local maximum of 1942 m/s (with respect to the rotating system). The return history is very nearly symmetric to the outbound trip history. The re-entry point is passed with the speed of 10,924 m/s, which is 150 m/s under the local excape velocity, referred to a central force field.

II.7.4 <u>Sensitivity of Periselenum to Orbit Injection Conditions and Mid-Course Impulses</u>. Knowledge of the quantitative relationship of trajectory deviations as function of parameter changes (erroneously occurring as well as purposely introduced) is a basic requirement for the design of the guidance system and magnitude choice of the engine for correction impulses.

There are seven parameters that can be changed at any point of the trajectory. They are the three coordinates each for position and velocity, and the time. However, only six of them are independent of each other, as e.g., trajectory changes due to time variations can also be expressed as being produced by a combination of position and velocity changes.

Of these six independent parameters, the most important ones are analyzed and discussed in the following paragraphs. Since the guidance system that controls the flight up to injection will be of the continuously correcting type, in general no large deviations will occur from the nominal trajectory, thus, the first partial derivatives may be sufficient to demonstrate the effects of deviations.

As to the time periods, where corrections may be expected, it is anticipated that these may occur at any time over the whole course; on the first leg primarily with the purpose of acquiring favorable lunar approach conditions, on the return leg exclusively to assure safe recovery of the capsule. Since return conditions are strongly influenced also by the maneuvers on the outbound course, the outbound maneuvers are to be selected with regard to both their effects on the periselenum and on the return conditions.

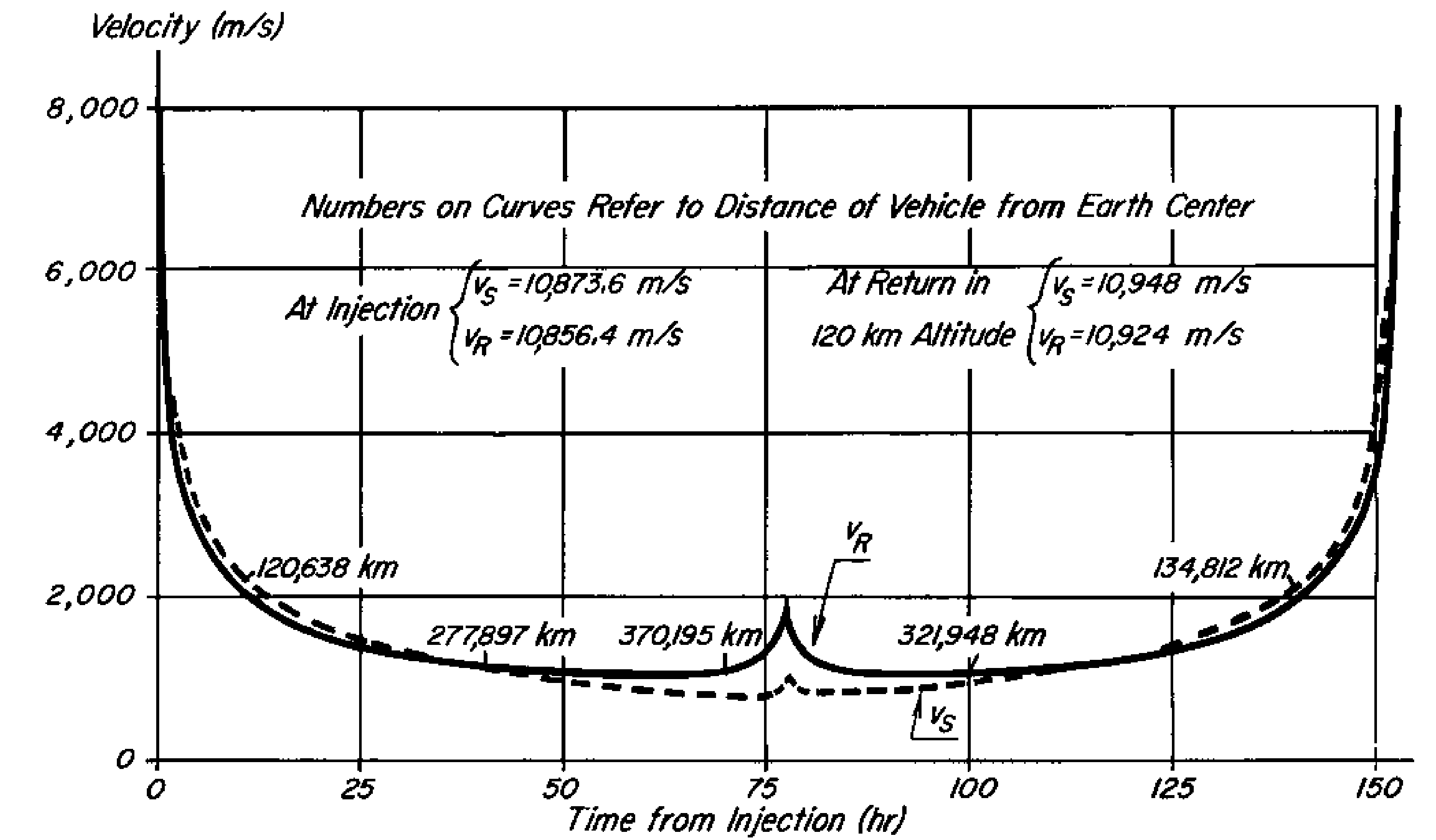


FIG. II.33 VELOCITY HISTORY OF CIRCUM-LUNAR ORBIT VERSUS FLIGHT TIME

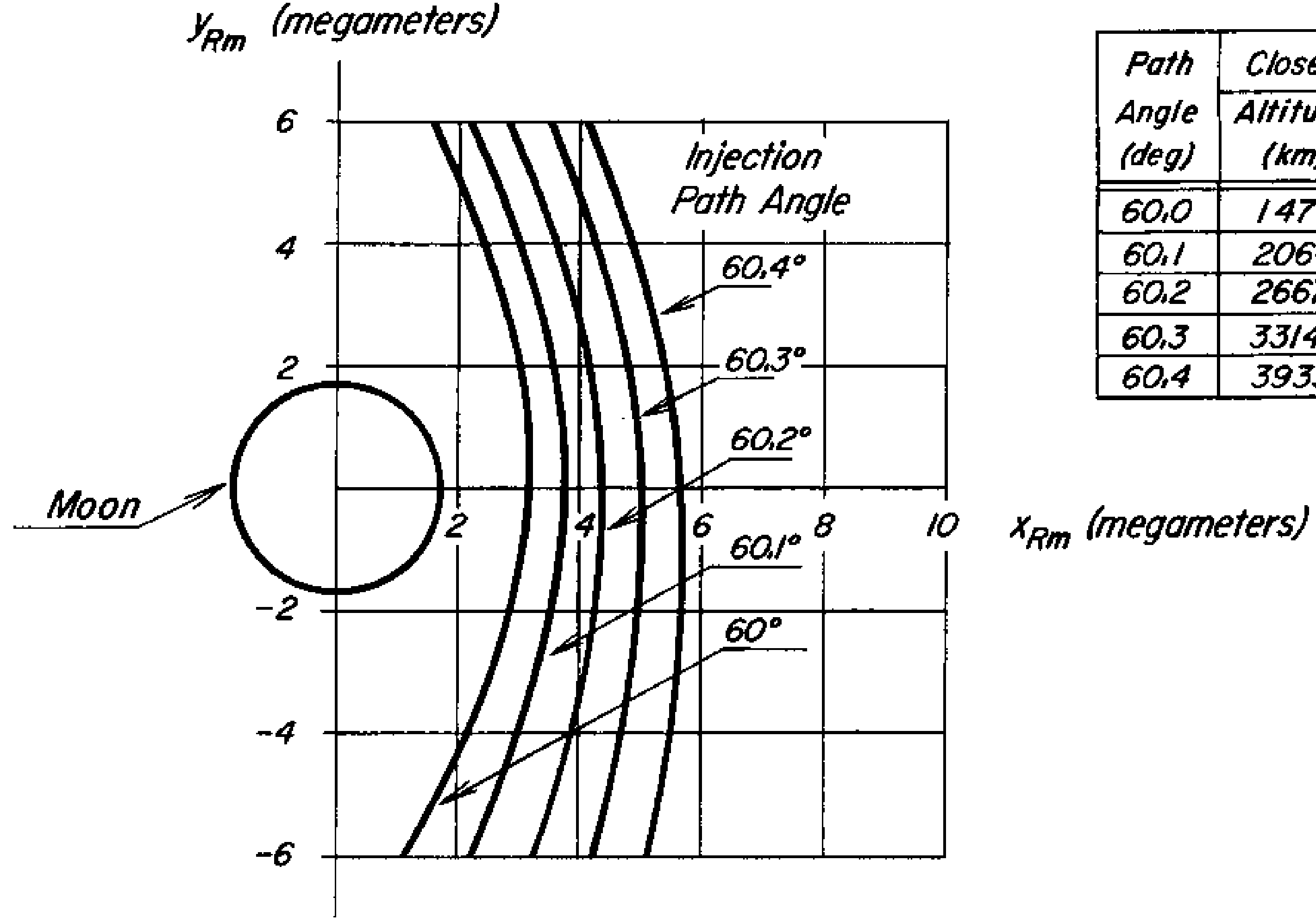
v_S = Velocity with Respect to Space-fixed System v_R = Velocity with Respect to Rotating System

The analysis covers the effect of deviations from standard values in (1) velocity magnitude, (2) velocity direction in the flight plane (pitch), and (3) velocity direction out of the flight plane (yaw or azimuth). For the injection point, information is also provided for effects from altitude deviations, lunar lead angle deviations and latitude deviations.

Before the partial derivatives are given, the character of the effects of deviations may be discussed by referring to two figures that show the behavior of the orbits near the moon. Figure II.34 depicts a family of orbits that originate with path angle deviations (pitch) applied at the injection point. All other parameters of the injection point are kept at standard values. While increasing the path angle from 60 to 60.4 from the vertical, the close approach distance to the moon increases by about 1.5 lunar radius, and the time to reach the periselenum is increased by about 1.5 hours. The flight branches around the moon are still fairly symmetrical with respect to the axis earth - moon for this magnitude of variation.

The second figure (Fig. II.35) shows the corresponding circumlunar flights for variations of the following injection parameters: Path angle in plane (pitch) by 0.1 degrees; azimuth (cross plane) by 0.1 degrees; velocity by 1.0 m/s; lunar lead angle by -0.1 degrees (corresponding to a launch time delay by 0.4 minutes); altitude variation by -1.0 km; and a latitude variation by 0.0872 degrees (corresponding to 10 km lateral displacement, maintaining standard azimuth and standard altitude). For the cases of azimuth and latitude variations, the orbit does not lie in the principal plane. Hence, the figure shows here only the projection of the lunar flight phase into the principal plane. The shift out of this plane, however, is small. The magnitude is indicated by the insert on this figure, showing the pierce point of all trajectories through the plane perpendicular to the principal one and going through the earth - moon line (x - axis).

In listing the partial derivatives in Table II.2, reference is taken to this perpendicular (x_R, z_R) plane. Distance variations from the standard trajectory and time variations are taken for the moments when the vehicle passes through this (x_R, z_R) plane.



Path Angle (deg)	Close Approach Altitude (km)	Close Approach Time (hr)
60.0	1471	77.500
60.1	2064	77.821
60.2	2667	78.226
60.3	3314	78.715
60.4	3933	79.072

FIG. II.34 EFFECT OF IN-PLANE PATH ANGLE VARIATION AT INJECTION ON GEOMETRY OF LUNAR APPROACH PHASE

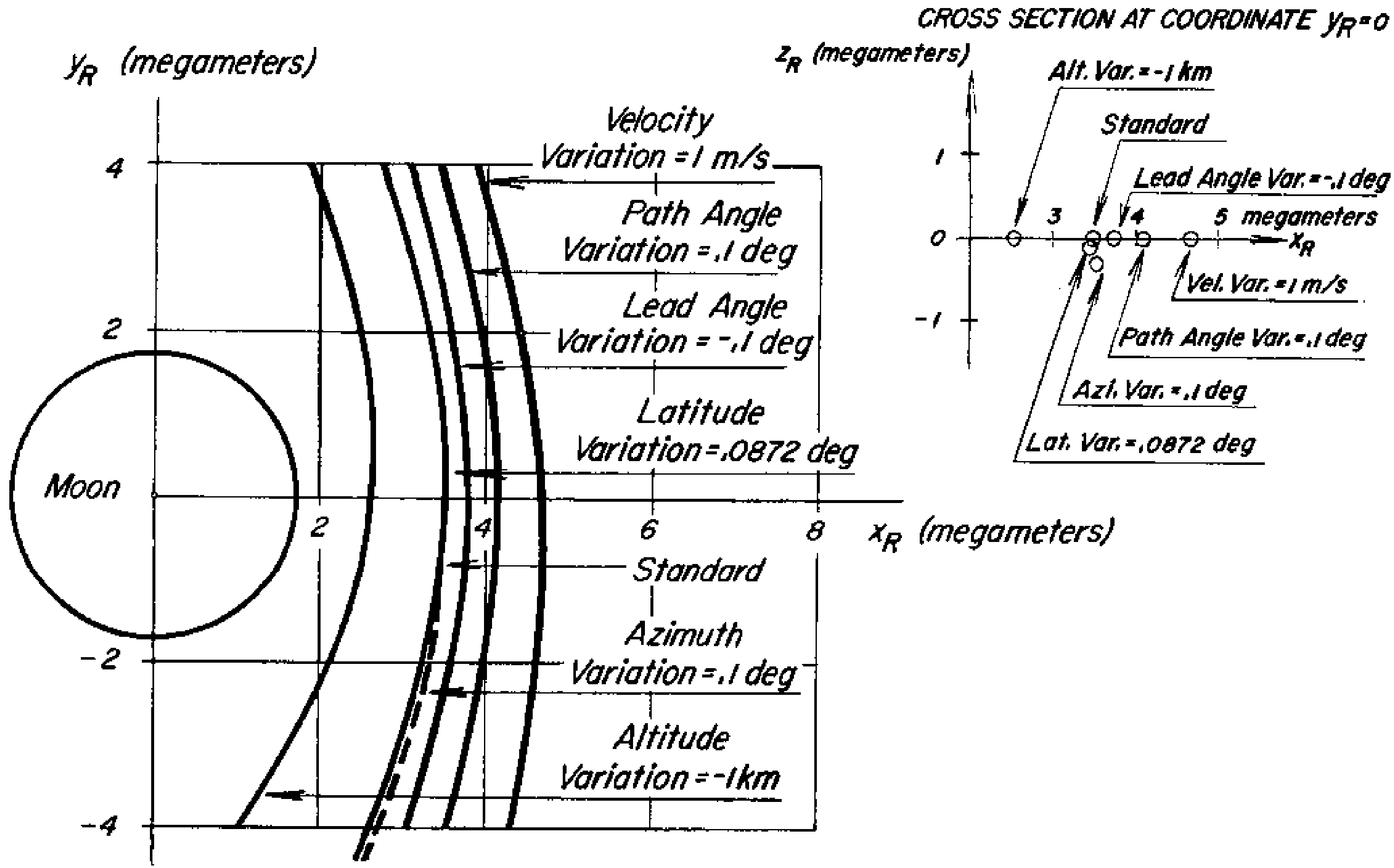


FIG. II.35 EFFECT OF VARIATIONS OF INJECTION PARAMETERS ON GEOMETRY OF THE LUNAR APPROACH PHASE

TABLE II.2

EFFECTS OF INJECTION ERRORS
ON DISTANCE AND TIME OF PERISELENUM

Variations at Injection from Standard	Variations from Standard at Periselenum	
	distance (km)	time (hrs)
Velocity (per m/s)	1,081.	.16
Path Angle (pitch) (per degree)	6,054.	3.77
Azimuth Angle (yaw) (per degree)	2,941.	.18
Altitude (per km)	924.	.27
Latitude (per km)	10.7	.00
Lead Angle (per degree)	2,993.	.60
or		
Injection Time (per minute)	748.	.15
or		
Ground Range (per km)	27.	.01

While the originating variations may be understood as errors, the following changes may be thought of as correction maneuvers. These are changes introduced at a later time of flight on the velocity vector for the purpose of achieving the flight mission. The changes here were applied at different times on the standard orbit. It is understood that in reality changes will be applied only if deviations are measured (e.g., by a tracking system from ground) from standard positions. This fact means that the status of flight, as it was assumed at any point of application, is not strictly true, but it is justified to assume that the erroneous flight status is still close enough to the standard so that the influential factors listed are applicable.

The effect of velocity-magnitude, path-angle (pitch) and azimuth-angle changes on the periselenum placement, for application of these changes at any time of the orbit, is graphically shown on the following two figures (II.36 and 37). One refers to time of flight as abscissa, the other to distance from earth center. The following two figures (II.38 and 39) depict the time deviations of the periselenum, again for the two referenced abscissas.

It is worthwhile to mention that some curves do not exhibit a monotonously decreasing behavior, but show proper maximum and minimum. This fact will be of importance for selection of favorable points of applying corrections.

<u>II.7.5 Sensitivity of Return Flight to Orbit Injection Parameters and Mid-Course Impulses</u>. The positioning and shape of the return leg of the orbit resulting from flight parameter changes at injection is first discussed on the basis of a typical example. Figure II.40 demonstrates the incoming phases of orbits which experienced variations of the (in plane) path angle at the injection point. Five trajectories are shown, corresponding to angular variations from 60 degrees to 60.4 degrees. The impact points on the earth, which are calculated here with the assumptions of no atmosphere, are representative for the later applied reference of atmospheric re-entry at 120 km altitude. Attention is drawn to the increased steepness of re-entry with increased injection path angle, and also to the great sensitivity of the impact point locations with angle changes. This is especially true if the ideal re-entry direction (near the tangential approach) is reached. Large time changes (more than 10 hours) result from variations of the injection angle, as illustrated by the curves.

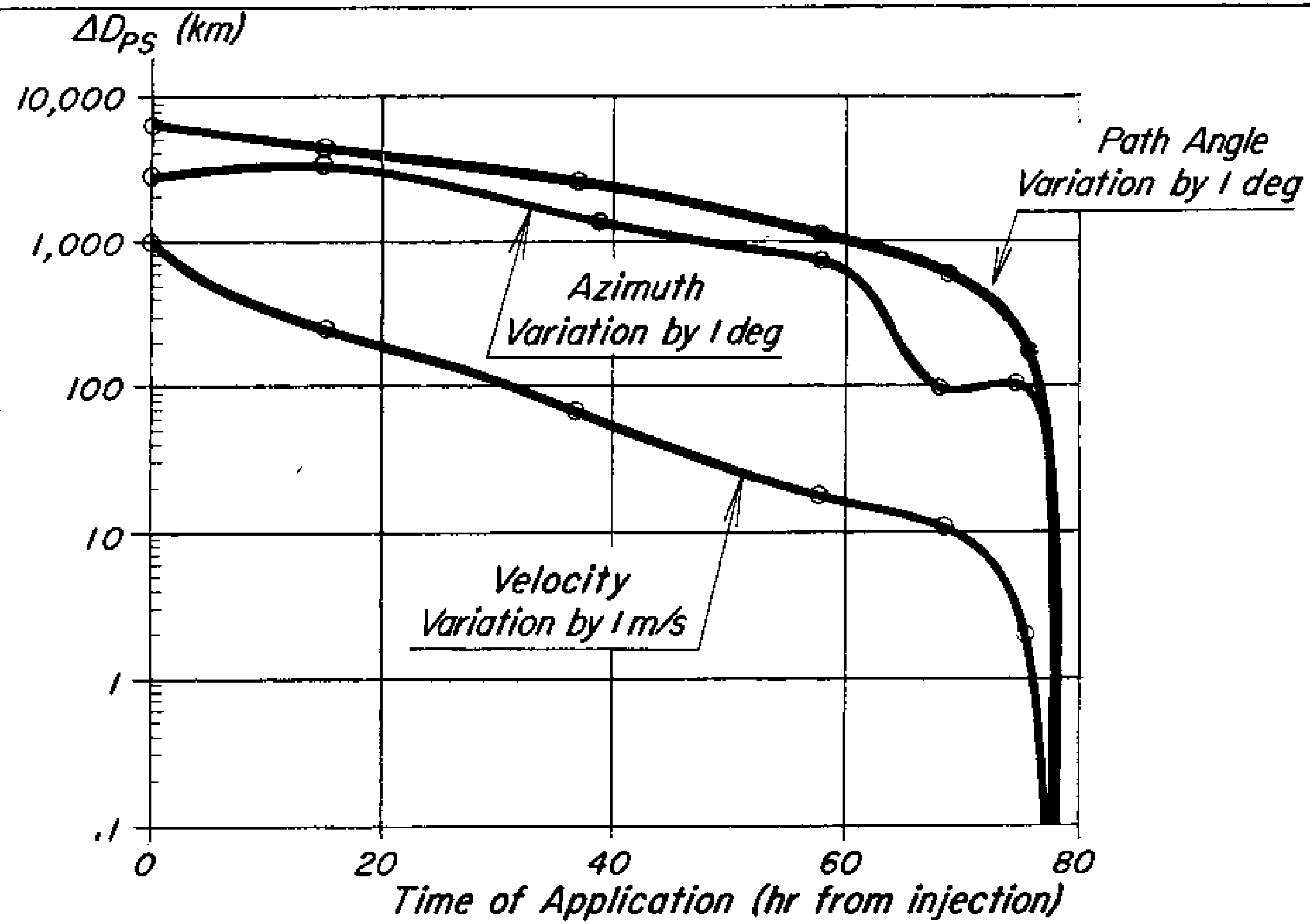


FIG. II.36 PERISELENUM DISTANCE VARIATION (ΔD_{PS}) AS EFFECTED BY VARIATIONS OF VELOCITY, PATH ANGLE AND AZIMUTH PLOTTED OVER TIME OF APPLICATION

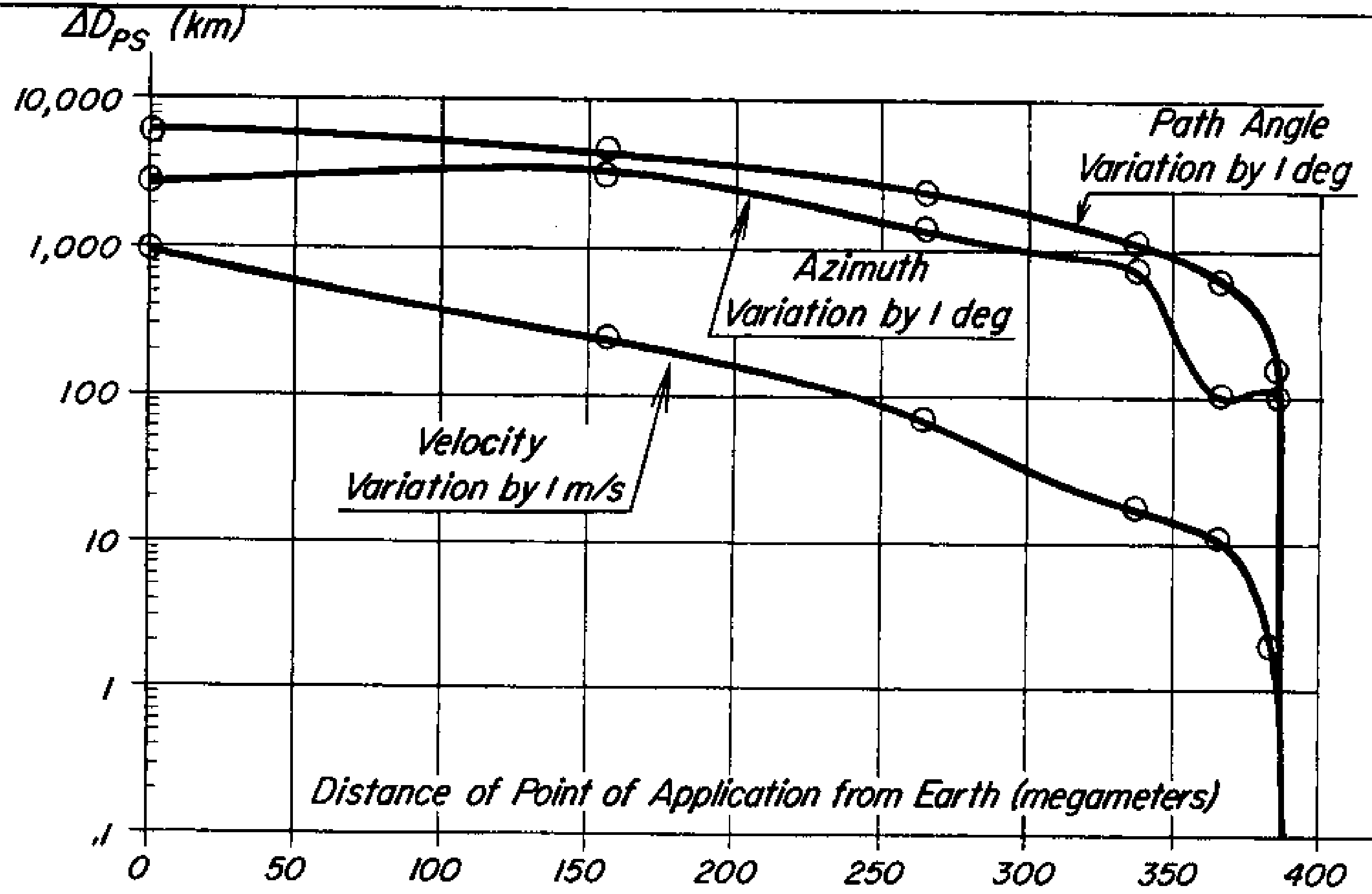


FIG. II.37 PERISELENUM DISTANCE VARIATION (ΔD_{PS}) AS EFFECTED BY VARIATIONS OF VELOCITY, PATH ANGLE AND AZIMUTH PLOTTED OVER DISTANCE FROM EARTH AT WHICH VARIATION IS APPLIED

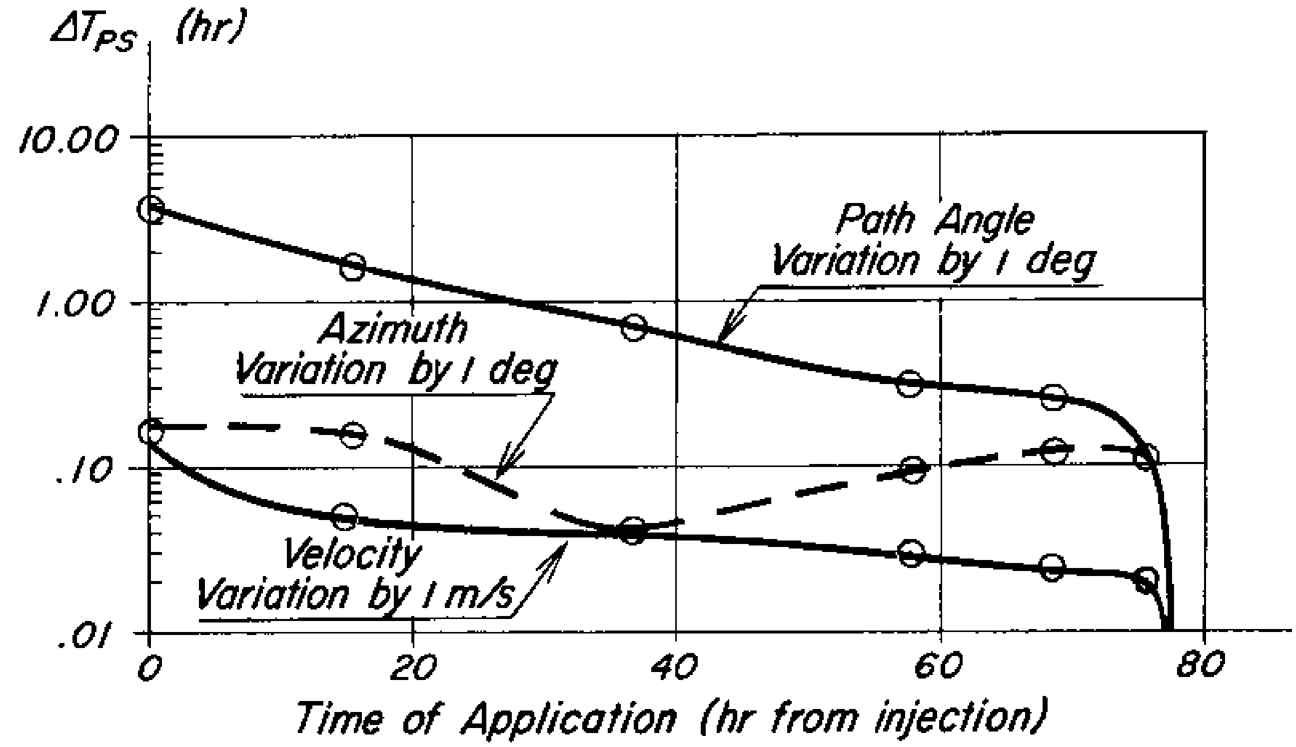


FIG. II.38 TIME VARIATION (ΔT_{PS}) IN REACHING PERISELENUM AS EFFECTED BY VARIATIONS OF VELOCITY, PATH ANGLE AND AZIMUTH PLOTTED OVER TIME OF APPLICATION

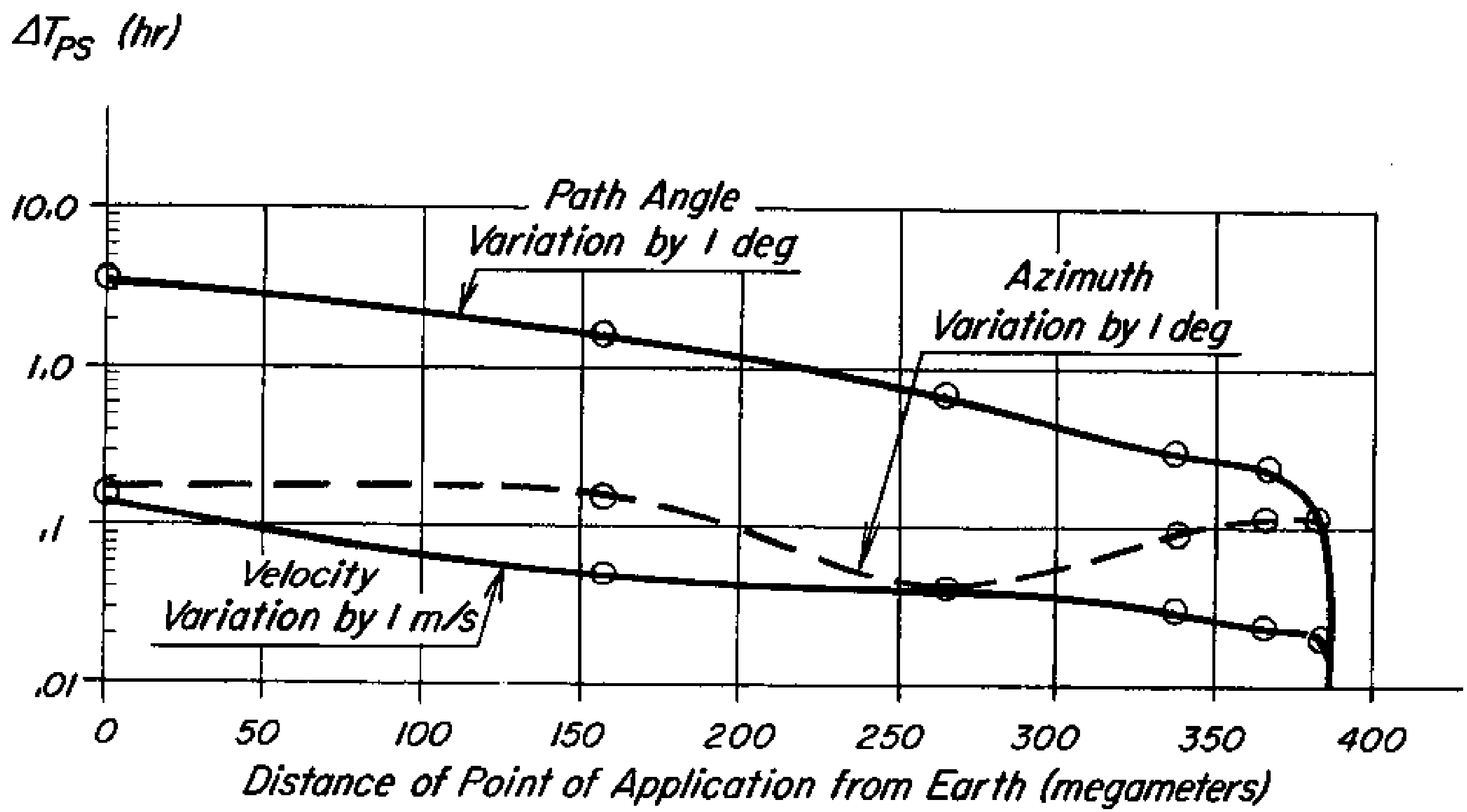


FIG. II.39 TIME VARIATION (ΔT_{PS}) IN REACHING PERISELENUM AS EFFECTED BY VARIATIONS OF VELOCITY, PATH ANGLE AND AZIMUTH PLOTTED OVER DISTANCE FROM EARTH AT WHICH VARIATION IS APPLIED

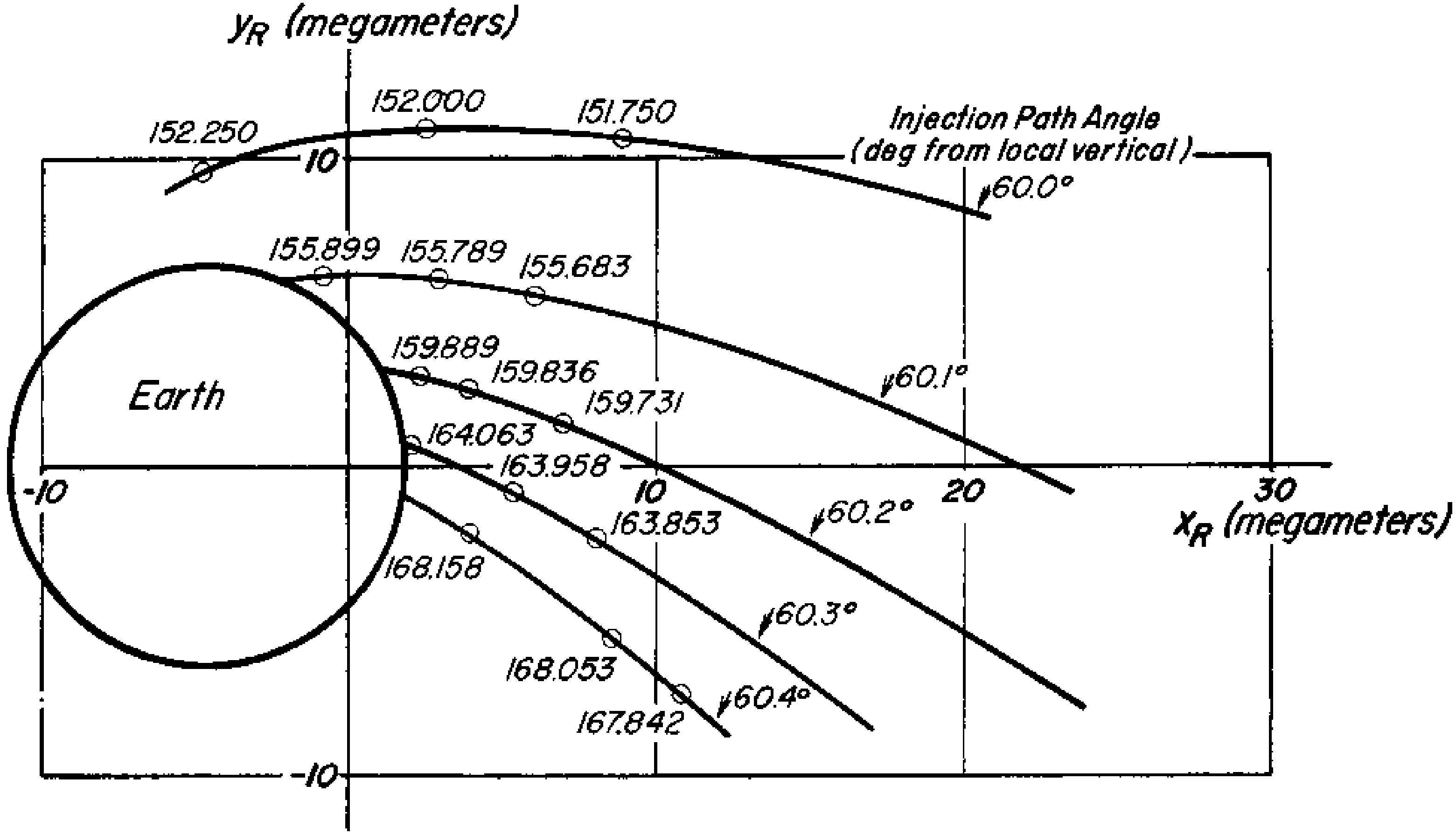


FIG. II.40 GEOMETRY OF RETURN PHASES AS EFFECTED BY VARIATIONS OF INJECTION PATH ANGLE

Figures on Curves Show Time from Injection in hr

The insert figure (Fig. II.41) depicts return orbits for the same variety of injection errors that were discussed above for the effects on the periselenum. The injection parameter deviations are taken with reference to the ideal trajectory injection conditions, which yield a re-entry angle of 97 degrees from the local vertical at the altitude of 120 km. The extreme sensitivity of re-entry with injection performance is quite obvious.

It is interesting to see that the errors connected with out-of-plane motion, as azimuth and latitude, have a relatively small influence on the in-plane behaviour. Their effect on the lateral displacement of the return leg, however, is large, as can be seen on the insert graph.

The insert graph depicts again a plane perpendicular to the principal orbit-plane, but this plane is, in contrast to that used for the periselenum, perpendicular to the line earth-moon. The location of this plane (which for short reference is called the R-plane for "return") is marked on the figure; it is tangential to the earth.

This plane is also referred to in the following table (Table II.3) which lists the effects of injection errors on the in-bound leg of the orbit.

Figures II.42 through II.45 are prepared to give help in analyzing the problem of selecting best time points for applying correction impulses. They give a survey of effects of unit-magnitude correction maneuvers on the displacement and on the shift of the trajectory pierce point in the R-plane, for application of this correction at any time between injection and passing the R-plane. For convenience, the presentation is again made for two abscissae, the time of travel and the distance from the earth center.

It is apparent that the factor of influence for correcting the return geometry is larger on the out-bound leg by two orders of magnitudes than on the in-bound leg. With regard to time corrections, there is very little that can be done on the in-bound leg.

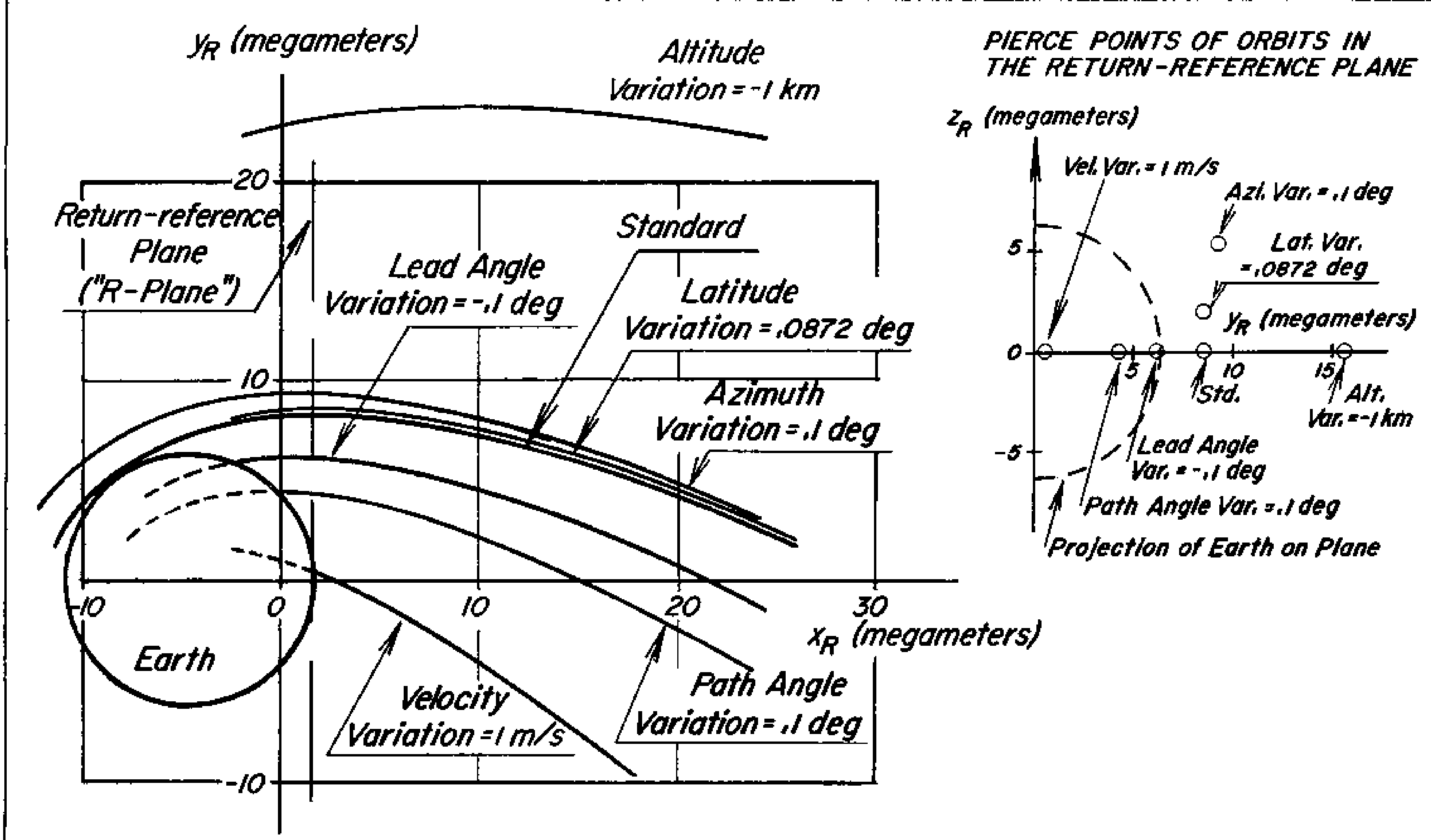


FIG. II.41 GEOMETRY OF RETURN PHASES AS EFFECTED BY VARIATIONS OF INJECTION PARAMETERS

TABLE II.3

EFFECT OF INJECTION ERRORS
ON DISTANCE AND TIME OF RETURN LEG

(Return-Coordinates Measured
in the Return-Reference Plane)

Variation at Injection from Standard	Distance Variation from Standard (km)	Time Variation from Standard (hour)
Velocity (per m/s)	18,017	5.44
Path Angle-Pitch (per degree)	29,397	40.5
Azimuth Angle (yaw) (per degree)	102,790	32.07
Altitude (per km)	14,528	4.66
Latitude (per km)	206.4	.01
Lead Angle (per deg)	24,360	19.60
or		
Injection Time (per minute)	6,090	4.90
or		
Ground Range (per km)	219.5	.18

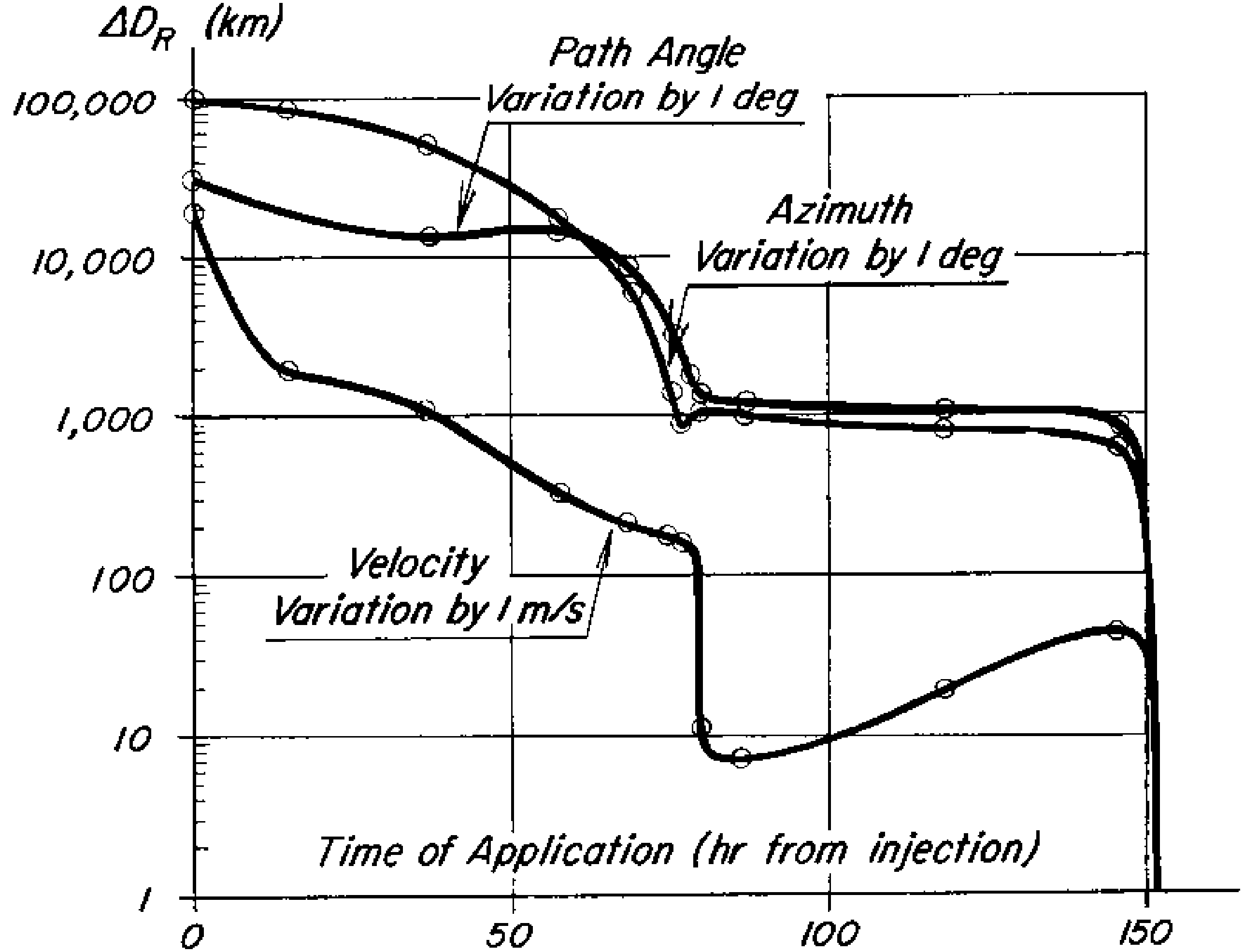


FIG. II.42 DISTANCE VARIATION (ΔD_R) OF RETURN PHASE FROM STANDARD RETURN PHASE, MEASURED IN RETURN-REFERENCE PLANE, AS EFFECTED BY VARIATIONS OF VELOCITY, PATH ANGLE AND AZIMUTH PLOTTED OVER TIME OF APPLICATION

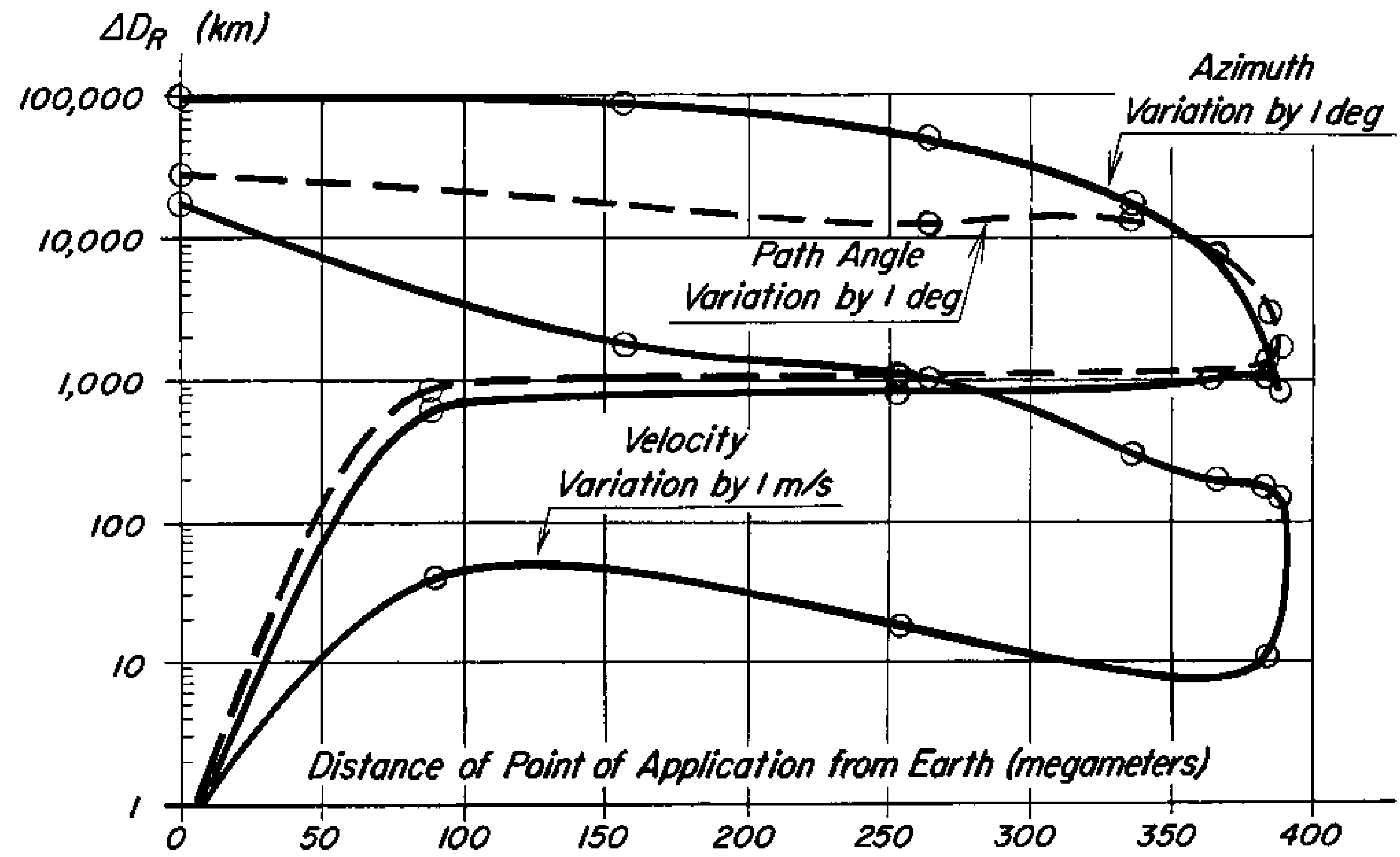


FIG. II.43 DISTANCE VARIATION (ΔD_R) OF RETURN PHASE FROM STANDARD RETURN PHASE, MEASURED IN RETURN-REFERENCE PLANE AS EFFECTED BY VARIATIONS OF VELOCITY, PATH ANGLE AND AZIMUTH PLOTTED OVER DISTANCE FROM EARTH AT WHICH VARIATION IS APPLIED

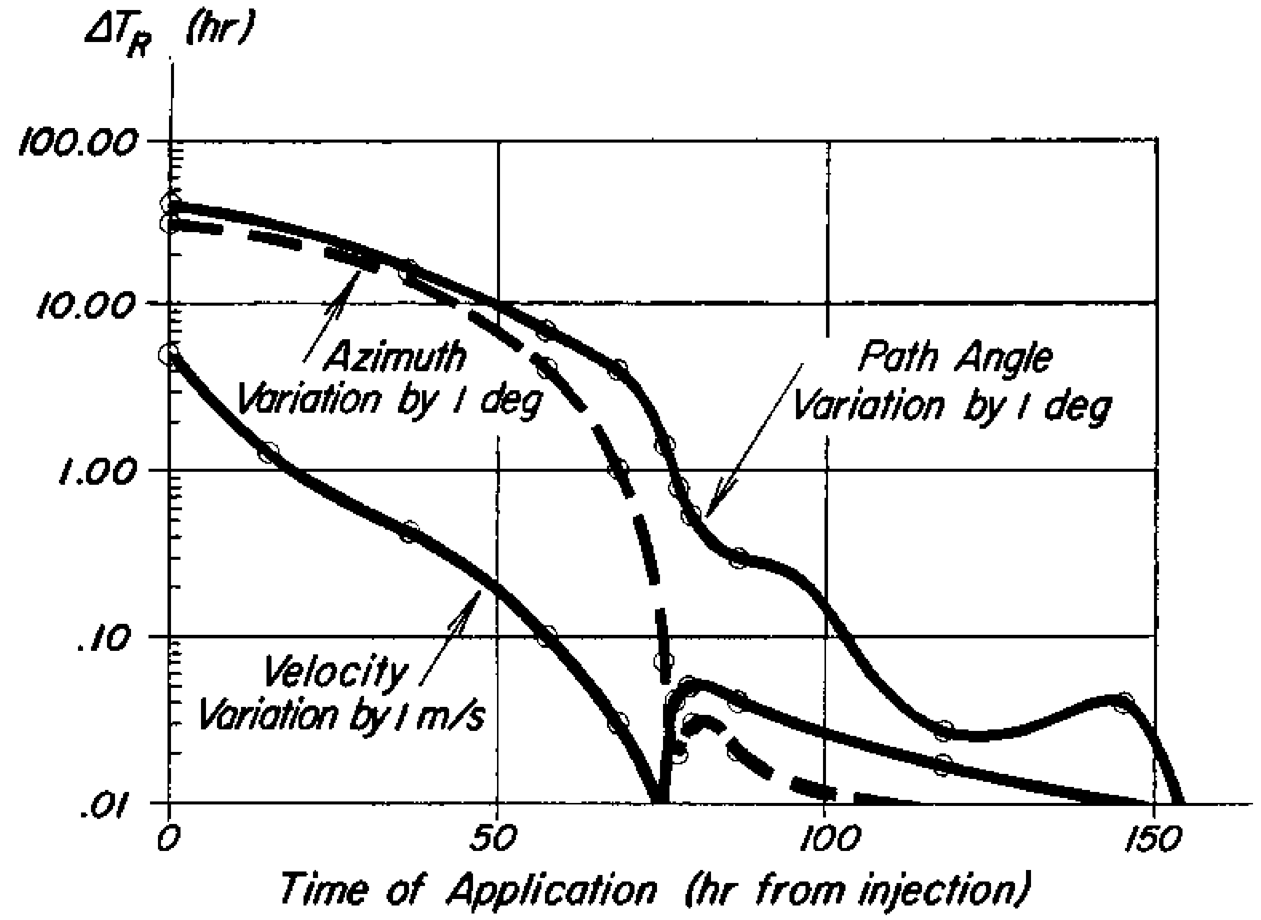


FIG. II.44 TIME VARIATIONS IN REACHING THE RETURN-REFERENCE PLANE AS EFFECTED BY VARIATIONS OF VELOCITY, PATH ANGLE AND AZIMUTH PLOTTED OVER TIME OF APPLICATION

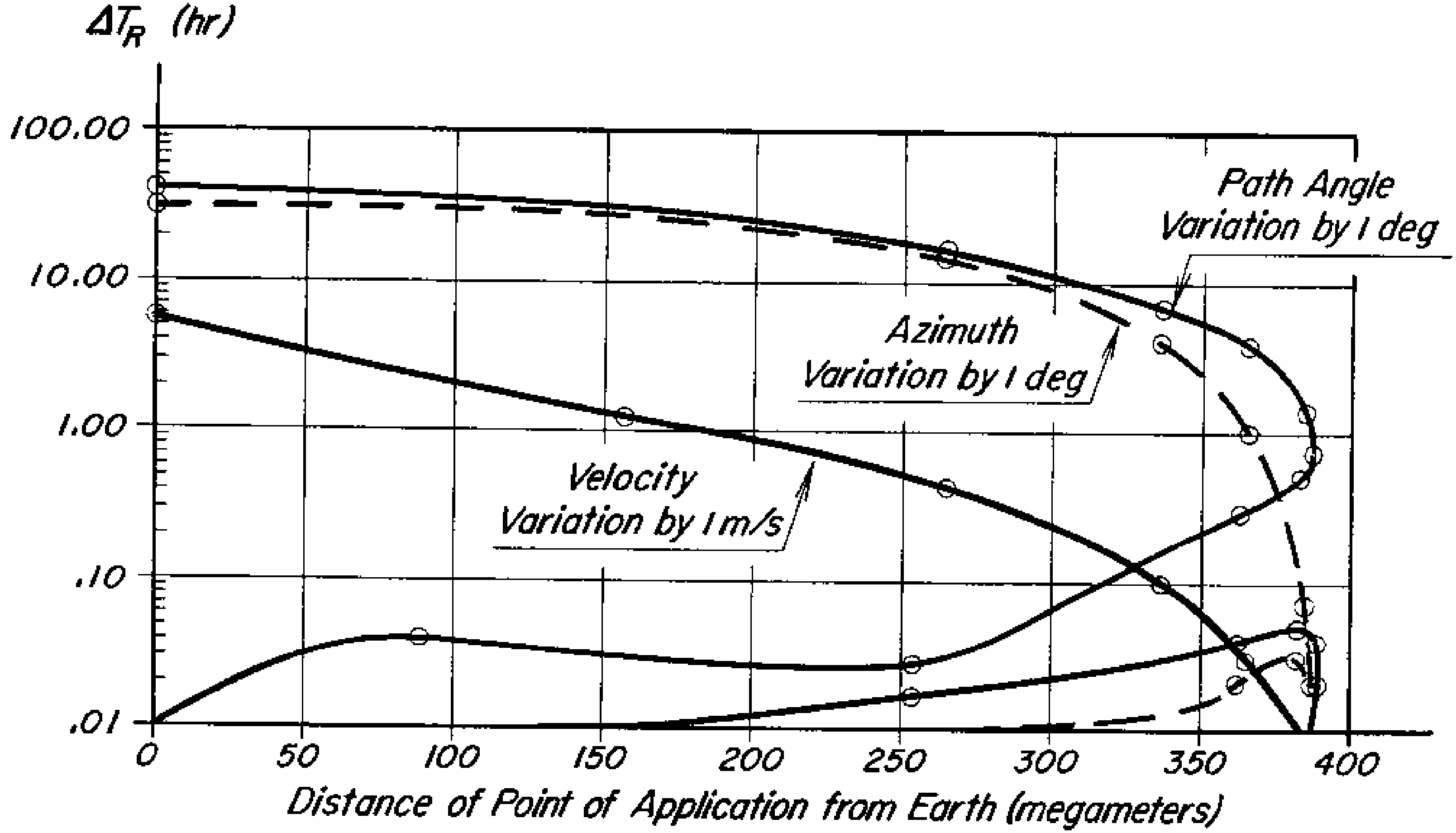


FIG. II.45 TIME VARIATIONS IN REACHING THE RETURN-REFERENCE PLANE AS EFFECTED BY VARIATIONS OF VELOCITY, PATH ANGLE AND AZIMUTH PLOTTED OVER DISTANCE FROM EARTH AT WHICH VARIATION IS APPLIED

II.7.6 Flight Mechanics of the Re-entry Phase. Investigations of the re-entry phase were made to find the degree of maneuverability and range of correction possibilities. The following restrictions and conditions were imposed:

(1) The re-entry velocity at 120 km altitude is 11,000 m/s.

(2) The re-entry trajectory is a single pass through the atmosphere. This means, after first re-entry, the body is not allowed to go higher than about 100 km.

(3) The maximum lift to drag ratio is somewhat less than unity.

(4) The total decelerations shall not be larger than 10 times gravity accelerations.

(5) The body ballistic factor is 500 lb/sq ft., as e.g., represented by the JUPITER re-entry body.

Observing these restrictions, it was found that feasible solutions to the re-entry problem would exist for all re-entry angles between 95.4 and 98.4 degrees (referred to the local vertical). More extended studies may result in a widening of these limits. This range of 3 degrees of path angle at 120 km altitude corresponds to a re-entry "corridor" of 109 km length in the earth-tangential plane, which was introduced in the former section as the return-reference plane.

To allow simultaneously for range and time corrections, the problem of maneuverability was checked on the two limiting re-entry angle trajectories. By making use of the maximum permissible lift (upward and downward) the total re-entry path could be varied to change the ground coverage by 10,000 km or a quarter of the earth circumference. This range variation could compensate for time variations of $\pm$ 3 hours of arrival.

Full use of this range correction possibility cannot be made, since the heat protection provisions are to be optimized for certain precalculated heat input histories. The trajectories for the shortest and longest maneuvered ranges differ in their heating histories to such a degree that probably different heat absorption schemes will be required for each.

The four trajectories, discussed previously, are presented on the five following figures. Figure II.46 illustrates their paths on the curved surface of the earth, and Figures II.47 through II.50 give for each trajectory the variation of the more important flight mechanical parameters.

II.8 LAUNCHING TIME CONSIDERATIONS

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

(1) From considerations of orbital mechanics, 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 once during each anomalistic month of approximately 27.5 days, which varies slowly, and secularly, in time. The declination minima themselves vary periodically and secularly. During one year, the periodic variation amounts to $\pm$ .50°. The average rate of change of declination near the instant of minimum declination is about 0.5° per day, both before and after the instant of minimum.

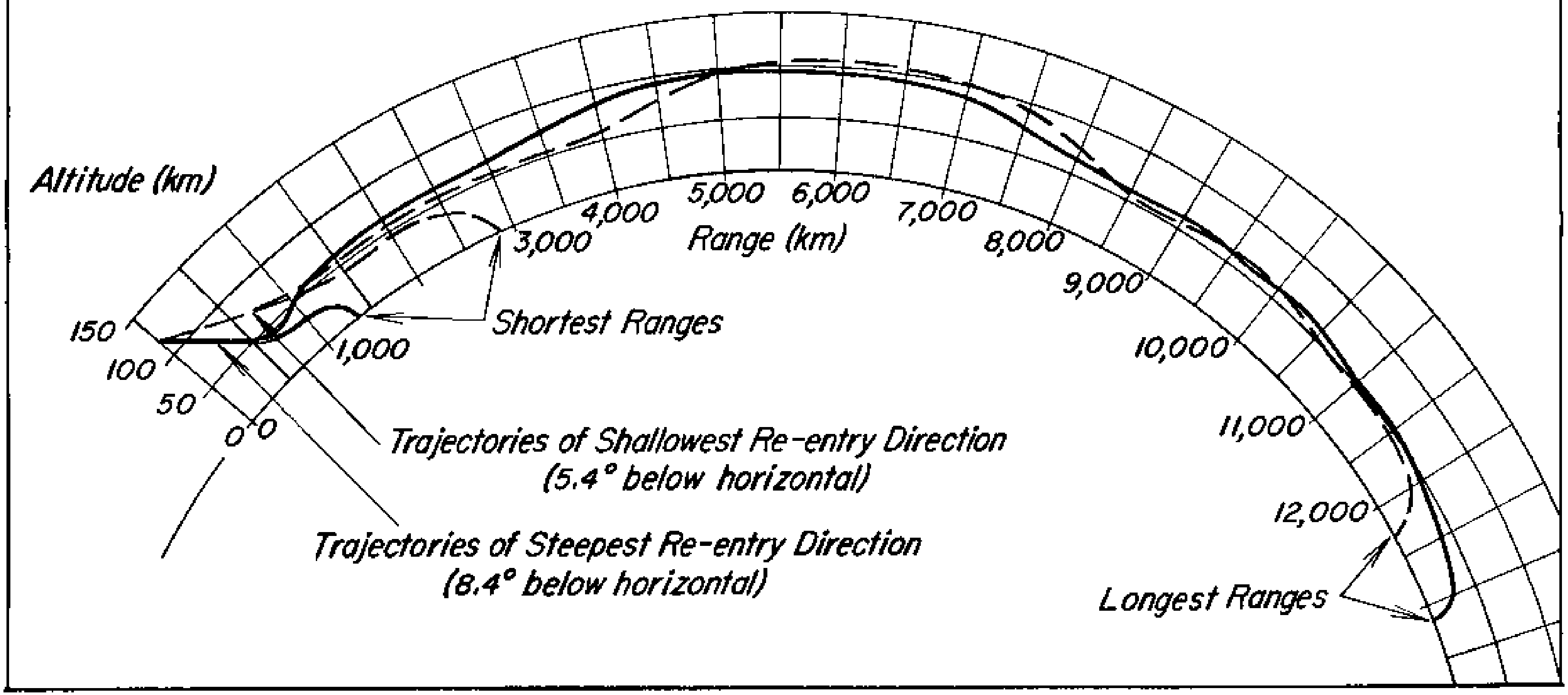
(2) The landing should take place on the terminator, at sunrise for the landing spot. This provides the longest working time using the solar power supply, which is half a moon day, or 14 earth days. The terminator passes the given landing spot once every synodic month, i.e., every 29.5 days, and moves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

If both requirements (1) and (2) should be exactly satisfied, favorable instants for launching occur every 4 or 5 years. Since this interval is too long to be satisfactory, one must be content in satisfying either one or the other condition 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, a preliminary consideration has shown that favorable instants with respect to requirement (1) would occur about once a year. If two days of delay are allowed, the two adjacent minima are also favorable. If at

FIG. II.46 GEOMETRY OF FOUR RE-ENTRY TRAJECTORIES, WITH LONGEST AND SHORTEST GROUND COVERAGE FOR TWO LIMITING RE-ENTRY DIRECTIONS AND DECELERATION RESTRICTED TO 10 G's

(Altitude Scale Exaggerated by Factor 10)

$W/C_D A \approx 500\ lb/ft^2$ $\quad$ $L/D < 1$



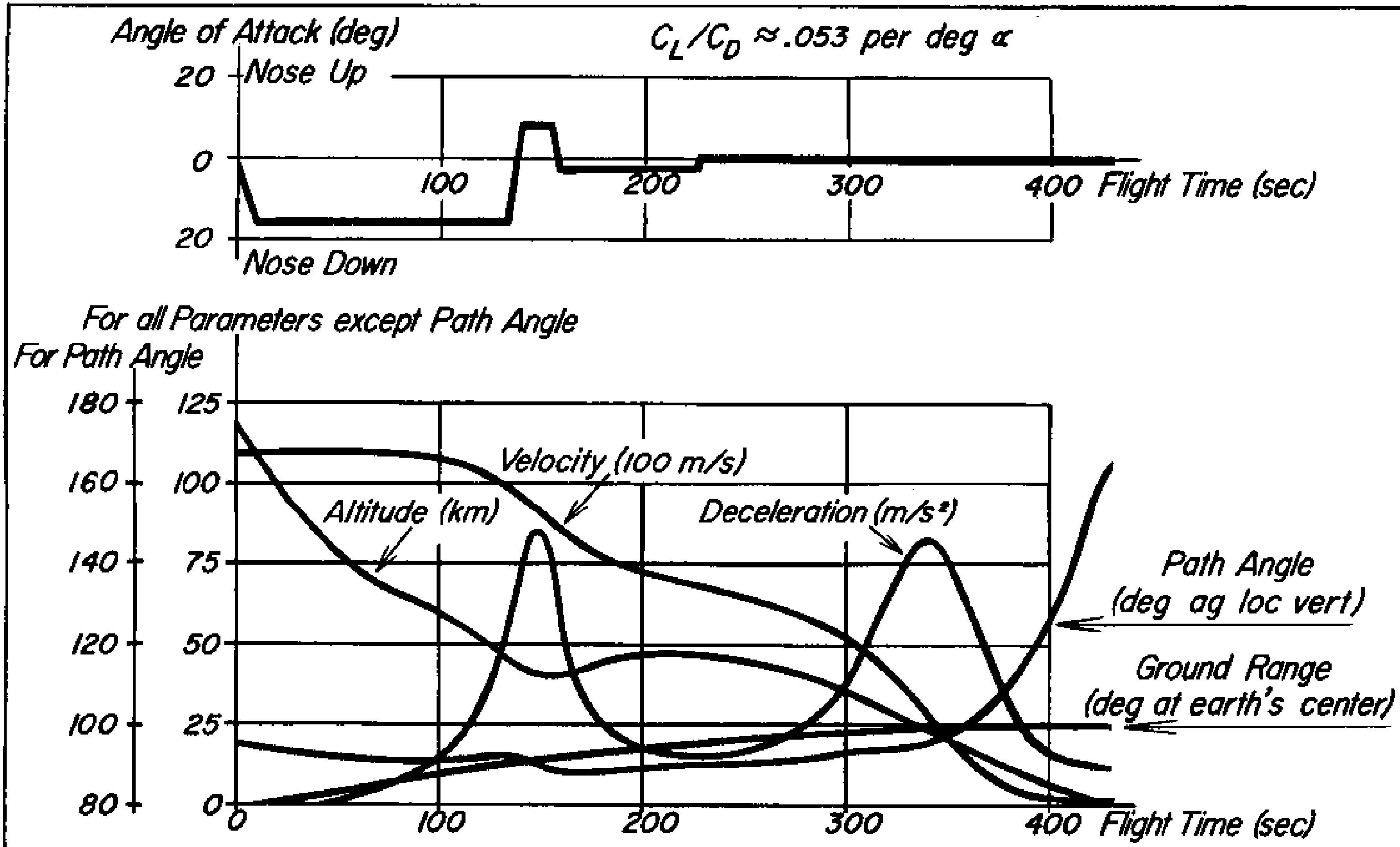


FIG. II.47 SHORT RANGE TRAJECTORY FROM FLAT RE-ENTRY PATH ANGLE
HISTORY OF FLIGHT MECHANICAL PARAMETERS
Re-entry Angle of 95.4 deg from Local Vertical

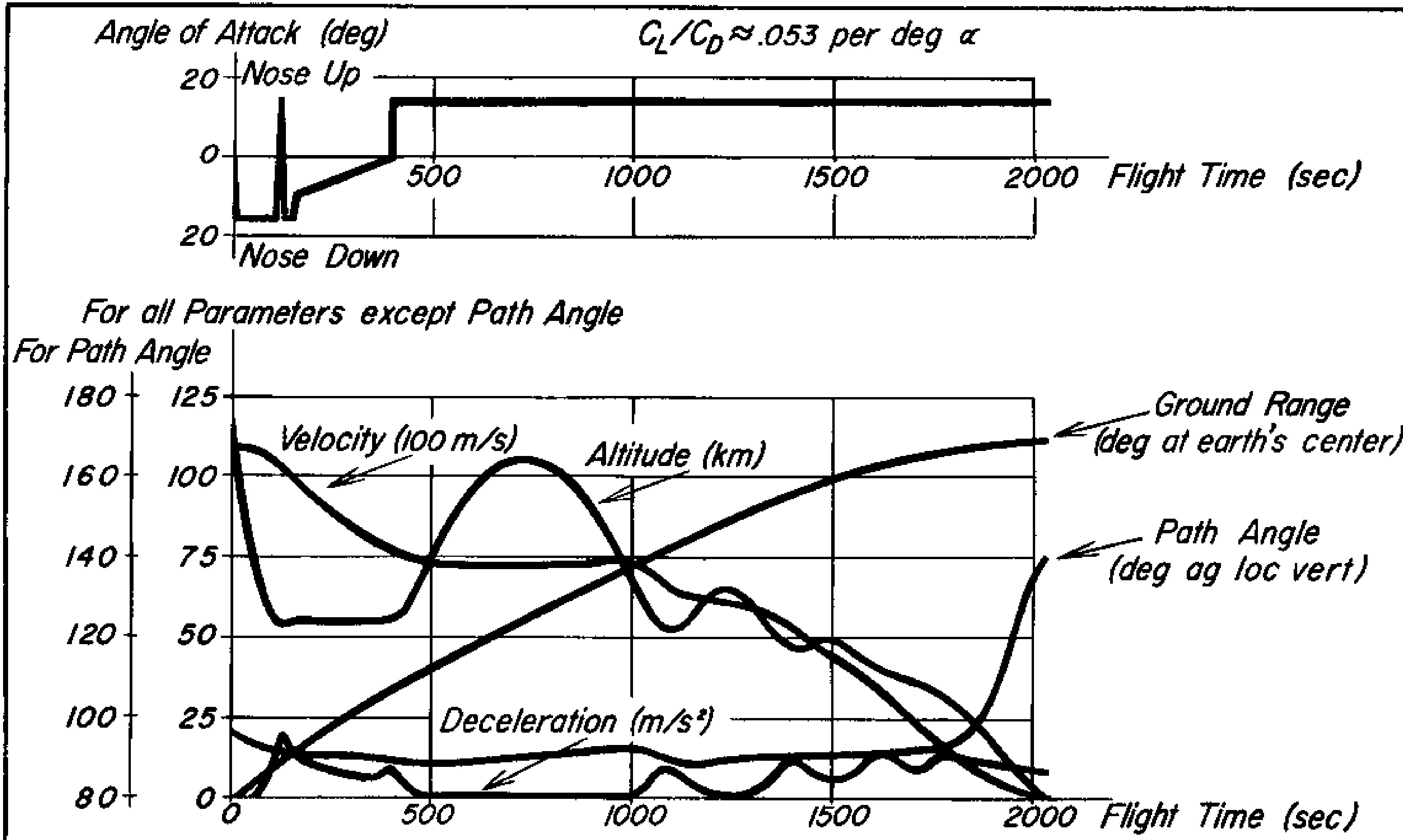


FIG. II.48 LONG RANGE TRAJECTORY FROM FLAT RE-ENTRY PATH ANGLE
HISTORY OF FLIGHT MECHANICAL PARAMETERS
Re-entry Angle of 95.4 deg from Local Vertical

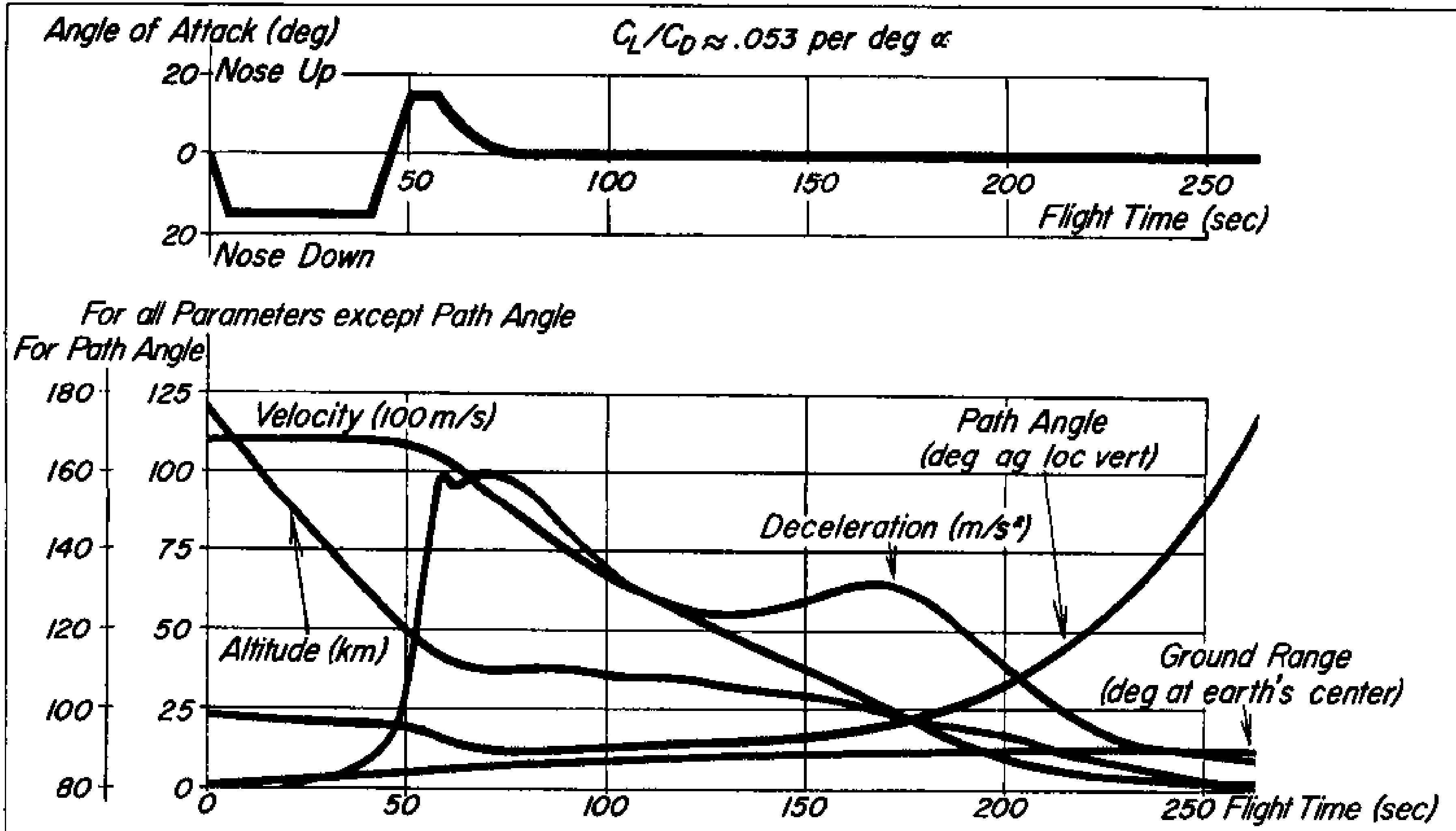


FIG. II.49 SHORT RANGE TRAJECTORY FROM STEEP RE-ENTRY PATH ANGLE
HISTORY OF FLIGHT MECHANICAL PARAMETERS
Re-entry Angle of 98.4 deg from Local Vertical

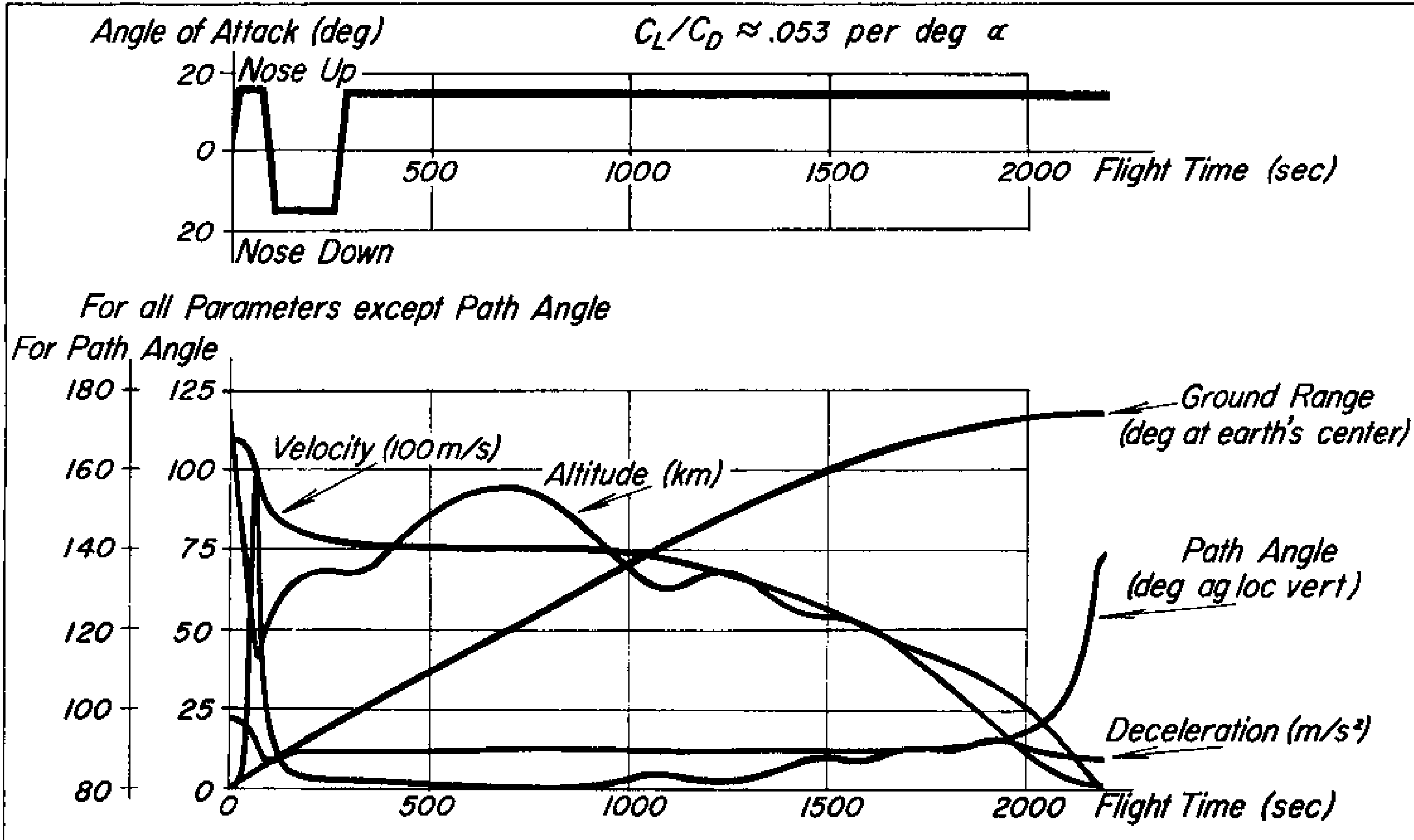


FIG. II.50 LONG RANGE TRAJECTORY FROM STEEP RE-ENTY PATH ANGLE
HISTORY OF FLIGHT MECHANICAL PARAMETERS
Re-entry Angle of 98.4 deg from Local Vertical

the same time one or two days of condition (2) could be waived, about 6 to 7 minima during the year could be considered as favorable.

