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A LUNAR EXPLORATION PROGRAM
BASED UPON SATURN-BOOSTED SYSTEMS

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This study has been extended for 6 months. In continuing this work, particular emphasis is being placed on the revisions required by modifications of the basic SATURN vehicle.

Chapters III, IV, and VI have been extracted and reproduced in this special publication for distribution to individuals concerned with the NASA Lunar Exploration Program.

Prepared by

Development Operations Division
Army Ballistic Missile Agency

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 & 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 & Control Laboratory, the Research Projects Laboratory, the Structures & 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

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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.

studying a great many reports on measurements useful in the exploration of the lunar surface, numerous contacts were made with other organizations and individuals, and problems of lunar observations were discussed at length. 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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CHAPTER III*

PAYLOAD PACKAGES FOR LUNAR LANDINGS AND CIRCUMLUNAR FLIGHTS

*(This chapter was taken from ABMA Report No. DV-TR-2-60, dtd 1 February 1960)

DEVELOPMENT OPERATIONS DIVISION
ARMY BALLISTIC MISSILE AGENCY

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.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

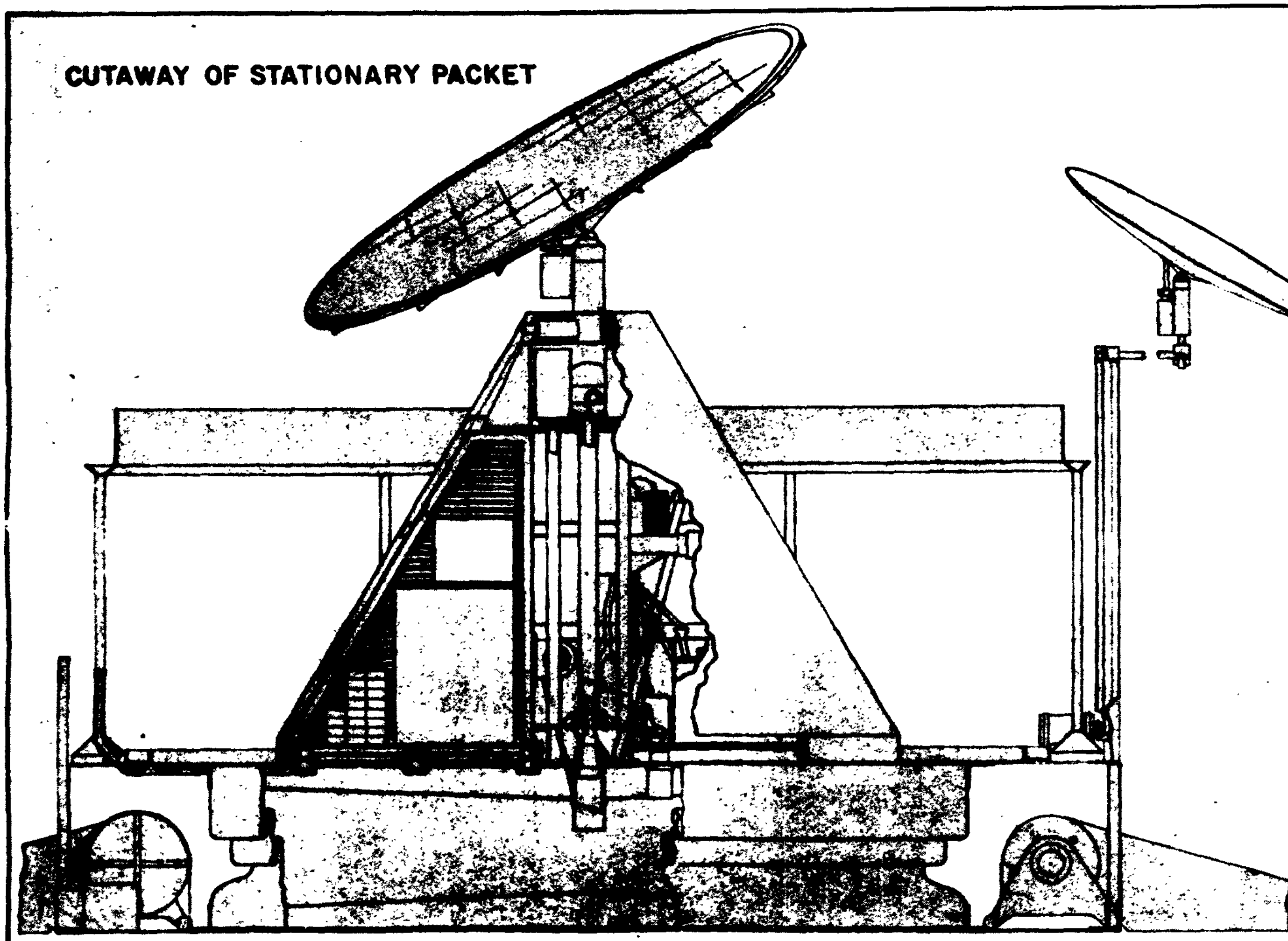


FIG III.3

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 $+130^{\circ}$ 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 [illegible]0 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.

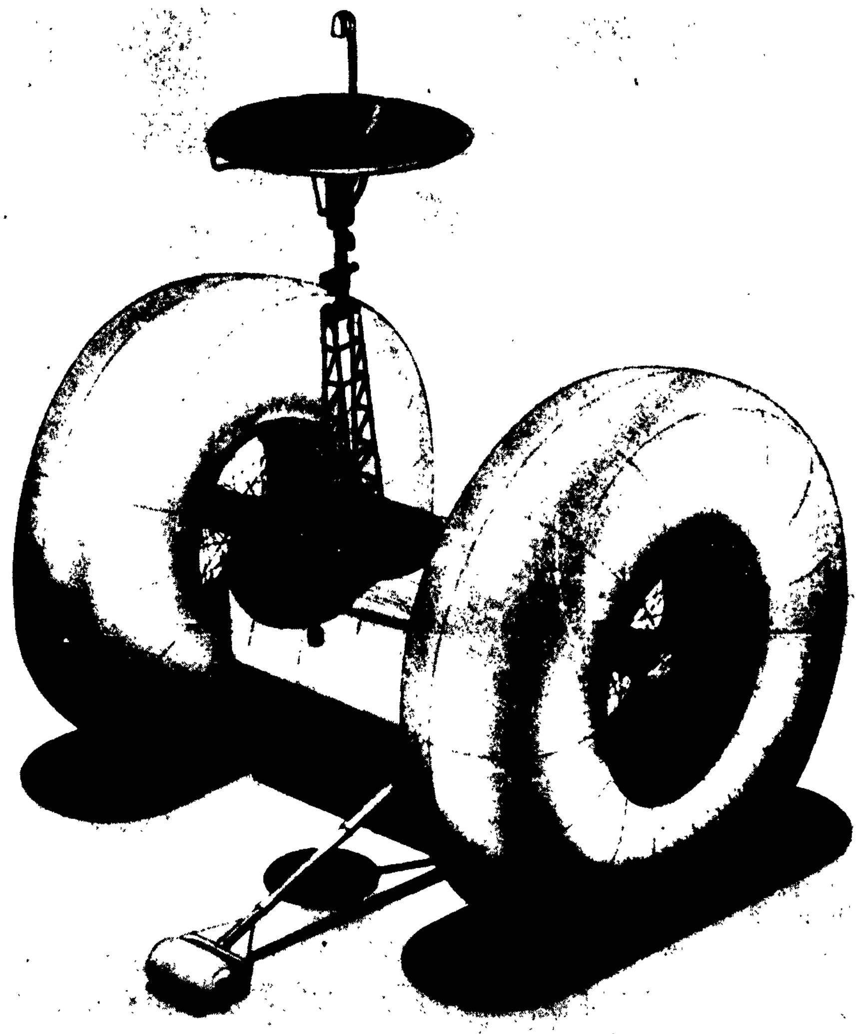
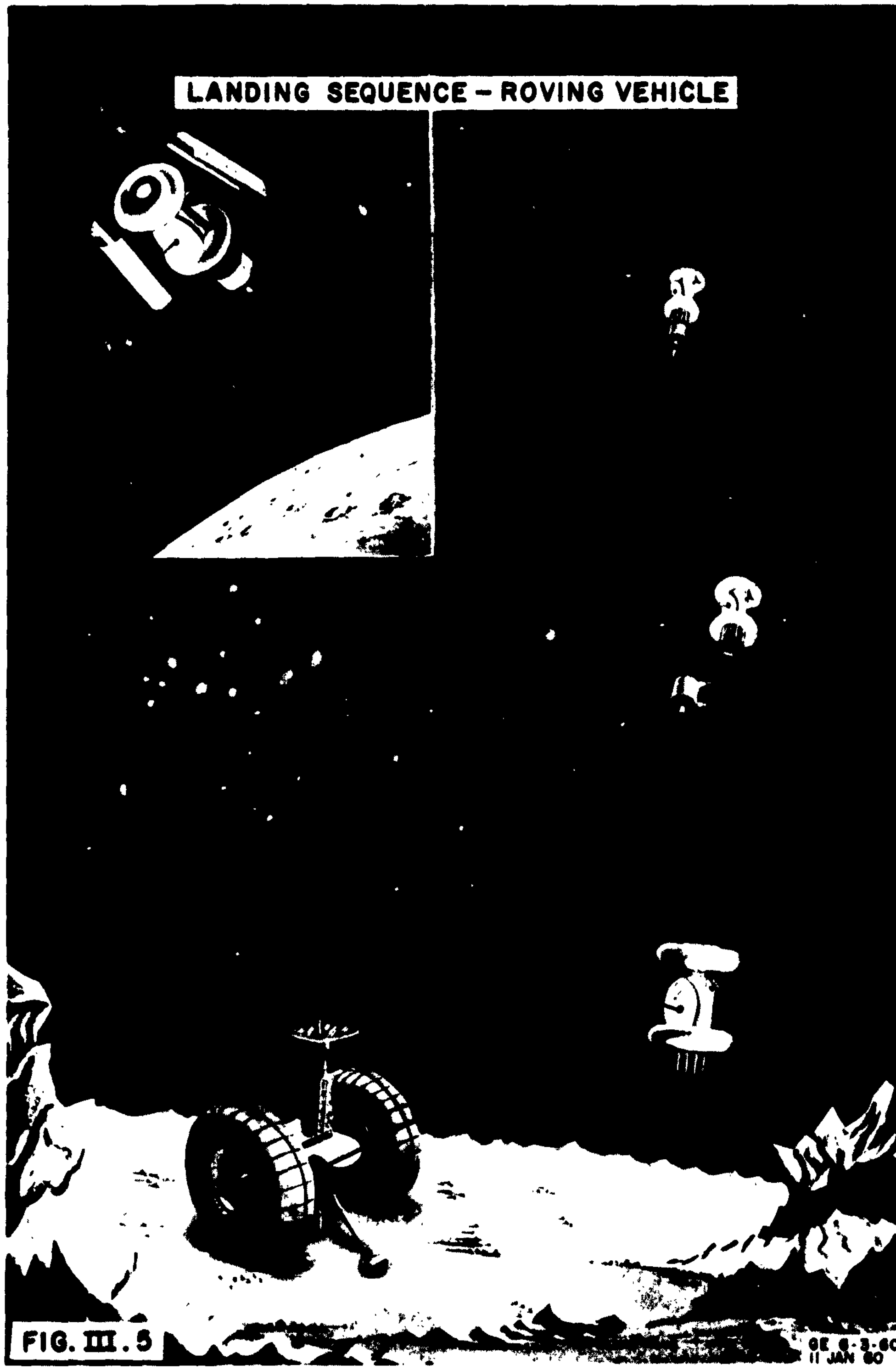


FIG III.4 GE6-5-60 11 JAN 60

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.



LANDING SEQUENCE – ROVING VEHICLE

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 mobilisation 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.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 temperatu› 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 x 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 semi-conductors. 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.

U. S. ARMY ORDNANCE MISSILE COMMAND
REDSTONE ARSENAL, ALABAMA

CHAPTER IV

LUNAR EXPLORATION*

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DEVELOPMENT OPERATIONS DIVISION
ARMY BALLISTIC MISSILE AGENCY

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. Commtzis

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

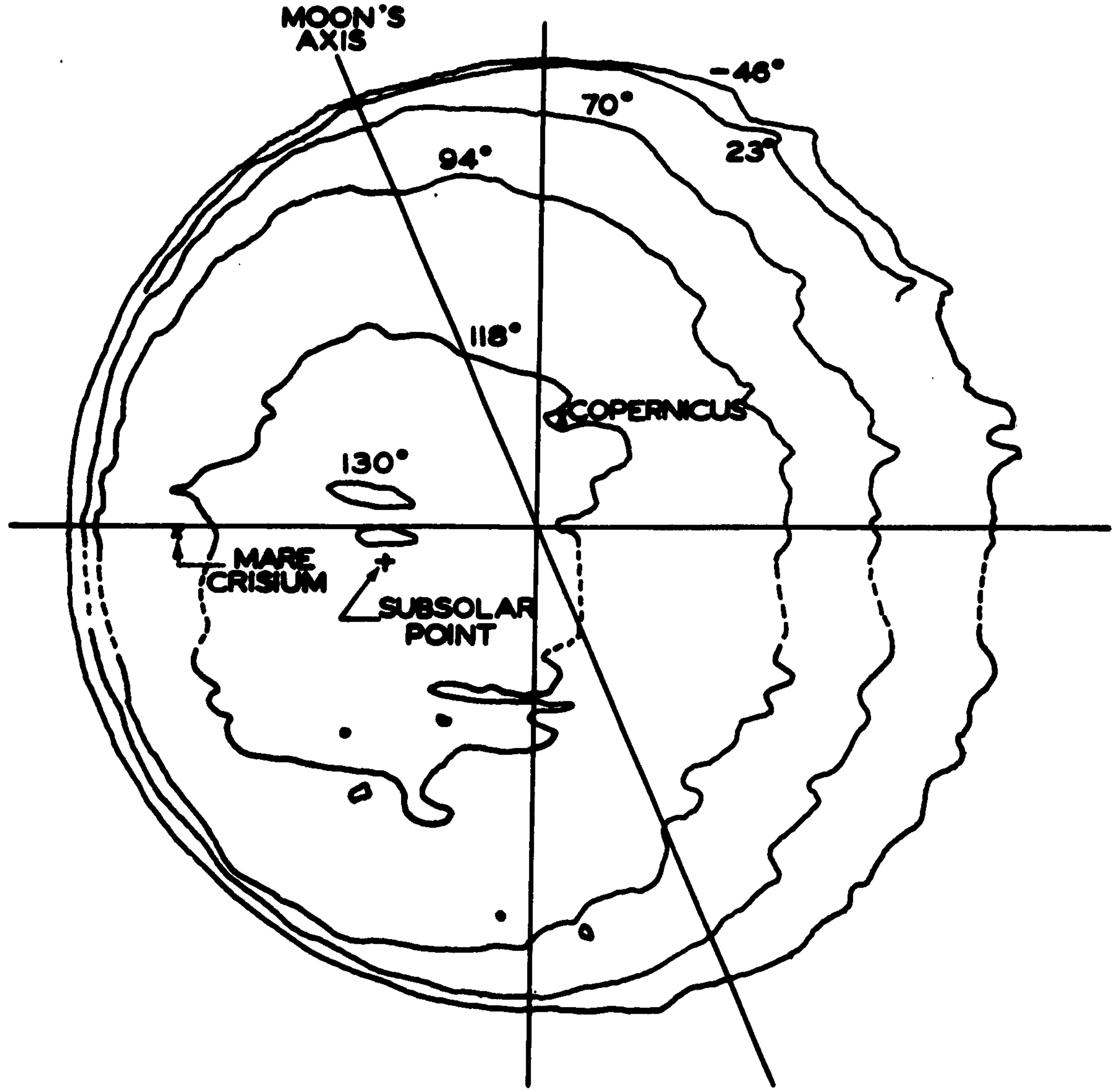
The proper functioning of equipment landed on the moon will be insured by properly taking into account the influence of lunar environment on equipment operation. Aspects of the lunar environment which are particularly important are lunar surface temperatures, the nature of the moon's surface, its topography, 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.

IV.4.2 Landing Sites for Roving Vehicle. 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,

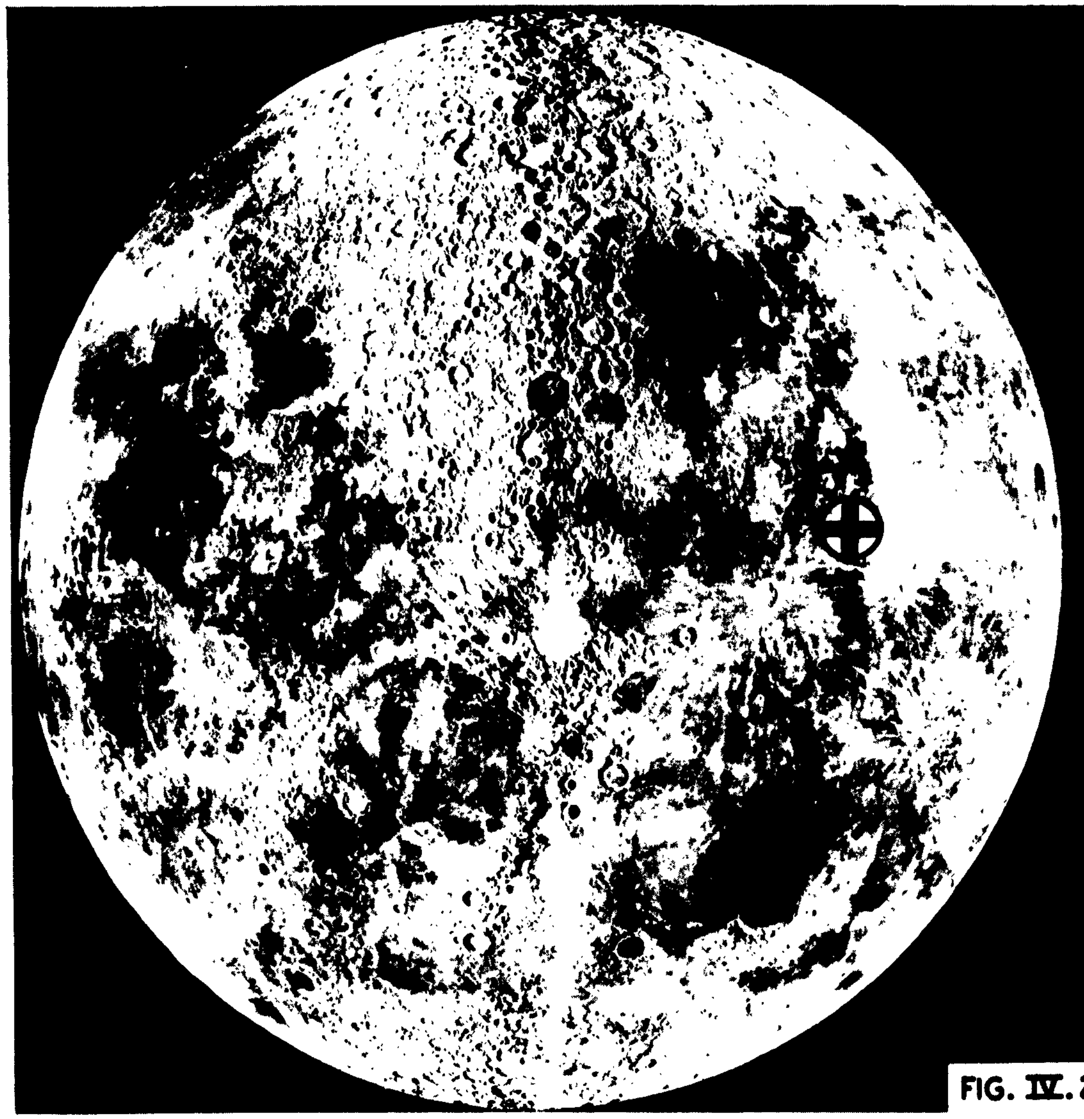

FIG. IV. 2

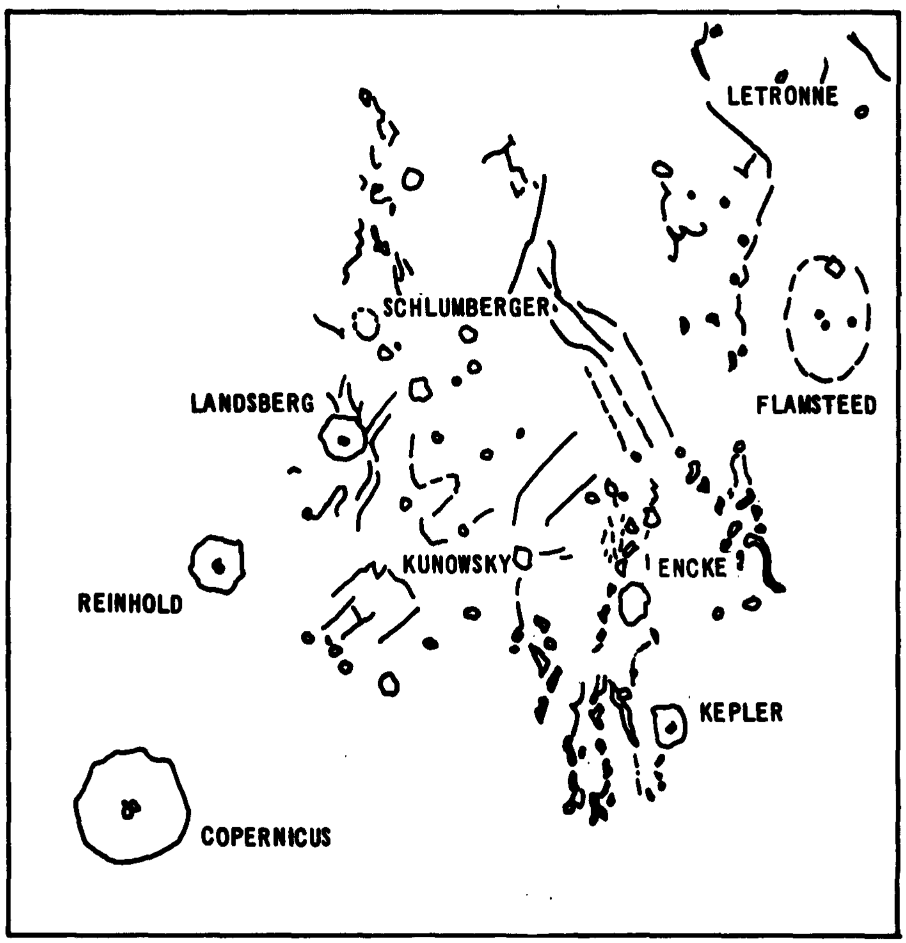

FIG. IV.3

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

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[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.