Lunar probes circumnavigating the moon may require an illuminated lunar back side. Combining this condition with condition (1) and eliminating condition (2), one faces the same conditions and favorable intervals as when 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.

CHAPTER III

PAYLOAD PACKAGES FOR LUNAR LANDINGS AND CIRCUMLUNAR FLIGHTS

INTRODUCTION (C)

This chapter presents a description of the various payload packages considered for the lunar landing and circumnavigational missions and concludes with a discussion of the thermal and materials problems facing the designer.

The performance of the SATURN B-1 as described in Chapter I provides an escape weight of 16,440 pounds. For the direct lunar landing and the circumnavigation missions, a usable payload weight of 10,150 pounds is available as indicated in the following weight breakdown:

Usable payload	10,150 lbs.
Guidance & control equipment and compartment	2,000
CENTAUR engines, tankage and structure	2,765
CENTAUR reserve fuel (3%)	1,425
Separation device (4th stage-payload)	100
SATURN boosted escape weight	16,440 lbs.

For the indirect lunar landing scheme, to be described more fully in Section III.5, the above weight breakdown would not be applicable.

III.1 DESIGN PHILOSOPHY (U)

The unknown effects of space and lunar environment on the materials used in the preliminary studies of the payloads herein described may require a later redesign or even a complete substitution of parts, components, or subsystems. Electronic components, wiring, fuel storage, movable elements, etc. must all be re-evaluated prior to the final design of the flight packages. For example, the propulsion system for the braking or maneuver stage is conservatively based on hypergolic propellants with a specific impulse of 300 seconds. If it is determined that the high impulse (I_{sp} = 420 sec) liquid hydrogen and liquid oxygen system (CENTAUR) can be developed to meet storability requirements and lunar environment operating conditions, then a considerable payload increase will be realized. In addition, use of the 20K CENTAUR engine would eliminate the requirement for developing a new engine.

In general, similar components have been selected for use in the several payload packages. The designs of the shock absorption devices for the stationary and roving packages consider the use of different schemes. This was done for the purpose of this report in order to describe two different possible approaches. However, the devices which will be used in the final designs of these packages will probably be similar to each other since approach schemes and weights are the same.

Investigation of lunar surface areas uncontaminated by braking rocket exhaust gases, desirability to operate through one or more lunar days, and the capability to return to the earth, established the design requirements for the payload packages to be placed on the lunar surface.

A desirable maximum deceleration of 20 earth "g's" for the stationary and roving vehicles has been used in the payload package design. Lunar temperature of +250 degrees Fahrenheit to -300 degrees Fahrenheit and the lunar vacuum environment establish other design conditions.

III.2 LUNAR LANDING VEHICLE (C)

The lunar landing vehicle for the stationary packet is shown in Figure III.1 as it would appear when mounted on Stage IV of the launch vehicle. To reduce as much as possible the propulsion and attitude control requirements in the landing vehicle, Stage IV and the major portion of the injection guidance package are separated from the lunar landing vehicle as soon after vernier correction as feasible. The braking stage engine considered in this study has a maximum thrust in the order of 20,000 pounds (20K). One example of such an engine is a variable thrust motor as developed and tested by Naval Ordnance Test Station (NOTS) with somewhat lower thrust ratings. An engine in the 20K thrust class can probably be made available to this program in time to be incorporated into the landing vehicle. The NOTS 20K engine will use the hypergolic propellant combination of hydrazine (N_2H_4) and nitrogen tetroxide (N_2O_4). Hypergolic fuels will simplify ignition for engine restarts. The guidance scheme described in Chapter II could also be accomplished with a constant thrust engine. A 20K hydrogen-oxygen engine of the type used in the fourth stage would increase the payload very considerably (see Section I.2).

The attitude control system for this braking stage consists of eight 10-pound thrust nozzles. Control of the velocity vector is obtained with two pairs of 100-pound thrust nozzles. The low thrust attitude and velocity control engines will use hydrazine in combination with a catalyst as a monopropellant. The hydrazine is carried anyway as one of the propellants for the braking stage engine.

The braking stage uses a pressure-feed system operated with high pressure gas, either helium or nitrogen. No special provisions are necessary for restarts with the hypergolic propellants. A positive displacement bladder will be used in the propellant tanks to provide a bubble-free head for engine start. The propellants are stored in four spheres inside the 120-inch diameter of the braking stage.

At the time of separation between braking stage and payload package, at approximately 60 meters above the lunar surface, the attitude control nozzles on the braking stage and venting of the pressurized fuel tanks will prevent collision of the two parts by pushing the separated stage to the side.

FIG. III . I
LUNAR LANDING VEHICLE
FOR STATIONARY PACKET

GE 6-6-60
11 JAN 60

The total weight (10,150 lbs) of the lunar landing vehicle is distributed as follows:

Propellants (for braking and three-stop approach scheme)	6,895 lbs
Flight propellant reserve	180
Engine and tankage	900
Separation device	50
Soft landing package	2,125
TOTAL	10,150 lbs

III.3 DESIGN OF THE STATIONARY PACKET (U)

The stationary landing packet is shown in Figure III.2. During launch, the packet is surrounded by an aluminum shroud for aerodynamic reasons. This shroud protects the payload from excessive heating during powered flight through the atmosphere. After leaving the dense atmosphere, the shroud is no longer required and is separated from the payload at burnout of the second stage. From injection to initiation of the terminal approach phase, communication with the earth is necessary. It is provided by two antennae, one omnidirectional antenna, and one disk antenna, which is swung out from the payload to clear the solar cell deck. An electric motor and gear drive actuate the antenna. Swing-out is initiated either by timer after ejection of the shroud or by command from the earth. The disk antenna is used after burnout of the fourth stage and completion of the vernier correction, when the vehicle is rotated 180 degrees and "backed" to the moon. During the flight to the moon, temperature sensitive components can be heated by energy from storage batteries, just as they are later heated during the lunar nights. Upon reaching the lunar vicinity, the terminal approach maneuvers are initiated.

After deceleration of the complete package by the braking stage engine, the burned-out braking stage is separated from the payload at about 60 meters altitude above the lunar surface. Normal separation devices such as explosive bolts or a Morman clamp mechanism will be used. The braking stage is pushed out of the payload free fall path by means of the attitude control

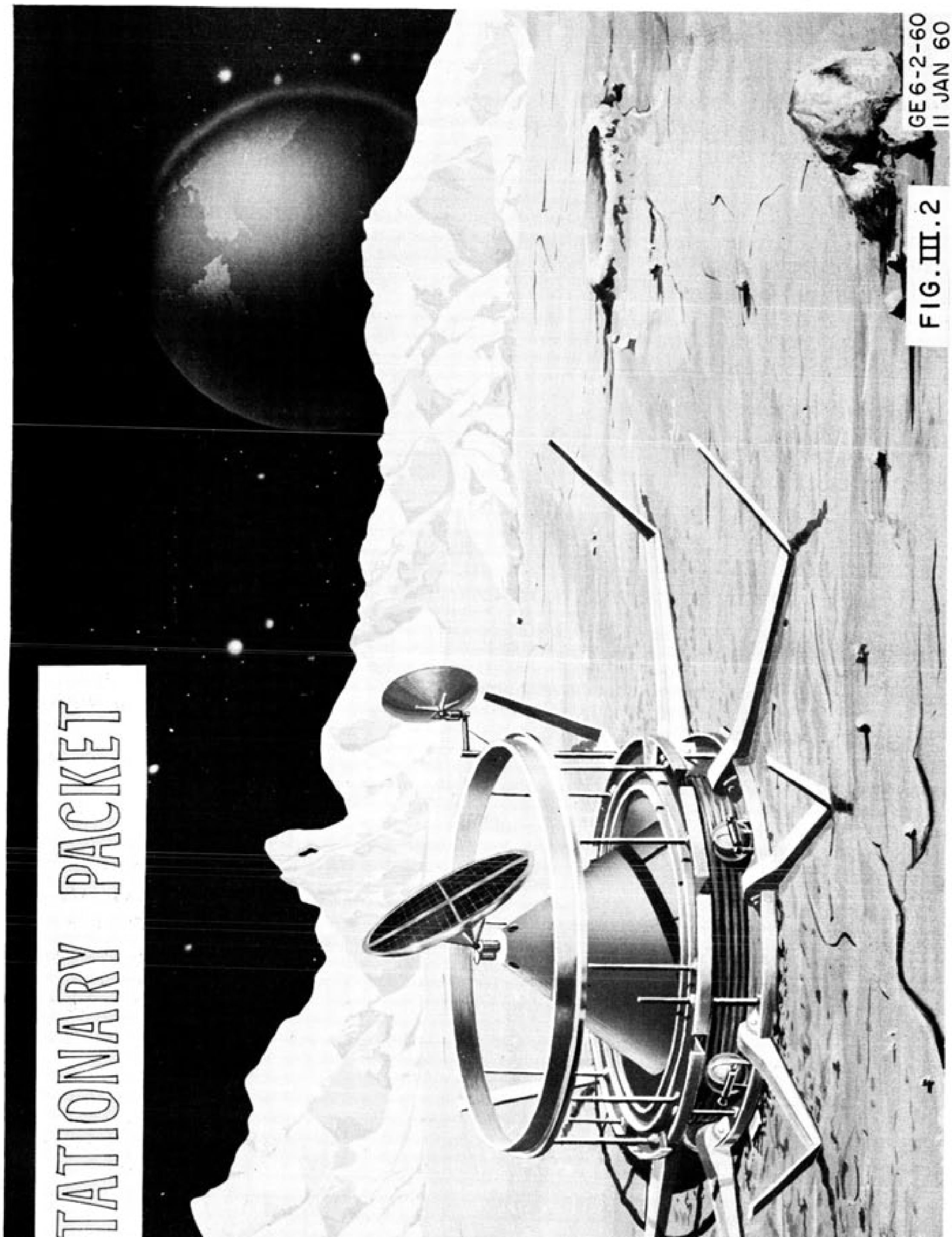


FIG. III.2

nozzles and additional venting of the propellant containers, which hold gas at 350 psi pressure. The payload package falls straight down (providing no lateral motion is present) and is attitude controlled. Immediately after separation from the braking stage, the impact shock absorber is inflated. This device consists of a gas bag below the payload base plate with a metal bottom. The metal bottom will prevent piercing of the bag by sharp rocks at lunar impact. The desired impact deceleration is accomplished by controlled venting of the bag. This assures an almost constant g-load during the deceleration period. After coming to rest, the gas bag will be split open to allow fast and complete release of the gas. Bags of this kind are presently being used for air-drops to cushion impact.

To provide stability against toppling of the payload package, eight fold-out arms are attached to the base of the package. These arms are folded within the shroud during powered flight of the SATURN stages I and II. As soon as the aerodynamic shroud is separated, the arms are unfolded by means of built-in springs at the points of attachment. The positive acceleration during the third stage flight opens the arms completely and then locks them by an automatic locking device. The outer part of the arms act as a crush device to absorb lateral motion at lunar impact. Three guide rods assure compression of the gas bag in the direction of the vertical axis of the payload. These rods are mounted to the bottom plate of the gas bag and pass through three guide holes in the base plate of the payload. They are rigid enough to withstand any lateral motion at impact.

Three mechanical jacks are attached to the lower metal shield of the gas bag. They provide for leveling the payload packet after landing. Leveling is necessary to assure operation of the cooling system, because the radiator surface must not "see" any part of the lunar surface. The jacks consist of a worm gear driven by an electric motor, and are actuated either by an automatic sensing device within the vehicle, or by remote control from the earth by command link.

Attitude control for the free fall after separation is provided by eight 2-lb thrust nozzles. These give pitch, yaw and roll control and use the same gas as the deceleration bag. The helium gas used for both devices is stored in four spheres under a pressure of 3000 psi.

The ring-shaped radiator surrounds the package and is oriented toward open space. A cooling system circulates liquid coolant from the different heat sources within the vehicle to the radiator, where the heat is dissipated to space. All cooled components will be held to a temperature less than 60°C. The cooling liquid has not yet been finally selected, but it will have a high specific heat, and will be noncorrosive.

The solar cell bank consists of a rigid metal mounting surface, a rotating and gimballing device, and the attached solar cells. The gimballing device and rotating device are both electrically driven, enabling the bank to follow the sun to within 30° of the horizon. At sundown, it will reorient itself for the beginning of the next lunar day. The sun's position is sensed by voltage measurements directly from the cells, or by sun seeker cells (photoelectric), which send their signals to the control device. To protect the radiator and solar cell bank from settling surface dust after landing, the components must be kept shielded for some time by a light cover, which can be readily removed.

Besides the solar power supply, the package will contain rechargeable zinc-silver oxide electric batteries. They will provide power to execute functions during and immediately after the landing maneuver, and during the lunar night. More details of the power supply will be given in Chapter VI.

The payload packet itself is a conical-shaped container. All instruments, guidance and control equipment, transmitter and receiver, and all scientific apparatus with the exception of the surface scanning TV cameras are mounted on the base plate (see Figure III.3). The cone acts as a shield from dust, meteors, and cosmic radiation. It has cutouts and access doors as required for assembly and operation of the instrumentation. The surface and subsurface sampling device is located along the center line of the package. It consists of a power drill, a sample conveyor, a viewing device, and a sample distribution system. The samples are distributed to different experiments as described in Chapter IV. Half of the payload packet is used for a temperature controlled compartment. The compartment is insulated by a multiwalled shield and is mounted by ceramic insulators. It houses the battery pack, transmitters and receivers, biological experiment, all the temperature sensitive instruments, and the gravimeter. A portion of the inner walls of the compartment are covered with cooling tubes having metallic contact with the walls. Heat is radiated from the interior to the walls. Excessive heat is removed by the cooling liquid. During nighttime operation heat losses are low because of the multiple walls. Thus, the internally produced heat is sufficient to maintain adequate temperatures inside

CUTAWAY OF STATIONARY PACKET

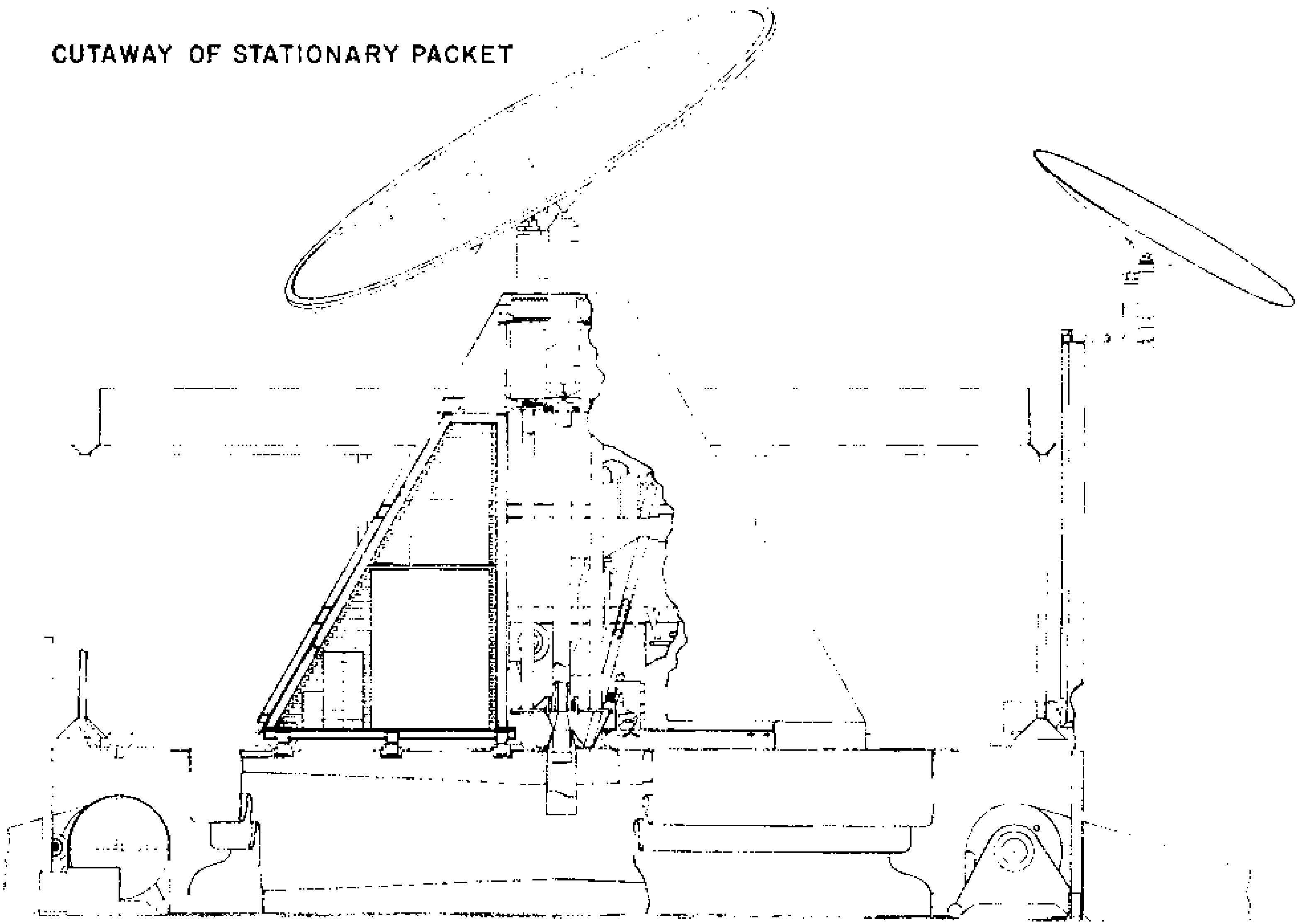


FIG III.3

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the compartment and to prevent freezing of the components. The cooling system is also connected to the video tubes of the TV systems outside of the temperature-controlled compartment to protect them from overheating during the lunar day. The rest of the payload compartment is occupied by other instrumentation and necessary guidance and control components for terminal approach. They are housed in individual containers. All components must be designed to withstand a 20 *g* deceleration and temperature cycling from -150° C to +130° C, except for the temperature-controlled components.

Approximately 700 pounds of scientific instruments can be placed in the stationary packet. A complete breakdown of the 2125 pounds allotted to the packet follows:

Shock Absorber and mounting	275 lbs
Guidance and control equipment	300
Attitude control (including pressurized gas and container)	100
Communications system	125
Power supply - batteries	130
Power supply - solar cells	35
Structure for solar deck	35
Cooling system	150
Structure	275
Available for scientific payload	700
TOTAL	2125 lbs

III.4 DESIGN OF THE ROVING VEHICLE

The second of the lunar landing vehicle designs is that of the roving vehicle shown in Figure III.4. This vehicle is based on the same payload package weight (2125 lbs) discussed for the stationary payload. The landing scheme for this payload is identical with the stationary package, as are some of the instruments used for obtaining lunar surface data. However, the similarity ends at this point.

Because a large percentage of its weight must go into the wheels and propulsion devices, the roving vehicle has a smaller total weight allowable for scientific instruments than the stationary packet. However, it is designed to move about on the lunar surface to provide sampling from a large area rather than a more complete examination of a single point.

No detailed knowledge exists as yet of the actual surface conditions, therefore, the requirements placed on the design of the vehicle might not be completely accurate, but they are believed to result in design features which should overcome any situation arising during the movement of the vehicle.

The general characteristics and requirements listed below were established for the vehicle considered herein:

(1) Withstand impacts of 20 "g"

(2) Operate under the lunar environment for at least one lunar day

(3) Have minimum range of 50 miles, negotiate slopes of 15°, pass over boulders 3 to 4 feet in diameter

(4) Be maneuver-controlled from earth

(5) Be capable of moving about on smooth rock or on a thick layer of dust

When mounted on Stage IV of the launch vehicle, the roving vehicle is inclosed by a protective cover and nose cone. The nose cone is ejected after Stage II cutoff, at which time an omnidirectional and directional antenna are exposed for communication with earth during the lunar flight. The protective cover remains in place to provide protection from micrometeorites during the journey.

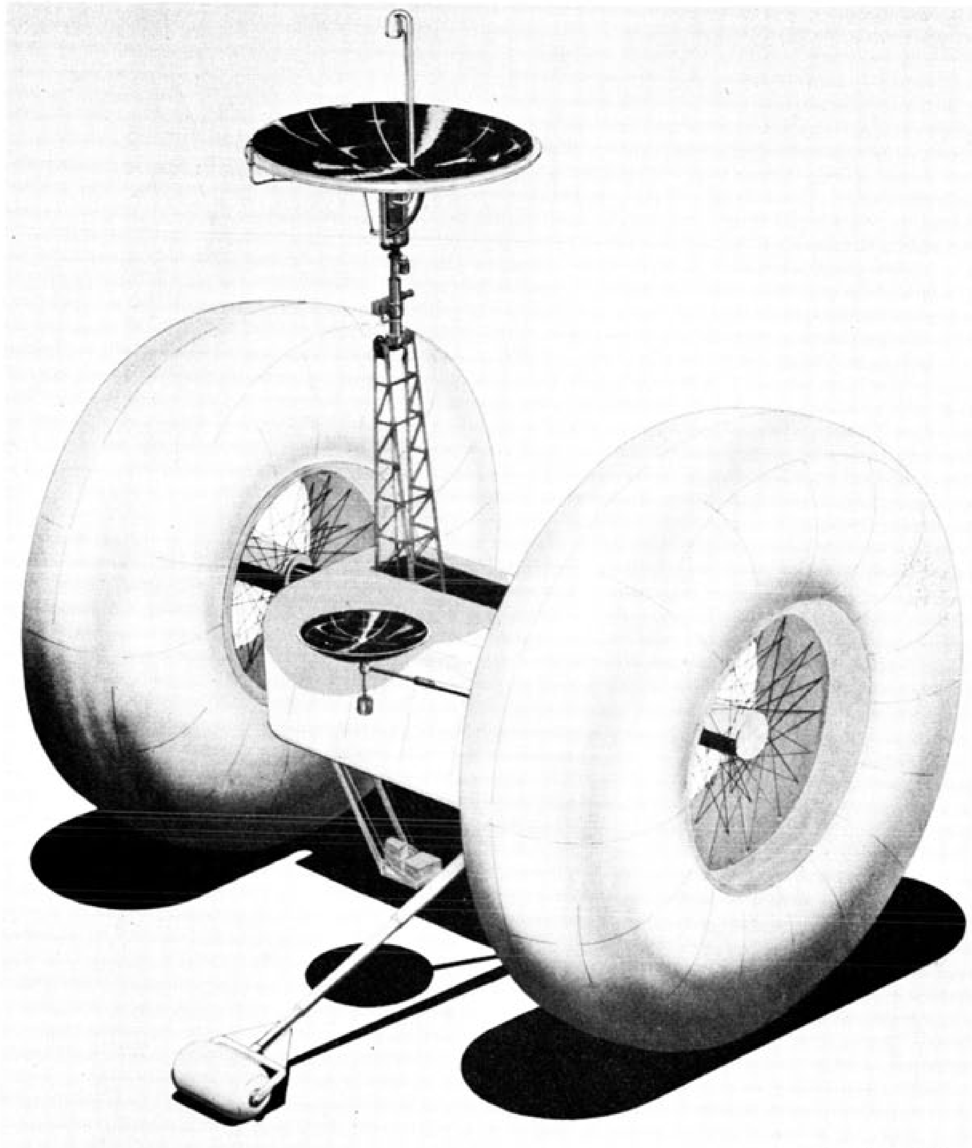

FIG III.4 GE6-5-60 11 JAN 60

LUNAR ROVING VEHICLE

The landing sequence for the roving vehicle is shown in Figure III.5. Prior to the beginning of the terminal approach, the protective cover is blown off the payload package by explosive charges. After the cover is blown off, the vehicle tires will be free to expand immediately to their full size at operating pressure. The vehicle will remain in this attitude throughout the braking and landing phases. Attitude control will be provided during these phases to insure that the payload package lands in an upright position and with minimum lateral motion. When the landing phase is complete and the vehicle has impacted on the lunar surface, it will topple over from the vertical axle attitude to a horizontal axle traveling position. The off-set center of gravity of the vehicle instrumentation package will serve to initiate this toppling motion. Blow-off or spring-loaded devices used to separate the shock absorber and final guidance and control equipment from the vehicle may be used to assist in initiating this toppling action. Once the vehicle is in normal horizontal traveling attitude, the torque reaction arm, the solar reflector, a TV camera on the reflector arm, and the antenna will be extended to operating position by a timing device, and the vehicle will be ready for travel under control of command signals from the earth.

The lunar roving vehicle consists basically of two 16-foot diameter inflatable tires connected by a dead axle. Electric vehicle drive motors will be located in the hub of each wheel. Vehicle track width will approximate 15 feet. The payload of the vehicle will consist of a package of electronic equipment which will be independently swung from the dead axle. The equipment will be shock-mounted in two planes. The center of gravity of the package will be offset from the axle to aid in orienting the vehicle from landing position to traveling position and to keep the package properly oriented during travel. Driving will be possible through employment or a torque reaction arm extending from the axle to the lunar surface which will trail the vehicle regardless of the direction of travel. When direction of travel is reversed the arm will rotate around the axle to its new position. A wheel with puncture-proof tire will be mounted on the end of the arm in contact with the lunar surface. The vehicle will be powered by a turbogenerator operating on a simple Rankine cycle with mercury as the working fluid. Heat source for the power plant boiler will be solar energy concentrated on the boiler by a parabolic reflector. Electrical energy produced by the turbogenerator will be utilized to drive the vehicle, operate all instrumentation as required, and provide power for the instrument package cooling system.

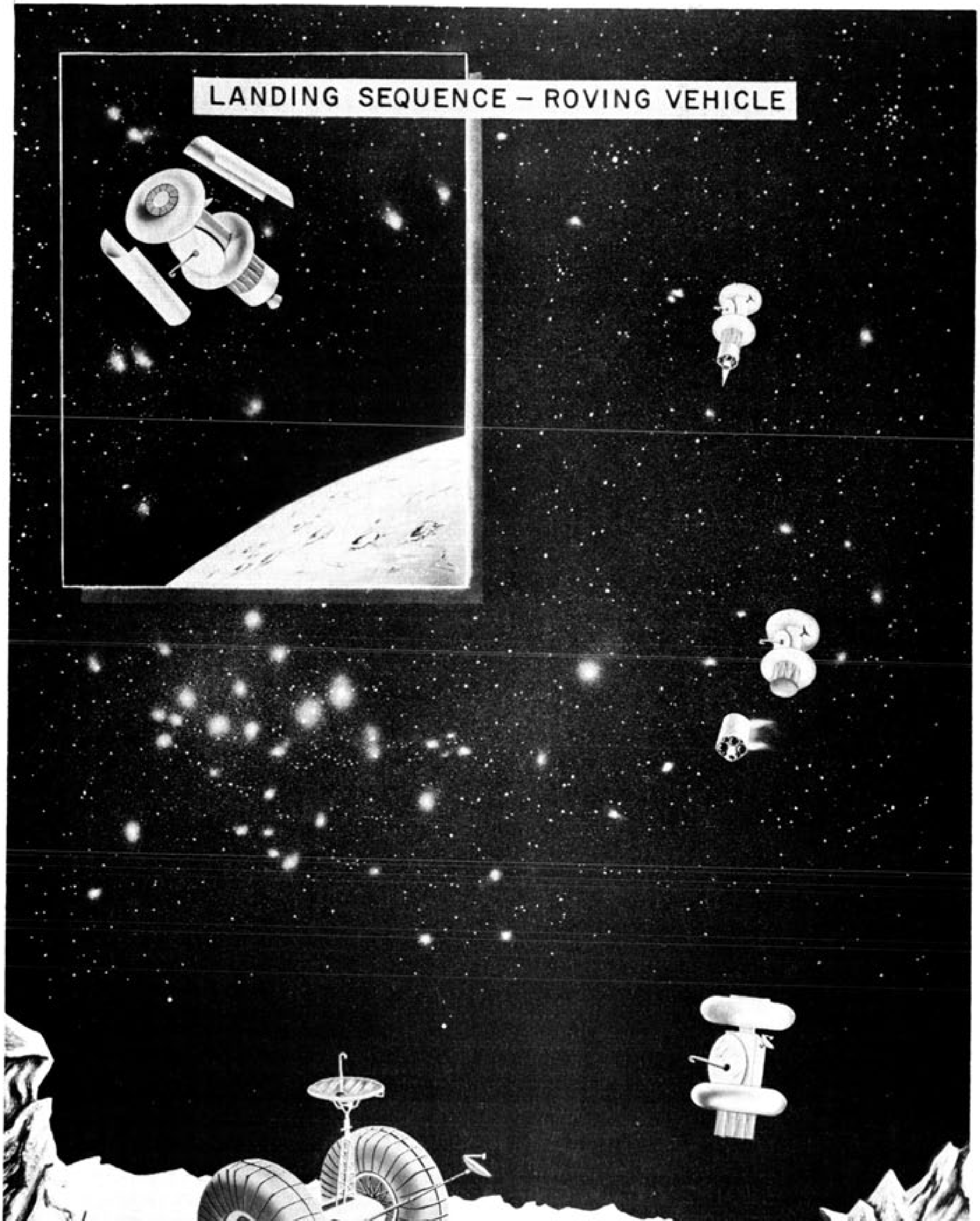

FIG. III. 5

The instrument package is free to rotate about the vehicle axle. As it changes position on the vehicle axle under the influence of lunar gravity, it will act as a pendulum weight to maintain the power plant in a vertical position with respect to the local surface.

The allowable weight of the payload package under the SATURN B-1 scheme has been established as 2125 pounds. This total weight will consist of the weight of the roving vehicle, plus the weight of the shock absorber, the final guidance and control and the attitude control systems which will be detached from the vehicle prior to operation. Design weight estimates are shown in the breakdown below:

Shock absorber and mounting	275 lbs
Guidance and control equipment	300
Attitude control equipment	100
Communications system	125
Power system and network (less batteries)	240
Rechargeable battery package	130
Cooling system	135
Vehicle structure	275
Driving mechanism, tires, wheels,etc.	250
Available for scientific payload	295
TOTAL	2125 lbs

The 295 pounds available for the scientific payload will adequately accommodate the instruments and associated equipment described in Chapter IV.

The primary function of the vehicle tires will be to provide a mobility capability during the full period of operation. However, the tires will also serve a secondary function in absorbing the secondary landing shock when the vehicle topples from its vertical landing position to the horizontal traveling position.

In the event a true vertical landing is not accomplished, the tires may have to absorb a part of the initial landing shock.

The proposed tire is basically a torus having an outer diameter of 16 feet, an inner diameter of 6 feet, and a tire cross section diameter of 5 feet. The tire will be constructed of mylar polyester film or a similar substance approximately 6 mils thick. A net of fine mesh titanium or stainless steel wires approximately 15 mils in diameter will be embedded in this material for added strength and protection from puncture by sharp objects. Operating pressure of the tires will be approximately 1/2 psia. Prior to launch, the vehicle tires will be collapsed for storage in the missile payload section. In this collapsed condition the tire will be inflated with the amount of air necessary to expand it to full size at 1/2 psia after the payload cover is blown off prior to the lunar landing sequence.

Determination of the maximum power requirements for mobilization of the roving vehicle was based on the following assumptions:

(1) Vehicle speed of 3 mph

(2) Zero tire deflection

(3) Friction loss due to tire-to-surface contact is zero (this includes tire material hysteresis loss and surface deformation)

(4) Normal travel will be over flat surface with maximum slope of 15°. Negotiation of 3 to 4-foot diameter boulders will be required. A boulder of this size could be encountered on a 15° slope.

Total continuous power requirements for movement at 3 miles per hour on a 15° slope was determined to be 0.38 horsepower (0.283 kw). In the event the vehicle encounters a boulder or ditch of the specificed sizes, the vehicle speed will be reduced accordingly, and since power delivered to the wheel will remain constant, down to almost zero speed, the torque will increase to negotiate the obstacle.

The vehicle will be driven by two separate electrical motors in order to accomplish the simple control system desired. One motor, along with a planetary reduction gear, will be housed inside the axle at each wheel hub. The motors will be driven by AC current from the turbogenerator of the main power plant. Vehicle control will be achieved by using selsyn units attached to each motor. When both selsyns are turning at the same speed their electrical outputs will be the same and will cancel out. When their speeds are different, their different outputs, when added, will have a net output which will operate a servo control mechanism that brings the two wheels to the same speed. External signals can be fed to the control mechanism to maneuver the vehicle. With a vehicle speed of only 3 mph, control of the vehicle can be simple yet satisfactory. Time delay effects of radio and TV signal travel from the moon to earth and back (approximately 2.6 seconds) tend to make precision control of the vehicle difficult. The value of a complex system, such as speed control and variable turning rate controls, is doubtful largely because of this time delay effect. A simple "on", "off", "reverse" control for each wheel can be used with no provision for speed control. Maneuver of the vehicle can be accomplished by stopping one wheel while the other continues to turn or by stopping both wheels and reversing one. The angular change of vehicle direction will be directly proportional to how long one wheel is stopped. Other important advantages of simple controls are lower weight and increased reliability.

It is expected that a vehicle of this kind, before being sent to the moon, will be tested out very thoroughly on terrestrial proving grounds resembling many different kinds of possible lunar surfaces. Improvements and modifications of the vehicle design will certainly result from this test program.

The lunar vehicle payload is considered as the instrumentation package which performs the desired investigation of the lunar environment. The instrumentation will be housed in an instrument package swung beneath the vehicle axle in the traveling position. All instruments requiring cooling to maintain suitable operating temperatures will be housed in an insulated and refrigerated compartment within the package. Some of the apparatus, such as the sample collector which does not require a controlled operating temperature, will be housed in an uncooled compartment of the package during travel and extended on an articulated arm for operation. Chapter IV contains a detailed description of the instrumentation, its placement within the package, and its operating sequence. The temperature sensitive apparatus can be heated during the flight to the moon by energy from storage batteries, just as they will be heated during the lunar night.

Instrument compartment cooling will be accomplished by a refrigeration unit operating on a reversed Brayton Cycle. The compressor will be driven by an electric motor which will be operated by power from the main power plant turbogenerator. After compression, the gaseous working medium will be routed to a radiator where heat will be dissipated at constant pressure. The design operating temperature of the radiator will be above the lunar surface temperature. Consequently, the radiator can emit heat on all surfaces and will not require constant orientation as the vehicle changes position. The working medium will be routed from the radiator to an expansion turbine where a considerable additional temperature drop will occur. The medium, now at its low temperature, will be injected directly into the instrument compartment for cooling. The work gained in the expansion turbine will be utilized to aid in driving the compressor which will reduce the net external power requirement to operate the system. The entire system, including cooling cycle and instrument compartment, will be hermetically sealed.

Using hydrogen as a working medium, this system will provide the instrument compartment with a gaseous atmosphere at a pressure of 5 psia. The gas will enter the compartment at 38° F and leave at 140° F. Instruments can be placed in the compartment so that those requiring the lowest operating temperature will be near the inlet side of the compartment and those with higher tolerable operating temperature will be near the warmer outlet side.

Preliminary calculations show that other working mediums such as air or nitrogen will result in approximately the same power requirements as for hydrogen. Final choice of a working medium will be governed by more detailed study of such characteristics as leakage susceptibility, chemical reactions within the instrument compartment and density as related to fluid friction losses in lines and rotating components.

The power plant for the roving vehicle is discussed in Chapter VI.

The foregoing design data are merely a summary of some aspects of a detailed report (DLS-TN-26-59) compiled by ABMA on the complex subject of the lunar roving vehicle. The report describes in considerable detail design characteristics of tires, drive power requirements, cooling system, power plant and shock absorbers.

III.5 ALTERNATE SCHEME FOR LUNAR LANDING VEHICLES - INDIRECT APPROACH (C)

The lunar landing vehicle described in Section III.2, the stationary packet described in Section III.3, and the roving vehicle described in Section III.4, are all based upon the terminal approach scheme described in Section II.6 of this report.

During the latter days of this study, an alternate method for landing lunar payloads has been considered. The method has been evaluated only superficially as yet, but it presents several interesting aspects.

In the terminal approach scheme previously described, the landing vehicle approached the moon and descended by direct braking until the final short free fall period. For purposes of discussion in this section, the previous method of placing a payload on the moon may be termed the "direct" approach.

The alternate method, to be discussed here, will be termed the "indirect" approach. Basically, the indirect approach requires that the CENTAUR fourth stage not be separated from the lunar payload after injection, but that it continue to the vicinity of the moon and be re-ignited to place the entire package into a lunar satellite orbit. After establishing the orbit the burned out CENTAUR engine and all excess guidance and control equipment will be separated from the payload satellite to reduce attitude control requirements. The satellite, in addition to its own scientific payload, will carry one or more landing vehicles which separate from the satellite upon ignition of a small solid propellant rocket, proceed to the lunar surface, and reduce their free fall velocity by a second solid propellant rocket prior to impact. The landing would probably be "harder" than that achieved by the terminal approach scheme described in Section II.7, but of course, the scheme itself is simpler.

Using the indirect scheme, without early separation of the fourth stage of the SATURN launch vehicle, 16,440 pounds would be injected into the lunar trajectory. The weight breakdown is shown below:

Fourth stage structure	1180 lbs
CENTAUR engine	1130

CENTAUR reserve fuel (3%) plus trapped residual fuel from injection	1880
Guidance and control equipment and compartment	2000
Braking propellants for orbit establishment	4020
Tankage for braking propellants	330
Engine restart mechanism and fuel	100
Usable payload	5800
TOTAL	16,440 lbs

To establish the satellite orbit, 1015 m/sec braking must be accomplished. A low (174 km) altitude, preferable for lunar surface observation, and a circular orbit have been assumed for this study. The small solid propellant rocket used for separation of the landing vehicles provides a velocity increment of 72 m/sec. A central angle of 90° has been chosen from separation point to impact point because it makes the location of the impact point insensitive to alignment errors of the separation rocket. However, this is not the minimum energy path. The second solid propellant rocket must provide 1705 m/sec braking as the landing vehicle approaches the lunar surface.

Of the 5800-pound usable payload shown in the above weight breakdown, approximately 4000 pounds is actually available for landing vehicles, the remaining 1800 pounds being required for structure, power supply, satellite guidance and attitude control equipment, and communications equipment. The 4000 pounds could be used for several small landing vehicles, or one large one.

Considering several small 1000-pound landing vehicles, or a single 3500-pound vehicle, the weight breakdowns would be approximately as follows:

	1000-lb vehicle	3500-lb vehicle
Separation solid propellant motor (I_{sp}=260 sec)	55 lbs	140 lbs
Braking solid propellant motor	575	1990
Payload:	(370)	(1370)
Power supply	50	120
Structure	40	150
Guidance equipment	60	100
Attitude control equipment	50	100
Shock absorption device	50	150
Communication equipment and antenna	40	100
Temperature control	30	200
Scientific instrumentation	50	450
Total	1000 lbs	3500 lbs

The advantages of the indirect approach scheme are as follows:

(1) Several landers can be carried in one flight which provides an opportunity to test several different approach methods, guidance schemes, solid versus prepacked liquid engines, etc.

(2) For approximately 400 pounds, a minimum performance scheme can be incorporated to return a vehicle from the satellite to the earth. This would permit return of scientific data (photos,etc.) and would provide experience for a moon-to-earth flight.

(3) In case of failure of the landing vehicles, optical observation from the satellite and high-quality slow-time picture transmission to earth may help to find the cause of the failure.

(4) The satellite relay station provides a means to communicate with a lander placed on the side of the moon away from earth.

(5) The lander can send high-quality TV pictures during descent to the satellite relay station much faster than to the earth because of the shorter distance. Using the satellite for storage, such pictures may give the cause of failures, or they may be part of the scientific experiment. It might be advisable to carry an elaborate payload instead of the second braking motor and send valuable information to the satellite before the "lander" is in this case destroyed upon impact. The flat approach to the lunar surface and the short distance to the satellite makes this scheme much more attractive than simple direct "hard impacts," which appears to have a minimum of technical and scientific value.

(6) Spot landings from the satellite orbit should be easier than via a direct approach. Besides, a careful evaluation of the desirability of a landing area can be made from the satellite before the landing is attempted.

(7) The flat approach to the lunar surface should simplify the landing: the jet does not obstruct the TV or radar view to the surface, the velocity component normal to the ground is small, and the horizontal component can be easily measured. With proper selection of a landing area, fairly high residual horizontal velocity components may be tolerable.

(8) The satellite itself carries scientific equipment to give area coverage, whereas the landing vehicles give spot coverage.

Even if all landing vehicles should fail, the satellite alone would be a fair success, providing a number of scientific and technical data from a vantage point close to the lunar surface.

Furthermore, experience gained in establishing a satellite in a lunar orbit and then sending forth a landing vehicle, will be valuable in later lunar and planetary exploration missions, when the landing vehicle must return from the surface to the satellite before departing for earth.

The disadvantages of the indirect approach scheme are as follows:

(1) A smaller amount of useful payload is actually landed upon the lunar surface per flight, because a number of guidance systems, of smaller vehicles, and of solid propellant rockets are used.

(2) The smaller landing vehicles cannot carry a roving vehicle.

(3) The actual landing may be rougher, because the guidance system is simpler, control of the engines is simpler, and a hovering phase is not provided.

(4) Total guidance effort for a satellite plus several landers is greater than in the case of one landing vehicle. Earth-to-moon trajectories for satellite and soft lander are virtually the same. Guidance accuracy required for a satellite in a predetermined orbit is comparable to the requirement for hitting the moon at a predetermined location.

(5) Re-ignition of the CENTAUR engine is required to place the satellite into orbit.

(6) Inclusion of a storage and relay link in the communication line might decrease reliability. On the other hand, the lander can now use simple omnidirectional antennas, which should increase reliability. The net effect is hard to estimate.

(7) Satellite orbital plane and vehicle landing point on the lunar surface must be correlated for convenient communications between lander and satellite, taking into account lunar rotation.

III.6 DESIGN OF THE CIRCUMLUNAR VEHICLE (C)

The circumlunar vehicle is shown in Figure III.6. Design of the manned circumlunar capsule is based upon the ablation-type nose cone re-entry vehicle using a controllable lift program. Flight testing of this vehicle on orbit return missions will begin years before the first lunar flight takes place.

The circumlunar vehicle will be designed for capability of 500 m/sec flight maneuvering. If a hypergolic propellant engine of 20K variable thrust is assumed, a total of 1600 pounds of propellant are required. The 20K engine will control the vehicle until re-entry begins, at which time the heavy guidance package, maneuver engine, tankage and other structure will be separated from the capsule.

The weight breakdown of the circumlunar vehicle is:

Tankage and engine	900 lbs
Guidance and control equipment (retained from injection G&C)	2000
Separation device	50
Propellants	1600
Capsule	7600
TOTAL	12,150 lbs

The capsule for the first flights will be unmanned but will be designed to carry two men on later flights. All systems for the manned operation will be installed and monitored during the initial flights to check their effectiveness.

The capsule flaps are folded within the shroud during the boost portion of the flight. Ejection of the shroud will occur prior to injection as in the case of the stationary packet landing vehicles. Extension and operation of the flaps is controlled by the re-entry guidance package within the capsule.

Because of the short duration of the flight, a chemical system for the control of humidity and CO_2 will be used. The oxygen will be obtained from stored LOX. No provision is made for cooking of food but fluids may be heated if desired.

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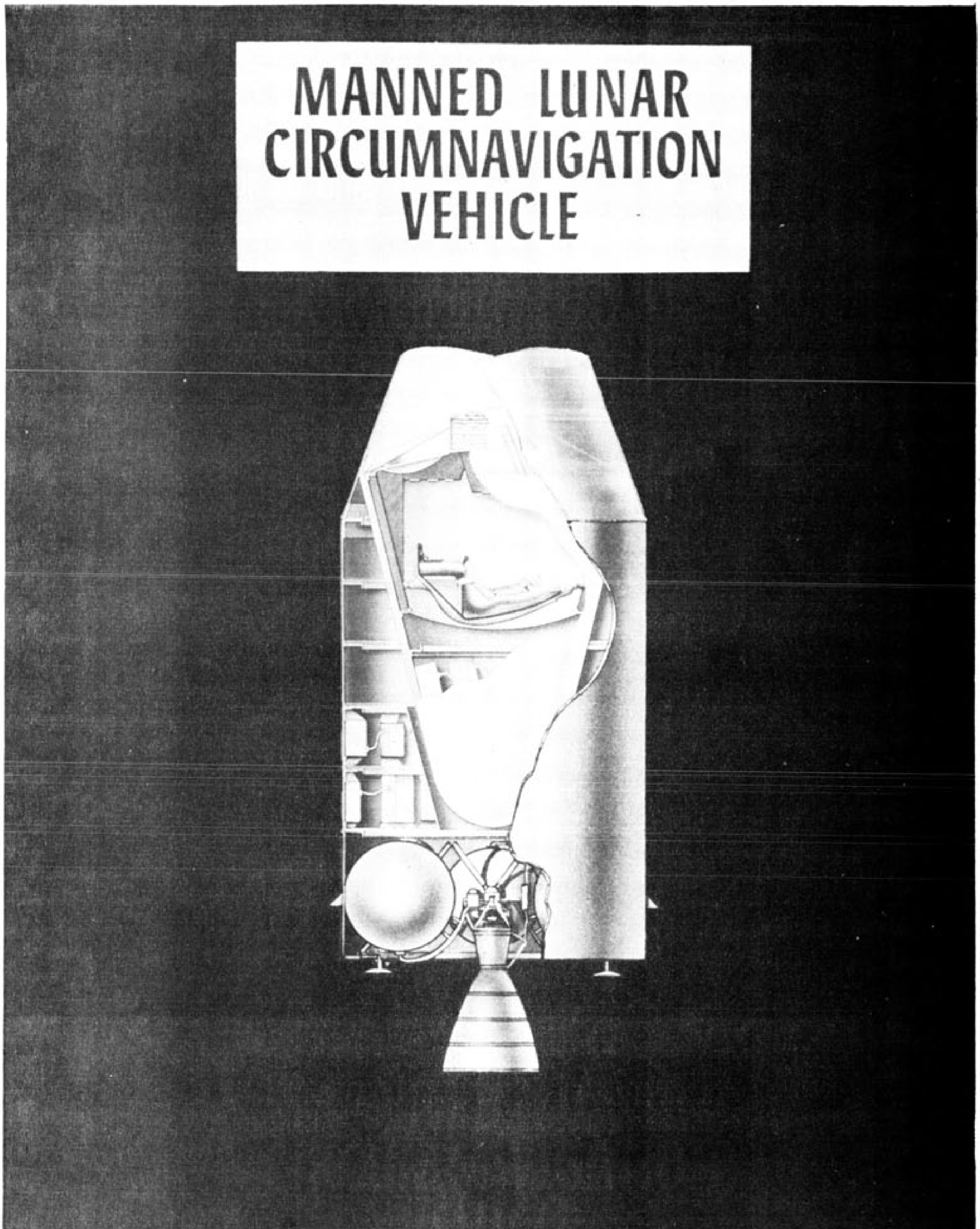

FIG. III. 6 GE6-1-60 11 JAN 60

[redacted]

The passengers will be provided with space suits which will allow them to venture into free space through the air lock in the base of the capsule.

It is envisioned that the shape of the capsule will be similar to that of the JUPITER nose cone design. The ablation material has not been finally selected at the present because many improved materials are becoming available through current tests. Also, materials which are satisfactory for the IRBM and ICBM nose cones may cause excessive internal temperatures, due to higher heating rates and soaking time which will be experienced with parabolic re-entry.

A drag chute is packaged in the base of the capsule to slow the final descent. There is also an inflatable flotation ring for water landings. In addition, the usual safety and survival gear needed while awaiting recovery of the capsule is included.

While the above description has been slanted toward the manned capsule, the unmanned flights will include much of the same equipment. Items such as liquid oxygen, water, and food could be reduced on unmanned flights, giving a greater instrumented payload capability for the first two or three flights.

Safety of the passengers during pad time and lift-off has been very carefully considered in connection with Project MERCURY. Many of the precautions and safety devices developed for that project will be applicable directly to the manned lunar circumnavigation project. Escape rockets attached to the payload shroud will lift it clear of the pad if necessary or will move it safely away from an aborted flight. The rockets will lift the payload to an altitude from which successful deployment of the chute may be accomplished. In a normal flight, jettisoning of the escape rockets will occur at the same time the shroud is jettisoned.

Approximately 250 pounds of scientific payload will be carried in addition to the two men. A breakdown by sub-systems gives the following weight picture for the 7600-pound capsule:

Structure	1400 lbs.
Re-entry heat protection	2800

Guidance and control equipment	400
Attitude control	200
Communications	250
Power supply	500
Recovery system (chute, floats, etc.)	400
Environmental control (temperature, humidity, etc.)	150
Passengers and suits	500
Subsistence	750
Available for scientific payload	250
TOTAL	7600 lbs

A very sizable increase of payload will be incurred if high energy propellants such as liquid hydrogen and liquid oxygen are used. It is anticipated that the problems of tank heat protection for a period of several days under space conditions will be successfully solved by the time the actual design is initiated.

III.7 DESIGN OF A MANNED LUNAR LANDING VEHICLE (U)

The high payload requirement of a project that includes manned lunar landing and return precludes any consideration of the use of the SATURN B-1 vehicle for a direct manned landing on the lunar surface and return to earth. There remains, however, the possibility of a SATURN manned lunar landing mission through an orbital refueling or assembly maneuver. Such a program will be discussed in Chapter V.

III.8 THERMAL DESIGN 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 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 many times from the earth. Results vary between 81° and 134° C.

A flat plate at moon distance from the sun that is a black body on the sun-exposed side with the incident solar radiation normal to this side, but is adiabatic on the opposite side, will have an equilibrium temperature of 123° C. A relatively small flat plate near the moon's surface at the subsolar point and parallel to the moon's tangent plane at that point that is a black body on both sides will have an equilibrium temperature of 123°C.

A flat plate that is suspended above the lunar surface parallel to the moon's tangent plane 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 radiation spectrum, but is black for 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, its equilibrium temperature will be higher than 123°C. This case is represented by a solar cell bank, which will assume a temperature of 190°C or more if no precautions are taken as described in Chapter VI.

Moon albedo has been deduced to be about 7.3% of the sunlight or about 87.6 $Kcal/hr\ m^2$. 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.

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 payloads primarily dependent upon power from solar cells during the lunar day, night operation must be severely curtailed. The problem 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.

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 significant source of energy available to maintain the temperature.

Exothermic chemical reactions become important when chemical sources are used for power. 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 desired constant temperatures, but from the standpoint of payload limitations materials could not be transported to the moon in large quantity for the purpose of thermal control. Endothermic chemical reactions for thermal control are in the same category.

III.8.1 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 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 back side of the solar decks is assumed to be highly reflective in the infrared. If naked solar cells are used without coating of the light-sensitive surface, 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, 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, their maximum temperature will be about 55°C. However, this condition is very optimistic and requires coating materials that are not known yet.

Another way to reduce the solar cell temperatures is to include some radiative area on the front 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 temperature 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 some form of energy, finally converted into heat, is transmitted 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 heat energy at high efficiency during the night. It is very difficult to design elaborate multiple shielding to automatically enclose the decks at sundown and thus conserve energy during the night. Chemical energy or battery-supplied electrical heaters are feasible for heating the cells, if a protective cover is applied. However, this entails a very high weight penalty. It appears that in the present case, it is very difficult to protect the solar cells during the lunar night due to power and weight limitations. Every attempt will be made to design solar cells to withstand a night temperature of about -150°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 this point because its surface must possess a high infrared emissivity. In addition, it must be shielded from reflections from the main body and 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 would be in equilibrium with the shell. Under operation, the radiator temperature would be almost the same as the shell temperature. To maintain the packet at 60°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°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 multiple 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 to within broader limits that represent the range outside of which mechanical damage would occur.

III.8.2 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 of the stationary vehicle, namely:

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

(2) If a closed thermodynamic cycle must be used for the locomotive power, a condenser will be required.

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

III.8.3 Major Research and Development Required in Connection with the Thermal Design. Research and development activity will be required in the following areas:

(1) Precise definition of the temperature limitations of the instruments.

(2) More accurate determination of the moon's temperatures.

(3) Coatings for solar cells that have the ability to reflect "unconvertible" portions of the solar radiation spectrum and that possess high infrared emissivity.

(4) Materials that have low emissivity at all wavelengths and can be applied to large surfaces. In addition, materials that have very low absorptivity in the solar spectrum and very high emissivity in the infrared. These materials should be relatively insensitive to handling and stable in vacuum when

exposed to the unfiltered solar radiation.

(5) Lightweight fluid circulation systems.

(6) Lightweight radiators .

(7) Lightweight multiple walled vessels.

(8) Radiator control systems.

(9) Determination of radiative geometrical factors of shapes that are not quoted in the literature.

(10) Analysis of two and three-dimensional heat flow with transient and radiation boundary conditions.

(11) Analysis of radiative transfer involving re-reflection between bodies of various shapes. (This problem has been solved for infinite flat parallel plates only.)

III.8.4 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 sourthern temperature zone. Every attempt will 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.9 ENGINEERING MATERIALS FOR USE IN THE LUNAR ENVIRONMENT

This section provides some recommendations for engineering materials which will be suitable for continuous exposure to the lunar environment, and which will remain usable without severe deteriorations.

The gravitational attraction at the moon's surface is 0.165 that of the earth. This is of prime importance for the designer, because rigidity can be achieved with less structural material.

The small gravitational pull and the sufficiently high thermal velocities of gas molecules and atoms during the lunar day are responsible for the fact that the moon has lost its atmosphere (if it ever had an atmosphere). Very small concentrations of some heavy and inert gases, such as xenon, krypton, etc., may exist above the lunar surface at pressures of the order of 10^{-8} to 10^{-13} mm Hg.

Although the impact frequency of large meteoritic bodies on the moon's surface is small, meteoritic dust is abundant and may represent a hazard of undetermined magnitude. The micrometeorite particle sizes are well below 1 mm in diameter, but the particles have velocities of the order of 11 to 70 km per second. Such velocities have not been simulated in the labora-tofy, and little is known about the effects of impacts. Some assumptions can be made from laboratory tests. The meteoritic dust concentration in the vicinity of the moon is assumed to be approximately 5×10^{-21} grams per cubic centimeter, and calculations on that basis reveal that the weight gain of the moon in 24 hours amounts to approximately 110 tons. The particle sizes are estimated to be between 0.005 to 0.01 millimeters in diameter.

The following engineering materials are expected to be useful under lunar environmental conditions:

Metals and Alloys

Under the conditions outlined above, most common structural metals and alloys will be suitable for use on the moon. Preference should be given to metals with low vapor pressure, extremely low gas diffusion rate, and a high strength to weight ratio. No penalizing requirements for high fatigue or creep resistance are necessary; however, pressurized containers will, to a very low degree, be subject to creep. Consideration should be given to increased fatigue during vibration in vacuum because there is no dampening effect of the atmosphere. Our technology is not sufficiently advanced to consider beryllium as a structural metal which has the highest strength to weight ratio. The uniformity of the metal and its alloys is still a problem which would affect reliability. The use of magnesium depends on the required lifetime. The sublimation during long-time application (years) must be considered.

Short-time temperature changes dictate a preference for structural materials with a low coefficient of thermal expansion. Steels and stainless steels will definitely have a range of application. Vacuum-melted or vacuum-annealed alloys may have some preference over alloys treated by conventional methods. From the list of stainless alloys, the types 202, 302, 304, 321, 347, 17-7 PH and 15-7 PH Mo are highly recommended. They exhibit favorable properties such as negligible vapor pressure, low gas diffusion rate, and a low coefficient of thermal expansion (α = 11 to 12 x 10^{-6}). The creep resistance and fatigue life is satisfactory for normal application. Solar radiation does not have any deteriorating effect. The resistance of these alloys against cosmic radiation is also good. Their toughness promises the best meteoritic erosion resistance. Titanium and titanium alloys also show very favorable data, particularly the alloys B120 VCA and 6 Al-4V. Their vapor pressure and rate of gas diffusion in the specified temperature range is negligible, and the coefficient of thermal expansion is low (α = 8.9 to 9 x 10^{-6}). Their resistance against solar and cosmic radiation, creep and fatigue is also very good. These alloys will be the most suitable materials for manned capsules and for living quarters on the lunar surface.

For further weight savings, many aluminum alloys may be very practical, particularly from the viewpoint of specific gravity and the ability to be surface treated by anodizing. The anodized surface can be colored by incorporating many inorganic pigments whose radiation resistance is superior to any organic dying agents or organic coatings. Such coatings may become important for temperature control in structures, but for this application more research is needed. The most suitable aluminum alloys are the following: 5456, H24, 5086, H34, X2020T6, 2014T3. Unfortunately, the meteoritic dust erosion resistance is expected to be low, but can be increased by anodized coatings. The coefficient of thermal expansion is also somewhat high (α = 23 x 10^{-6}). These aluminum alloys, with anodized surface coatings for meteoritic dust erosion protection, in combination with a lead shield for X-ray radiation protection, may be the most suitable covering for manned capsules. For special applications, many other metals and alloys can be considered. For X-ray radiation shielding lead and heavy alloy (90% tungsten, 6% nickel, 4% copper) can be used, however, they are not suitable for structural purposes. Vapor coating of precious metals such as gold, silver, etc., will have to be considered for solar mirror power plants. Such metals are not, however, suitable as structural materials.

Nonmetals (Including Ceramics, Glasses and Cermets)

From the large number of ceramics, a careful selection of materials must be made which can withstand the severe temperature shock which occurs during the transition period between the lunar day and the lunar night. Again, preference will be given to ceramics with low coefficients of thermal expansion and relatively high conductivity. The sintered high temperature oxides such as aluminum oxide, magnesium oxide, stabilized zirconium oxide, glazed and unglazed porcelain, and others, will be suitable, particularly as dielectrical materials for antennas and power circuit devices. These sintered ceramics are gas-tight, and their radiation resistance is good. For long-time application, however, an additional glazing with a radiation resistant glass (i.e., lead oxide-cadmium oxide-boron oxide) may be helpful in order to eliminate possible changes in the dielectric properties under long-time exposure to cosmic radiation. Some radiation resistant glasses for lenses, windows, and filters, are commercially available. Quartz glass is recognized as the best transmission medium for the solar radiation spectrum. However, cosmic radiation and X-rays cause lattice defects and subsequent discoloration in a relatively short time. The exposure of quartz windows for optical measurements will be confined to a relatively short time as shutters will be used to protect the windows during periods of non-operation.

Our present knowledge of the influence of cosmic radiation upon ceramic materials and glasses is very sketchy because the simulation of high-energy cosmic radiation in the laboratory is extremely difficult. Ceramic materials such as glass wool, and fiberglass mats will be the most suitable materials for thermal insulation. Multilayer materials, made from alternating layers of fiberglass mats and high reflective metal foils such as aluminum foil or silver foil, are the best known materials commercially available for insulation of containers for cryogenic liquids. The vacuum of the lunar environment assures a low rate of heat transfer. The moon's surface may be covered with a mineral dust or powder of unknown grain size and chemical composition. According to investigations from several authors, this surface layer seems to have very good thermal insulation properties and may be the ideal "domestic" material for thermal insulation of storage buildings and living quarters on the moon.

The use of cermets cannot be predicted at the present time, but most of the cermets will be usable under the lunar environment. All carbides, borides, silicides, and nitrides of the transition metals will be usable under the lunar environment.

Their thermal shock properties are much better than those of the ceramics. These materials will be most useful in the following application: meteoritic-dust-resistant linings for capsules, cutting blades and drill bits for mineral drilling and moving equipment, linings for bonding gears, etc. where no lubrication can be provided.

Organic Materials

Due to the environmental conditions on the moon, organic materials will not be applicable for long-time usage. The effect of vacuum and temperature extremes causes breakdowns of the molecular structure, decomposition, and vaporization at high temperatures, and brittleness of the weakened structures at low temperatures. The cosmic and solar radiation intensifies these effects. Organic materials cannot be considered as safe and reliable structural materials. For short exposure time applications such as space suits some silicone rubbers, silicone plastics and teflon compounds can be considered. The life expectancy of such materials can be increased by reinforcing them with radiation resistant fiberglass or metal fibers. In areas shielded against solar and cosmic radiation, the deteoriation of such materials can be retarded considerably. As mentioned previously, organic dielectrics as used on earth will be replaced by ceramic dielectrics. For example, organic dielectrics on copper must be replaced by fluoridized copper coatings having some flexibility and adherence to the copper conductor. On aluminum conductors, anodized surfaces provide excellent electrical insulation, also aluminum fluoride coatings have shown attractive properties. In cases where organic dielectrics cannot be replaced, shields must be provided to protect organic material. Mylar has relatively good low temperature and vacuum stability and flexibility. The cosmic radiation stability is unknown.

Another case where organics cannot be replaced by inorganic materials is that of lubricants. However, lubricity in vacuum can be obtained by replenishing the lubricant lost by vaporization. Over 1000 hours of load bearing operation in high vacuum have been obtained by proper selection of silicone grease-based lubricants. In space or on the moon, the bearing must be shielded and an adequate grease reservoir must maintain the lubricity film on the sliding surfaces. High speed, high load bearings capable of operating for one year under vacuum conditions are now undergoing development; they show promising results. For low speeds and intermittent operation, molybdenum disulphide and tungsten disulphide are acceptable. They can provide lubrication at elevated temperatures (up to 300° C), low temperatures in vacuum, and under cosmic and solar radiation conditions.

Special Materials

There may be applications for special materials, e.g., solids of a certain crystal structure and degree of atomic or ionic order such as transistor materials, solar cells or semiconductors. As mentioned previously, some glasses and ceramics undergo changes under long-time exposure to cosmic radiation. This applies also to semi-conductor, transistor and solar cell materials where the lattice microstructure is of vital importance. No general statements can be made at this time because the influence of cosmic radiation is not known. Shielding of these materials will probably be necessary in order to maintain reliable operation.

This section has covered only general aspects and suggestions for materials applicable under lunar environment. Details for specific structures can be provided when requirements are known. Many of the suggested structural materials are presently under investigation and valuable data are being obtained.

CHAPTER IV

LUNAR EXPLORATION

IV.1 PRELIMINARY CONSIDERATIONS CONCERNING LUNAR EXPLORATION

Before a SATURN vehicle is used to land a large payload softly on the moon, much will have been learned about the moon from experiments transported by less sophisticated vehicles.

This suggests 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 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.

The broad objectives of the experiments are two-fold. The first, and initially paramount objective, is the augmentation of scientific knowledge concerning the moon and its surroundings. A secondary objective is the development of technology for later application to manned lunar operations. The vigorous pursuit of the first objective is probably as expeditious as any means of pursuing the second. Therefore, in the following discussions, the scientific objectives have been emphasized.

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 paragraphs.

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.

Many of the instruments used will surely be modifications of instruments used on previous scientific packages. The nature of the modifications required will in many cases be dictated by this previous experience, and cannot be predicted in advance.

Early lunar probes and impact vehicles, both from the United States and USSR, have been carefully sterilized to reduce the danger that the lunar surface will be contaminated with living organisms from the earth.[1] 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.

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.

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[1] Publication #698. Jet Propulsion Laboratory, August 1959
R. W. Davies & M. A. Comuntzis

The elimination of serious outgassing is particularly troublesome on the roving vehicles. Thus an attempt should be made to make all required atmospheric measurements with instruments on the stationary packets. Extreme care should be used in designing the stationary packets to avoid the use of materials with excessive outgassing. This requirement could be relaxed for the roving vehicles where less emphasis would be placed on atmospheric measurements.

IV.2 INFLUENCE OF THE LUNAR ENVIRONMENT

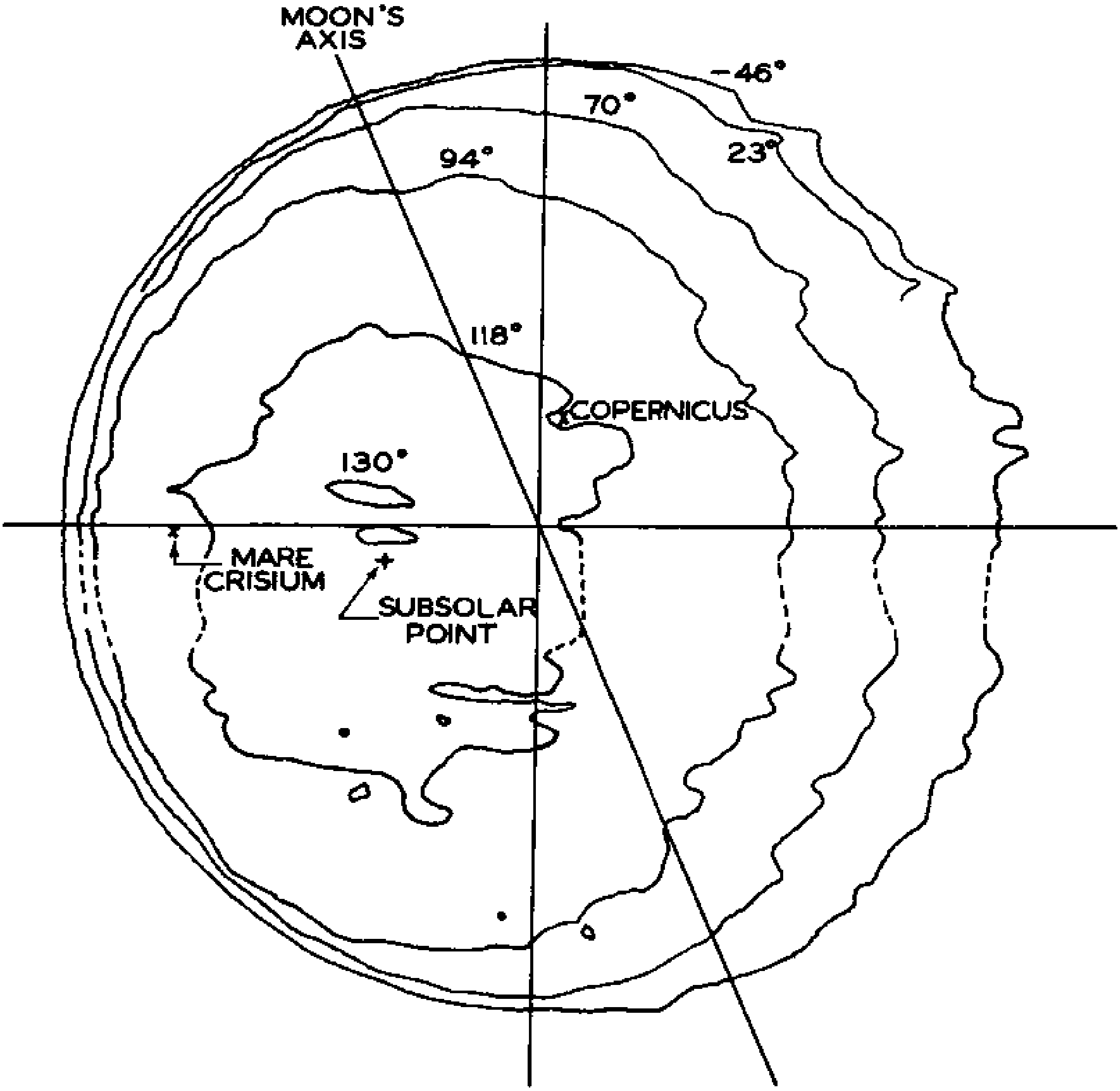
The proper functioning of equipment landed on the moon will be insured by properly taking into account the influence of lunar environment on equipment operation. Aspects of the lunar environment which are particularly important are lunar surface temperatures, the nature of the moon's surface, its topography, and its lack of atmosphere. Unfiltered cosmic and solar radiation and micrometeorite bombardment will become important considerations for projects involving manned landings and long time exposures to the environment.

The high maximum surface temperatures and the minimum night temperature experienced by an unprotected vehicle are discussed in Section III.8. Figure IV.1, taken from Sinton's work[1], shows an example of the distribution of surface temperatures on the moon.

Of all the environmental influences experiences by the scientific apparatus, that imposed by the temperature on the electronic circuitry is the most serious. In particular, transistorized circuits and photomultiplier tubes have narrow operating ranges. The gravimeter will require thermal control even though it will be designed to operate either at 0 or 60 degrees C. On the other hand, the following instruments are quite insensitive for the lunar day - night temperature range: Geiger-Mueller counters, certain magnetometers, and mass-ion spectrometers of the conventional magnetic-electrical field separation type.

Some of the features of the lunar surface having a bearing on roving vehicle design are: surface roughness and competence, obstacles such as large blocks of debris, crevices

[1] Private communication from W. Sinton, Lowell Observatory.



ISOTHERMS ON THE LUNAR DISK

FIG. IV. 1

or rills, the behavior and the thickness of any dust layer.

Continuous bombardment of the moon by meteors during geologic time must certainly have caused a pitted and irregular surface. Impacting with their space velocity unimpeded by atmosphere, even the smallest particles produced craters and debris. Then too, a considerable mixing of the meteoric material with that of the moon will have taken place.

The question of competence of the moon's surface arises often, especially in the popular literature. There is no basis, presently, to assume that the moon is covered with thick layers of loose dust through which the vehicle will sink. Dust in a vacuum compacts very closely and is able to bear loads like any competent material. Over geologic time, one expects that the dust particles, stripped of their gaseous film, would form a strong bond. Urey, Kopal and Alter[1] give estimates of the dust thickness ranging from 1 to 30 cm. In the upland regions, greater accumulations of dust may exist in crevices and valleys.

In and around the explosion craters, debris has been scattered which offers some obstacle to the maneuvering of a mobile vehicle. Large blocks which are widely spaced can be avoided, but debris on the order of a few feet in size will require that the vehicle be designed to traverse them. The danger from obstacles about the size of the vehicle, not discernible in telescopic observation, may be avoided during the descent phase through the use of high resolution television. The earth-bound operator will have the opportunity to land the vehicle away from obstructions which might tip the vehicle or place it in an unfavorable position.

Crevices or rills on the moon's surface present a problem which cannot be eliminated entirely by vehicle design. The problem may be solved to some degree by using large diameter wheels with wide treads and wide wheel base, thus eliminating the need for the vehicle to avoid the smaller rills. During the descent phase, the high resolution television view obtained may be used to chart a course for the roving vehicle away from the wider crevices.

Lack of appreciable atmosphere is a problem which becomes more serious as the complexity of the exploration apparatus increases. In the SATURN surface vehicles, the drill-sampler

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[1]Private communications to R. Jastrow, NASA.

mechanism, bearing surfaces in the mobile unit, and plastic components are especially susceptible to this aspect of the lunar environment (see Figure III.9).

The drill-sampler mechanism is made up of a number of components, some of which are required to operate with a high load whereas others are not. Where light loads and short periods of operation are concerned, wear and galling may not destroy the function of the component. Apparatus handling only a few ounces of material for short periods may utilize unprotected bearings or moving parts. A recent development by Columbia Broadcasting System laboratories[1] indicates that certain bearing materials have long lifetimes in vacuum (approximately 10^{-7} mm Hg). Certain plastics, found by experiment to be unsatisfactory in high vacuum,[2] will not be used.

IV.3 OBJECTIVES AND PROGRAM OF SCIENTIFIC INVESTIGATIONS

A diverse group of instruments is 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.

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

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[1] Private communication from J. W. Christensen, Columbia Broadcasting System.

[2] National Research Corporation Contract #DA-19-02-ORD-4635; Army Ballistic Missile Agency: 1958.

reconnaissance system; 2) gravimeter; 3) seismic apparatus; 4) thermal measurement devices; 5) special sample collecting equipment; 6) X-ray fluorescence and diffraction apparatus; 7) alpha and gamma radiation counters; and 8) gas analysis equipment. 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 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, may be carried in the stationary packet and roving 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 a plasma probe.

Measurements of the lunar magnetic and electric fields are also included in the stationary packet. 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.4 LANDING SITES

The proposed SATURN exploration program on the moon begins with the landing of a stationary scientific packet of considerable complexity on the moon. This initial landing of an unmanned scientific laboratory will be followed by landings of roving vehicles capable of surveying a large area of the moon's surface.

The site suggested for landing the stationary packets is the location where the nominal earth - moon trajectory terminates with a vertical incidence on the moon's surface. From the flight mechanics point of view, this is the easiest point for landing. It also corresponds to the greatest probability of success on early flights. The discussions in Chapter II assume such a landing site.

The roving vehicle will be best utilized by landing it within roving distance of several strikingly different features of lunar topography. If such a site is chosen, more stringent demands are made upon the approach guidance system, since the site may not correspond to a vertical incidence for the earth-moon trajectory. However, drawing upon the experience gained from landing the stationary packets, adapting the approach scheme to the more ambitious objectives of the roving vehicle should be practical. Operating the roving vehicle in an area away from the equatorial region makes easier the task of thermal control of the vehicle and its instrumentation.

<u>IV.4.1 Landing Sites for Stationary Packet</u>. The stationary packet makes possible a study of the moon's gross properties and an analysis of its surface and subsurface materials. It provides as well a visual survey of the landing area. Scientific information obtained by the initial landing, plus the functional information concerning the behavior of the vehicle components in the lunar environment, will prove to be valuable for later vehicle design.

Mechanical considerations, such as those involved in gravity orientation of directional antennas to earth admit a large portion of the lunar disc as suitable for landing. On a stationary packet, where unwieldiness of the structure is not of great concern, the addition of thermal protection hardware allows the vehicle to rest in the thermally unfavorable equatorial regions.

The scientific program, likewise, permits the selection of sites from a large area on the lunar disc. Except for supposed "hinge line" areas around the larger Maria where it is thought that local outgassing may occur infrequently, the location of the stationary vehicle is not critical for studying the lunar atmosphere. The program involving the "moon tide" gravimeter, as well as early seismometer experiments, may be carried out in any part of a vast area on the lunar disc.

The area chosen for the initial soft landing is shown in Figure IV.2. The nominal earth - moon trajectory terminates in a vertical incidence at a point in the area bordered by the craters Kepler and Landsberg. A more detailed view of the surface features present in the landing area is shown in Figure IV.3. The floor of the Oceanus Procellarum in this region is marked by several large post-mare disturbances. One may suppose that the area for miles around these disturbances, both large and small, abounds in debris unobservable by an earthbound telescope. Likewise, remnant of pre-mare craters forming low ridges on the lunar plain may offer local obstacles to a successful landing.

The interior of post-mare craters and the area immediately around them will likely contain much fractured, metamorphosed material, as well as debris of significant size compared to the size of the packet. Landing sites of this type will be avoided in the initial vehicle. Not only would such a landing tax the ability of the vehicle to obtain lunar samples, but it might confuse and make even more difficult the proper interpretation of data from the mineral analysis.

During the terminal phase of lunar flight, the vehicle will be guided away from ridges, rills and the interior of post-mare craters. The earth-based operator landing the stationary packet will have a number of suitable landing sites available. Unlike the roving vehicle, in which the irregularity of the surface is of concern, the stationary packet requires only a small site of relative smoothness on which to land.

<u>IV.4.2 Landing Sites for Roving Vehicle</u>. The roving vehicle is designed for a range of over fifty miles. Landing sites within such distances of interesting selenographic features are highly desirable.