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

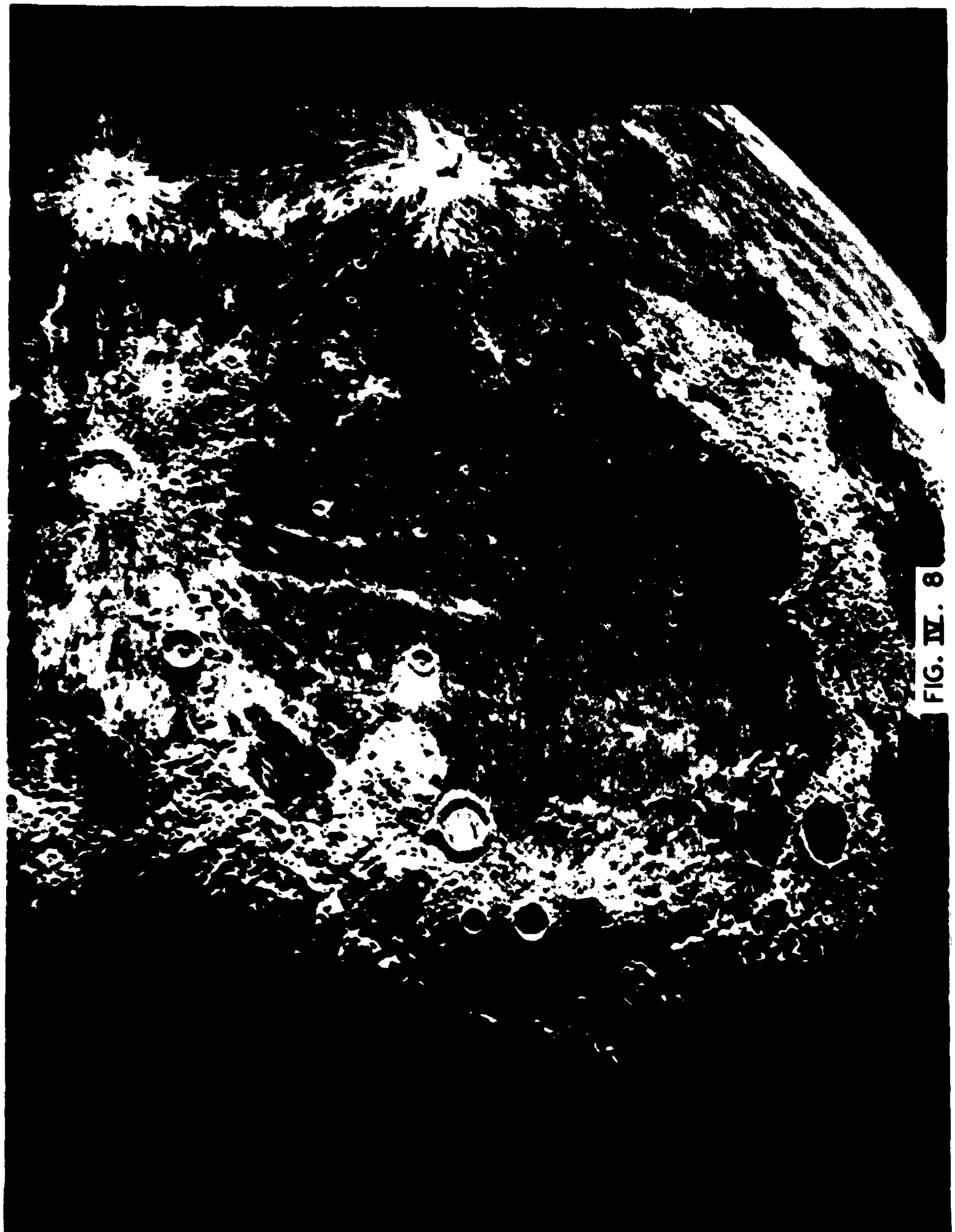

FIG. IV. 8

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

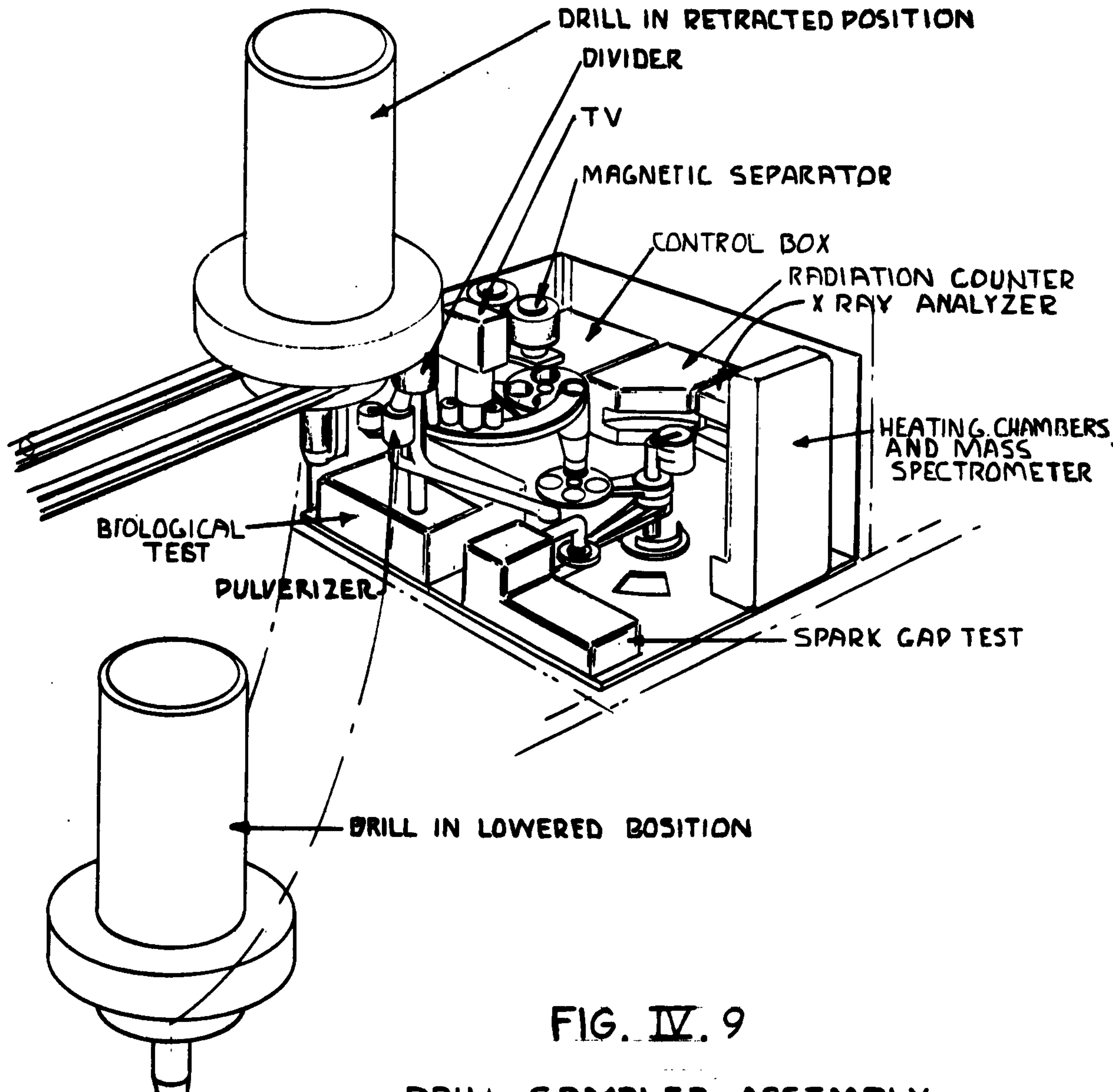


FIG. IV. 9

DRILL SAMPLER ASSEMBLY

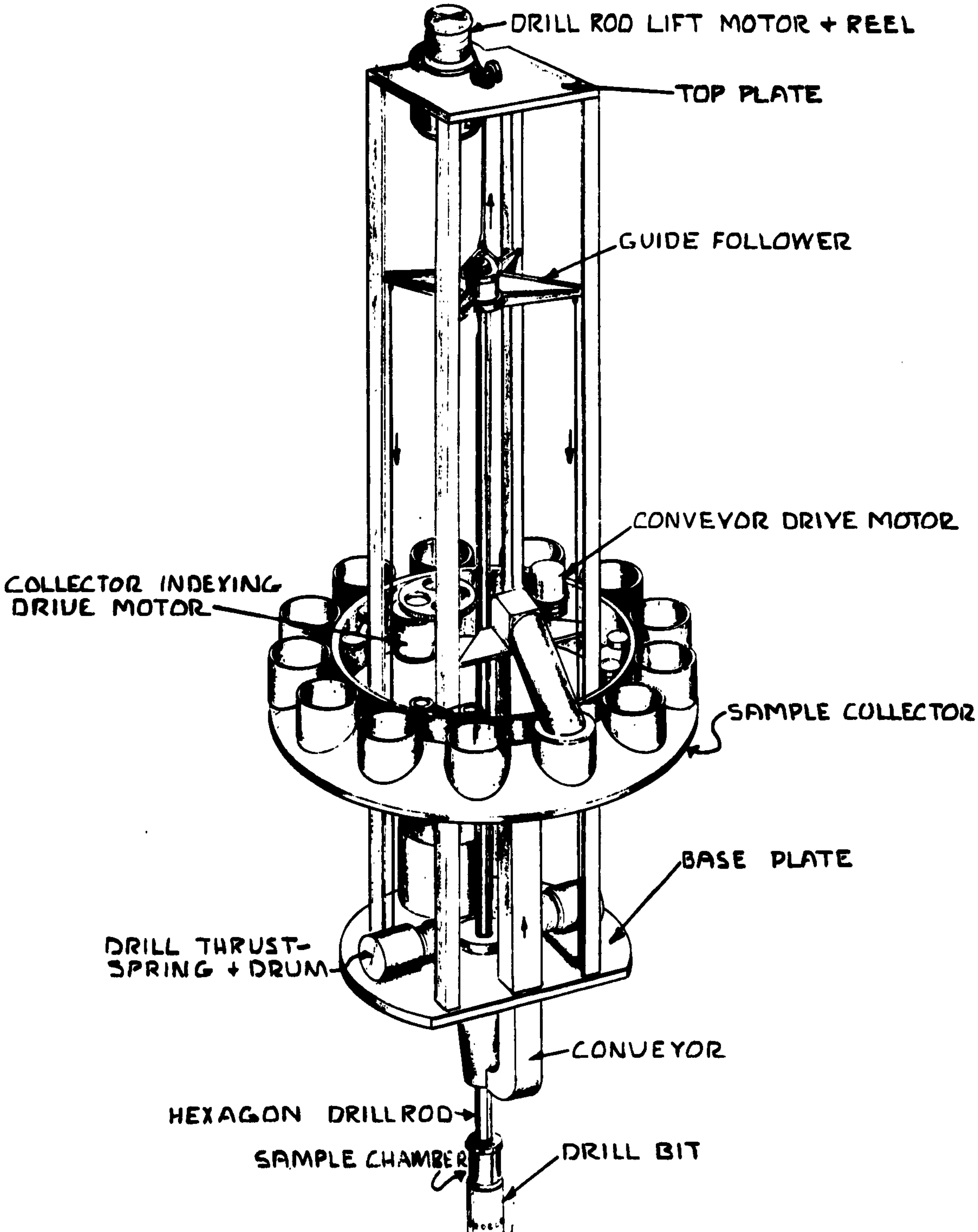


FIG.IV.10 DRILL UNIT COMPONENT ARRANGMENT

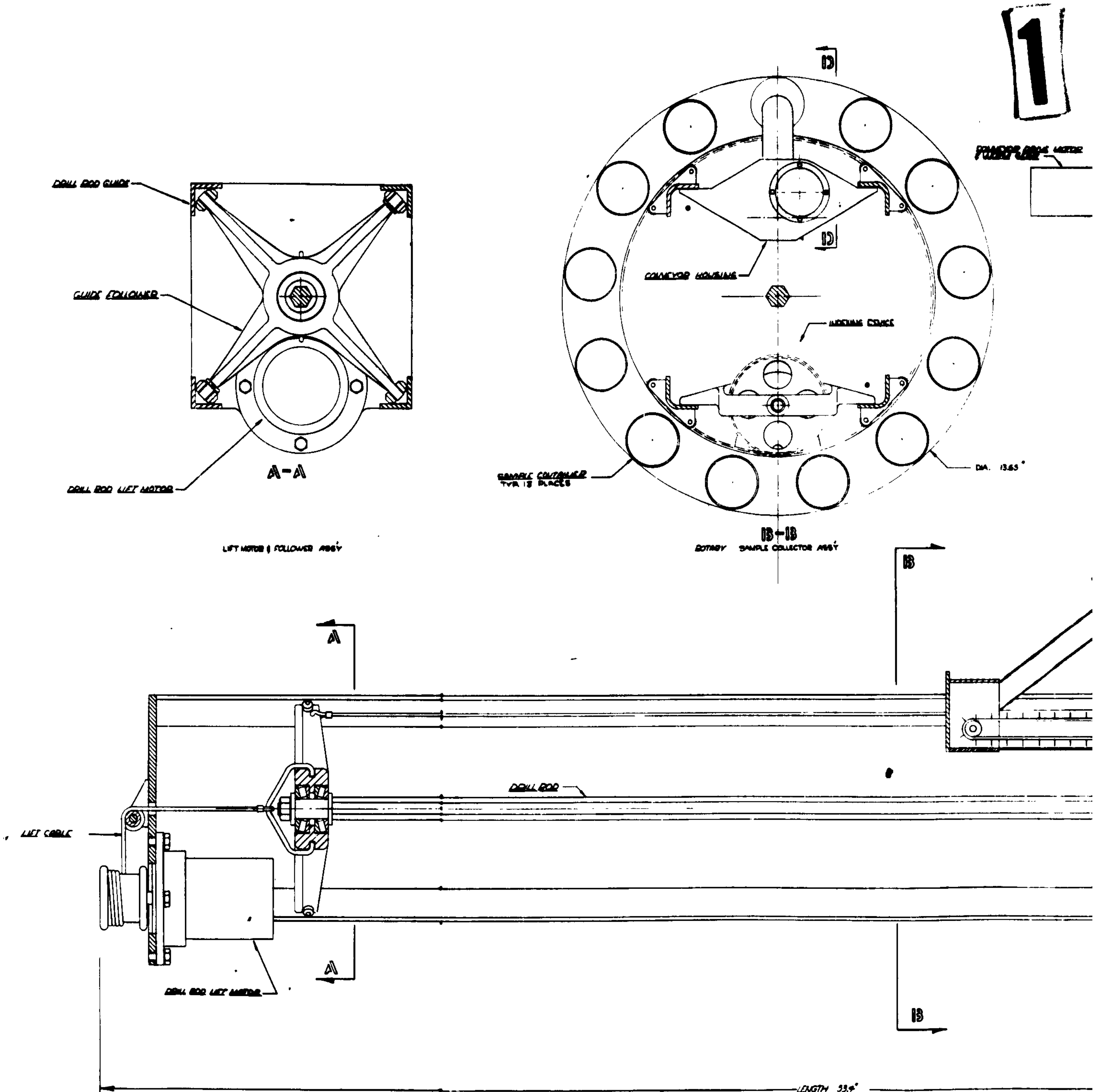


FIG. IV. 11

DRILL & SAMPLE COLLECTOR

(LUNAR ROVING VEHICLE)

2

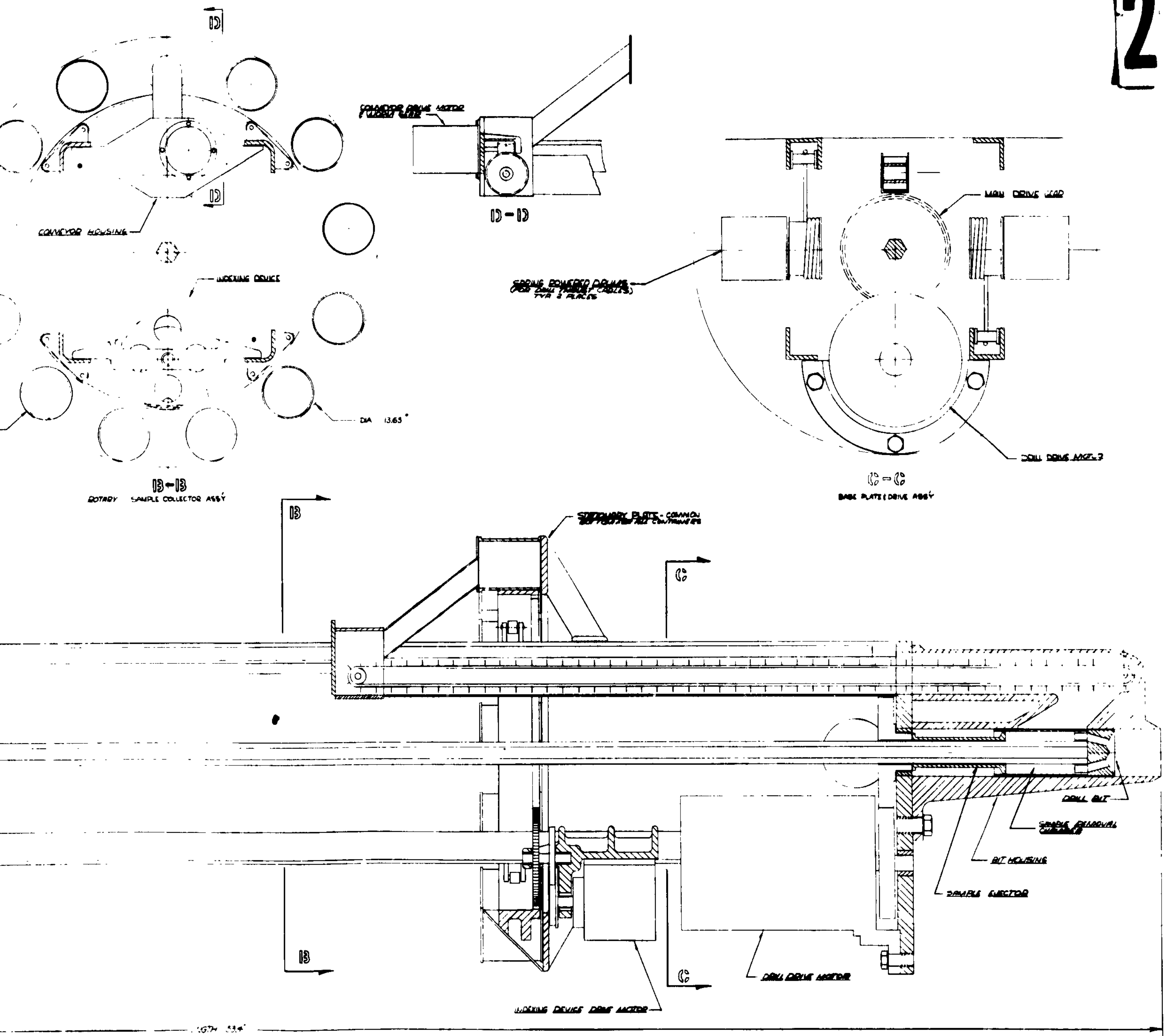


FIG. IV. 11

DRILL & SAMPLE COLLECTOR

(LUNAR ROVING VEHICLE)