Several suitable landing sites are described briefly in the following pages. These represent only a few of the sites which have been suggested by individuals in publications, meetings,

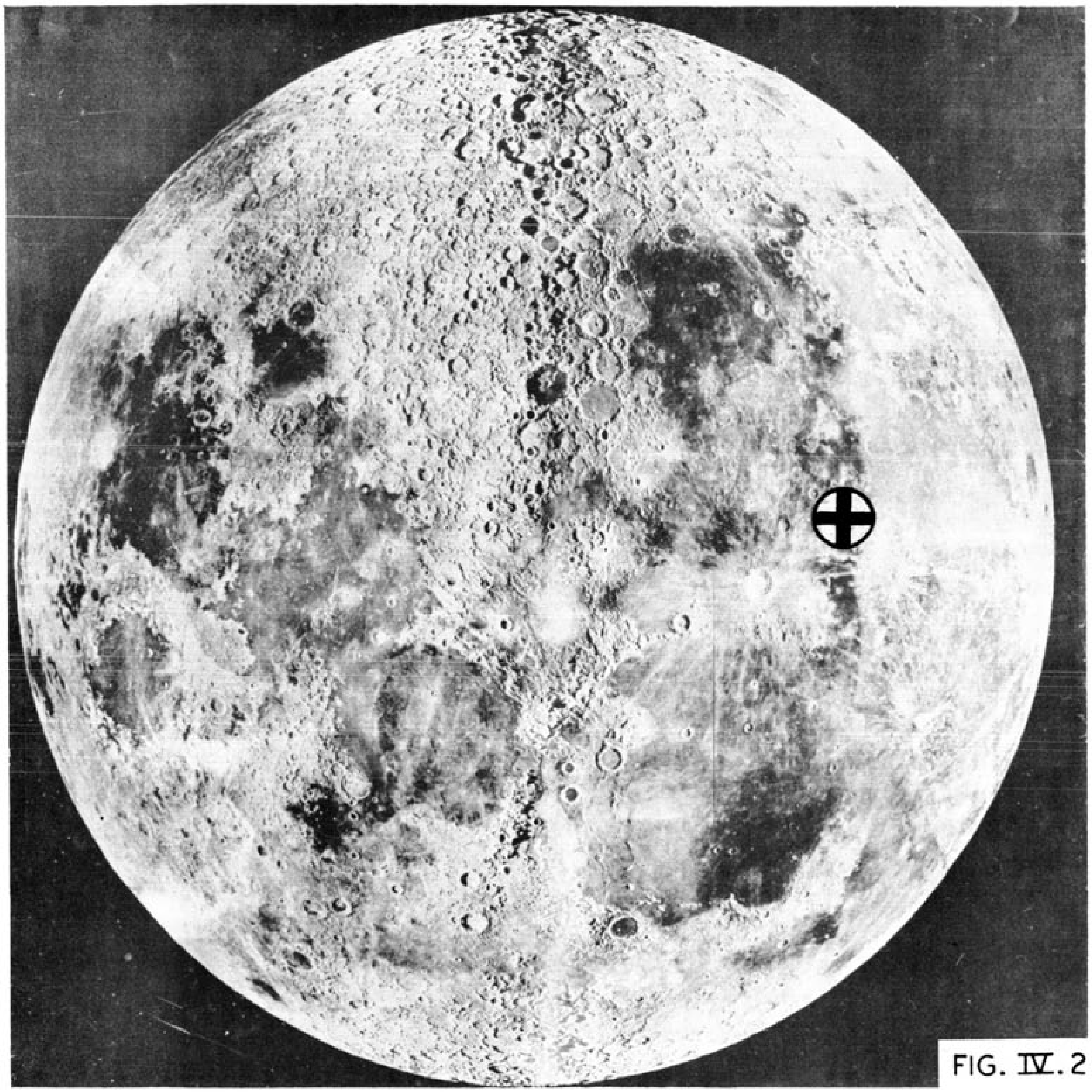


FIG. IV.2

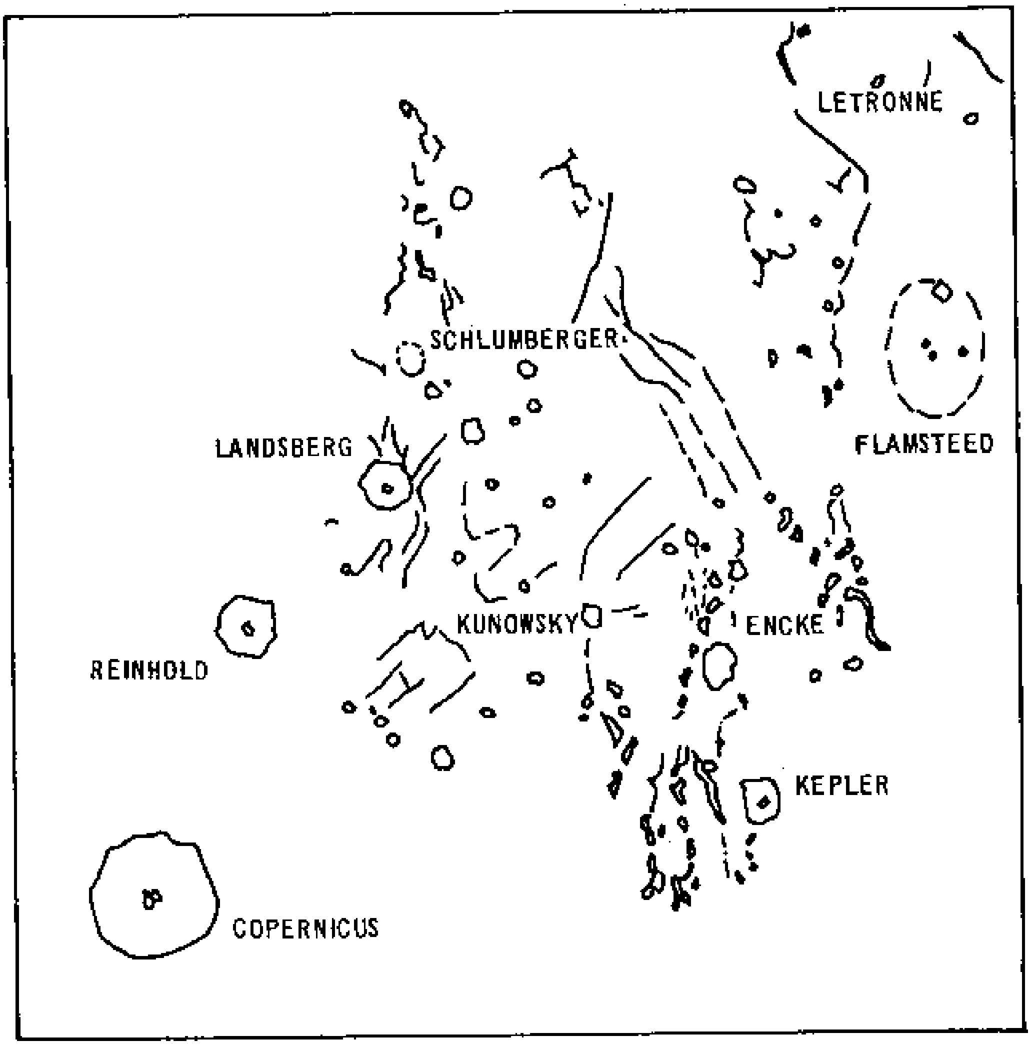


FIG. IV.3

DETAIL OF INITIAL LANDING SITE (AFTER LAMECH) etc., during the past few years. While the list of interesting sites is certainly incomplete, those included here represent the most often mentioned landing areas.

One such spot might be near the straight wall in Mare Nubium, another in Mare Imbrium near the Apennine Mountains, or the Alpine Valley region of the Alps. From another point of view, a roving vehicle landed in a large walled plain, such as Alphonsus, will be capable of searching the area for local outgassing or evidences of a past vulcanism.

The landing sites selected for discussion here are in widely separated areas of the moon; one (Stadius-Copernicus) is in the equatorial region, two are in the southern hemisphere (Straight Wall and Alphonsus) and two are in the northern hemisphere (Mare Imbrium and Mare Frigoris). See Figure IV.4.

Stadius-Copernicus (Figure IV.5)

Many small chain craters exist in the region of Stadius-Copernicus on the south limb of Mare Imbrium. This area has been mentioned as one that might offer suitable natural abodes for manned occupation of the moon.[1] For a future roving vehicle, the crater Stadius offers a landing site within vehicle range of the lengthy crater chain to the northeast. The roving vehicle in this site will employ a maximum of equipment for reconnaissance purposes.

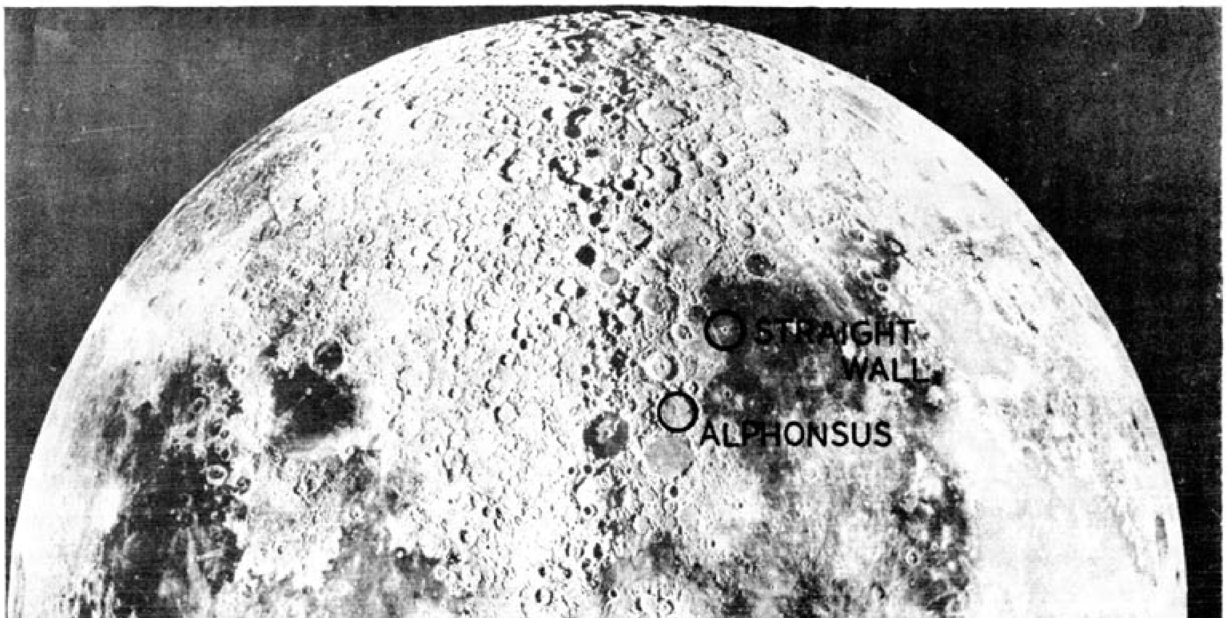
Straight Wall (Figure IV.6)

This remarkable structural feature is a fault some 60 miles in length with a vertical displacement of approximately 800 feet. Small craters lie on both sides of this fault and have been reported by Moore.[2] The structure lies near the middle of a hexagonal pre-mare ring which is traceable in the photograph.

A roving vehicle, landed on the downthrow side of this structure, will observe some 800 feet of vertical section of the mare with high resolution television. Presumably the vehicle may be modified to obtain samples for analysis near the base of the displacement. Information from the analysis of

[1] Scientific Aspects of the Lunar Surface. Dinsmore Alter, Proceedings of the Lunar & Planetary Exploration Colloquium, Vol I, No. 1, May 1958.

[2] The Moon - Wilkins & Moore. Faber & Faber, Ltd., London.


STRAIGHT WALL
ALPHONSUS
STADIUS - COPERNICUS
PALUS - PUTREDINIS
PLATO
HARPALUS
ARISTOTELES
MARE FRIGORIS
FIG. IV.4

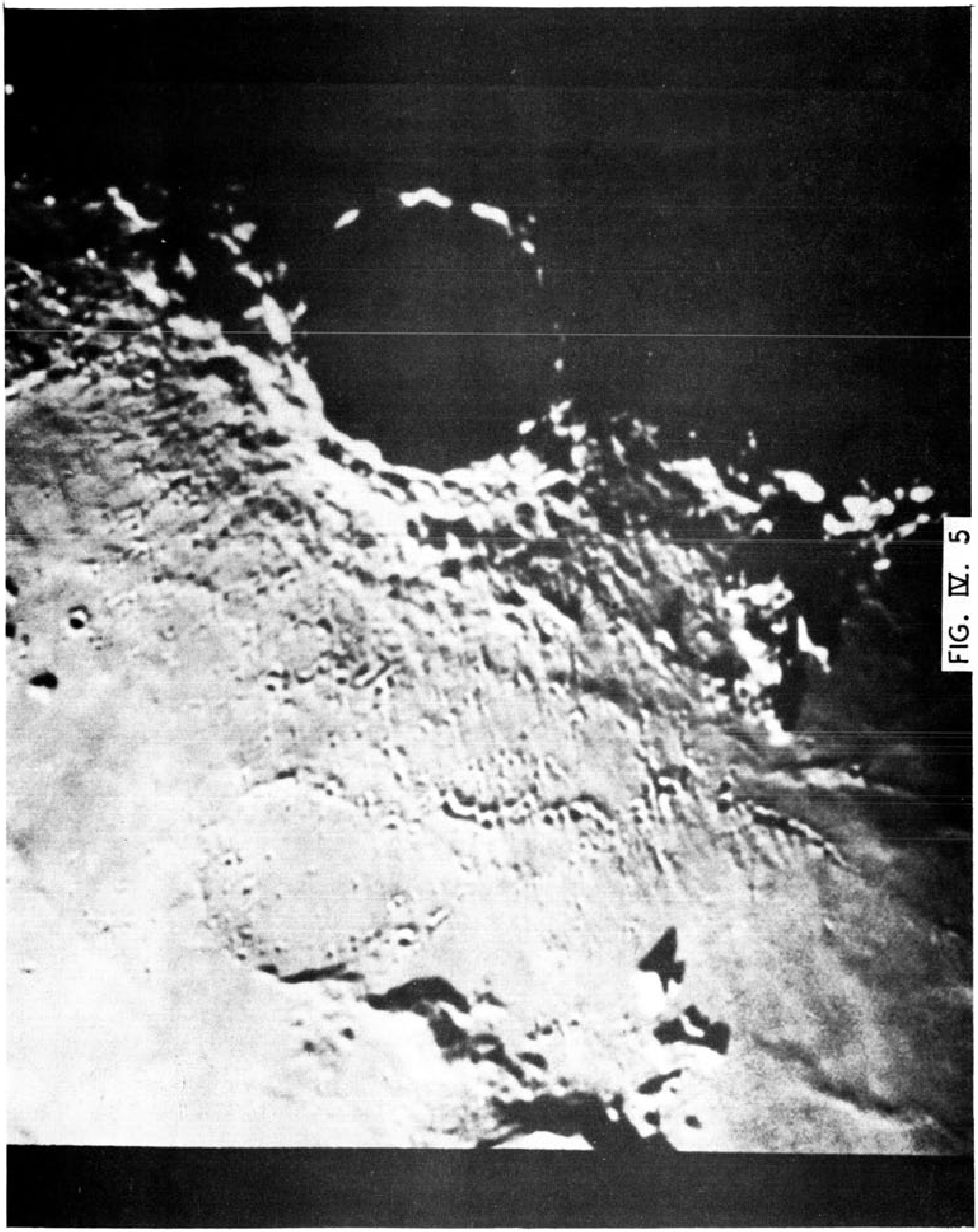


FIG. IV. 5

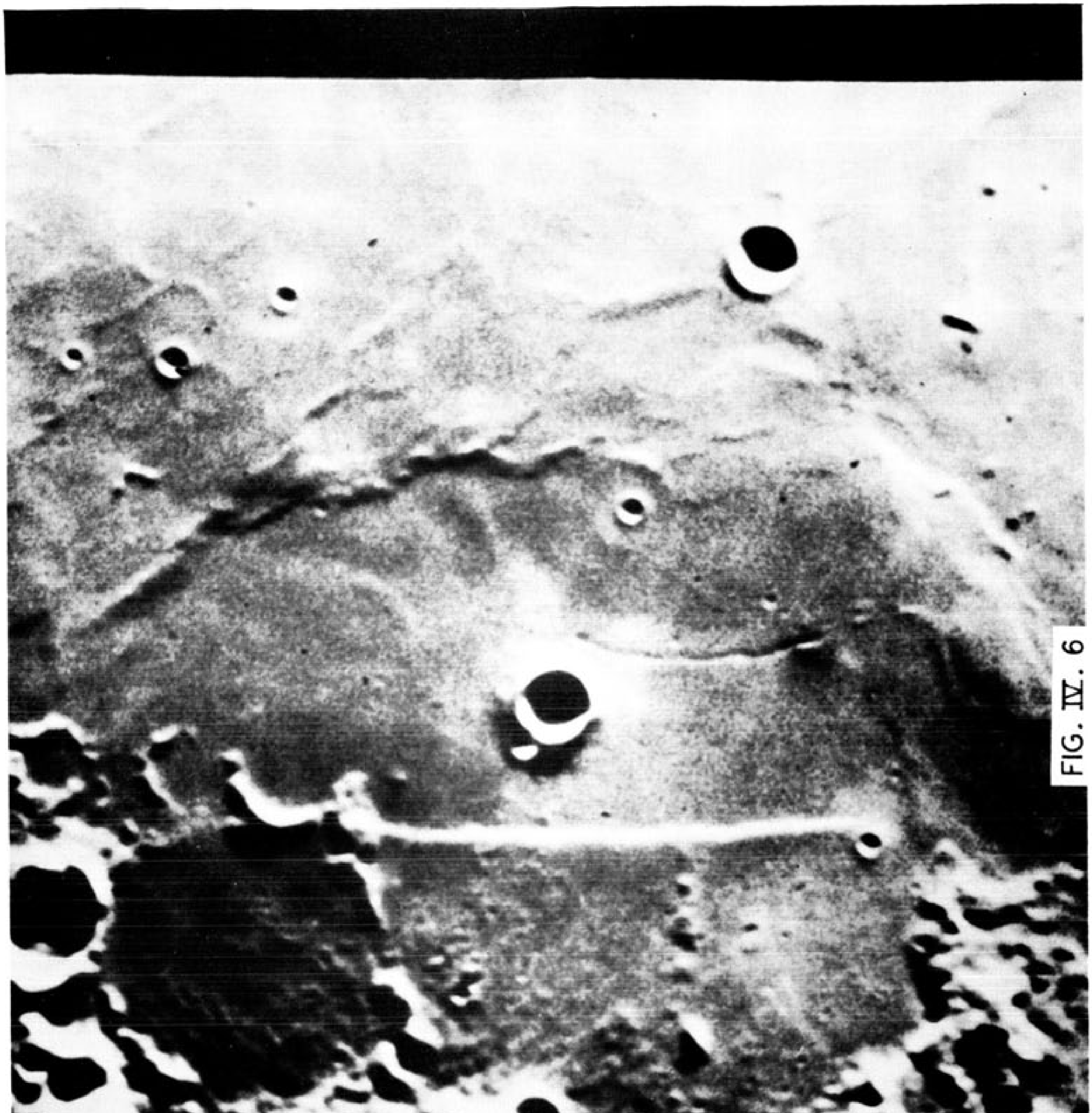


FIG. IV. 6

such material, coupled with an analysis of the surface under the vehicle, may give a comparison of surface composition and composition at depths below the surface.

Alphonsus (Figure IV.7)

Alphonsus lies to the north and west of the straight wall and borders the Mare Nubium. It is one of a family of three well-known walled plains which have been the subject of much observation recently. During the previous year, much publicity was gained by Kozyrev[1] who reported evidence of outgassing on the central mountain chain of Alphonsus.

The crater, or walled plain, has a diameter of 70 miles. In places the walls rise to 7000 feet above the floor. Inside the crater near the center is a mountain. Near the west wall a cleft connects a number of very dark spots. Wilkins[2] reports that these dark spots vary in intensity but are generally well seen under high illumination.

The occurence of gas, and the nature of the surface materials and structure may be investigated by a roving vehicle equipped with color television, equipment for mineralogical study and gas analysis.

Mare Imbrium (Fig. IV.8)

Considerable attention has been drawn to Mare Imbrium by Prof. Harold Urey in his use of this area to illustrate the meteor impact hypothesis. Spurr[3] used the same area to illustrate the origin of lunar surface features from other causes.

Within the bounds of the Mare Imbrium are a number of suitable landing sites. In particular, an interesting site is the region comprising the Palus Putredinis and adjacent Apennine Mountains where the roving vehicle may pass from the mare to the mountain range. The northwest border of the mare is formed by the Alps Mountains, a curious jumble of blocks on the lunar surface. On the north flank of Mare Imbrium, the

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[1] Proceedings of the Lunar and Planetary Colloquium, Vol I, No. 4, January 1959 (Alter, D.)

[2] The Moon - Wilkins & Moore, Faber & Faber, Ltd., London.

[3] Geology Applied to Selenology - J. E. Spurr. Science Press, Lancester, Pa.

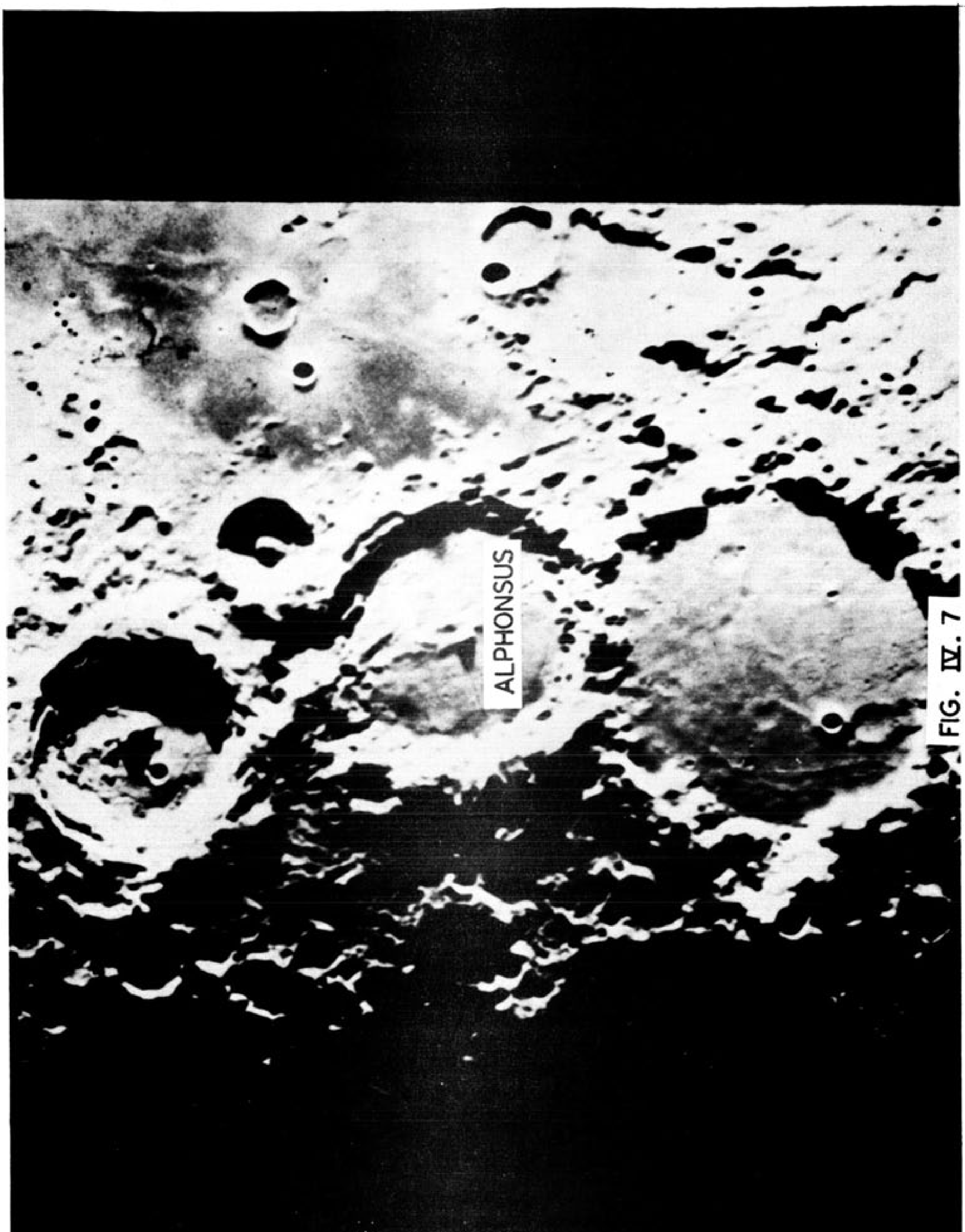

FIG. IV. 7

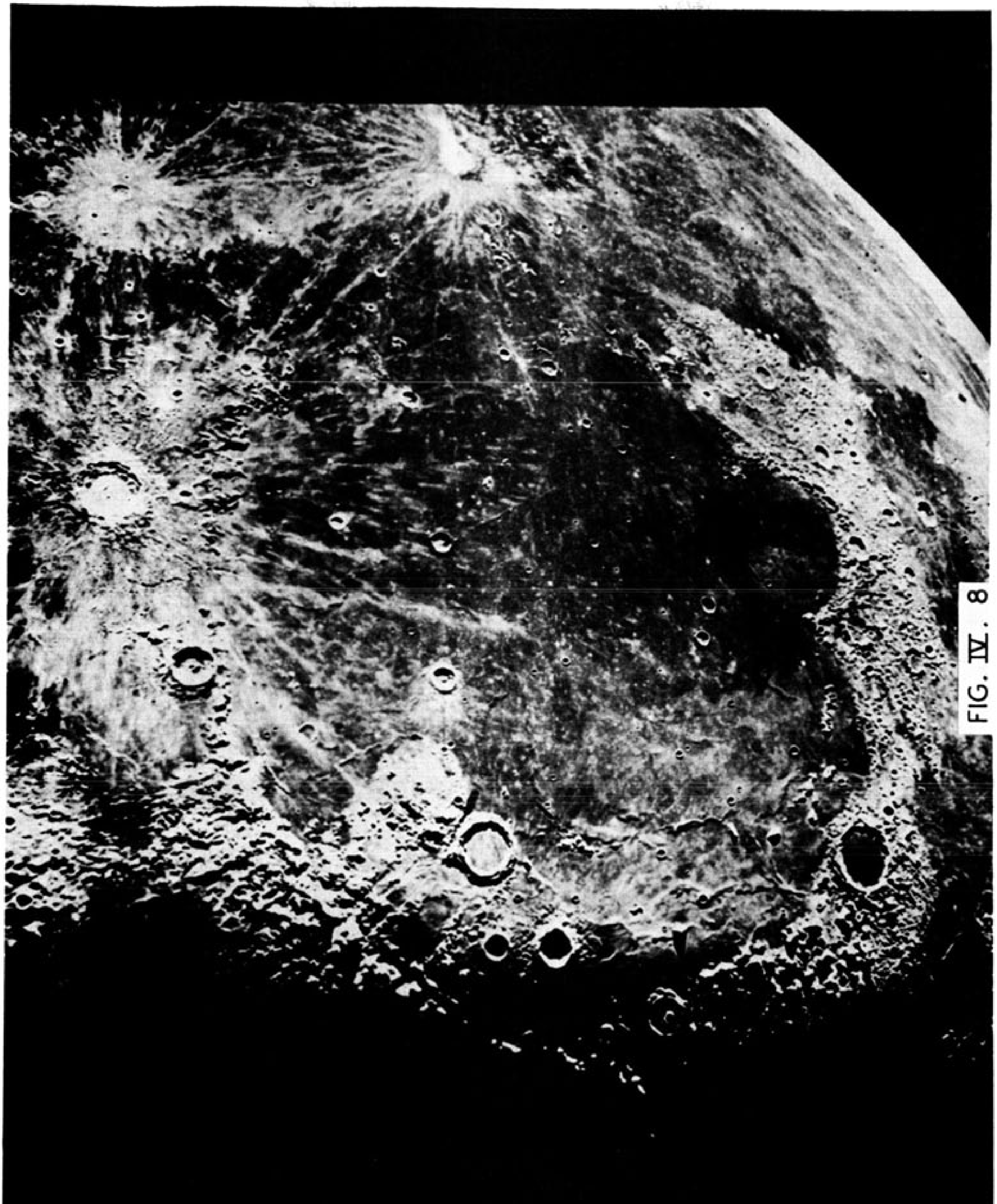

FIG. IV. 8

Teneriffe Mountains, remnants of a large crater, rise up to 8000 feet above the plain. Both areas offer a variety of topographic features. In the same area is Plato, a walled plain some 60 miles in diameter. The interior is remarkably smooth. Several small craters and spots are observable when the sun is high.[1] A landing within the walled plain, Plato, would allow a sampling of the apparently flooded crater floor. The large smooth area in Plato would likewise offer no obstacle to maneuvering the vehicle over a large area on the moon. The small craters in Plato's floor might be examined visually at close range in order to determine the origin of these post-mare disturbances.

Mare Frigoris

Tombaugh[2] has suggested a manned landing in the western flank of Mare Frigoris in the region of Aristoteles. He suggests that the ridges from Aristoteles may be made up of tunnels caused by the freezing of the lava surface, leaving the molten interior to break out of the base and form a hollow structure. Such structures, it is supposed, may be utilized as natural protection for lunar explorers.

In the same vicinity is Egede, a diamond-shaped crater with low broken walls rising at most to 400 feet. East and north of Egede is the Alpine Valley. This remarkable feature may be a graben, resulting from the thrust of the north wall of Mare Imbrium when asteroidal impact gave it birth, or perhaps it results from a ricochet at the time of impact. Wilkins[1] indicates a cleft down the center of the Alpine Valley. It is feasible to traverse this feature by the roving vehicle. An item of interest here is a televised view of the north and south walls. Tombaugh[2] suggests that since the sun's rays do not impinge on the north wall, the effects of solar radiation might be seen by comparing the two.

On the east flank of Mare Frigoris, von Braun has indicated a likely region for a manned landing near Harpalus. It is interesting to note that these two widely publicized manned

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[1] The Moon - Wilkins & Moore; Faber & Faber, Ltd., London.

[2] Presentation to the American Rocket Society, March 1958.

landing areas are in the Mare Frigoris, a region far away from the severe high temperature maximum in the equatorial regions.

IV.5 SCIENTIFIC INSTRUMENTATION

The individual instruments which are proposed for the stationary packet and the roving vehicle are discussed in the sections to follow.

IV.5.1 Television System. A television system is a versatile, multipurpose tool for the lunar scientific mission. It may be used to observe the lunar surface, to observe the drill samples, to monitor instrument operation, and to select the route of the roving vehicle.

Technology[1] in special purpose television is quite advanced. The assumption may reasonably be made that television will be used in lunar vehicle to photograph the lunar surface prior to the first SATURN flight. However, the high payload capacity of the SATURN vehicle allows the additional use of television as a multipurpose instrument.

A television system with a linear resolution of one part in 10^3 can provide high resolution color pictures of the lunar surface within perhaps a mile of the vehicle. Surface characteristics, such as fracture patterns, roughness, extrusives and small topographic features, important to a study of lunar relief and its causes, 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.

By observing the drill samples with appropriate magnification and under different light conditions, such as white light, ultraviolet light, etc., the size, shape and texture of the particles may be determined. Likewise, the presence of some organic material and minerals which fluoresce under ultraviolet may be determined.

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[1]This is Tiros; Reference Handbook for the Tiros I Meteorological Satellite System (RCA); NASA Contract No. DA-36-039-SC-78902, 20 November 1959.

With a low resolution camera mounted in the vehicle, the operation of moving parts can be observed as required (see Tables IV.2 and IV.3).

As many as six cameras (Figures IV.31 & IV.32) may be used in the stationary vehicle for various purposes. Five of these cameras require high resolution and the sixth, the one which is used as a monitor, does not. However, for economy of manufacture and for simplicity of operation, all six may have the same technical specifications.

It is expected that approximately 74 watts will be required to operate one camera, its associated circuitry, and the transmitter for television transmission. However, modulation methods must be investigated and experimental work performed before a final decision is made as to the optimum modulation scheme for the different types of television pictures. This choice will influence the final power requirement.

Digital modulation schemes would unquestionably render the highest economy in terms of power consumption. However, the demand for best possible picture quality during the descent phase and during the observations of the drill samples makes analog transmission desirable, particularly, since in this case image enhancement techniques can be applied to the electronically stored frames for later picture improvement and study.

Figure. IV.31 represents the system if analog modulation is used and Figure IV.32 represents the system if digital techniques are used.

IV.5.2 Lunar Sampling Apparatus. The lunar surface composition is unknown, but it may reasonably be assumed that the top few centimeters of this surface will be mixed with cosmic dust accumulated during the life of the moon. 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 moon. Over great lengths of time, with the absence of appreciable atmosphere, this debris may have bonded together to form a conglomerate of mixed composition. The maria were likely made up of flows of igneous material, evolving in the vacuum environment and forming a frothy texture.

To facilitate investigation of lunar composition, a device will be included to obtain, process, examine and distribute samples of lunar material for analysis.

A number of methods may be used to obtain a sample of the lunar surface. Some of these methods such as a scoop, clamshell, brush, magnetic or electrostatic devices, are limited to sampling loose materials. Since there is a possibility that the surface is solid, the method used in the preliminary design is a rotary drill. This apparatus, with its accessory equipment, obtains a sample from the surface and from selected depths. The system should operate satisfactorily with a loose or solid surface.

It is desirable that the sampler keep the surface material separate from the subsurface material, and that it 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 indigeneous lunar material. Preliminary studies indicate that a bit with a slightly concave cutting head and a sample chamber mounted atop the bit allows 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.

An automatic drilling device 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 effort will be necessary to develop a workable apparatus for lunar sampling. Successful operation of bearing surfaces under high loads in a vacuum environment may require development of special lubrication techniques. As with other mechanical apparatus used in the lunar environment, care must be taken to utilize materials and components tested and proven under vacuum conditions and to test the actual apparatus in vacuum.

With the solar power system 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 operator, monitoring the program via television. Since the sampler will operate from the auxiliary power source of the roving vehicle, power is available for a number of holes to be drilled along the vehicle path. It appears that the lifetime of the apparatus will depend more on wear than the power available.

Two models of the sampling mechanism are shown in this chapter. The first model, used in the stationary packet is shown in Figure IV.28. The second model, a modification of the first, is used in the roving vehicle. This model (Fig. IV.10) is described in some detail in the following pages. Components of both systems, as illustrated, will undergo engineering refinements before incorporation into the final vehicle design.

The estimated weight, dimensions and power requirements of the first model are given below. With optimum materials and configurations, the weight requirement should be lowered considerably.

Estimated Weight	50 lbs (drill)
Power Requirement	200 watts (drill)
Size (retracted)	Cylinder, 6" dia. x 24 to 36" (drill)

In a retracted position, the drill and case are housed in a vertically mounted cylinder located in the base of the stationary packet (Figure IV.28). In drilling, the drill stem and case are lowered, the drill resting firmly against the lunar surface. In recovery, the drill stem is retracted, and the drill bit rises into the drill case. (Figure IV.28 includes the components of the sampling system and should be referred to while reading the following description.)

The sample obtained by the drill is collected in a spring-positioned cylinder located above the bit. On recovery, the cylinder is pushed against a bearing, exposing the sample which may be spun or scraped into the conveyor (15). The sample is then moved to the sample splitter (1) where it is examined by television and magnifier with a regular light source (4) or ultraviolet lamps (3). At the same time, part of the sample drops from the sample splitter into a sample holder (5), biological experiment (10) and the sample preparation bin (7).

The sample holder consists of an expendable cup located in a cutout of the sample holder arm. This arm may move to any position for pickup, analysis or reject. When the sample holder cup is in reject position, a dumping device (24) removes the cup and a new cup is inserted (25).

The sample collector system for the roving vehicle consists of the following components:

Drill Sampler Assembly	(Fig. IV.9)
Drill Unit	(Fig. IV.10)
Sampler with Conveyor & Collector	(Fig. IV.11)
Rotary Grid	(Fig. IV.12 & IV.13)
Pulverizer	(Fig. IV.14)
Sample Holder	(Fig. IV.15)
Drill Sampler Flow Diagram	(Fig. IV-16)

The general arrangement of the drill unit components are shown in Figure IV.10, and details are shown in Figure IV.11.

The drill unit is designed with four vertical guides for lowering and raising the drill. This structure also serves as a support for the following components:

Drill Rod Lift Motor and Reel

Guide Follower

Drill Thrust-Spring Assemblies (2)

Drill Motor

Drill Bit Housing

Conveyor and Drive Motor

Sample Collector and Drive Motor

The drill is a rotary type with the drill rod supported by the guide follower at its upper end. Two thrust bearings in the guide follower transmit the thrust applied to the drill rod when drilling or to the drill rod assembly when lifting.

A downward force of 10 to 30 pounds is applied to the drill rod through the guide follower by two thrust springs which are located on the base plate. Each spring applies a

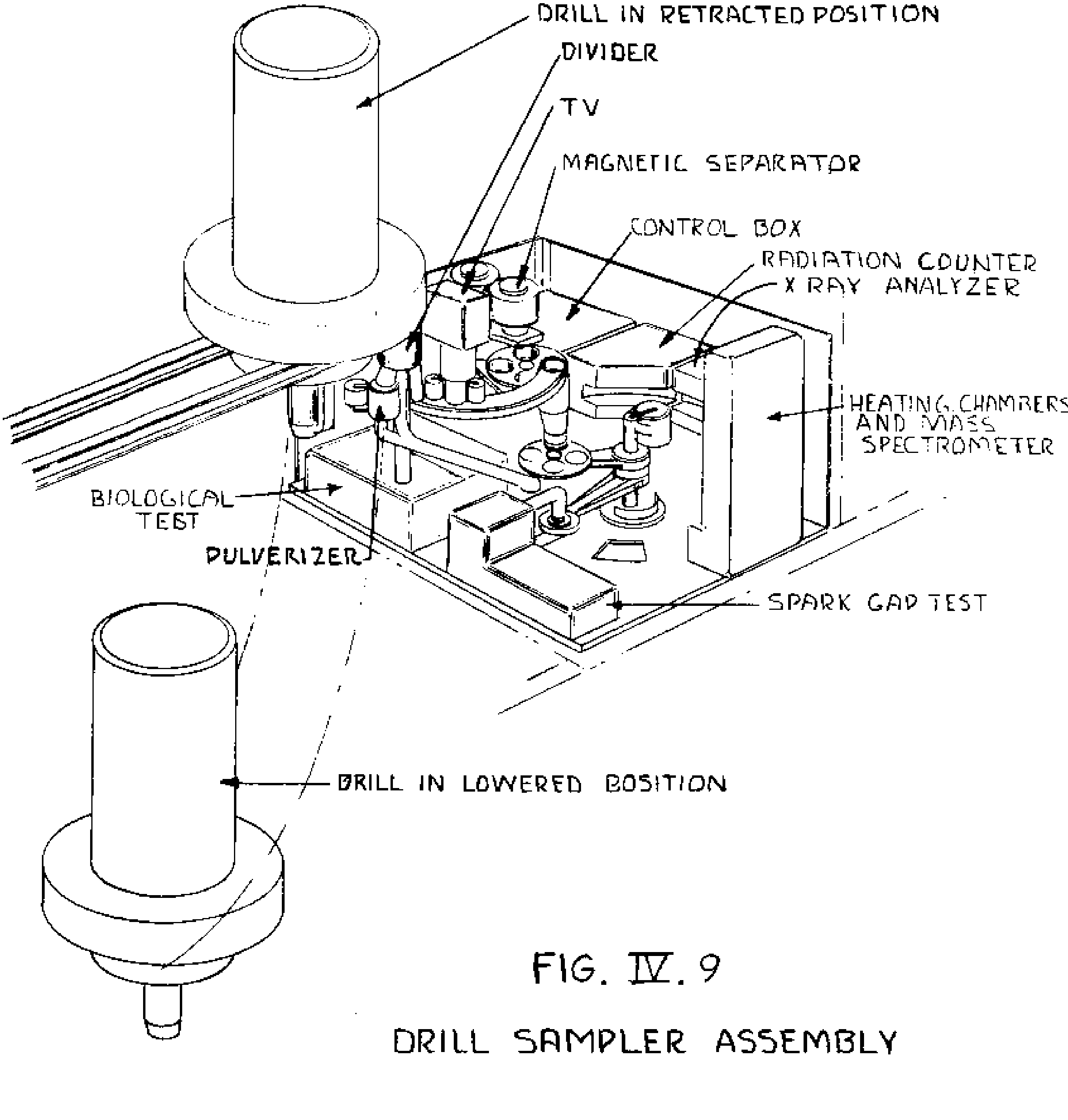


FIG. IV. 9

DRILL SAMPLER ASSEMBLY

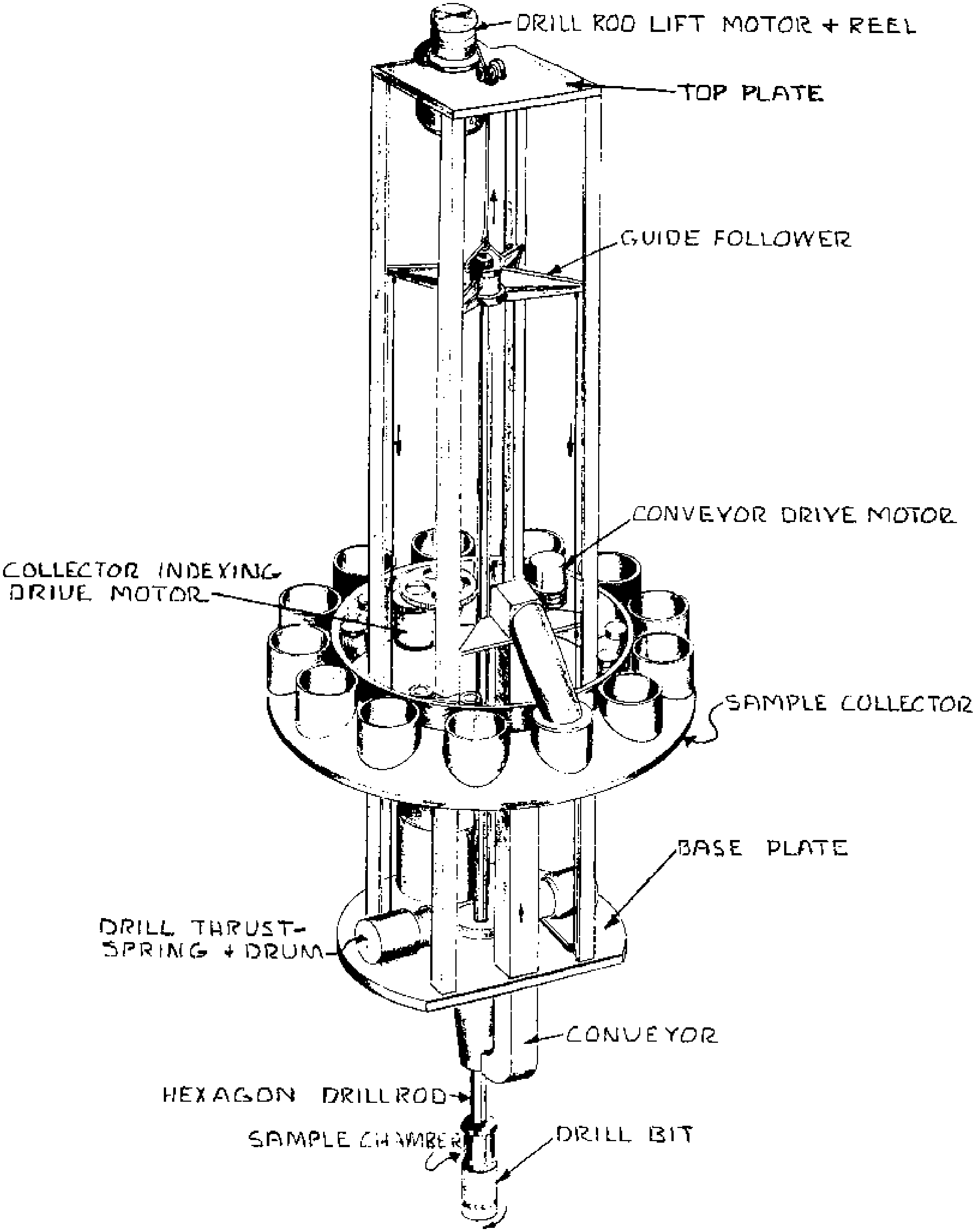


FIG.IV.10 DRILL UNIT COMPONENT ARRANGMENT

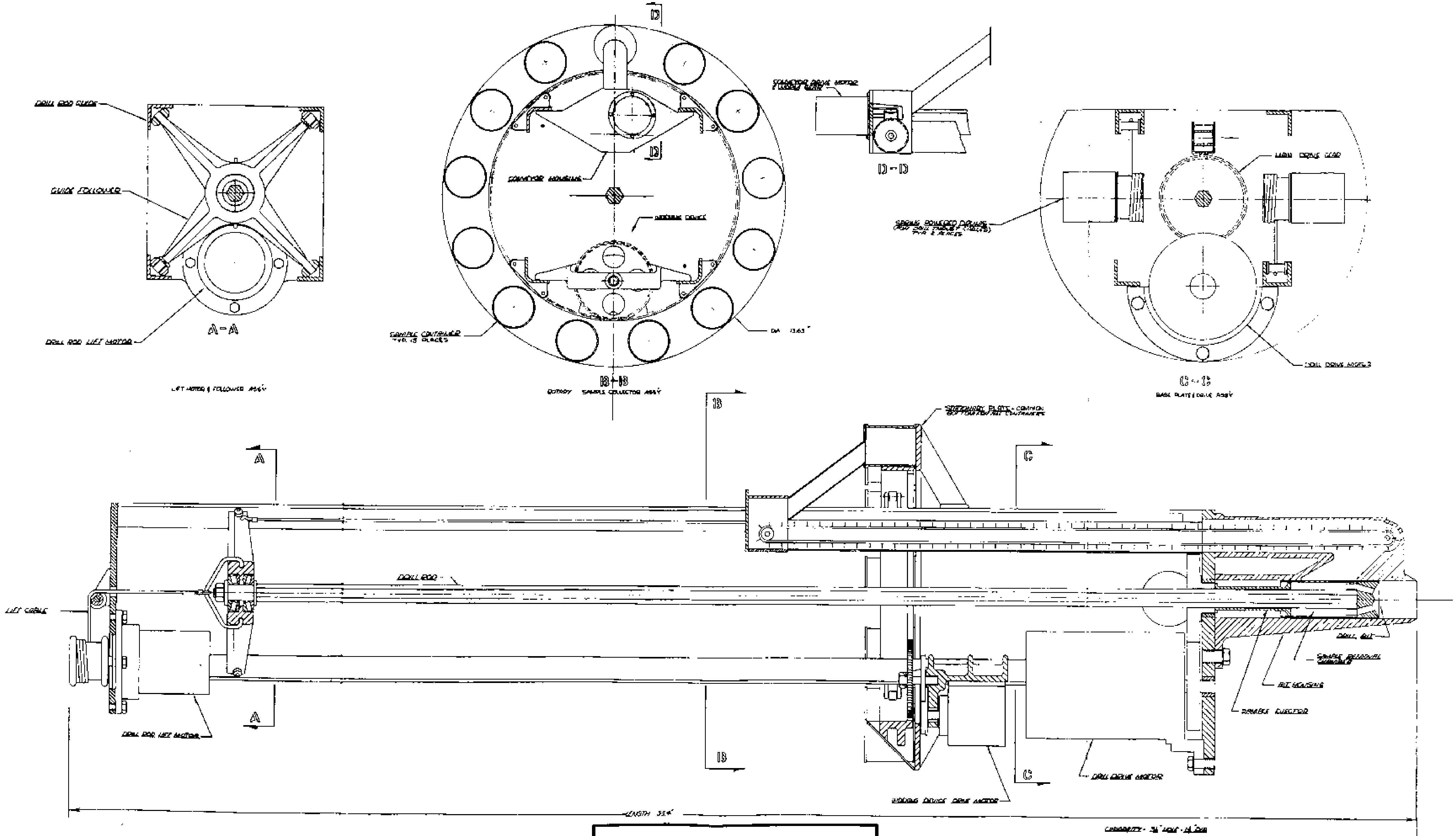


FIG. IV. 11

DRILL & SAMPLE COLLECTOR

(LUNAR ROVING VEHICLE)

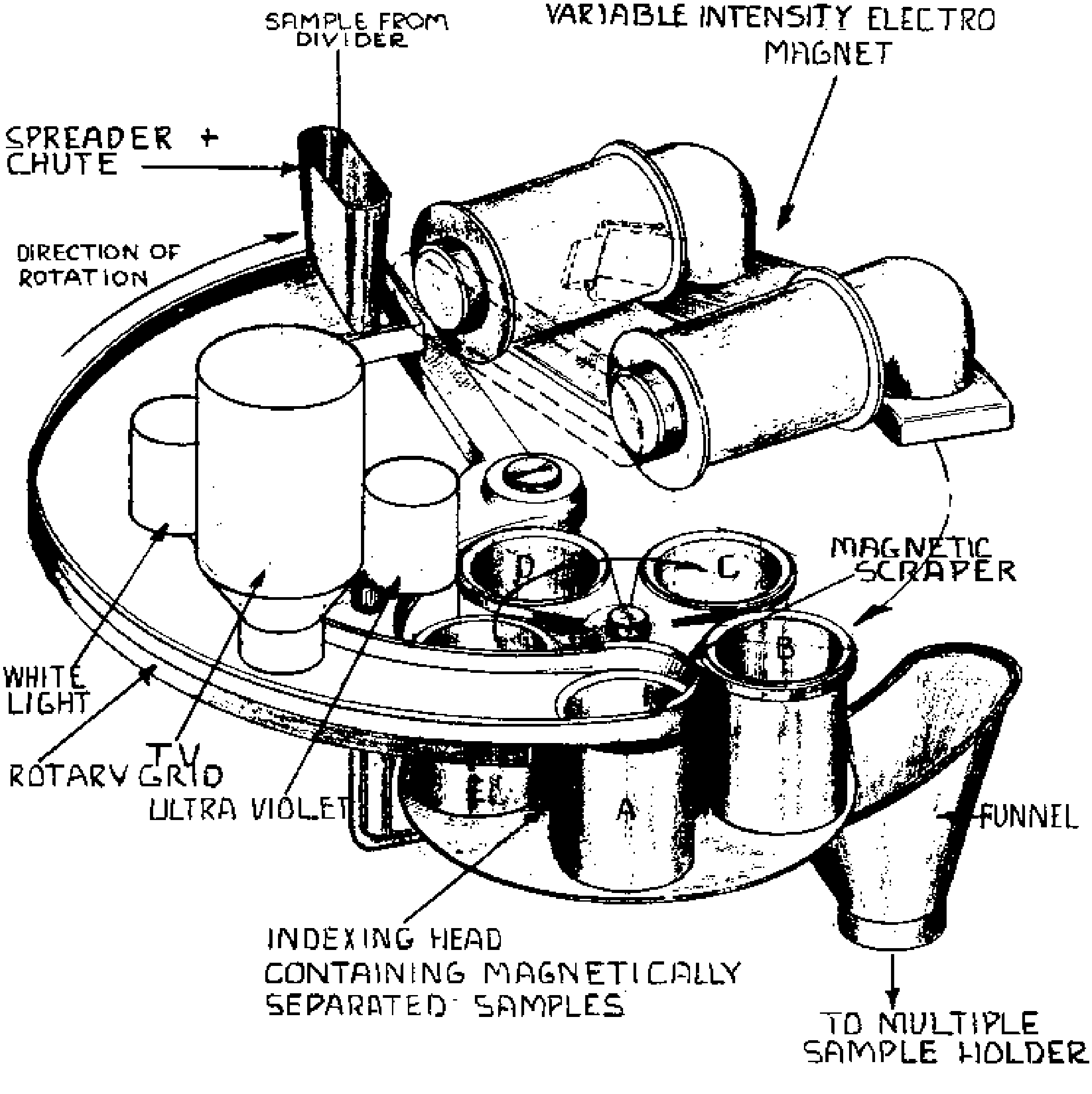


FIG.IV.12 GENERAL ARRANGMENT - VISUAL + MAGNETIC INSPECTION EQUIPMENT

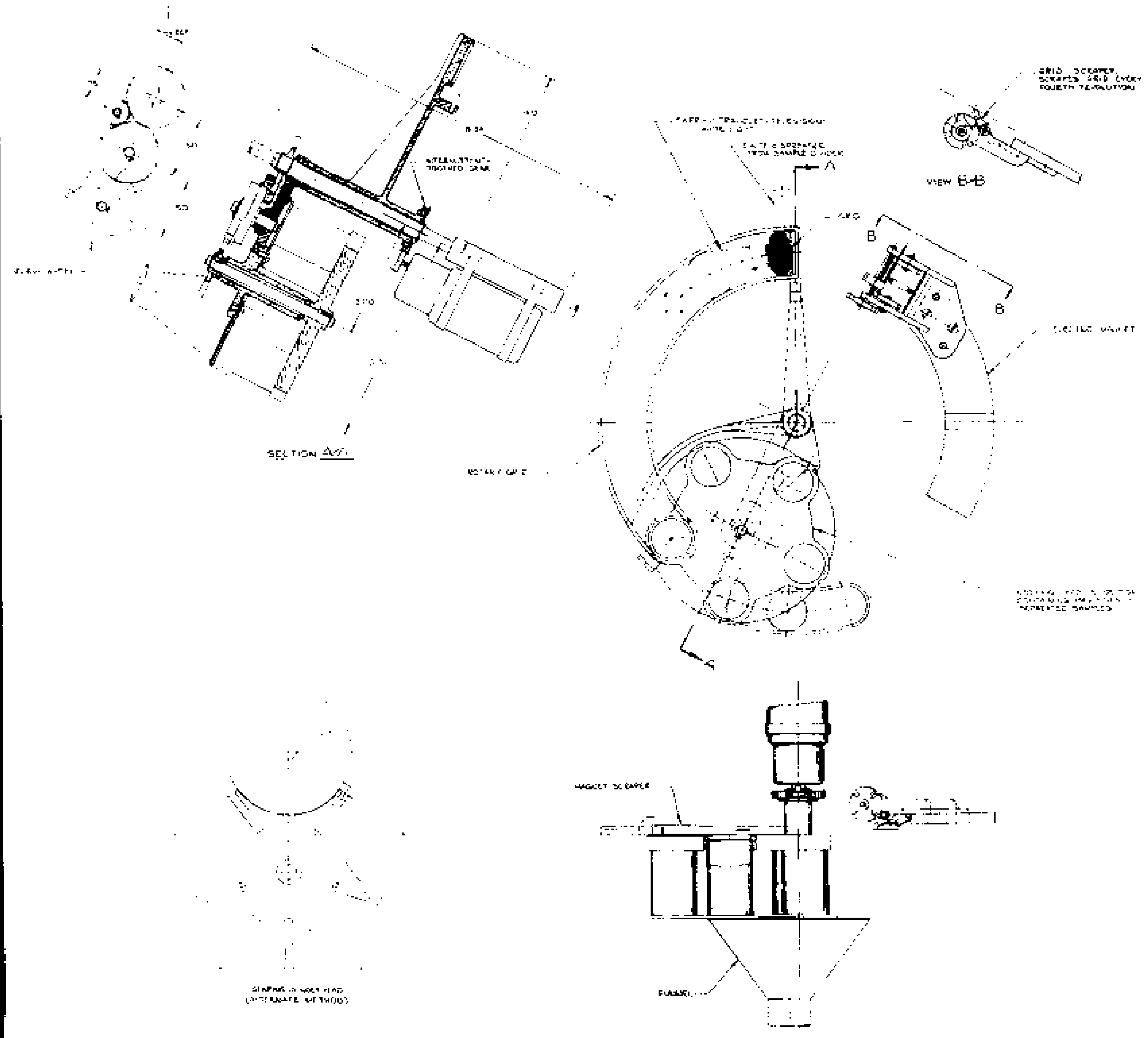
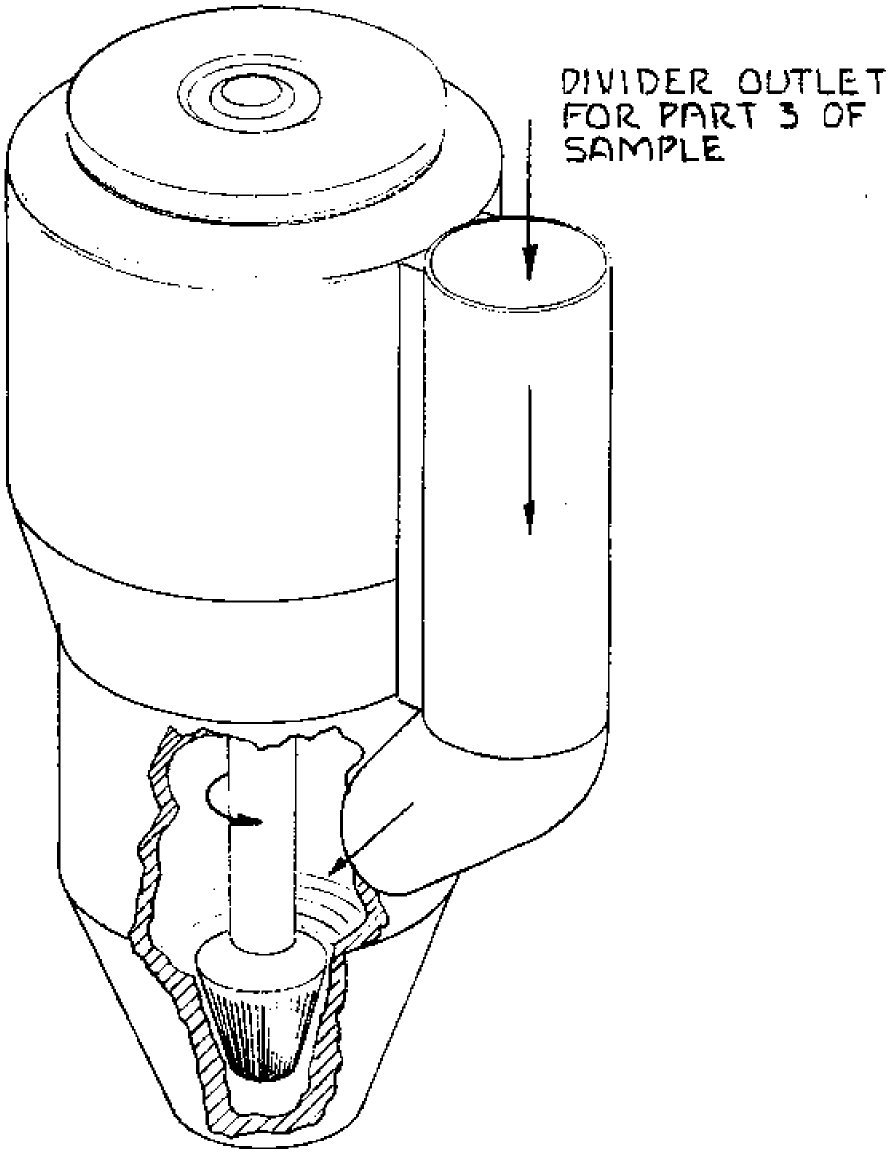


FIG. IV, 13

VISUAL & MAGNETIC INSPECTION EQUIPMENT

INSTRUMENT COMPARTMENT OF
THE LUNAR ROVING VEHICLE



SAMPLE PULVERIZER

FIG. IV. 14

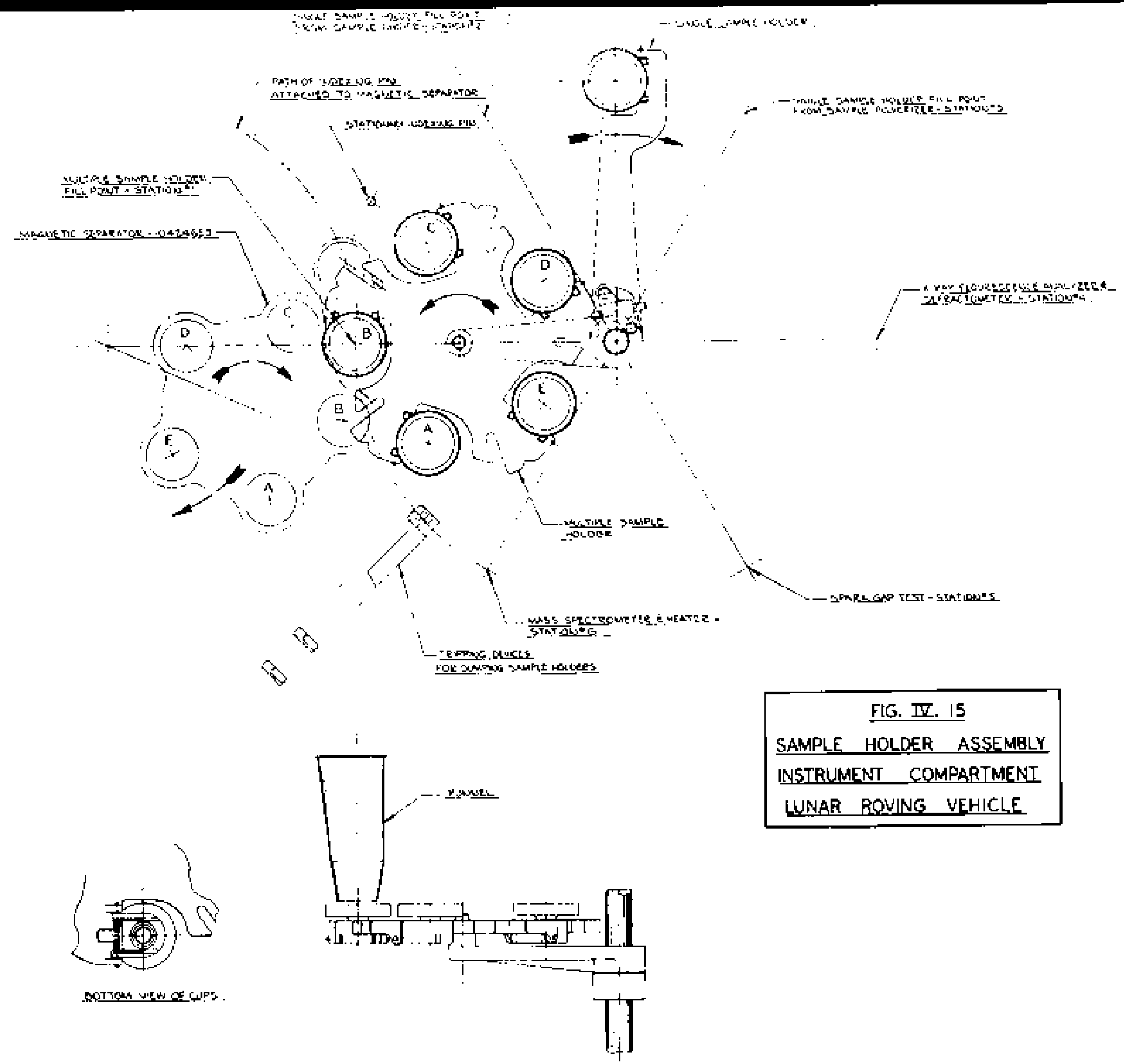


FIG. IV. 15
SAMPLE HOLDER ASSEMBLY
INSTRUMENT COMPARTMENT
LUNAR ROVING VEHICLE

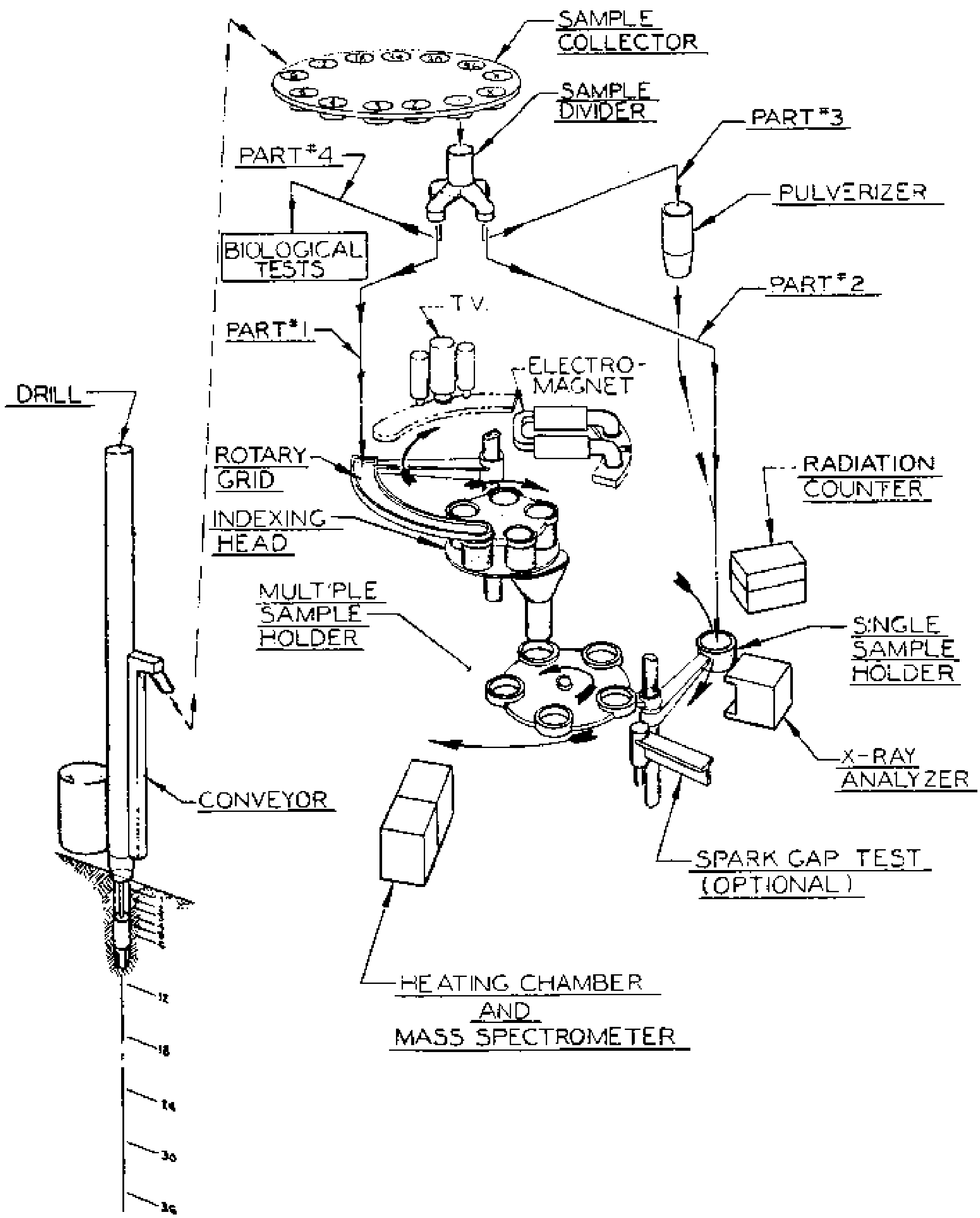


FIG. IV. 16
DRILL SAMPLER
FLOW DIAGRAM

ELECTRIC MOTORS REQUIRED FOR SAMPLER

1) Drill - Drive Motor
   Task: turn drill rod
   500 watt motor

2) Conveyor - Drive Motor
   Task: operate conveyor at approx. 1 ft per sec. $\frac{1}{2}$ lb sample
   0.34 watt equiv. load
   use 10 watt motor

3) Sample Collector - Indexing Motor
   Task: rotate collector at 4 rpm requiring 2 ft-lb, approx.
   1.135 watt equiv. power
   use 10 watt motor

4) Drill Rod - Lift Motor
   Task: lift drill rod at approx. 4 in. per sec. against 30 lb
   13.6 watt equiv. load
   use 50 watt motor to absorb gearing losses and in drill rod becomes stuck

5) Rotary Grid - Drive Motor
   Task: turn separator at 6 rpm requiring 1 ft-lb torque
   0.85 watt equiv. load
   use 10 watt motor as in case (3)

6) Multiple Sample Holder Arm - Drive Motor
   Task: turn arm at 1 rpm requiring 4 ft-lb
   0.567 watt equiv. load
   use 10 watt motor as in case (3)

7) Single Sample Holder Arm - Drive Motor
   Task: turn arm at 1 rpm requiring 4 ft-lb
   .567 watt equiv. load
   use 10 watt motor as in case (3)

8) Drilling & Sample Collecting Unit - Lifting Motor
   Task: lift unit 4.5 ft in 30 sec., wt. 20 lbs lunar
   4 watt equiv. load
   use 20 watt motor

9) Pulverizer - 25 watt motor

rotary force to a cable drum which in turn pulls the drill rod assembly down for drilling.

The drill rod is raised by the drill rod lift motor and reel which is mounted on the top plate. The drill rod is suspended by a stainless steel cable which is attached to the guide follower at one end and is wound at its other end onto the lift reel. When the drill rod is fully retracted, the guide follower turns off the lift motor switch. Simultaneously, the reel is prevented from rotating by a mechanical brake; this also prevents the drill rod from being lowered by the thrust springs. The brake is electrically released during the drilling sequence by the sequence timer (not shown) which regulates the drill unit as it performs its task.

The drilling torque is supplied by the drill motor through a gear reducer. The main gear has a hexagonal hole through which the drill rod slides as it is lowered and raised during drilling operations.

A purely mechanical method of extracting the sample from the hole must be used. The drill bit is so designed that as the particles are cut, the combined rotary and compressive action between the bit and the lunar material forces the particles through the channels to the top side of the bit. The cuttings are retained in this position by the sampler chamber.

When the proper depth has been reached for any given sample the depth sensing device in the drill rod lift assembly will cause the drill rod to be pulled up. During the last three inches of travel the sample ejector is shouldered in the bit housing; as the bit continues to rise the sample is forced into the conveyor chamber (Figure IV.11). The drill continues to rotate during ejection, and the combined forces discharge the sample into the conveyor inlet. Once the drill drive motor has been turned on, it runs until all samples have been drilled and conveyed to the collector.

The conveyor is started by the same control used to start the drill rod lift motor. This action takes place each time the drill is pulled out of the hole. The conveyor is a simple bucket-type lift system. The buckets are attached to a steel cable which is driven by an electric motor at the top pulley. The bottom side of each bucket is sloped to insure that the sample is deflected into the outlet chute as it is dumped. Samples reaching this point may either be rejected or placed into the collector.

The collector (Figure IV.11) is a disc with containers for holding the samples. The identity of each sample is maintained by a mechanical arrangement (Figure IV.10). The containers are mounted on a rotating ring gear which is indexed to position the containers beneath the conveyor outlet chute. A stationary plate serves as a common bottom for all containers. The stationary plate has a hole which serves as the sample outlet to the divider.

After the samples are in the collector, the drill unit is retracted (Figure IV.9). In the retracted position, the collector indexing device (Figures IV.9 & IV.11) positions the first container over the outlet chute to the divider. As shown in Figure IV.16, the sample enters the divider and is discharged in four equal parts. Part #1 goes to the rotary grid; part #2 goes directly to the sample holder; part #3 goes to the particle sizing machine; and part #4 goes to the biological test chamber.

As part #1 comes out of the divider, it is deposited on the rotary grid (Figure IV.12). The rotary grid holds the sample during visual inspection through television and during magnetic classification of particles with the electro-magnet.

The rotary grid is powered by an electric motor through an "intermittent-tooth-gear drive." The motor runs continuously, and the intermittent-tooth-gear allows the rotary grid to pause for one minute after each revolution for visual inspection through television.

When the sample from the divider drops onto the rotary grid, it is spread evenly over a light and dark surface etched with small squares. This surface moves into position under a magnifier and a small television camera. A small lamp is turned on, and two television frames are taken of the sample. These are transmitted back to earth. A measure of the size of the particles, as well as their texture and shape, may be obtained from this observation.

In the same manner, a shortwave and longwave ultraviolet lamp may be used to excite fluoresce in the sample. Two television frames may be taken in each case. A comparison of these pictures with those taken in ordinary light may give a general impression of the percentage of particles fluorescing in either ultraviolet range.

The rotary grid also serves as an apparatus by which a rough magnetic separation of the mineral particles may be performed. By varying the intensity of the electro-magnet, several separations could be made. After each separation, the grid rotates under the television camera. If a suitable amount of material is removed by the magnet, this will be evidenced when the televised image is compared to the original. The samples removed by the electro-magnet may be preserved individually for analysis. After each separation by the electro-magnet, the material is scraped and dropped into an indexed container. The containers have their counterparts on the multiple sample holder (Figures IV.15 and IV.16). These samples are retained in the multiple sample holder until a command is given which indexes the cups into position for analysis.

Part #1 (Figure IV.16) of each sample is processed in the same manner. Part #2 of each sample goes directly to the single sample holder. Before moving into position for analysis, the sample may be leveled and tamped by a simple mechanical arrangement.

The single sample holder is pivoted on one end to rotate about a vertical axis. The arm is driven by an electric motor through several positions by a Geneva indexing device. These positions are equally spaced around a complete circle as follows:

(1) Direct sample position (sample part #2)

(2) Ground sample position (sample part #3)

(3) X-ray analyzer

(4) Spark gap test

(5) Mass spectrometer

(6) Reject position

When operating on a programmed cycle, the sample holder arm will start at position #1, the direct sample fill point. If operated by command, the holder will start at position #2. After the sample holder passes through the testing positions, the sample is rejected and the cycle may be repeated again.

The pulverizer (Fig. IV.14) provides a crushed sample on command to position #2 as outlined previously. The pulverizer will be used only when it is thought that the original sample is not suitable for the X-ray apparatus. A programmed sequence involving the use of the pulverizer places the sample holder in position #2. From this position, the sample moves to position #3 and then to reject position.

Part #4 goes directly to the biological test chamber and is not handled by the sampler mechanism.

The size of the modified drill and sampler apparatus is indicated in the drawings. Power requirements are shown on Table IV.1. Weight estimate for the entire system is 75 pounds.

IV.5.3 Fluorescence Spectrometer and Diffractometer. Near the base of the instrument container, 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 such as those illustrated in Section IV.5.2.

For a number of years automatic spectrometers have been used in control laboratories where large numbers of samples must be analyzed for the same sets of elements, each of which is variable only over a limited range of concentration. Both North American Phillips and the Applied Research Laboratories have built automatic X-ray spectrometers. A description of devices built by both companies is aptly described by Cullity[1] as follows:

(1) Single - channel type - An instrument of this kind is manufactured by North American Phillips Company and is called the Autrometer. It uses a flat analyzing crystal in reflection and a scintillation counter as detector. Corresponding to the elements A, B, C,...to be detected are the wavelengths λ_A, λ_B, λ_C, ...of their characteristic spectral lines, and to these correspond certain diffraction angles $\theta_A, \theta_B, \theta_C$... at which these wavelengths will be diffracted by the crystal. The counter is designed to move step-wise from one predetermined

[1]Cullity, B.D.; Elements of X-ray Diffraction, Addison-Wesley Publishing Company, Reading, Massachusetts (1956)

angular position to another rather than to scan a certain angular range. The various elements are determined in sequence: The counter moves to position $2\theta_A$, remains there long enough to accurately measure the intensity of the beam from A, B, and so on. At each step the intensity of the beam from the sample is automatically compared with the intensity at the same time from a standard, and the ratio of these two intensities is printed on a paper tape. The instrument may also be adjusted so that the actual concentration of the element involved is printed on the tape. As many as 12 elements per sample may be determined.

(2) Multi-channel type - manufactured by Applied Research Laboratories and called the X-ray Quantometer. The analyzing crystal is a bent and cut LiF or NaCl crystal, used in reflection. Near the sample is a slit which acts as a virtual source of divergent radiation for the focusing crystal. Eight assemblies, each consisting of slits, analyzing crystal, and counter, are arranged in a circle about the centrally located X-ray tube; seven of these receive the same fluorescent radiation from the sample, while the eighth receives fluorescent radiation from a standard. Each of these seven assemblies forms a separate "channel" for the determination of one particular element in the sample. In channel A, for example, which is used to detect element A, the positions of the crystal and counter are preset so that only radiation of wavelength λ_A can be reflected into the counter. The components of the other analyzing channels are positioned in similar fashion, so that a separate spectral line is measured in each channel. The eighth, or control, channel monitors the output of the X-ray tube.

In this instrument, each counter delivers its pulses, not to a scaler or ratemeter, but to an integrating capacitor in which the total charge delivered by the counter in a given length of time is collected. When a sample is being analyzed, all counters are started simultaneously. When the control counter has delivered to its capacitor a predetermined charge, i.e., a predetermined total number of counts, all counters are automatically stopped. Then the integrating capacitor in each analyzing channel discharges in turn into a measuring circuit and recorder, and the total charge collected in each channel is recorded in sequence on a chart. The quantity indicated on the chart for each element is the ratio of the intensity of a given spectral line from the sample to that of a line from the standard, and the instrument can be calibrated so that the concentration of each element in the sample can be read directly from the chart recording. Because the total fluorescent energy received in each analyzing counter is related to a fixed amount of energy entering the control counter, variations in the X-ray tube output do not affect the accuracy of the results.

Although an X-ray fluorescence instrument has not been built for lunar application, a few brief studies have been made by industry concerning the probably 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" x15" in instrument case

Sampling time: 1 hour per analysis

A sketch of a fluorescence apparatus, taken from this study and modified for the sampler mechanism, is shown in Figure IV.17.

Another study[2] indicates that the operating temperature range for such equipment is between -20 to +60°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

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

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

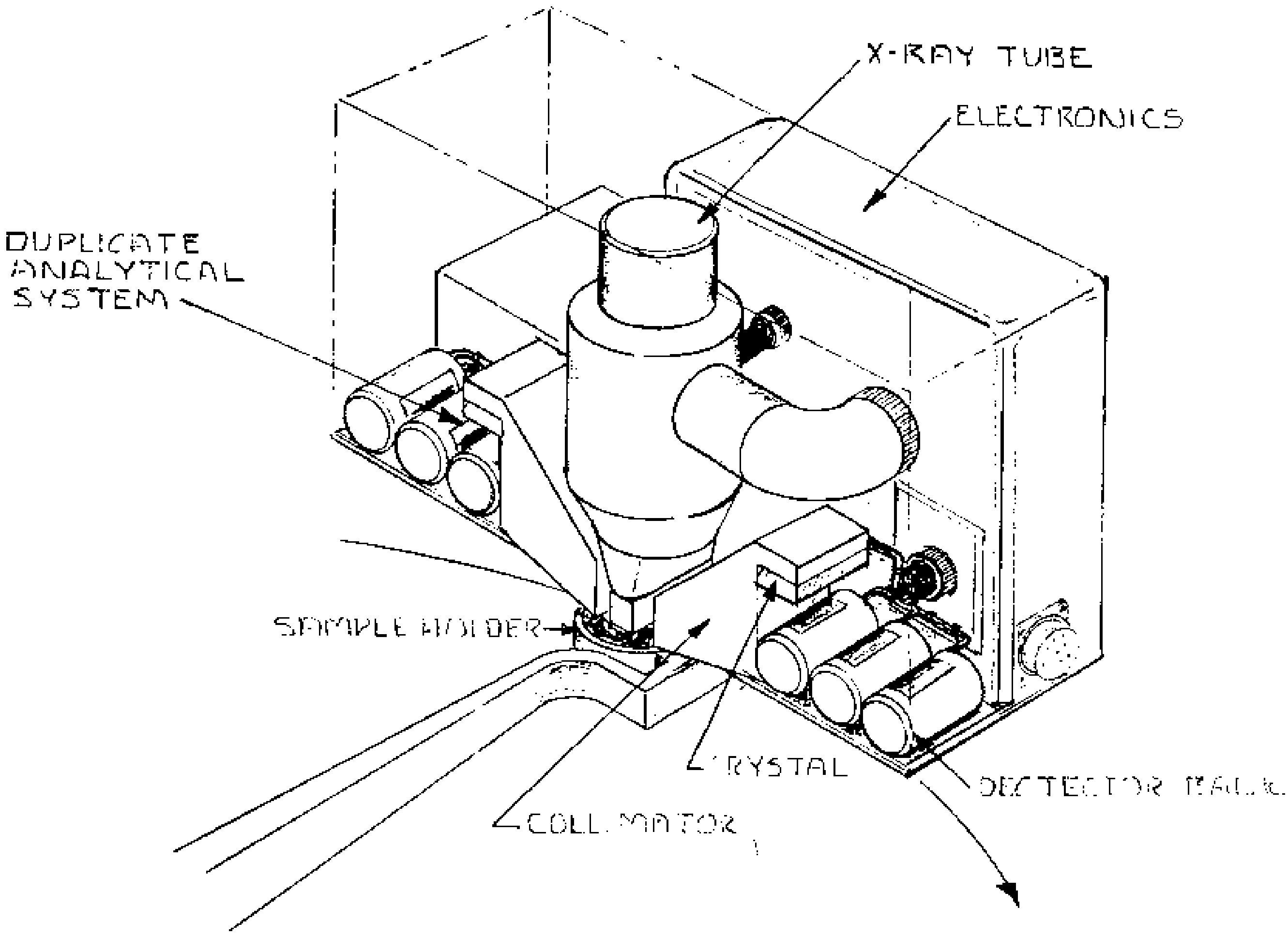

FIG. IV. 17

X-RAY SPECTROMETER

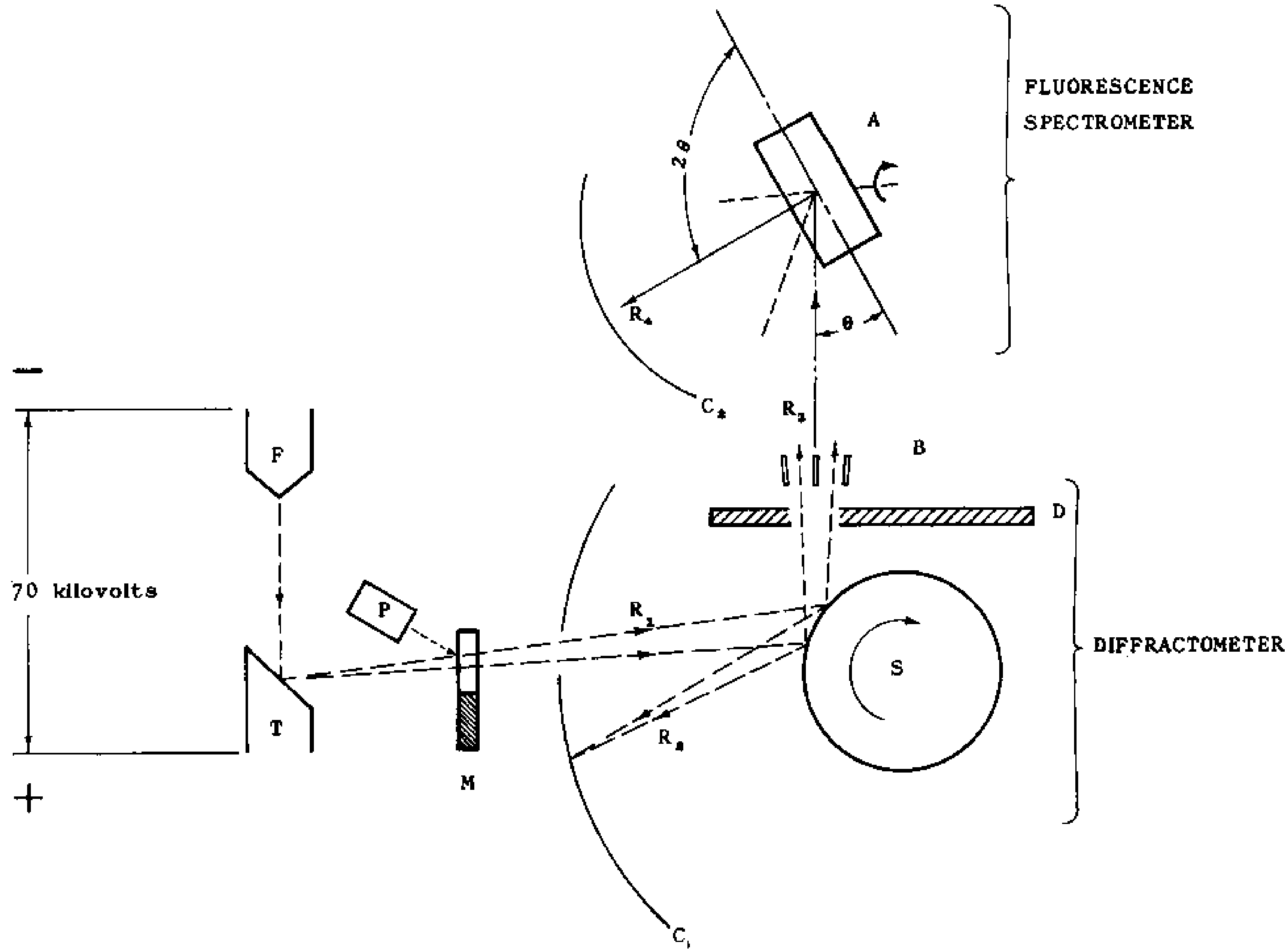

FIGURE IV.18 SCHEMATIC DIAGRAM OF ESSENTIAL FEATURES OF X-RAY DIFFRACTOMETER AND FLUORESCENCE SPECTROMETER

X-RAY DIFFRACTOMETER AND FLUORESCENCE SPECTROMETER

(Figure IV.18)

F, filament of X-ray tube; T, tungsten target of X-ray tube; R , 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 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 θ and the reflection angle 2θ are automatically coupled in the $\theta - 2\theta$ relationship; P, phototube monitor viewing zinc sulfide screen on M; D, lead shield; B, Soller slit.