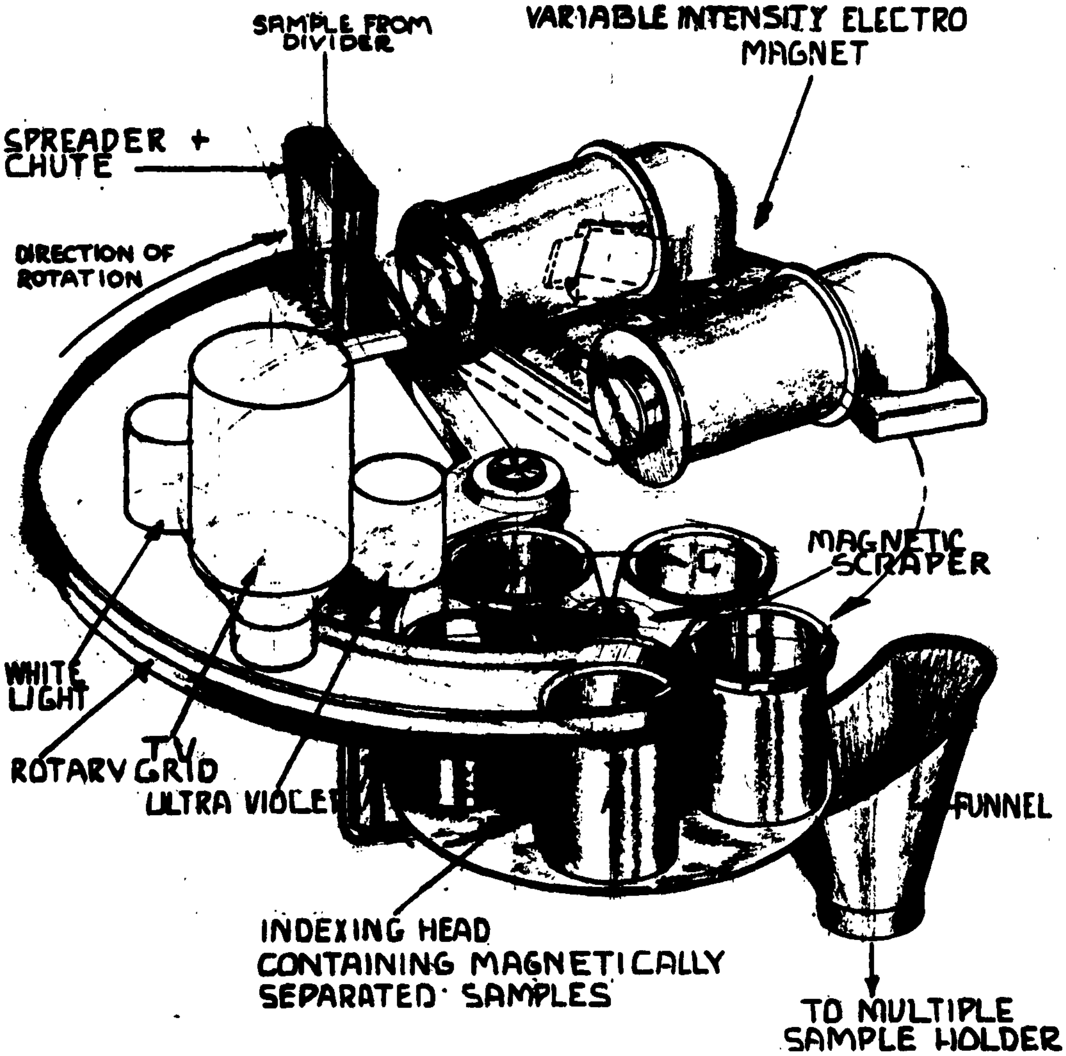


FIG IV.12 GENERAL ARRANGMENT - VISUAL + MAGNETIC INSPECTION EQUIPMENT

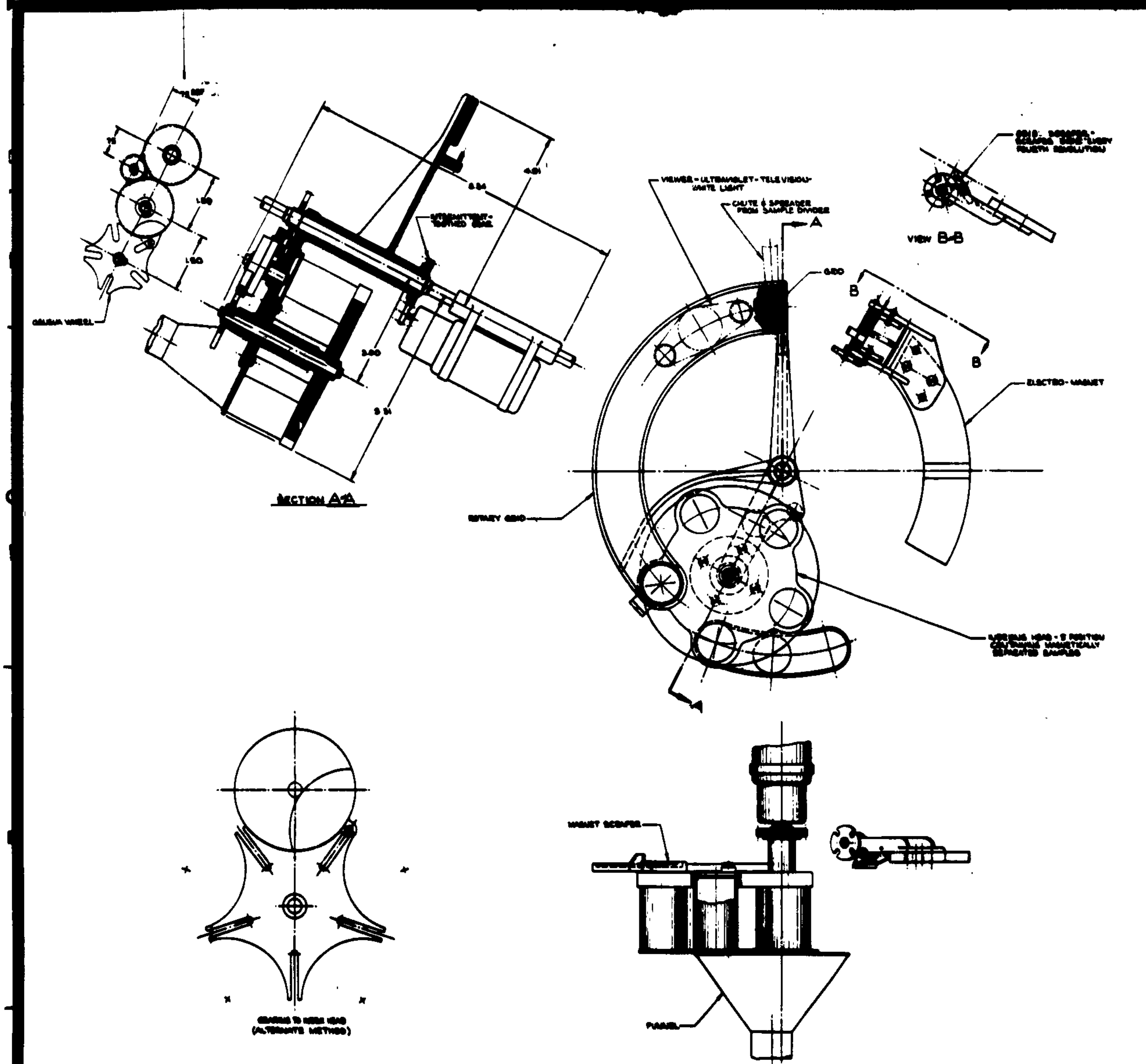
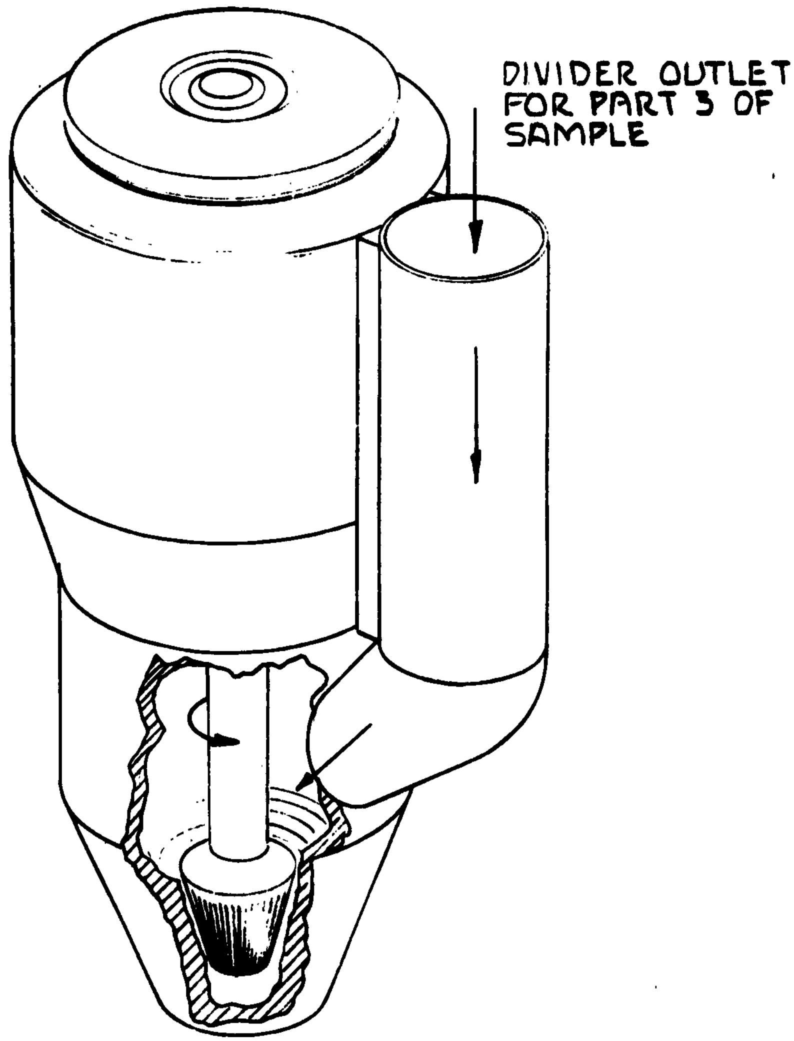


FIG. IV. 13

VISUAL & MAGNETIC INSPECTION EQUIPMENT

INSTRUMENT COMPARTMENT OF
THE LUNAR ROVING VEHICLE



SAMPLE PULVERIZER

FIG. IV. 14

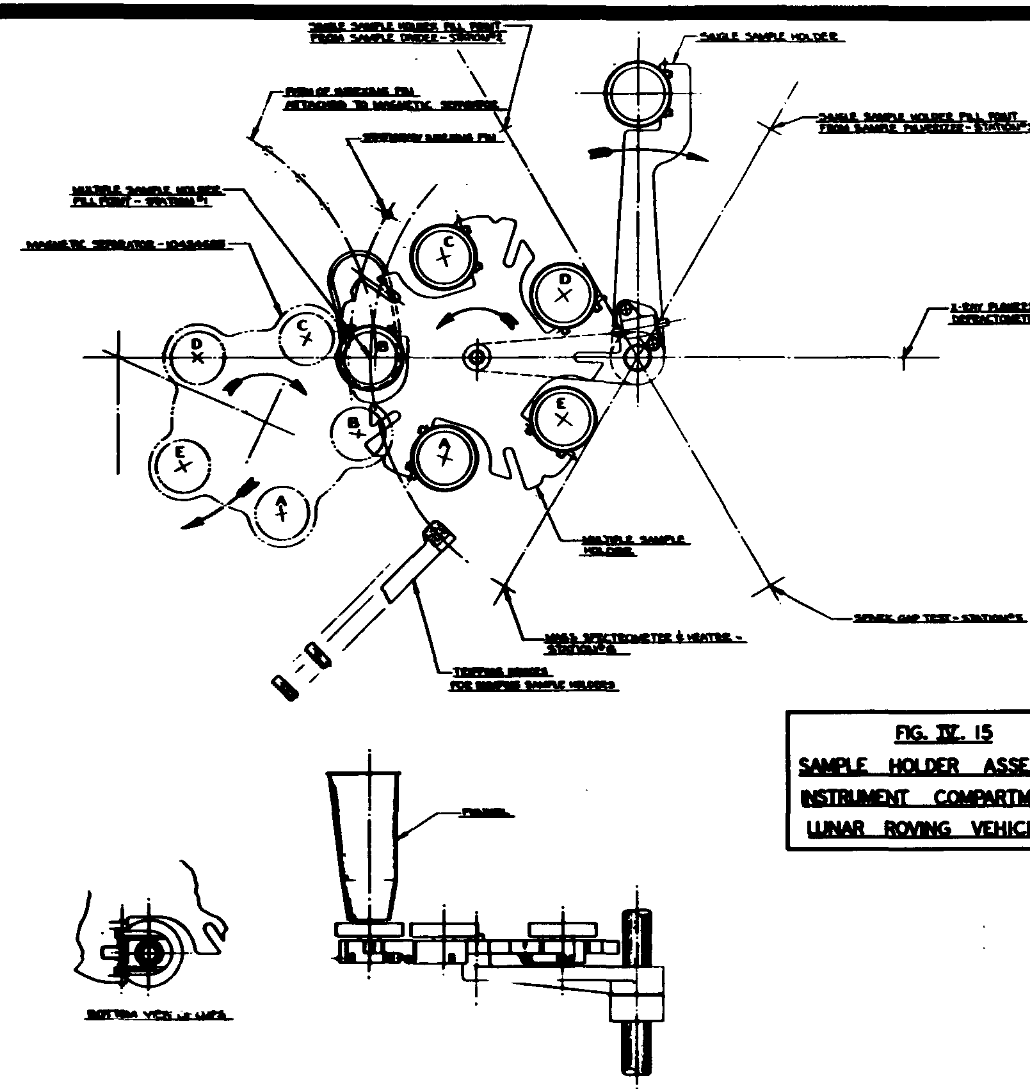


FIG. IV. 15
SAMPLE HOLDER ASSEMBLY
INSTRUMENT COMPARTMENT
LUNAR ROVING VEHICLE

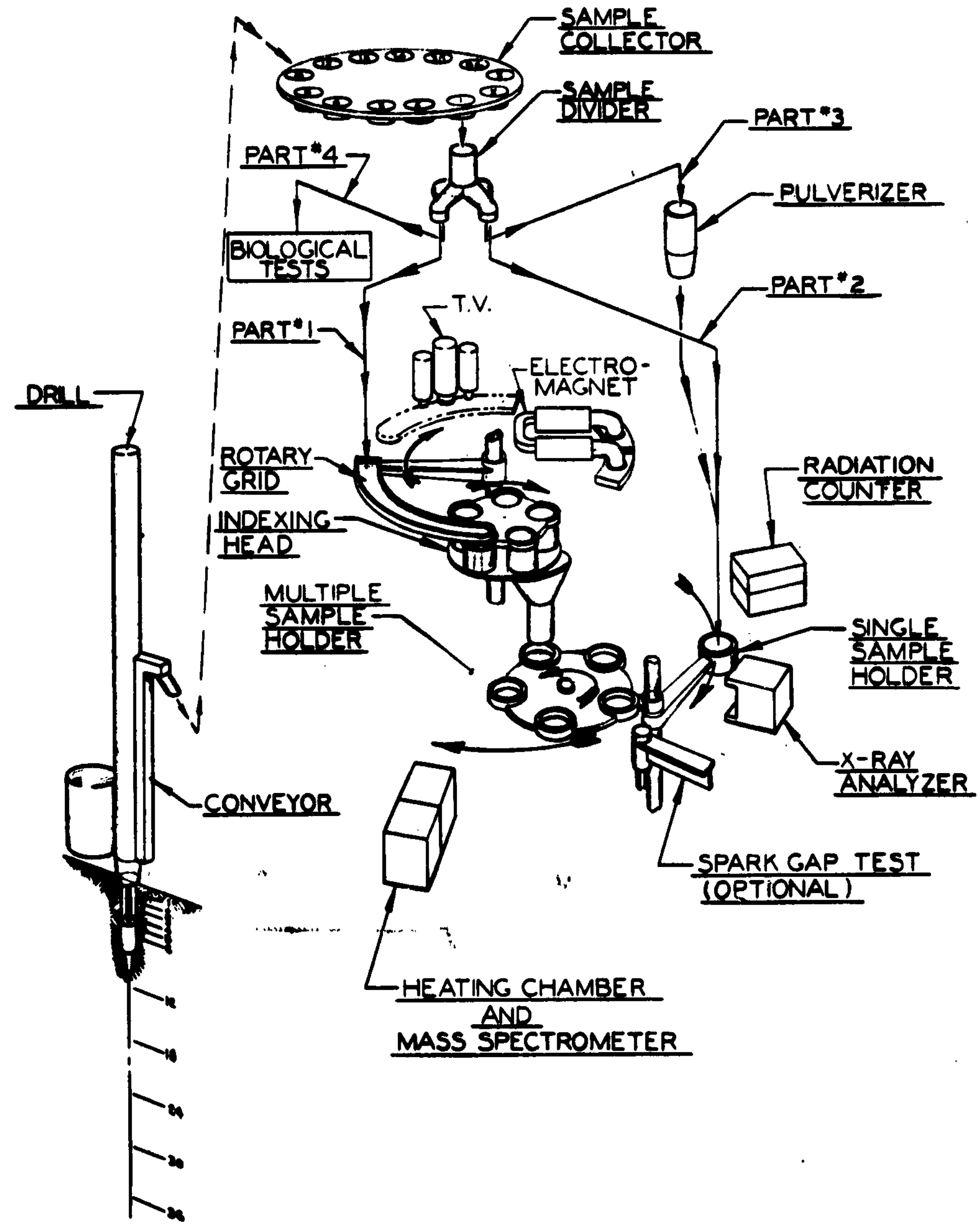


FIG. IV. 16
DRILL SAMPLER
FLOW DIAGRAM

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. ½ 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 pulveriser 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 analysing 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 θ_A, θ_B, θ_C... at which these wavelengths will be diffracted by the crystal. The counter is designed to move step-wise from one predetermined

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[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 - 2½ 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):

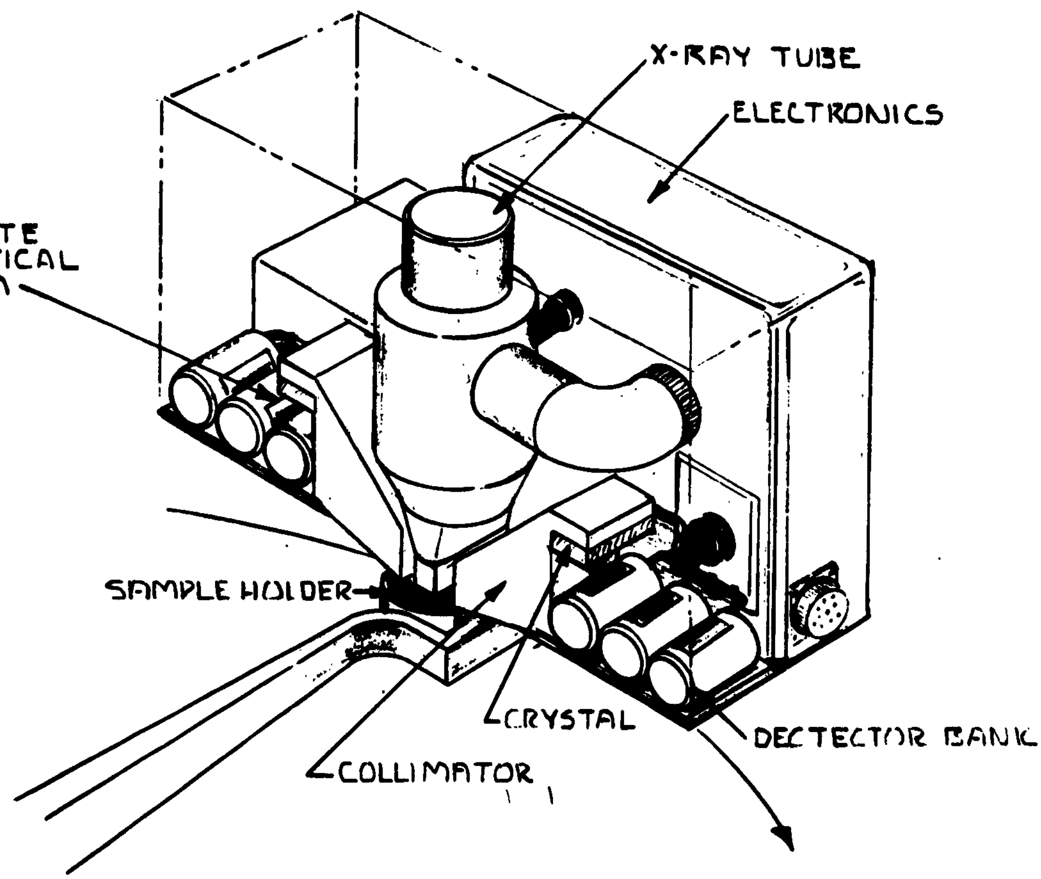

FIG. IV. 17

X-RAY SPECTROMETER

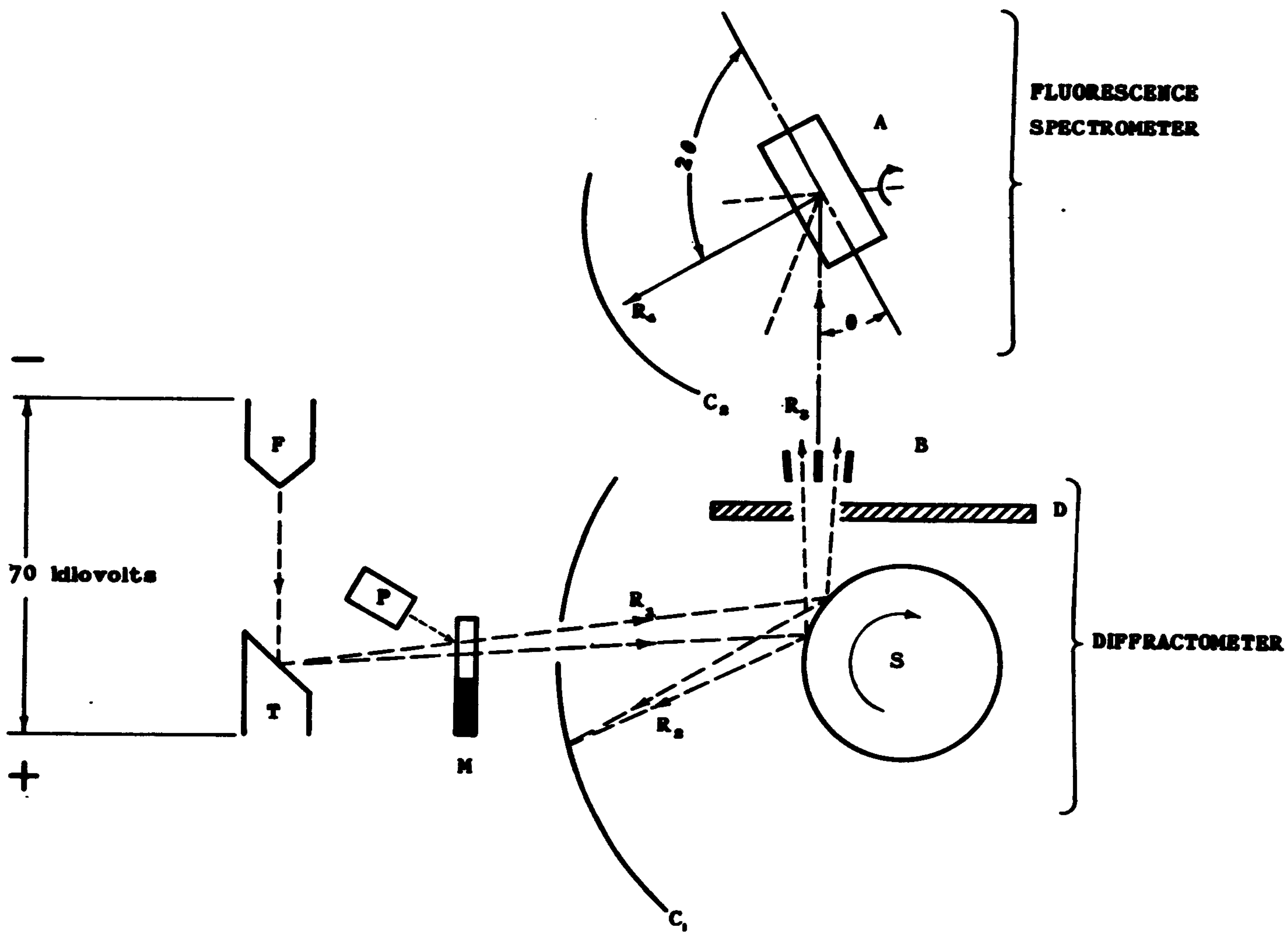

FIGURE IV. 18 SCHEMATIC DIAGRAM OF ESSENTIAL FEATURES OF X-RAY DIFFRACTOMETER AND FLUORESCENCE SPECTROMETER

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 θ-2θ relationship; P, phototube monitor viewing zinc sulfide screen on M; D, lead shield; B, Söller 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_2 , which is analyzed by the crystal A automatically placed in the proper Bragg angle position during some part of its rotation.