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 is to identify chemical elements in the sample by observing their fluorescence spectra induced by proper wavelengths incident radiation. Both functions, diffraction and fluorescence spectrometry, are performed by a single instrument represented in principle by Figure IV.18. 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.18 operates as a diffractometer during about one-half of 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_3, 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 make 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 12400/7000 = 0.18 Å. Since the tungsten K-alpha radiation is at 0.209 Å and the tungsten L-alpha$_1$ radiation is at 1.47 Å, both wavelengths will be present with good intensity.

The rotating analyzing crystal A of Figure IV.18 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.18) 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 Å to measure any aluminum K-alpha fluorescence radiation (8.3 Å).

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-hr of energy is 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 legend for Figure IV.18). 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.[1]

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

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

required in the instrumentation shown by Figure IV.18. Thus, the rotation frequency of M and crystal analyzer A would have to be speeded to several rotations per second.

Alternate methods of analysis should be looked into where the sample preparation problem may be lessened. Some of these methods have been mentioned in some detail by interested workers in the past for the problem of analyzing the lunar crystal material.[1] Helvey[2] has recommended an "automatic" spectrograph using spark or oxyhydrogen excitation source with integrating capacitance network for several channels, each covering an element which may be of interest.

<u>IV.5.4 Radioactivity as a Means of Lunar Survey</u>. Nuclear radiation promises to be quite useful in surveying the lunar landscape. The use of nuclear radiation may be categorized as active or passive, in a manner analogous to radar search. In the active survey, the material to be examined is bombarded with a stream of particles and the reflected radiation or the radiation arising from nuclear reactions is examined, whereas in the passive survey, the nuclear radiation arising from radioactive materials naturally present is examined.

The passive survey requires less equipment since it is not necessary to provide a source of radiation, and is perhaps best for an initial examination of the lunar surface. It would, of course, be quite feasible to employ small natural radiation sources in an active survey, but these are not subject to easy control and would present a shielding problem if passive surveys were desired.

The lunar surface is expected to contain only elements which are available on the earth. A study of meteorites has shown them to be only slightly radioactive, so that little radiation will arise from cosmic ray bombardment of the material of which the lunar surface is comprised. The meteoric data do not indicate directly, however, the extent to which the gamma background will be enhanced by short-lived excited states created by cosmic ray bombardment.

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[1]Private communication from C. N. Scully, North American Aviation (NAA Proposal MD 59-254).

[2]Private communication from T. C. Helvey, Radiation, Inc.

A first set of measurements will perhaps contain surprises. The proposed instrumentation should be versatile enough to cover a wide range of the unexpected.

Alpha Particle Measurements

The detection of natural alpha radiation from lunar material yields useful information about its nature.[1] Only the heavier isotopes, with mass numbers greater than 208, are expected to contribute to the alpha decay arising from the lunar surface The energies of the alpha particles emerging from the heavy nuclei will almost all have energies from 4 to 10 mev.

The accurate spectroscopic measurement of the energy distribution of the alpha radiation would in principle permit the identification of the emitting nuclei. However, a number of factors make such detailed measurements extremely difficult, if not impossible. For example, suppose an experiment is performed with a pure sample of some alpha emitter. The alpha particles are emitted isotropically. If the sample is of finite thickness, the alphas will lose energy in the source, with the amount of energy loss being determined by the initial energy of the alphas and the path length within the source. Since the energy lost by alphas in various materials is well known, it is possible, in principle, to calculate the distribution in energy and direction of alpha particles emitted from a known alpha emitter in a particular configuration. However, good spectroscopic techniques would require careful preparation of sources small enough to minimize the self-absorption problem. This procedure naturally reduces the emission intensity from the source and increases noise and background problems. Accordingly, it is preferable in a beginning experiment to perform a simple alpha measurement, avoiding many of the problems of a detailed spectroscopic analysis.

The background for the alpha measurement is expected to consist of cosmic rays of all energies, and the beta and gamma radiation from the lunar surface. One of the best means of eliminating unwanted background is by means of a "dE/dx" counter. Because the alpha particles of interest are stopped in less than 100 microns of scintillator material, whereas the background is much more penetrating and subject to lower

[1] Private communication from Patrick Hurley, Massachusetts Institute of Technology.

ionization loss, the use of a scintillator layer just sufficient to stop the alphas is quite effective in eliminating effects of the more penetrating background radiation. An arrangement of a possible detection system is given in Figure IV.19.

The alphas are expected to produce flashes in the scintillator. These flashes are detected and amplified by the photomultiplier tube and are sent through a discriminating circuit which removes pulses which are too large or too small to correspond to the alphas of interest. In general, the gamma and beta pulses can be eliminated because of their smallness. Very energetic cosmic rays will not be counted because of their low specific ionization. The Cerenkov effects in the glass are expected to be small. Primary cosmic ray alphas which just manage to penetrate the vehicle walls and reach the scintillator will be counted and contribute to the background. Protons will not be counted at all. Heavier nuclei will be counted if they have the right speed to produce a pulse of the proper height, but heavier nuclei are expected to be rare.

The use of RCA 6810-A multipliers with silver activated ZnS scintillators seems appropriate for alpha counting. The principle emission spectrum of the ZnS (Ag) is at 4500 Å and the maximum response of the 6810-A is at 4400 Å.

A possible modification of Figure IV.19 is shown in Figure IV.20. This new arrangement permits a larger information rate but suffers from the fact that the pulses are smaller and more varied in size. The arrangement of Figure IV.21 is effective in reducing the problem of internal noise in the photomultiplier tubes if we wish to take full advantage of the technique of Figure IV.20. Figure IV.22 represents a working arrangement which is useful in eliminating cosmic background. New developments in semiconductor detectors may make it possible to eliminate the photomultiplier tube and to replace the ZnS (Ag) with a thin layer of silicon covered with a gold deposit.[1]

Beta and Gamma Detection

Beta and gamma radiation are emitted from many of the unstable heavier isotopes as well as from lighter isotopes such as ${}_{6}C^{14}$ and ${}_{19}K^{40}$. Because of the greater penetrating power

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[1]Fowler, Session N, 1959 Winter Meeting of the American Physical Society. Detector noise may be a problem, however.

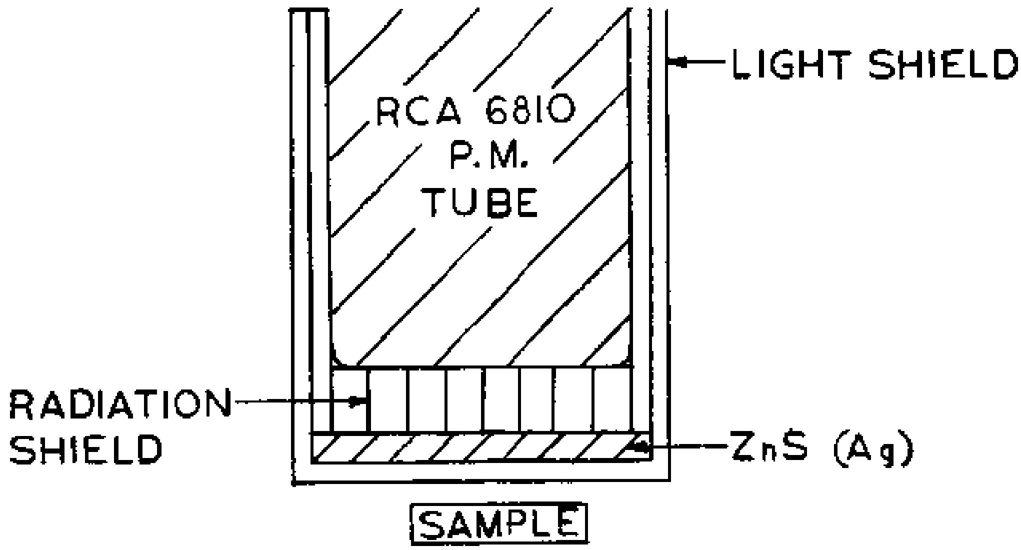

FIG IV. 19 A SIMPLE ARRANGEMENT FOR ALPHA DETECTION

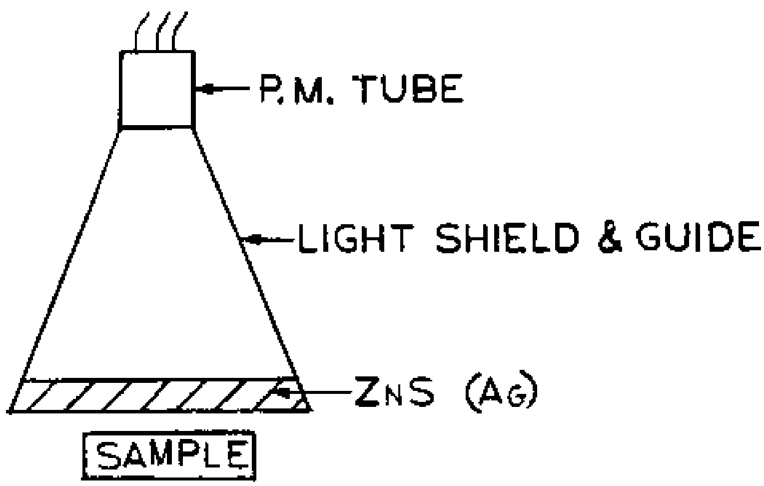

FIG. IV. 20 A MODIFIED ARRANGEMENT TO INCREASE THE RATE OF ALPHA COUNT

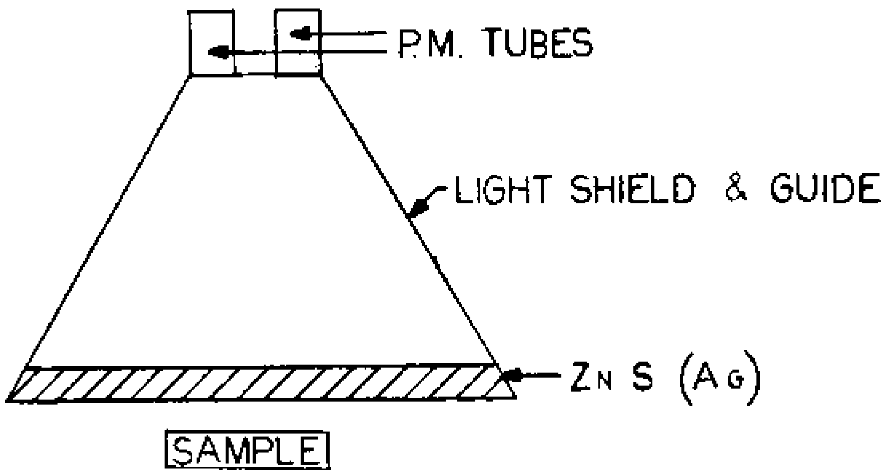

FIG.IV.21 AN ARAANGEMENT USING TWO PHOTOMULTIPLIERS IN COINCIDENCE TO REDUCE BACKGROUND COUNT.

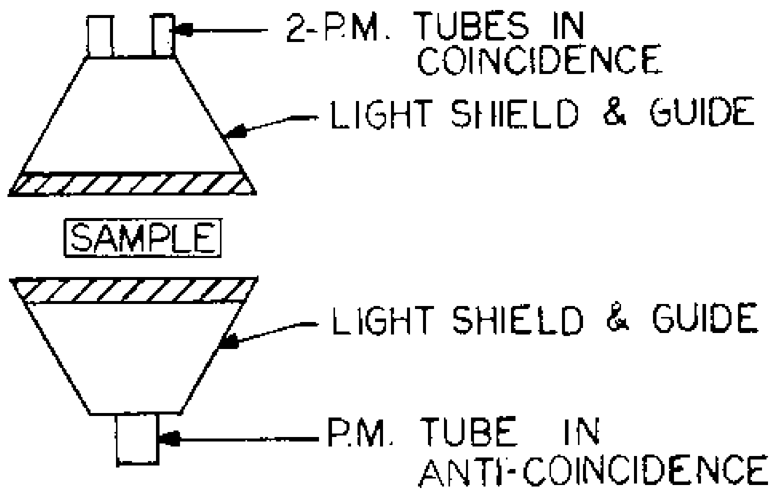

FIG.IV.22 A POSSIBLE WORKING ARRANGEMENT FOR LUNAR ALPHA COUNTING of these radiations, they convey information from farther below the surface than do alphas. For example, a one-mev beta particle can penetrate 0.4 gm/cm^2 of Al, and a one-mev gamma can penetrate roughly 16 gm/cm^2. Because of their greater penetrating power, gamma rays are more difficult to detect. A Geiger tube, for instance may count 97% of the betas, and only 0.5% of the gammas passing through it.

Because of the great differences in the range of the alpha, beta, and gamma radiations from nuclei, conventional field survey instruments usually do an efficient job of differentiating between them by means of shields covering a Geiger tube. With all sliding shields removed, the counter is sensitive to alphas, betas, and gammas; with the first thin aluminum shield in place, the counter is sensitive to betas and gammas; and with an additional heavier iron shield roughly 1/8" thick in place, the counter is sensitive to gammas only. This crude method, though the instrumentation can be quite simple and rugged, has the serious shortoming of being unable to distinguish one particle from another or to provide a good energy spectrum. Further, the alpha and beta counts are available only after a subtraction process. This subtraction introduces additional statistical errors and may fail to show one type of radiation if it is present in amounts small compared to the others.

It would be desirable to have the same discrimination afforded by the (dE/dx) alpha counter. The case of electrons and photons, is less fortunate, however, because for each (dE/dx) range chosen for electron or photon detection, there is at least one photon, alpha, or heavier nucleus with energy which will produce the same ionization in the crystal. Suppose, for example, a crystal is chosen with the right thickness to stop one-mev betas. A one-mev proton will also be counted, for although it will traverse only a small part of the crystal, it will deposit one-mev of energy and produce roughly the same ionization. There are some crystals which show a certain discrimination between particles, but they serve only as partial solutions to the problem.

An examination of the (dE/dx) curves shows that it is possible to choose an energetic proton which will deposit the same amount of energy within the crystal as a one-mev proton. The softer component of the cosmic radiation can be eliminated by means of a shield in front of the scintillator as shown in Figure IV.19. If there is a continuous energy distribution, some charged particles will penetrate the shield with an energy appropriate for producing background interference.

In addition to the background problems mentioned above, there is the additional background resulting from inelastic collisions of cosmic rays with nuclei and from X-rays associated with the removal of electrons from atoms by cosmic ray ionization.

Although the magnitude of the background problem for beta and gamma measurements is uncertain, these measurements should perhaps be attempted. The general arrangement of the beta and gamma counters would be photomultiplier tubes with appropriate scintillators attached. Transistorized pulse sorters (multi-channel analyzers) would be used to reduce background.

IV.5.5 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. Direct measurements of temperatures at various distances below the surface are desirable.

Measurements can be performed by boring holes into the ground to various depths into which temperature sensors are placed. This experiment should be conducted by the stationary packet as well as the roving vehicle.

To obtain an estimate of the depths to which temperature variation may be expected on the moon, an analysis of heat conductivity and the nature of the lunar surface material by Wesselink[1] will be used.

The following notations and definitions are adopted:

T = absolute temperature

x = depth below the lunar surface

t = time

c = specific heat per unit volume

[1] A. V. Wesselink; Bull. Astron. Inst. Netherlands, 10:351 (1948).

k = coefficient of heat conduction

P = period of temperature variation, which equals the synodic month

λ = $2 \left(\pi \frac{k}{c} - P\right)^{\frac{1}{2}}$, wavelength of harmonic heat wave

ξ = $\frac{x}{\lambda}$

F = flow of heat across a square centimeter per min.

A = amount of heat absorbed by a square centimeter of the surface per minute

σ = Stefan-Boltzmann constant

The lunar surface is supposed to radiate as a black body. The change in heat content of an internal element of volume equals the net amount of heat energy conducted through its walls. This statement of the law of conservation of energy is expressed by the following two equations:

$$c \frac{\partial T}{\partial t} = k \frac{\partial^2 T}{\partial x^2} \tag{1}$$

$$\sigma T_o^4 = A + F_o \tag{2}$$

The index 0 indicates surface values (x = 0). At any depth we have further:

$$F = k \frac{\partial T}{\partial x} \tag{3}$$

With given boundary conditions, these three equations determine a unique solution.

It is convenient to introduce ξ instead of x . The following equations are equivalent to (1) and (3).

$$\frac{\partial T}{\partial t} = (4 \pi P)^{-1} \frac{\partial^2 T}{\partial \xi^2} \tag{4}$$

$$F = (4 \pi P)^{\frac{1}{2}} (k c)^{\frac{1}{2}} \frac{\partial T}{\partial \xi} \tag{5}$$

Equation (5) for the surface combined with Equation (2) gives:

$$\sigma T_o^4 = A + (4 \pi P)^{-\frac{1}{2}} (k c)^{\frac{1}{2}} \left(\frac{\partial T}{\partial \xi}\right)_o \tag{6}$$

Suppose the temperature distribution at $t = 0$, $T(\xi, 0)$, and the temperature variation at the surface $T(0,t)$, to be given. It is required to find the periodic solution. Starting with a provisional initial distribution, Eq. (4) can be integrated numerically. When the integration is completed, $\left(\frac{\partial T}{\partial \xi}\right)_o$ can be found as a function of time. $F_o = \sigma T_o^4$ can be plotted against $\left(\frac{\partial T}{\partial \xi}\right)_o$. These two quantities proved to be reasonably proportional to each other, as is to be expected from Eq. (6), if $A = 0$. From the factor of proportionality, $(k c)^{\frac{1}{2}} = 0.0080$ was found.

The specific heats per gram of minerals have an average of about 0.20. The average density of the moon is 3.33. The density of the surface is likely to be smaller than this average value. Wesselink has assumed a surface density $\rho = 2.0$. The specific heat per cm is then $c = 2(0.20) = 0.40$. Combining this value of c with the result $(kc)^{\frac{1}{2}} = 0.0080$, it is found that $k = 16 \times 10^{-6}$ cal cm^{-1}min^{-1}deg^{-1}.

The wavelength of a harmonic heat wave with a period equal to the synodic month is then found to be $\lambda = 14.5$ cm. This result shows how little the variations in temperature penetrate below the surface. The computed value of k is very low. It seems that the assumption of the density value is high. A more realistic assumption would be $\rho = .125$ and $c = 0.25$. Using the result $(kc)^{\frac{1}{2}} = .008$ obtained from measurements, the value obtained for k is 2.7×10^{-3} cal cm^{-1}min^{-1}deg^{-1}, and for λ is 240 cm. It can be seen that the uncertainties in the knowledge of the surface of the moon are greater than a factor of 10.

From this analysis, it is suggested that six temperature measurements be made in each drill hole. They should be made as follows: one surface measurement and five measurements at 2 cm, 6 cm, 17 cm, 50 cm, and 150 cm. Measurements should be taken once per hour except during the day-night transition, where more frequent samples are desired. On the roving vehicle, the measurements may be continued for the transmitter lifetime after drilling the last sample hole.

The sensor, or sensors, for the surface measurements and for the subsurface measurements at 2 cm, 6 cm, 17 cm depths should have a range from +130°C to -130°C. The sensors for the subsurface thermal probe at 50 cm should have a range of about ±50°C, and the probe at 150 cm should have a range of ±10°C.

In each location, the sensor will have to be protected from direct radiation. For the surface measurements the protection will consist of a hemispherical shield with the following characteristics: the internal surface will be black with respect to infrared radiation, and the external surface will reflect 7% of the solar radiation. The sensor, located in the center of the hemisphere, will be placed against the moon's surface.

Drilled holes exposed to sky radiation, as may be the case in a roving vehicle, should be covered by a shield which possesses the following characteristics: the reflectivity of the internal surface should be as high as possible; the reflectivity of the external surface should also be high. A good approximation to such a surface would be a shield which is covered internally with gold or silver and externally by flame sprayed aluminum oxide (Rokide A).

In the case of the stationary packet, where the holes will be located underneath the vehicle, the shield would have to possess surfaces which are highly reflective in the infrared region.

Subsurface thermal probes will be equipped with bell-shaped shields fitted around the thermocouples to eliminate the necessity of shielding the entire wall of the hole.

The subsurface thermal probe for the stationary vehicle is shown in retracted position in the instrumentation assembly drawing (see Figure IV.28). On the roving vehicle, the probe (not shown) is incorporated in the drill sampler assembly.

The power requirement and weight of the temperature sensors is negligible. The drive motor for probe extension will require 10 watts, and the weight of the apparatus will be less than 3 pounds.

IV.5.6 Lunar Gravimeter. A desirable experiment using a modified "Earth Tide" gravimeter may be performed on the moon to obtain some measure of the moon's elasticity.

An "Earth Tide" gravimeter as used on the earth's surface may vary in weight from 45 to 250 pounds. The LeCoste Laboratory model is mounted on a 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.

According to a manufacturer,[1] the weight of a lunar tidal gravimeter based on the "zero length" spring principle is about 30 pounds, excluding electronics and thermal control. In such an instrument, a mass is attached to one end of a beam which is pivoted at its other end. The beam can move in a vertical plane. It is held in a nearly horizontal position by a spring attached to the mass and to a point one beam length directly above the pivot. The spring is wound so that its elongation is equal to the distance between the points where it is attached. If one defines the initial length as the actual physical length minus the elongation, this type of spring has zero initial length. The period of the spring can be given any value by appropriate choice of the spring material. The device can also be used as a long period vertical seismograph.

If the lunar gravimeter can be made to operate at 60°C ± 0.1° C during the daylight period and at 0° C ±0.1° C during the night period without damage from thermal hysteresis effects, a considerable advantage will be gained in weight and power. Research is presently underway on this problem.

During the time the sampler mechanism is being used in the stationary packet, the gravimeter will remain clamped to avoid damage. In the roving vehicle, no provision is made for a tidal gravimeter. For purposes other than accurate tidal observation a lightweight torsion apparatus, thermally protected, might be utilized. Such a device need not weigh over a few pounds; it would have a negligible power requirement.

Depending upon the type apparatus used, the seismometer and the gravimeter may perhaps be combined into one instrument package. The structural supports, clamps, springs and measuring

[1]Private communication from Mr. LeCoste, LeCoste & Romberg Co.

components evolving from the gravimeter research and test may be used for both instruments.

The power requirement for the gravimeter, including electronics but excluding thermal protection, will not exceed 6 watts. Signal output will be between 0 and 6 volts DC. Since the moon's period of rotation is slow, sampling need not be at a high rate.

To summarize, it appears feasible to make a gravimeter with a weight of approximately 50 pounds including mounting and thermal protection, that will perform satisfactorily in the lunar payload. This relatively low weight, of course, would be possible only after a rigorous research program in materials and thermal control.

IV.5.7 Seismic Apparatus. A seismometer will be placed aboard pre-SATURN lunar landing vehicles to detect the presence of moonquakes. Seismometers will be built, hopefully, to withstand a "modified hard landing" and still be able to operate on the moon. Later experiments, such as those aboard the SATURN vehicles, could involve the placing of several seismometers on the moon's surface. If the moon has sufficient internal activity, the experiment will allow an investigation of the deep subsurface character of this semi-planet. With instruments capable of operating for very long times, perhaps the impacts of larger meteorites on the lunar surface could be detected. Likewise, an active seismic system placed aboard the landing vehicle may be used to study the character of the subsurface in the vicinity of the landing.

The moon has been regarded by most selenologists 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 within the moon is not of sufficient magnitude to cause a melting or plastic flow of sizable portions of the moon's interior.

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.

Two independent efforts are presently being funded by NASA in regard to seismic apparatus for pre-SATURN vehicles. The seismic systems being developed are passive and, at the present state of study, have the following characteristics:

Lamont Geological Observatory Instrument:

Type - three component instrument

Response - 10 cycles to 0.05 cycles per second

Operating temperature - thermal control - can be made to operate at gravimeter range

Size - 8" x 5"

Orientation - self orienting

Cal Tech Instrument:

Type - single component instrument

Response - 1 second period

Operating Temperature - -55°C to +55°C

Output - 1 microvolt maximum across 1000 ohms

Orientation - operates in any orientation

Telemetry bandwidth - 5 cycles per second

Weight - 10 lbs (including batteries)

Size - 6" diameter - 6 to 8" length

In the SATURN stationary packet, a passive seismic system will be included. Instruments resulting from the above mentioned development work will probably have been tried previously in the lunar environment at the time they are launched by a SATURN vehicle.

A small seismic instrument might also be fitted on the end of the subsurface thermal probe which is lowered into a drilled hole. With a proper shield on the top of the instrument, advantage may be taken of this natural thermal controlled compartment. This instrument, if used, should have a separate communication circuit from the vehicle mounted apparatus.

An active seismic system measuring the seismic waves created by artificial sources should be placed on the roving vehicle to investigate the near subsurface conditions. Such an apparatus has already been considered by one company.[1] On the roving vehicle, a series of microphones (geophones) may be unreeled as the vehicle moves. These may be rewound and used repeatedly during a traverse. Periodically, small explosive charges could be thrown or dropped from the vehicle, artificially generating seismic waves for each measurement.

For a study of subsurface conditions in the sampling area, a four-geophone system with amplifiers and power supply 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". Measurements to depths of 200 to 400 feet could be made with two or three geophone stations placed by the roving vehicle over 300 to 500 feet. These geophones would weigh a total of 3 to 4 pounds.

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

The source of seismic energy 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 designed to record moon tremors over long periods of time, after being used as an

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[1] Personal communication from Southwestern Industrial Electronics, Houston, Texas.

active system.

For protection against shock and vibration of sensitive seismic components during the flight phase, any of several methods have been suggested. These are: freezing in low melting point medium; volatile solvent enclosure; mechanical clamping; mechanical positioning.

Thermal protection for the seismic apparatus is desirable. In the SATURN stationary packet, the passive instrument would be mounted in the thermally controlled compartment with the gravimeter. The roving vehicle will make use of its active apparatus during the day, when no protection will be required.

IV.5.8 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. Of course, 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. It has been observed (Soviet Sputniks) that about an order of magnitude decrease per day in residual gas pressure occurs in operating space vehicles.

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 for 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.

As an auxiliary instrument to determine atmosphere particle density and monitor the mass-ion spectrometer, a Redhead gauge[1] 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.23 illustrates schematically the essential features of the proposed 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 square array of small charge-accumulation plates Q ... 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 resulting in a pulse in triode which finally yields an amplified pulse that can be measured and counted. All switching and voltages is done electronically.

The pertinent data for the mass-ion spectrometer are:

Weight: 8 pounds (not including power supply)

Volume: about 45 cubic inches (not including power supply)

Power: about 20 watts during operation

Sampling frequency: on command

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[1] N.W.Spencer, University of Michigan, is also considering this gauge for particle density measurements in space vehicles.

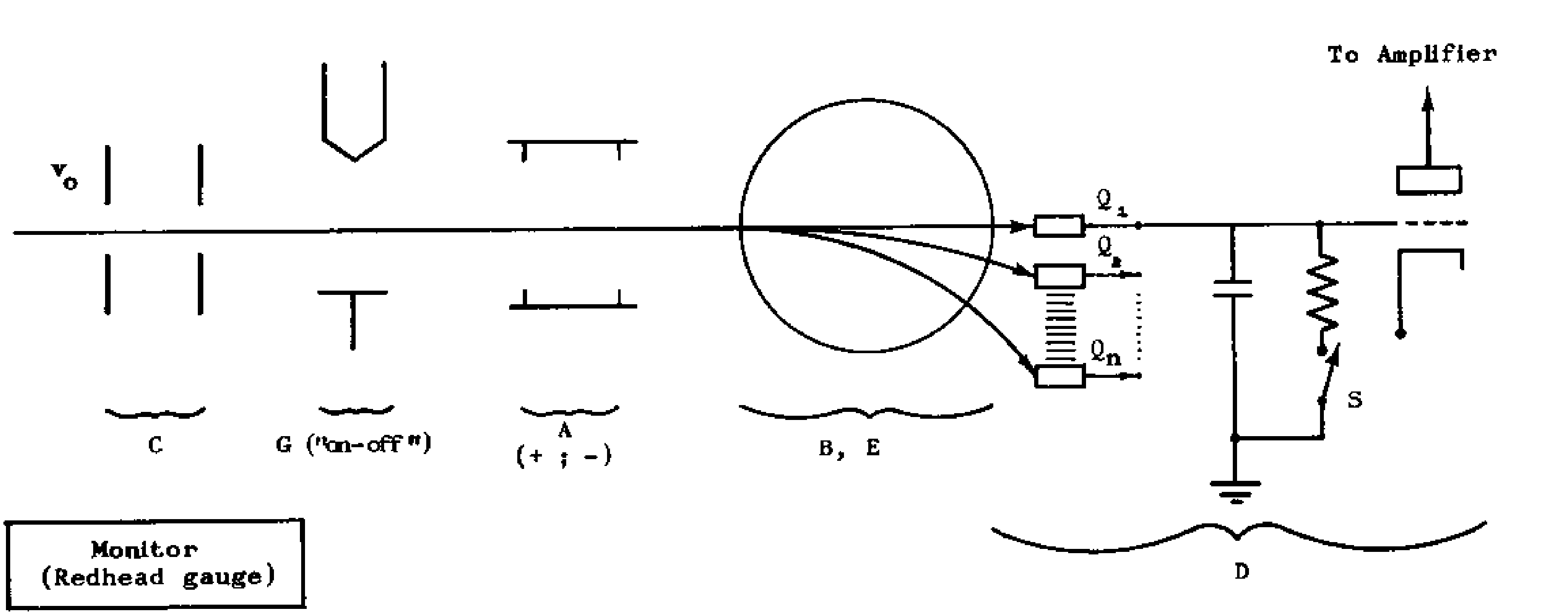

FIGURE IV.23 PROPOSED MASS-ION SPECTROMETER WITH CHARGE INTEGRATION FOR EXTREMELY LOW INTENSITIES

The pertinent data for the Redhead ion gauge monitor are:[1]

Weight: about 8 pounds (including all electronics)

Range: 10^{-8} to less than 10^{-11}mm Hg

Volume: about 20 cubic inches

Power: less than 3 watts on 10% duty cycle

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

An additional measurement, which may be performed by a mass spectrometer, is an analysis of the volatiles present in a lunar sample.[2] Since a sufficient vapor phase sample must be presented to the apparatus, the sample to be analyzed must be sealed in a small chamber connected to the instrument while being heated. This may be accomplished by providing a small divided chamber fitted with soft gold seals, having a heating element set in the upper portion. When the sample has passed the X-ray analyzer, the holder is placed in position between the two halves of the chamber. The chamber is clamped on the sample holder while the heating element "boils" a small portion of the sample. After analysis, the chamber opens so that the sample holder may move to the next position.

IV.5.9 Plasma Experiment. The plasma wind emerging from the sun, if existing, will be studied in interplanetary space by a number of experiments. Professor Rossi and his associates at the Massachusetts Institute of Technology have suggested a plasma probe to observe this phenomenon. The Jet Propulsion Laboratory is also preparing an instrument for such purposes.

The interaction of this wind with the moon raises many interesting questions. The observation has been made that the details of this interaction may determine how much atmosphere

[1]Private communication to A. H. Weber, St. Louis University, from NRC Equipment Corporation, Newton, Mass (Aug 59).

[2]The importance of similar measurements were emphasized in a personal communication from Harold C. Urey.

the moon may have.[1]

The use of the same instruments on the moon which were used in space near the moon will allow comparisons to be made, and the interactions to be studied. Particular interest will center around the periods when the instrument is on the limb of the moon as seen from the sun.

MIT Experiment

The purpose of the plasma probe experiment, in the form developed at MIT, is to measure directly the plasma densities and motions in the solar system as functions of position and time. 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.24.

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, grid 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.

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[1] J. A. Herring and A. L. Licht - "The Solar Wind" (First International Space Science Symposium, Nice, France, 11-15 Jan 1960).

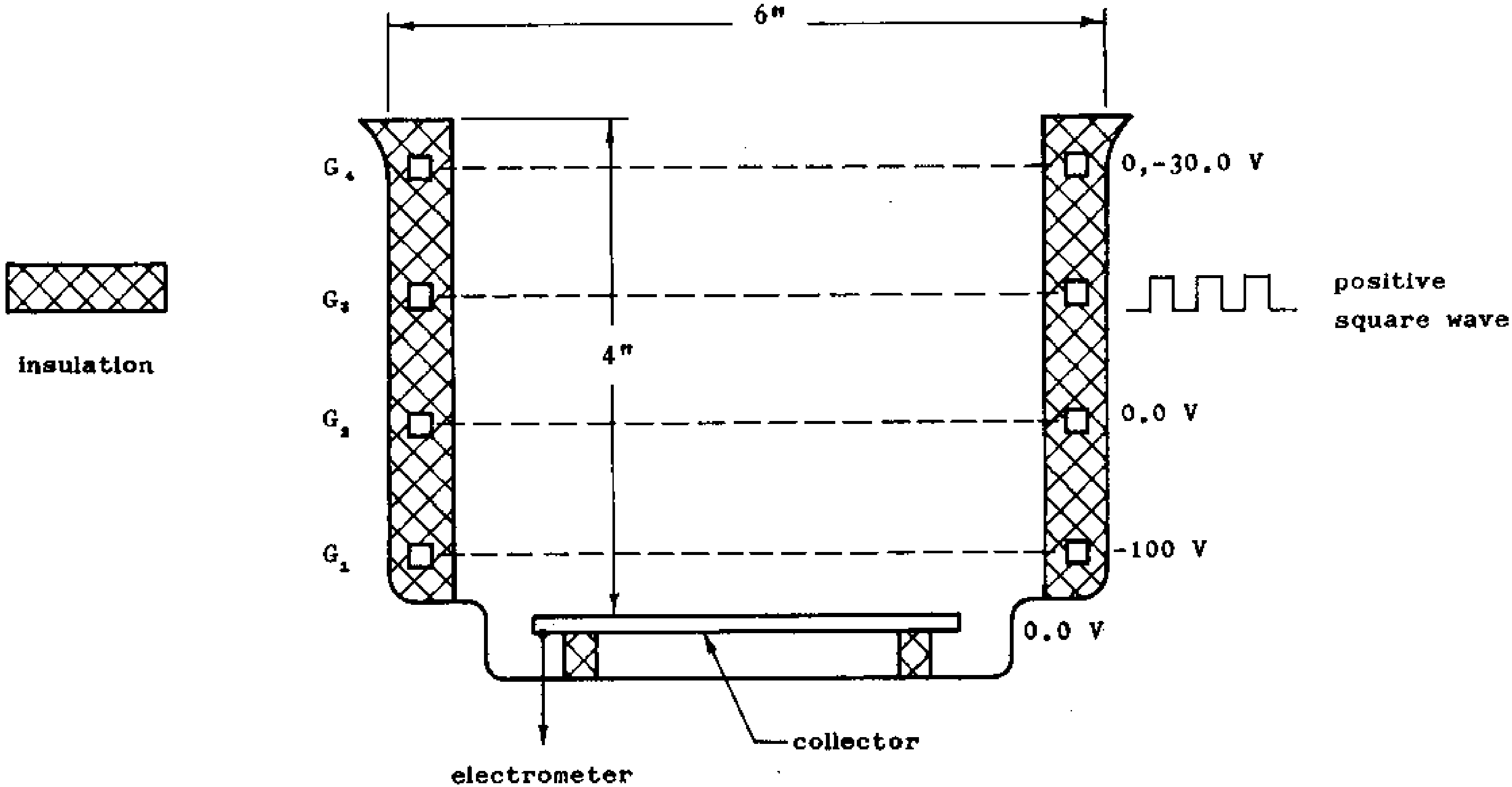

FIGURE IV.24 DIAGRAM OF THE PLASMA PROBE

The power required for the present instrument is about 1.5 watts. The weight of one system, exclusive of the power source 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 volts DC at 200 ohms.

JPL Experiment[1]

This JPL instrument for investigating the interplanetary plasma can be understood with the aid of Figure IV.25. Charged particles which enter the instrument are deflected by an electric field which is approximatly transverse to the particle velocity. Those particles with a particular charge sign and within a certain range of energy and angle of incidence will be deflected onto the charge collector. Particles which enter the instrument with the wrong charge sign, energy, or angle of incidence will hit the analyzer walls and thus will not be recorded. The charge and energy distribution of the particles entering the device can be determined by varying the sign and magnitude of the deflecting voltage with time.

Six of these instruments might be carried on the stationary packet or roving vehicle to obtain data on the direction of plasma flow. The totwl weight for six analyzers, their electronics, and the supporting structure is estimated at 9 pounds, a considerable portion of which is attributed to the capacitive modulators. The power requirement for six instruments is about one watt.

IV.5.10 Magnetometers. The moon's magnetic field, if it is measurable, is likely to be very weak.[2] Therefore, a sensitive, lightweight instrument will be necessary. Even the influence of the stationary packet or roving vehicle which carries the 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

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[1] Private communication from M. Neugebauer, JPL.

[2] Colginov, Eroshenko, Zhuzgov, Pushkov, Tyurmina - First International Space Science Symposium, Nice, France, January 1960. M. Neugebauer, Phys. Rev. Letters 4, 6 (1960).

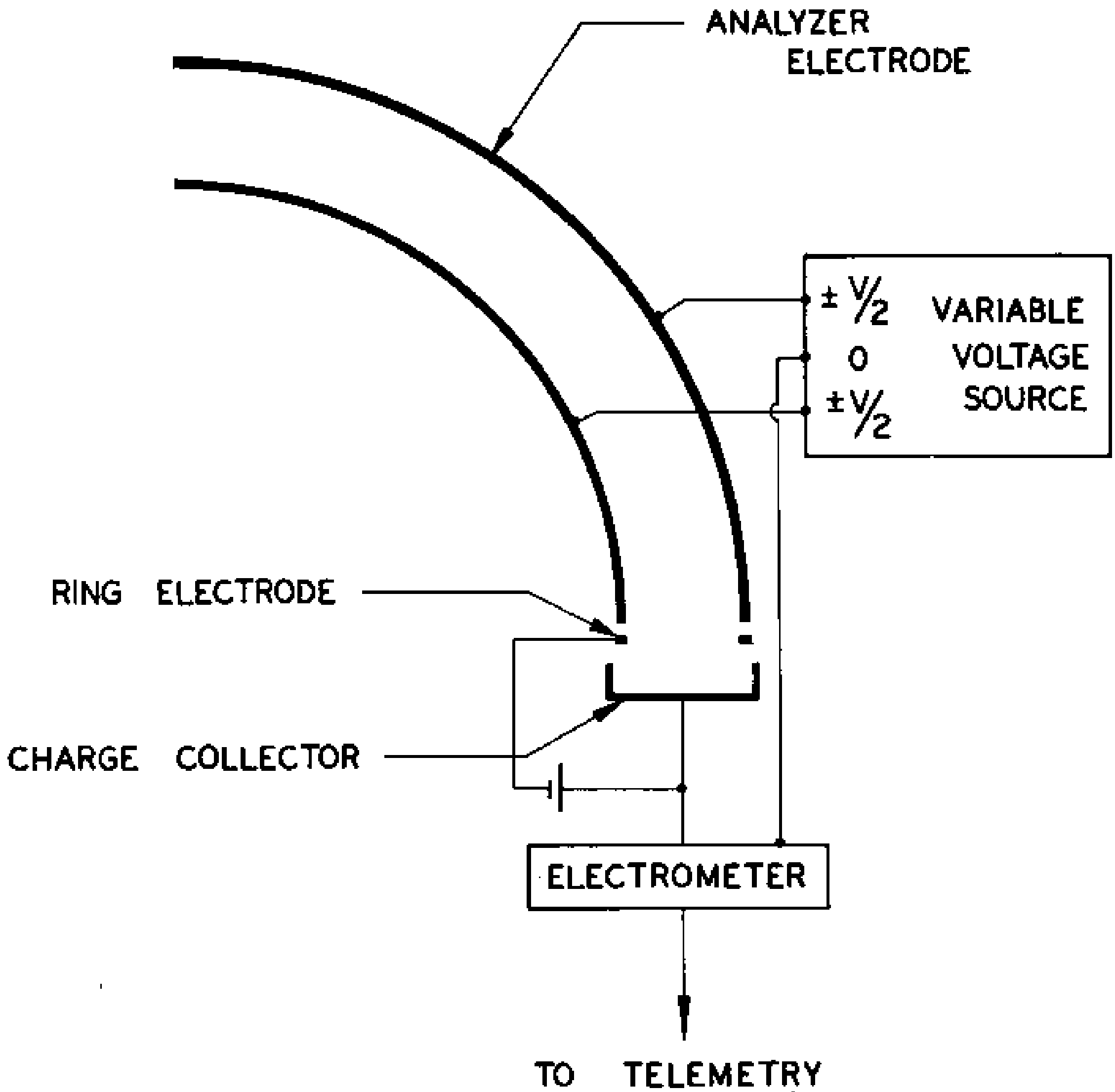

FIG. IV.25

SOLAR CORPUSCULAR RADIATION SPACE PROBE INSTRUMENTATION 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 oersted). The high operating frequency of 2.8 mc per oersted (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:[1]

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

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

[1]Texas Instrument Company Proposal to Nasa, 24 March 1959.

IV.5.11 Measurement of the Electric Field on and Near the Moon[1]. 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. They are mechanically rotated 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 may range from 10 ohms to 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 is 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 potentials 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 to measure this potential difference, 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

[1]Cornell Aeronautical Laboratory, Inc. proposals to ABMA (1958).

are used, the packages can be kept relatively small, about 12 cubic inches, with a weight of about 8 ounces, and one watt of power.

IV.5.12 Biological Experiment. An instrument to detect micro-organisms on other planets, particularly Mars, is being developed by Dr. Wolf Vishniac[1] of Brookhaven National Laboratory and Yale University.

In this instrument, Martian dust is blown into the apparatus through a small pipe when a valve is operned to an evacuated chamber. This dust is carried by the gas stream into a number of vials containing nutrient solutions. One of the vials will contain pure water. The growth of organisms in the media is detected through changes in pH and in turbidity of the solution.

The change in pH and turbidity due to biological cause appears graphically as a curve that differs in form from the curve which portrays chemical causes. It is further expected that the amount of dust added to the medium would be too small for appreciable chemical reactions. The pH meter proposed is of relatively standard design. A pair of small electrodes are placed in each vial.

The power requirement of the pH meter is insignificant. The turbidities of the media are measured by passing a light beam through the individual media, and measuring the absorption of light. The light source can be a small bulb, similar to a flashlight bulb. One simple photocell would probably be required for each vial. These could be read successively.

The basic detection technique for the lunar case would be essentially the same as that for the Martian case. The mode of innoculation of the media must be different, and different media might be chosen. Pure water again would probably be one choice. The others would be dilute aqueous solutions.

Innoculation might be made through the following tentative scheme. A series of small samples obtained by the drill from various depths would be dropped in a container. The container lid would be closed and sealed by some appropriate mechanical arrangement. Pure water would then be allowed to enter the chamber. Enough water would be added to insure a liquid as

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

a vapor phase. A simple piping and valving system would innoculate each vial of medium with water which had filtered through the lunar sample.

Other possible uses for portions of the water which has passed through the lunar sample may be considered. Presumably any soluble chemicals in the sample would be in the solution, as well as any micro-organisms. Various chemical tests could be run with portions of the sample in addition to the biological experiment. In particular, chromotographic methods of detecting organic compounds might be considered.

The problem of on-board control experiments may be raised. It is possible that there would be merit in having a second chamber identical to that into which the lunar sample is introduced. No sample would be placed in the second, but its lid would be opened, closed, etc., in the same manner as the other container. The liquid released into it would be transmitted to a medium appropriate for growth of earth organisms. Thus, any appreciable contamination remaining after sterilization of the package would be detected experimentally.

For the lunar application, an observation of the turbidity and pH of the media in the vials might be made once per hour. The signal available for telemetry in both cases could be an electrical potential. During the growth of large populations of bacteria, the turbidity might make a change in optical density from 0 to perhaps 0.1, and the pH might change one-half pH unit. A 10 cm^3 volume of each medium would be an appropriate volume for the tests proposed. The pressure in the vial is not at all critical, so long as liquid phase is maintained.

Thought should be given to illuminating the contents of some of the vials to provide a suitable environment for the growth of photosynthetic organisms. A simple tungsten lamp would be quite satisfactory.

IV.6 OPERATIONAL REQUIREMENTS FOR INSTRUMENTS AND MEASUREMENT PROGRAM

Placing the soft landing vehicles at the selected site at the beginning of the lunar day will provide the maximum operating time for instruments powered by solar energy apparatus. A few instruments may be operated over the lunar night. This, however, requires stored electrical energy and active heating of the components. Likewise, other instruments may remain inoperative during the night period and may be reactivated during the following day.

The instruments aboard the stationary packet may be divided into separate groups according to their required operating lifetimes. These are:

(1) Instruments accorded a high priority and given thermal protection during the lunar night. Operating time is continuous for as long as possible. Apparatus in this category are:

- (a) Gravimeter
- (b) Seismometer
- (c) Thermal probes
- (d) Communication equipment

(2) Instruments accorded a high priority but used only during the first lunar day. These are as follows:

- (a) Sampler with visual and magnetic study equipment, sample handling mechanism
- (b) X-ray fluorescence spectrometer
- (c) Radioactivity counter
- (d) Biological experiment
- (e) Television system

(3) Instruments accorded a lower priority. These instruments are required to operate during the lunar day and during a short period of the lunar

night. Such instruments, if possible, should be reactivated during the second lunar day. The following instruments fall into this category:

(a) Plasma probe

(b) Magnetometer

(c) Mass-ion spectrometer

(d) Redhead density monitor

(e) Electric field probes

Most instruments aboard the roving vehicle will be used only during the lunar day. Apparatus which is designed primarily to treat and analyze samples of lunar materials, for instance, may be used and allowed to freeze out after the end of the vehicle traverse. Other instruments may be used advantageously for longer periods. These are the seismic apparatus and the thermal probe used for continuous operation during the lunar night by affording thermal protection to a very small communications compartment. The seismic system, an optional item aboard the roving vehicle, may be designed to operate as a passive system after the vehicle has ceased roving. A small seismic sensor, designed to fit on the base of the thermal probe, may be placed into the last hole drilled by the sampling apparatus. If, at depth, the temperature remains above the freeze-out point for the sensor, the device may be used without active heat control. The amplifier and transmitter, however, would depend on heat from operation with possibly some additional heat to function.

The operational temperature ranges for the proposed instrumentation, based on present studies, are as follows:

(1) Lunar day-night range (-150°C to +130°C)

(a) Thermal probe

(2) -20°C to +60°C

(a) X-ray apparatus

(3) Controlled temperature at either 0° or at 60°C

(a) Gravimeter

(b) Seismometer

(4) Controlled temperature around 24°C

(a) Biological experiment

(5) Within instrument compartment limits (0°C to 60°C)

(a) Television system

(b) Magnetometer

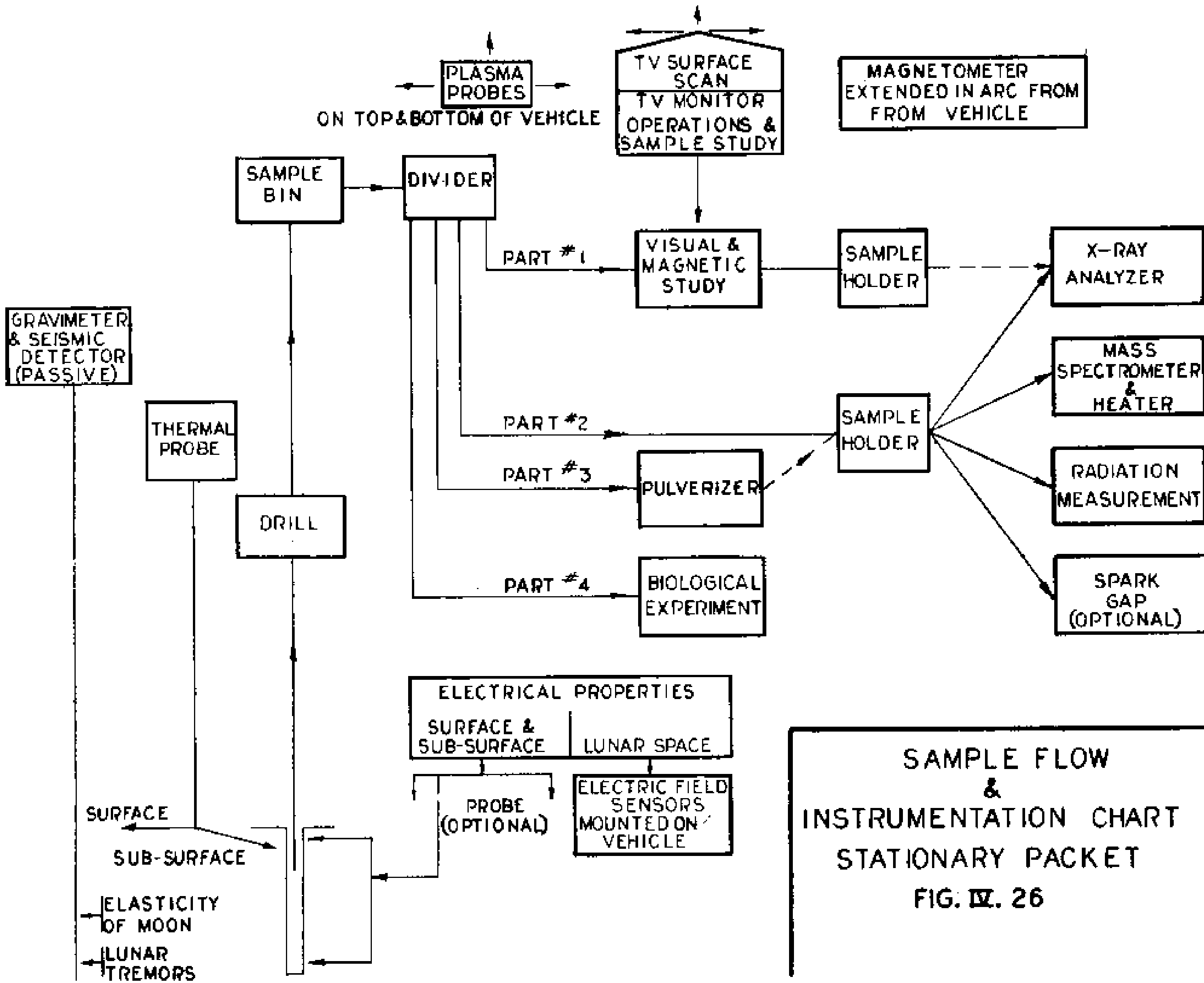
(c) Mass-ion spectrometer

(d) Redhead density monitor

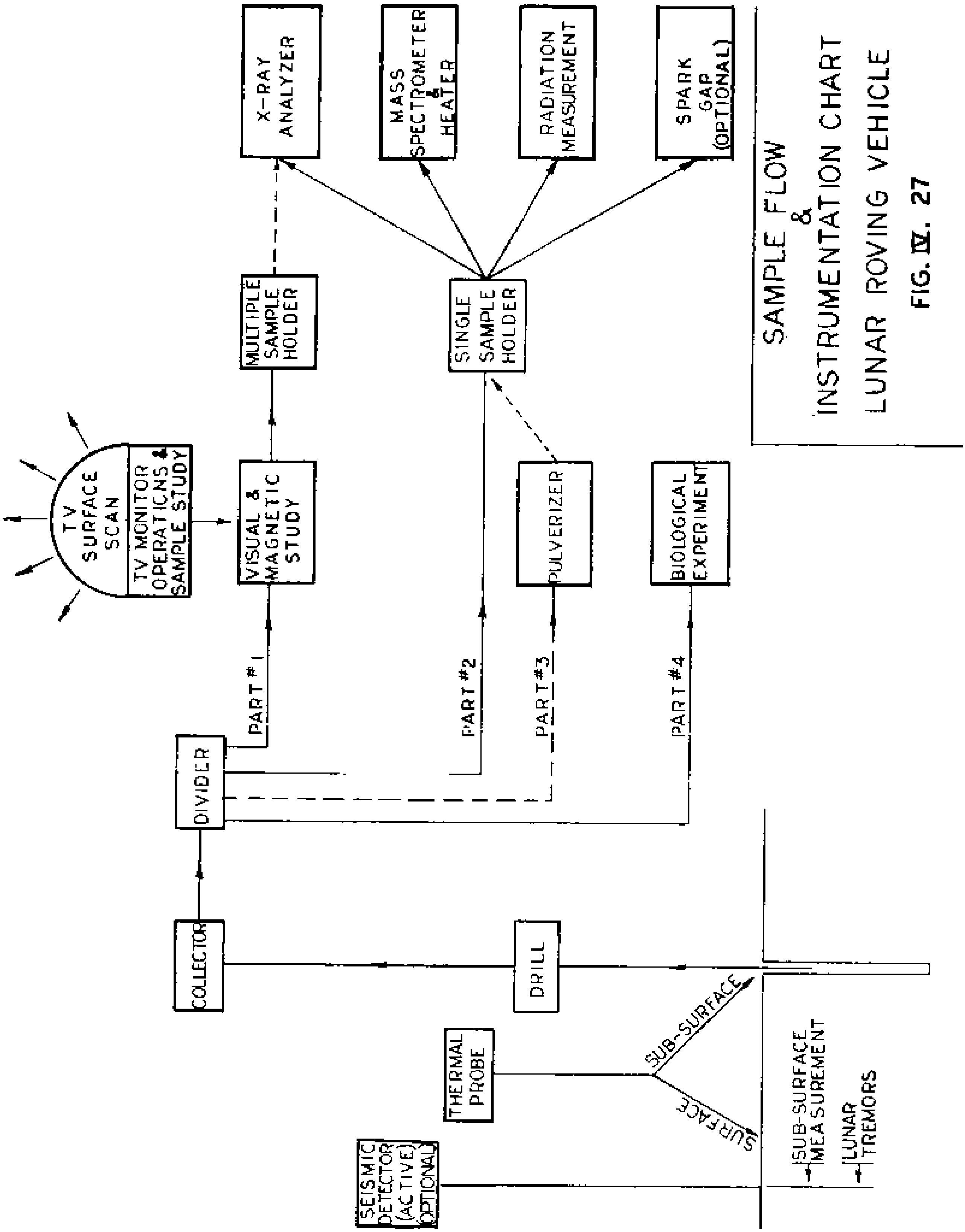
(e) Electric field probes

By selecting the optimum materials and components in making an instrument, the temperature range at which it may operate with stability may be increased. Instruments which require careful temperature calibration may still be used on the moon through proper selection of their temperature sensitive elements. A program of study now underway on the gravimeter follows this approach. The thermally controlled compartment on the stationary packet may be held between 0°C and 60°C with present design considerations. The use of instrument heaters which are thermostatically controlled may keep the temperature of the instrument parts to within $\pm$ 0.01°C of the compartment temperature, if this close control should be necessary. Power, weight, and size required by the scientific instrumentation and accessory equipment are given in Table IV.1. In order to follow the sequence of operation, performed by the scientific apparatus on both the stationary and roving vehicles, Figures IV.26 and IV.27 may be read with Tables IV.2 and IV.3.

Power requirements for instrumentation and communication preclude the use of all instruments simultaneously. Tables IV.2 and IV.3 show a possible sequence of operation which works within the power supply and communication restrictions. The power demand for that group of instruments used to obtain and analyze the lunar material is the highest, followed by those instruments which investigate atmospheric phenomena. The smallest power demand is from instruments accorded a high priority for continuous long-time measurements.



SAMPLE FLOW
&
INSTRUMENTATION CHART
STATIONARY PACKET
FIG. IV. 26



SAMPLE FLOW & INSTRUMENTATION CHART
LUNAR ROVING VEHICLE
FIG. IV. 27

INSTRUMENT OR ACCESSORY	WEIGHT # S*	WEIGHT # R*	POWER-WATTS S*	POWER-WATTS R*	REMARKS	
Thermal Probes:						Lunar Month Operation
Surface probe	1	1	1	1	Sensor extended from payload.	
Subsurface probe	3	3	1	1	½in. flexible rod, 40" length.	
Subsurface probe reel	2	2	10	10	Used to extend probe.	
Gravimeter (Tidal)	50	-	6	-	Special compartment, 18"x15"x15".	
Seismometer:						
Passive system	10	-	1	-	Size- Cylinder 8" dia. x 5".	
Active system (optional)	-	8	-	2.4	Size (amplifier 6"x6"x12") Can be used as passive system later.	
Mass-Ion Spectrometer (ea)	8	8	20	20	Size- 45 cu. in.	Lunar Day - Part Night
Redhead gauge	8	8	3	3	Size- 20 cu. in. (Monitor)	
Heating chamber	5	5	25	25	Size- 2" dia. x 5". (Sample study)	
Plasma Probes (each)	2	2	12	12	Size- Each 6" dia. x 4".	
Magnetometer(Helium vapor)	4	4	4	4	Size- Sensors 2" x 2".	
Electric Field probes	1	1	1	1	Size- 12 cu. in. Extended from vehicle.	
Biological Experiment	5	5	2	2	Size- 40 cu. in.	

TABLE IV. 1

* S = Stationary Packet; R = Roving Vehicle

INSTRUMENT OR ACCESSORY	WEIGHT # S*	WEIGHT # R*	POWER-WATTS S*	POWER-WATTS R*	REMARKS	
Television system:						LUNAR DAY OPERATION
Transmitter	10	10	70	70	Size -220 cu. in.	
Cameras	30	20	24	16	Each camera: 150 cu. in; 5; 4W	
Sampling apparatus:	60	75				
Drill unit motor			200	500	See Figures IV-28 and 29 for	
Drill lift motor			50	50	Stationary Packet	
Sample lift motor			10	10		
Drill and Collector lift			--	20	See Figures IV-11 and IV-13 for	
Rotary Grid motor			--	10	Roving Vehicle	
U.V. and Magnetic sampler			30	30		
Sample holder motor			10	10		
Pulverizer			25	25		
X-Ray Fluorescence Spectrometer	40	--	30	-	Size- 800 cu. in.	
Spectrometer and Diffractometer	--	50	-	750	Size- approx. 1000 cu. in. Operates on cycle as combination instr.	
Alpha Radiation Counter	10	10	1	1	Cone shaped. 1000 sq. cm. at base Can be used for surface scan	
Support Structures, accessory hardware, communication equipment, program control, automation components, etc.	200	70			Stationary Packet: 240 pounds Available for additional nightime power supply and duplication of certain communication and instrument components.	
Total weight of instruments and accessories	459	290				

TABLE IV.1

* S = Stationary Packet ; R = Roving Vehicle.

TABLE IV. 2

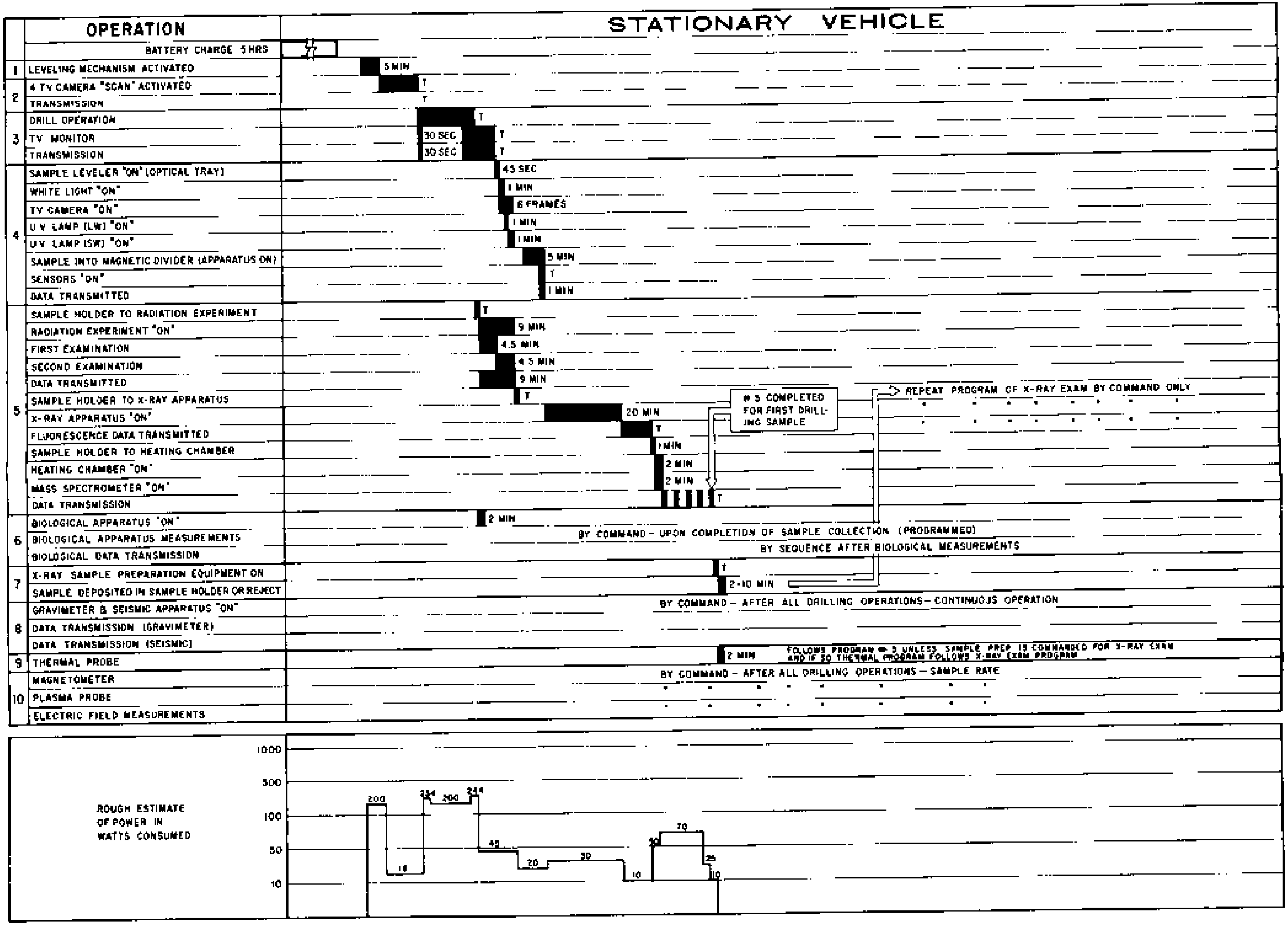
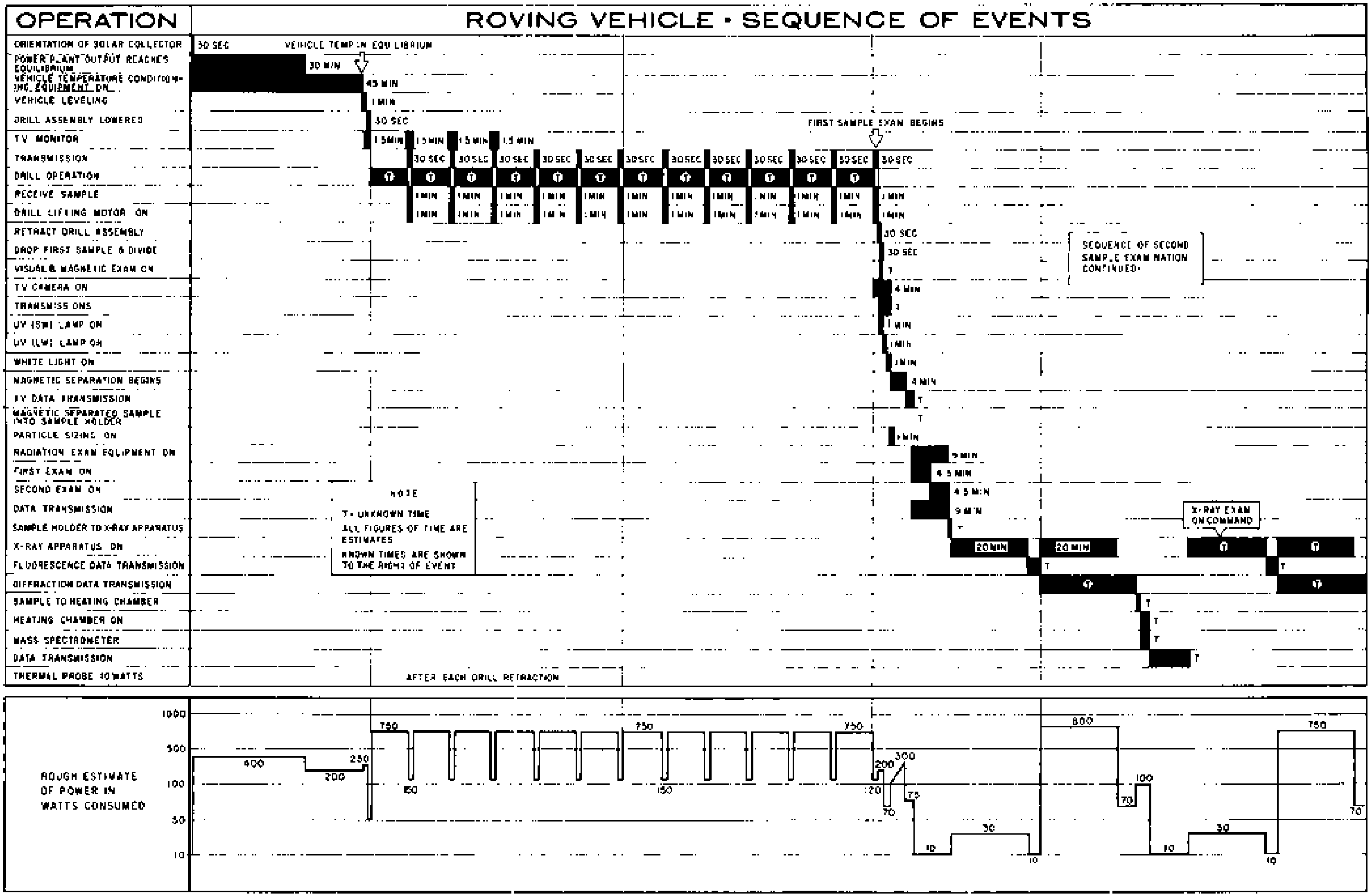

OPERATION
STATIONARY VEHICLE
BATTERY CHARGE 5 HRS
1 LEVELING MECHANISM ACTIVATED
2 4 TV CAMERA "SCAN" ACTIVATED
TRANSMISSION
3 DRILL OPERATION
TV MONITOR
TRANSMISSION
4 SAMPLE LEVELER "ON" (OPTICAL TRAY)
WHITE LIGHT "ON"
TV CAMERA "ON"
U V LAMP (LW) "ON"
U V LAMP (SW) "ON"
SAMPLE INTO MAGNETIC DIVIDER (APPARATUS ON)
SENSORS "ON"
DATA TRANSMITTED
5 SAMPLE HOLDER TO RADIATION EXPERIMENT
RADIATION EXPERIMENT "ON"
FIRST EXAMINATION
SECOND EXAMINATION
DATA TRANSMITTED
SAMPLE HOLDER TO X-RAY APPARATUS
X-RAY APPARATUS "ON"
FLUORESCENCE DATA TRANSMITTED
SAMPLE HOLDER TO HEATING CHAMBER
HEATING CHAMBER "ON"
MASS SPECTROMETER "ON"
DATA TRANSMISSION
6 BIOLOGICAL APPARATUS "ON"
BIOLOGICAL APPARATUS MEASUREMENTS
BIOLOGICAL DATA TRANSMISSION
7 X-RAY SAMPLE PREPARATION EQUIPMENT ON
SAMPLE DEPOSITED IN SAMPLE HOLDER OR REJECT
8 GRAVIMETER & SEISMIC APPARATUS "ON"
DATA TRANSMISSION (GRAVIMETER)
DATA TRANSMISSION (SEISMIC)
9 THERMAL PROBE
10 MAGNETOMETER
PLASMA PROBE
ELECTRIC FIELD MEASUREMENTS
5 MIN
T
30 SEC
30 SEC
45 SEC
1 MIN
6 FRAMES
1 MIN
1 MIN
5 MIN
1 MIN
9 MIN
4.5 MIN
4.5 MIN
9 MIN
20 MIN
1 MIN
2 MIN
2 MIN
5 COMPLETED FOR FIRST DRILLING SAMPLE
REPEAT PROGRAM OF X-RAY EXAM BY COMMAND ONLY
2 MIN
BY COMMAND - UPON COMPLETION OF SAMPLE COLLECTION (PROGRAMMED)
BY SEQUENCE AFTER BIOLOGICAL MEASUREMENTS
2-10 MIN
BY COMMAND - AFTER ALL DRILLING OPERATIONS - CONTINUOUS OPERATION
2 MIN
FOLLOWS PROGRAM # 5 UNLESS SAMPLE PREP IS COMMANDED FOR X-RAY EXAM AND IF SO THERMAL PROGRAM FOLLOWS X-RAY EXAM PROGRAM
BY COMMAND - AFTER ALL DRILLING OPERATIONS - SAMPLE RATE
ROUGH ESTIMATE OF POWER IN WATTS CONSUMED
1000
500
100
50
10
200
18
244
200
244
45
20
30
10
50
70
25
10

TABLE IV.3


OPERATION
ROVING VEHICLE • SEQUENCE OF EVENTS
ORIENTATION OF SOLAR COLLECTOR
POWER PLANT OUTPUT REACHES EQUILIBRIUM
VEHICLE TEMPERATURE CONDITIONING EQUIPMENT ON
VEHICLE LEVELING
DRILL ASSEMBLY LOWERED
TV MONITOR
TRANSMISSION
DRILL OPERATION
RECEIVE SAMPLE
DRILL LIFTING MOTOR ON
RETRACT DRILL ASSEMBLY
DROP FIRST SAMPLE & DIVIDE
VISUAL & MAGNETIC EXAM ON
TV CAMERA ON
TRANSMISSIONS
UV (SW) LAMP ON
UV (LW) LAMP ON
WHITE LIGHT ON
MAGNETIC SEPARATION BEGINS
TV DATA TRANSMISSION
MAGNETIC SEPARATED SAMPLE INTO SAMPLE HOLDER
PARTICLE SIZING ON
RADIATION EXAM EQUIPMENT ON
FIRST EXAM ON
SECOND EXAM ON
DATA TRANSMISSION
SAMPLE HOLDER TO X-RAY APPARATUS
X-RAY APPARATUS ON
FLUORESCENCE DATA TRANSMISSION
DIFFRACTION DATA TRANSMISSION
SAMPLE TO HEATING CHAMBER
HEATING CHAMBER ON
MASS SPECTROMETER
DATA TRANSMISSION
THERMAL PROBE 10 WATTS
30 SEC
VEHICLE TEMP IN EQUILIBRIUM
30 MIN
45 MIN
1 MIN
30 SEC
1.5 MIN
30 SEC
1 MIN
FIRST SAMPLE EXAM BEGINS
4 MIN
9 MIN
4.5 MIN
20 MIN
SEQUENCE OF SECOND SAMPLE EXAMINATION CONTINUED
X-RAY EXAM ON COMMAND
NOTE
T = UNKNOWN TIME
ALL FIGURES OF TIME ARE ESTIMATES
KNOWN TIMES ARE SHOWN TO THE RIGHT OF EVENT
AFTER EACH DRILL RETRACTION
ROUGH ESTIMATE OF POWER IN WATTS CONSUMED
1000
500
100
50
10
400
200
250
750
150
200
300
120
70
75
10
30
800
100
70

Weight and size are obtained, in large part, by educated guesses. They are based on a knowledge of present instrument configurations. Certain instruments, for instance gravimeter and seismic apparatus, may be combined into one packet, while others, to effect a weight saving, may be modified for combined functions. The total weight of the proposed instrumentation is approximately 260 pounds for the stationary packet and 220 pounds for the roving vehicle. Support structures, accessory hardware, communication equipment, automation components, etc. use an additional 200 pounds in the stationary packet and the balance of the available weight in the roving vehicle. An additional 240 pounds capacity is available in the stationary packet to provide added power or instrumentation. Instruments in category (3) discussed previously can be provided with power to operate well into the night periods by using a portion of the weight capability for batteries. If desired, a duplicate system for certain instruments and components is also possible.

The sequence tables (IV.2 and IV.3) are self-explanatory. At the bottom of each table is shown the peak power demand during one sampling period. Likewise indicated is additional information, such as the relative time of operation of the apparatus and the data transmission time.

IV.7 INSTRUMENTATION ASSEMBLY

The following components for the stationary packet payload assembly are illustrated in Figures IV.28 and IV.29:

(1) Sample bin

(2) Television camera

(3) Ultraviolet lamp

(4) Clear light

(5) Sample holder

(6) Magnetic separator

(7) Sample preparation (crusher)

(8) X-ray analyzer

(9) Mass spectrometer

(10) Biological experiment

(11) Radiological experiment

(12) Rossi Faraday plates

(13) Magnetometer

(14) Drilling apparatus for sampler

(15) Spiral sample conveyor

(16) Batteries

(17) Gravimeter

(18) Television system

(19) Transmitter

(20) Further apparatus as required

(21) Television surface scanner

(22) Guidance components (for terminal guidance only)

(23) Command receiver

(24) Dumping device for sample holder

(25) Device for insertion of new cup in sample holder

(26) Thermal probe

(27) Motor for spiral sample conveyor

(28) Motor for sample holder

(29) Coiled tubing for temperature control

(30) Further apparatus as required

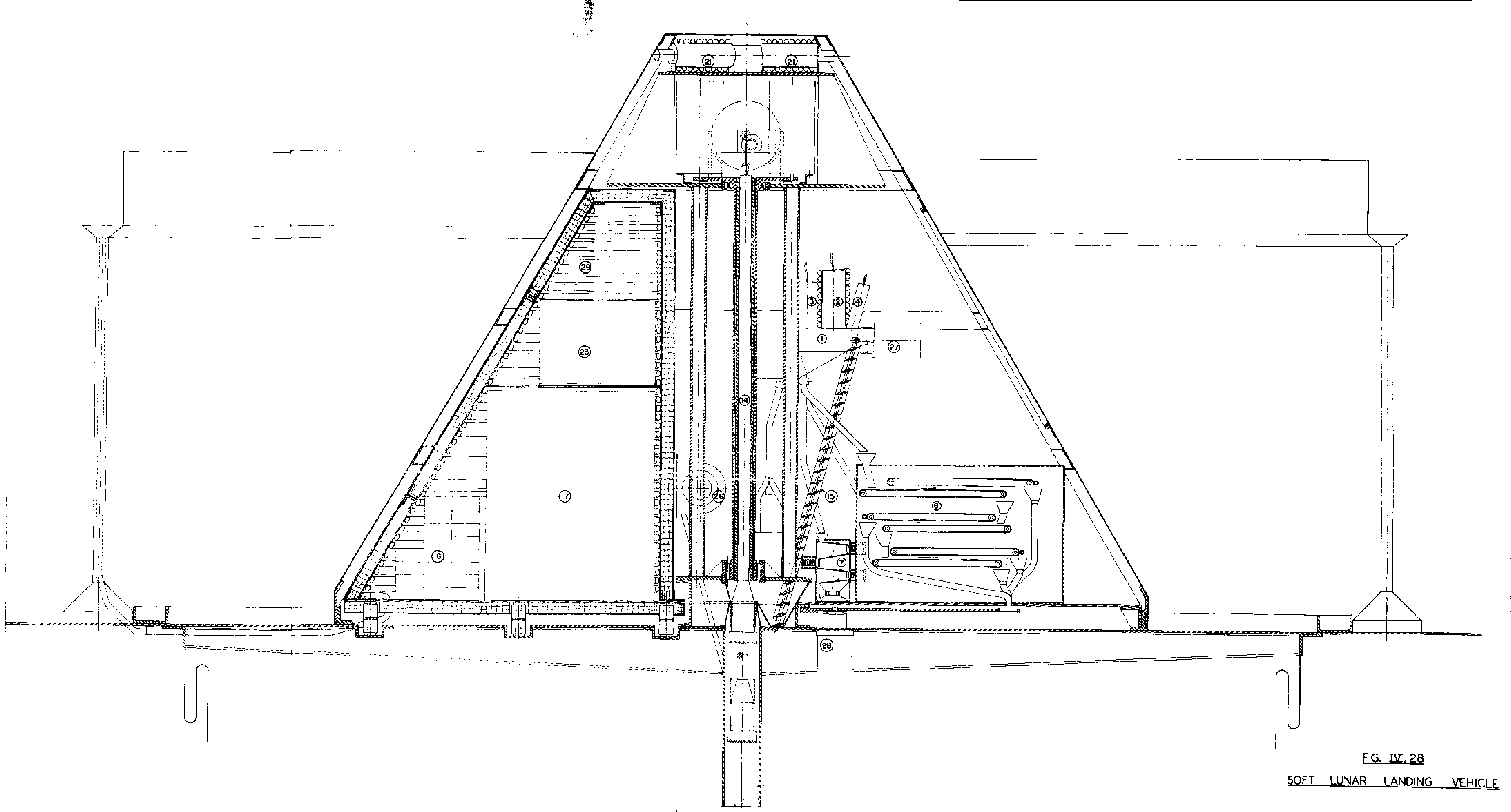


FIG. IV. 28

SOFT LUNAR LANDING VEHICLE

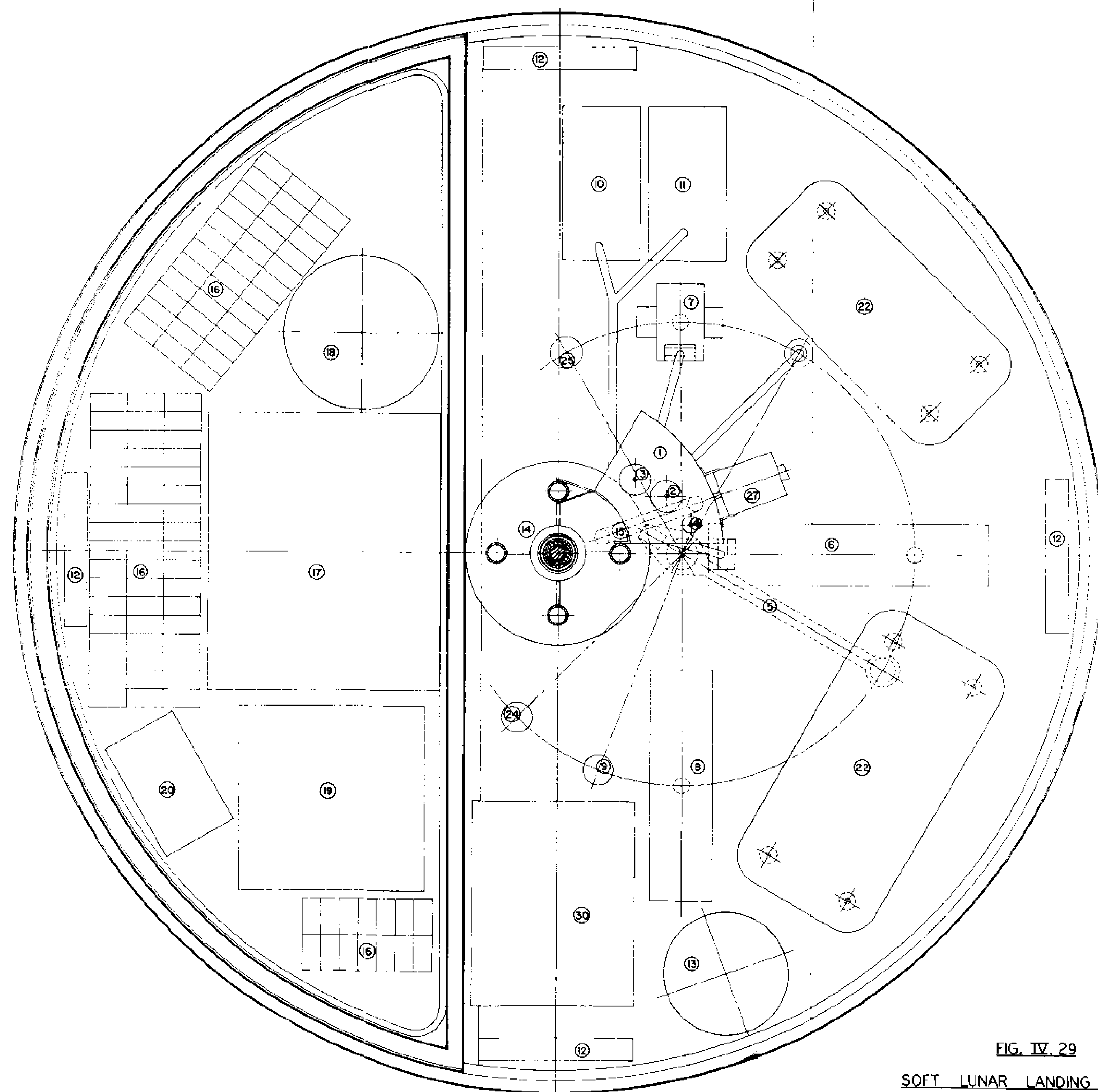


FIG. IV. 29

SOFT LUNAR LANDING VEHICLE

The following components of the roving vehicle payload assembly are illustrated in Figure IV.30 :

(1) Drill

(2) Drill drive motor

(3) Drill lift motor

(4) Drill thrust springs

(5) Drill sequence control

(6) Conveyor

(7) Conveyor drill motor

(8) Sample collector

(9) Collector drive motor

(10) Sample divider

(11) Pulverizer

(12) Rotary grid

(13) Rotary grid drive motor

(14) TV Monitor

(15) Ultraviolet light source

(16) White light source

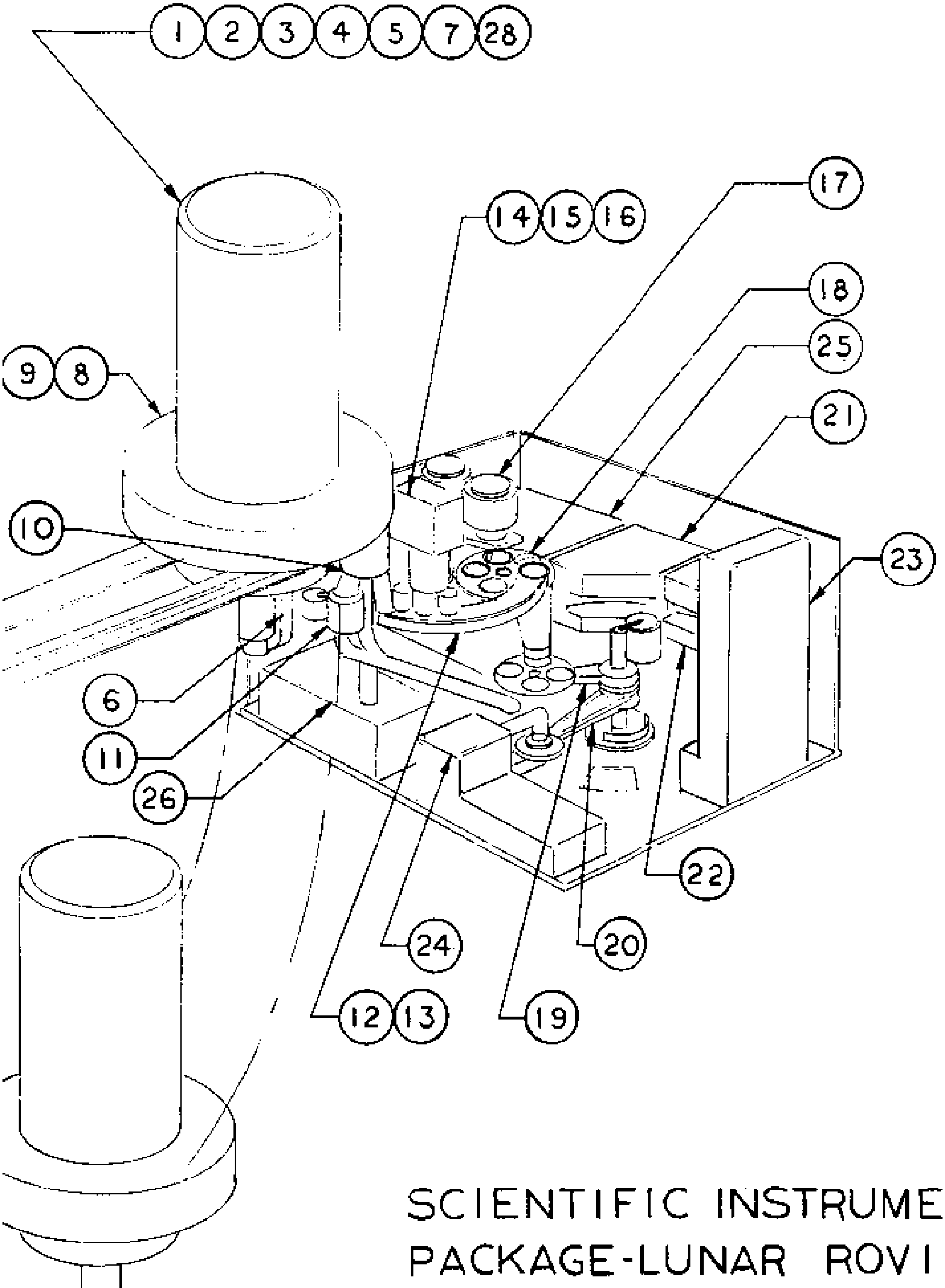
(17) Electro-magnet

(18) Sample containers

(19) Multiple sample holder and drive motor

(20) Single sample holder and drive motor

(21) Radiation counter



SCIENTIFIC INSTRUMENT PACKAGE-LUNAR ROVING VEHICLE

FIG. IV. 30

(22) X-ray analyzer

(23) Mass spectrometer and heating chamber

(24) Spark gap spectrometer (optional)

(25) Communication and control equipment

(26) Biological experiment

(27) Seismic apparatus (optional)(not shown)

(28) Thermal probe (not shown)

IV.8 THE COMMUNICATIONS SYSTEM

The communications system for the Soft Lunar Landing program will be required to handle a variety of data from injection to the end of the experiment.

During the mid-course phase of the trajectory, the system will be required to transmit missile performance data. Then during the terminal guidance phase, if a television system as previously described is used, it must transmit picture data from the vehicle to the earth and command data from the earth to the vehicle. Later, after landing has been accomplished, the system must transmit scientific data and television pictures from the moon to earth and transmit control commands from the earth to the moon.

An evaluation of the information rate required to transmit the data indicates that the mid-course telemetry data rate and the scientific data rate will be approximately the same, and that the television transmissions will require a much greater bandwidth than either of these.

Such considerations indicate that one should plan to use the same transmitter for mid-course telemetry and for scientific data transmission after the vehicle has landed on the moon. The high information rate of the television system and the corresponding increase in transmitter power required suggests a second transmitter designed for the television function. It is quite probably that the two transmitters will operate near the same frequency and therefore only one antenna will be required for both transmitters after the arrival on the moon.

The most favorable frequency band for the lunar communication system seems to be 1000 to 2000 Mc, if only favorable propagation conditions are considered. Stringent requirements for highest transmitter efficiency and lowest weight, however, would call for lower frequencies, where the application of transistors becomes possible. On the other hand, vehicle design problems (antenna size to achieve 20° beamwidth) make even higher frequencies desirable. Therefore, the final choice of the carrier frequency will be influenced mainly by future developments in the field of high frequency transistors. Studies are continuing to find the best compromise.

At least three ground stations will be required on the ground to permit continuous contact with the lunar-based transmitters. Additional study and cost evaluation will be required to determine whether each ground station will be complete with data recording, reduction, evaluation, and command capabilities.

Telemetry System

The telemetry system will transmit vehicle performance data during the mid-course and terminal phase of lunar flight. On the moon's surface, it will transmit scientific and vehicle data from the landed vehicle.

In view of the similarity of measurement programs for the stationary packet and roving vehicle missions, almost identical telemetry systems for the two missions are considered. Where necessary, modification may be made in the telemetry system to accommodate variations in measuring requirements.

System Considerations

An analysis of the measurement quantities indicates that some 100 separate pieces of information must be telemetered to the earth to evaluate certain physical phenomena of the moon and lunar vehicle operations. The results of the measured quantities or the data to be telemetered must be transformed to a form suitable for transmission; in other words, the most suitable type of multiplexing and modulation which will require a minimum RF radiated power. Of the known modulation techniques pulse code modulation (PCM)[1] appears to offer the most advantage

[1] Information transmission by means of a code of a finite number of symbols representing a finite number of possible values of the information at the time of sampling.

if minimum radiated power and flexibility in data reduction are the primary considerations. Except for data from the seismic experiment the data to be transmitted are of a slowly changing nature, and are well suited to electronic sampling. The sampled data can then be encoded by PCM techniques and transmitted at a rate to be selected.

For expedient data handling on the receiving end, the pulse coded data has an advantage over analog in that it can be fed into a digital computer for fast data reduction.

The frequency of occurrence of seismic events on the moon is not known; however, when and if the seismic apparatus is stimulated, the characteristics of the data expected are generally known from a study of earth seismic records.

This data is a transient type and may be analyzed for rise time, amplitude and other pertinent characteristics on-board the vehicle using predigestion techniques. The resulting data can then be fed to an electronic memory along with the time of occurrence and read into the telemetry system when requested by a programming device, or by command as desired.

An alternative solution for on-board analysis is amplitude distribution measurement of the seismic signal. Thus, the expected amplitude range is subdivided into several "slices" (e.g. 10 or 15) and the percentage of time during which the signal falls within each of these intervals is measured. The interrogation rate can easily be tailored to the demands of the telemetry multiplexer, no separate memory device is needed.

Both methods of on-board analysis might give a survey of the seismic activity on the moon. In case further studies indicate that these analysis methods alone cannot give a complete description of all characteristics of scientific interest, short samples of seismic data could be transmitted in addition. For this real-time transmission, a television carrier could be used, as indicated in Figure IV.31. The transmission (or recording) periods would be triggered by the seismic events.

The tentative choice of PCM means that the data will be transmitted as binary coded, time multiplexed samples (a sample is defined as a voltage level corresponding to a measured value). The binary coded pulses must then be transmitted to the earth over an RF carrier. The modulation technique will be that which will provide a maximum signal to noise ratio at the receiver. Of the known types, past experience indicates

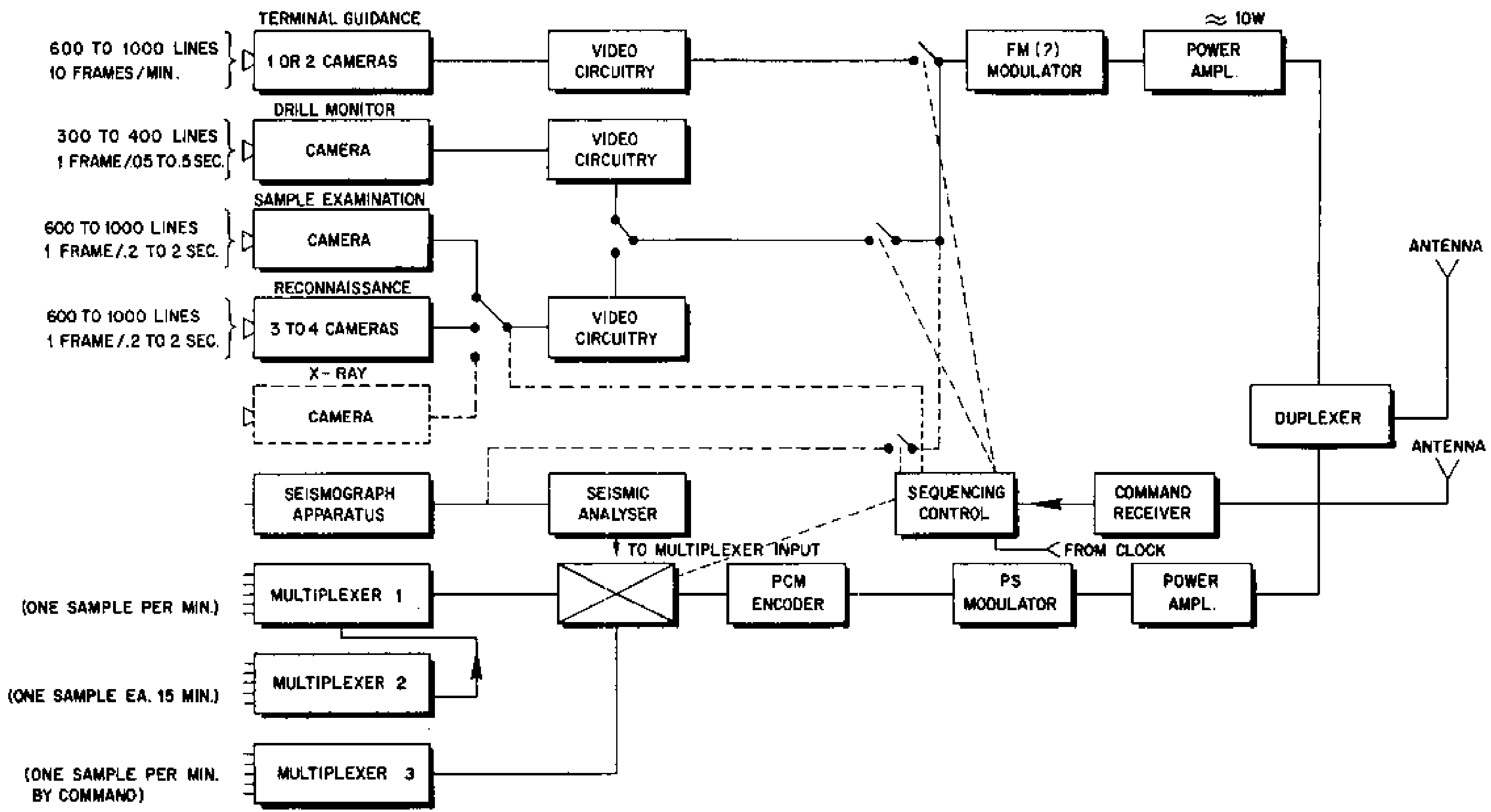


FIG. IV. 31

COMMUNICATION SYSTEM FOR SOFT LUNAR LANDING (STATIONARY VEHICLE)

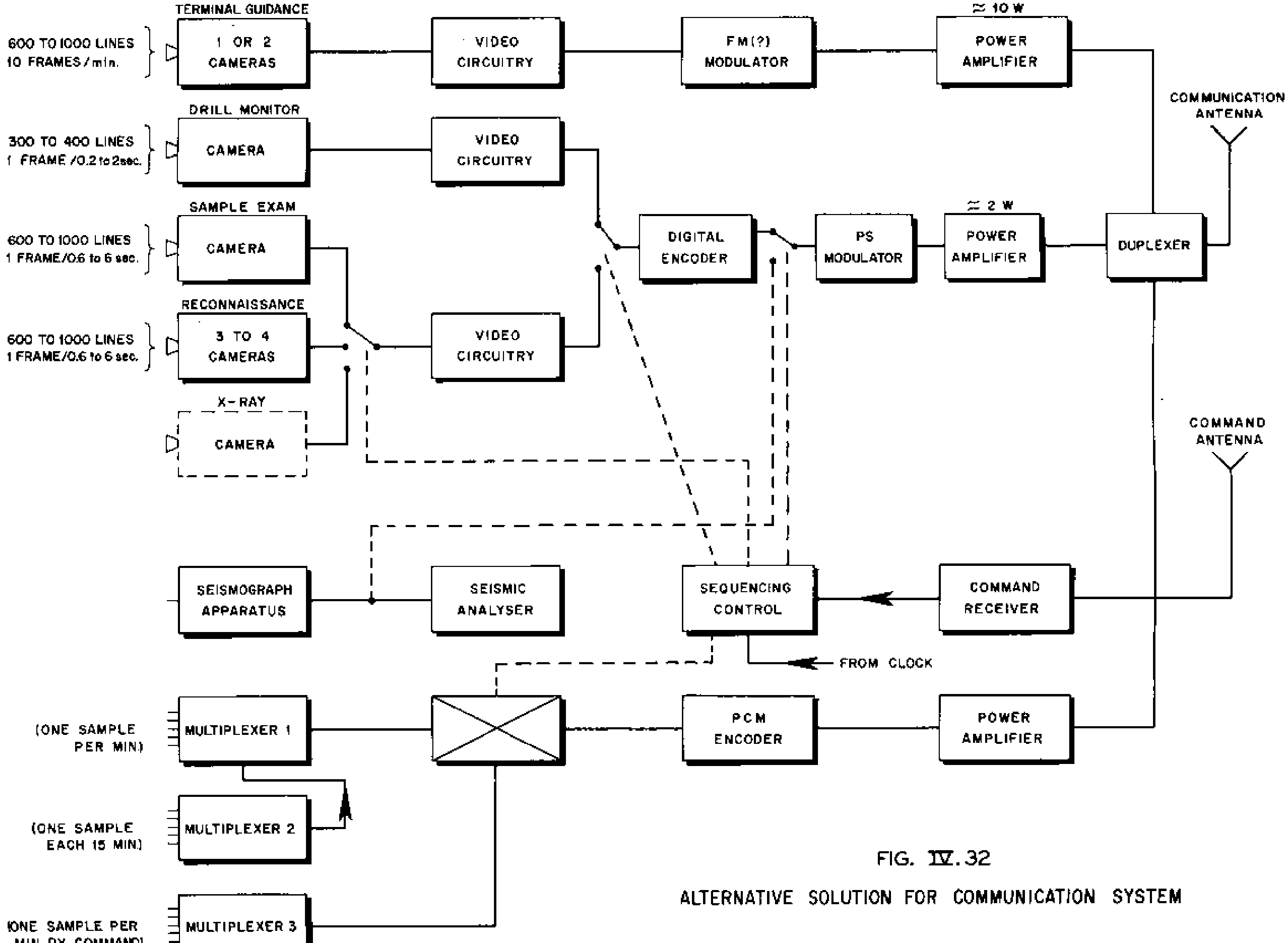


FIG. IV.32

ALTERNATIVE SOLUTION FOR COMMUNICATION SYSTEM either frequency or phase shift modulation (FM or PM) may be the better choice.

There is a growing trend in the use of PCM-FM and PCM-PM methods of data transmission on missile and aircraft test ranges. Standards have been established and it is envisioned that PCM-PM telemetry design and components will be readily available for lunar vehicles and ground receiving stations.

In considering such parameters as transmitter power, orientation problems, etc., one sees that a compromise must be made in the choice of RF radiator. Calculations indicate that an antenna system which can concentrate the RF energy into a relatively wide beam (20°) may have sufficient advantage over an omnidirectional system to warrant its use.

CHAPTER V

MANNED CIRCUMLUNAR FLIGHTS AND LUNAR LANDINGS

V.1 OBJECTIVES OF MANNED MISSIONS (U)

The ultimate goal of the lunar exploration program will be the establishment and operation of a manned lunar base. However, many problems must be solved prior to man's setting foot on the lunar surface. The first phase of the over-all program is to determine the nature of the moon and its environment. The scientific exploration program described in Chapter IV is expected to accomplish this objective. The second phase is to provide a system which may, during any portion of its operation, assure the safe return of human passengers to the earth. Only then will man be capable of traveling to the moon.

Development of a reliable system for transportation of man to the moon and back to earth will be accomplished in several steps. First, a vehicle must be designed which can safely re-enter the earth's atmosphere. The proven JUPITER nose cone, plus simulated return flights during the early testing of the SATURN vehicle, will be the bases for development of the re-entry vehicle. Second, unmanned flights to circumnavigate the moon and return to earth will provide data on the actual conditions encountered on the round trip. Third, manned circumlunar flights will present the first tests for a human passenger traveling to the vicinity of the moon and returning to earth. Fourth, and finally, manned lunar landings will be accomplished.

This chapter will describe some of the problems associated with circumlunar flights, manned landings on the moon and recovery of a manned capsule on earth. The circumlunar flights can be accomplished with the SATURN B-1 vehicle as described in Section III. The manned lunar landings must be accomplished by orbital fueling or assembly maneuvers.

V.2 CIRCUMLUNAR FLIGHTS (U)

The first circumlunar flights will be unmanned. Therefore, the complete flight system will be automatic or remotely controlled from the earth. These early flights will provide necessary data for later manned missions and will test the over-all system. Animal-carrying flights will follow, providing physiological data. Scientific instruments will probably be carried to measure the moon's fields and the space environment. In addition, motion picture and still cameras will record the approaches to the moon and earth, and details of the lunar surface, including the side away from the earth.

Later manned flights will emphasize the capability of the passengers to observe and interpret events during the flight and details on the lunar surface. Scientific instrumentation will be carried as required to complement the abilities of the passengers. Since much will have been learned about the moon and conditions in space prior to the manned flights, any instrumentation named at this time would be strictly speculative.

Because the first circumlunar flights will probably be accomplished to establish system tolerances, a weak encounter, that is, a miss on the order of twenty lunar diameters, is desirable. Any type of stronger encounter will introduce complications to the flight due to lunar influence which may be an undesirable hazard, especially in the first flights. Later circumlunar flights will range within a few kilometers of the lunar surface to provide near passes for more favorable experimental and viewing purposes, as described in Section II.7.

A typical total flight time of approximately ten days has been determined for the first flights. A slightly perturbed elliptical orbit is followed for a weak encounter. In the case of the strong encounter, however, generally all elements of the trajectory will be altered, thus introducing many complications into the system. The complications may result in extended orbital flight time, either inadvertently or directed. Inadvertent conditions can occur if the return flight skips out of the earth's atmosphere due to trajectory errors and might involve an additional time approximating that of flight to the lunar vicinity. It is also possible, in the event of an undesirable earth approach, that the flight might be purposely extended. In any case, the inclusion of quantities of life essentials sufficient for extended orbital flight or ground survival will be desirable for the first few missions.

Even from a scientific standpoint, a weak encounter with the moon during the first few flights may not be entirely unfavorable. At a distance of about sixty to seventy thousand kilometers, a 25 cm telescope will have a resolution of about 150 meters. Valuable lunar pictures may be obtained and scientific experiments may be conducted, in addition to establishing procedures and testing equipment for manned landings to follow.

V.3 MANNED LUNAR LANDINGS (C)

The objective of this section is to discuss various aspects of manned lunar landing missions. As stated earlier, the ultimate objective, at least for the next one or two decades, will be that of establishing a manned lunar base. Before man will be ready to land on the moon's surface a large number of earlier lunar research flights should and will be made as described in Chapter IV. In the following discussion, it is assumed that the flights have been accomplished and that the necessary research and environmental data have been collected and analyzed.

One of the first items to be developed is the transportation scheme for delivering personnel and cargo from the earth's surface to the moon and finally returning the personnel to the earth. Many possibilities exist. However, from the standpoints of total energy required and travel time, only two schemes appear desirable, as shown in Figure V.1.

The first scheme is the direct approach, that is, a vehicle would depart the earth's surface and proceed directly to the lunar surface, using a braking rocket or landing stage for the final descent maneuver. The second scheme shown is that of proceeding first into an earth orbit and later departing the orbit for the flight to the lunar surface, again using a braking stage for descent. In either scheme the flight time from the earth or earth orbit to the moon will be basically the same, in the order of two to two and one-half days, utilizing a relatively high energy trajectory.

The direct scheme, which is the most straightforward, has several major advantages:

(1) It offers the shortest flight time from the earth's surface to the lunar surface since an orbital stop-over is not required.

(2) It eliminates the necessity of developing techniques and equipment for orbital rendezvousing, propellent transfer and vehicle checkout.

It does, however, have a limitation which the orbital scheme does not have. That is, the payload size which can be delivered to the moon is limited to the size carried by a single earth-departing vehicle.

EARTH - MOON TRANSPORTATION SCHEMES

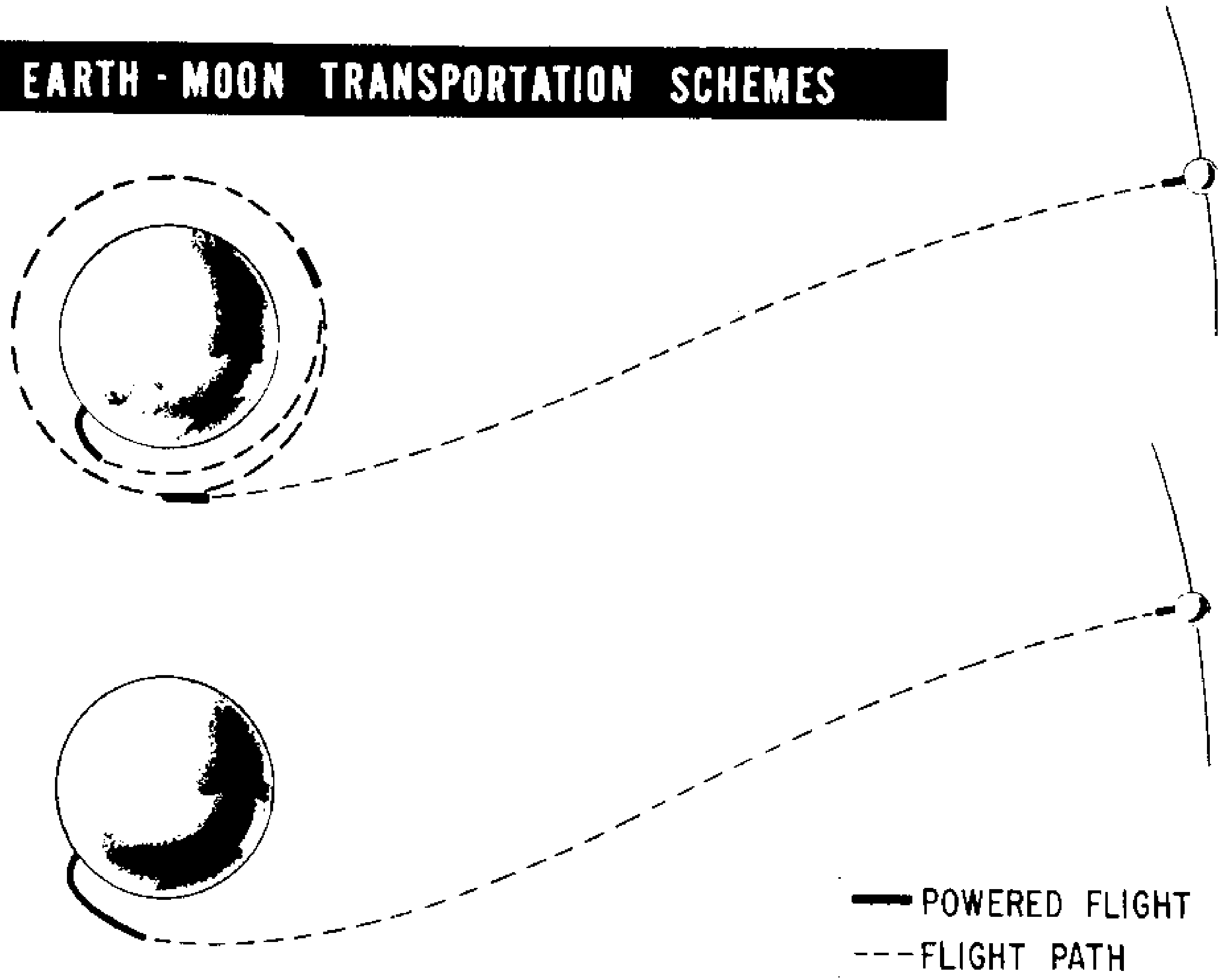


FIG. V.1

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In the orbital scheme, much heavier payloads can be transported into orbit than to the moon, assuming a constant carrier vehicle size. By accumulating payloads in orbit, it is possible to construct or refuel an orbit-departing vehicle which would have many times the lunar landing payload capability of a single earth-departing vehicle.

To illustrate this point for the manned lunar landing and return for a two-man crew, assuming that the two men will have an immediate return capability upon landing on the moon, see Figure V.2. The direct scheme would require a six-stage vehicle with a liftoff thrust of eight to twelve million pounds, depending on upper stage propellents and design. Such a vehicle could softly land on the moon a payload of approximately fifty thousand (earth) pounds which is adequate to return a two-man capsule to earth. For the orbital scheme it would require an orbit-departing vehicle weighing approximately 400,000 pounds to softly land a 50,000-pound payload on the moon. The number of earth-launched vehicles required to provide the 400,000 pounds of payload into orbit, necessary for the orbit-departing vehicle, depends on the size of available booster systems. The two million pound thrust vehicle shown in Figure V.2 is a nominal size of a future (growth potential) SATURN vehicle. As shown, a total of six earth-departing flights are required (one for the orbit-departing vehicle and five refueling tankers) plus a vehicle to carry the refueling and checkout crew into orbit and return them to earth, if a crew is not already available in orbit. There is, however, a practical limit to this scheme:

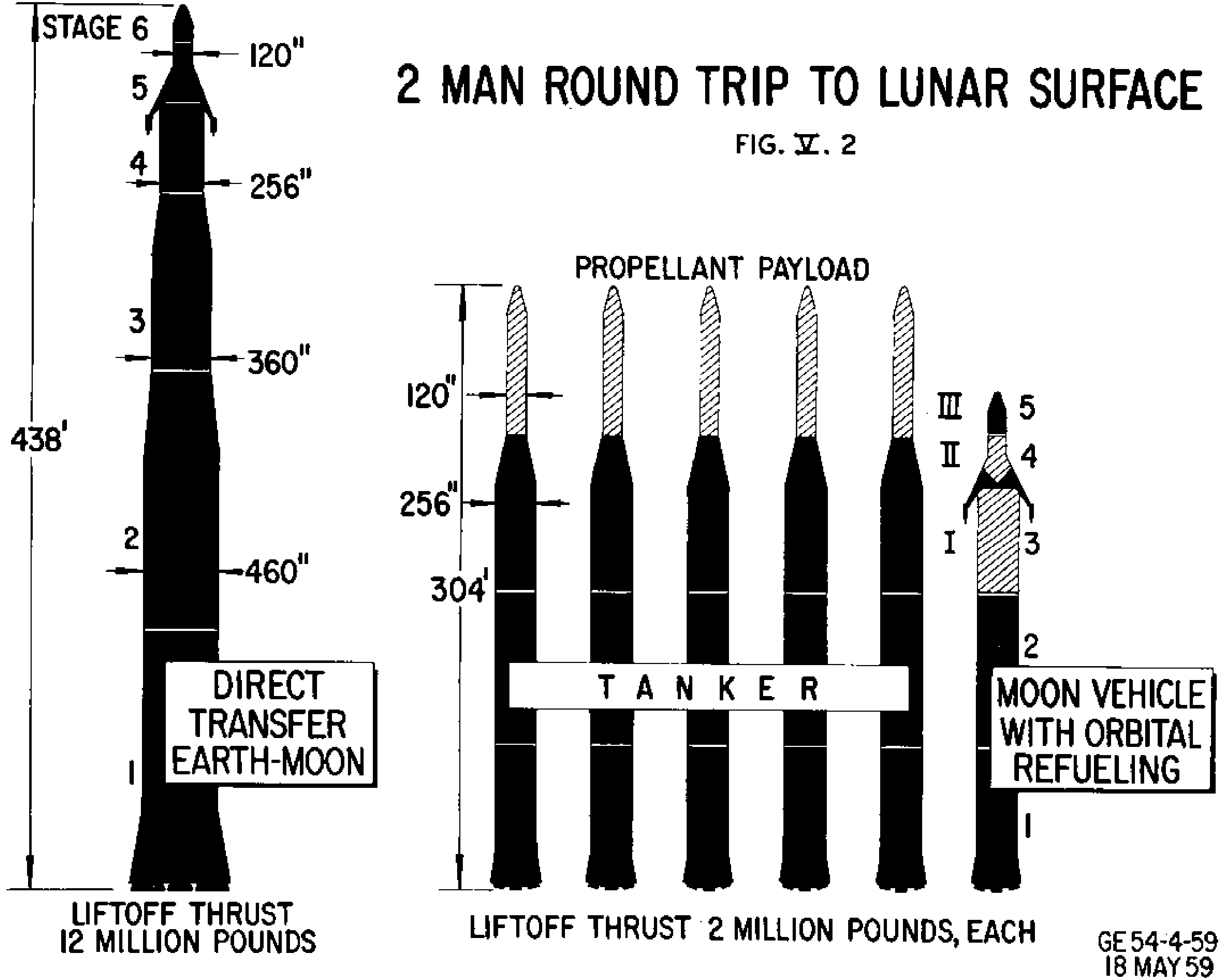
(1) If the orbital payload capability of a single vehicle is less than 45,000 pounds, orbital assembly as well as orbital refueling will be required.

(2) If the SATURN B-1 is considered to be the largest vehicle available, a total of twelve vehicles will be required plus transportation for the orbital crew. This would result in a rather high launch rate since it would be desirable that the orbital operation be completed in a short time (less than six months).

(3) For vehicles smaller than the SATURN B-1, this scheme is not considered feasible due to the requirement for a large number of vehicles, requirement for a considerable amount of orbital construction or assembly, and the extremely high launch rates.

2 MAN ROUND TRIP TO LUNAR SURFACE

FIG. V. 2



The SATURN B-1 vehicle, with the performance characteristics quoted in Section 1.2, could be used for the manned lunar exploration mission. If the SATURN C, as shown in Figure V.3, is developed it could be used for the lunar mission. The SATURN C will consist of a nominal 2 million pound thrust booster and will utilize high energy (H_2 + O_2) propellants in all upper stages. The payload capability for such a vehicle would be:

300-mile orbit	=	80,000 lbs
Escape	=	34,000 lbs
Lunar soft landing	=	11,000 lbs

The SATURN could be available for operational use at least three years, and possibly five or more years, earlier than the eight to twelve million pound thrust vehicle. Therefore, if a manned lunar landing and return is desired before the 1970's, the SATURN vehicle is the only booster system presently under consideration with the capability to accomplish this mission.

In addition to the orbital scheme for manned lunar landing and return, the SATURN offers another possibility to accomplish the mission. It does not, however, provide a manned landing on the moon with an immediate earth-return capability. The scheme is as follows:

(1) Four SATURN C vehicles would be launched from the earth directly to the moon, each softly landing an 11,000 pound payload consisting of a pre-packaged liquid propulsion system. Each of these propulsion systems would have a 10,000 pound thrust engine and 9,000 pounds of usable propellant.

(2) One SATURN C would be launched directly to the moon, and would softly land a crew of two persons in an earth-return capsule. The return capsule would weigh 8,000 pounds plus 3,000 pounds of supplies and equipment.

(3) The landing stage tankage of the manned vehicle would be removed and replaced by the four pre-packaged propulsion systems. For the first manned landing, the crew would be required to perform the task of replacing the empty landing stage tankage with the pre-packaged system before the return flight would be possible.

[redacted]

[redacted]

SATURN

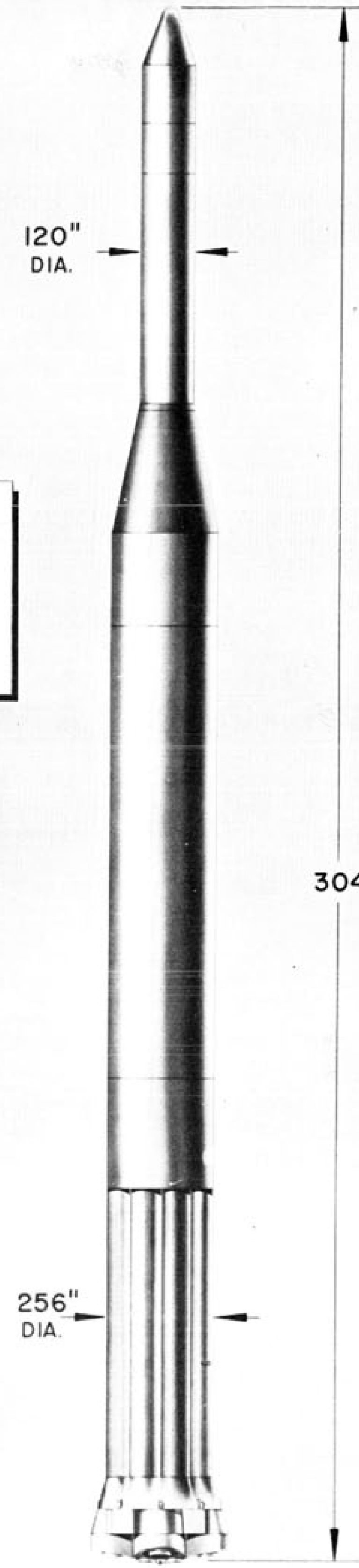


FIG. V. 3

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[redacted]

This scheme, although feasible, is not considered desirable, at least for the early manned landing attempt.

To determine the type and quantity of vehicles required for manned lunar missions, it is necessary to first establish the objectives of the mission and the method by which it will be carried out. If the objectives are such that a manned landing (with a crew of two or three people) are required only once every year or so, the transportation system will be considerably different from one which must establish a 20-man lunar observatory or base and continually support it. Since the final objectives are not clearly defined, the remainder of this section will be devoted to defining and discussing components of a space transportation system.

Such a transportation system should provide for carrying a considerable amount of cargo to the moon as well as manned round trip transportation. Even a minimum type base would require large quantities of construction material, equipment and life support essentials. The life support essentials alone are 450 pounds per man month. Therefore, the transportation system components described later will include cargo vehicles as well as manned vehicles.

In addition to the basic carrier vehicles, the SATURN B-1 or SATURN C, the following additional components of the space transportation system are envisioned:

(1) Manned earth orbital return vehicle. The vehicle will provide round trip transportation for the orbital crew and transportation into orbit for the moon-bound personnel. The vehicle, shown in Figures V.4 and V.5, will be a re-entry body containing a storable liquid propellant engine of approximately 6K thrust for rendezvous, orbital maneuvering, and separation when returning to the earth's surface. The re-entry body will contain seats for personnel; communication equipment; guidance system for ascent, rendezvousing, and return; as well as life supporting essentials and survival equipment. It will be capable of housing a few personnel for a short time and will become a temporary satellite itself. For long periods of time and with more personnel, additional housing must be provided, such as empty propellant containers fitted to serve as living quarters. The most promising re-entry configuration for early operational availability is considered

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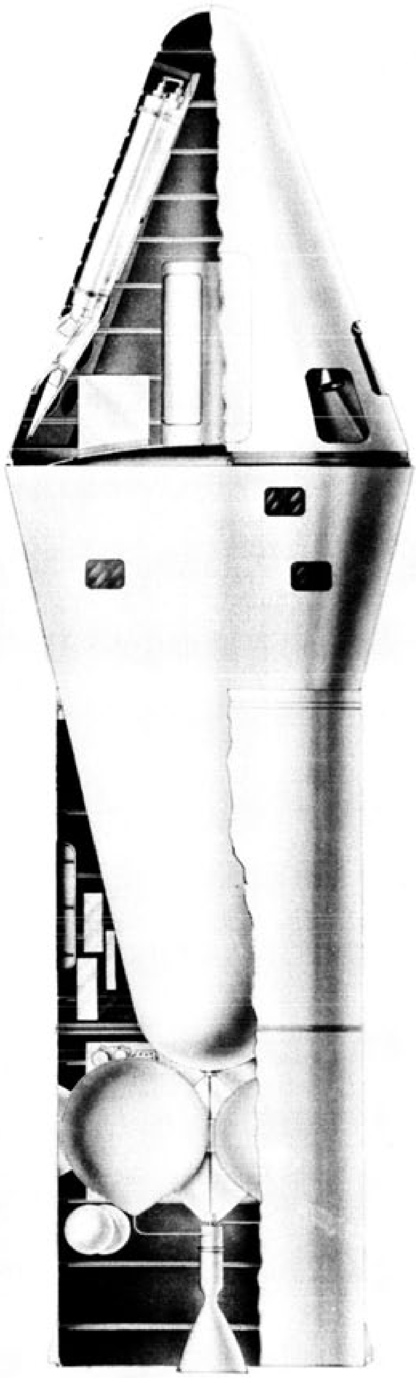

FIG. V.4

MANNED ORBITAL TRANSPORT

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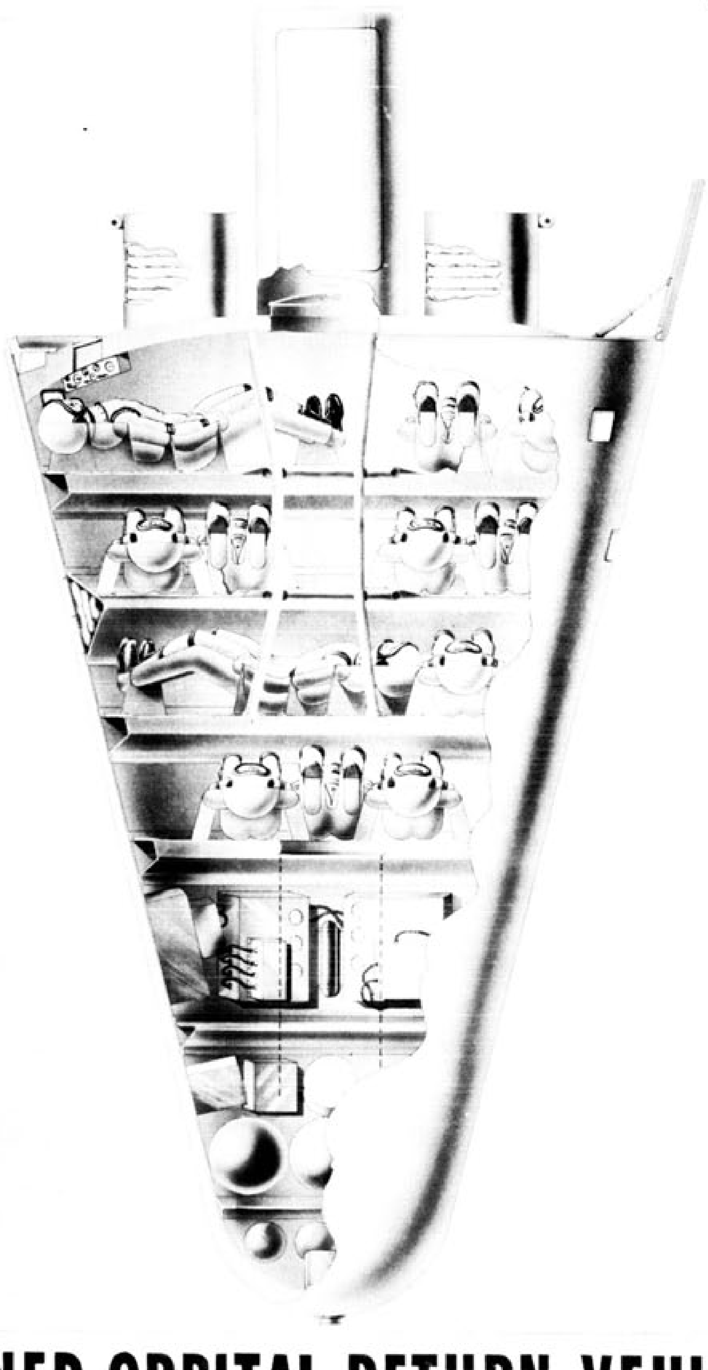

FIG. V.5

MANNED ORBITAL RETURN VEHICLE

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[redacted]

to be a variable lift-drag ballistic vehicle. Although it will not provide the desired re-entry maneuverability it is considered adequate to compensate for possible re-entry dispersions. Such a vehicle weighing approximately 18,000 pounds with an additional 6,000 pounds of propulsion system could be carried into orbit by a SATURN B-1 with an additional 11,000 pounds of cargo for life support essentials or equipment. The vehicle illustrated in Figures V.4 and V.5 also includes: seating capacity for up to sixteen personnel, escape rockets in case of vehicle malfunction during ascent, air lock for space entrance and exit (which would be removed before re-entry) drag flaps for re-entry maneuvering, parachutes for final deceleration and water impact, and a flotation tube for additional buoyancy and stability. The capsule will float in the upright position even without the flotation tube.

(2) Orbital refueling vehicle. The orbital refueling vehicle will be required to transport H_2 and O_2 into orbit, provide at least short time storage, and contain pumping and metering equipment which will operate in a zero "g" environment. No details are presently available on such a system; however, studies and designs are presently in progress at this agency. The orbital refueling scheme is shown in Figure V.6.

(3) Direct lunar landing vehicle. Although no personnel will be carried to the moon early in the program, it is very desirable to provide such a capability for cargo, especially since the total flight time is much shorter and for emergency purposes time may be the most important parameter. Figure V.7 shows a lunar landing vehicle.

(4) Orbit-launched lunar vehicle. The orbit-launched lunar vehicle could be used for either cargo or personnel (See Figure V.8). The basic vehicle is comprised of a tandem stage arrangement using two high energy ($H_2 + O_2$) stages for cargo missions. The first stage would provide the impulse from orbit velocities to escape, and the second stage, with controllable thrust engines, for mid-course correction and lunar landing (See Figure V.9 for the landing vehicle). For manned application the 50,000 pounds of payload capability would be utilized for

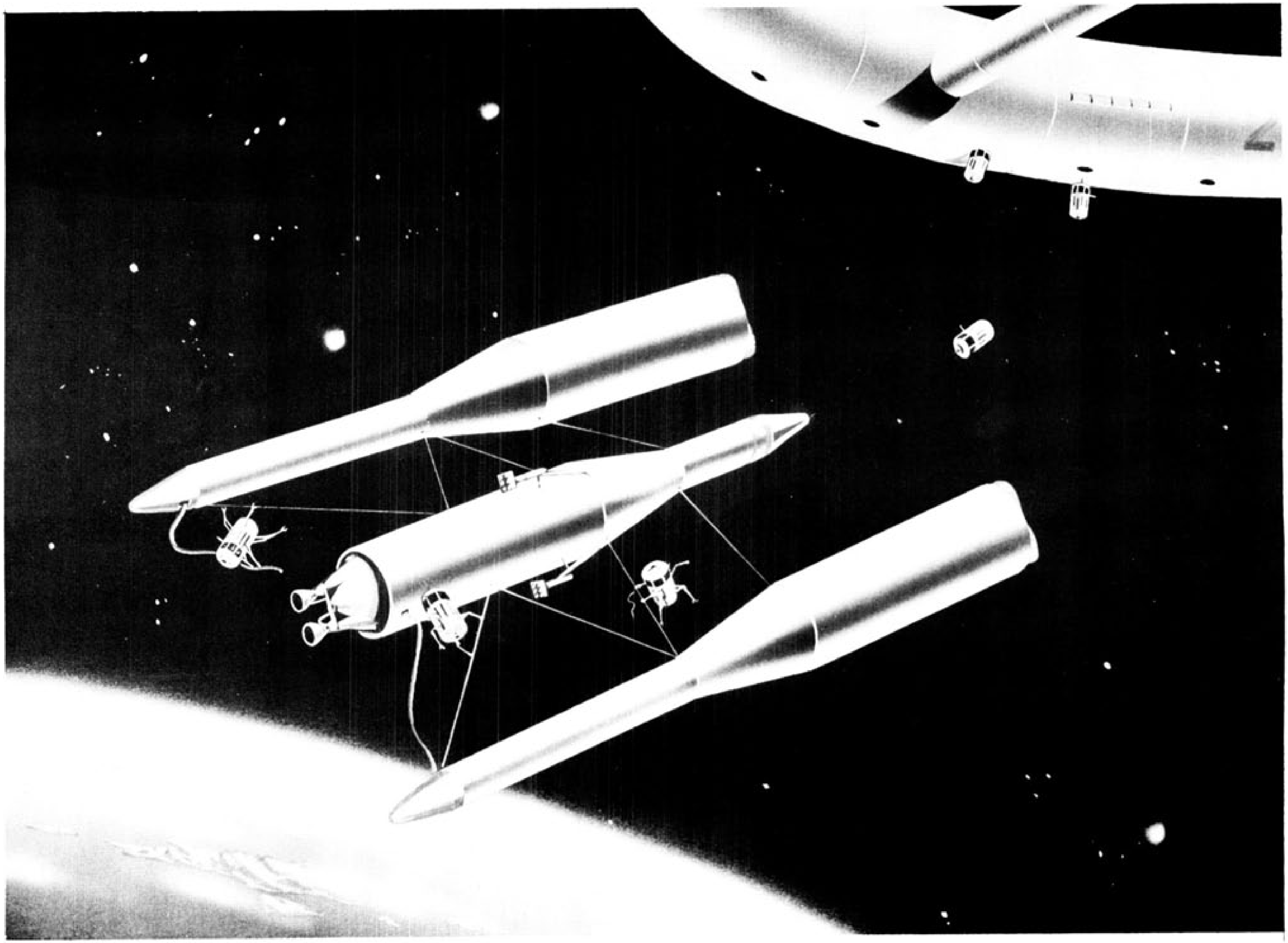
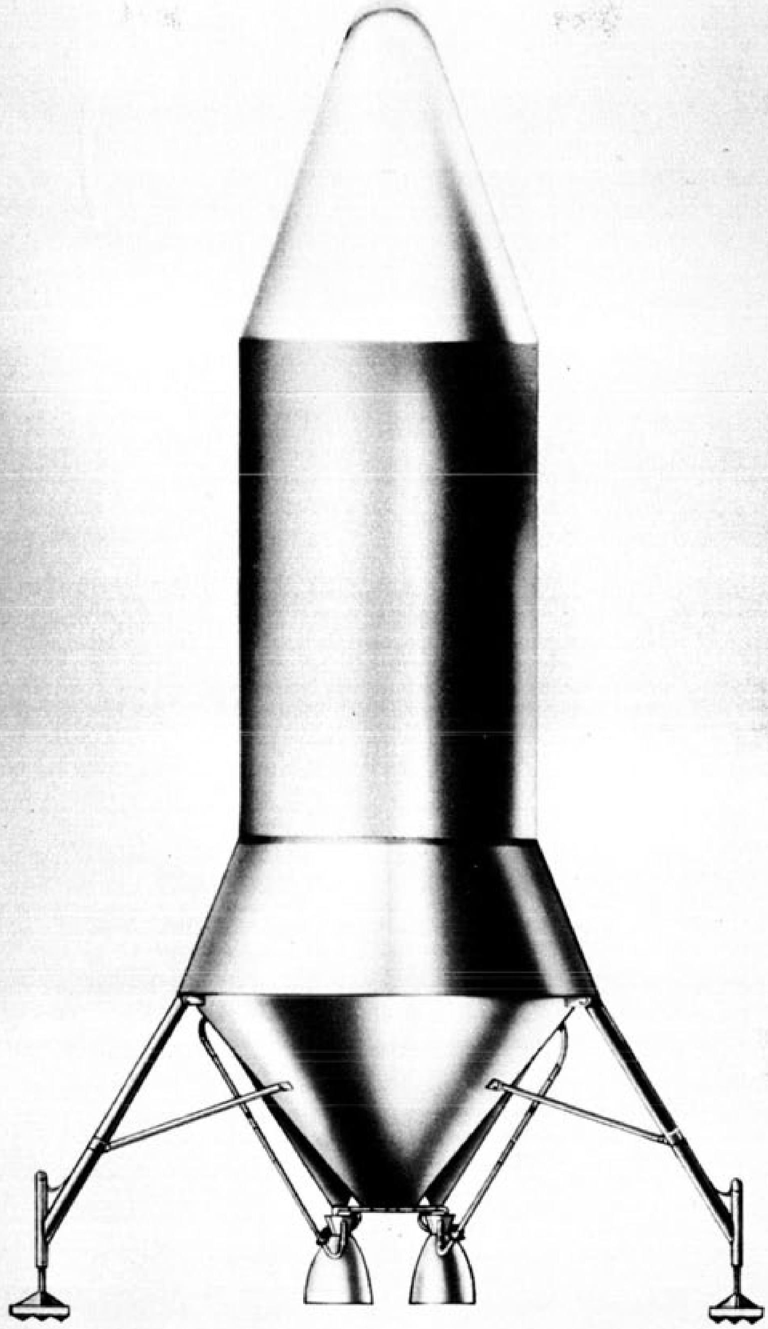


FIG. V. 6 ORBITAL REFUELING SCHEME



LUNAR LANDING
Vehicle

FIG. V.7

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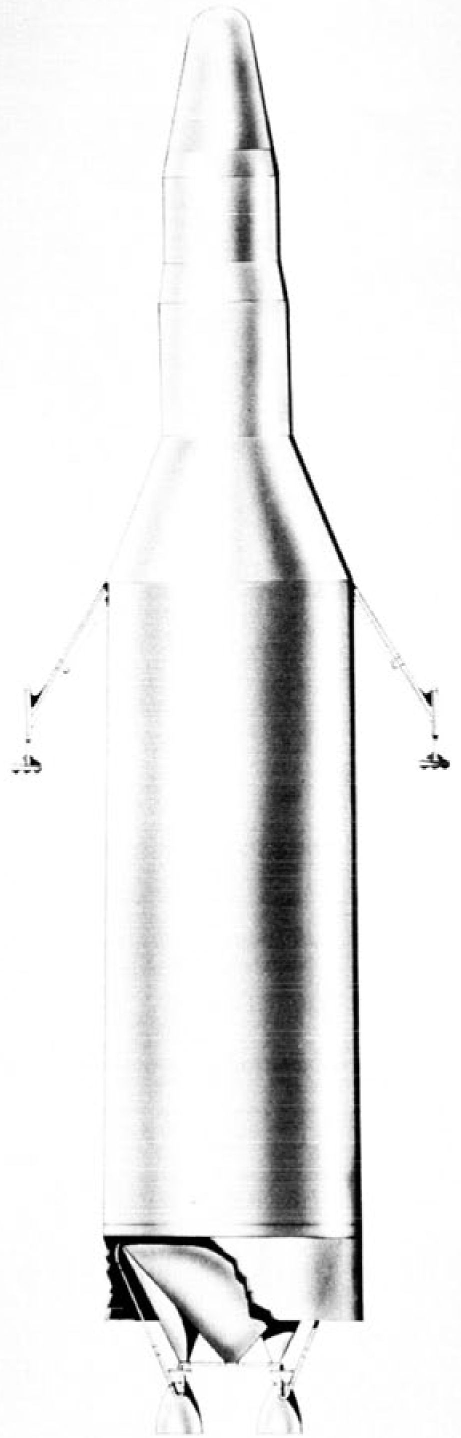

ORBITAL LAUNCHED
Lunar Return Vehicle

FIG. V.8

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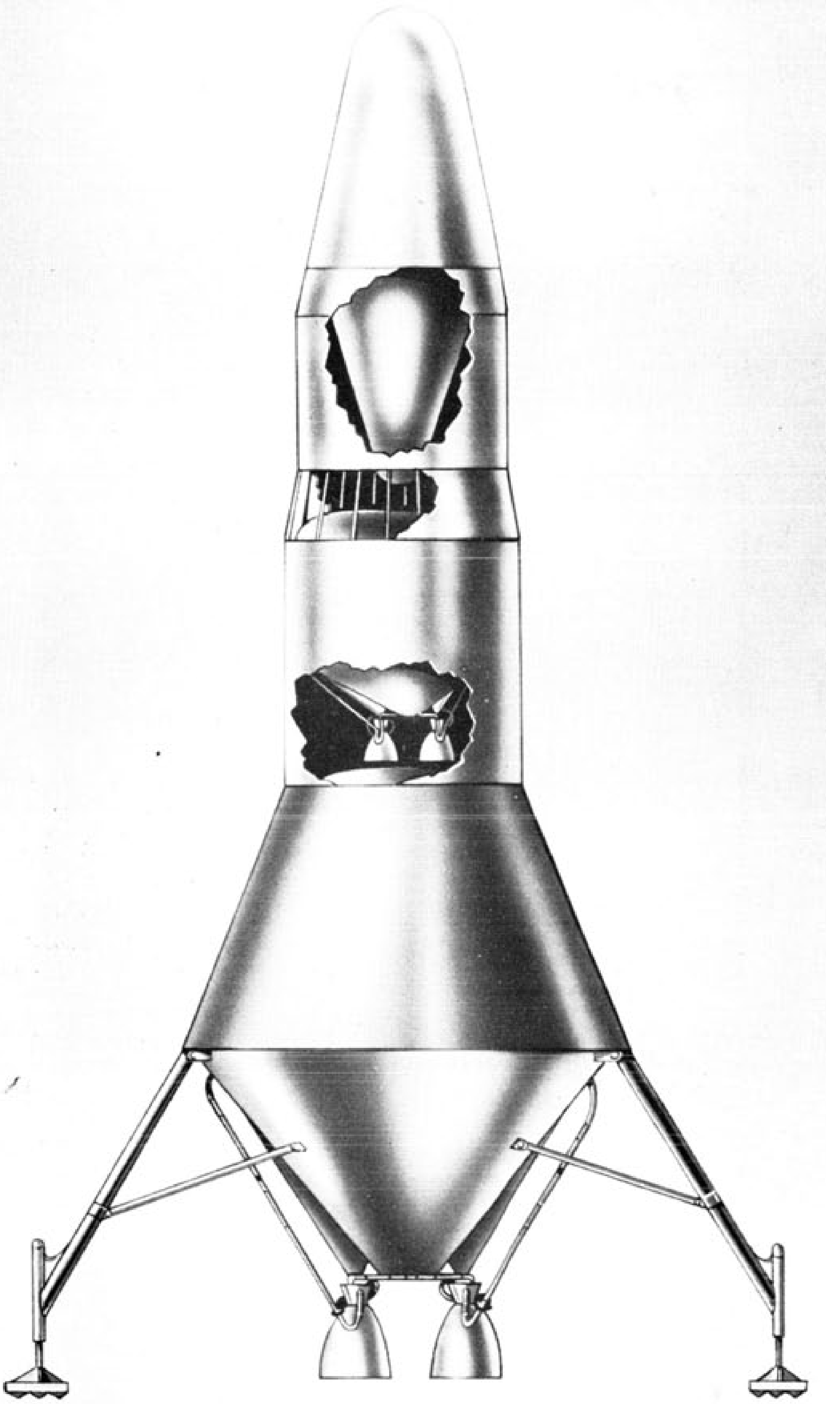

LUNAR LANDING *Vehicle*

FIG. V.9

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a two to four-man earth-return vehicle as shown in Figure V.10. A weight and thrust summary of the complete vehicle is given in Table V.1. Such an orbit-launched vehicle could be transported into orbit by one SATURN C. The third stage of the carrier vehicle would be used as the first stage of the orbit-departing vehicle. The third stage tanks would be emptied during ascent and would be refilled along with the empty lunar landing stage by later tanker flights. Several of the components as well as many of the schemes for this vehicle will have been developed by the time it is ready for use. For example, if the third stage of the SATURN C is required for orbit departure, it would be available and proven; mid-course and terminal guidance would have been developed from earlier unmanned landings; the return capsule would have been used for earlier manned lunar circumnavigation flights; and the landing propulsion system could possibly be a multiple arrangement of earlier direct landing vehicles. In addition, the lunar departing stage would be completely tested on earth before delivery to the moon. As mentioned earlier, considerable cargo will probably be required on the moon regardless of the magnitude of the program. Therefore, the first orbit-departing vehicle probably will be devoted to cargo transportation, rather than to a manned mission.

In addition to the various vehicles (or components) of the space transportation system, the following development areas will require considerable effort:

(1) Orbital rendezvousing

(2) Orbital refueling

(3) Manned survival on lunar surface

(4) Extended storage of O_2 and H_2 in space environment

(5) Complete earth - lunar communication system

(6) Preparation, packaging, and storing of life support essentials for lunar crews.

(7) Selection and training of orbit and lunar crews

FIG. V. 10

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TABLE V.1

Weight and Thrust Summary for
Orbit-Departing Lunar Vehicle

Thrust to leave orbit (H_2 + O_2 engine)	200,000 lbs
Weight from orbit	400,000 lbs
Cutoff weight of orbit departing stage (1st)	160,000 lbs
Stage weight of 1st stage (engine, tankage, etc)	20,000 lbs
Ignition weight of landing stage (2d)	140,000 lbs
Cutoff weight of 2d stage	60,000 lbs
Stage weight of 2d stage	9,000 lbs
Thrust of 2d stage (H_2 + O_2 engine)	100,000 lbs
Gross payload landed on moon (for cargo)	51,000 lbs
Lunar earth-return ignition weight (3d stage)	46,500 lbs
Cutoff weight of 3d stage	14,000 lbs
Thrust of 3d stage (storable propellant)	40,000 lbs
Gross payload of 3d stage (manned re-entry capsule)	8,000 lbs

V.4 MANNED LUNAR CAPSULE RECOVERY (U)

A speedy, reliable and efficient system for recovery of the manned lunar capsule from the ocean after return from a lunar circumnavigation flight is required. The utmost speed and reliability are absolutely essential in this operation. The "loss of the horse for lack of a nail" would be intolerable at this point. The proposed equipment and techniques are based on experience gained through use of present state of the art equipment in the actual recovery of the JUPITER and JUPITER C nose cones. The equipment presently aboard typical Navy rescue and/or fleet tug vessels was originally designed for salvage of large stranded vessels and submarines and is not of optimum design for retrieving comparatively small nose cones or capsules. The proposed system of recovery offers what is believed to be the most infallible method of capsule recovery over the widest range of sea conditions and capsule configurations. The system may be readily optimized for any given capsule configuration from 120 to 160 inches in diameter, from 16 to 25 feet in length, and from 3 to 6 tons in weight. This system utilizes a hydraulically controlled crane together with specially designed retrieving and handling devices.

Based on past experience in nose cone recovery and an evaluation of the advantages and disadvantages of the present equipment and techniques, the following general characteristics and requirements for a manned lunar capsule recovery system may be stated:

(1) Fail-safe method of tracking and spotting the descending capsule.

(2) Rapid transport of retrieving equipment to the point of impact.

(3) Speedy deployment of retrieving equipment.

(4) Versatile equipment for operation under extreme variations in sea conditions.

(5) Fail-safe method of securing the floating capsule.

(6) Sensitive and rapidly reacting equipment controls with the utmost reliability.

(7) Short coupling between recovered capsule and retrieving equipment.

(8) Speed and simplicity in equipment operation.

In the present state of the art, the precise point of impact of a returning capsule cannot be controlled. Therefore, extensive fleet deployment in the general impact area may be necessary. The deployed fleet consisting of recovery and auxiliary ships must be equipped with both radar and visual detection equipment and augmented by search aircraft. Predictable impact accuracy will determine the number of ships in the fleet that must be equipped with recovery equipment.

The actual recovery ships must be fast and highly maneuverable. Ships of the new destroyer class seem to best meet these requirements in addition to having sufficient seaworthiness to navigate and operate in any sea condition in which a recovery could conceivably be attempted. This class of ships and the general arrangement of recovery equipment on board is shown in Figures V.11 and V.12.

Each recovery ship should be provided with two teams of frogmen equipped to perform underwater inspection and to attach special retrieving gear when necessary. The equipment for each of these teams should include a motorized rubber life raft and reliable two-way radio communication with the mother ship. The diving team must be trained to remove the parachutes, marker buoys and any other impedimenta which might interfere with recovery operations. During actual retrieving operations under normal conditions the frogmen are not required to perform any tasks involving physical contact with the capsule or equipment. This is an important safety consideration, especially when operations must be performed in rough sea conditions.

The proposed lifting device for removal of the capsule from the water and placing it on the ship's deck is a hydraulic crane of the Bucyrus-Erie type as shown in Figure V.12.

A hoop net is proposed as the retrieving device to be used in conjunction with the hydro-crane (See Figure V.13). The hoop net is cylindrical in shape and should have a diameter and length approximately four feet greater than the capsule to be recovered. Determination of final dimensions will depend upon the capsule dimensions and whether or not it will float on its side or nose. Dimensions of the nets may be varied for

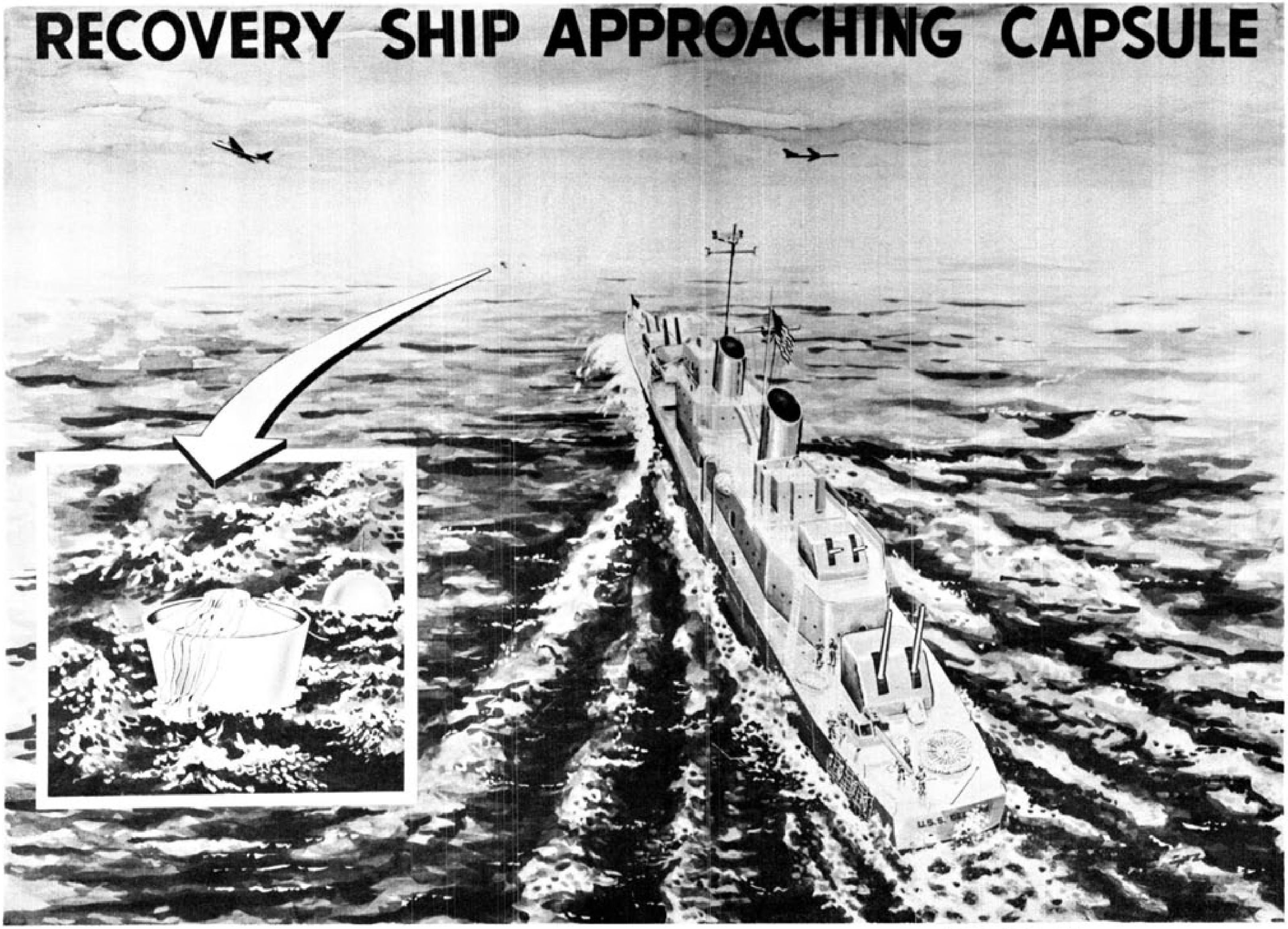


FIG. V. 11

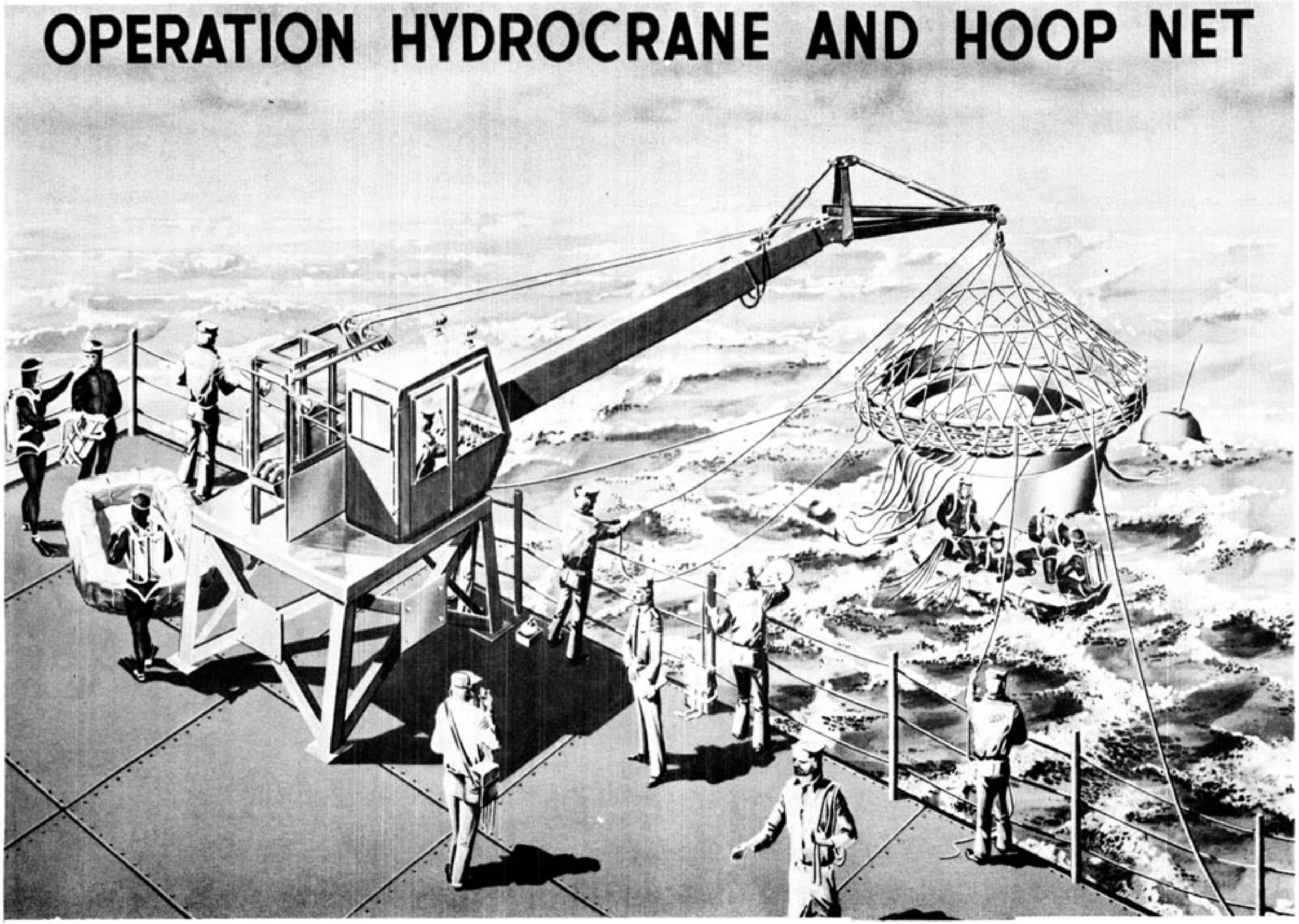
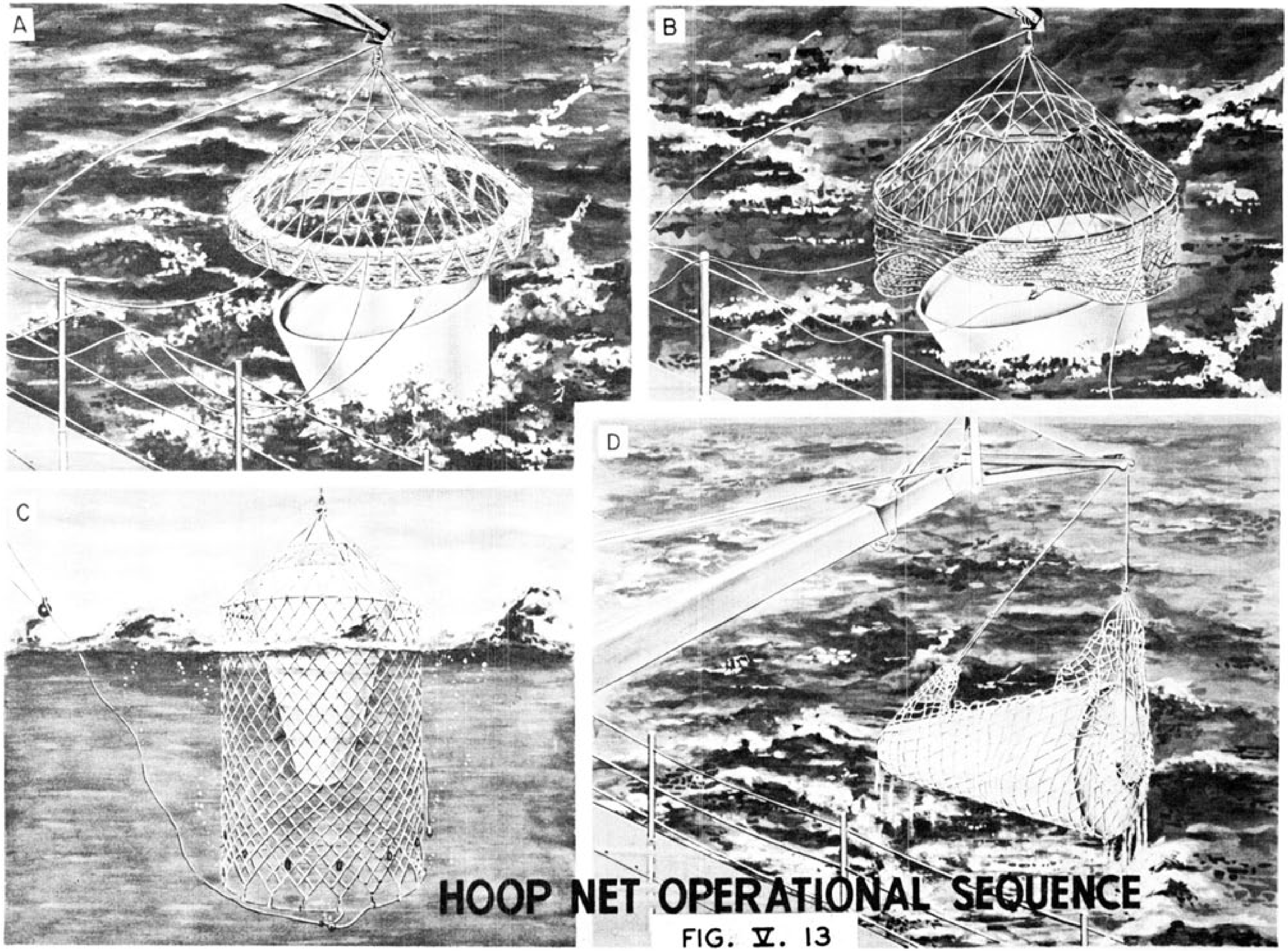


FIG. V. 12



HOOP NET OPERATIONAL SEQUENCE

FIG. V. 13

recovery of different sized capsules but operating characteristics will remain the same. The net may be constructed of any fibrous material, such as hemp or nylon, with sufficient tensile strengh to carry the load of the particular capsule to be recovered. The net is held in shape and given body by incorporation of several collapsible elastic hoops along its length. When extended, these hoops hold the net in shape while it is lowered over the capsule. Weights or sinkers placed near the bottom of the net cause it to sink rapidly over the capsule. The bottom or open end of the net will be equipped with several cable clinches with unidirectional action through which a gathering or closure cable will be operated to effect closure of the net. The closure cable will be attached to a boom line which may be temporarily secured aboard ship during the lowering operation. After the net has been lowered over the capsule, a weight of about 100 pounds attached to a pulley is placed on the boom line and allowed to run down the cable and effect closure of the net by its inertia. The weight is then pulled back up the cable by a retrieving hand line and detached from the cable. With the capsule thus secured in the net, the crane operator may remove all slack from the line connected to the closure cable and lower the boom to reduce the distance between boom tip and cargo to effect a short coupling which, together with jib boom design, will minimize the pendulum and gyrating effects of the capsule and net. The capsule and hoop net are then hoisted aboard the ship with the capsule in approximately a horizontal position. This operation is illustrated in Figures V.12 and V.13. The hoop net fulfills the stated requirements for speed of deployment and operation, versatility, reliability, and fail-safe method of securing the capsule. In addition, it has operational capabilities over a wide range of sea and weather conditions as well as capsule conditions such as distortion and severe structural damage.