A high atomic number metal, tungsten with $Z = 74$, is used as target material in order to obtain sufficiently energetic X-ray photons (that is, sufficiently short wavelengths) to excite fluorescence in as many elements as possible. In principle, the excitation of tungsten by 70-kilovolt electrons in the X-ray tube 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 $12400/7000 = 0.18$ A. 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

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[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.

IV.5.4 Radioactivity as a Means of Lunar Survey. 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

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[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 A 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.

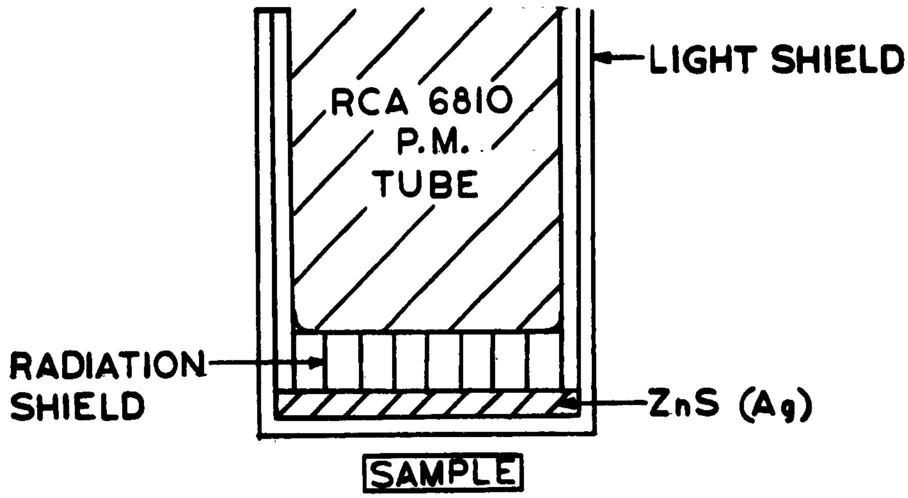

FIG IV. 19 A SIMPLE ARRANGEMENT FOR ALPHA DETECTION

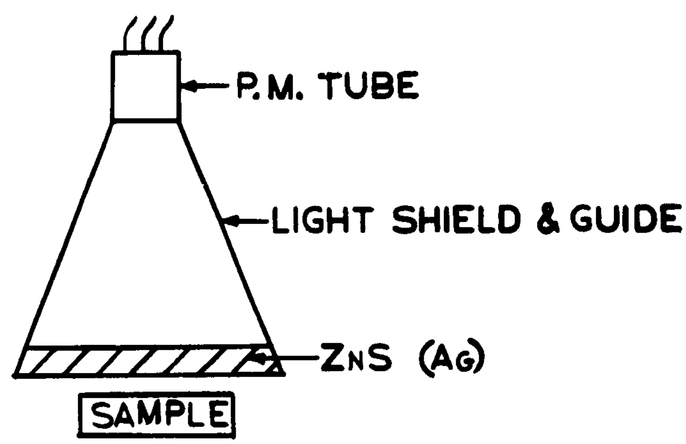

FIG. IV. 20 A MODIFIED ARRANGEMENT TO INCREASE THE RATE OF ALPHA COUNT

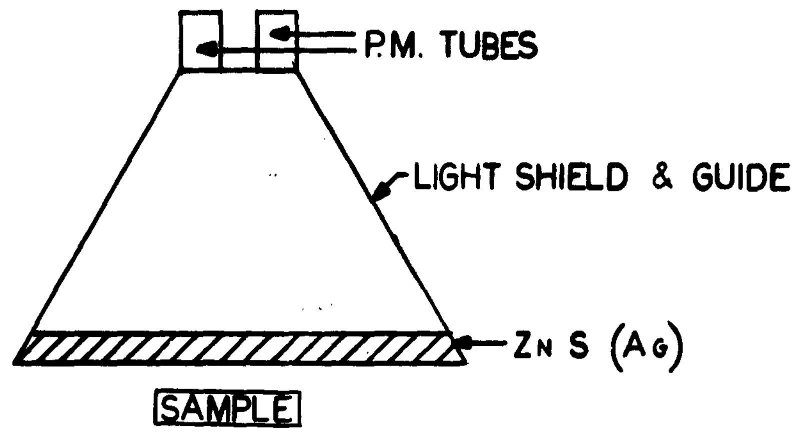

FIG.IV.21 AN ARAANGEMENT USING TWO PHOTOMULTIPLIERS IN COINCIDENCE TO REDUCE BACKGROUND COUNT.

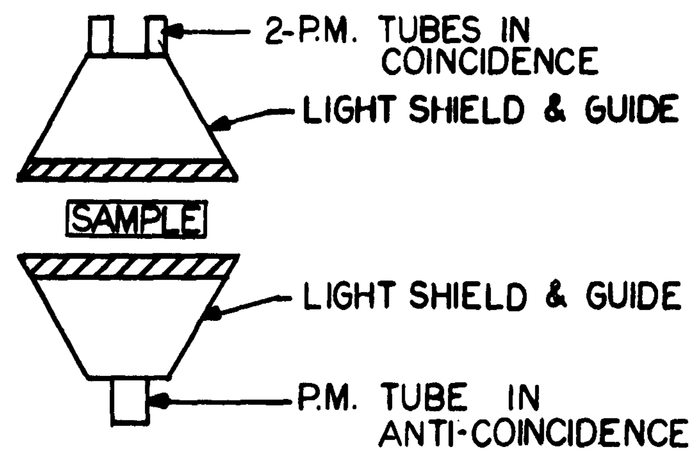

FIG.IV.22 A POSSIBLE WORKING ARRANGEMENT FOR LUNAR ALPHA COUNTING of these radiations, they convey information from farther below the surface than do alphas. For example, a one-mev beta particle can penetrate 0.4 gm/cm^2 of Al, and a one-mev gamma can penetrate roughly 16 gm/cm^2. Because of their greater 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 shortcoming 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

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[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:

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

$$\sigma T_0^4 = A + F_0 \tag{2}$$

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 $k = 16 \times 10^{-8}$ 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^{-8} 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 absorged 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 $Q \ldots Q_n$ connected to R-C circuits which may be discharged after variable and adjustable time of charge accumulation by electronically closing switch S which causes discharge of capacitor and, in turn, creates a cathode-to-grid potential 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

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

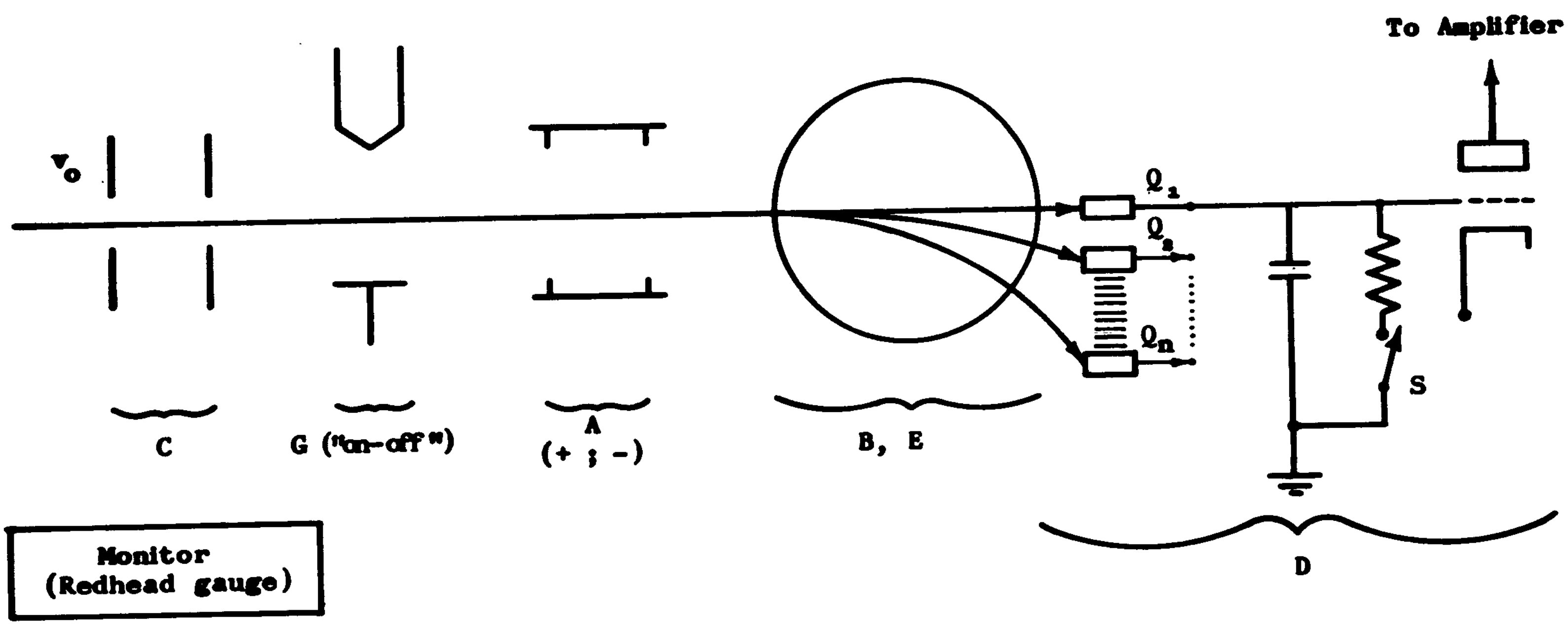

FIGURE IV.23 PROPOSED MASS-ION SPECTROMETER WITH CHARGE INTEGRATION FOR EXTREMELY LOW INTENSITIES

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

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[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.

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

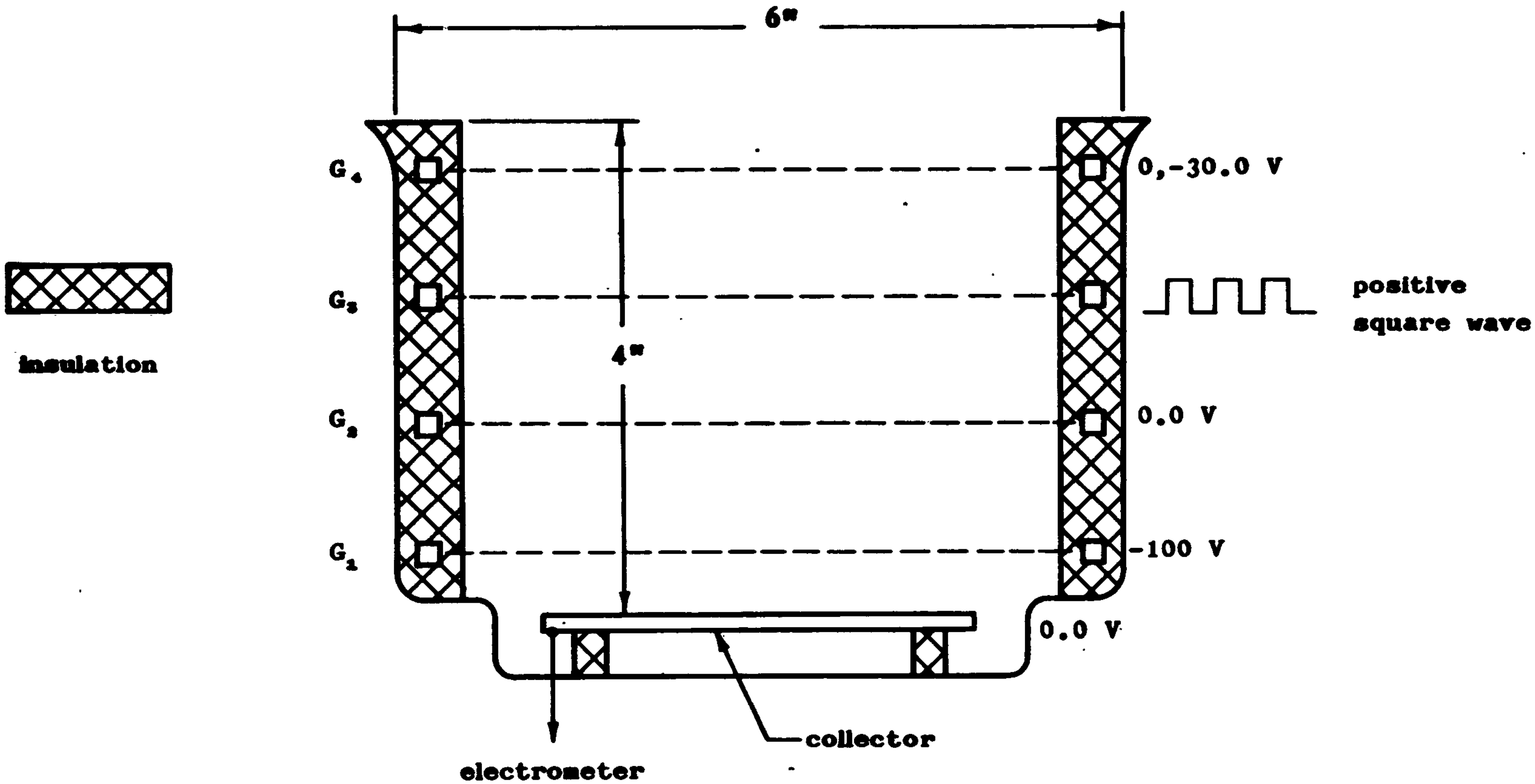

FIGURE IV.24 DIAGRAM OF THE PLASMA PROBE

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).

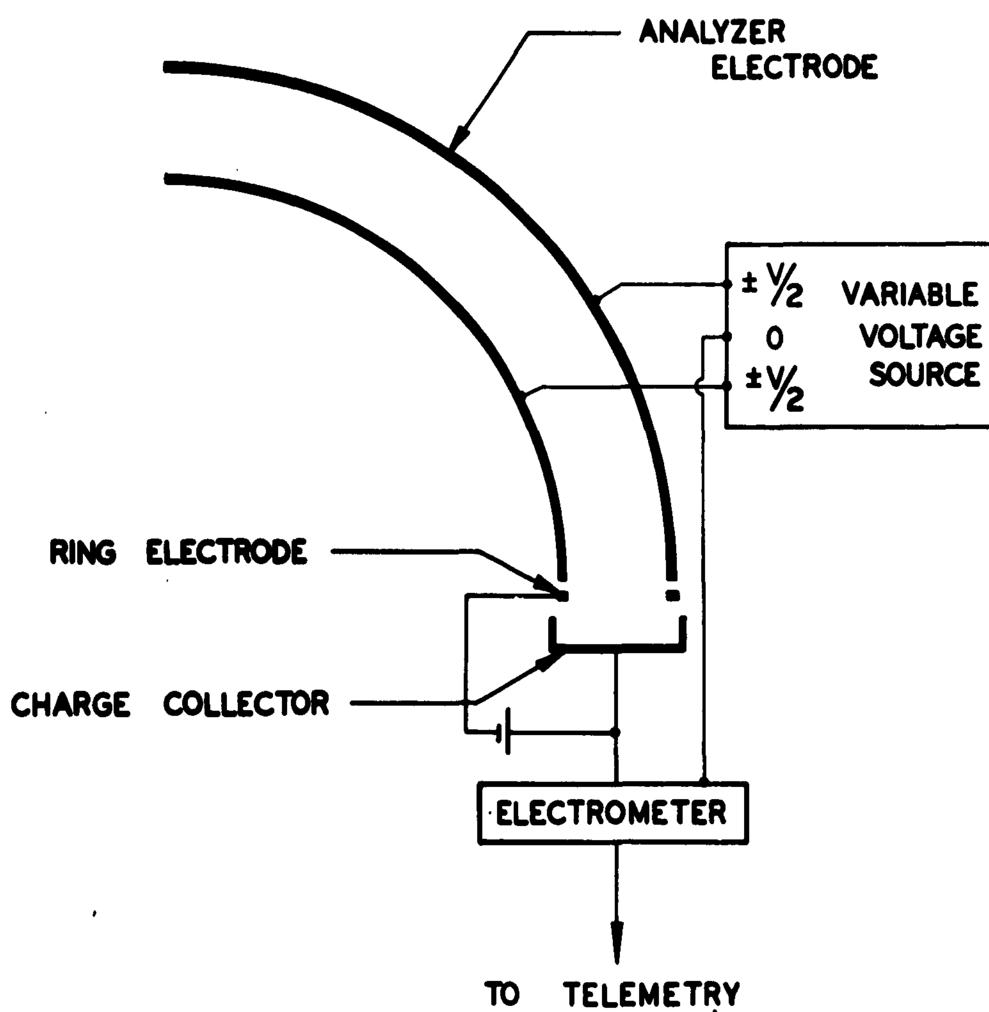

FIG. IV.25

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

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[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