Normal operational sequence for the proposed recovery system is outlined as follows:

(1) Location of capsule

(2) Nearest recovery ship proceeds to point of impact (Figure V.11)

(3) Deployment of frogman team (Figure V.12)

(4) Removal of parachutes and flotation buoy

(5) Positioning of recovery ship and hoop net (Figure V.13A)

(6) Release of net (Figure V.13B)

(7) Closure of net (Figure V.13C)

(8) Retrieval of capsule (Figure V.13D)

CHAPTER VI

POWER SUPPLIES

VI.1 CRITERIA FOR SELECTION OF POWER SYSTEMS

The selection of an optimum power system for each phase and sub-phase of the lunar surface exploration program has been based on the specific mission demands such as:

(1) Requirements for primary and/or secondary (storage) system

(2) Required peak and average power capacity

(3) Required operational time

(4) Anticipated environmental conditions

(5) Weight and volume limitations

(6) Degree of reliability and simplicity required

(7) Required safety and economy

In view of the above requirements, the proposed power systems have been selected after comparison of the merits and disadvantages of the following possible competitive systems:

(1) Primary and secondary electro-chemical storage cells

(2) Open cycle and solar regenerative fuel cells

(3) Open cycle and solar regenerative turbo-electric units

(4) Photovoltaic converters

(5) Thermoelectric and thermionic converters

(6) Nuclear energy conversion systems

For each application within the program the foregoing systems were rated for:

(1) Reliability and ruggedness under the imposed environmental conditions

(2) Compatibility with mission requirements (such as weight, volume, etc.)

(3) Development state of the art and availability

(4) Safety and economical limitations

Nuclear energized power systems were eliminated as a possibility for the lunar surface exploration program because:

(1) They would render sensitive on-board radiation measurements extremely difficult, if not impossible.

(2) The weight penalties to provide adequate protection of personnel, in addition to the required complex control systems, appeared restrictive at this time.

(3) Operational control, agreements, allocation of Atomic Energy Commission controlled materials for test and for operational units, and extensive personnel training would have to precede the use of any nuclear system.

(4) The present cost of nuclear materials alone would far exceed the cost of the proposed systems for this program.

(5) It is undesirable to contaminate the surface of the moon with radioactive material.

VI.2 AUXILIARY POWER FOR SATURN INJECTION VEHICLE AND CIRCUMLUNAR VEHICLE

SATURN Stages I, II, and III. An analysis of all present ballistic missile programs has shown that auxiliary power is furnished by either a high rate battery or by a hot gas turbogenerator. These two systems offer a distinct weight advantage over other systems for the high rate, short duration power requirements of ballistic missile boosters. Although the weight and volume limitations for units in the first stages are not so critical, certain environmental conditions may be most severe during this interval.

Since the success of each mission hinges entirely on the performance of the power system, especially during the initial boost stages, extreme reliability and ruggedness are mandatory requirements. The zinc-silver battery system represents the most highly developed power unit to date for this application. Because it is rugged and requires no moving parts, it is estimated that such a system has attained a 99.95% reliability status as compared to about 96% for the nearest competitive system. The extremely high reliability of the battery system is verified by the fact that no failures in this system have occurred during the many firings of the REDSTONE and JUPITER programs.

As shown in Figure VI.1 for short duration booster applications up to 20 minutes operating time, the zinc-silver battery system is competitive with the solid propellant auxiliary power unit system. The energy to weight comparison in Figure VI.1 is based on a typical 5-kilowatt system. This applies to Stages I, II, and III, since the elapsed time from take-off to cut-off of the third stage is about 12 minutes, and the estimated power requirements of the three stages are as follows:

SATURN Stage I	6100 watts
SATURN Stage II	2500 watts
SATURN Stage III	2500 watts

It is not foreseen that any appreciable development costs will be incurred in providing batteries since a complete line of zinc-silver batteries are available that have been flight tested. On the other hand, the rather high development cost necessary to provide a turbogenerator system for this specific application does not seem warranted. Therefore, it is planned that each of the first three stages of the booster vehicle will carry its own zinc-silver battery to fulfill its power requirements independent of the other stages. The discharged battery is dropped with the expended stage allowing maximum vehicle performance. This plan also simplifies the power distribution, decreases cable weights and avoids the necessity of breaking power cables at each separation.

SATURN Stage IV for Landing Package. The guidance scheme for landing the stationary packet and roving vehicle provides that the fourth stage of the SATURN booster will be separated after injection. The fourth stage power requirements are limited to approximately 15 minutes of actual operation at an estimated level of 1800 watts.

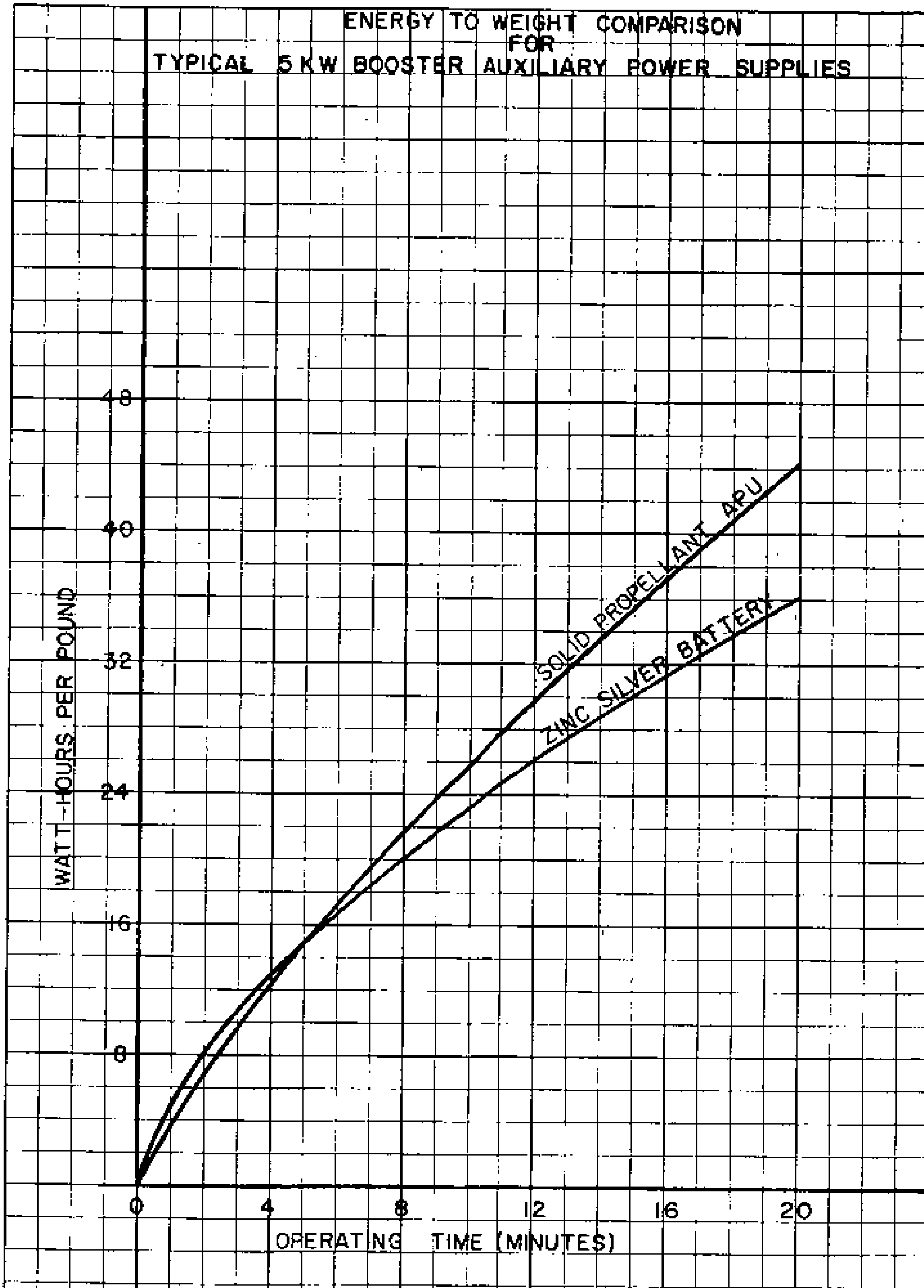

ENERGY TO WEIGHT COMPARISON
FOR
TYPICAL 5 KW BOOSTER AUXILIARY POWER SUPPLIES
WATT-HOURS PER POUND
48
40
32
24
16
8
0
4
8
12
16
20
OPERATING TIME (MINUTES)
SOLID PROPELLANT APU
ZINC SILVER BATTERY

FIG. VI. 1

For the reasons stated previously, it is proposed that a zinc-silver battery be used in this stage.

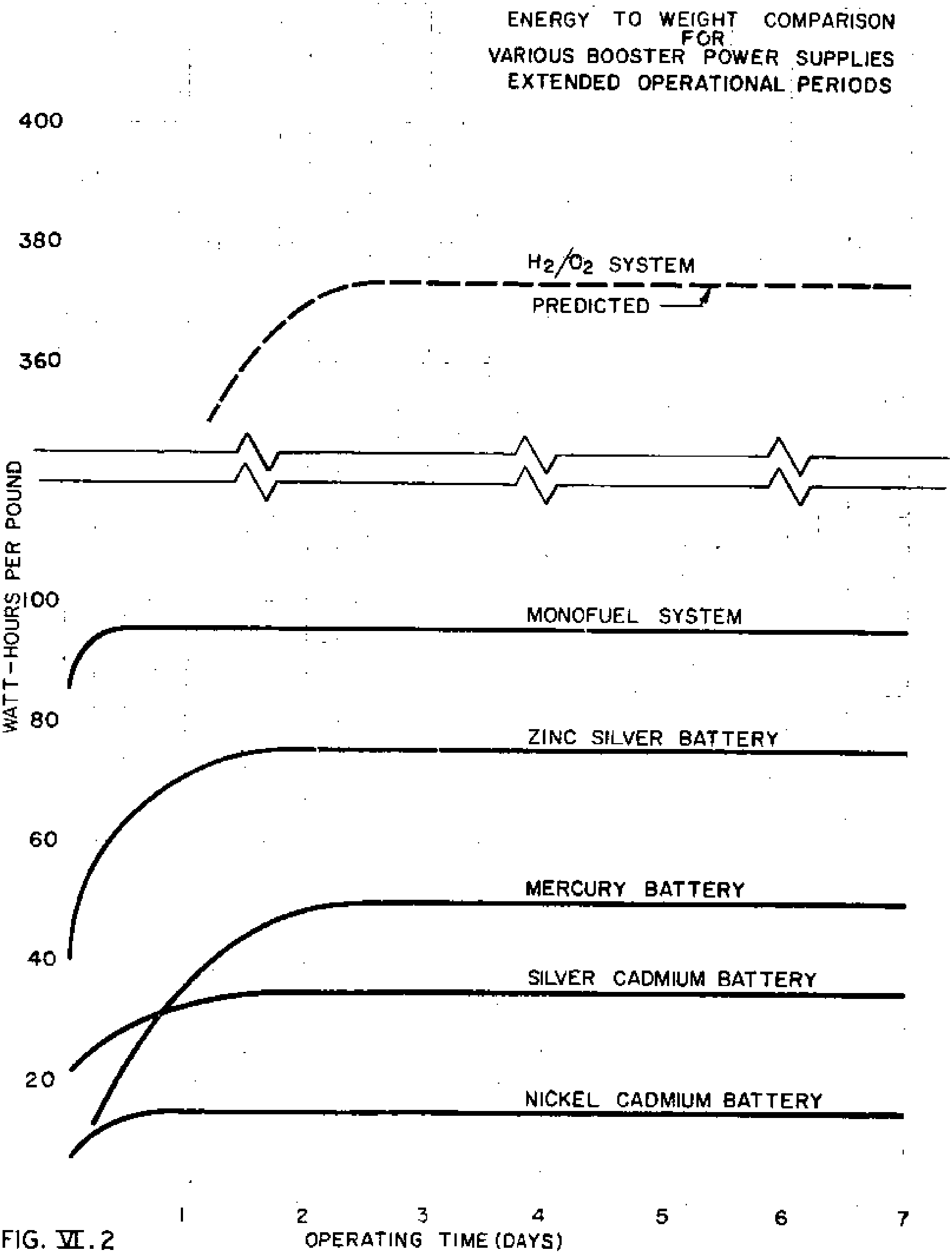
SATURN Stage IV and Circumlunar Vehicle. As presently planned, the fourth stage will separate from the circumlunar vehicle two to ten hours after take-off. The circumlunar vehicle is expected to require seven days to complete its flight. Although the first two flights will be unmanned, this basic vehicle will ultimately be used for manned flights. Therefore, the initial design should provide for the requirements of a two-man crew.

During part of its seven-day flight, this vehicle will be in the shadow of the earth or the moon. Therefore, any auxiliary power system which depends on solar energy would have to include a storage system for operation during the shadow periods. Seven days is a rather short operating time for a solar powered system and the weight of such a system becomes comparable to that of other systems. The necessity for maintaining the solar collector in a fixed position with respect to the sun adds to the complexity of the vehicle and is, therefore, a disadvantage.

A manned vehicle requires attitude control, CO_2 and water removal, a cooling system, and electrical power. Studies have shown that on the basis of electrical power requirements alone, a liquid hydrogen/oxygen open cycle turbine system offers a weight advantage over battery systems and over other open cycle systems (see Figure VI.2). In addition, with a modest increase in weight, the same system can provide the power for attitude control, CO_2 and water removal, and cooling. It is expected that the problem of storage of H_2 and O_2 in a space environment will be solved by the time of these flights.

The electrical load is expected to be about 500 watts in addition to the other loads mentioned above.

The fourth stage energy requirements for guidance and control equipment, tracking, telemetering and cooling is estimated to be 10,800 watt-hours for electrical and 6,000 watt-hours for cooling. It is planned to supply this load with the power supply aboard the circumlunar vehicle. The estimated load breakdown on this power supply is tabulated as follows:



ENERGY TO WEIGHT COMPARISON FOR VARIOUS BOOSTER POWER SUPPLIES EXTENDED OPERATIONAL PERIODS
400
380
360
100
80
60
40
20
WATT-HOURS PER POUND
H_2/O_2 SYSTEM
PREDICTED
MONOFUEL SYSTEM
ZINC SILVER BATTERY
MERCURY BATTERY
SILVER CADMIUM BATTERY
NICKEL CADMIUM BATTERY
1
2
3
4
5
6
7
OPERATING TIME (DAYS)

FIG. VI.2

Stage IV (To Injection)	Requirement		Duration	Watt-Hours
	Electrical			10,800
	Cooling			6,000
Circumlunar Vehicle				
	Electrical	500 w	7 days	84,000
	Cooling	550 w	7 days	92,400
			TOTAL	193,200

The best estimates from industry indicate that with rather extensive development, a hydrogen/oxygen turbine system can be built that will weigh 1.9 pounds per horse power-hour. On this basis, a system that will meet the above requirements will weigh 492 pounds.

It is estimated that this system can provide attitude control and CO_2 and water removal with an increase in weight of 115 pounds. In the event that the circumlunar vehicle uses liquid hydrogen/oxygen for propulsion a substantial weight saving can be realized by combining the fuel storage for propulsion and auxiliary power systems.

The primary reason for the selection of the hydrogen/oxygen system is its weight advantage brought about by the low specific fuel consumption as compared to other high energy fuels. A comparison of the minimum attainable specific fuel consumption of various fuels is tabulated below:

Fuel	Minimum SFC (lb/hp-hr)
Ethylene Oxide	5.5
Hydrazine	4.0
Gasoline/Oxygen	3.0
Hydrogen/Oxygen	1.0

Braking Stage for Landing Package. It is not anticipated that the braking stage will require any appreciable amount of auxiliary power. Since it separates from the lunar vehicle only a few seconds before impact on the lunar surface, it is proposed that the small requirements for engine and attitude control be supplied from the vehicle auxiliary power system.

VI.3 LUNAR STATIONARY PACKET POWER SUPPLY

After separation from the fourth stage, the landing vehicle, including the braking stage, will travel for about 58 hours before landing on the lunar surface. The guidance and control equipment and instruments to be carried aboard is shown on the network block diagram of the stationary packet in Figure VI.3.

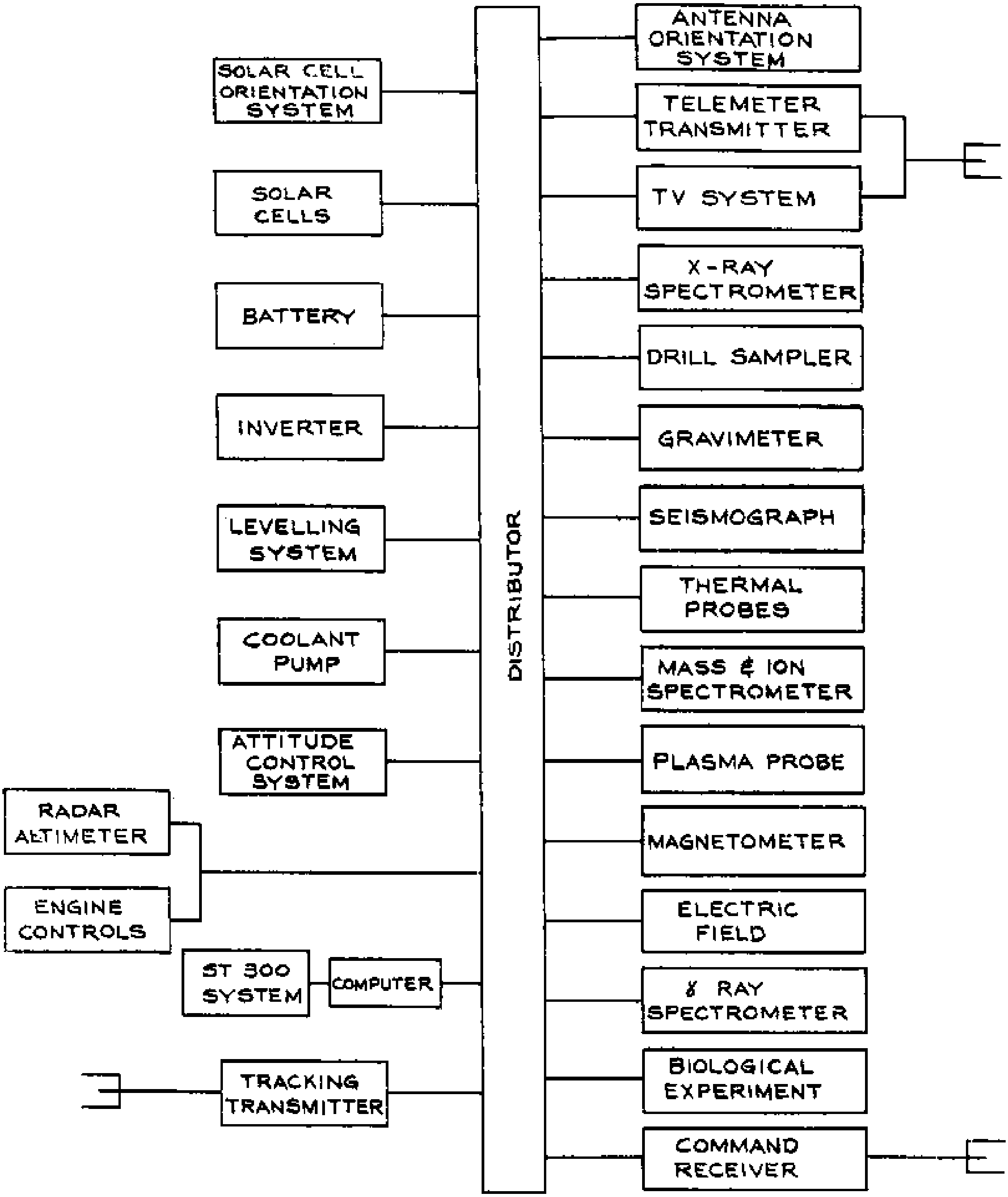
The ST-300 stabilized platform will operate for a maximum of ten hours after fourth stage separation, and for about thirty minutes before landing at an estimated power level of 150 watts.

The only equipment that will be operating for the entire flight is the attitude control system, and the tracking and telemetry systems. The average power required for this equipment is estimated to be 120 watts. During the period just preceding the landing the TV system will operate, requiring about 100 watts of power.

In addition to the TV system, the radio altimeter will be operating at a power level of about 100 watts. The total energy requirements during flight are listed below:

10 hours at 270 watts	=	2700 watt-hours
48 hours at 120 watts	=	5760 watt-hours
0.5 hours at 470 watts	=	235 watt-hours
Total		8695 watt-hours

Immediately after landing the packet must go through an erection phase before the solar powered system goes into operation. This phase will consist of antenna orientation, solar collector orientation, leveling of the packet, initial sequencing of instrumentation, and command functions initiated from the earth. Power required for this phase is not expected to exceed 400 watts for 15 minutes. This raises the total energy required prior to operation of the primary power source to 8795 watt-hours.



NETWORK BLOCK DIAGRAM OF STATIONARY PACKET

FIG VI. 3

An H_2O_2 system was considered but has the following disadvantages:

(1) Power requirements are reduced to a level where optimum energy to weight ratios cannot be realized.

(2) Exhaust from an open cycle system may contaminate the lunar surface.

(3) An additional energy storage system is still required for lunar night operation, thereby complicating power distribution and further derating the over-all system energy to weight ratio.

(4) Possible hazard due to landing the vehicle with fuel tanks aboard.

The low power level demand for stored energy during the in-flight and lunar night phases make a battery system to provide both requirements a logical choice.

The estimated continuous power required for the scientific instrumentation, telemetering transmitter, command receivers, and circulating pump for the cooling system is 35 watts. In addition, there are instruments that are operated by command from the earth or that are programmed in a predetermined sequence. The largest electrical load is presented by the drill sampler which is operated by command and requires 200 watts. The total expected operating time of the drill is five hours. The TV system, also operating on command, requires 100 to 200 watts for a few seconds for each transmission. Although the TV system will be operated many times, the total operating time is short, being on the order of five or ten minutes. Other instruments require from one to thirty watts intermittently. The average power required is estimated to be fifty watts for the lunar day operation.

During two lunar nights it is desirable to operate a part of the scientific equipment within the limits of the available power. In addition, it is necessary to heat the equipment that is sensitive to low temperatures.

Studies performed in connection with this and other projects indicate that the most reliable system for meeting these requirements is composed of photovoltaic converters and rechargeable zinc-silver oxide batteries.

Tests have shown that within the temperature limits of the instrument compartment of -10° C to +60° C the zinc-silver oxide battery yields an energy to weight ratio far above that of any other available battery. At low discharge rates, mercury batteries can deliver up to 50 watt-hours per pound at room temperature as shown in Figure VI.2. However, at the lower temperature limit of -10° C the output is down to 15 to 20 watt-hours per pound. On the other hand, recent tests indicate that selected zinc-silver oxide cells can operate at much lower temperatures without appreciable depreciation of efficiency. This is illustrated in Figure VI.4 where typical cells were discharged at reduced temperatures and delivered 87.6 watt-hours per pound.

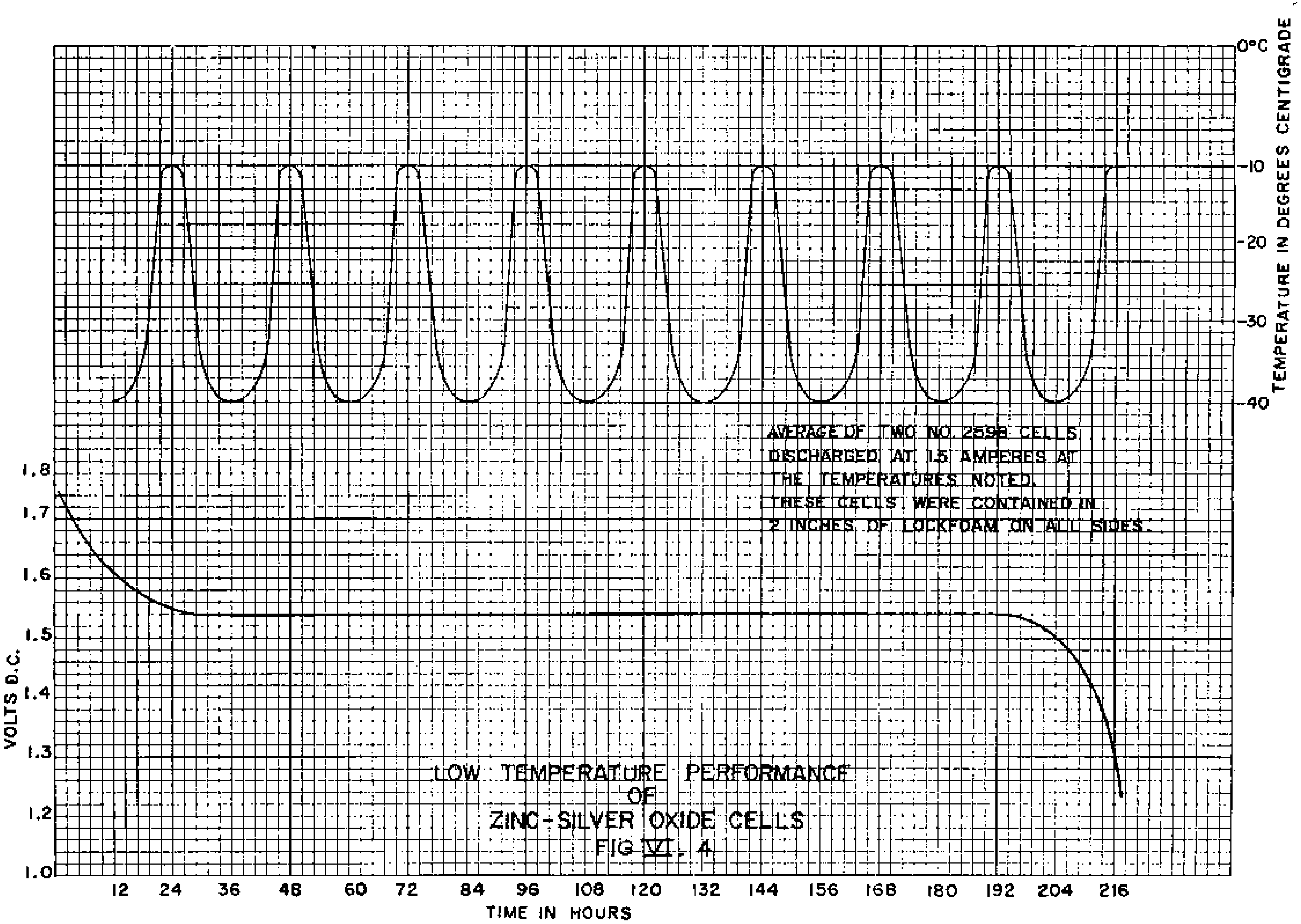
This cell is capable of delivering high peak demands of a hundred times its average rate for this requirement with only a slight decrease in its energy to weight ratio. Furthermore, it can be fully charged and discharged about five cycles and deliver 90 per cent of its initial capacity.

It is anticipated that the battery will be sealed with an automatic pressure control for operation in a vacuum. This pressure seal plus the derating necessary for the charge-discharge cycling is expected to reduce the output to 75 watt-hours per pound.

The in-flight energy requirement was 8795 watt-hours. If we increase this by 10 per cent for reserve capacity, the total required is 9675 watt-hours. Based on 75 watt-hours per pound, the total battery weight is 129 pounds.

By recharging this same battery during the lunar day, it can provide 26 watts of power continuously during the lunar night for heating and instrumentation and yet maintain its 10 per cent reserve. Approximately 130 per cent of the battery discharge capacity must be supplied by the solar energy converter to recharge the system. Therefore, during one lunar day 11,420 watt-hours would be required, necessitating an average battery recharge power capacity of 34 watts from the solar system.

The total power capacity required of the solar energy converter is the sum of the instrumentation load during the day and the battery recharge load which is 50 + 34 = 84 watts. Protection of the solar cells from micrometeorite erosion can be provided by placing thin glass slides directly over each cell. These slides may be optically coated to reject light in the unusable wavelengths and to increase the emissivity of the cells



LOW TEMPERATURE PERFORMANCE
OF
ZINC-SILVER OXIDE CELLS
FIG VI. 4

thereby improving the temperature control of the system. There remains the possibility of cell damage from meteorites so large that glass slides would not afford protection. It is proposed, therefore, that the solar cell system be overdesigned by 20 per cent to allow for such possible damage. This increases the power capacity to 101 watts.

Silicon solar cells with conversion efficiencies of 10 per cent at 30°C appear to be most feasible at this time. However, it is quite possible that high efficiency cells employing other materials, such as gallium-arsenide, may become available for this application. Such an event would enhance the operation at elevated temperatures and reduce the required area and weight of the system.

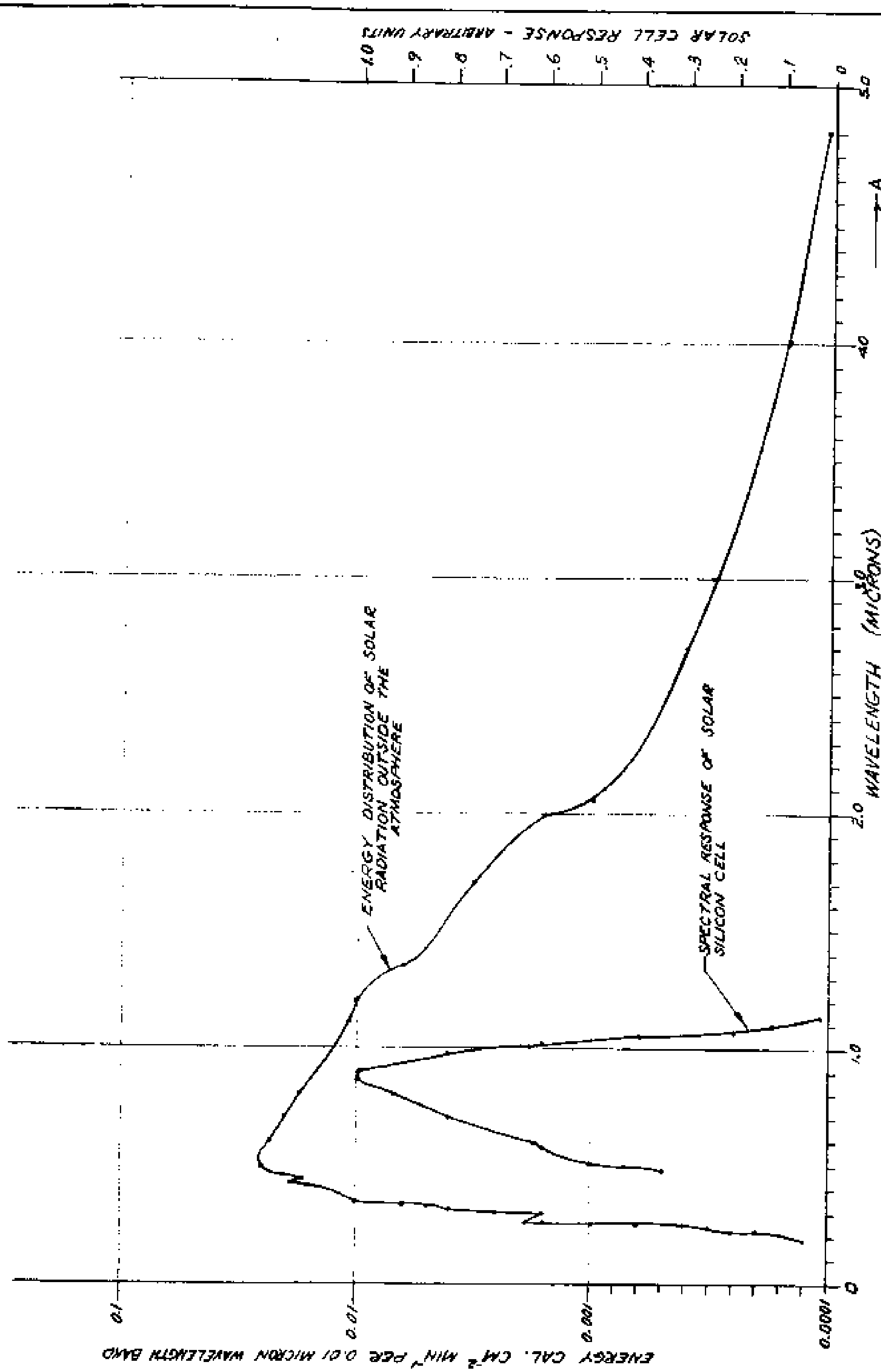
Solar cells have been successfully used for many space applications. The characteristics of static devices plus the experience that has been gained in testing, handling, and mounting solar cells makes it possible to construct a highly reliable power system from these cells. Therefore, a solar cell system was selected for the first lunar surface exploration attempt.

Orientation of the solar cell array with respect to the sun is not overly critical. Since the cell output varies approximately as the cosine of the angle of incidence of the solar radiation, a system that can maintain the solar cell bank within five degrees of a plane that is perpendicular to the sun's rays is completely adequate. A solar aspect detector composed of four silicon cells operating in a closed loop servo system can provide the proper orientation with an estimated power expenditure of 0.5 watts.

With careful radiator design and employment of proper cell coatings, it is expected that the worst solar cell temperature encountered will be about 100°C during lunar day operation. The curve in Figure VI.5, taken from the Smithsonian Tables, shows the energy distribution of solar radiation at various wavelengths and the spectral response of a typical silicon solar cell. It can be seen that the cell is sensitive to a relatively narrow band of the solar spectrum. Optical coatings that are capable of blocking the wavelengths that are not utilized by the cell and that can increase the cell emissivity in the far infrared region can drastically lower the equilibrium temperature of the cell. Development work sponsored by ABMA shows promise of this objective being achieved in the near future.

FIG VI.5

SPECTRAL DISTRIBUTION OF SOLAR ENERGY AND SPECTRAL RESPONSE OF SILICON SOLAR CELL



A cell that operates at 10 per cent efficiency at 25° C will convert about 6.2 per cent of the solar energy at 100° C. Blocking diode losses and mismatch of cells will further reduce the conversion efficiency to 5.2 per cent. The conversion of 101 watts of solar energy by a system of 5.2 per cent efficiency requires 15 square feet of active cell area. Allowing for about 20 per cent of the active solar cell area for spacing between the cell strips, the total solar bank area required is 18.8 square feet. The weight of the cells, cement, and mounting tray for this area is approximately 35 pounds. The supporting structure for this area would increase the weight to 70 pounds.

In summary, the total weight of the auxiliary power system to supply power for the stationary packet for two lunar days and nights is 199 pounds.

VI.4 LUNAR ROVING VEHICLE POWER SUPPLY

VI.4.1 System Considerations. In the preliminary study it was considered desirable to use the same power system for the instrumentation of this vehicle as for the stationary packet. It was proposed that energy requirements for vehicle propulsion and drilling be supplied by a turbogenerator fueled with hydrazine.

A more detailed study of the roving vehicle concept has changed the power requirements and operational times considerably. This, in addition to the changes in the in-flight concept has required a complete re-evaluation of the roving vehicle demands. Keeping in mind that more time could be allowed to develop a more nearly optimum power system for this application, the re-evaluation studies included a state of the art survey on more exotic systems. Therefore, it appears quite feasible that the system proposed herein can be developed for reliable performance within the time limitations of the program.

The most advanced system for converting solar energy to usable power, other than solar cell systems, is the closed-cycle, mercury-driven, turbogenerator system. The basic system shown in Figure VI.6 depicts the system selected for the roving vehicle. Working fluids such as sulphur or sodium appear quite attractive from a theoretical efficiency standpoint; however, after considering a great number of possible fluids, it is proposed that a mercury vapor Rankine Cycle system be used.

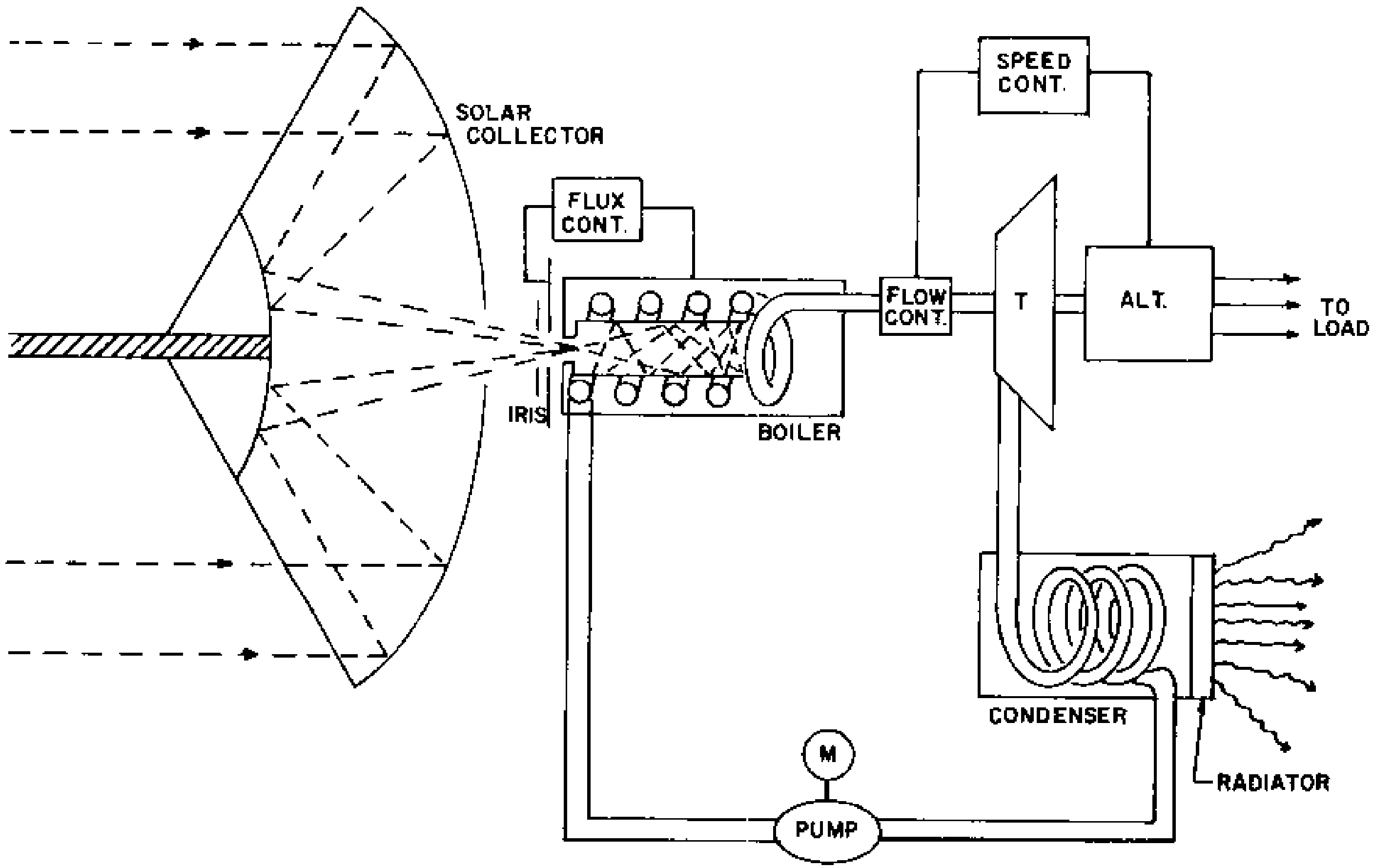


FIG. VI. 6

CLOSED CYCLE SOLAR POWER SUPPLY

An estimate of the electrical requirements of the various components in the vehicle are given in Table VI.1.

Figure VI.7 is a schematic of the power system. During the flight phase, it is planned to operate the telemetry and communications equipment from the battery. A 100-watt static inverter will provide closely regulated 400 cps AC power to the ST-300 stabilized platform. After landing on the lunar surface this inverter will furnish a small amount of power to the instruments. During normal operation, 400 cps AC power will be provided to the drive system and cooling system by the turbo-alternator. This can be rather coarsely regulated, on the order of ± 5 per cent. In addition, the alternator will furnish power to an AC to DC regulated power supply which will feed the DC loads and maintain the battery on charge. However, if a shadow obsures the solar collector, the battery will power a 3-phase static inverter of 850 watts and 400 cps which will furnish power to the drive system and cooling system. The battery will also provide power for communications and telemetry. An estimate of the weight of the power system and network components is given in Table VI.2.

The overall network block diagram of Figure VI.8 depicts the various components and systems composing the electrical system for the roving vehicle.

VI.4.2 Secondary Batteries. Since the capacity of the proposed solar powered turbogenerator system for surface operation of the roving vehicle is greater than that required by the vehicle during its 58-hour flight, consideration was given to the operation of this power system to satisfy the in-flight requirements of the vehicle. However, closer examination of such a scheme reveals some serious disadvantages which are listed below:

(1) The necessity for orientation of the solar collector during flight would impose an additional requirement on the configuration of the roving vehicle and the braking stage.

(2) The gyro effect of the turbine would cause disturbances in the attitude control system of the flight vehicle.

(3) The collector and radiator would have to be folded against the vehicle in order to withstand the landing shock.

TABLE VI.1

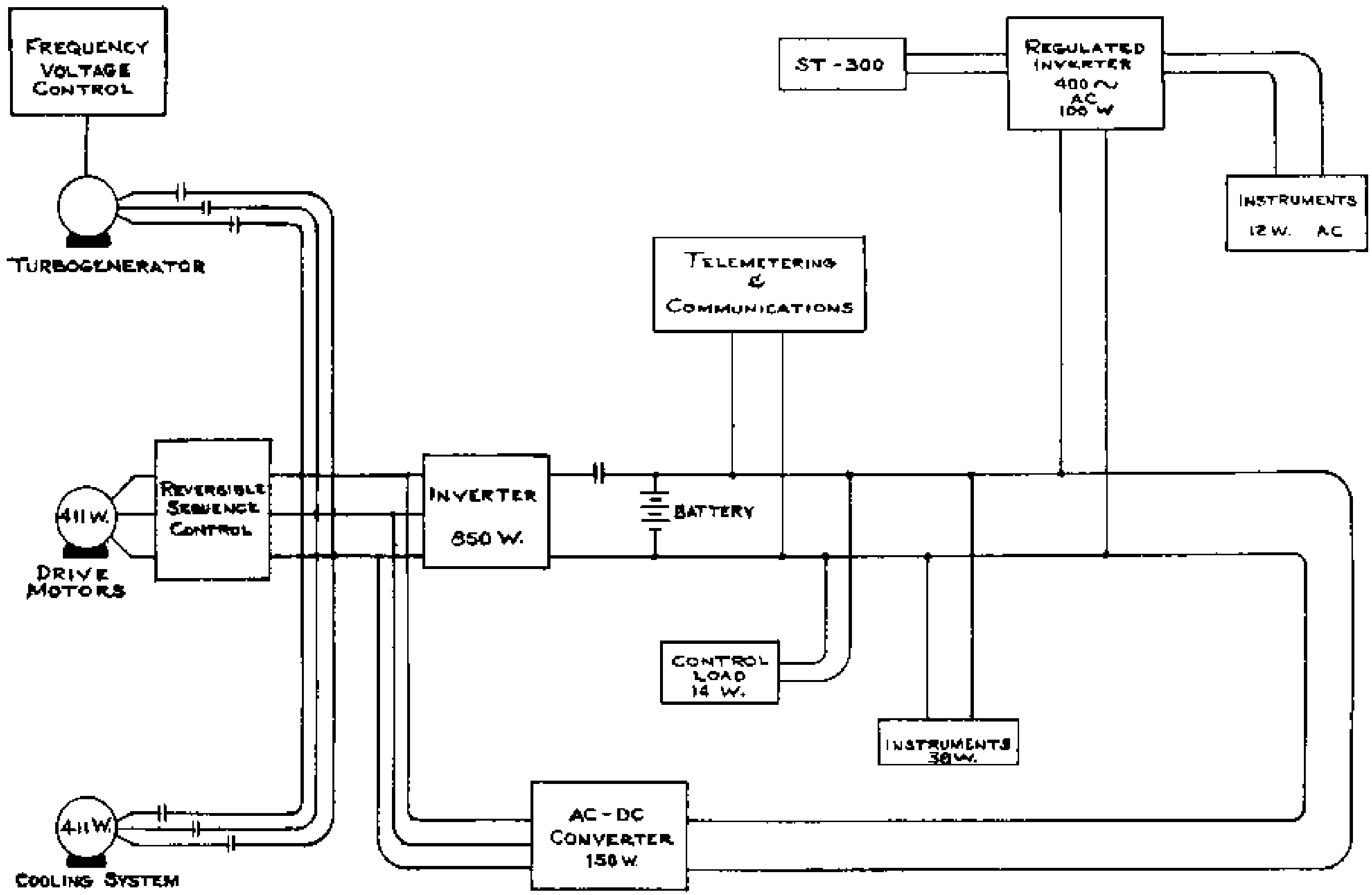
ROVING VEHICLE ELECTRICAL POWER REQUIREMENTS

Precision Regulated AC System -	Output	Input
Load:	AC Watts	DC Watts
Instrumentation	16	
Source: (from DC system)		
400 cps Inverter (70% Eff.)		23

DC System (Regulated) -	Output	Input
Load:	DC Watts	AC Watts
Instrumentation	38	
Vehicle Control	14	
Battery Charge	39	
400 cps Inverter (70% Eff.)	23	
Total	114	
Source: (from Solar Conversion AC System)		
AC to DC Converter (85% Eff.)		134

Solar Energy Conversion System -	Output
Load:	AC Watts
AC to DC converter	134
Cooling System	411
Vehicle Drive System	411
Total	956*
Source: (from Solar Energy Collector)	

*The solar energy conversion system output at 956 watts, less the cooling load, will be available for operation of the drill unit and X-ray diffractometer when the vehicle is at a standstill.



SCHEMATIC,
ELECTRICAL POWER SYSTEM
FIG. VI. 7

TABLE VI.2

WEIGHT OF POWER SYSTEM AND NETWORK COMPONENTS

	Pounds	
Closed Cycle Solar Converter		
Mirror, Ribs and Attachments	60	
Boiler Assembly	12	
Turbogenerator and Enclosure	25	
Condenser and Radiator	30	
Feed Pump	4	
Boiler and Load Controls	10	
Piping and Hardware	15	
Orientation System (sensors and drive)	9	
Solar Converter Total		165
Battery		131
Inverter, 850 watts, 400 cps		15
Inverter, 100 watts, 400 cps		6
AC to DC Converter, 150 watts		4
Distributor		30
Cables and Connectors		20
TOTAL		371

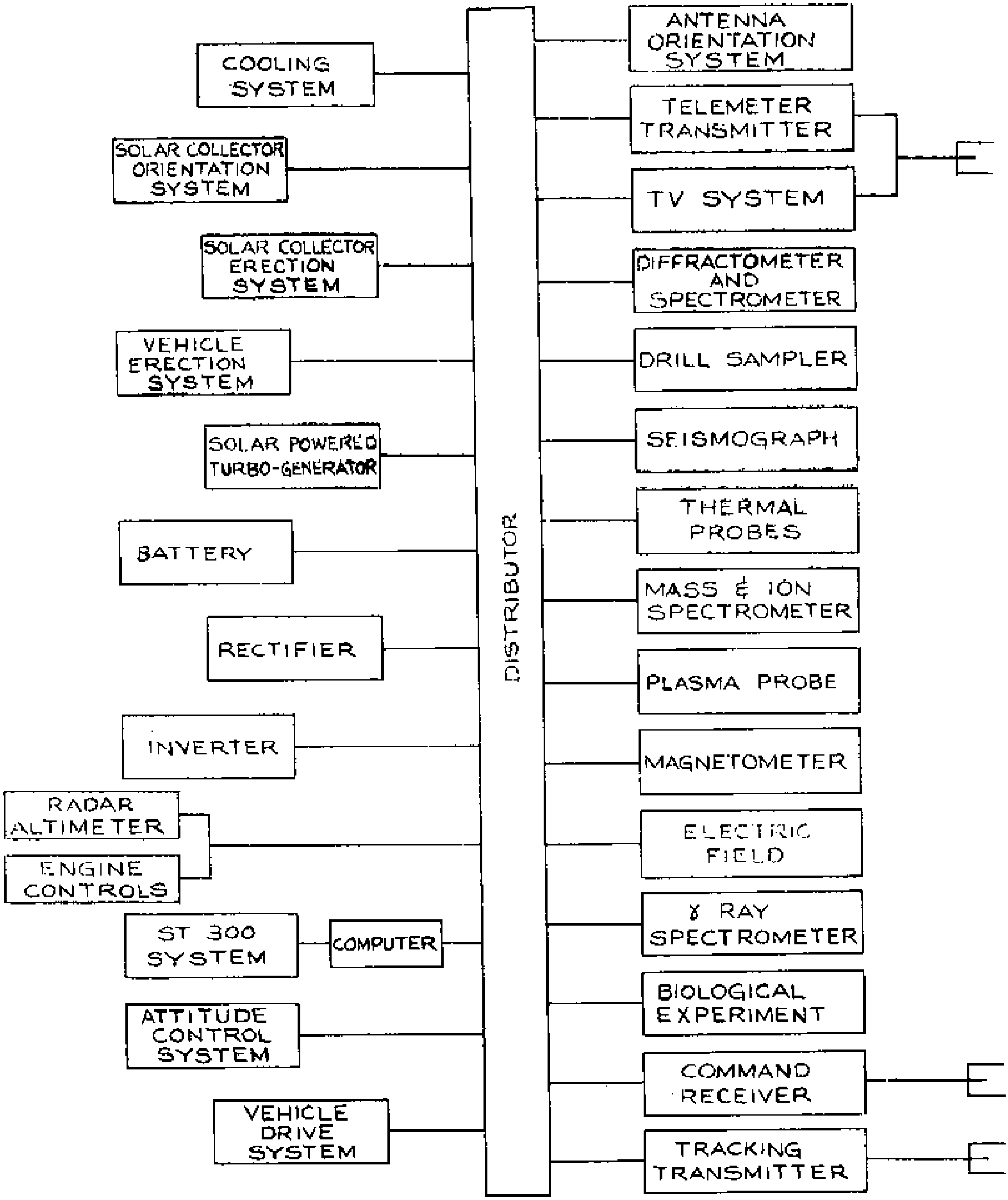


FIG. VI. 8

NETWORK BLOCK DIAGRAM OF ROVING VEHICLE

(4) The requirement for a battery would not be eliminated since power is needed immediately before landing when the solar conversion system is secured for the landing; immediately after landing for erection of solar collector and radiator and for initial programming of instrumentation; and operation of the drive system in the event that the vehicle lands or moves in the shadow.

Continuous alignment of the solar collector with the sun within the high accuracy required for full power output may not be possible while the vehicle is moving over rough terrain. A battery will make up for this deficiency of the solar conversion system by supplying power through an inverter to the drive motors. Furthermore, a battery that is kept charged during the day could be used for heating and operation of a limited amount of instrumentation during the night in a manner similar to that planned for the stationary packet.

In view of the above considerations, it is planned to use a zinc-silver oxide battery similar to the one proposed in Section VI.3.

The in-flight power requirements will be the same as those for the stationary packet and the erection phase power requirement after landing is estimated to be the same as that of the stationary packet giving a total of 8795 watt-hours. In addition, sufficient energy must be provided to operate the driving system for 15 minutes in the event that the vehicle lands in a shadow. A minimum amount of communications and telemetry equipment must also operate during this period. The total power level is estimated to be 500 watts. This adds 125 watt-hours and a further increase of 10 per cent for reserve capacity brings the total to 9812 watt-hours. Based on 75 watt-hours per pound, the total battery weight is 131 pounds.

VI.4.3 Closed Cycle Solar Converter. For the solar energy conversion system considered, boiler temperature and pressure as well as condenser temperature and pressure seriously affect the system performance; thus, they must be optimum within practical limitations. The system component efficiencies and limitations as well as practical collector and radiator performance must be compromised to attain an optimum over-all system. The weights given in Table VI.2 for the solar conversion system are not minimum since it was felt that some weight should be sacrificed to provide high strength and rigidity as protection against vibration and landing shock.

As shown in Table VI.1, the net system output must be 956 watts. However, an additional 25 watts for regulation and system control and approximately 20 watts for orientation of the solar collector must be included. This gives a gross requirement of 1001 watts. Preliminary calculations indicate that this system will have an over-all efficiency of 17.2 per cent.

The maximum theoretical efficiency for any heat engine is determined by the absolute temperatures of the working fluid as received and as released by the engine and is expressed by the equation:

$$\text{Carnot Efficiency} = \frac{T_1 - T_2}{T_1}$$

The boiler and condenser temperatures selected for this application are 1200° F and 480° F, respectively. Converting these temperatures to the Rankine scale

$$\text{Carnot Efficiency} = \frac{1659 - 939}{1659} = 43.4\%$$

The over-all system efficiency is determined by the product of the individual component efficiencies and the Carnot efficiency. Component efficiencies are tabulated as follows:

Generator	85%
Turbine	60%
Solar Collector	80%
Insulation & Pump	97%

The system efficiency is given by:

$$\text{Efficiency} = .434 \times .85 \times .60 \times .80 \times .97 = 17.2\%$$

Had this system efficiency been based on the Rankine cycle efficiency, it would have been 16.1 per cent which is in very close agreement.

Rankine cycle thermal efficiency is defined by:

$$\eta_{rc} = \frac{h_1 - h_2}{h_1 - h_f}$$

where h_1 = enthalpy at turbine inlet conditions (BTU/#),

h_2 = enthalpy at turbine exhaust conditions,

h_f = enthalpy of mercury liquid at condenser conditions.

The Mollier diagram for mercury gives enthalpies at inlet and outlet conditions as 161 BTU/# and 126 BTU/#. Enthalpy for liquid mercury at 480° is 16 BTU/#. Therefore, $\eta_{rc} = 24.1\%$ and

$$\eta_{system} = .241 \times .85 \times .80 \times .97 = 16.1\%$$

VI.4.4 Components

VI.4.4.1 Alternator. The alternator selected for supplying the output electrical power from the solar-converter will have the following characteristics and ratings:

(1) Three-phase, wye-connected, 115/200 volts, 400 cycles

(2) Output 1.25 KVA at 0.8 power factor

(3) Efficiency 85% at 24,000 RPM

(4) Combination permanent magnet and brushless induction generator type

(5) Frequency regulation ± 5%

(6) Voltage regulation ± 3%

(7) Weight - 12 lbs.

Systems based on frequencies higher than 400 cps offer additional weight and volume savings for components. However, since most of the present guidance and control equipment is based on 400 cps, the proposed system was selected on this basis.

In selecting the smallest and lightest alternator for the subject application many operating requirements and physical limitations were considered. DC generators were quickly eliminated as a possibility due to:

(1) Commutation difficulties that would arise at rotational speed compatible with efficient turbine operation.

(2) Unreliability due to excessive brush wear at high speeds and high temperatures.

(3) Need for additional cooling of rotor windings. Therefore, an AC system was considered to be most practical and the two types of basic alternators that appeared most feasible for the subject application were:

(a) The axial air-gap, permanent magnet alternator.

(b) The brushless, induction alternator requiring no rotor windings.

Both machines are adaptable to high speed operation and efficient cooling and are quite competitive in efficiency and weight for a given output. The brushless alternator has a slight advantage in weight for the 1 KW size, however, for the purpose of this study, it appears that either machine would perform within an 83 to 88 per cent efficiency range and would weigh from 8 to 12 pounds.

It is felt that a machine combining the principles of both types of alternators would afford optimum performance when variations in operating conditions are considered.

The basic design would be based on that of a permanent magnet machine which offers the advantages of positive self-excitation, simpler construction, and accommodation of greater number of poles. However, to allow for operation with somewhat larger air gaps, to provide voltage control under load changes and/or power factor changes, it is proposed that the brushless induction principle also be incorporated into the machine.

Although machines are under development that should be capable of operation at temperatures of 750° F, it is anticipated that liquid mercury from the working fluid system will be circulated through coils in the alternator stator to provide cooling for higher reliability. In addition, it is expected that the alternator will be sealed in a housing with the turbine so that liquid mercury can be used as a bearing lubricant and coolant. It is expected that a machine weighing 15 pounds and operating at an efficiency of 85 per cent can be realized which will provide the above features.

VI.4.4.2 Turbine. The turbine proposed for this system is a three-stage impulse machine with an expected efficiency of 60 per cent. A single stage turbine would call for a pitch line speed over 70 per cent greater than a three-stage machine with equal heat releases per stage. To compromise rotational losses and mechanical wear with efficient thermal operation and alternator usage, a speed of 24,000 rpm appeared most desirable.

Temperatures and pressures selected provide the desired turbine operation within the practical limitations of the system's components. Although high boiler pressure to condenser pressure ratios give high turbine efficiency, it imposes bucket erosion, high material strength and greater pump losses on the system. Therefore, an inlet pressure of 200 psia was selected which provides only a nominal amount of superheating of the mercury vapor. In accordance with condenser design practice a pressure of 2 psia was chosen to provide adequate feed pump inlet pressure and to insure reasonable pressure drops.

VI.4.4.3 Boiler. The upper temperature limit for such a system is governed by the maximum allowable stress at turbine wheel speeds compatible with reasonable efficiency and metallurgical limits of boiler and pipes. Since a high boiler temperature affords greater possible efficiency, the maximum temperature of 1200° F was selected. This provides a reasonable safety margin and allows the use of lighter weight materials. A slight amount of superheating appears desirable; however, this only requires the boiler output pressure to be 200 psia.

The boiler will be of lightweight construction utilizing thin wall steel tubing with a coating of copper or silver to provide better wetting and heat conductivity. The exterior surface of the boiler will be coated to improve its absorptivity.

As can be seen in Figure VI.9, the boiler is rigidly mounted to the collector to provide accurate concentration of solar energy.

VI.4.4.4 Solar Collector. In order to provide sufficient solar energy for the system selected, a paraboloidal mirror for concentrating the light on the boiler, as shown in Figure VI.9, was chosen. Since the sun appears as a disc subtending an arc of 32 minutes, the diameter (d) of the sun's image in the focal plane of the paraboloid is given by

$$d = 2 f \tan 16'$$

where f = focal length of the concentrator. It can also be shown that such a concentrator with a flat collector (boiler) has a concentration ratio (c) defined as the ratio of heat flux at the focus to that received by an equivalent plane surface and is given by the relation

$$c = 46{,}100 \sin^2 (\theta)$$

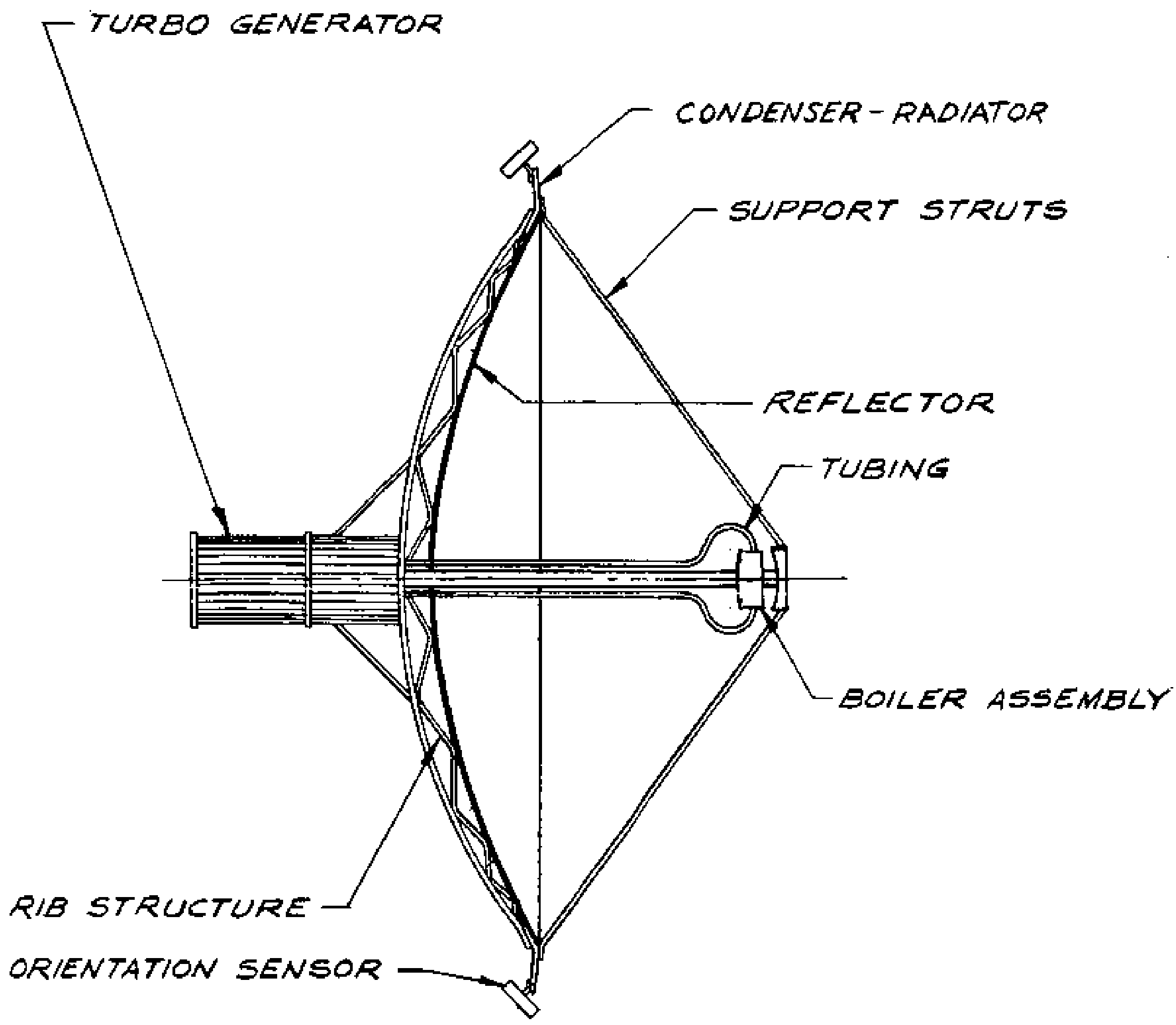
where (θ) = angular aperture or rim angle of the mirror.

Figure VI.10 shows the theoretical performance of a paraboloidal concentrator. The temperatures were based on a perfect reflector, black body collector, and 140 milliwatts per square centimeter incident solar radiation. Practical considerations of construction inaccuracies, orientation errors, weight, rigidity, and efficiency led to the selection of a mirror with an effective concentration ratio of approximately 600.

Assuming a boiler absorptivity of 0.9 and a mirror reflectance of 95 per cent, and allowing for construction and misalignment errors of approximately 0.5 degrees, the proposed collector will have an efficiency of 80 per cent. Since the over-all system efficiency is 17.2 per cent and, based on 140 milliwatts per square centimeter of radiant solar energy, the required mirror diameter will be 92 inches with a focal length of 40 inches.

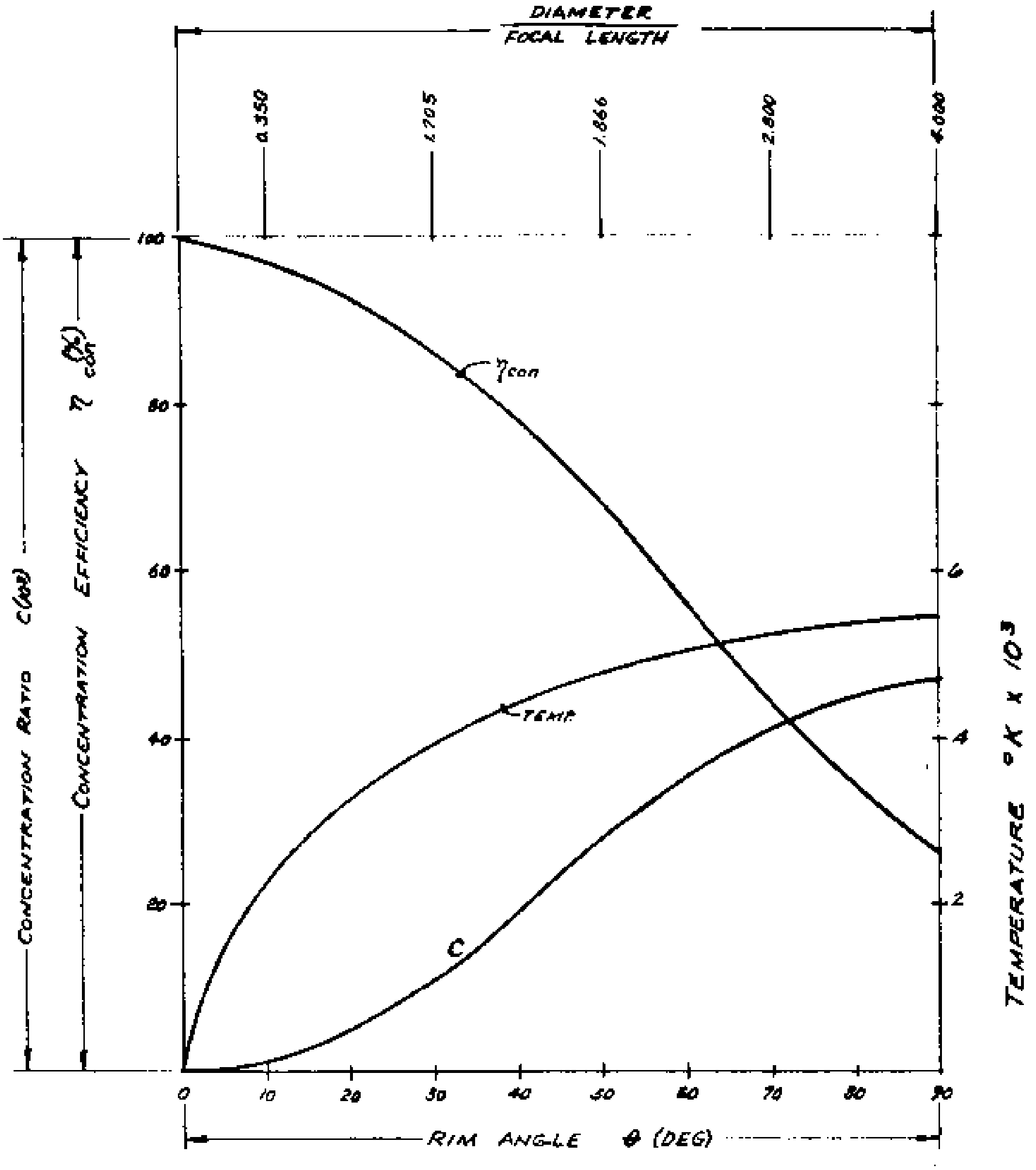
The collector will be of ribbed construction (similar to a radar antenna) covered with about 200 panels of very lightweight reflector material. The boiler will be cylindrical with one side flat to reduce the problem of re-radiation and yet to allow a reasonable target collection surface.

FIG VI .9



SOLAR CONCENTRATOR
SKETCH

(NOT TO SCALE)



THEORETICAL PERFORMANCE OF PARABOLOIDAL CONCENTRATOR

FIG. VI. 10

VI.4.4.5 Radiator. The radiator surface must be coated with stable materials which increase its emission properties at the temperatures required. It is desirable and anticipated that multiple layer coatings can be applied which not only increase the emissivity but also decrease the absorption at shorter wavelengths. A satisfactory material would be one which has an absorptivity of approximately .25 below one micron wavelength and an emissivity of approximately 0.9 above three microns wavelength. In view of the above properties, it has been estimated that an additional 85 milliwatts per square centimeter of radiator surface must be radiated to account for heat absorption from the sun and moon. The relationship of output power to required radiating surface can be expressed by the following equation:

$$\frac{P}{A_r} = \left(\frac{\eta}{1-\eta}\right)\left(e\,\sigma\,T_r^4 - \frac{Q_A}{A_r}\right)$$

where:

P = net power output in watts

A_r = radiating surface area in square centimeters

$\eta = \frac{P}{Q}$ = over-all thermal efficiency

Q = total heat input to power package in watts

e = thermal emissivity, dimensionless

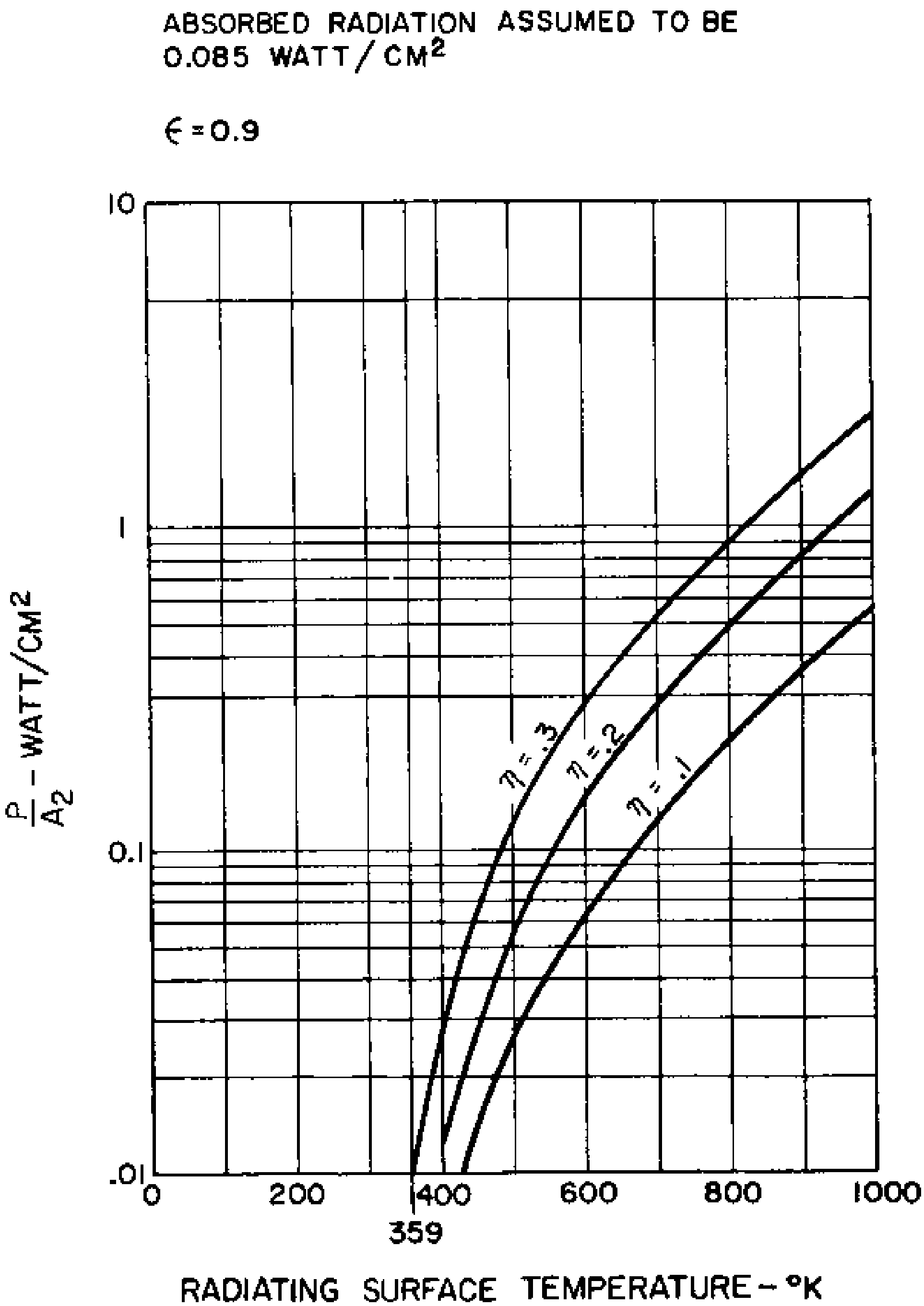
σ = Stefan - Boltzmann constant = 5.73×10^{-12} $\frac{\text{watts}}{\text{cm}^2\,(^\circ\text{K})^4}$

T_r = radiator temperature in degrees K

Q_A = absorbed radiation heat in watts

Figure VI.11 shows the ratio of output power to radiator area as a function of radiating surface temperature for various thermal efficiencies in the range expected. Since the size of radiating surface has a large effect on over-all weight, it is desirable to have as high a value as possible of this power output per unit area. The desirability of having both a high efficiency and a high radiating surface temperature is thus

RADIATING SURFACE AREA AS A FUNCTION OF SURFACE TEMPERATURE AND SYSTEM EFFICIENCY



apparent, and the particularly strong effect of radiating surface temperature is evident with a lower limit under these conditions of 359°K. For this application, 522°K has been selected as the operating temperature of the radiator. With a system efficiency (exclusive of collector) of 21.5%, from the curve in Figure VI.11, the required radiator area required is 15.4 square feet. It is proposed that this be an integral part of the collector as shown in Figure VI.9.

VI.4.4.6 Orientation System. The solar concentrator and turbogenerator will be mounted on a dual axle assembly, the axes being 90° displaced, so that the concentrator can be in two planes (rotation not required). Each axle will be capable of swiveling the assembly from a geared servo motor for each axis. Because the required rotational speed will be very low, this can be accomplished with very small multipole motors.

The servo system will be a dual positional control type and the loops may be over-damped to provide stability since rapid response should not be required. Movement in each axis will be controlled by a matched pair of solar aspect sensors mounted diametrically opposite on the outer edge of the reflector and directed radially with respect to the associated rotation axis. Thus, each sensor will provide equal output when the concentrator is directed accurately at the sun. A change of sun direction will cause one sensor to increase its output while its mate's output decreases. Both outputs feeding into a difference bridge amplifier and discriminator will cause the associated servo motor to drive to correct the differential signal. The system will then stabilize when both sensor outputs become equal which occurs only when the concentrator focal axis is directed toward the sun.

It is estimated that this system will require 20 watts and will weigh approximately 9 pounds.

VI.4.5 Restarting Mercury-Cycle System. The exposure of the mercury-cycle system (as well as other fluid systems) to the extreme cold of a lunar night imposes several problems in preparing the system for operation during the second lunar day. The boiler, condenser, tubing and joints must be fabricated from high strength materials capable of withstanding severe thermal shocks that may occur during the dark-light transitions. In addition, energy storage (either chemical or thermal) must be provided to safely resume operation the succeeding day.

Storage of thermal energy was not considered feasible at this time because:

(1) The long storage time (14 earth days) imposes severe insulation and re-radiation problems.

(2) Elaborate mechanical devices would be required to retract the boiler and lines and to cover the radiating surfaces to avoid undue heat loss.

Therefore, it is proposed to store energy chemically in the form of battery capacity for restarting the system. This energy will be required for re-orienting the collector either upon command or automatically and possibly for preheating critical portions of the system to avoid rupture due to "hot spots" that may occur in the thawing system. To alleviate the problem of "hot spots" it is anticipated that the solar collector will not be fully oriented at sunrise as the ambient temperature increases. Only partial alignment of the collector will be maintained to bring the temperature up slowly.

Investigations have not been completed to determine the characteristics of an energy storage system for restarting the mercury cycle system described herein.

CHAPTER VII

PROJECT IMPLEMENTATION

An operational program, such as described in this report, will be implemented by a number of concurrent phases. The SATURN vehicle system, together with supporting facilities and techniques of operation, must be developed and fabricated. Parallel with this development, the specific components and techniques required by the lunar missions must be designed, developed, and fabricated. Systems integration must progress concurrently. Finally, after testing of the complete system, the lunar flight missions must be accomplished.

Development of the SATURN vehicle system has been described in detail in other reports.[1] In the present study, emphasis was placed on the development of the space system, while only those portions of the vehicle development were described which are of specific interest for lunar missions. The development of the SATURN vehicle, which is a firm assignment to this agency, will progress according to a schedule which is in agreement with the space system development schedule as proposed in this chapter.

If lunar vehicles are to be launched from CY 1964 on, provisions for ordering and funding of the launch vehicles must be made in the very near future.

Details of the implementation of SATURN lunar projects are discussed in the following paragraphs.

VII.1 PROPOSED ORGANIZATION PLAN

The SATURN Lunar Project, as described in this report, will be anchored in several segments of NASA. The scientific objectives will be formulated by representatives of the scientific community. The Office of Space Flight Programs will implement the program of lunar exploration. The launching vehicle and its guidance will be the responsibility of the Office of Launch Vehicle Programs. The payload package, besides serving

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[1]SATURN System Study II, ABMA Report DSP-TM-13-59, 13 Nov 59, and other reports referenced there.

the primary purpose of the flight, must be perfectly adapted to the launch vehicle; it must be designed for temperature and other space environmental conditions, and it must be built, tested and checked out in concurrence with the launch vehicle. It is logical to assume that the Huntsville Center will contribute to the payload development under the direction of the Office of Space Flight Programs.

A detailed scheme of all the project management phases of the SATURN Lunar Project cannot be drawn at the present time. However, it can be shown how the launch vehicle part of the lunar project will fit logically into the Management Plan for the SATURN Vehicle Systems which exists presently at the Huntsville Center. An Office for SATURN Systems Management was established at this agency in the summer of 1959 when the SATURN project became a firm assignment to ABMA. Its responsibility includes the coordination of the development of the various stages, of the guidance and launching systems, of the funding and scheduling, and of the integration of SATURN-boosted systems like the Dynasoar and the 24-Hour Communication Satellite.

The management plan for SATURN vehicle systems, which evolved from previous experience with REDSTONE and JUPITER systems, is illustrated in Figure VII.1. The solid lines indicate presently existing offices and sections. The dashed lines show organizational elements which are proposed in order to complete the management plan for the SATURN Lunar Project.

The heart of the project will be the Lunar Project Objectives Coordination Board. This board will be responsible for the fulfilment of the objectives of the project - which includes responsibility for concurrency of the various developments of launch vehicle, guidance systems, payload package with scientific instruments and communication devices, tracking and computing systems, ground receiver stations, and data evaluation. The board will contain members of NASA Headquarters, the Field Centers, and possibly representatives of manufacturers. Individual members of the board will be responsible for parts of the SATURN Lunar Project, while the Chairman of the Board will be Project Director and will hold over-all project responsibility.

PROPOSED ORGANIZATION PLAN

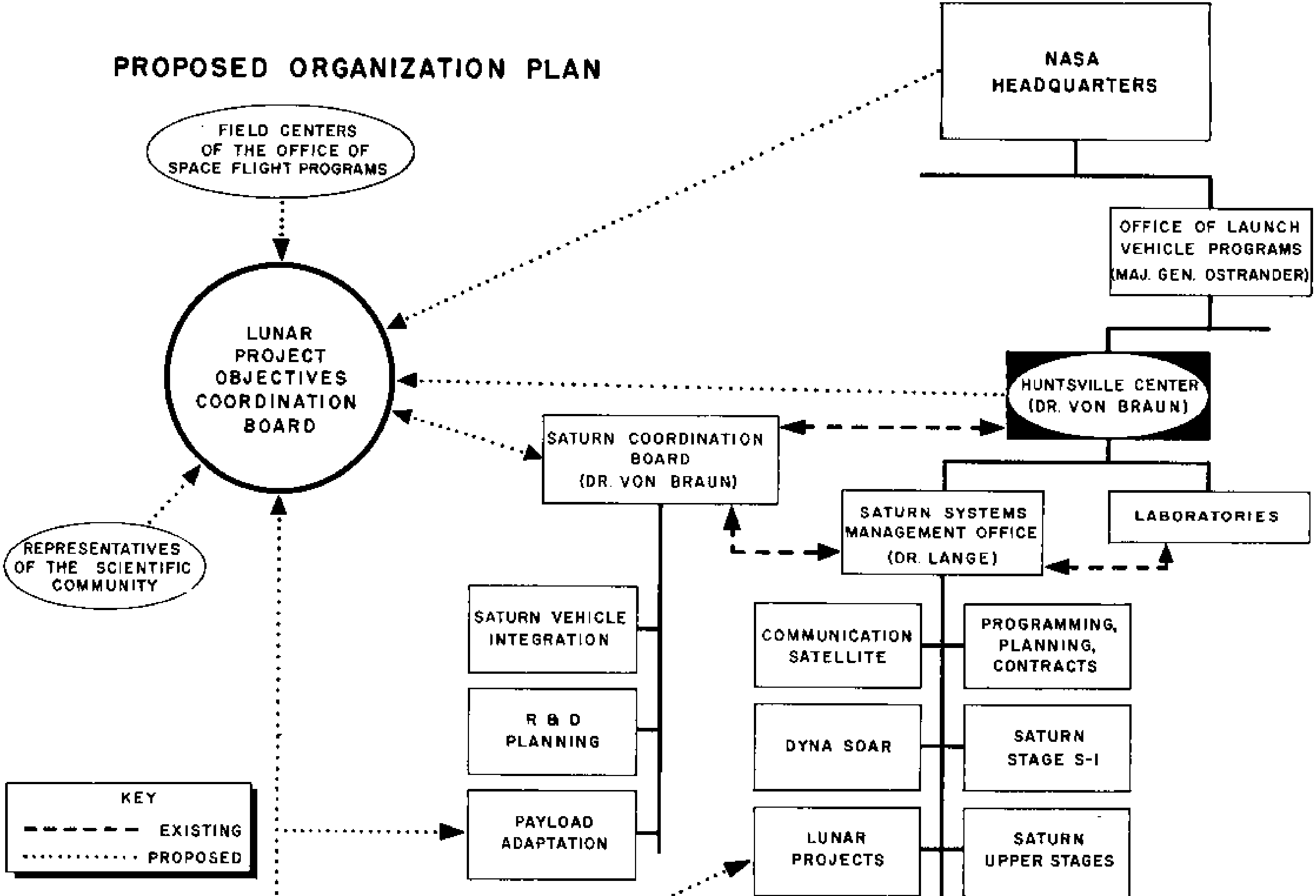

FIG. VII.1

VII.2 FUTURE EFFORTS REQUIRED FOR THE SATURN LUNAR PROJECT

This report, attempting to give a comprehensive account of lunar projects to be carried out with the SATURN vehicle, necessarily suffered from the fact that basic design and performance data of the carrier vehicle were not firm at the time when these data had to be used for payload computations. While the report was based on the SATURN B-1 configuration, it became very likely later during the report-writing period that the SATURN C-1 configuration, and as a logical next step the C-2 configuration, will be the carrier vehicles for the SATURN Lunar Project.

As a continuation of the SATURN Lunar Project study, it is proposed that a first-priority effort be extended to the re-evaluation of performance data on the basis of SATURN C-1 and C-2 configurations. In connection with this work, a refinement of trajectory and design data should take place. The present study considers only co-planar orbits for moon and lunar vehicle; no careful trajectory optimization has been made with respect to propellant consumption and guidance accuracy; the injection maneuver, mid-course correction scheme, and lunar approach should be refined. Descent and landing techniques need a more thorough study; lunar circumnavigation by the manned vehicle, and subsequent return to the earth's surface require further study.

The SATURN Lunar Project, when its basic design and performance data are established, requires further effort in two broad categories: first, a number of research and development problems should be solved in the near future in order to create the technical basis for the project; second, the organization and implementation of the project should begin as soon as possible so that the lunar project develops concurrently with the development schedule of the SATURN vehicle.

Research and development problems of the first category are listed below:

(1) Structural designs applicable to lunar vehicles

(2) Storability of liquid H_2 and O_2 under prolonged space conditions

(3) Re-ignition of LH-LOX engines under space conditions

(4) Behavior of materials and components under space conditions

(5) Further refinement of mid-course and lunar approach guidance schemes.

(6) Return-to-earth guidance and re-entry into atmosphere

(7) Adaptation of existing stabilized platform to prolonged operation in space

(8) Modification of existing guidance and control computers

(9) Design of long-range and short-range lunar altimeters.

(10) Radio command system for in-flight and lunar use

(11) Investigations of radio and optical tracking systems for lunar flights

(12) Antenna design and fabrication for use in lunar environment

(13) Investigations of television system for terminal approach and lunar surface operation

(14) Exhaust flame attenuation studies (for terminal approach scheme)

(15) Investigations of timers, pulse code modulators, multiplexers, readout and memory circuitry for payloads operating in the lunar environment

(16) Studies of solar cell operation under lunar conditions

(17) Studies of closed cycle solar power systems for lunar use

(18) Studies of auxiliary power units for circumlunar flights

(19) Investigations of a no-gimbal inertial reference system for lunar terminal approach scheme

(20) Investigations of horizon seekers for terminal approach

(21) Investigations of optical approach velocity meter for terminal approach

(22) Investigations of sun sensors for attitude control of vehicle in flight, and for solar bank on lunar surface

(23) Studies of heat transfer problems during flight to the moon

(24) Thermal design studies of stationary and roving payloads

(25) Structural design studies of stationary and roving payloads

(26) Design studies of components of the stationary and roving payloads, such as shock absorber, instrument compartment cooling system, tires, drive and control mechanism

(27) Studies of designs for manned capsule for circumlunar flights

(28) Design studies of drill, sampling system, and sample processing device for stationary and roving payloads

(29) Design studies of scientific instruments carried by lunar payloads

(30) Design studies for receiving, recording, and data processing systems on earth

It should be recognized that the scientific instrumentation research and development schedule requires immediate attention. Assuming temporarily that individual instruments were available in final form, much time would still be required to design their packaging and their auxiliary equipment, such as power supplies and telemetry systems, and then to fabricate the necessary packages, to test the entire assembly, and to make required modifications. The time needed for these activities for a two-thousand pound lunar package may not be appreciably less than the time required to develop the SATURN vehicle.

The instruments, however, are presently not in anything like final form. Thus, instrument development must be pursued with great urgency if the instrument packages are to be ready for flight when their launching vehicles are available. Fortunately, the designs of several instruments have been initiated by NASA contracts. Some of these are being designed first for flights on other vehicles. More assignments, especially to develop instruments for the SATURN program, should be made at the earliest possible time.

The second category of necessary efforts includes the establishment of an organizational structure within NASA, supported by industrial contractors, that can implement the lunar project with high expediency.

It is obvious that the research and development effort, as reflected by the above list, requires the cooperation of many segments of NASA. Considerable benefit can be drawn from earlier lunar projects, like the ATLAS AGENA and the ATLAS CENTAUR, and from satellite projects like the Military 24-hour Communication Satellite[1] presently under study by the Huntsville Center and the Signal Corps. Valuable experience in radio guidance, inertial guidance, satellite and lunar probe tracking, solar power supplies, satellite and space probe instrumentation, environmental testing, packaging, and data handling exists at various places within NASA. Capabilities for the design, manufacturing, and testing of large structures have been demonstrated.

It will be the task of the SATURN Lunar Project Director to distribute the required effort to the various segments of NASA, and to outside contractors, in such a way that maximum over-all efficiency is ascertained.

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[1]SATURN 24-Hour Communication Satellite System, U. S. Army Ordnance Corps, U. S. Army Signal Corps, 30 November 1959.

The over-all project requirements, and a time schedule for accomplishing them, are shown on Figure VII.2.

The proposed SATURN Lunar Project flight schedule, showing the sequence of payloads and the projected firing dates, is illustrated in Figure VII.3.

It must be emphasized that the schedule shown in Figure VII.2 is predicated on the assumptions (a) that work in certain areas can begin immediately, (b) that the necessary funding is available when needed, and (c) that full responsibilities are assigned within the very near future. Any unforeseen difficulties or delays will reflect in the completion dates of the program.

VII.3 TEST AND TRAINING PROGRAM

Implementation of a lunar exploration program such as described herein will require an extensive plan for testing of all system components and for training personnel. Typical examples of the many aspects of the test program are:

(1) Testing individual components of the integrated payload package

(2) Testing the operation of an integrated payload in simulated lunar environment

(3) Drop testing of payload package from airplanes

(4) Payload package handling and transportation

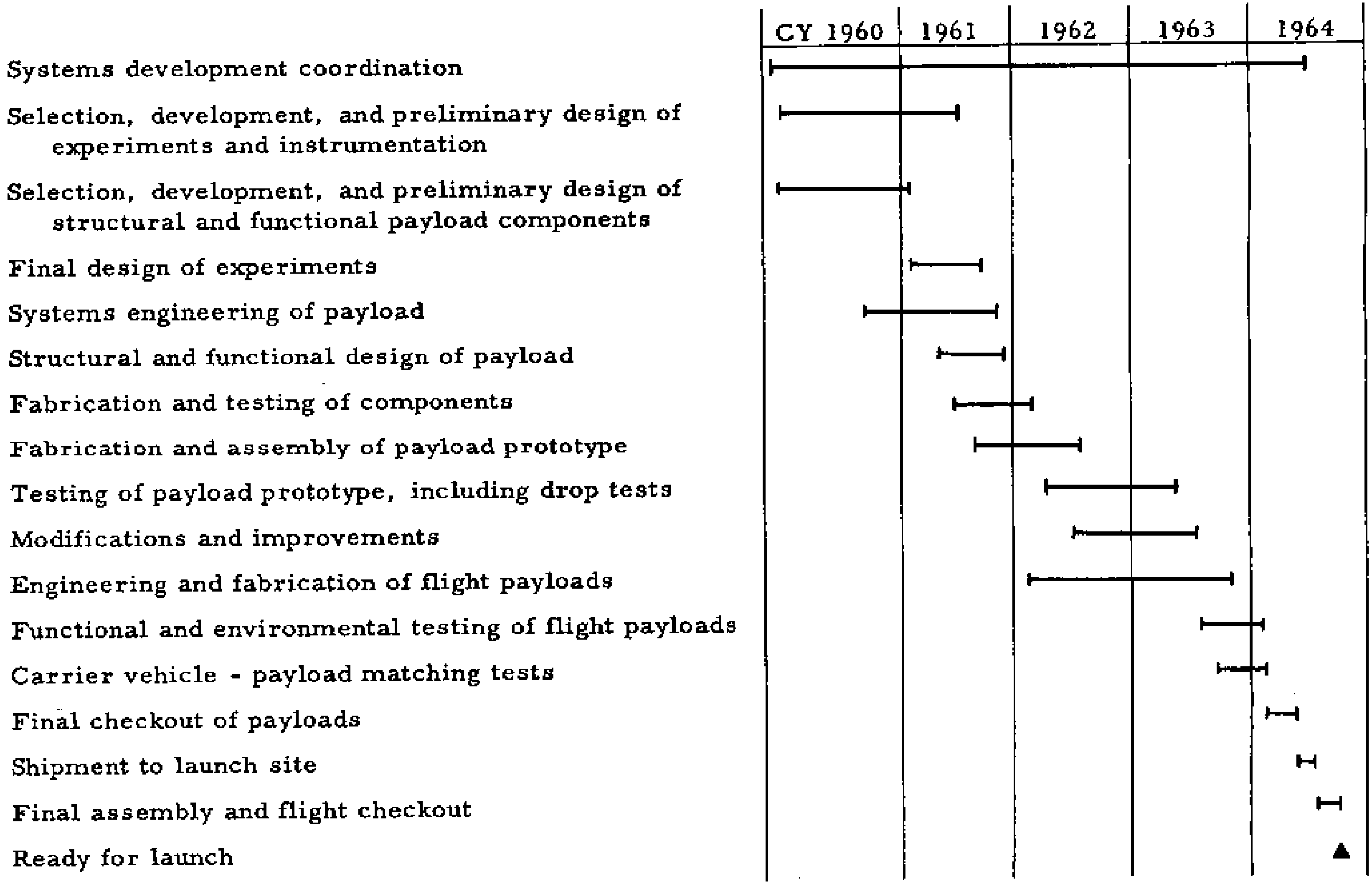
(5) Training: vehicle launching and landing crew; crew to operate the lunar roving vehicle via TV link; or manned operations (flight crew, recovery crew).

For many of the tests, equipment and experience are available at ABMA and at other segments of NASA. For other tests, new equipment is necessary (e.g., lunar environment chamber).

Some special remarks follow regarding testing of the unmanned landing vehicle and the payload:

Figure VII. 2

SATURN LUNAR PROJECT REQUIREMENTS



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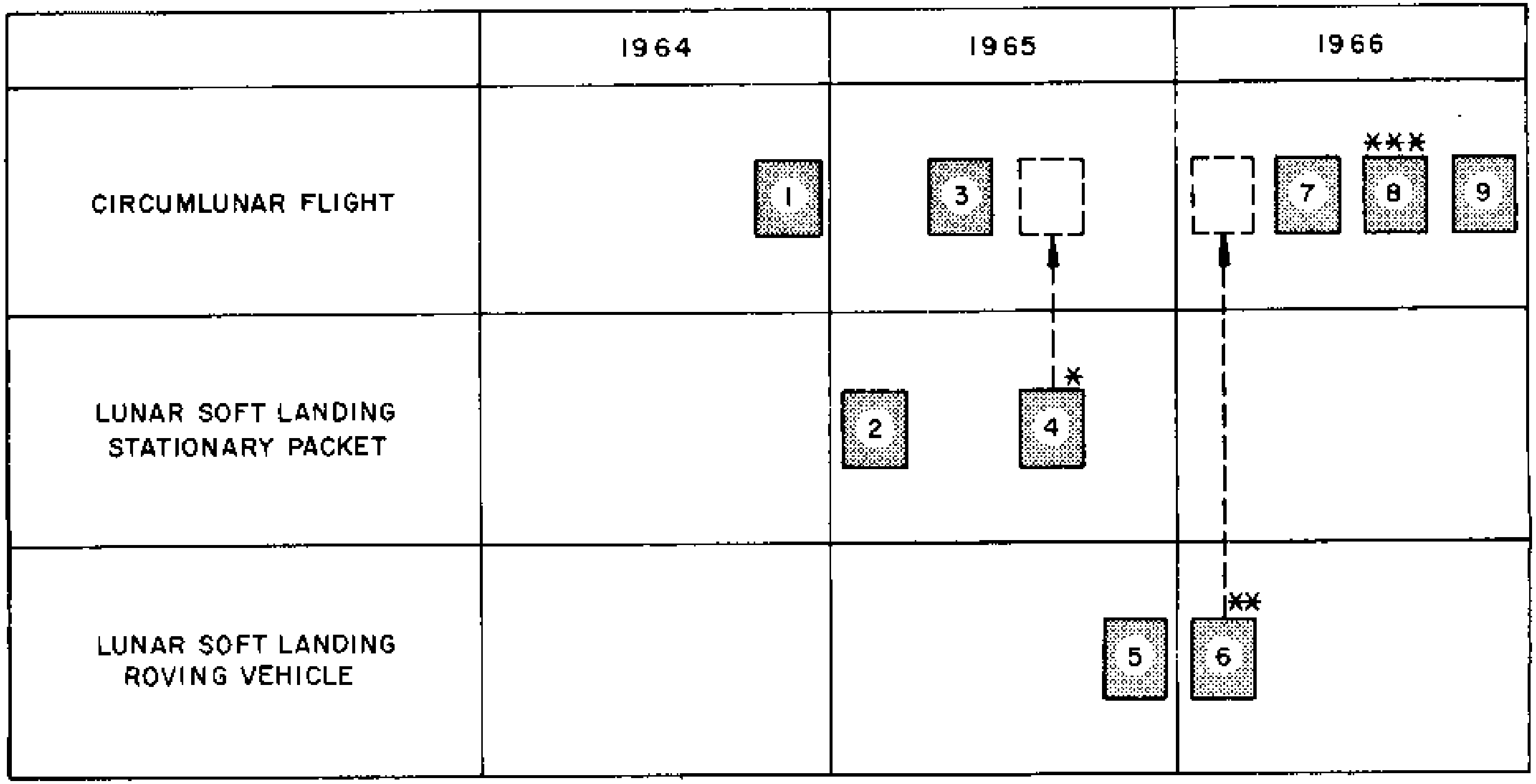
SATURN LUNAR PROJECT FLIGHT SCHEDULE
(BASED UPON SATURN C-2)



* FLIGHT NO. 4 COULD BE USED FOR A CIRCUMLUNAR FLIGHT IF FLIGHT NO. 2 IS SUCCESSFUL AND ALL DESIRED DATA IS OBTAINED.

** FLIGHT NO. 6 COULD BE USED FOR A CIRCUMLUNAR FLIGHT OR A SOFT LANDING STATIONARY PAYLOAD IF FLIGHT NO. 5 IS SUCCESSFUL.

*** FLIGHT NO. 8 COULD BE USED FOR ANY MISSION AS REQUIRED.

FIG. VII. 3

CONFIDENTIAL

Vacuum Chamber Tests

The purpose of the chamber tests is to allow an evaluation of the effects of vacuum and temperature environment on components and individual instruments. Dependent upon the size and kind of individual parts, these tests can be accomplished in relatively small chambers under high vacuum. As integrated payload packages are developed, testing in larger chambers will become necessary, but since the individual components will have been evaluated, only the over-all functioning must be proven and evaluated. As the final step of this program, the complete packet will be tested and remotely operated in a large chamber for the duration of one lunar day. Among other things, this testing will permit evaluation and optimization of the cooling system. The chamber should simulate the complete lunar environment, as far as this is practical. If available, satellites might also be used for selected environmental tests.

General requirements for the vacuum test facilities are listed below:

(1) The first facility should be a vacuum chamber large enough to enclose a vehicle 24' x 20' x 24' high at pressures of 10^{-8} mm Hg or below. The following capabilities should be incorporated into the chamber:

(a) Movable overhead solar optical radiation spectrum capable of 420 BTU/h/ft^2 intensity. This solar radiation source should be constructed so that the emitted rays are parallel to one another.

(b) A heat sink of LN_2, or some other suitable cryogenic fluid to simulate the temperature conditions of outer space, should be located along the walls and portions of the ceiling.

(c) An infrared radiation source covering the chamber floor and lower walls to simulate radiation from the lunar surface.

(d) Instrumentation for all required measurements (temperature, pressure, etc.) on the vehicle and throughout the chamber.

(e) Provision for simulation of lunar surface conditions (dust, etc.) so far as compatible with operation of the vacuum pumps.

(f) Provision for impact testing of shock absorber.

(2) The second facility should be a relatively small vacuum chamber, (for example 6' dia. x 6' long), with vacuum as far below 10^{-6} mm Hg as possible, for component testing. An ultraviolet and infrared radiation source should be incorporated within this chamber.

Braking Stage Engine Testing

During testing of the braking stage, the lunar descent should be simulated as far as possible. After the usual testing of tanks, valves, pressurization equipment, control nozzles, and after static firing of the main thrust chamber, the braking stage should be suspended from a helicopter and allowed to descend slowly, while engine, attitude control system, guidance, altimeters, and TV systems are in operation.

Drop Testing

After all the previous tests are performed, the entire payload package, including the braking stage, should be drop tested from an airplane. Altitude and air speed of plane must be calculated to achieve the same deceleration and impact conditions as experienced in the lunar landing. These tests could be accomplished in one of the desert areas in the United States. After successful drop the payload should perform the missions required on the moon, such as obtaining samples, analysis, TV operation, responding to remote control signals, etc. This would be the ultimate earth-bound test to determine the soundness and reliability of the proposed systems.

The testing program of the hardware for the manned flights will be similar to that described for the unmanned vehicle and payloads. In addition there will be a series of unmanned flights, and animal-carrying flights, including recovery. A thorough training program to include electro-mechanical landing simulation and actual drop tests with human passengers would precede a manned flight.

VII.4 GROUND SUPPORT, LAUNCH SITE FACILITIES, AND OPERATIONS

General

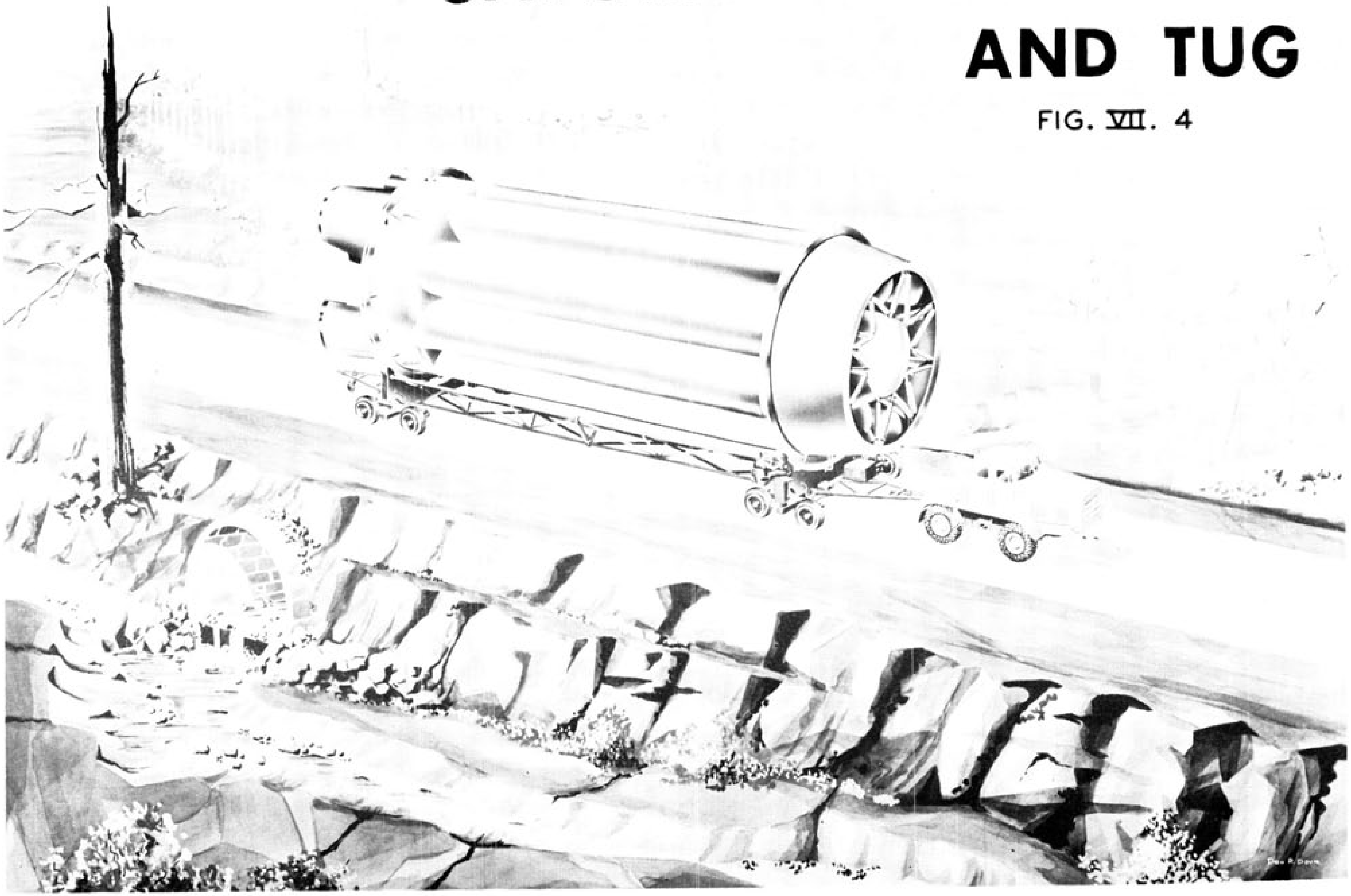
Logistical support of the launching of lunar mission vehicles from the Atlantic Missile Range (AMR) will present major transportation problems. The major missile components (booster, upper stages, and payload) must be shipped to AMR from different sections of the country and all of these components have dimensions which exceed the maximum capabilities of conventional air, rail or road carriers. Special equipment and special routing must be provided. The over-all schedule will require movement of these items at times when other transporation activities are at a peak. Several requirements will be discussed in the following paragraphs to illustrate the problems involved.

Booster

The SATURN booster to be transported from Huntsville to AMR in an assembled condition will be approximately 256 inches in diameter and approximately 82 feet in length. Since the booster will be moved on its transporter, the over-all dimensions will be even larger. Figure VII.4 shows the booster on its transporter. The only way to transport such a large, expensive and vulnerable item from ABMA to AMR is by waterway, as shown in Figure VII.5. The barge transportation to be described will assure safe handling of the cargo but will require between two and three weeks for the transportation phase alone. It will be necessary to partially build or reinforce about 10 miles of roadway at Redstone Arsenal and about 2 miles of roadway at AMR. Obstacles such as power and telephone lines must be removed, and for the actual movement, all other road traffic must be blocked or rerouted. Since the barge must be returned to ABMA with the recovered booster and since maintenance time for barge and equipment must be allowed, it is obvious that firings at two-month intervals will require at least two barges with operating personnel to maintain continuous operations.

SATURN TRANSPORTER AND TUG

FIG. VII. 4



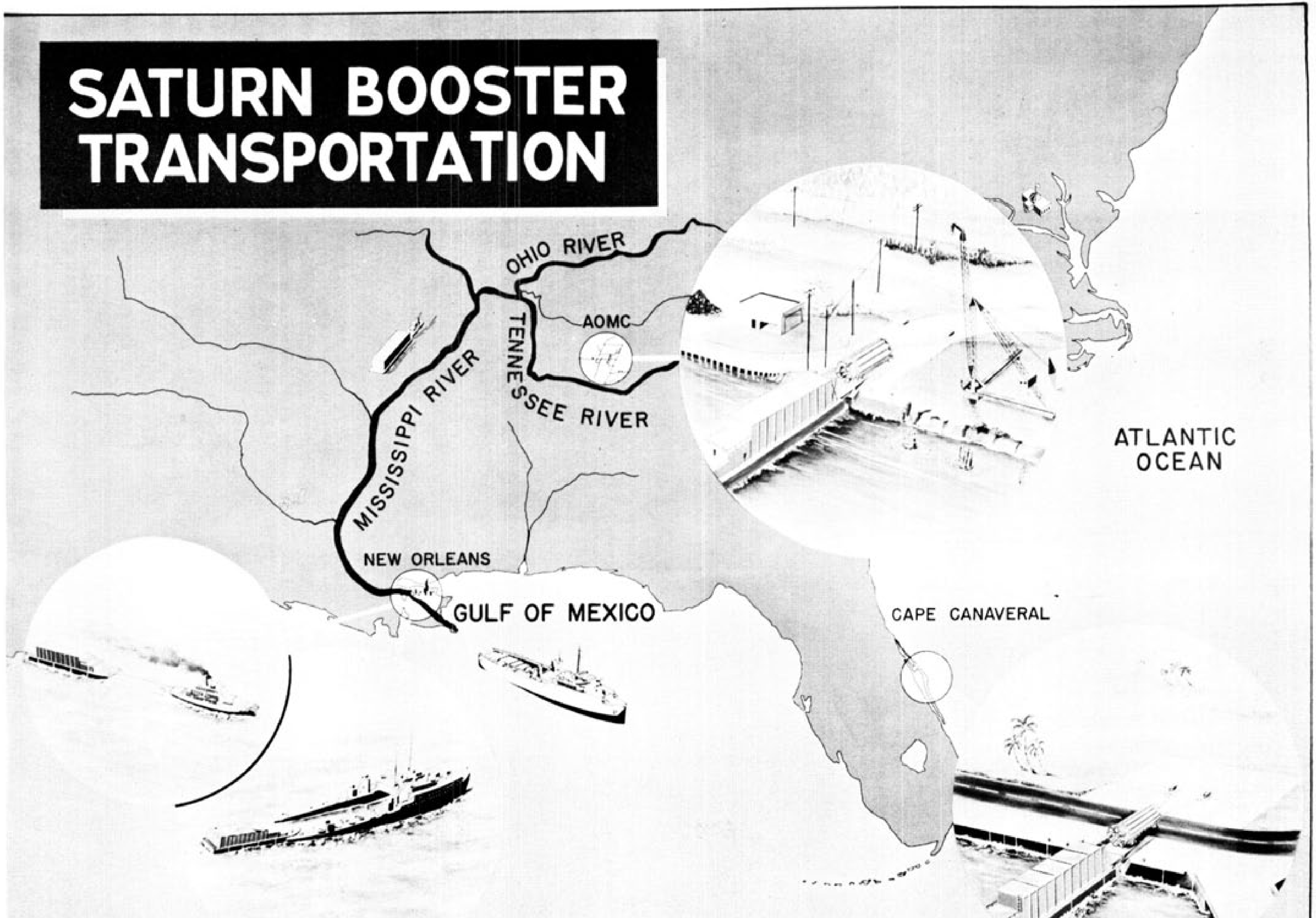

SATURN BOOSTER
TRANSPORTATION
OHIO RIVER
AOMC
TENNESSEE RIVER
MISSISSIPPI RIVER
NEW ORLEANS
GULF OF MEXICO
ATLANTIC
OCEAN
CAPE CANAVERAL

FIG. VII. 5

Upper Stages

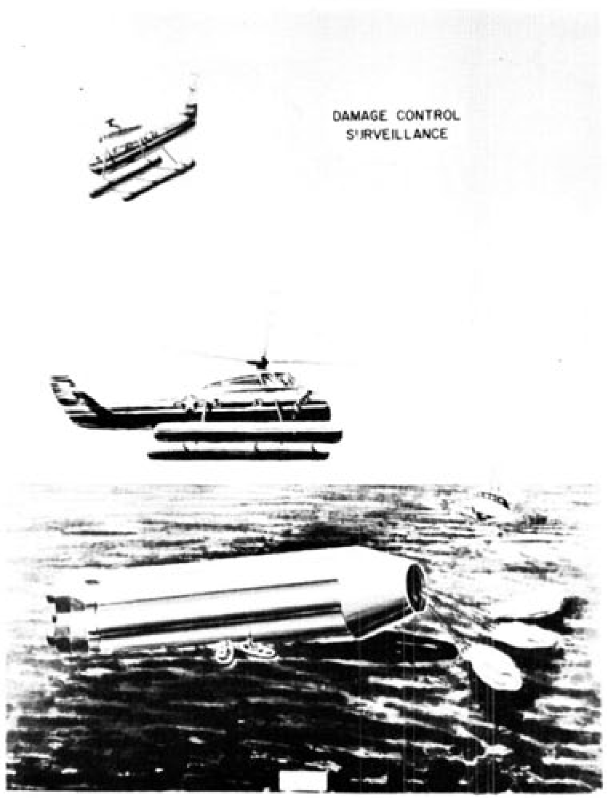
Transportation of stages of the order of 220 inches diameter over long distances from an inland area to AMR presents a critical logistical problem. Air transportation by the largest presently available cargo aircraft (C-133) is not possible. Water transportation in a manner similar to that used for the booster would be the simplest method but is not always possible. Road transportation may be possible but would require special routing and traffic restriction. Rail transportation would be even more difficult. Therefore, transportation by a lighter-than-air craft (blimp) has been considered but may prove to be economically impractical. No conclusion has been reached on a solution to this problem at the present time.

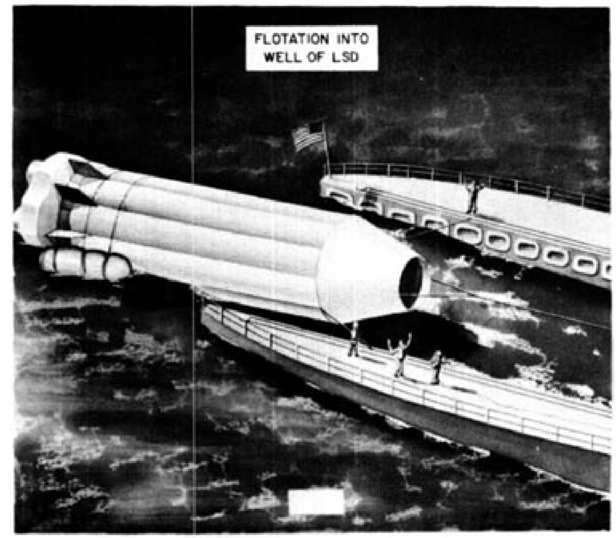
Booster Recovery

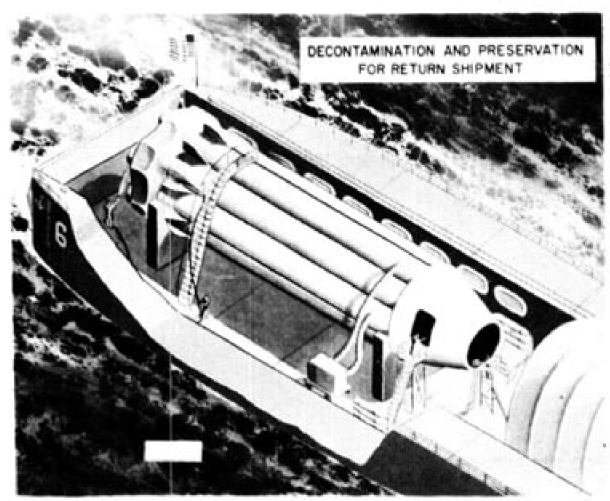
Handling of the booster after a successful landing and recovery also constitutes a major operation. The techniques of spotting and water recovery of re-entry nose cones are well known. However, the water recovery of a small and compact nose cone is far simpler than that of a huge, delicate booster which contains propellant residuals, and therefore, presents a safety hazard to equipment and personnel. A Landing Ship Dock (LSD) is proposed as the main item of the recovery equipment. Flushing, inspection, preservation and disassembly of delicate parts will be done on-board the ship by a special crew during the return trip (see Figure VII.6). Later the missile will be transferred to a river barge for return to Huntsville. A suitable harbor with proper crane facilities, such as New Orleans, will be used for the transfer. Preparation, recovery action and return to New Orleans will require an LSD for about one week, and supporting ships for a portion of this time.

Launch Site

The SATURN Launch Complex (Figure VII.7) contains all the necessary facilities for handling, storing, servicing, checkout, erection and launching of the SATURN vehicle as well as the required administration and logistical facilities and special research laboratory facilities to support the various projects to be carried out during the SATURN program.

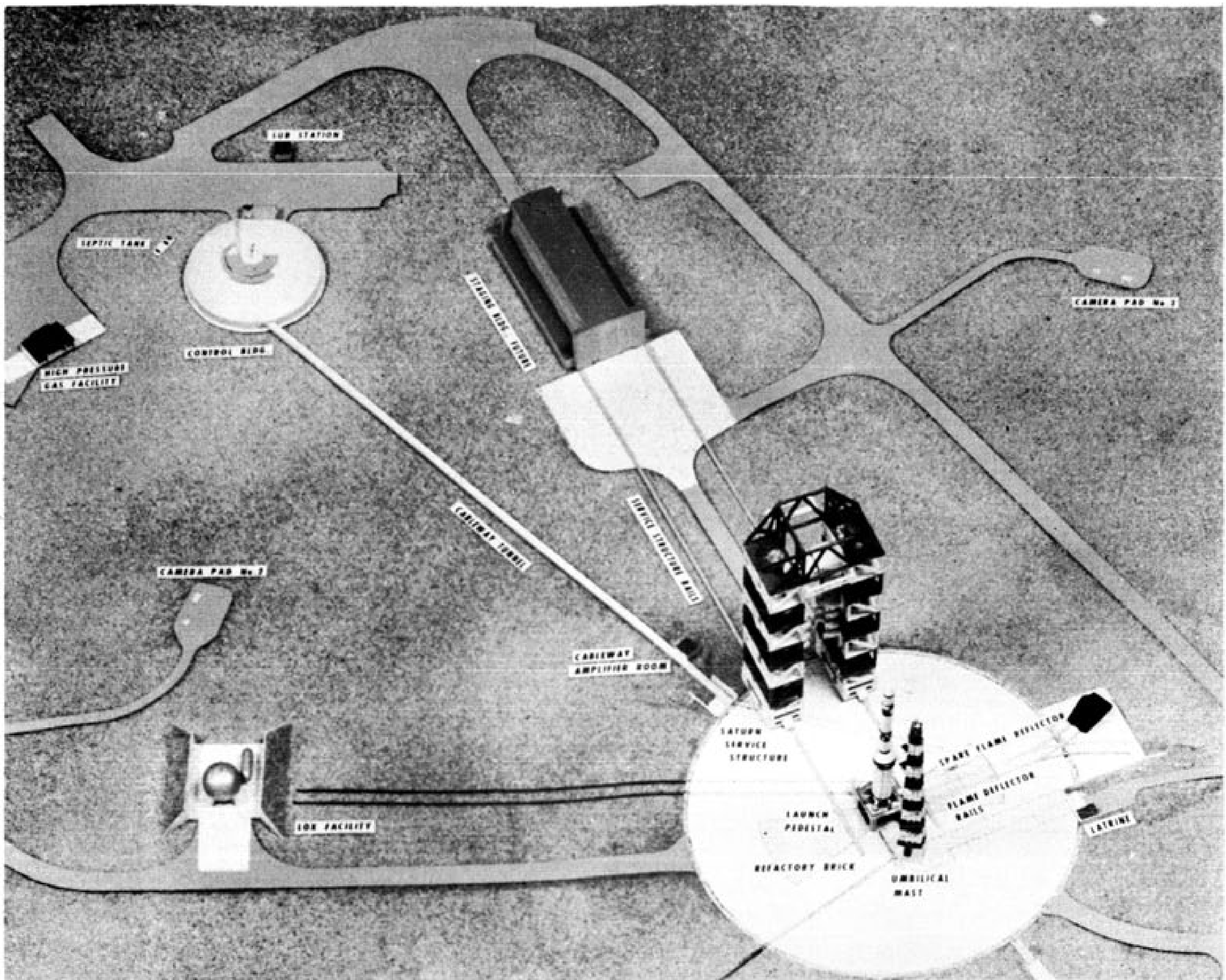






SATURN BOOSTER RECOVERY SEQUENCE

FIG. VII. 6



SATURN LAUNCH SITE
FIG. VII. 7

Upon arrival at the barge basin the booster and transporter are unloaded from the barge and towed to the booster assembly building, where final assembly details and horizontal checkout are accomplished. From here the booster passes through the staging building to the launch pad for erection. The upper stage assemblies are handled in a similar manner from the airfield (if delivered by air) to launcher.

One launch pad facility is sufficient to support SATURN firings at approximately two-month intervals. Therefore, based upon the presently projected firing schedule, one pad facility will be sufficient to support all the lunar payload missions. However, the propellant storage and transfer facilities are designed with the capability of supporting two launch pads at alternate intervals when required. This will be a requirement when the lunar payload missions are superimposed on other SATURN programs scheduled.

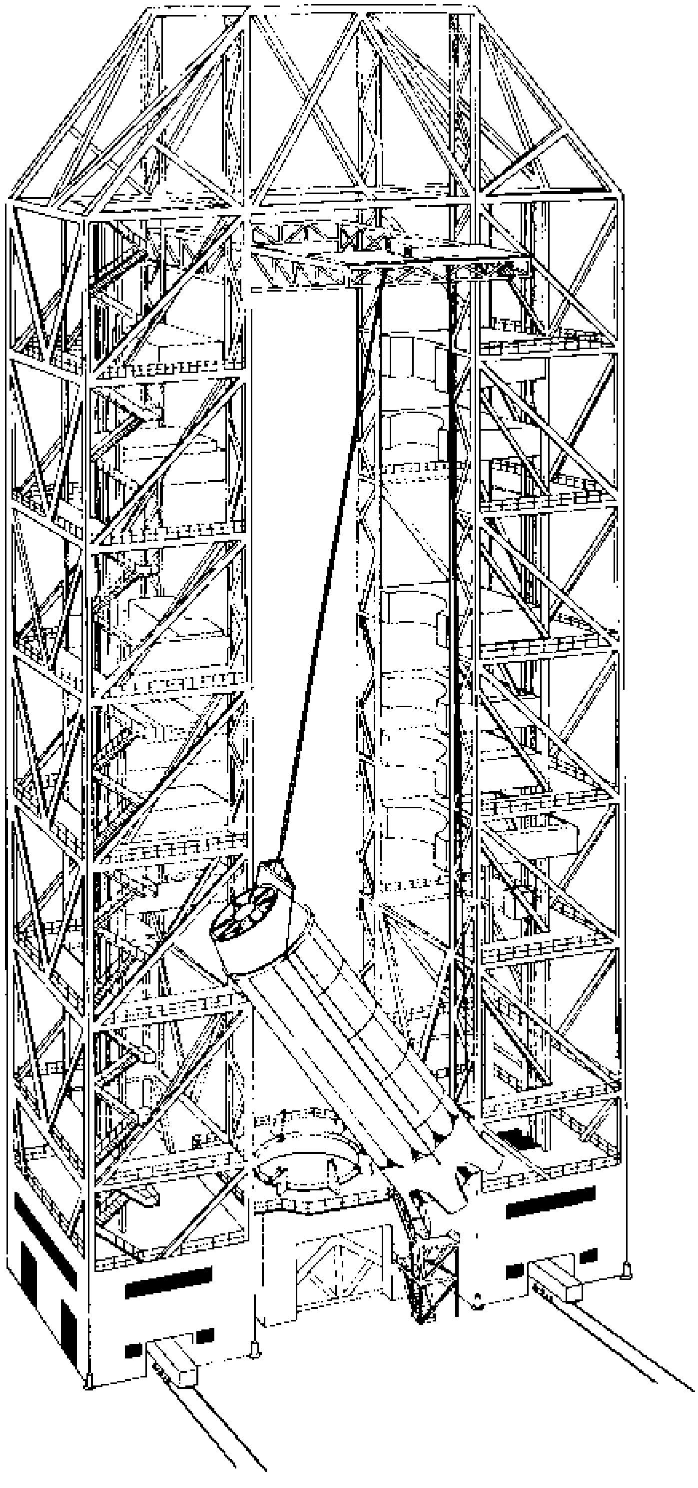
Booster Handling and Checkout

The SATURN booster can be erected with the help of the track-mounted gantry-type service structure (see Figure VII.8). Thereafter, a pneumatic leak and function check is made utilizing gaseous nitrogen. The engine service operation follows, during which an engine may be exchanged in the booster while in the vertical position.

Details on upper stages handling and checkout are not yet finalized. However, requirements for the lunar program will be similar to those for the standard SATURN, shown in Figure VII.9.

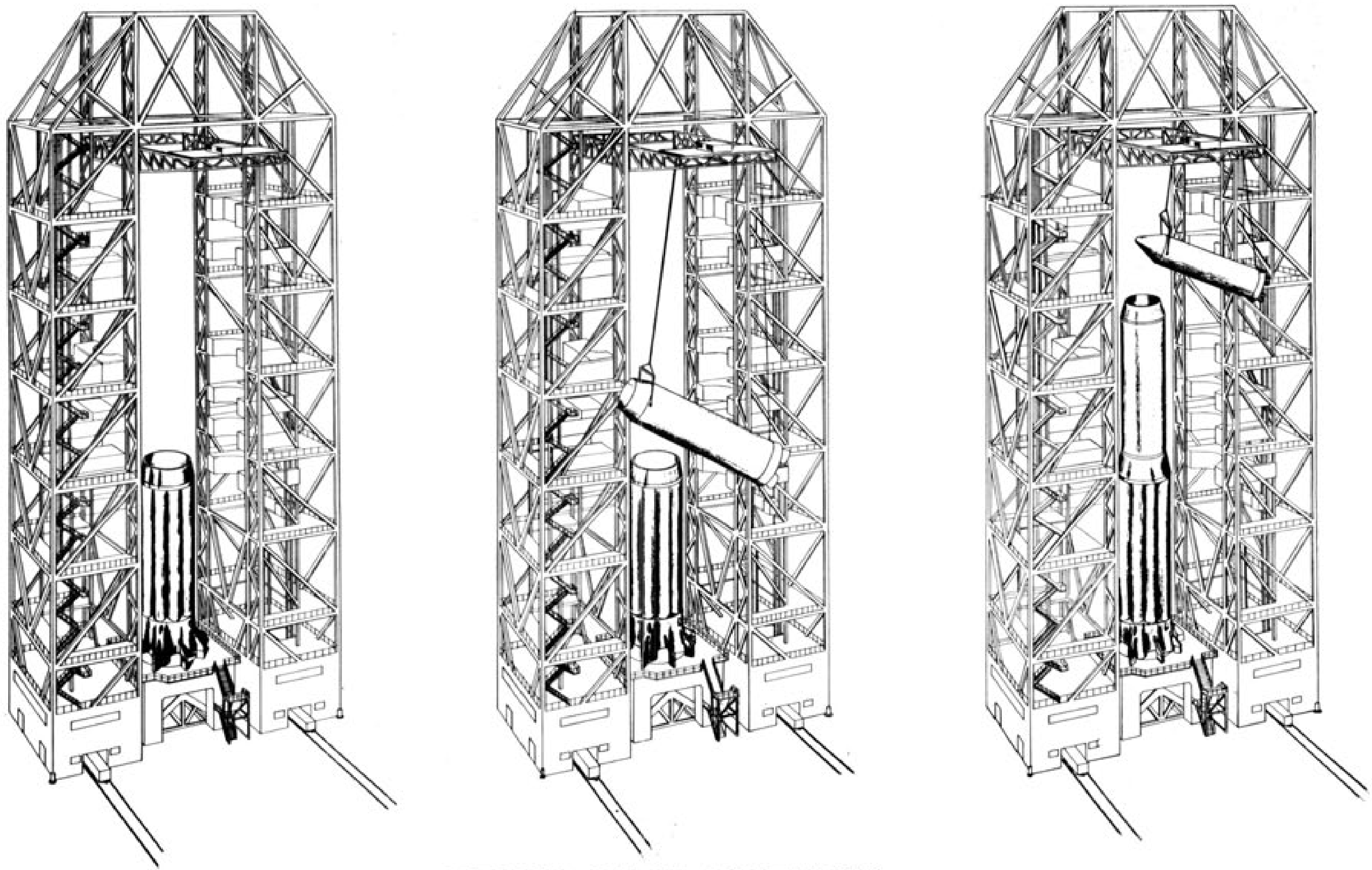
Launcher

The SATURN launcher is a reinforced concrete and steel structure 42 feet square and 27 feet high. It has eight support arms (see Figure VII.10). Four supports 90° apart are cantilevered at the outboard engines and are retracted horizontally after the valid commitment signal is given to permit the engine shrouds to clear the missile during lift-off. The other four supports are dual purpose support and hold down points located at 45 degrees between the outboard engines. Hold down is accomplished by a toggle linkage which is activated when the retractable arms are all fully retracted. In event of malfunction of one or more of the retractable supports, all four supports may be returned to position under the missile thrust frame prior to engine cut-off.



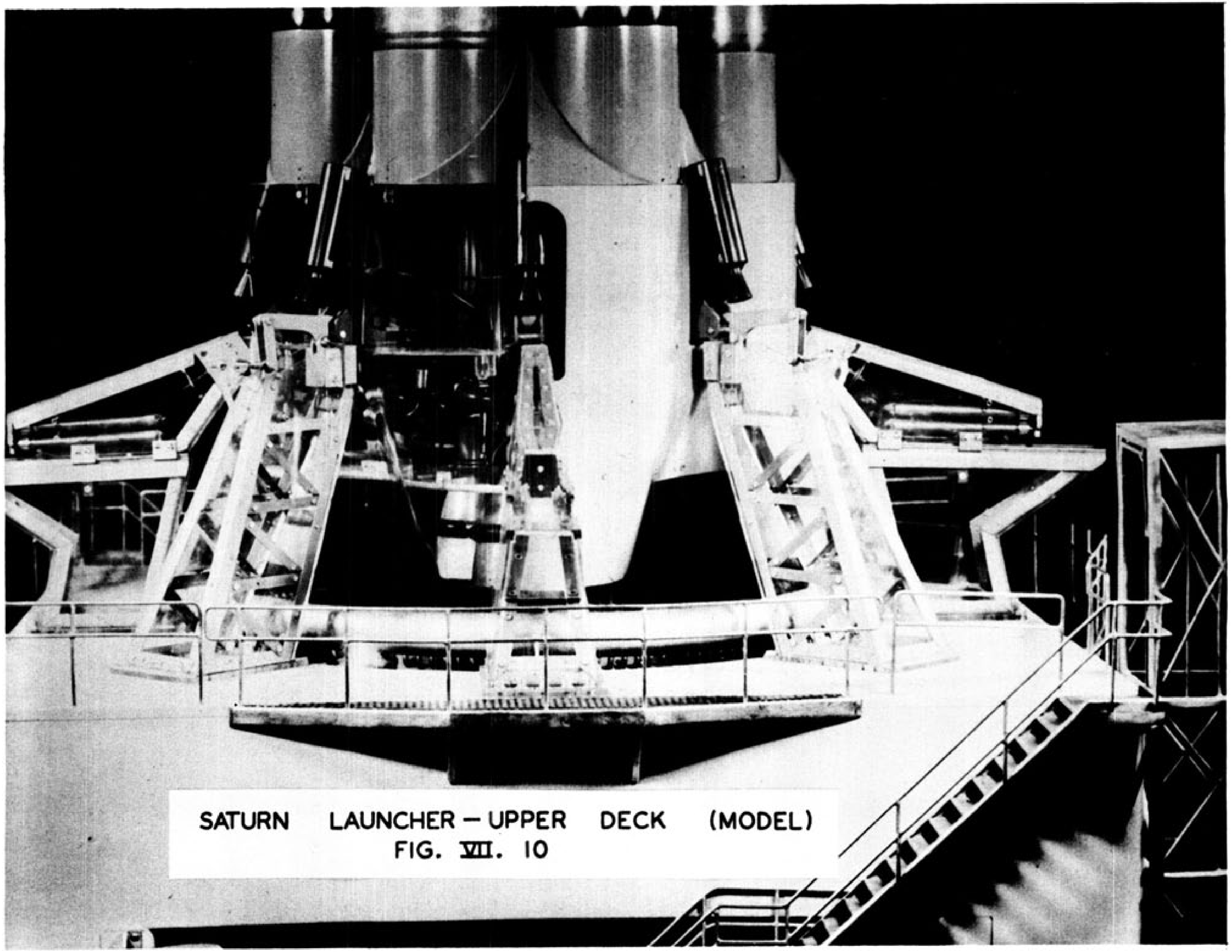
SERVICE GANTRY & ERECTION

FIG. VII. 8



SATURN STAGE ASSEMBLY

FIG. VII. 9



SATURN LAUNCHER – UPPER DECK (MODEL)
FIG. VII. 10

Umbilical Tower

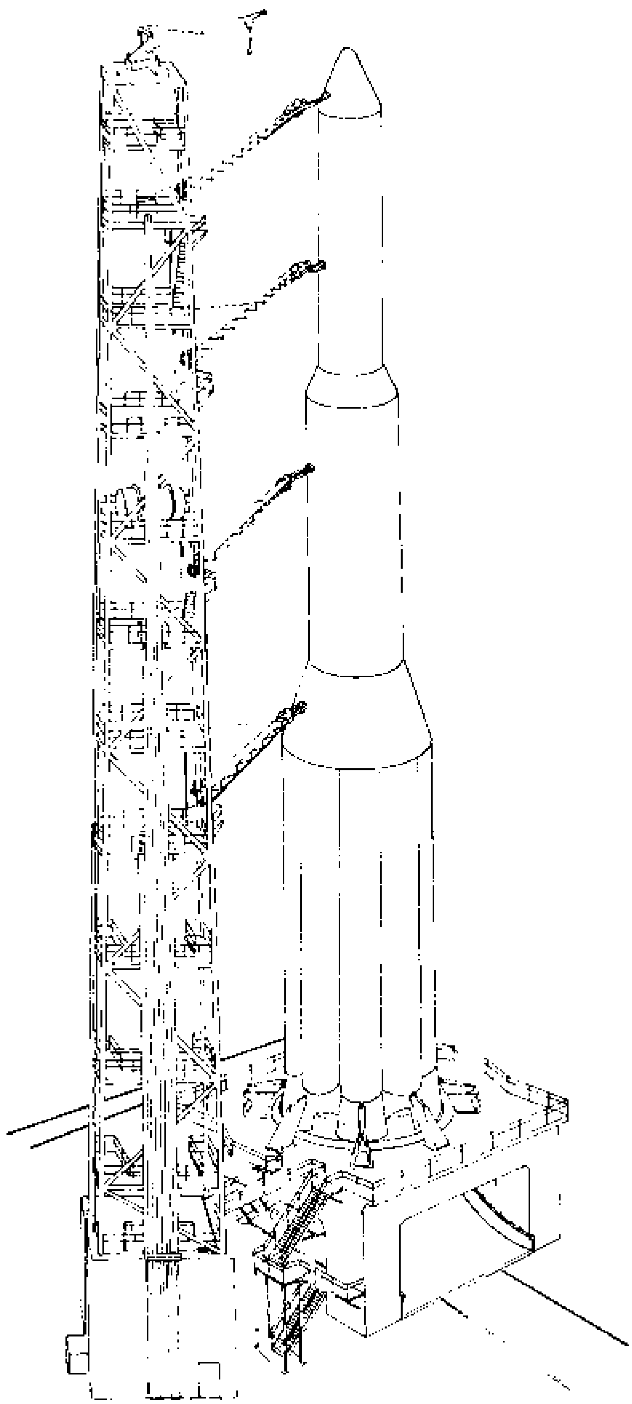
The umbilical tower (Figure VII.11) is used to support and service the umbilical arms as well as to house and support the various electrical cables, pneumatic and LOX replenishing lines, liquid nitrogen cooling tanks, mechanical refrigeration units, the ground hydraulic unit, and the pneumatic and electrical distribution station, which are required to service the booster and upper stages prior to launching. The tower is 240 feet high and 24 feet square at the base. The bottom 27 feet of the tower is enclosed to provide for two airconditioned equipment rooms. Above the 27 foot level the four-tower columns slope inward to a 10-foot square at the top. Tower facilities include safety ladders and service platforms at 20-foot intervals, a 2000-pound capacity personnel and small hardware elevator, and a 3000-pound capacity electric hoist at the top for handling lines, cables and the umbilical arms.

Propellant Storage and Handling

The propellant loading system is shown in Figure VII.12. The LOX facility consists of protective revetments, foundation, and partial weather protection for LOX storage and transfer. The system can serve two pads at different intervals.

The 125,000 gallon capacity tank is insulated, but not vacuum jacketed. The transfer system consists basically of a 2500 gpm, 400 foot head pressure centrifugal cryogenic pump with associated valving necessary to transfer LOX through 750 feet of 8-inch uninsulated aluminum transfer line. Manifold lines connected to the gantry service structure load the upper stages. A 13,000 gallon, vacuum-jacketed tank is used to replenish the various stages through insulated feed lines. The LOX transfer system is automated and is initiated and controlled from the blockhouse propellant loading panels during transfer sequence.

RP-1 fuel is stored in two insulated cylindrical tanks of 30,000 gallon capacity each. The booster is serviced by two 1000 gpm pumps operating at 175 psi head pressure through 1000 feet of 8-inch diameter transfer line. A 600 gpm filter/separator unit insures proper filtration of fuel and a minimum of entrained water. The fuel transfer system is automated and is initiated and controlled from the blockhouse by the fuel loading panels.



THREE STAGE SATURN
WITH UMBILICAL TOWER

FIG. VII. 11

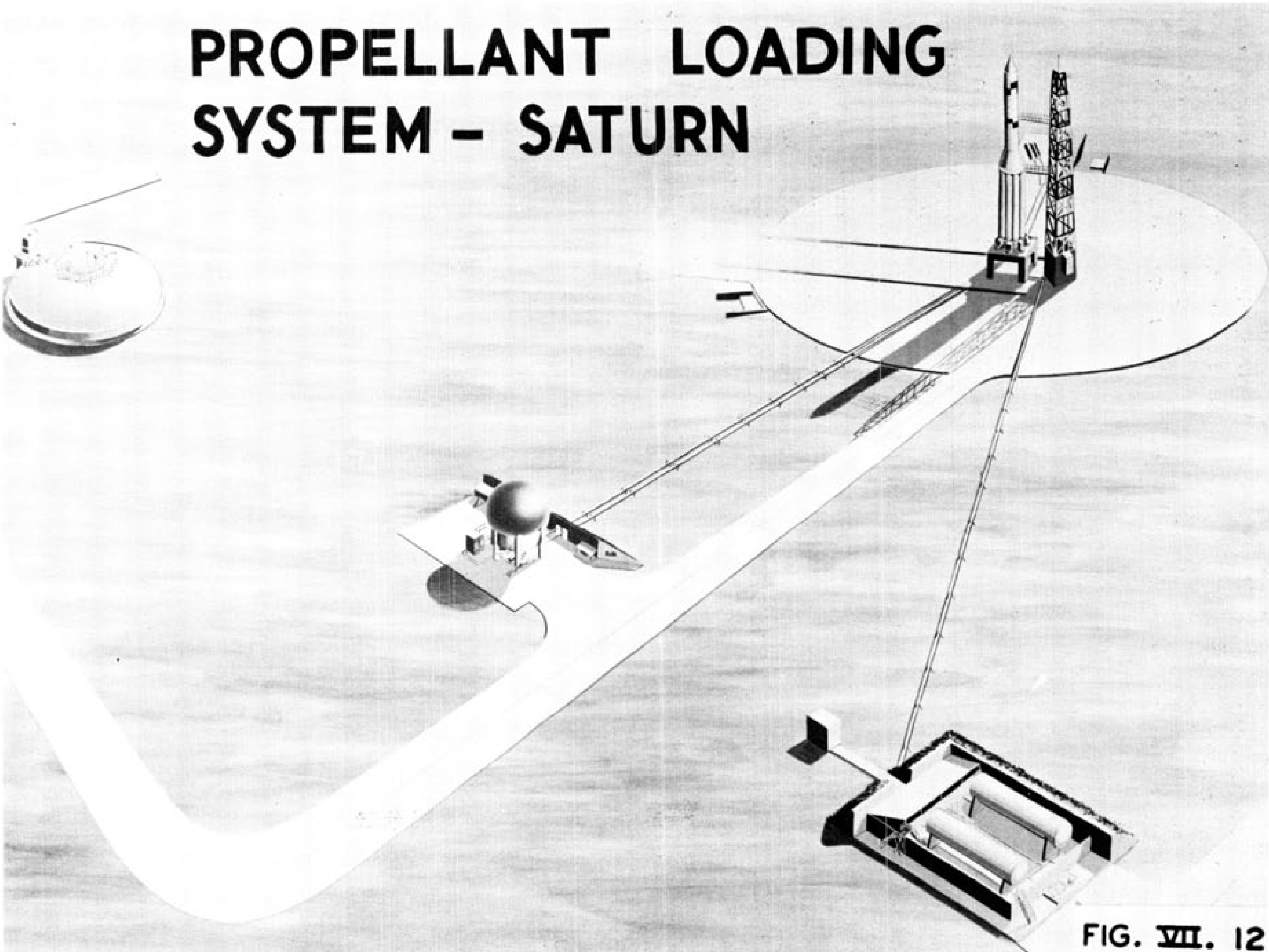


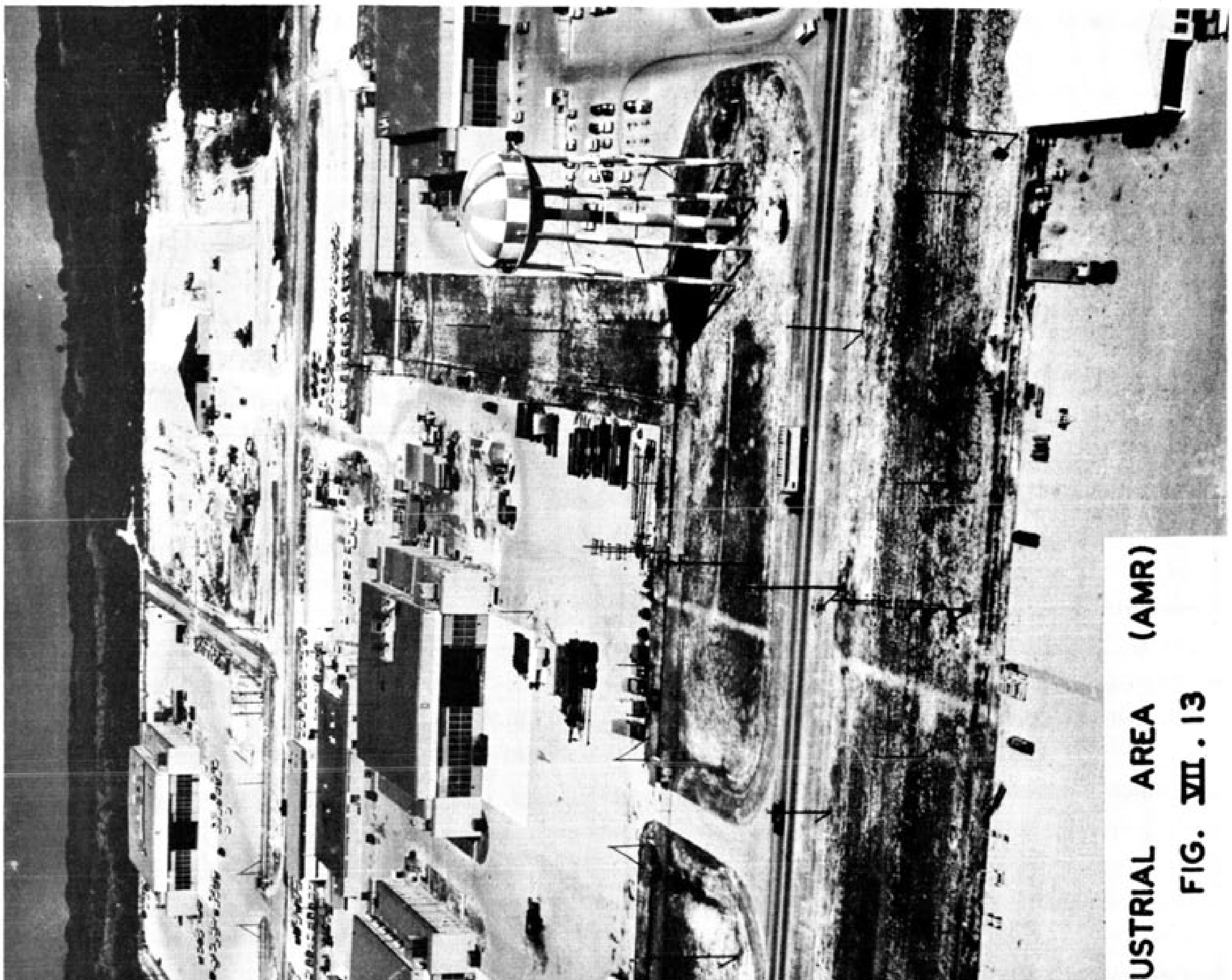
FIG. VII. 12

Studies and initial designs are now underway for liquid hydrogen facilities. The first installation will have a capacity of 90,000 gallons, and will be expanded later to a total capacity of 180,000 gallons. Details such as pipe size, type and size of pumps, insulation, and type of storage tank are being considered in the present study.

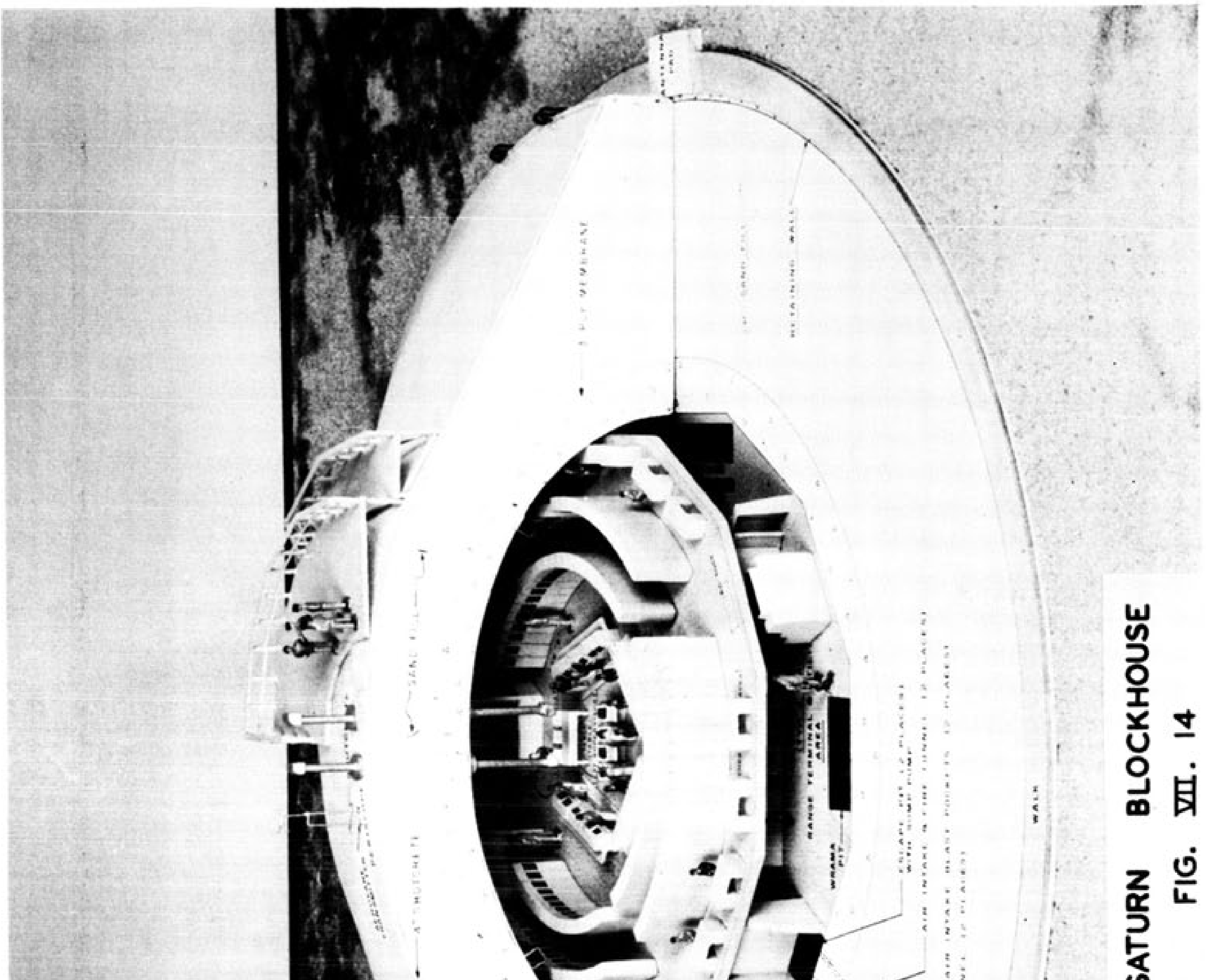
Checkout and Launch Operations

Initial checkout of the guidance and control components will be made in the Industrial Area (see Figure VII.13). This area contains two hangers, an engineering laboratory, administration offices and a supply building, which will be available for support of the SATURN lunar program.

Final checkout will be made at the SATURN launch site. If horizontal checkout is required, it will be accomplished in the stage building. The final checkout will be made with the assembled vehicle in the vertical position on the pad. Final checkout of the vehicle, including all upper stages, will be under the technical supervision of the Huntsville Center. All operations will be controlled from the blockhouse (see Figure VII.14). After completion of the final checkout, and upon completion of the countdown, the vehicle will be launched.



INDUSTRIAL AREA (AMR)
FIG. VII . 13



SATURN BLOCKHOUSE
FIG. VII. 14

APPENDIX A

GUIDANCE COMPUTER SYSTEM

FOR INERTIAL INJECTION

(C)

A block diagram representing the computations to be performed by the guidance computer during the inertial injection phase is shown in Figure A.1.

The accelerometer outputs are applied to a digital resolver in order to effect the proper rotation of the coordinate system. The resolving angles are time programmed into the computer for use during the appropriate portions of the powered flight. The transformation equations which must be solved by the digital resolver are:

$$\dot{\xi}_n = \dot{\xi}_1 \cos\Delta\epsilon_n \pm \dot{\eta}_1 \sin\Delta\epsilon_n \tag{1}$$

$$\dot{\eta}_n = \dot{\eta}_1 \cos\Delta\epsilon_n \mp \dot{\xi}_1 \sin\Delta\epsilon_n \tag{2}$$

In this expression, $\dot{\xi}_1$ and $\dot{\eta}_1$ are the velocity components obtained directly from the integrating accelerometers mounted on the inertial platform with an initial orientation such that the ξ measuring direction is in the flight plane and elevated at an angle of ϵ_0 degrees from the horizontal at the launch point. Therefore, $\Delta\epsilon_n$ ($n = 1, 2, 3, \ldots$) is measured with respect to this initial orientation and $\dot{\xi}_n$ and $\dot{\eta}_n$ are the velocity outputs from the resolvers. For a counter clockwise rotation of the coordinate system, the positive sign in Equation (1) and the negative sign in Equation (2) are used. When the rotation is clockwise the signs are reversed.

Time programs for standard $\dot{\eta}$ and $\dot{\xi}$ velocities are used. These programs are computed from accelerations measured in a Cartesian coordinate system space-fixed in direction and moving in a standard gravitational field. The origin of the coordinate system is initially located at the launch site and is initially oriented to coincide with the platform orientation at launch.

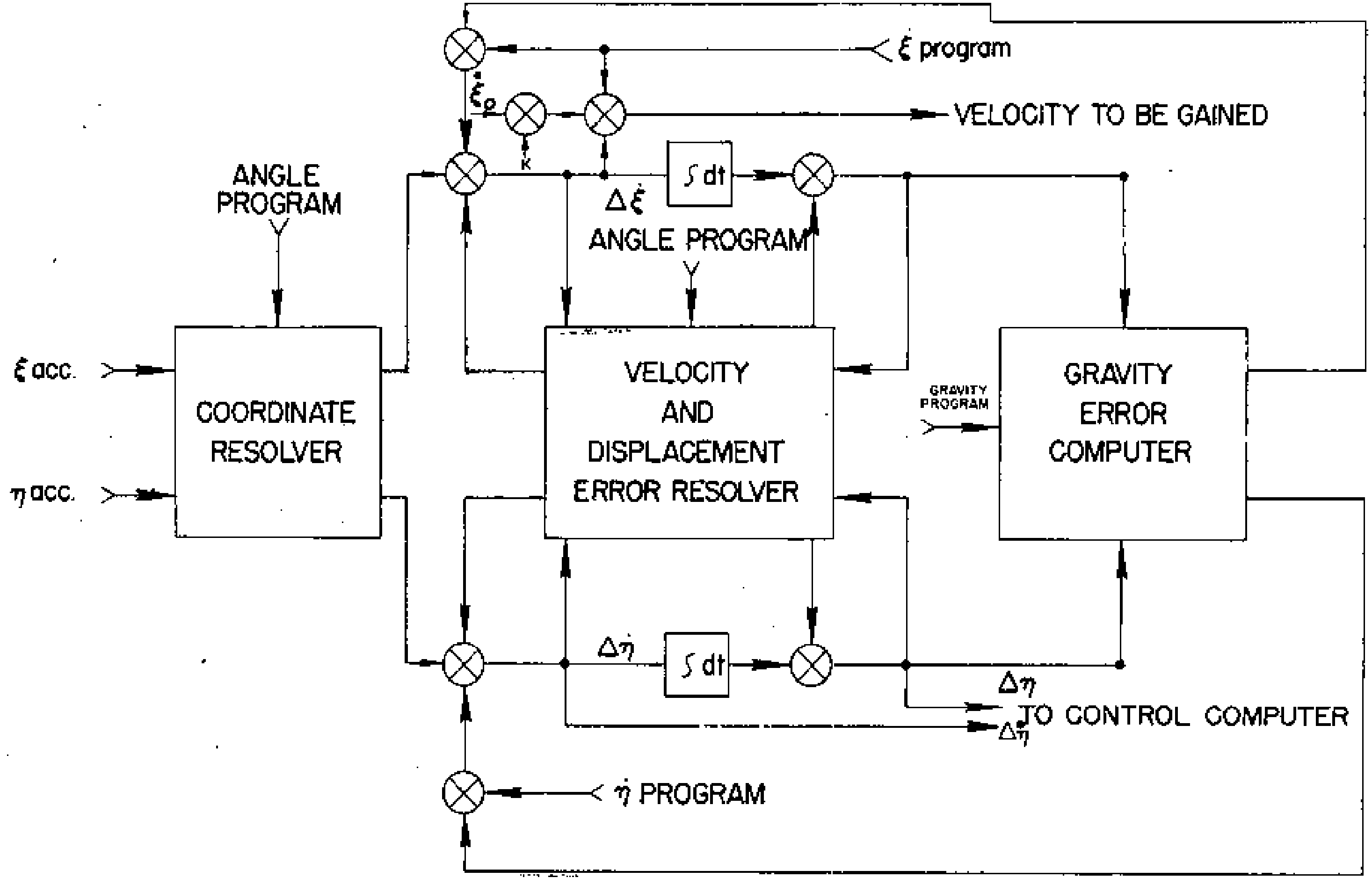


FIG. A.1
INJECTION GUIDANCE COMPUTER

~~CONFIDENTIAL~~

The components of acceleration may be expressed as follows:

$$\ddot{\xi}_p = \ddot{\xi}_s + \ddot{\xi}_{gs} \tag{3}$$

$$\ddot{\eta}_p = \ddot{\eta}_s + \ddot{\eta}_{gs} \tag{4}$$

where $\ddot{\xi}_s$ and $\ddot{\eta}_s$ are the components of vehicle acceleration under standard conditions of thrust, drag, and gravity; $\ddot{\xi}_{gs}$ and $\ddot{\eta}_{gs}$ are the components of gravity acceleration experienced by the missile flying the reference trajectory.

The actual missile may follow a deviated path resulting from non-standard thrust and drag conditions. The gravity acceleration experienced by the actual missile will then be different from the standard. The acceleration components, $\ddot{\xi}_m$ and $\ddot{\eta}_m$, which will be sensed by the accelerometers in the actual missile may be expressed as

$$\ddot{\xi}_m = \ddot{\xi} + \ddot{\xi}_g \tag{5}$$

$$\ddot{\eta}_m = \ddot{\eta} + \ddot{\eta}_g \tag{6}$$

where $\ddot{\xi}$ and $\ddot{\eta}$ are the components of the deviated missile acceleration under conditions of perturbed thrust, drag, and gravity; $\ddot{\xi}_g$ and $\ddot{\eta}_g$ are the components of gravity acceleration experienced by the missile flying the perturbed path.

The difference between the standard acceleration and the actual acceleration then becomes:

$$\ddot{\xi}_p - \ddot{\xi}_m = \left(\ddot{\xi}_s - \ddot{\xi}\right) + \left(\ddot{\xi}_{gs} - \ddot{\xi}_g\right) \tag{7}$$

$$\ddot{\eta}_p - \ddot{\eta}_m = \left(\ddot{\eta}_s - \ddot{\eta}\right) + \left(\ddot{\eta}_{gs} - \ddot{\eta}_g\right) \tag{8}$$

Expressed in incremental form for small deviations these equations become:

$$\Delta \ddot{\xi}_m = \Delta\ddot{\xi} + \Delta \ddot{\xi}_g \tag{9}$$

$$\Delta \ddot{\eta}_m = \Delta\ddot{\eta} + \Delta \ddot{\eta}_g \tag{10}$$

Since the accelerometers are inertial devices they measure only acceleration resulting from thrust and drag. Hence, the difference between the outputs from the accelerometers and the standard program ($\Delta\ddot{\xi}_m$, $\Delta\ddot{\eta}_m$) indicates a deviation from the reference trajectory which is in error by the amounts of $\Delta\ddot{\xi}_g$ and $\Delta\ddot{\eta}_g$. Consequently, the indicated deviations ($\Delta\ddot{\xi}_m$, $\Delta\ddot{\eta}_m$) must be corrected by this amount in order to produce the true deviations.

The standard ξ and η programs are expressed as velocities since the accelerometers are integrating devices, thus

$$\dot{\xi}_p = \int (\ddot{\xi}_s + \ddot{\xi}_{gs}) \; dt \tag{11}$$

$$\dot{\eta}_p = \int (\ddot{\eta}_s + \ddot{\eta}_{gs}) \; dt \tag{12}$$

The indicated deviations are

$$\Delta\dot{\xi}_m = \int (\Delta \ddot{\xi}_a + \Delta \ddot{\xi}_g) \; dt \tag{13}$$

$$\Delta \dot{\eta}_m = \int (\Delta \ddot{\eta}_a + \Delta \ddot{\eta}_g) \; dt \tag{14}$$

The acceleration errors resulting from the non-standard gravitational acceleration are functions of the ξ and η displacement deviations and may be written as

$$\Delta\ddot{\xi}_g = F_1 (\Delta\xi, \Delta\eta) \tag{15}$$

$$\Delta\ddot{\eta}_g = F_2(\Delta\xi, \Delta\eta) \tag{16}$$

The first order terms of the total differentials are then

$$\Delta\ddot{\xi}_g = \frac{\partial\ddot{\xi}_g}{\partial\xi}\Delta\xi + \frac{\partial\ddot{\xi}_g}{\partial\eta}\Delta\eta \tag{17}$$

$$\Delta\ddot{\eta}_g = \frac{\partial\ddot{\eta}_g}{\partial\xi}\Delta\xi + \frac{\partial\ddot{\eta}_g}{\partial\eta}\Delta\eta \tag{18}$$

The ξ and η displacement deviations ($\Delta\xi$, $\Delta\eta$) may be approximated by continuously integrating the indicated respective velocity deviation from launch. If then, the above partial derivatives are programmed into the computer, the gravity errors can be approximated and used to correct the indicated deviations. This computation is included in the diagram of Figure A.1. In practice a more detailed investigation of the nature and magnitudes of these gravity partials may indicate the possibility of programming a series of constants rather than a continuously varying function. This will result in a desirable simplification. A more detailed investigation is required in order to establish the best method of making these gravity corrections.

The sum of the ξ velocity program and the gravity corrected ξ velocity deviation, $\Delta\dot{\xi}$, is compared with a ξ velocity presetting, "$\dot{\xi}_0$", and a preset constant "K" to provide a velocity-to-be-gained to facilitate CENTAUR booster cut-off and subsequent ignition of the vernier engines. The constant "K" is employed to compensate for thrust decay of the CENTAUR. The cut-off equation may be expressed as

$$\dot{\xi}_{gain} = \dot{\xi}_p + \Delta\dot{\xi} + \dot{\xi}_0 + K \tag{19}$$

when $\dot{\xi}_{gain}$ becomes zero, the CENTAUR cut-off signal is initiated and the vernier engines are ignited.

The vernier cut-off equation takes the form

$$\dot{\xi}_{gain} = \dot{\xi}_p + \Delta\dot{\xi} + \dot{\xi}_0 \tag{20}$$

when $\dot{\xi}_{gain}$ becomes zero, the vernier cut-off signal is initiated.