[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

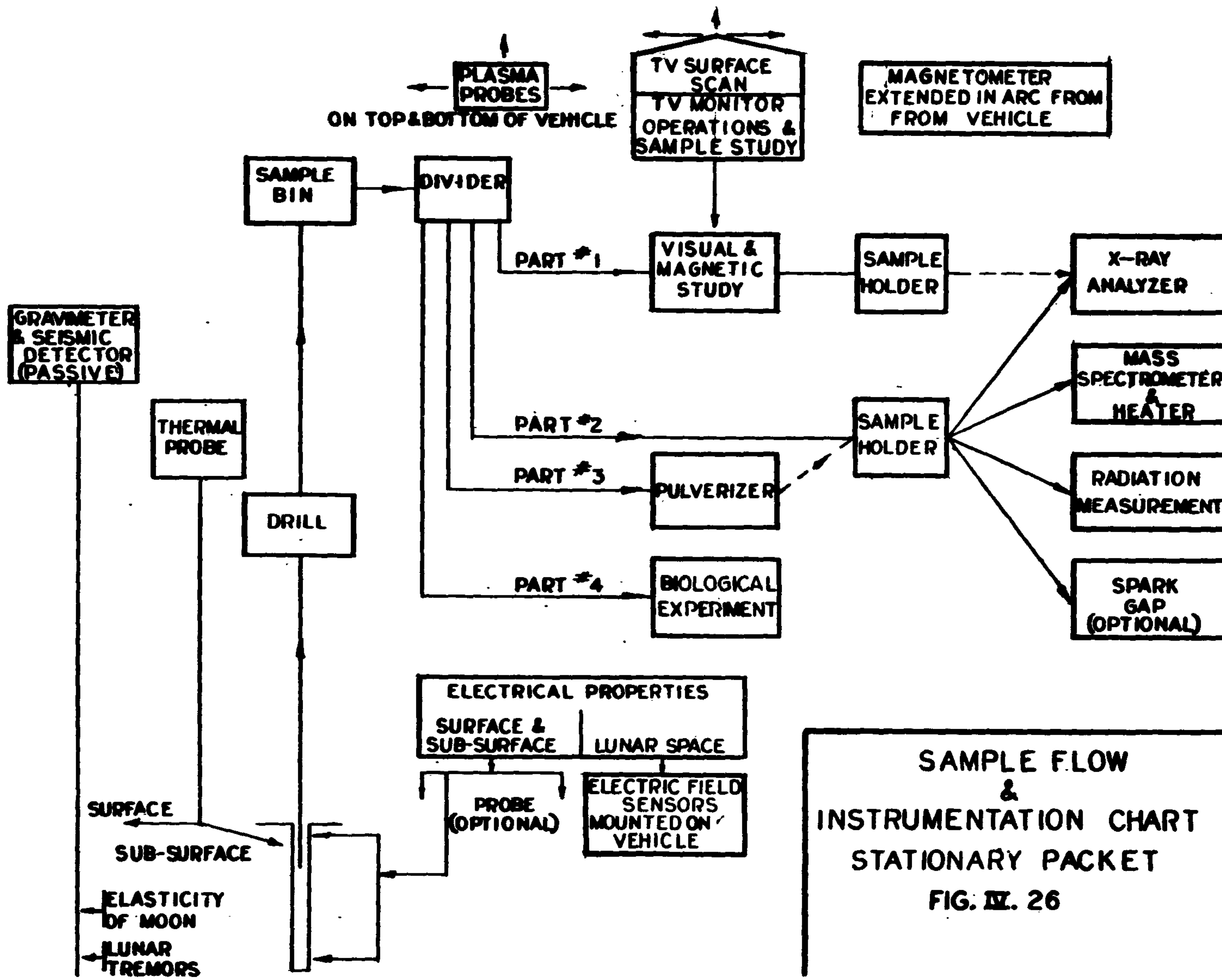
(c) Mass-ion spectrometer

(d) Redhead density monitor

(e) Electric field probes

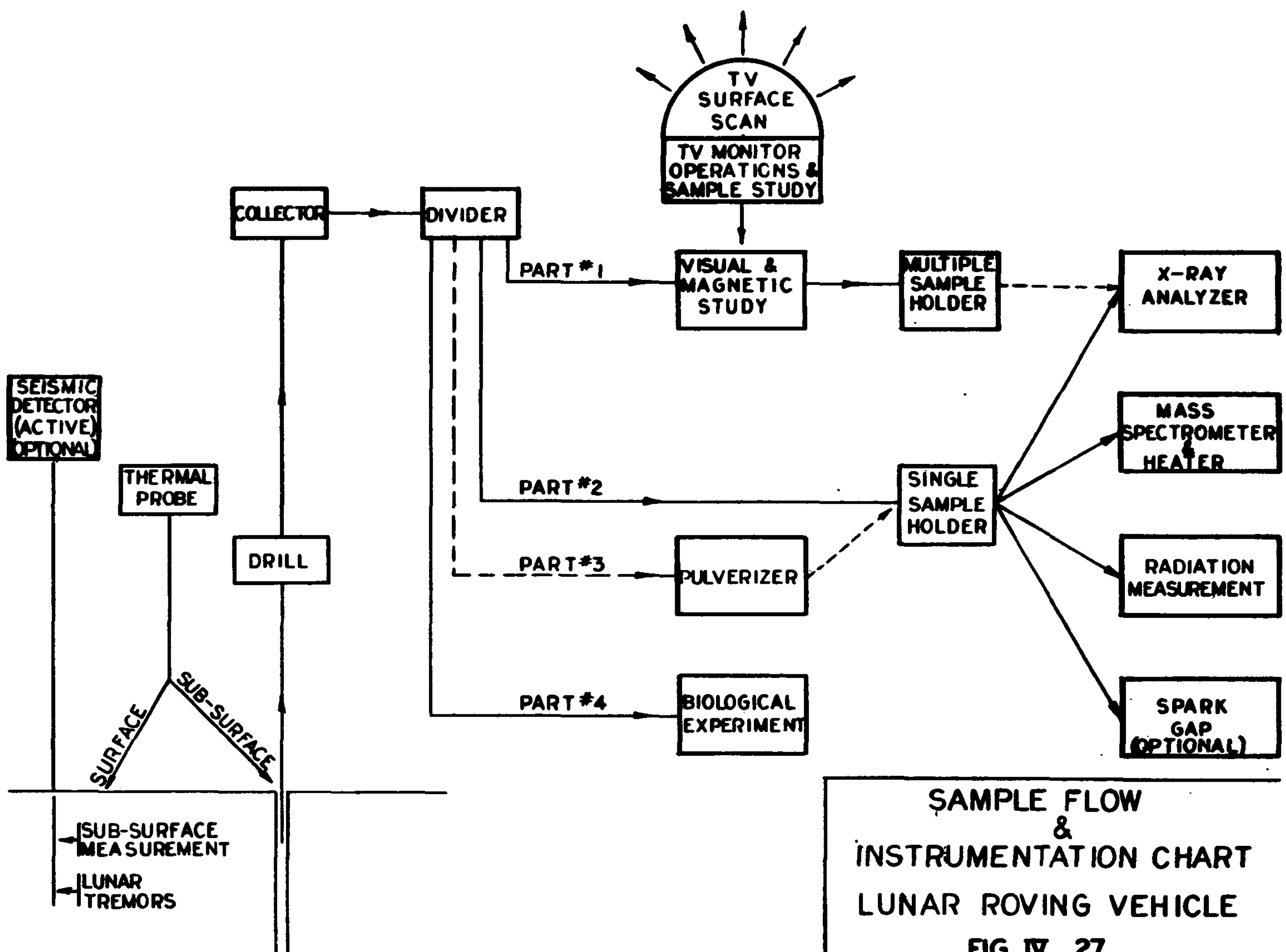
By selecting the optimum materials and components in making an instrument, the temperature range at which it may operate with stability may be increased. Instruments which require careful temperature calibration may still be used on the moon through proper selection of their temperature sensitive elements. 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	¼in. 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

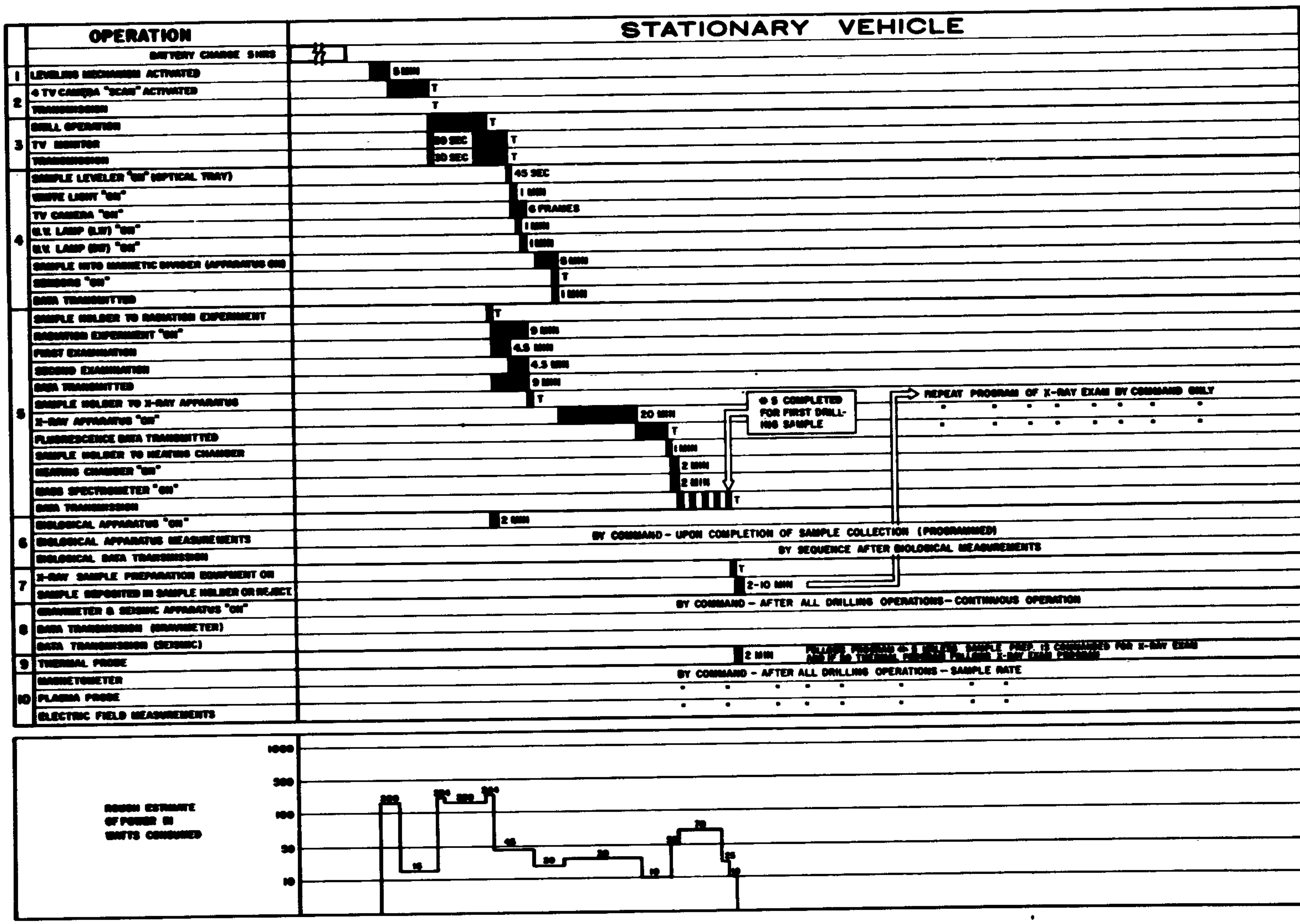
* S = Stationary Packet; R = Roving Vehicle

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

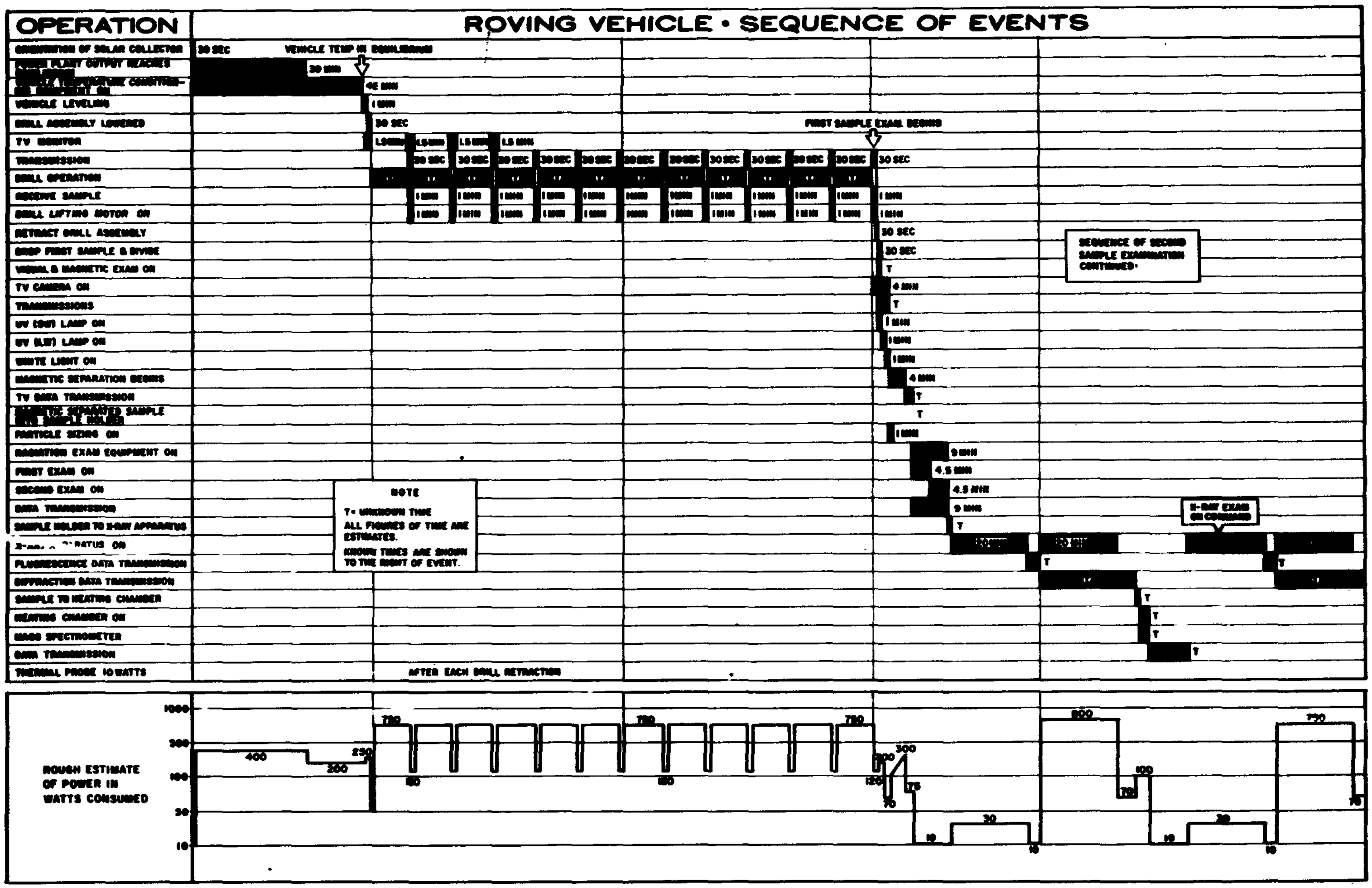
* S = Stationary Packet ; R = Roving Vehicle.

TABLE IV. 2



STATIONARY VEHICLE
OPERATION
BATTERY CHARGE 5 HRS
LEVELING MECHANISM ACTIVATED
4 TV CAMERA "SCAN" ACTIVATED
TRANSMISSION
DRILL OPERATION
TV MONITOR
TRANSMISSION
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
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
BIOLOGICAL APPARATUS "ON"
BIOLOGICAL APPARATUS MEASUREMENTS
BIOLOGICAL DATA TRANSMISSION
X-RAY SAMPLE PREPARATION EQUIPMENT ON
SAMPLE DEPOSITED IN SAMPLE HOLDER OR REJECT
GRAVIMETER & SEISMIC APPARATUS "ON"
DATA TRANSMISSION (GRAVIMETER)
DATA TRANSMISSION (SEISMIC)
THERMAL PROBE
MAGNETOMETER
PLASMA PROBE
ELECTRIC FIELD MEASUREMENTS
5 MIN
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
2 MIN
2-10 MIN
2 MIN
T
⊕ 5 COMPLETED FOR FIRST DRILLING SAMPLE
REPEAT PROGRAM OF X-RAY EXAM BY COMMAND ONLY
BY COMMAND - UPON COMPLETION OF SAMPLE COLLECTION (PROGRAMMED)
BY SEQUENCE AFTER BIOLOGICAL MEASUREMENTS
BY COMMAND - AFTER ALL DRILLING OPERATIONS - CONTINUOUS OPERATION
BY COMMAND - AFTER ALL DRILLING OPERATIONS - SAMPLE RATE
ROUGH ESTIMATE OF POWER IN WATTS CONSUMED
1000
300
100
30
10
15
45
20
30
10
70
25

TABLE IV.3


OPERATION
ROVING VEHICLE • SEQUENCE OF EVENTS
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
PARTICLE SIZING ON
RADIATION EXAM EQUIPMENT ON
FIRST EXAM ON
SECOND EXAM ON
DATA TRANSMISSION
SAMPLE HOLDER TO X-RAY APPARATUS
FLUORESCENCE DATA TRANSMISSION
DIFFRACTION DATA TRANSMISSION
SAMPLE TO HEATING CHAMBER
HEATING CHAMBER ON
MASS SPECTROMETER
DATA TRANSMISSION
THERMAL PROBE 10 WATTS
VEHICLE TEMP IN EQUILIBRIUM
FIRST SAMPLE EXAM BEGINS
AFTER EACH DRILL RETRACTION
NOTE
T = UNKNOWN TIME
ALL FIGURES OF TIME ARE ESTIMATES.
KNOWN TIMES ARE SHOWN TO THE RIGHT OF EVENT.
SEQUENCE OF SECOND SAMPLE EXAMINATION CONTINUED.
X-RAY EXAM ON COMMAND
ROUGH ESTIMATE OF POWER IN WATTS CONSUMED

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

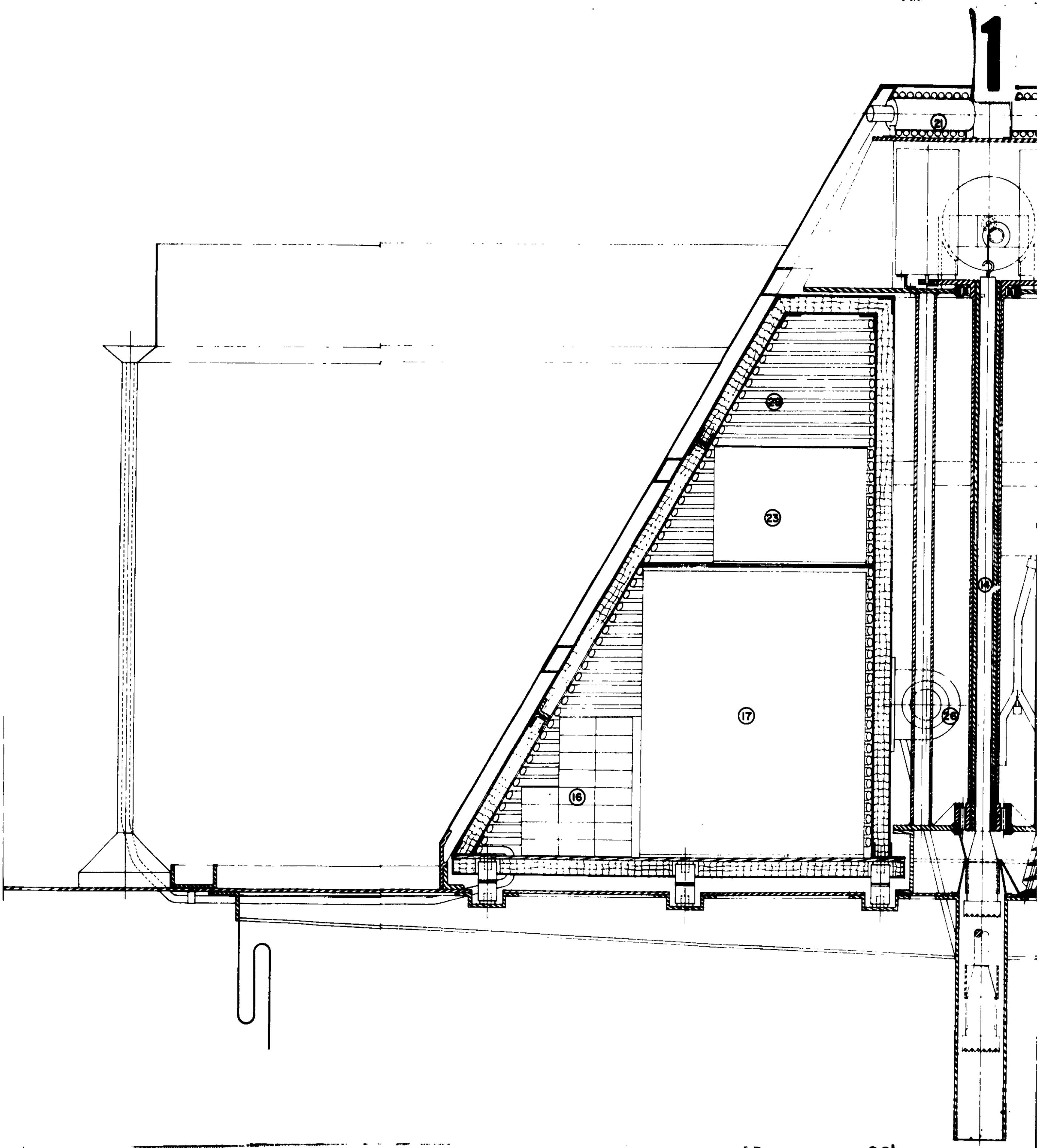
(26) Thermal probe

(27) Motor for spiral sample conveyor

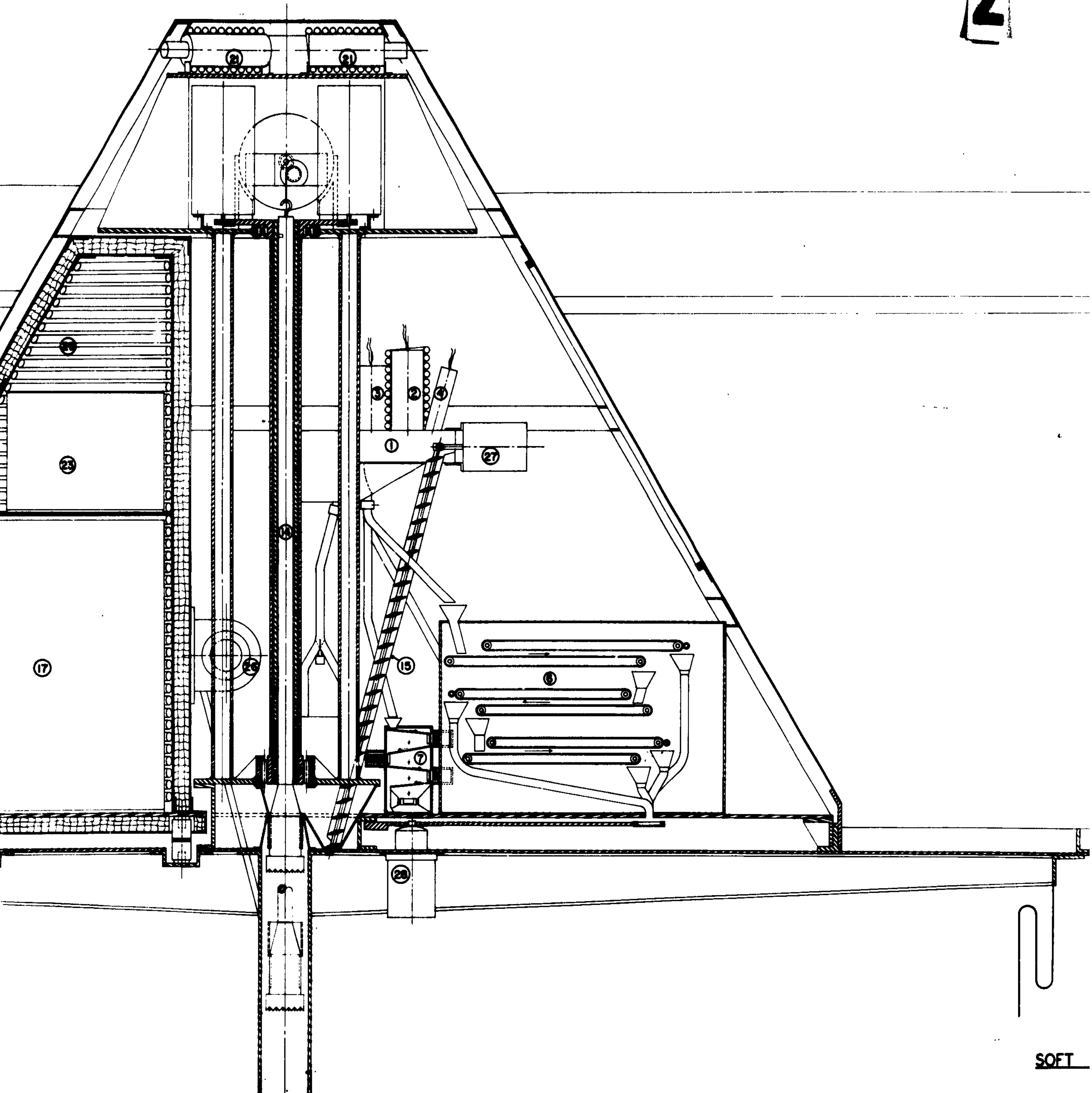
(28) Motor for sample holder

(29) Coiled tubing for temperature control

(30) Further apparatus as required


1
21
23
17
16
14
26

2



SOFT

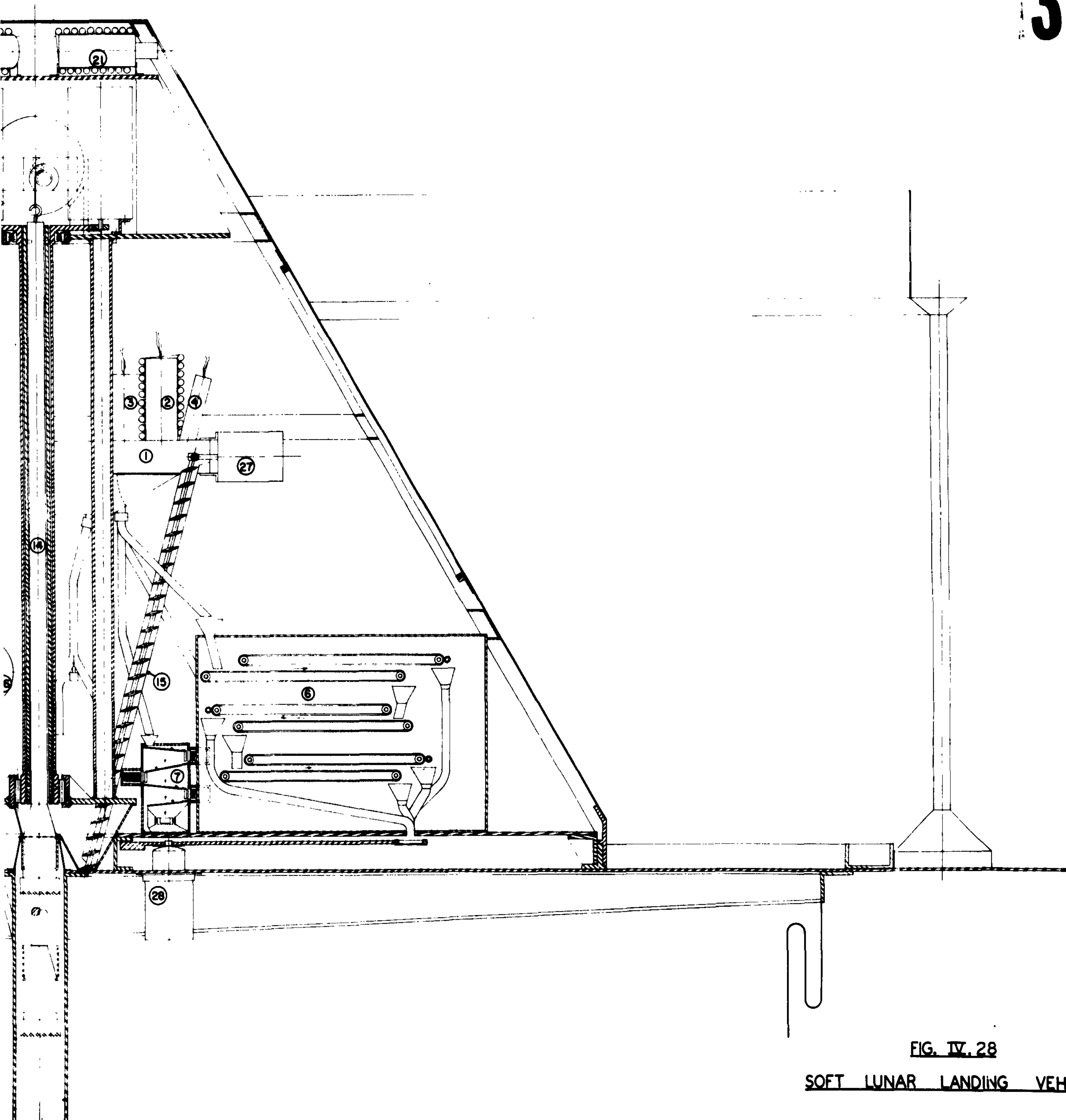
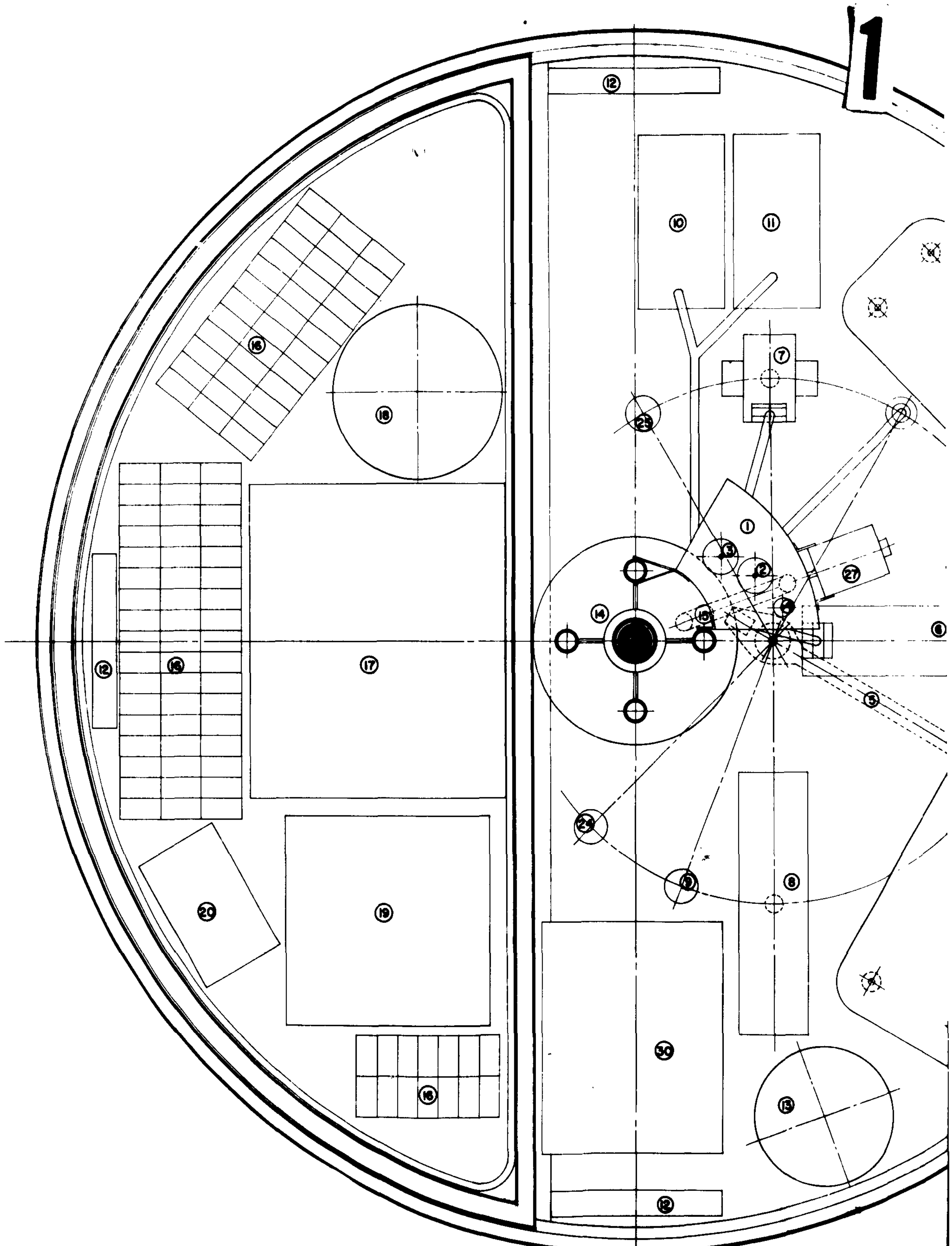


FIG. IV. 28

SOFT LUNAR LANDING VEHICLE



2:25

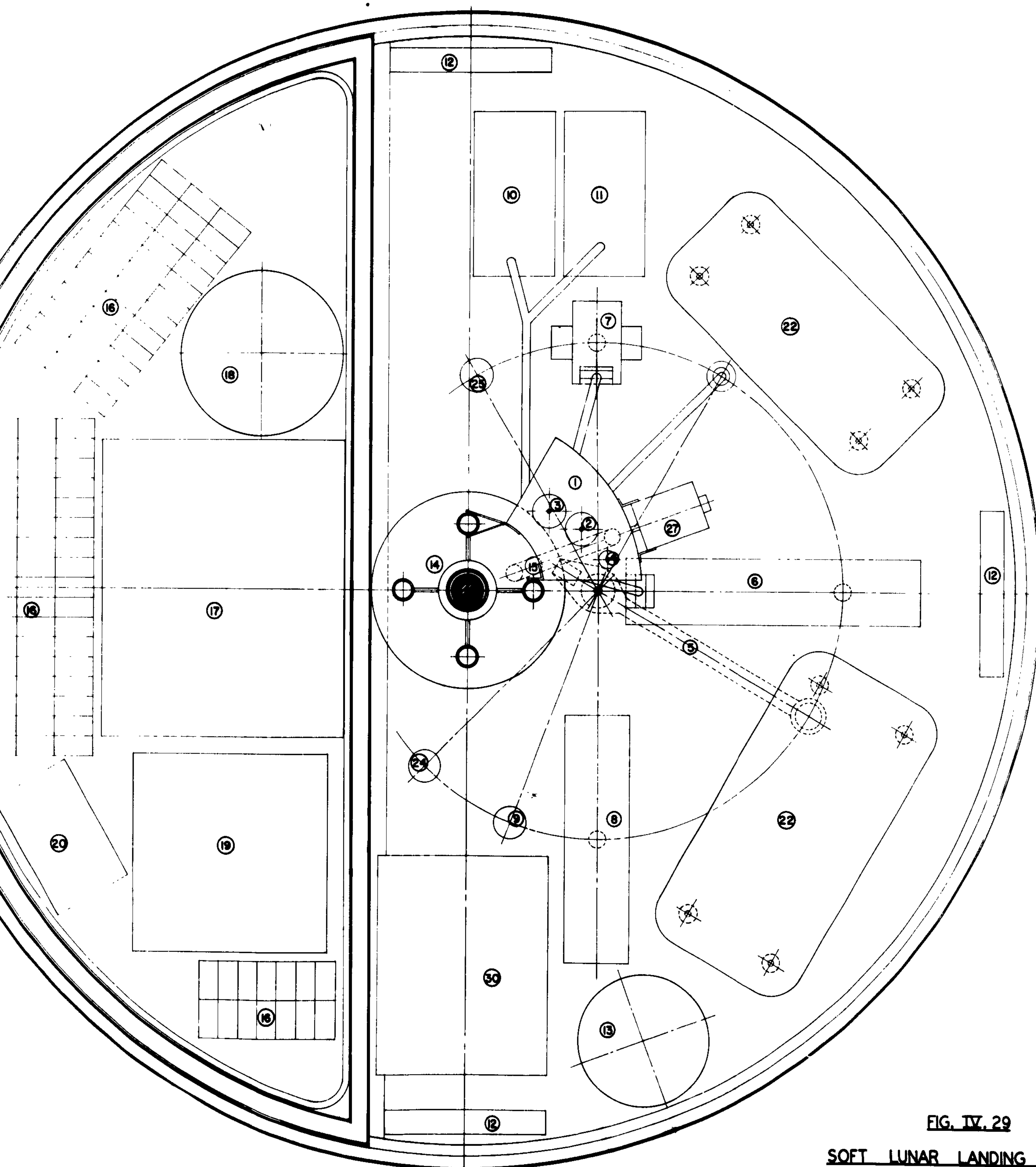


FIG. IV. 29

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

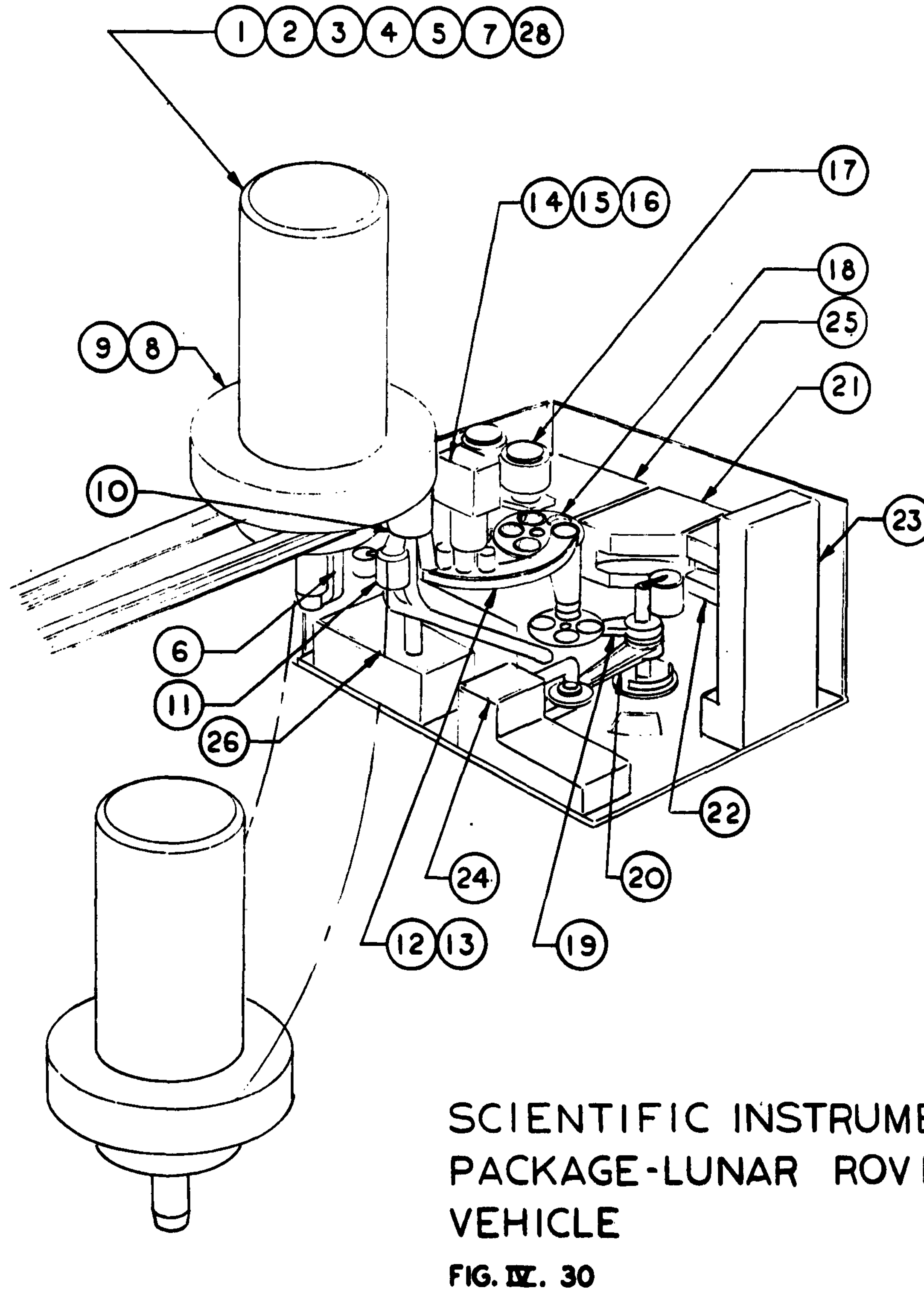
(17) Electro-magnet

(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 tailered 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

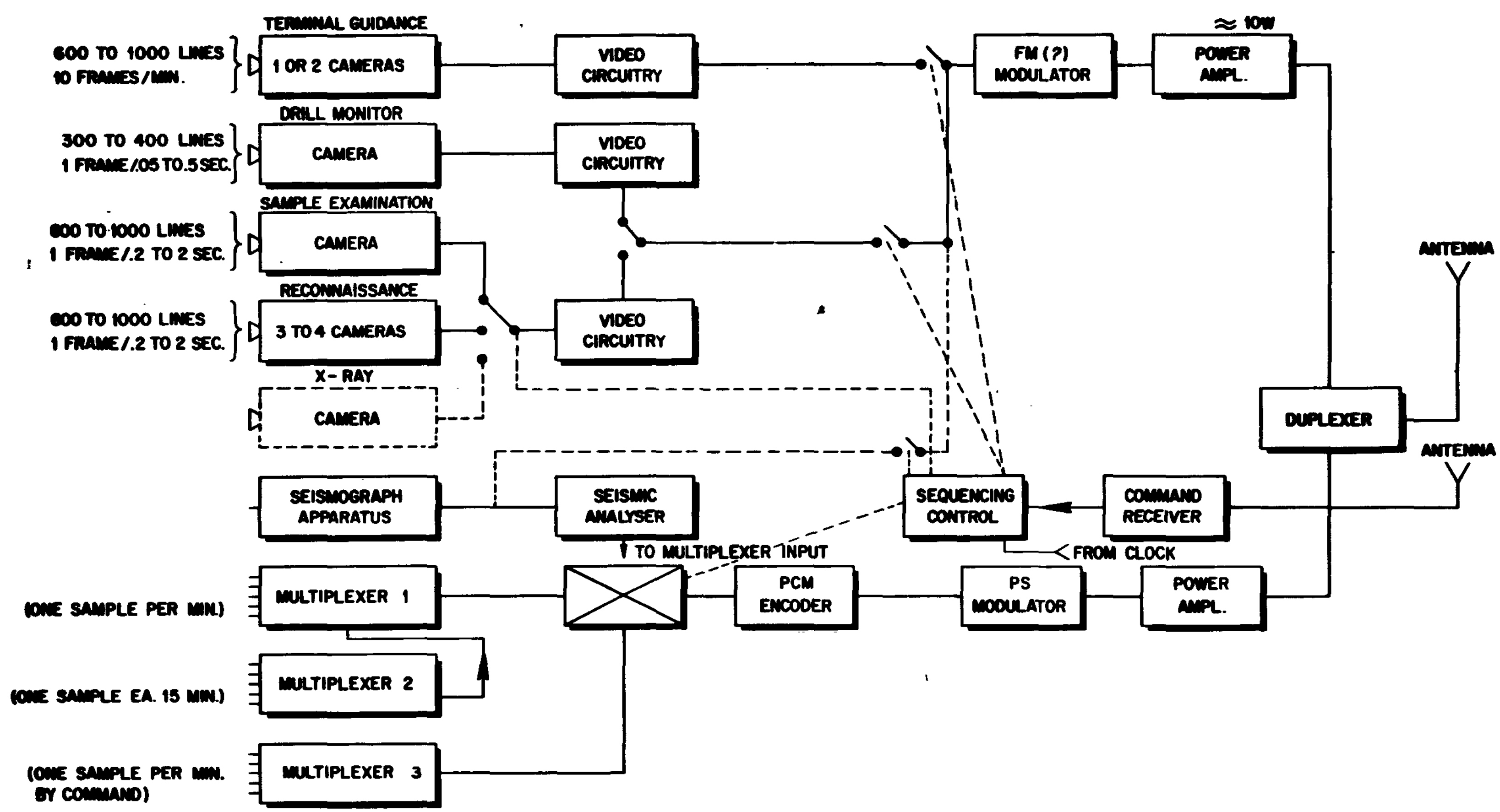


FIG. IV. 31

COMMUNICATION SYSTEM FOR SOFT LUNAR LANDING (STATIONARY VEHICLE)

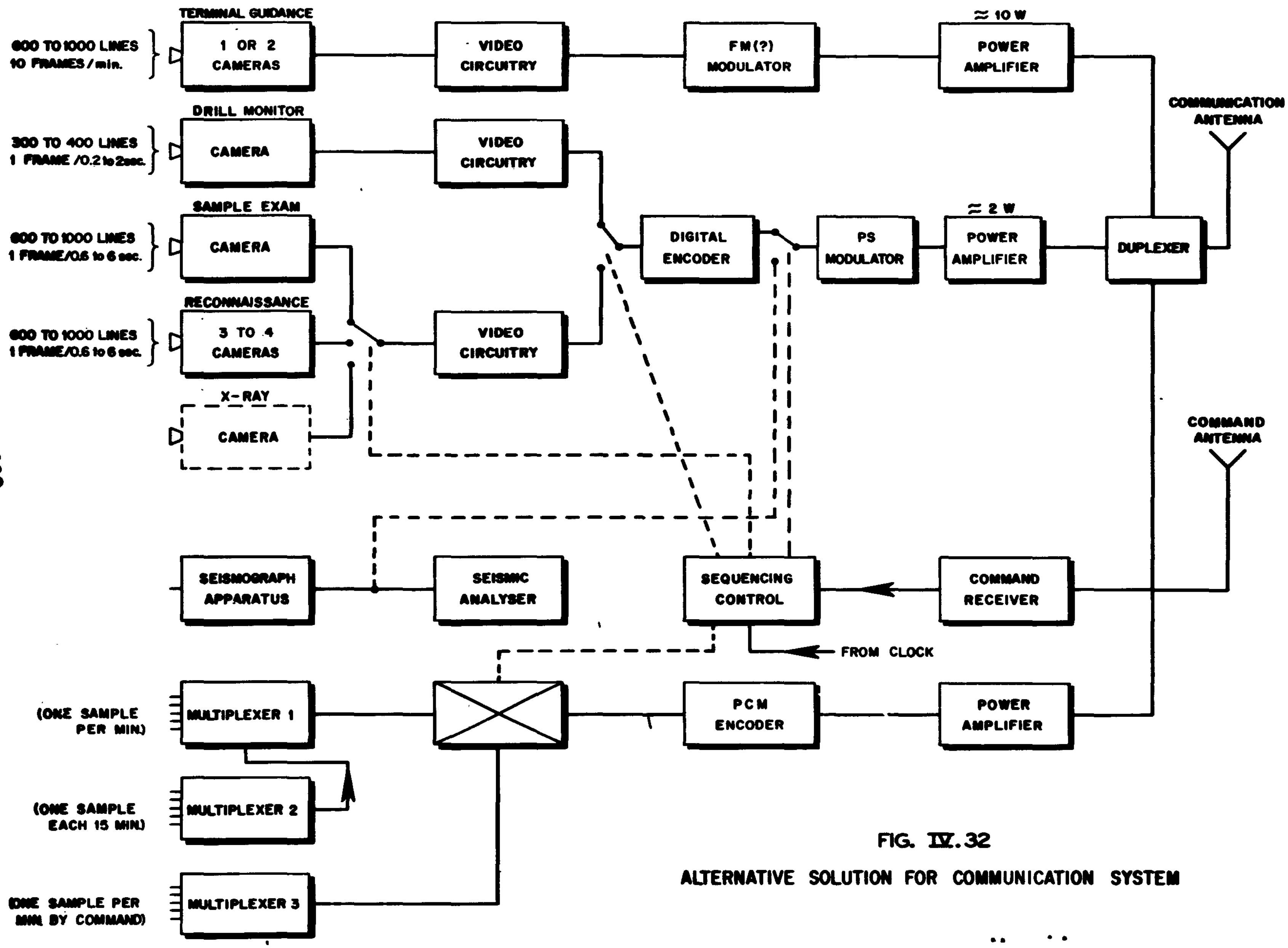


FIG. IV.32

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.