A more detailed investigation of the nature of the cut-off equation required may indicate the necessity for the inclusion of an η term. Moreover, a cut-off equation similar in form to that described later for the radio vernier phase may prove more applicable. Further studies must be made before this can be ascertained.

Since the actual missile may fly a perturbed path, it will deviate in velocity and displacement from the reference. These deviations must be transformed to the new coordinate system each time a coordinate transformation is made, and will appear as initial conditions. The transformation equations are similar to the ones used to transform the outputs from the accelerometers. These equations may be written as

$$\Delta\xi_n = \Delta\xi_{n-1}\cos\Delta\epsilon_n + \Delta\eta_{n-1}\sin\Delta\epsilon_n \qquad (21)$$

$$\Delta\eta_n = \Delta\eta_{n-1}\cos\Delta\epsilon_n - \Delta\xi_{n-1}\sin\Delta\epsilon_n \qquad (22)$$

$$\Delta\dot{\xi}_n = \Delta\dot{\xi}_{n-1}\cos\Delta\epsilon_n + \Delta\dot{\eta}_{n-1}\sin\Delta\epsilon_n \qquad (23)$$

$$\Delta\dot{\eta}_n = \Delta\dot{\eta}_{n-1}\cos\Delta\epsilon_n - \Delta\dot{\eta}_{n-1}\sin\Delta\epsilon_n \qquad (24)$$

These equations are solved for each rotation during the flight. The angle is varied in the same manner as was done in the transformation of the accelerometer outputs.

The η velocity and displacement deviations are applied to the control computer and the vehicle is caused to fly a path which will continuously reduce these deviations to zero.

During the powered flight a gravity tilt program is employed to cause the missile to fly the standard trajectory. In case one of the SATURN engines fails, the gains of the control system will be adjusted to be compatible with the magnitude of the deviations arising from such an occurrence.

A simplified block diagram of the cross range guidance system is shown in Figure A.2. Null guidance is used throughout the powered flight. The output from the ζ integrating accelerometer is integrated to obtain the ζ displacement. Both ζ velocity and displacement are then applied to the control computer which causes the missile to be constrained to the desired flight plane. It may be possible in the case of the cross range computer to employ analog devices. This possibility must be investigated further before it can be definitely decided if this would be more advisable or economical than using a portion of the main digital computer.

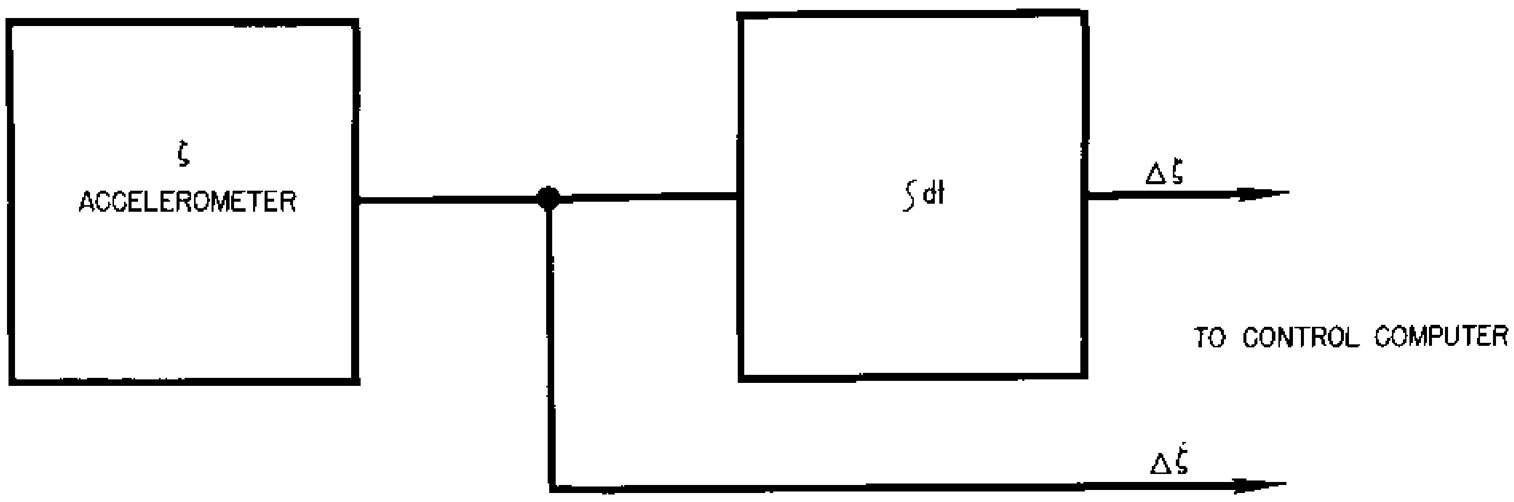


FIG. A.2

**CROSS RANGE
INJECTION GUIDANCE SCHEME**

APPENDIX B

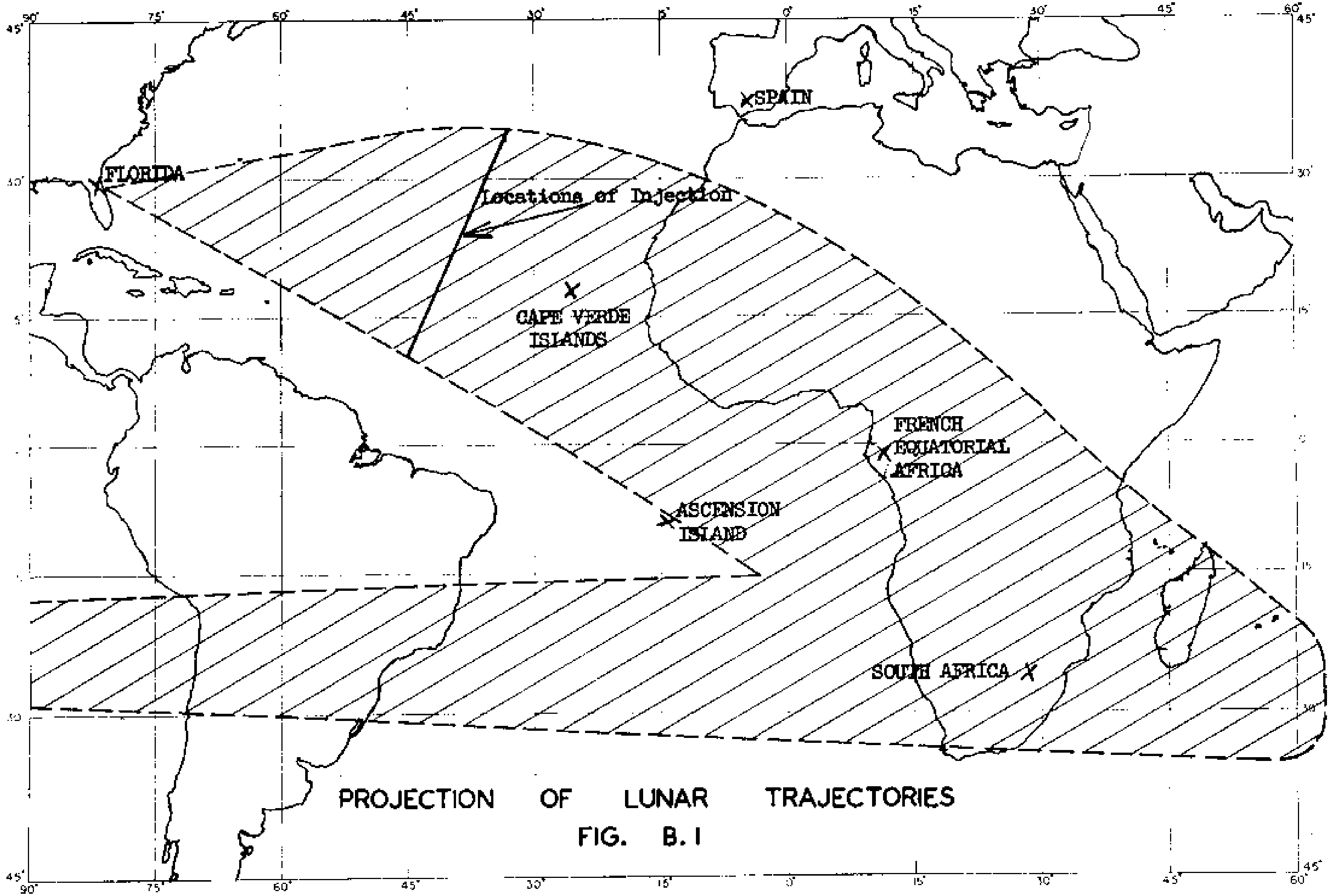
POSSIBLE LUNAR TRACKING STATION LOCATIONS

After injection into the lunar trajectory, the rapidly increasing altitude of the lunar vehicle will make it visible for tracking purposes to all locations south of the forty degrees North latitude. During the early hours immediately after powered flight, however, the relatively low altitude greatly restricts the visibility of a given trajectory.

For a first orientation six possible flight paths have been investigated, covering the range of variations feasible with the SATURN B-1 configuration. The launch azimuth, measured East from North, the injection altitude, and the injection path angle from local vertical for these trajectories are shown below. The velocity of each is 100 m/s below escape velocity.

	Azimuth (deg)	Path Angle (deg)	Altitude (km)
1	70	70	898
2	70	90	118
3	90	80	476
4	90	90	121
5	110	80	491
6	110	90	139

The launch azimuth may vary by forty degrees, causing a wide variation in the projection of the trajectory upon the surface of the earth. The twenty degree range in path angle and the 780 kilometer range in injection altitude also produce a similar variation in the altitude-time history of the vehicle, and hence the area of visibility. The envelope of the projections of the six trajectories upon the surface of the earth is shown in Figure B.1. A number of possible tracking locations are also shown. Of these locations, Spain, South Africa, and Florida are sites already proposed for space tracking facilities. Ascension Island is the location of an FPS-16 radar, and the Cape Verde Islands and French Equatorial Africa are locations



PROJECTION OF LUNAR TRAJECTORIES

FIG. B.1

chosen for their proximity to the region of injection points.

Figures B.2, B.3, & B.4 show the envelopes for the six trajectories of the elevation-time histories of each of the aforementioned station locations. A ten-degree horizon mask is shown as a practical visibility limit imposed by local horizon obstacles and by strong refraction effects.

It will be seen that Spain, the Cape Verde Islands, and Ascension Island in some cases acquire the vehicle slightly earlier because of their proximity to the injection point. In other cases, however, they may suffer a delay of up to four hours before acquiring continuous tracking, due to the low altitude and rapid ground coverage in the early hours. The South African location, however, acquires only slightly after the other stations, and in all cases then maintains continuous tracking.

The Florida station can acquire at best five hours after injection. This is representative of the time that stations in the continental United States will begin to be applicable.

Another factor of importance in tracking feasibility is the angular rate required of high-gain antennas in order to follow the vehicle. The envelope for the six trajectories of angular rates required by the South African and Cape Verde Island stations (one near injection point and one more distant) is shown in Figure B.5. For comparison the maximum angular rate possible in the slow, high-precision mode of the 85-foot Goldstone, California, high-gain antenna is approximately six degrees per minute.[1]

In summary, the choice of trajectory parameters will heavily influence the possible locations of tracking stations applicable during the early hours after injection for lunar flight missions. However, the South African location presently under construction would appear to permit, at least for the present vehicle configuration, continuous tracking beginning very shortly after injection. The angular tracking rates encountered during this early phase should present no problem to presently proposed high-gain antennas.

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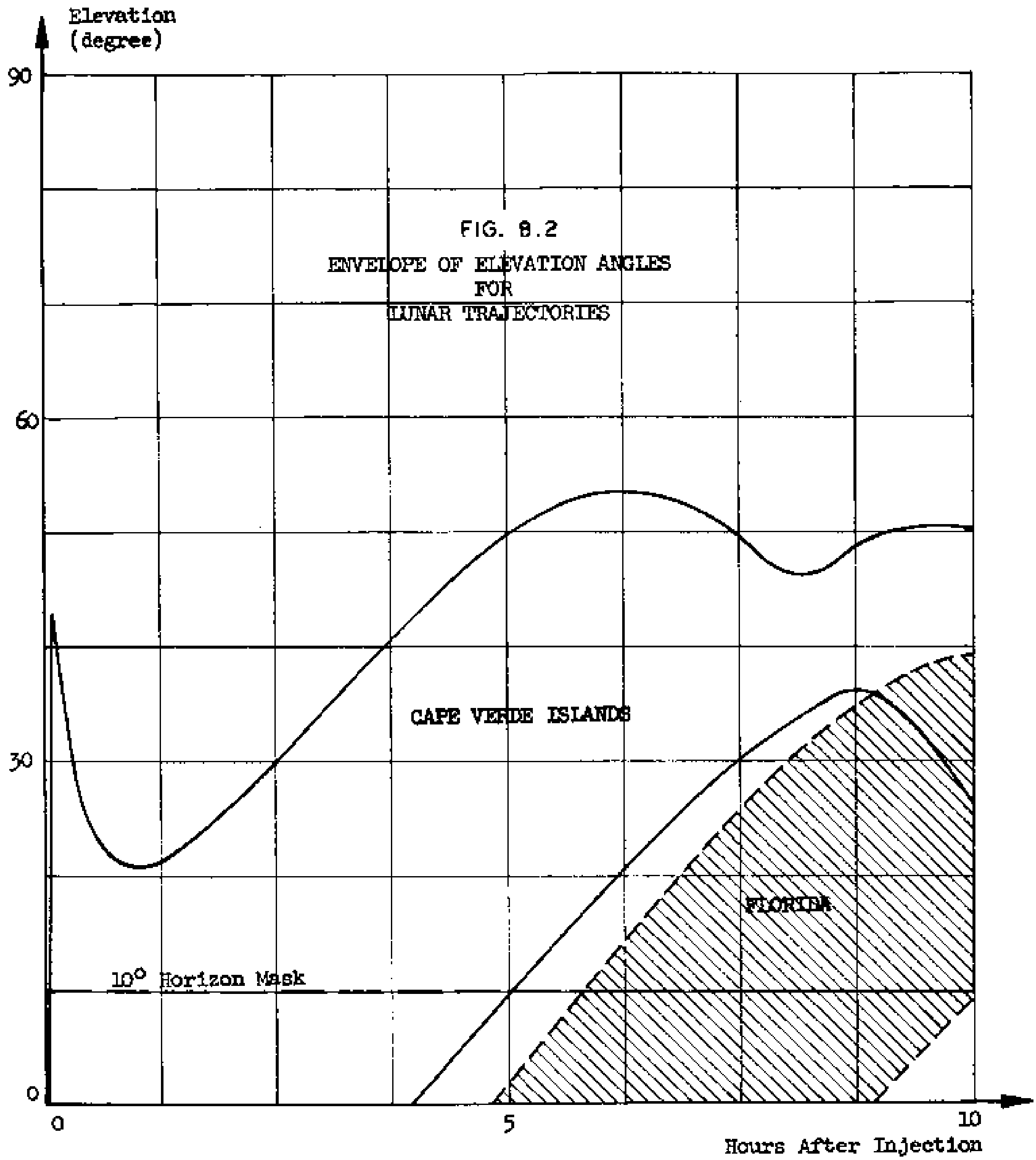
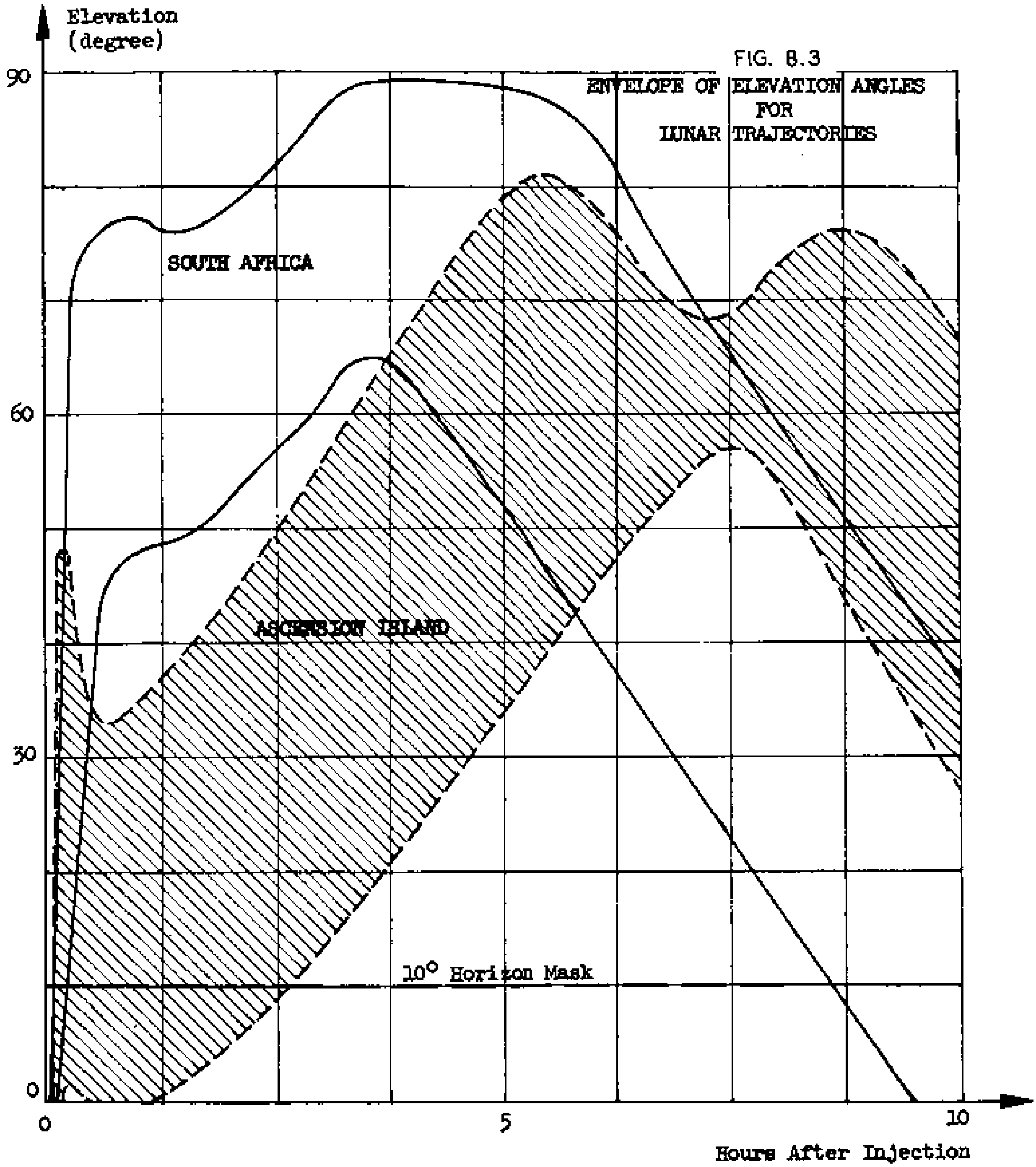
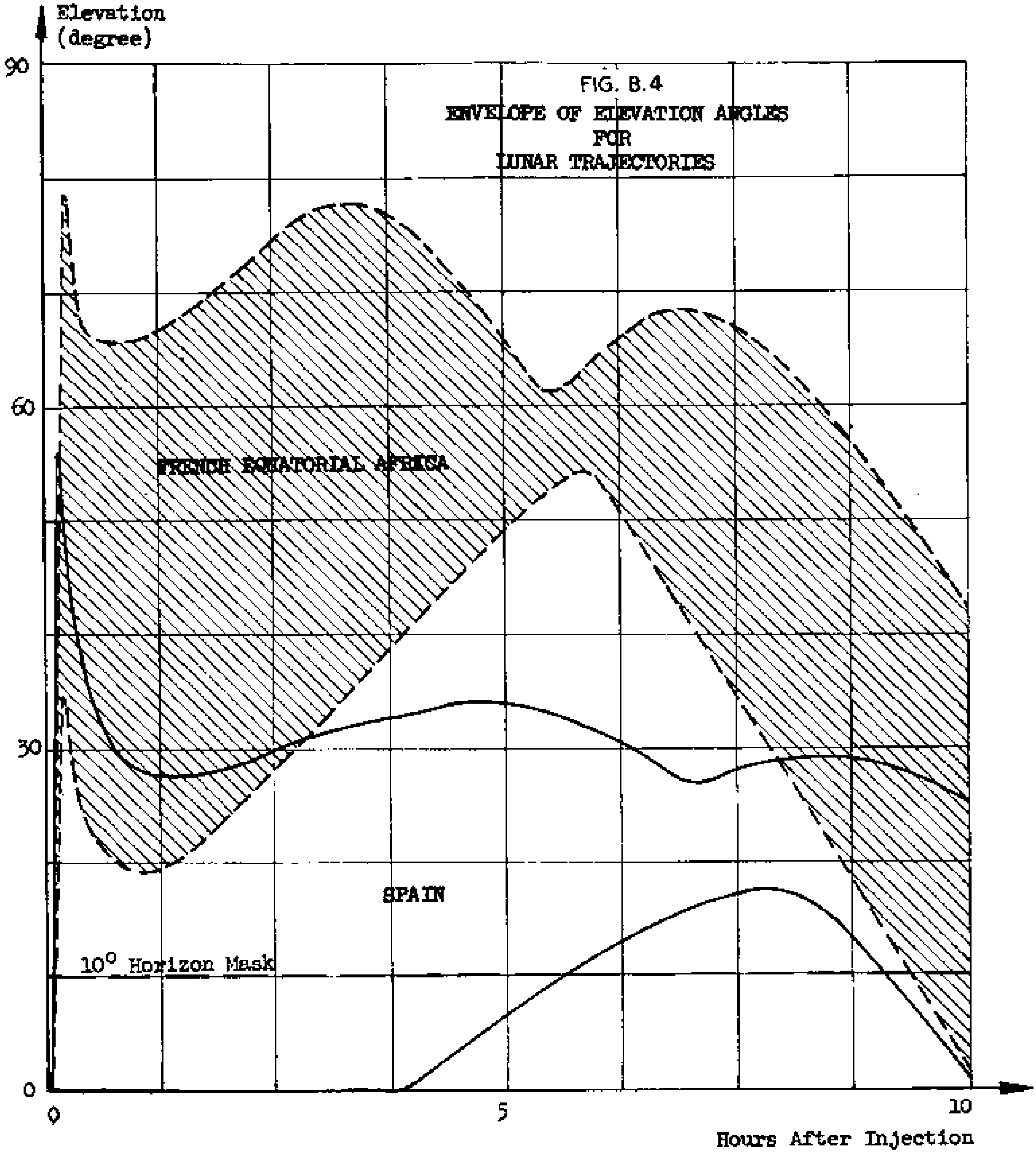
[1] Space Research Summary No. 1; Jet Propulsion Laboratory, 1 August 1958, p.57.


Elevation
(degree)
90
60
30
0
FIG. 8.2
ENVELOPE OF ELEVATION ANGLES
FOR
LUNAR TRAJECTORIES
CAPE VERDE ISLANDS
FLORIDA
10° Horizon Mask
0
5
10
Hours After Injection


Elevation
(degree)
90
60
30
0
FIG. B.3
ENVELOPE OF ELEVATION ANGLES
FOR
LUNAR TRAJECTORIES
SOUTH AFRICA
ASCENSION ISLAND
10° Horizon Mask
0
5
10
Hours After Injection



Elevation
(degree)
90
60
30
0
FIG. B.4
ENVELOPE OF ELEVATION ANGLES
FOR
LUNAR TRAJECTORIES
FRENCH EQUATORIAL AFRICA
SPAIN
10° Horizon Mask
0
5
10
Hours After Injection

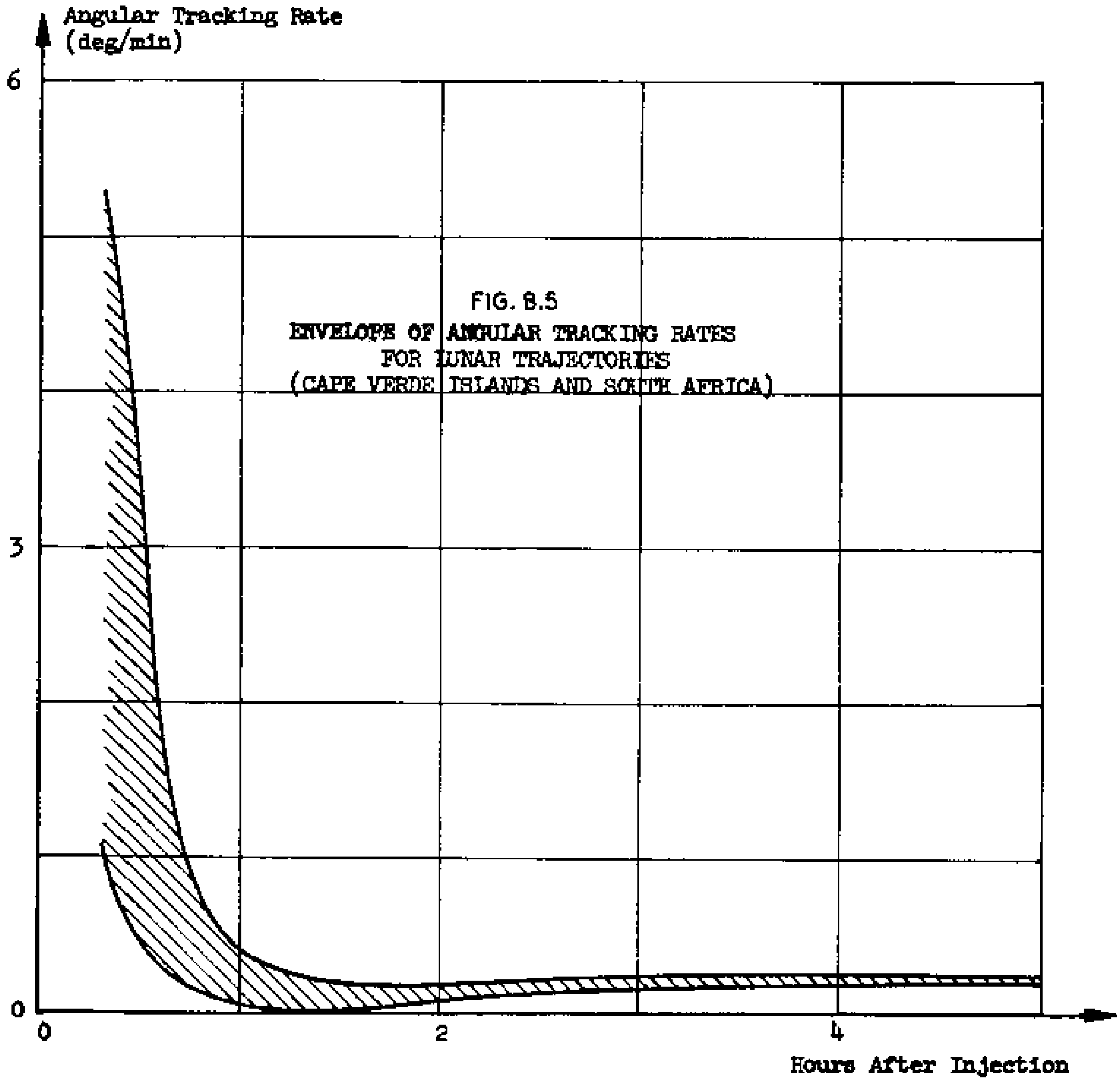


FIG. 8.5
ENVELOPE OF ANGULAR TRACKING RATES
FOR LUNAR TRAJECTORIES
(CAPE VERDE ISLANDS AND SOUTH AFRICA)

APPENDIX C

THE DORA TRACKING SYSTEM

Principle of Operation

DORA is a precision CW-phase comparison and Doppler system employing techniques that have proved successful in short baseline systems. The use of recent developments in frequency standards and investigation into phase stability of low frequency propagation make a long base line system feasible.

Each DORA ground system consists of three or more stations located to give a suitable geometry. Two phase-coherent frequencies are transmitted from one of the ground stations to a transponder on the vehicle, and offset or doubled in frequency (whether offset or doubled is immaterial to the principle) with the coherence maintained and transmitted from the vehicle to the ground receivers. Frequency offset has the advantage of greater flexibility in use of frequencies while frequency doubling has the advantage of simplicity of the transponder.

The frequencies received by the ground receivers are delayed in phase by an amount which is proportional to the frequency and to the travel time of the wave. By comparing the phase of each frequency received with the phase of a reference frequency, representing a phase delay of zero, a phase delay which is known to be less than 2π may be determined directly. But, generally, the information is ambiguous in the sense that it is uncertain whether the phase delay at each received frequency is within the range 0 to 2π or within 2π to 4π, etc. However, by comparing a coincidence of the two reference frequencies with the same coincidences of the two received frequencies, the ambiguity can be resolved over a time interval representing the period of the difference between the two frequencies (The offset or doubling of the frequency returned from the transponder must be taken into account.). For instance, the beat frequency of the two transmitted signals can be compared in phase with the beat frequency of the two received signals. Then a phase delay of 2π corresponds to a travel time equal to the period of the beat frequency. The frequency, or if needed, a number of frequencies, can be chosen as any value necessary to resolve the ambiguities for a specific case.

If a receiver is located at the transmitter site, the slant range and velocity component in the direction of the station can be determined from information returned from the vehicle. The reference frequencies in this case are derived from the same source (atomic standard) as those transmitted to the vehicle. At the receiver sites away from the transmitter, the reference frequencies are derived from phase stable frequency standards of the same type as used with the transmitter.

The information obtained at a receiver site which is away from the transmitter contains the algebraic sum of the slant range from transmitter to vehicle plus the slant range from vehicle to the receiver and the algebraic sum of the velocity components of the vehicle in the direction of the transmitter plus the velocity component in the direction of the receiver.

The system as described above requires only one transmitter for each system. For greater flexibility, each station should have transmitting as well as receiving capability; then any combination of three or more stations could be used since any of the stations could interrogate the transponder. Also, if a transmitter is located at each receiver site, the system could be used in a different mode by operating each receiver-transmitter site as an independent unit. Use of the three sets of frequencies will give greater accuracy since the error of the master station is not included in the measurements of the other two stations. The transponder, however, is made more complex and it must radiate three times the power over that required for the first mode of operation.

The most obvious technique by which the two phase-coherent frequencies may be transmitted is by amplitude modulation of a carrier by an appropriately lower frequency, both frequencies being derived from the same standard to provide the phase coherence. Unwanted sideband may be suppressed or left for other uses. The frequency difference between two transmitted frequencies determines the range of ambiguity resolution.

Data reduction for the proposed system is comparatively simple. The data from each station can be transmitted to a central data processing point. The data from each station consists of the phase delay and phase rate (Doppler) of the transmitted wave as seen by that station and compared with that station's reference standard, rate of change of phase delay, and time. This can be transmitted in digitalized form at time intervals of 0.1 second if necessary. Another possible means of transmitting the data is by direct analog transmission of the compared phase over a low frequency link having a minimum of propagation variation. The method preferred will depend upon the exact application and results of further investigation. Translation of the data into range and velocity components relative to each station is performed at the central point. With the information in that form, the problem is then the solution of a relatively simple geometrical problem. This solution can be obtained with high accuracy by employing a system designed for existing Doppler systems.

Requirements for communications between stations are for time synchronization and in maintaining phase reference between frequency standards. With the accuracies attainable with atomic standards, a continuous phase reference may not be needed but it is needed when the standards are first put into operation, or after being shut down for servicing.

Appendix D

SOFT LUNAR LANDING SCHEMES, EMPLOYING RADIO ALTIMETER FOR THE CONTROL OF POWERED DESCENT (C)

Many investigations have been undertaken to determine theoretically the best approach for lunar or planetary landing maneuvers. For instance, conditions for lowest propellant consumption have been studied and optimum guidance laws have been derived. Most of these investigations, however, do not indicate a practical way to apply the results with guidance equipment as it can be built today.

In this study an attempt was made to tackle the problem from the instrumentation side, looking for a simple guidance scheme that employs only equipment which will be available within the next few years.

D. 1. General

Under the assumption that no active radar beacon (or similar homing device) is available at the desired lunar landing site the radio altimeter will offer the best possibility for altitude measurements during the landing maneuver. Especially for lunar trajectories resulting in perpendicular or nearly perpendicular approach to the moon's surface, the pencil beam type altimeter with a beamwidth of a few degrees seems to be the most favorable instrument. However, inevitable shortcomings of the radio altimeter will influence the basic layout of the landing scheme.

If the braking operation is to be initiated at a preset altitude by command from the radio altimeter, the necessary altimeter range is determined by the total distance of the braking maneuver. This distance will be at least 100 kilometers for the SATURN landing vehicle with a 20,000 pound thrust engine. Using radio altimeters for such long ranges has the following disadvantages and restrictions:

1. Apart from instrument errors, relatively large altitude errors will be caused by the unevenness of the reflecting surface. For instance, an altimeter with a beamwidth of 4 degrees will cover an area of about 180 km^2 of the moon's surface, as seen from an altitude of 200 km. This area may contain elevations and depressions of several hundred meters. Consequently, the actual distance to the landing site can also differ by several hundred meters from the altimeter indication, even in case the altimeter is designed to measure the average distance to the area covered by the altimeter beam.

2. Since the location of the surface area covered by the radio beam will continuously change with altitude and vehicle attitude the measured range will fluctuate. Consequently, no precise velocity information can be derived from the range information.

3. During engine operation all radio measurements (and optical measurements including TV) are disturbed by the rocket exhaust. Flame attenuation caused by ionization of the hot gas may possibly make radio measurements impracticable, at least at the high altitudes encountered during the first part of the braking maneuver. To be on the safe side it is subsequently assumed that all altitude measurements are only made during thrustless periods.

As a consequence of these properties of the radio altimeter, it appears that the only practicable sequence for the powered descent will be the "open loop" procedure of:

a. Ignition of the braking rocket at preset altitude by the altimeter and thrust application according to a thrust program, possibly controlled by accelerometers. Ignition altitude and thrust program are precalculated from nominal values of approach velocity, cut-off altitude, cut-off velocity and engine performance data.

b. Free fall due to the moon's gravity until touch down.

If the sequence a-b is applied only once, large deviations in cut-off velocity and cut-off altitude are encountered for three reasons:

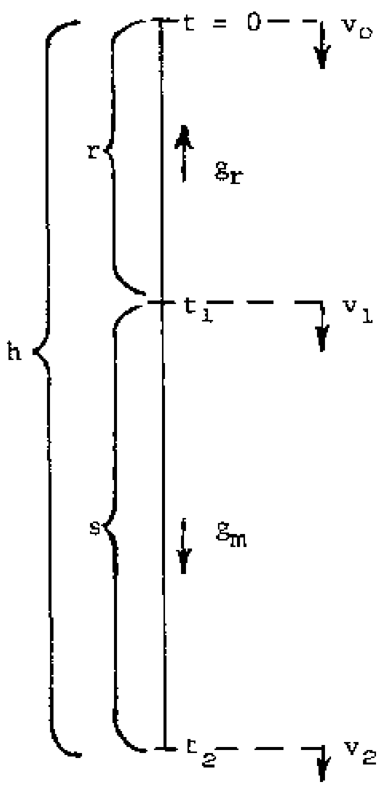
1. Because of the high ignition altitude, large dispersion in ignition altitude will occur. These dispersions produce errors in cut-off height of the same magnitude, namely several hundred meters.
2. Deviations of the approach velocity from the nominal value for which the thrust program is calculated are retained during thrust application and produce both errors in cut-off velocity and cut-off altitude. The altitude error is proportional to the burning time.
3. Deviations from the precalculated deceleration program produce errors in cut-off velocity and cut-off altitude. Both errors also increase with burning time or braking distance.

A free fall period immediately before touch down (see "b" above) should be inserted for separation of the braking stage and to reduce contamination of the landing area. Narrow tolerances must be placed upon the cut-off height and velocity of the preceding thrust period if the touchdown velocity is to be kept within a few m/s. It is easily shown that the errors arising from the above mentioned "one-stop" scheme would surpass those tolerances.

Much more favorable results will be obtained if the sequence a-b (subsequently designated as "braking phase") is employed several times before touchdown. Then, the altimeter errors will decrease for each braking phase with decreased altitude. In addition, the following error analysis shows that all other errors can be minimized by proper choice of ignition altitude, burning time, and free fall time for each of the braking phases.

D. 2. Error Analysis

First, one single braking phase is considered, consisting of a thrust period and a free fall period, as shown:



Definitions:

v_o = approach velocity

v_1 = cut-off velocity

v_2 = end velocity

r = braking distance

s = free fall distance

h = total distance

g_r = deceleration, generated by braking thrust, assumed to be constant over distance r

g_m = acceleration due to moon's gravity, assumed to be constant over r

t_1 = thrust duration

t_{ff} = free fall duration = $t_2 - t_1$

$\Delta v = \int_o^{t_1} g_r \, dt = g_r t_1 = v_o - v_1$

$a = g_m / g_r$

From the equation of motion the end velocity is obtained:

$$v_2^2 = v_o^2 + g_r^2 t_1^2 - 2g_r t_1 v_o + 2g_m h - 2t_1 g_m v_o + g_m g_r t_1^2 \tag{1a}$$

$$= v_o^2 + (\Delta v)^2 - 2v_o (\Delta v) + 2g_m h - 2 \frac{g_m}{g_r} v_o (\Delta v) + \frac{g_m}{g_r} (\Delta v)^2 \tag{1b}$$

For obtaining the dependency of v_2 from variations in g_r, Δv, t_1, h the equations (1) are partially differentiated, yielding coefficients for Taylor series. If g_r and t_1 are considered to be independent variables (consequently $\Delta v = g_r t_1$ = dependent) it follows from Equation (1a):

$$\frac{\partial v_2}{\partial g_r} = \frac{t_1^2 (g_r + g_m/2) - t_1 v_o}{v_2} = t_1 \frac{\Delta v (1 + a/2) - v_o}{v_2} \tag{2a}$$

$$\frac{\partial v_2}{\partial g_r} = 0 \quad \text{for} \quad \Delta v = \frac{v_o}{1 + a/2} \tag{3a}$$

$$\frac{\partial^2 v_2}{\partial g_r^2} \approx \frac{t_1^2}{v_2} \tag{4a}$$

$$(\Delta v_2)_{g_r} \approx \frac{\Delta g_r}{g_r} \Delta v \frac{a (\Delta v)/2 - v_1}{v_2} + \left(\frac{\Delta g_r}{g_r}\right)^2 \frac{(\Delta v)^2}{2 v_2} \tag{5a}$$

$$\frac{\partial v_2}{\partial t_1} = \frac{t_1 (g_r^2 + g_m g_r) - v_o (g_r + g_m)}{v_2} = \frac{-v_1}{v_2} (g_r + g_m) \tag{6a}$$

$$\frac{\partial v_2}{\partial t_1} = 0 \text{ for } v_1 = 0 \quad \text{or} \quad \Delta v = v_o \tag{7a}$$

$$\frac{\partial^2 v_2}{\partial t_1^2} \approx \frac{g_r^2 + g_r g_m}{v_2} \tag{8a}$$

$$(\Delta v_2)_{t_1} \approx -\Delta t_1 \frac{v_1}{v_2} (g_r + g_m) + (\Delta t_1)^2 \frac{g_r^2 + g_r g_m}{2 v_2} \tag{9a}$$

In case Δv and g_r are the independent variables (consequently $t_1 = \frac{\Delta v}{g_r}$ = dependent) equation (1b) yields

$$\frac{\partial v_2}{\partial g_r} = \frac{\Delta v}{v_2} \frac{g_m}{g_r^2} \left(v_o - \frac{\Delta v}{2} \right) \tag{2b}$$

$$\frac{\partial v_2}{\partial g_r} = 0 \quad \text{for} \quad \Delta v = 2\ v_o \tag{3b}$$

$$\frac{\partial^2 v_2}{\partial g_r^2} \approx \frac{\Delta v}{v_2} \frac{2 g_m}{g_r^3} \left(v_o - \frac{\Delta v}{2} \right) \tag{4b}$$

$$\left(\Delta v_2 \right)_{g_r} \approx \left(\frac{\Delta g_r}{g_r} - \left(\frac{\Delta g_r}{g_r} \right)^2 \right) \frac{a\ (\Delta v)}{v_2} \left(v_o - \frac{\Delta v}{2} \right) \tag{5b}$$

$$\frac{\partial v_2}{\partial (\Delta v)} = \frac{(1 + a)\ (\Delta v - v_o)}{v_2} = - \frac{v_1}{v_2} (1 + a) \tag{6b}$$

$$\frac{\partial v_2}{\partial (\Delta v)} = 0 \quad \text{for} \quad v_1 = 0 \quad \text{or} \quad \Delta v = v_o \tag{7b}$$

$$\frac{\partial^2 v_2}{\partial (\Delta v)^2} \approx \frac{1 + a}{v_2} \tag{8b}$$

$$(\Delta v_2)_{\Delta v} \approx \frac{\Delta(\Delta v)}{\Delta v} (1 + a) \frac{v_1 \cdot (\Delta v)}{v_2} + \left(\frac{\Delta(\Delta v)}{\Delta v} \right)^2 (1 + a) \frac{(\Delta v)^2}{2\ v_2} \tag{9b}$$

From Equations (1a) or (1b) also results:

$$\frac{\partial v_2}{\partial v_o} = \frac{v_o - \Delta v\ (1 + a)}{v_2} \tag{10}$$

$$\frac{\partial v_2}{\partial v_o} = 0 \quad \text{for} \quad \Delta v = \frac{v_o}{1 + a} \tag{11}$$

$$\frac{\partial^2 v_2}{\partial v_o^2} \approx \frac{1}{v_2} \tag{12}$$

$$(\Delta v_2)_{v_o} \approx \Delta v_o \frac{v_o - \Delta v\ (1 + a)}{v_2} + (\Delta v)^2 \frac{1}{2\ v_2} \tag{13}$$

$$\frac{\partial v_2}{\partial h} = \frac{g_m}{v_2} \tag{14}$$

$$\frac{\partial^2 v_2}{\partial h^2} \approx 0 \tag{15}$$

$$(\Delta v_2)_h \approx \Delta h \frac{g_m}{v_2} \tag{16}$$

The physical meaning of the above quantities Δg_r, Δt_1, $\Delta(\Delta v)$, Δh and Δv_2 is:

Δg_r = deviation of the deceleration from the programmed magnitude due to errors in thrust level or deviations of the vehicle mass from the nominal value.

Δt_1 = errors in thrust duration. For the case of programmed thrust duration timing errors can be neglected, because program timers can be built with a high degree of accuracy. But for simplicity, the effects of irregularities in thrust build up and/or thrust decay are expressed as errors in burning time.

$\Delta(\Delta v)$ = deviations of the integral $\int_0^{t_1} g_r\, dt$ from the nominal value

Δh = deviations of the total distance of the braking phase caused by altimeter errors. Δh is the sum of instrument errors and errors due to the unevenness of the reflecting surface. For all braking phases except the last one both altimeter measurements limiting the braking phase contribute to Δh.

Δv_2 = deviations of the end-velocity, caused by any one of the previous disturbances.

D. 3. Error Discussion

The above equations show that all errors Δv_2 caused by the various disturbances are inversely proportional to the end velocity v_2. Furthermore, equations (3a), (7) and (11) reveal that the errors Δv_2 caused by disturbances Δg_r, Δt_1, $\Delta(\Delta v)$ and Δv_0, can be minimized by proper choice of Δv for each braking phase. The condition for that is $\Delta v \approx v_0$, if $a = \frac{g_m}{g_r}$ is of small magnitude. The condition $\Delta v \approx v_0$ means a full stop, or nearly a full stop after thrust application.

From this result, and from the fact that the altimeter errors will decrease with decreasing altitude, the conclusion can be drawn that subdivision of the powered descent into three braking phases will essentially reduce the effects of all errors and thus relax the requirements for the terminal guidance instrumentation. The three braking phases will have different effects with respect to error reduction:

First braking phase. The main purpose of this phase is reduction of the approach velocity with the effect that the subsequent phases are reduced in length. Principal error sources are uncertainties in the altimeter measurements. Equation (16) shows that the velocity v_2 after the free fall period should be as high as possible. This can be reached by a long free fall time t_{ff} or by a relatively high cut-off velocity *$v_1 = v_0 - \Delta v$, the latter way being less costly with respect to propellant consumption.

* Because of the high altitudes of the first braking phase relatively large altimeter errors are inevitable and will constitute the primary error source. For this reason, further error reduction by satisfying the condition $v_1 \approx 0$ (i.e., $\Delta v \approx v_1$, see equations (3), (7) and (11)) is not as important here as it is for the following braking phases.

Second braking phase: The main purpose is compensation of the relatively large velocity error remaining from the preceding phase. Equation (11) shows that this can be done best by satisfying the condition $\Delta v = v_o (1 + a)$ i.e. by inserting a nearly full stop. Since $a = \frac{g_r}{g_m}$ is of small magnitude, this condition minimizes at the same time errors due to deviations from the thrust program, as seen from equations (3a) and (7). The reduced ignition altitude and braking distance of this phase result in another reduction of errors caused by thrust deviations and reduction of errors arising from altimeter uncertainties.

Third braking phase: As mentioned before, this phase is inserted to bring the vehicle to a full stop or nearly full stop at low altitude (e.g. 50 to 100 m). Free fall from this altitude is used for separation and removal of the braking stage, and to reduce contamination of the landing site by the rocket blast. Since the initial velocity is very small the ignition altitude is low and the thrust duration short. This fact, and the condition $\Delta v \approx v_o$ result in small variations in cut-off altitude and end-velocity. Furthermore, small errors in the initial velocity v_o, as a result of the minimization of Δv_2 during the second phase, lead to small variations in cut-off altitude.

D. 4. Numerical Evaluation

The terminal guidance scheme described in Section II.6 utilizes results of the previous part of this appendix for error minimization. The layout of the three braking phases (see Fig. II.16) was based on the following tolerances:

Tolerances in h due to altimeter uncertainties

$\Delta h = \pm 1000$ m for the first braking phase

$= \pm 100$ m for the second braking phase

$= \pm 10$ m for the third braking phase

Tolerance in approach velocity: (at beginning of the first braking phase)

$\Delta v_0 = \pm 10$ m/s

Tolerance in thrust duration due to irregularities of thrust build up and thrust decay:

$\Delta t_1 = 0.05$ s

The tolerable error in cut-off altitude (nominal s = 69 m) and touch-down velocity (nominal $v_2 = 15$ m/s) of the third braking phase are assumed to be:

$$\Delta s = \pm 10 \text{ m} \tag{17}$$

$$\Delta v_2 = +5 \text{ m/s}$$

The numerical procedure for the error calculation was as follows: starting with the last assumption (17), permissible values of Δv_2 were also computed for the first two braking phases. From these the errors due to the above listed tolerances were subtracted. From the difference, permissible values of Δg_r and $\Delta(\Delta v)$ were calculated for two different cases:

Case α: The braking impulse is controlled by thrust duration (timer error can be neglected as error source) and thrust level. The equation (5a) gives the accuracy with which the thrust level must be maintained. The resulting magnitude $\Delta g_r / g_r$ shows what effort in thrust control would be necessary (e.g. control by simply maintaining constant chamber pressure, or by a closed loop system using accelerometer as sensor).

Case β: The magnitude $\Delta v = \int g_r \, dt$ is controlled. For this case equation (9b) gives the accuracy to which Δv must be held. The permissible $\Delta(\Delta v)/\Delta v$ value reveals whether simple methods such as control of propellant quantity or accurate methods such as cut-off command by an integrating accelerometer must be employed. Equation (5b) gives the accuracy to which g_r must simultaneously be held. The permissible $\frac{\Delta g_r}{g_r}$ will be larger than it is in case α.

Thus the tolerances were determined to be:

Case α: $\frac{\Delta g_r}{g_r} \leqq 2\%$ for first and second braking phase

$\leqq 5\%$ for third braking phase

Case β:

$$\left.\begin{array}{l} \frac{\Delta(\Delta v)}{\Delta v} \leqq 0.1\% \\[2ex] \frac{\Delta g_r}{g_r} \leqq 5\% \end{array}\right\} \text{ for first and second braking phase}$$

$$\left.\begin{array}{l} \frac{\Delta(\Delta v)}{\Delta v} \leqq 2\% \\[2ex] \frac{\Delta g_r}{g_r} \leqq 10\% \end{array}\right\} \text{ for third braking phase}$$

The above listed tolerances determine nature and performance of the guidance instrumentation needed in addition to the altimeter. Although an integrating accelerometer becomes necessary for the control of Δv, case β seems preferable because it permits larger tolerances in g_r.

The design of suitable altimeters, accelerometers and programming devices lies well within the state of the art and can be based on experience with existing equipment. Consequently the main effort of further development can be concentrated to achieve the utmost reliability under the environmental conditions of the lunar mission.

The tolerances quoted above are based on a total retardation of approximately 200 seconds as compared with a continuous "one-stop" braking maneuver. Longer retardation (by insertion of more free fall time) would result in even larger permissible tolerances. Thus a wide range of compromises is possible between retardation time and guidance accuracy.

However, prolonged approach retardation has the disadvantage that more propellants are needed for the generation of the necessary braking thrust, because the retarded vehicle is exposed to the gravitation field of the moon for a longer time. The excess impulse is roughly proportional to the retardation time. For the landing maneuver described in Section II.6 the excess consumption of propellants is approximately 200 lbs It has to be emphasized, however, that this figure is derived from a consideration of strictly vertical descent and therefore applies only if no lateral correction has to be made during the landing maneuver.

The different conditions for the more realistic case where lateral corrections have to be included are treated in the next part of this appendix.

D. 5. Lateral Corrections

During the terminal guidance phase the need for changes in approach direction can arise for two reasons:

1. The approach velocity may contain a lateral component. Even in case nominal impact is perpendicular an appreciable horizontal component will be caused by injection and midcourse errors. This velocity component has to be eliminated before touch-down.
2. A capability for lateral displacements is required in order to land at a visually selected spot.

   Even if a pre-selected landing site should be chosen, lateral corrections would still be necessary to reduce errors with respect to the pre-selected site. In this case, the lateral correction capability should be at least equal to the expected errors due to injection and midcourse tolerances.

   If the decision concerning the exact landing spot is made immediately before starting the landing maneuver as a result of TV observation, the lateral correction capability may be smaller than the possible deviations due to injection and midcourse errors. The terminal guidance scheme described in Section II.6 is based on this second concept.

The simplest and most economical way to change the approach direction is by tilting the vehicle during the braking operation rather than using a separate nozzle system for lateral acceleration. In case lateral displacements are to be made, the correction procedure will consist of a lateral acceleration to produce a predetermined lateral velocity, and a subsequent deceleration to nearly zero horizontal velocity. See Figure D.1.

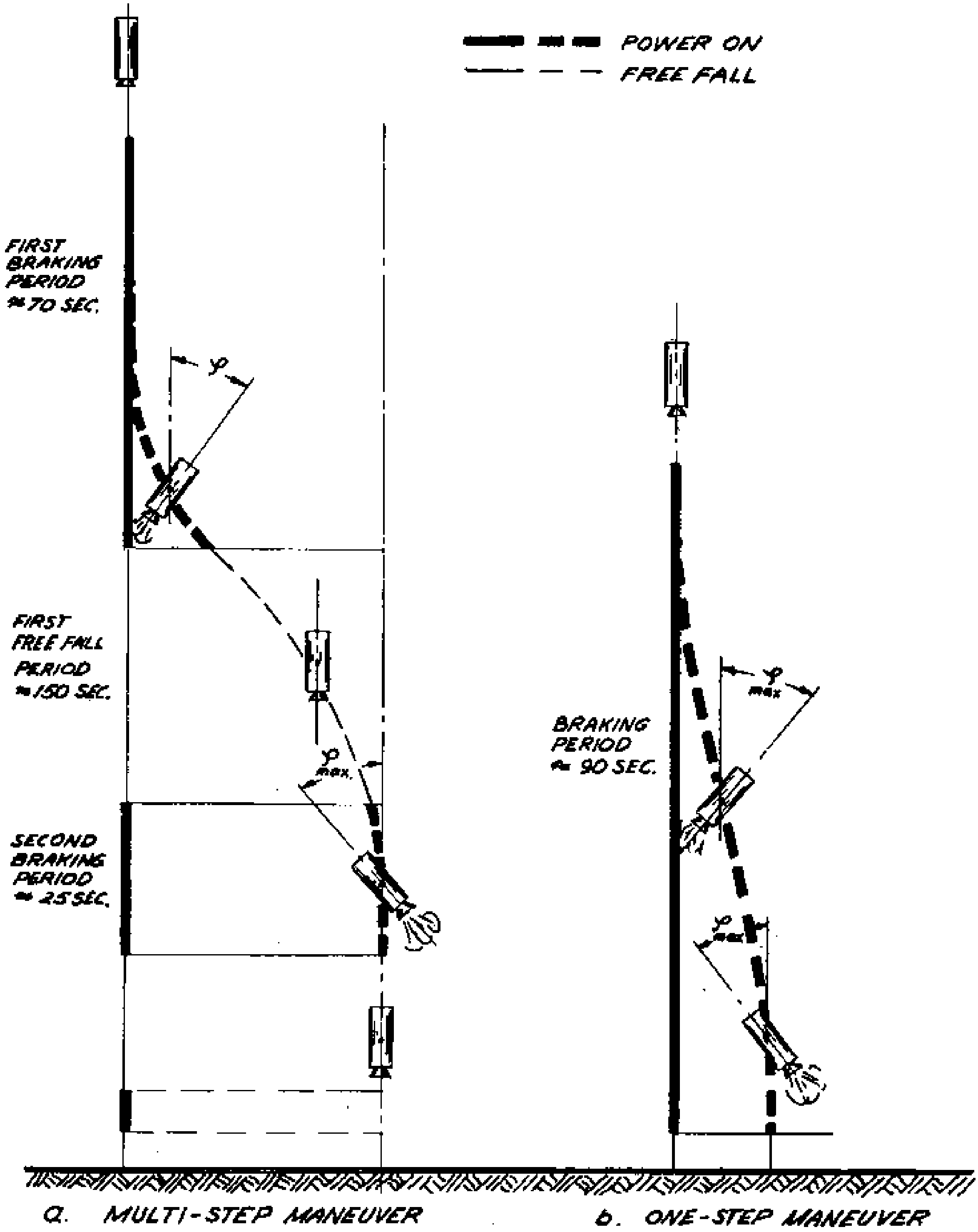


FIG. D.1 LATERAL MANEUVERS

The additional impulse needed for lateral displacement is proportional to $(1-1/\cos\varphi)$, φ being the tilt angle measured against the local vertical (see Fig. D.1). It is easily seen that for unretarded approach, large angles φ are necessary to achieve a given lateral displacement, resulting in high propellant consumption.

The second free fall period of the retarded approach however, is very advantageous with respect to fuel economy because it permits the build-up of lateral displacement without propellant consumption. Figure D.2 shows the total weight penalty to achieve a given lateral displacement. The weight penalty is defined to be zero for a "one-stop" maneuver with a 20K engine, if no lateral correction is made. For comparison the propellant requirements for "one-stop" with a 20K engine is also shown. The graph indicates that the retardation maneuver renders best economy for lateral displacement above 20 kilometers. The upper limit is given by the maximum angle φ, i.e. by the angular freedom of the platform. In this range the fuel economy is very close to the theoretical optimum.

D. 6. Summary

The most significant property of the described terminal guidance scheme is that by insertion of free fall periods according to a predetermined program automatic compensation of guidance errors is achieved. This error correction is very effective for errors in the initial conditions (e.g. approach velocity at the beginning of the terminal phase) and also with respect to tolerances in the execution of the programmed landing maneuvers. This self-compensating effect reduces greatly the necessary number and accuracy of guidance functions which in turn increases the simplicity and reliability of the system.

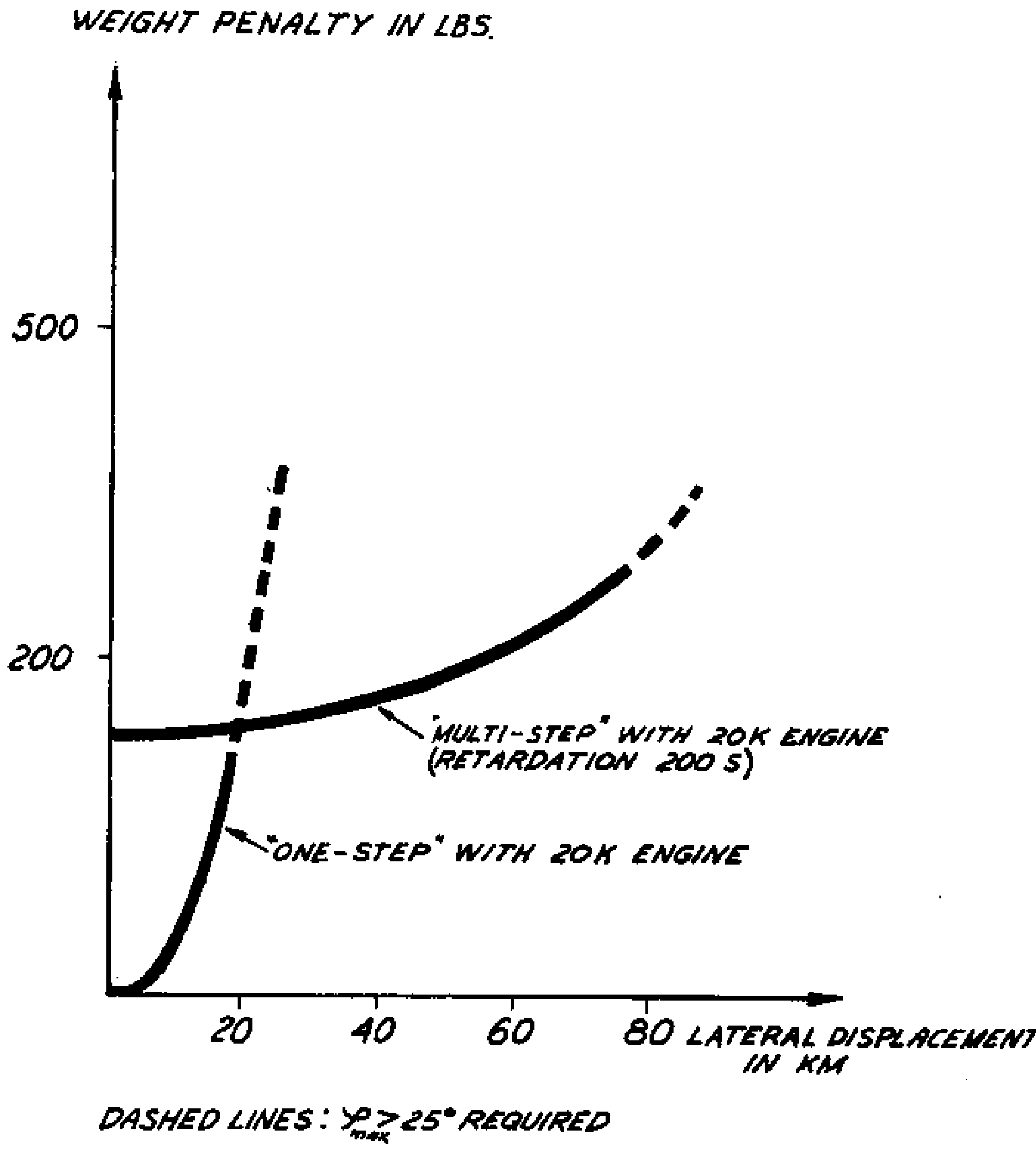


FIG. D.2 PROPELLANT CONSUMPTION FOR LATERAL CORRECTION

Except for the determination of the horizontal approach velocity, the only external measurements necessary are altimeter measurements to trigger the predetermined steps of the operational program. No measurement of the vertical approach velocity is required. A relatively simple autopilot is sufficient for program execution rather than a decision making on-board computer. The braking engine operates with full thrust at all times and need not be throttable. The permissible tolerances in thrust level (5 to 10%) require no unusually precise thrust regulation.

The limitations of the above described "minimum effort" scheme lie in the assumption of

(1) A nominal trajectory, which terminates in perpendicular or nearly perpendicular approach to the lunar surface. This trajectory has also been chosen for other reasons.

(2) A short free fall period to touch-down, consistent with the requirement for separation and removal of the braking engine.

Both limitations, however, are not inherent in the principles involved and can be eliminated while the advantages of the scheme are maintained. With a few modifications and some additional guidance equipment the present scheme can be extended to non-perpendicular approach.

Another modification of the described scheme which is envisioned for later lunar missions provides a small throttable engine of one to three thousand pounds thrust. This second engine would operate after cut-off or separation of the main braking engine, thus permitting very soft landings by closed loop control of the final portion of the descent until touch down. It is expected that suitable guidance sensors (altimeter and velocity meter for very low altitudes) can be developed within the next few years.

Studies are continuing to determine optimum conditions for these modifications.

APPENDIX E

HORIZON SEEKER PERFORMANCE IN MEASURING ALTITUDE (C)

The opening angle, ε , of the horizon seeker, HS , is a measure of height above the lunar surface, h , if the radius of the moon, R_L , is known. See Figure II.18.

$$\sin \frac{\varepsilon}{2} = \frac{R_L}{R_L + h} \tag{1}$$

$$h = R_L \left(\frac{1}{\sin \frac{\varepsilon}{2}} - 1 \right) \tag{2}$$

Equation (2) is plotted in Figure II.21 for $\frac{h}{R_L}$ vs. ε . The accuracy with which ε can be measured is limited. The effect of a variation $d\varepsilon$ on the altitude is:

$$dh = - \frac{R_L \cos \frac{\varepsilon}{2}}{2 \sin^2 \frac{\varepsilon}{2}} \, d\varepsilon \tag{3}$$

Eliminating ε by introducing (1) such that

$$dh = f\,(h)\, d\varepsilon \tag{4}$$

results in

$$dh = -h \, \frac{R_L + h}{2 R_L} \sqrt{\frac{2 R_L}{h} + 1} \; d\varepsilon \tag{5}$$

Expressing Equation (5) in relative altitude

$$a = \frac{h}{R_L} \tag{6}$$

$$dh = -h\ (1+a)\left(\frac{2}{a} + 1\right)^{\frac{1}{2}} \frac{d\varepsilon}{2} \tag{7}$$

The relative error in altitude $\frac{\Delta h}{h}$ as a function of altitude with a constant error $\Delta\varepsilon$ in ε is then

$$\delta_h = \frac{\Delta h}{h} = -\ (1+a)\left(\frac{2}{a} + 1\right)^{\frac{1}{2}} \frac{\Delta\varepsilon}{2} \tag{8}$$

This function is plotted in Figure II.22 for $\Delta\varepsilon = 0.1°$.

Considering a useful angle

$$10° < \varepsilon < 120°$$

with an absolute accuracy of

$$\Delta\varepsilon = 0.1°$$

the altitude can be determined to better than 0.1% neglecting errors in the moon's radius, R_L. The influence of R_L on altitude is described by

$$dh = \left(\frac{1}{\sin\frac{\varepsilon}{2}} - 1\right) d\ R_L \tag{9}$$

Eliminating ε by introducing Equation (1) results in

$$dh = f(h)\, dR_L \tag{10}$$

and

$$dh = \frac{h}{R_L}\, dR_L \tag{11}$$

The relative error in altitude being

$$\delta_h = \frac{\Delta h}{h} = \frac{\Delta R_L}{R_L} \tag{12}$$

For

$$\Delta R_L = 8 \text{ km}$$

$$\delta_h = \frac{8}{1735} = 0.0046 = 0.46\%$$

$\Delta R = 8$ km is supposed to cover (a) the uncertainty in the radius of the moon, (b) the error introduced by the unevenness of the lunar surface, and (c) the uncertainty of the difference in surface terrain between the subvehicle point and the point where the line of sight of the optical sensor is tangent to the moon.

Appendix F

HORIZON SEEKER

PERFORMANCE IN MEASURING VERTICAL VELOCITY (C)

The information on height available through the horizon seeker can be differentiated numerically by a digital computer.

The average vertical velocity would be

$$\dot{y}_{avg} = \frac{h_1 - h_2}{T}$$

Both values of altitude, h_1 and h_2 are subject to errors Δh_1 and Δh_2. The time interval is likewise in error by ΔT:

$$\dot{y}_{avg} + \Delta\dot{y}_{avg} = \frac{(h_1 \pm \Delta h_1) - (h_2 \pm \Delta h_2)}{T \pm \Delta T}$$

$$\dot{y}_{avg} + \Delta\dot{y}_{avg} = \frac{(h_1 - h_2) \pm (\Delta h_1 - \Delta h_2)}{T} \left[1 \pm \frac{\Delta T}{T}\right]^{-1}$$

$$\dot{y}_{avg} + \Delta\dot{y}_{avg} = \frac{h_1 - h_2}{T} + \frac{h_1 - h_2}{T}\left[\mp\frac{\Delta T}{T} + \left(\frac{\Delta T}{T}\right)^2 \mp \left(\frac{\Delta T}{T}\right)^3 + \ldots \mp \ldots\right] \mp$$

$$\pm \frac{\Delta h_1 - \Delta h_2}{T} \pm \frac{\Delta h_1 - \Delta h_2}{T}\left[\mp\frac{\Delta T}{T} + \left(\frac{\Delta T}{T}\right)^2 \mp \left(\frac{\Delta T}{T}\right)^3 + \ldots \mp \ldots\right]$$

The error in average vertical velocity $\dot{y}_{avg}$ due to errors in altitude and time interval is thus

$$\Delta\dot{y}_{avg} = \frac{h_1 - h_2}{T}\left[\mp\frac{\Delta T}{T} + \left(\frac{\Delta T}{T}\right)^2 \mp \left(\frac{\Delta T}{T}\right)^3 + \ldots \mp \ldots\right] \pm \frac{\Delta h_1 - \Delta h_2}{T} \pm$$

$$\pm \frac{\Delta h_1 - \Delta h_2}{T}\left[\mp\frac{\Delta T}{T} + \left(\frac{\Delta T}{T}\right)^2 \mp \left(\frac{\Delta T}{T}\right)^3 + \ldots \mp \ldots\right]$$

and the relative error would be

$$\partial \dot{y}_{avg} = \frac{\Delta \dot{y}_{avg}}{\dot{y}_{avg}} = \left[\mp \frac{\Delta T}{T} + \left(\frac{\Delta T}{T}\right)^2 \mp \left(\frac{\Delta T}{T}\right)^3 + \ldots \mp \ldots\right] \pm \frac{\Delta h_1 - \Delta h_2}{h_1 - h_2} \pm$$

$$\pm \frac{\Delta h_1 - \Delta h_2}{h_1 - h_2}\left[\mp \frac{\Delta T}{T} + \left(\frac{\Delta T}{T}\right)^2 \mp \left(\frac{\Delta T}{T}\right)^3 + \ldots \mp \ldots\right]$$

This rather lengthy expression may be reduced by neglecting all products and powers of errors. The third term disappears completely and the first shrinks into one:

$$\partial \dot{y}_{avg} \approx \mp \frac{\Delta T}{T} \pm \frac{\Delta h_1 - \Delta h_2}{h_1 - h_2} \qquad (1)$$

Since the relative error in the time interval, $\frac{\Delta T}{T}$ may be assumed to be smaller than 10^{-3} the main source of error in the average vertical velocity is due to errors in altitude indicated by the second term in equation (1).

Regardless of these errors the average vertical velocity, $\dot{y}_{avg}$, is only an approximation of the true vertical velocity $\dot{y}$:

$$\dot{y} = \dot{y}_{avg} + \sigma$$

This is shown in Figure II. 25 by plotting the true vertical velocity, $\dot{y}$, of an actual trajectory against time. Time is measured from the instant of impact being equal to zero seconds. Superimposed on this curve is the average vertical velocity computed from the altitudes of the same trajectory. The altitude, h, itself is plotted in Figure F. 1. The trajectory used here is one with perpendicular ballistic impact of 2722 m/s on the moon. The numerical differentiation is performed with a fixed time interval of 15 seconds.

The diagram shows a maximum error of the measurement of the vertical velocity 1.6%, which is very reasonable. This leaves the

[redacted]

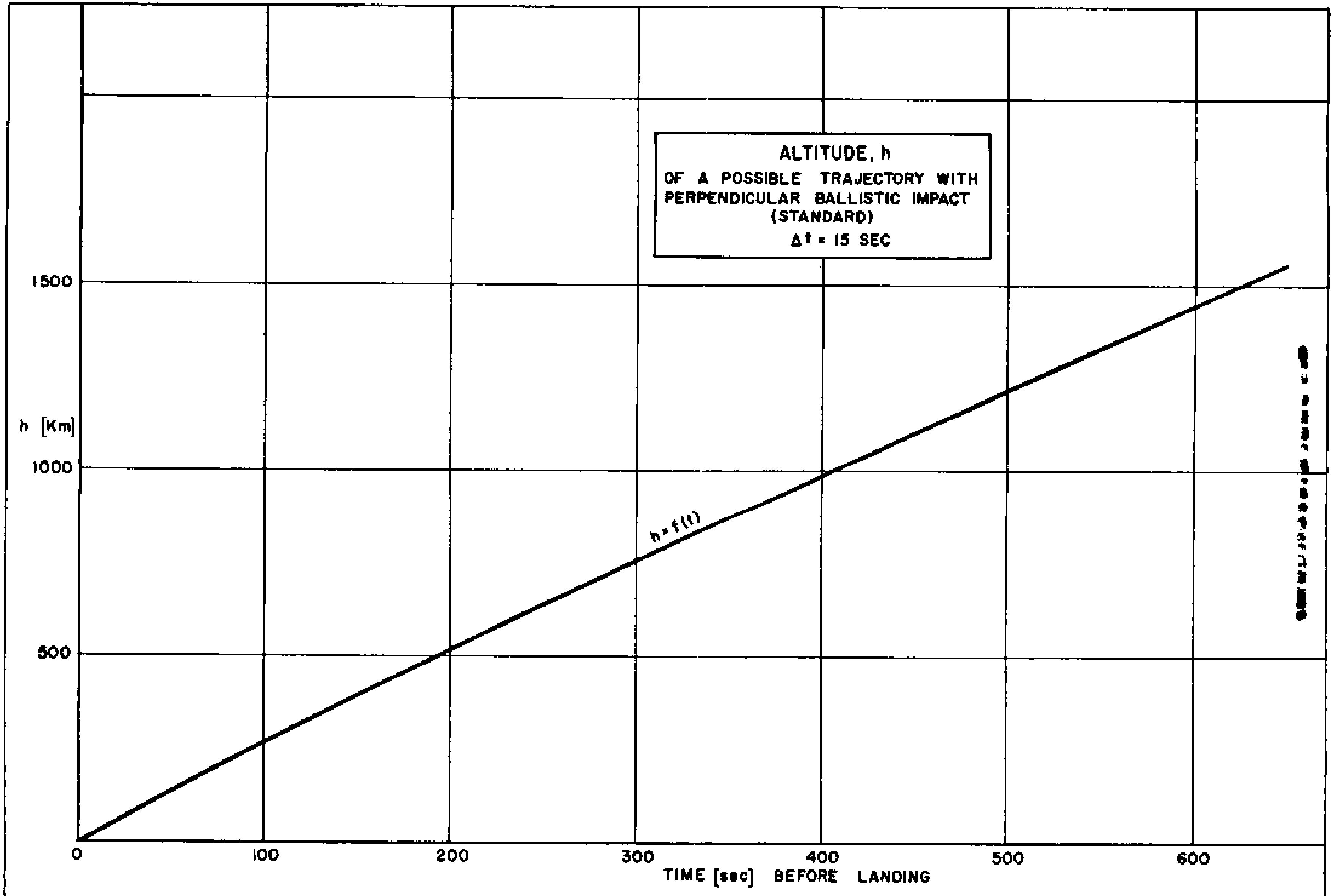


FIG. F.1

[redacted]

error $\partial \dot{y}_{avg}$ to be investigated further. It is the second term of equation (1) that gives reason for worry since it contains one of the most unfavorable expressions that can come up in error analysis, namely a difference.

Abbreviating equation (1)

$$\partial \dot{y}_{avg} = \mp \frac{\Delta T}{T} \pm \eta$$

for

$$\eta = \frac{\Delta h_1 - \Delta h_2}{h_1 - h_2}$$

one can rearrange η by using $\partial h = \frac{\Delta h}{h}$ then

$$\eta = \frac{h_1(\partial h_1) - h_2(\partial h_2)}{h_1 - h_2}$$

or

$$\eta = \frac{(\partial h_1)h_1 - \xi(\partial h_1)h_2}{h_1 - h_2}$$

Respectively

$$\eta = \partial h_1 \frac{h_1 - \xi h_2}{h_1 - h_2}$$

where

$$\xi = \frac{\partial h_2}{\partial h_1}$$

the ratio between the relative errors in the two altitudes.

Since ξ may be positive or negative the partial error η in the average vertical velocity may, under unfavorable circumstances, become much larger than the error in altitude.

η will become smaller with increasing $(h_1 - h_2)$, i.e., for large differentiating intervals. The error will be acceptable at great altitudes, but not in the lower range. Assuming the altitude range between 1500 km and 300 km, we find for intervals as long as 200 km around the 1500 km mark, and for about 100 km interval at the 300 km mark the following errors (assuming the most unfavorable $\xi = -1$):

$$\eta = \partial h_1 \frac{385 + 271}{385 - 271} = \partial h_1 \ 5.75$$

and

$$\eta = \partial h_1 \frac{1537 + 1339}{1537 - 1339} = \partial h_1 \ 14.5$$

In other words the largest errors in average vertical velocity over the range from 1500 km to 250 km altitude would be from about 6 times the relative error in altitude to about 14 times the relative error in altitude. At a maximum error in altitude of 0.5% this would be from 3% to 7%.

These maximum errors are not very probable though possible.

Appendix G

EQUATIONS OF MOTION
DURING DESCENT TOWARD LUNAR SURFACE (C)

During the powered periods of the descent, the following are the equations describing the motion of the vehicle. See Figure G.1.

Assumption: The curvature of the lunar surface is negligible; thus vertical and horizontal motion may be treated independently.

Vertical motion:

$$F \cos (\varphi + \beta) - m(t) g_L(y) - m(t)\ddot{y} = 0$$

With the simplifying assumption of $(\varphi + \beta)$ being small and the lunar gravitational field $g_L(y)$ being constant over the altitudes, y, the above equation reduces to

$$\frac{F}{m(t)} - g_L = \ddot{y}$$

Torque equation:

$$\ddot{\varphi} + c_2 \beta = 0$$

Mass as function of time:

$$m(t) = m_0 - k_1 \int F dt$$

Thrust ON-OFF (expressed symbolically)

$$(\Delta \dot{y} = \Delta \dot{y}_0) \supset (\sim F)$$

Horizontal motion:

$$F \sin (\varphi + \beta) - m(t)\ddot{x} = 0$$

with the simplifying assumption of $(\varphi + \beta)$ being small, the above equation reduces to

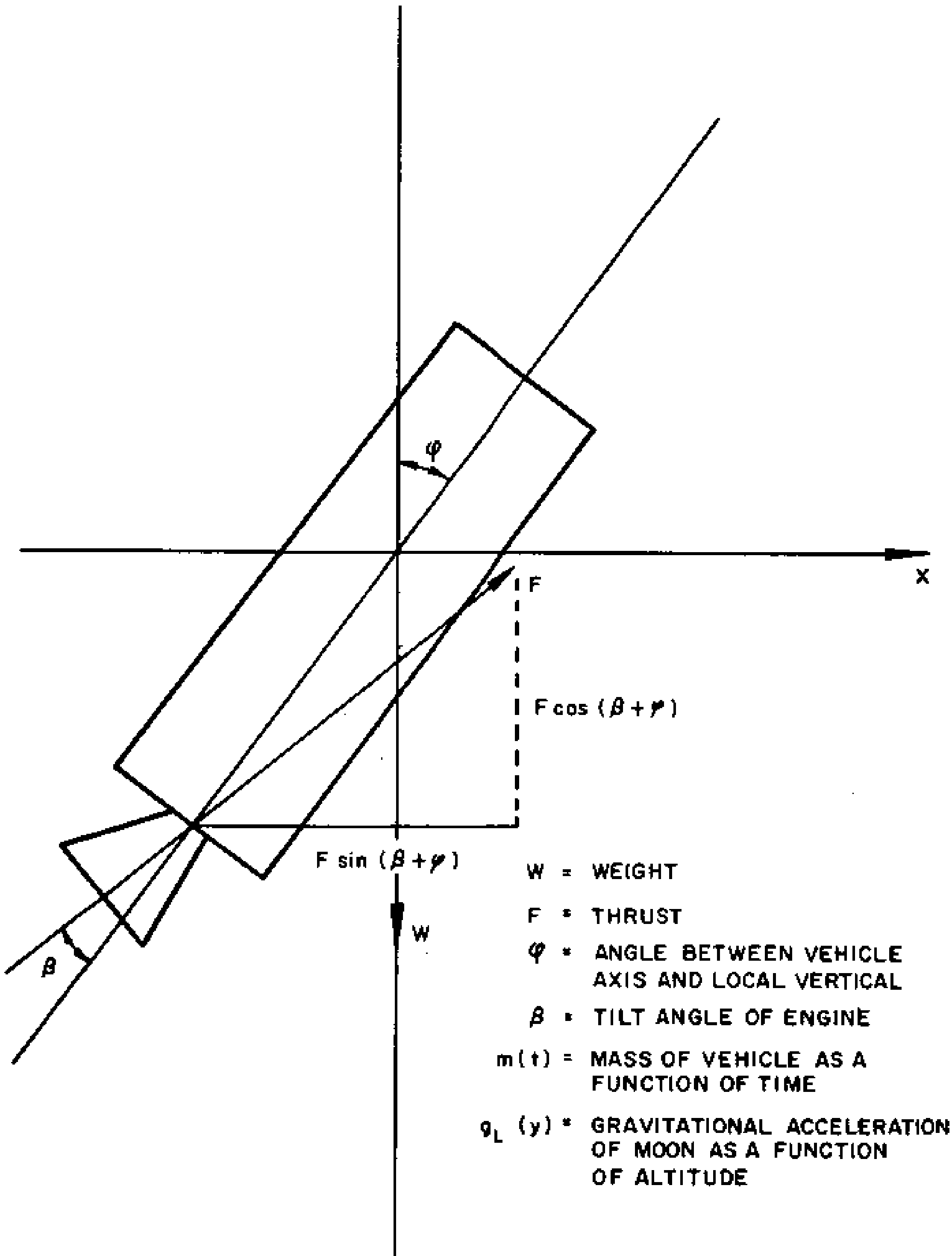


FIG. G.1

GEOMETRIC RELATIONS BETWEEN VEHICLE AND LUNAR COORDINATES

$$\frac{F}{m(t)}(\varphi + \beta) = \ddot{x}$$

Control function:

$$\beta = a_0\varphi + a_1\dot{\varphi} + e_1(\Delta\dot{x} - \Delta\dot{x}_0)$$

This holds for the first and second decelerating periods.

Δx_0 will demand the engine to swing out to an angle β. (β will be limited to between 4° and 8°.) The vehicle will thus be tilted. After about one second β goes back to zero, i.e., the engine lines up again with the longitudinal axis but the vehicle remains in a tilted position described by the angle φ between LA and LV. With $\Delta\dot{x}$ building up the maneuver will be reversed at the end of the powered period when $\Delta\dot{x}$ gets to be equal to $\Delta\dot{x}_0$. The engine then swings out again, this time in the opposite direction, in order to eliminate φ. Thus the vehicle will be lined up with the local vertical when the engine cuts off. Measurements for the new decelerating period may begin immediately.

$$\beta = a_0\varphi + a_1\dot{\varphi} + e_0(x - x_0) + e_1(\Delta\dot{x})$$

This holds for the third deceleration period.

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