ARMY ORDNANCE MISSILE COMMAND
REDSTONE ARSENAL, ALABAMA

CHAPTER VI*

POWER SUPPLIES

*(This chapter was taken from ABMA Report No. DV-TR-2-60, dtd 1 February 1960)

DEVELOPMENT OPERATIONS DIVISION
ARMY BALLISTIC MISSILE AGENCY

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.

<u>SATURN Stage IV for Landing Package.</u> 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.

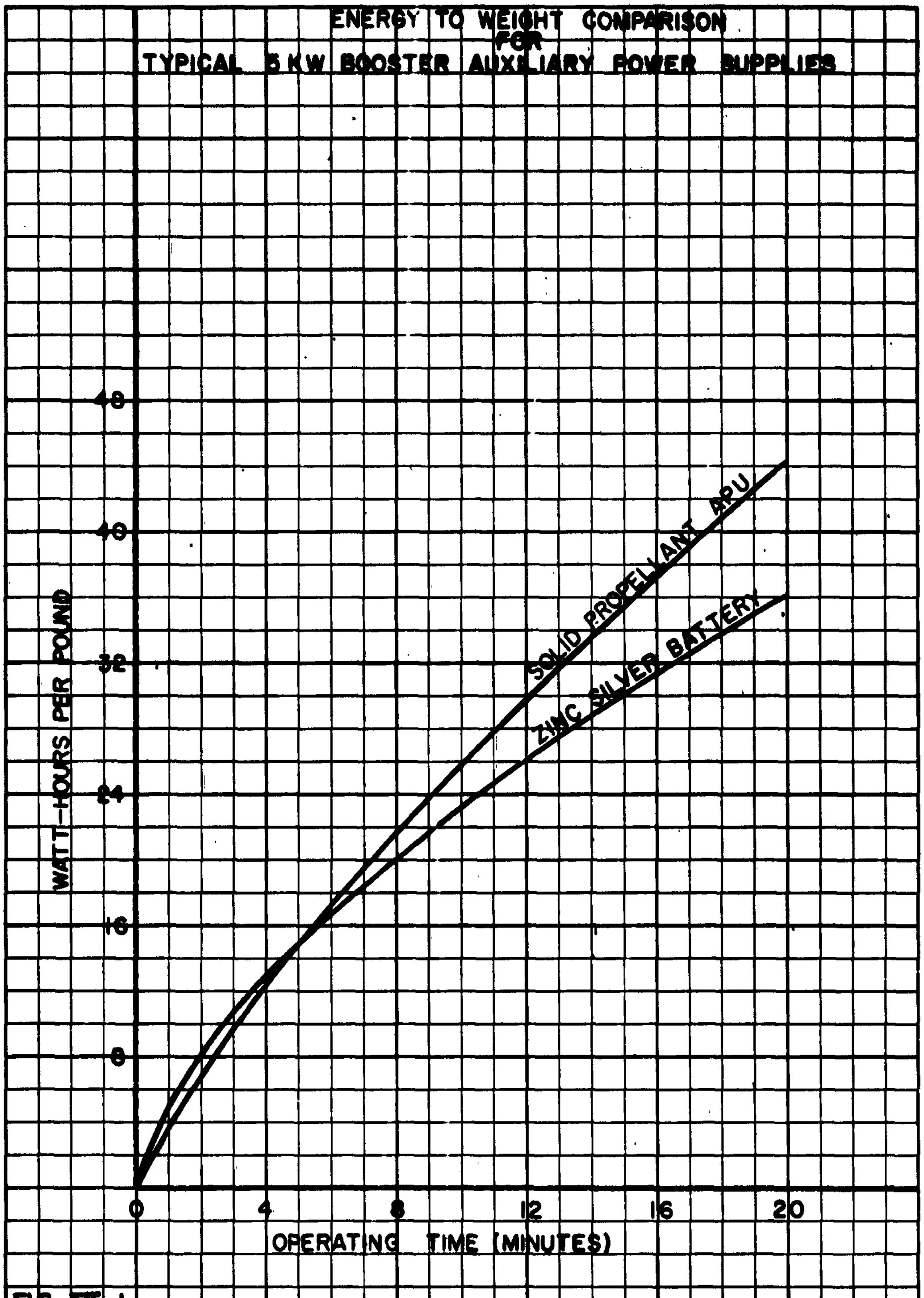

ENERGY TO WEIGHT COMPARISON
FOR
TYPICAL 5 KW BOOSTER AUXILIARY POWER SUPPLIES
SOLID PROPELLANT APU
ZINC SILVER BATTERY
WATT-HOURS PER POUND
48
40
32
24
16
8
0
4
8
12
16
20
OPERATING TIME (MINUTES)

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:

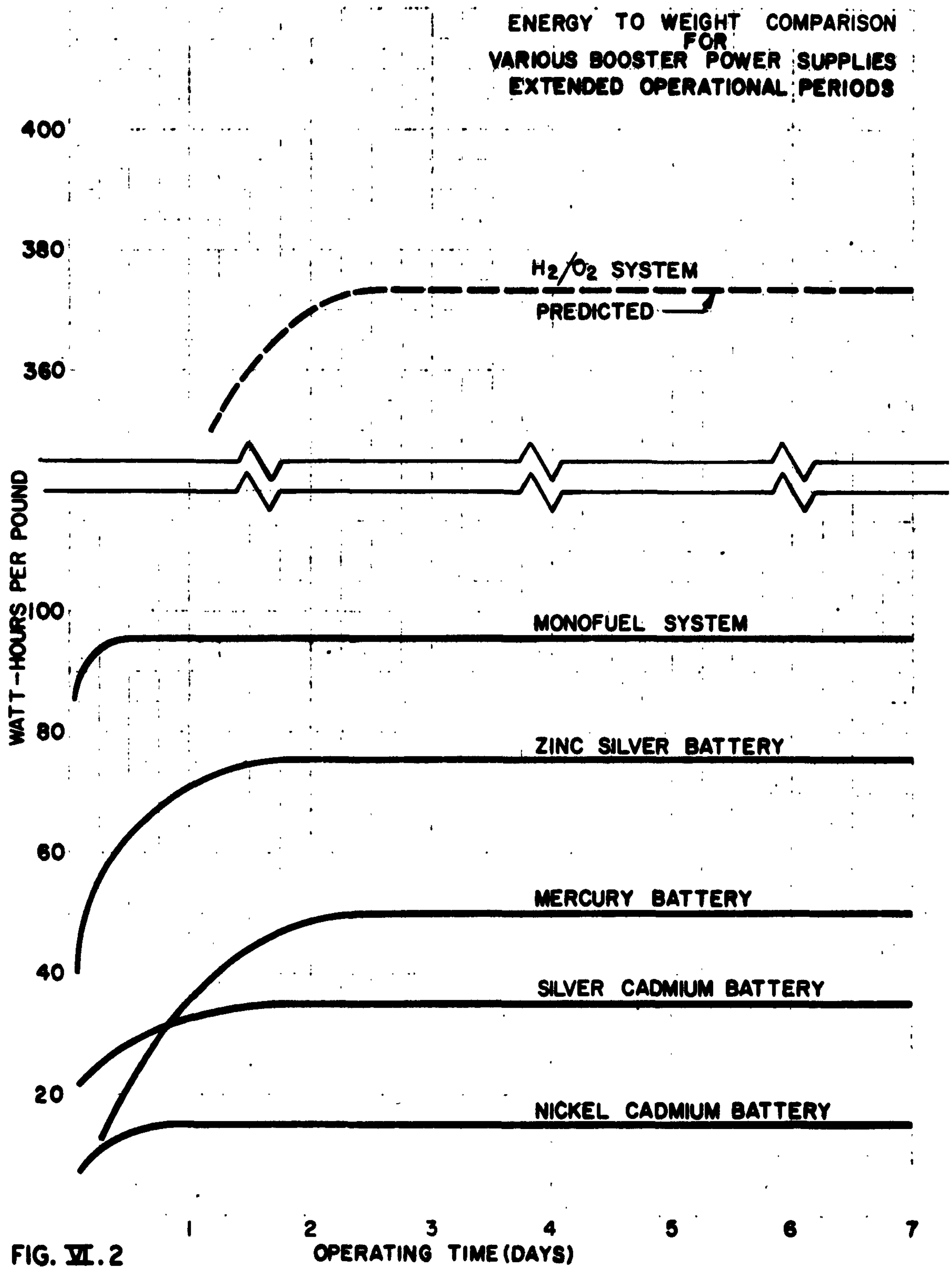

ENERGY TO WEIGHT COMPARISON FOR VARIOUS BOOSTER POWER SUPPLIES EXTENDED OPERATIONAL PERIODS
400
380
360
100
80
60
40
20
WATT-HOURS PER POUND
H_2/O_2 SYSTEM
PREDICTED
MONOFUEL SYSTEM
ZINC SILVER BATTERY
MERCURY BATTERY
SILVER CADMIUM BATTERY
NICKEL CADMIUM BATTERY
1
2
3
4
5
6
7
OPERATING TIME (DAYS)

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.

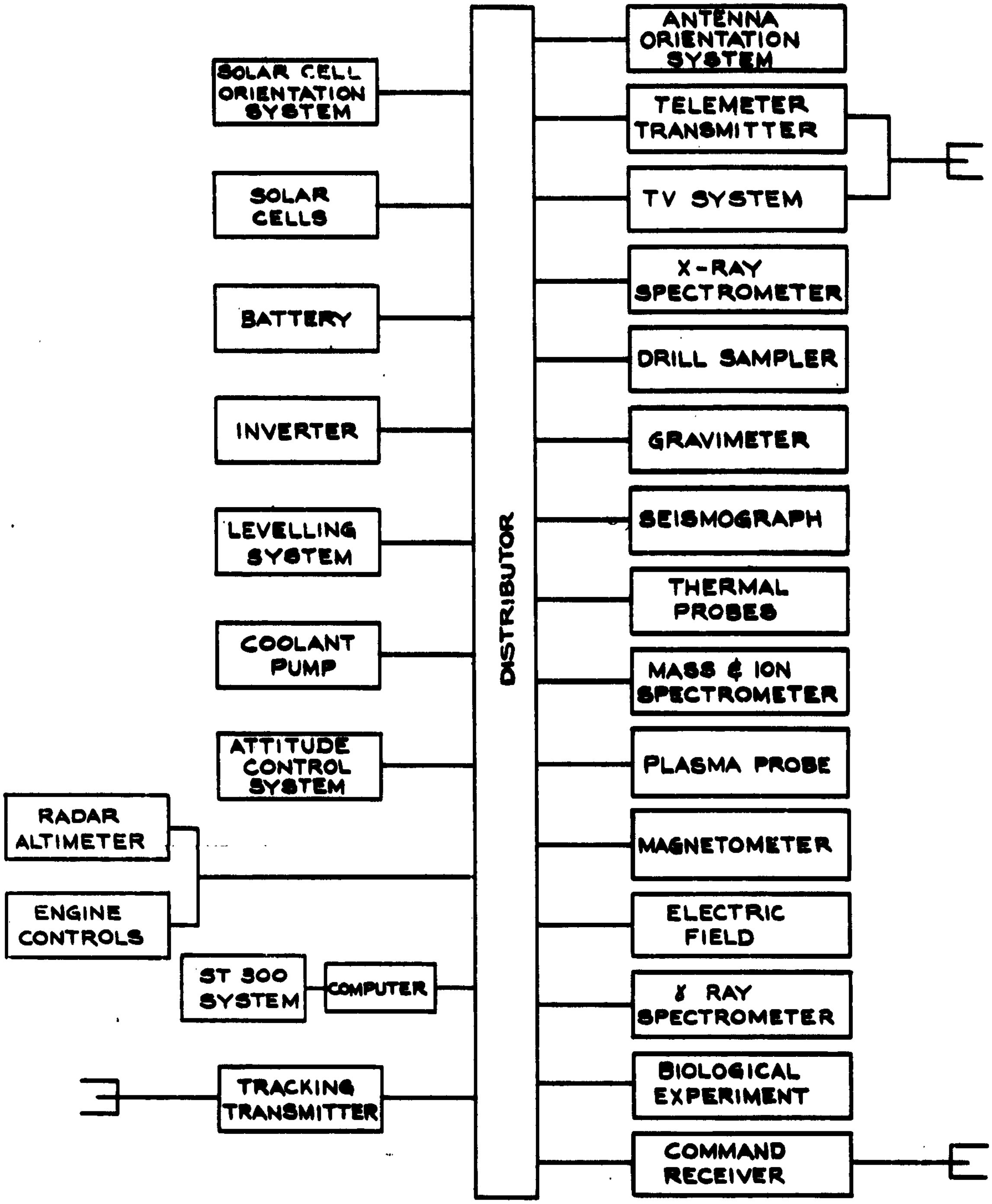
The ST-300 stabilized platform will operate for a maximum of ten hours after fourth stage separation, and for about thirty minutes before landing at an estimated power level of 150 watts.

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.

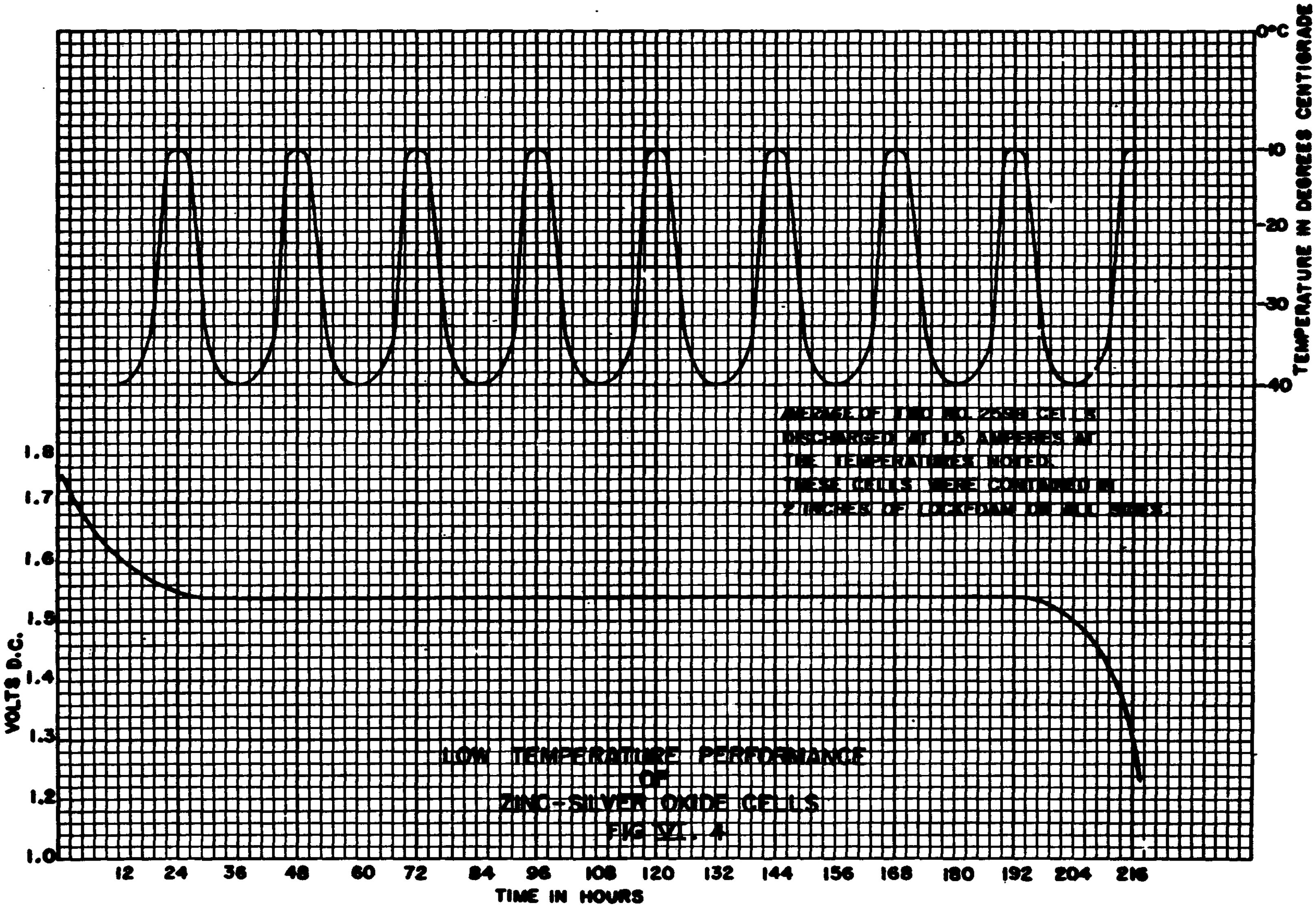
This cell is capable of delivering high peak demands of a hundred times its average rate for this requirement with only a slight decrease in its energy to weight ratio. Furthermore, it can be fully charged and discharged about five cycles and deliver 90 per cent of its initial capacity.

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

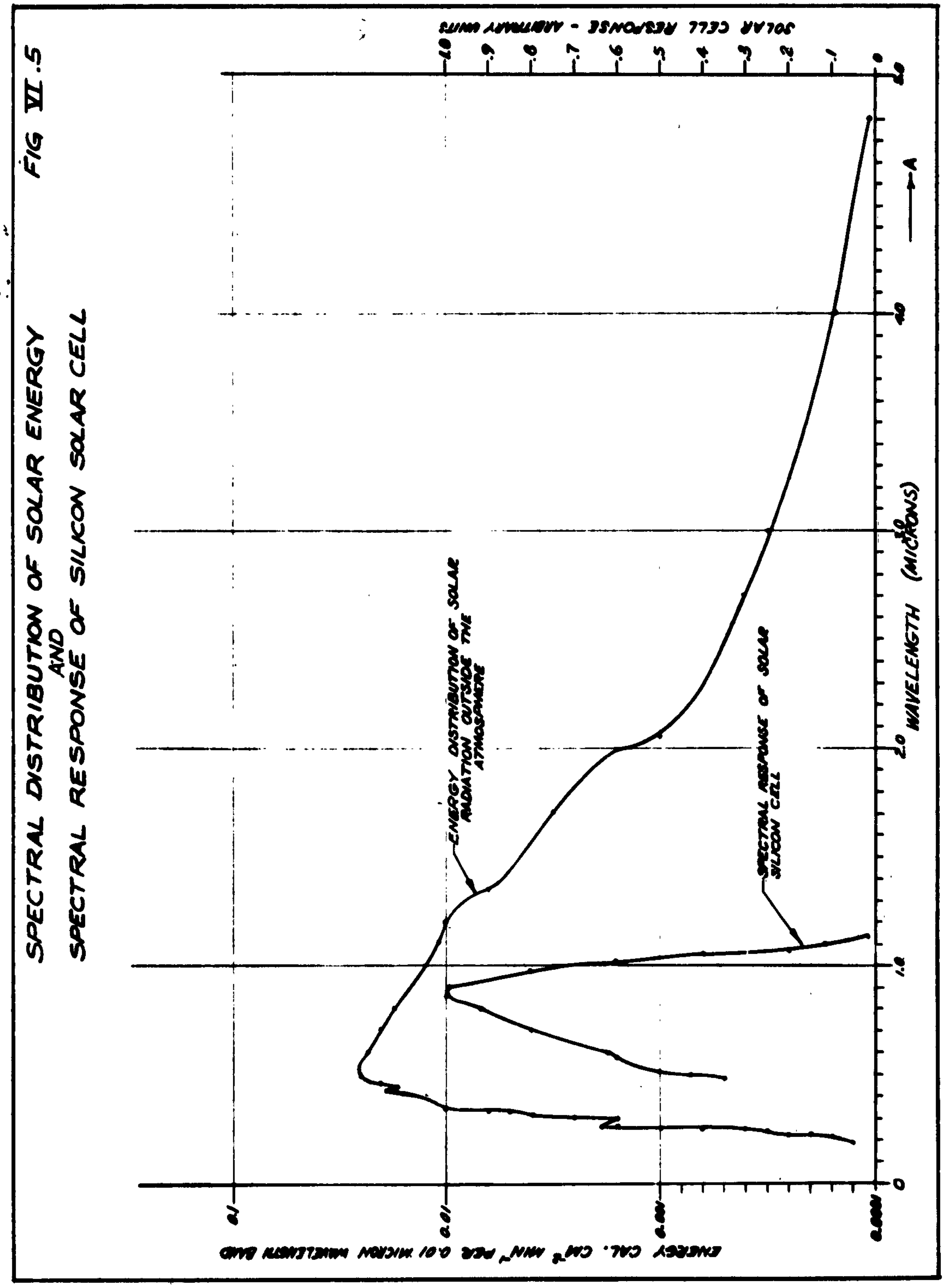
thereby improving the temperature control of the system. There remains the possibility of cell damage from meteorites so large that glass slides would not afford protection. It is proposed, therefore, that the solar cell system be overdesigned by 20 per cent to allow for such possible damage. This increases 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.



SPECTRAL DISTRIBUTION OF SOLAR ENERGY AND SPECTRAL RESPONSE OF SILICON SOLAR CELL

FIG VI.5

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.

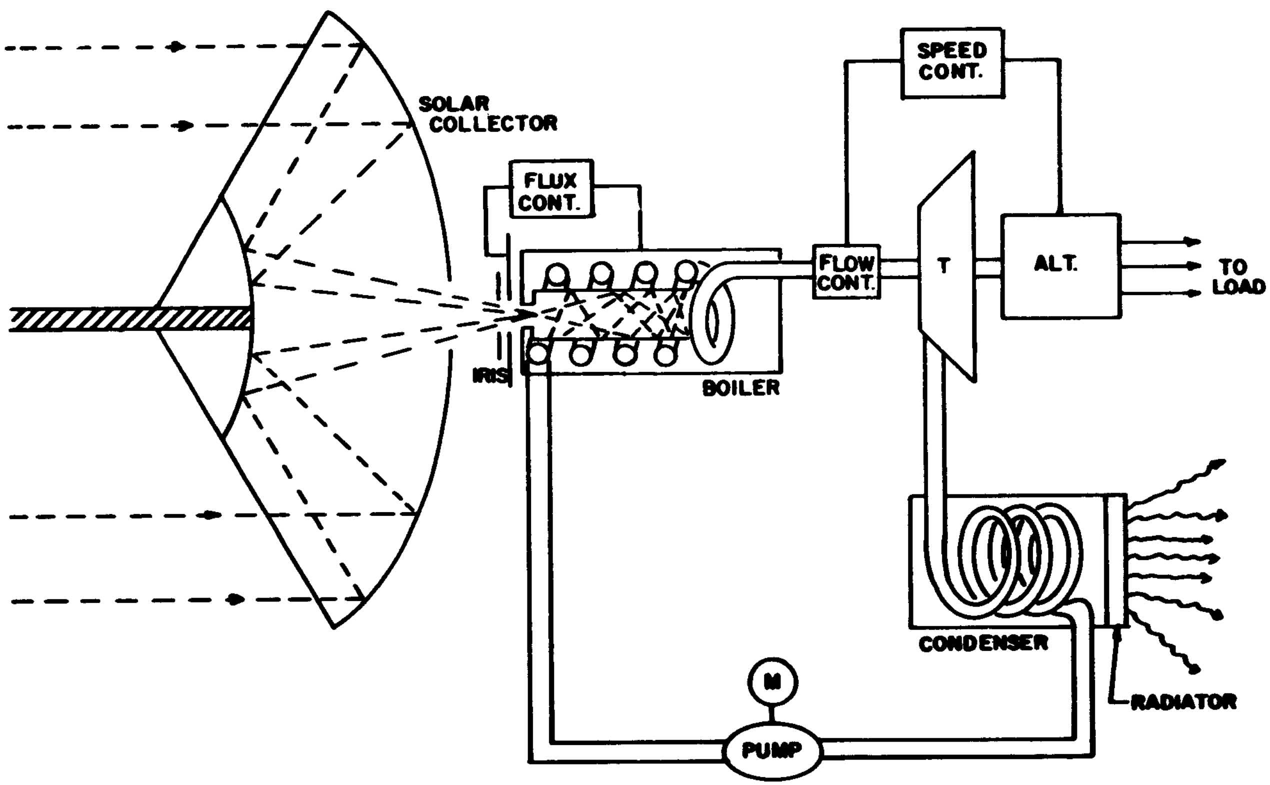


FIG. VI. 6

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.

<u>VI.4.2 Secondary Batteries</u>. 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

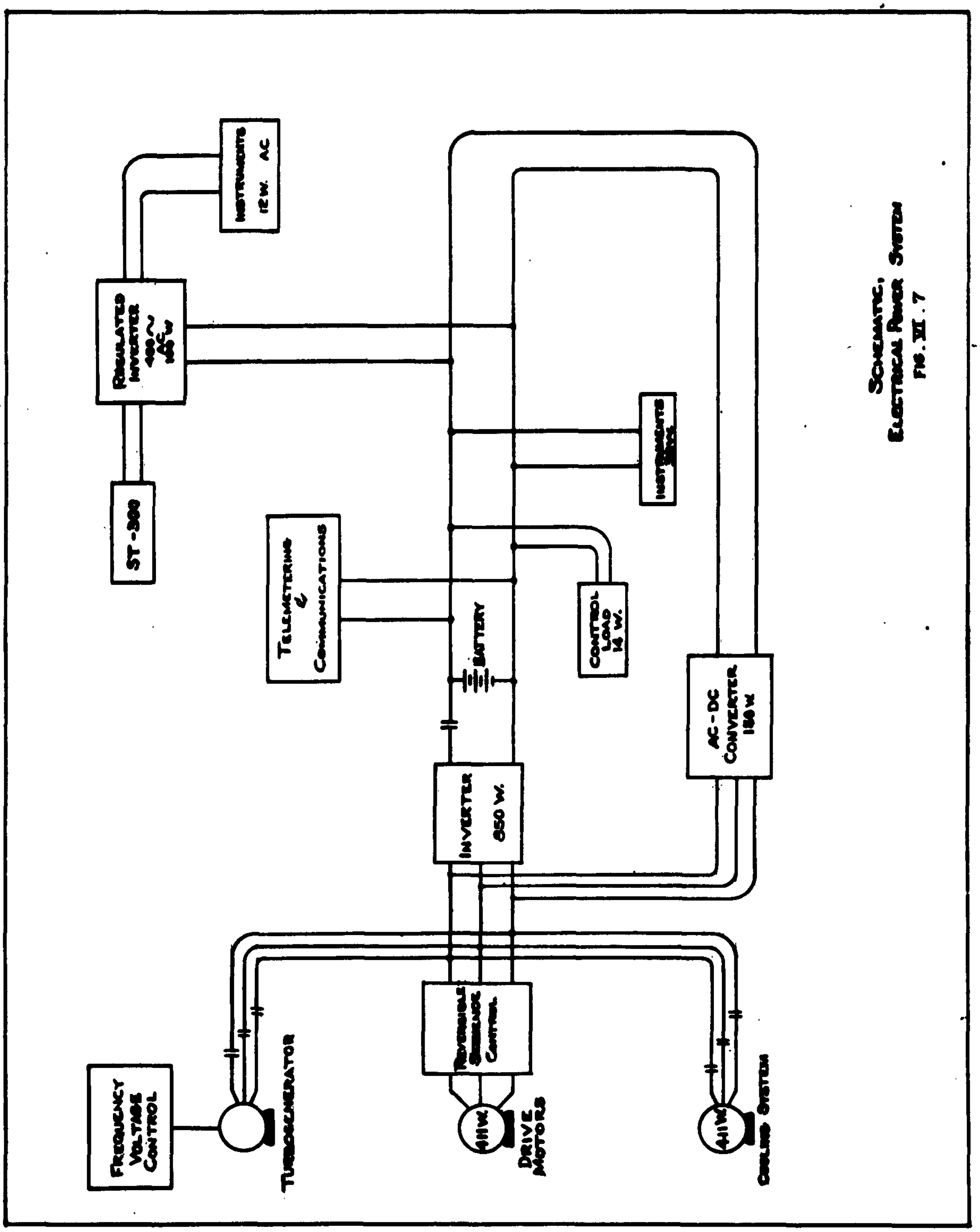
ROVING VEHICLE ELECTRICAL POWER REQUIREMENTS

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

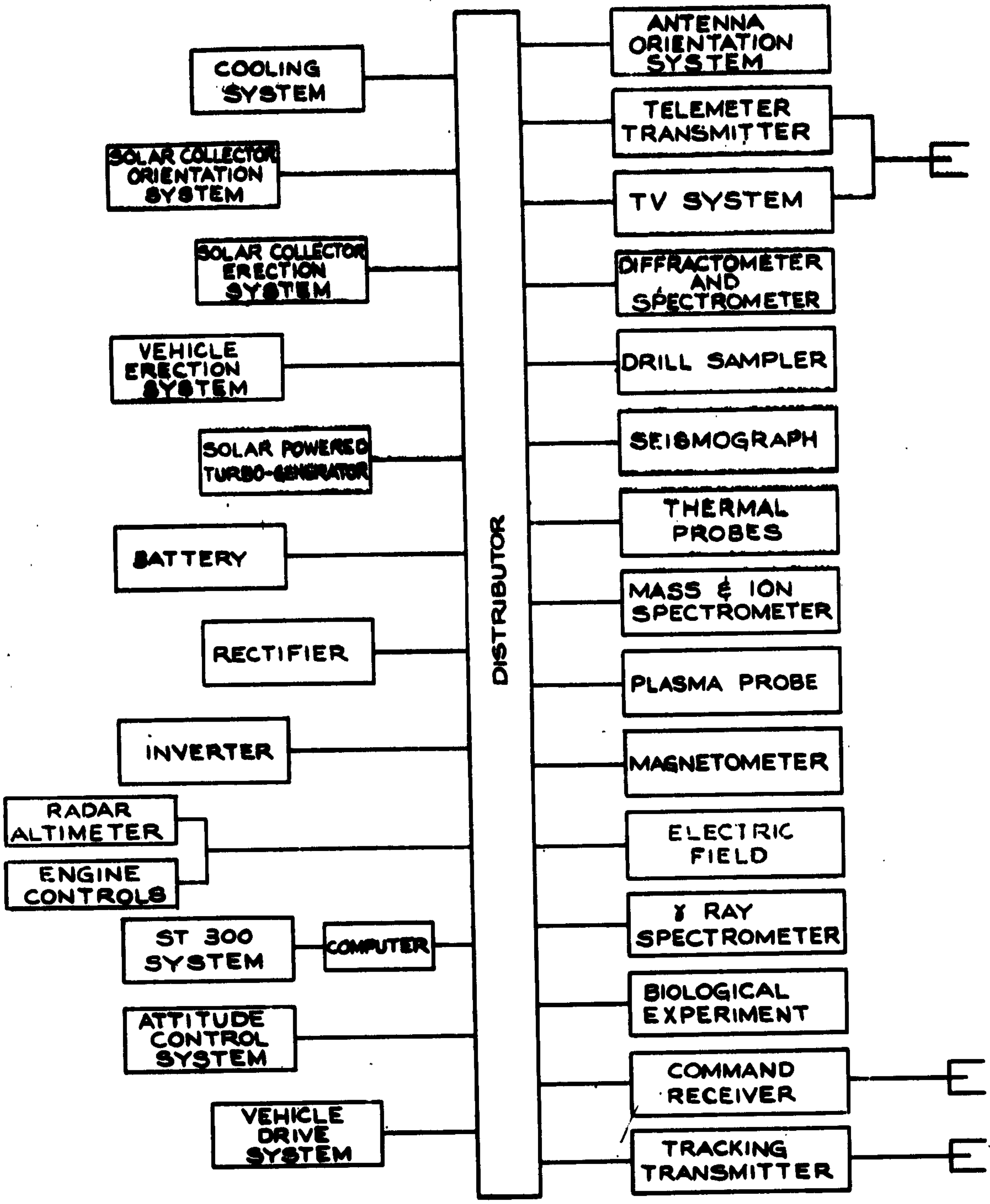


FIG. VI .8

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.

<u>VI.4.3 Closed Cycle Solar Converter</u>. 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, η_{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

$$d = 2f \tan 16'$$

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

$$c = 46{,}100 \sin^2 (\theta)$$

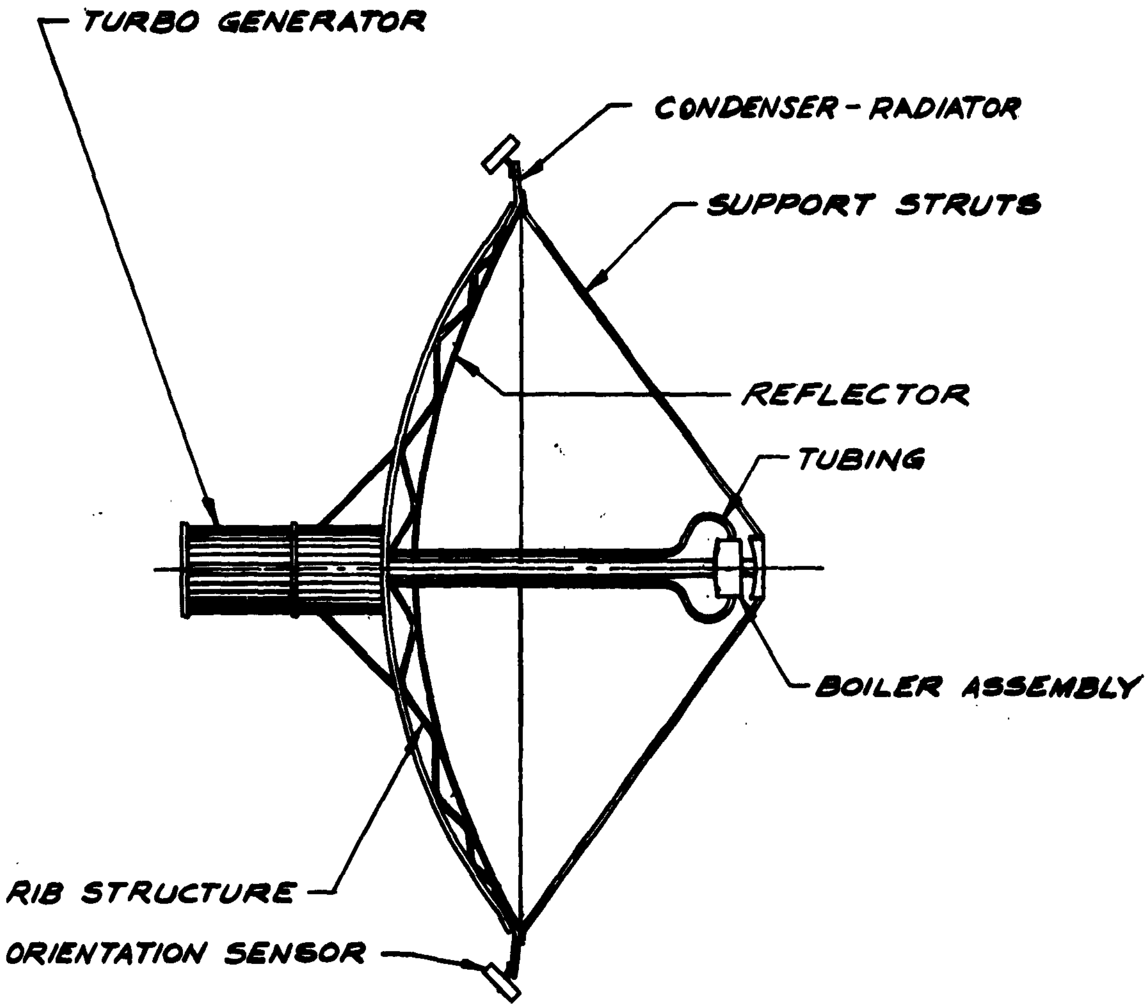
where (θ) = angular aperture or rim angle of the mirror.

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.

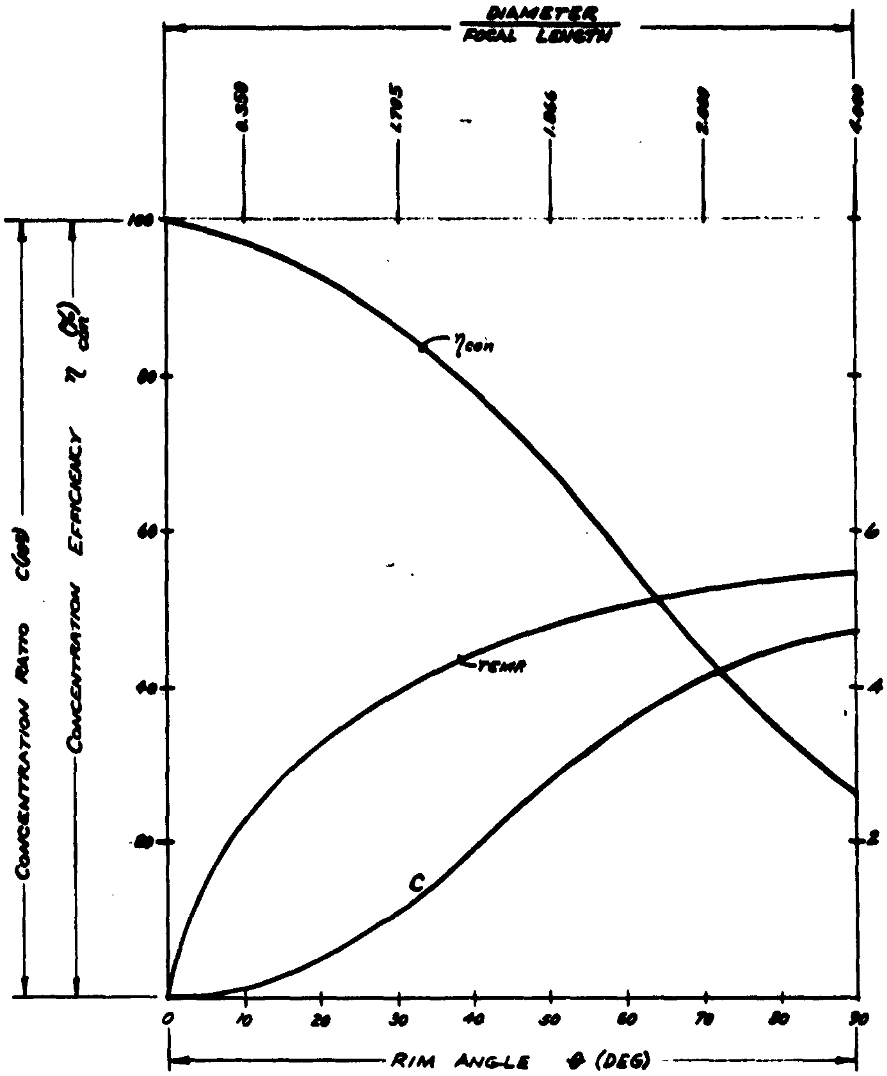
The collector will be of ribbed construction (similar to a radar antenna) covered with about 200 panels of very lightweight reflector material. The boiler will be cylindrical with one side flat to reduce the problem of re-radiation and yet to allow a reasonable target collection surface.

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

η = $\frac{P}{Q}$ = 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} $\frac{\text{watts}}{\text{cm}^2\,(°K)^4}$

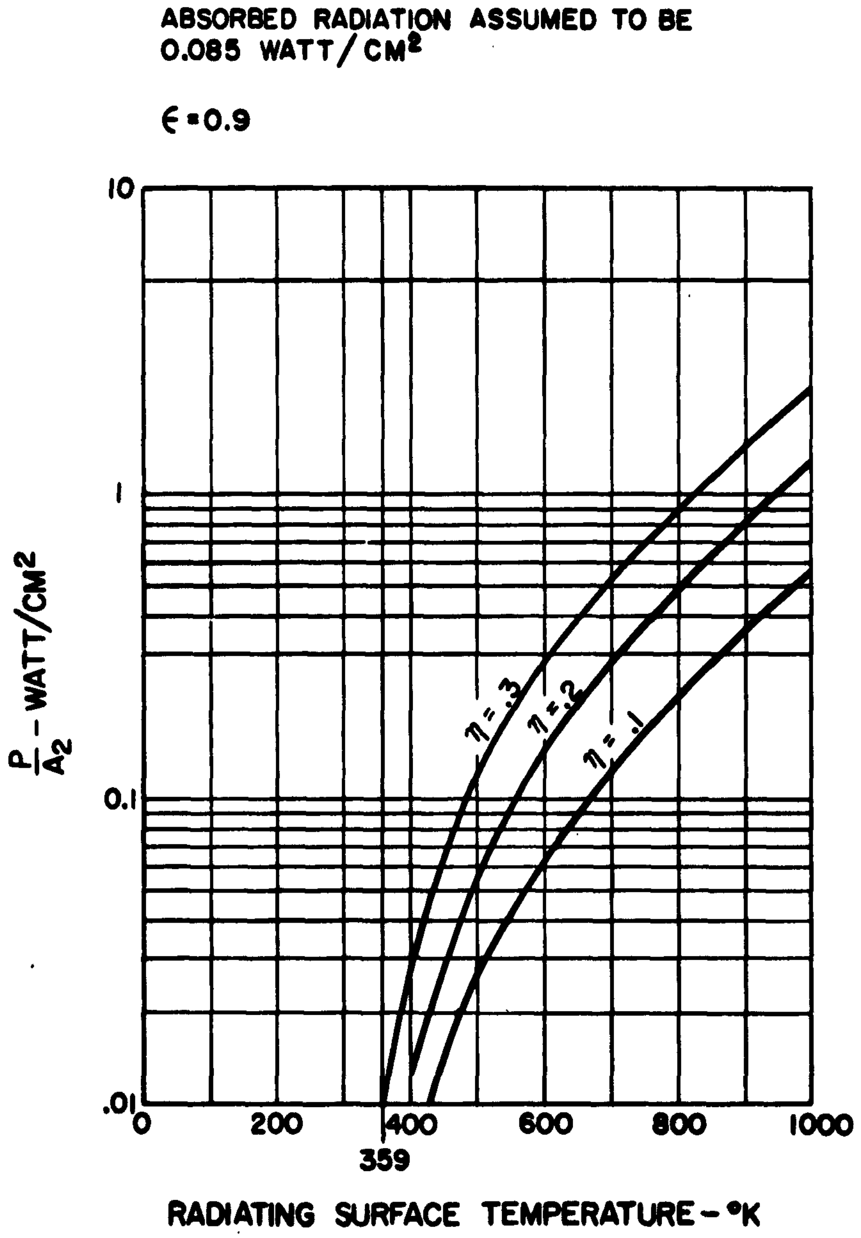
T_r = radiator temperature in degrees K

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

FIG VI .11

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.