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APOLLO

GUIDANCE AND NAVIGATION

MASSACHUSETTS INSTITUTE OF TECHNOLOGY

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APOLLO SPACECRAFT GUIDANCE SYSTEM

by
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ABSTRACT

The guidance and navigation problems inherent in the Apollo mission are discussed. The phenomena to be employed in the solution of these problems are considered. Many of the design features of the equipment which will implement the solutions of these problems are described. The system organizations and the installation configurations for this hardware in both the Apollo Command Module and the Apollo Lunar Excursion Module are presented. In the discussions, elements of the development program and design improvements of Block II over Block I hardware will be revealed.

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1. INTRODUCTION

The Primary Guidance and Navigation Problem

The goal of the Apollo Project is to place a human exploration team on the moon and return them safely to earth. The project is well outlined in reference 1. In brief, a spaceship consisting of three modules is launched on a trajectory to the moon with an enormous rocket. The Command Module is designed for atmospheric re-entry and is the home for the three man crew during most of the trip. The Service Module provides maneuver propulsion, power, expendable supplies, etc. The lunar excursion module is the vehicle which actually makes the lunar descent. It carries two of the three-man crew while the other two modules remain in lunar orbit. The Apollo Guidance and Navigation System is the on-board equipment which is used for the determination of the position and velocity of this spacecraft and for the control of its maneuvers. A set of this guidance equipment is contained in both the Command Module and the Lunar Excursion Module. Each set consists of a device for remembering spatial orientation and measuring acceleration, an optical measuring device, displays and controls, equipment to make an interface with a spacecraft control system and indicators, and a central digital data processer.

Organization of Paper

It is the objective of this paper to give a brief description of the Block II Apollo Guidance and Navigation System. Before proceeding, some of the guidelines of the development of this system will be reviewed. Also, the development program which has led to the Block II design will be described briefly. The main differences of Block II compared with Block I are summarized. Section number two reviews the guidance and navigation tasks which must be performed in carrying out an Apollo mission. This material defines the requirements on the hardware. The remaining sections each describe portions of the Block II Guidance and Navigation System.

Design Guidelines

In executing the design of the Apollo Guidance System, there have been many beliefs which one could describe as guidelines for the design. Of the many, four stand out as guidelines which have been particularly strong and consistently present.

The first guideline is that the guidance and navigation system shall be self-sufficient. The system should be designed with the capability of completing the mission with no aid from the ground. Three principles led to this concept at the beginning of the program. The first of these principles is the general appeal of the idea that the crew should be provided with all the means to be the master of its destiny. The second principle is the tendency to supply redundant means of performing each of the functions necessary for crew survival. Thus, although the on-board guidance and navigation system can perform all the guidance and navigational requirements of the mission and the various possible aborts with no aid from the ground, it is planned to have redundant navigational capability in the ground tracking networks. The third principle which makes the guideline of system self-sufficiency attractive is that the effort expended on the Apollo development will advance the state of the art in the manner that will make it of the most general usefulness for various projects and missions that may occur in the future.

System flexibility is a second guideline that was recognized early in the development. During the initial studies it was easy to determine that the Apollo guidance and navigation problem was an extremely complicated one having many phases. In fact, there are many alternative ways of performing the mission. Recognition of this complexity and variety urged great flexibility in system design so that modifications of the trajectories or plans would affect only computer programming. At the same time emphasis was given to eliminating features of the system which appeared to give it flexibility but which really gave little advantage.

The third guideline emphasizes making proper and appropriate use of both man and machine in performing the guidance function.[2,3] The design does not have excessive data displayed to the crew nor does it require the crew to perform dynamic loop closures that are either tedious or high-speed. On the other hand, the man's ability at pattern recognition is employed to establish the space reference by using stars and to make navigational sightings on landmarks. Further, the system provides a good capability for allowing the crew to freely exercise its various options and carry out human decisions efficiently.

The fourth guideline favors using advanced components and techniques in the design but is against overstepping the limits of the state of the art in the design.

Development Program

The development program which led to the Block II design began at the Laboratory in early 1961. It began as a study program with a very modest funding level which was initiated before the American Government decided to pursue the manned lunar exploration program with vigor. During this early study, as shown in figure 1, the basic concepts of the Apollo Guidance and Navigation System were defined and the initial error studies made. This study gave us a good uderstanding of the essentials of the problem as it existed during the middle of 1961. At that time a vigorous design and development program was initiated at the Laboratory.

The first step in this effort was the preparation of a development plan. The projected schedules were very tight. It was determined then that the best method of performing to the tight schedule and yet making the lunar trip with the system of mature design would be to plan on a system configuration change. Therefore, Block I was as quickly as possible defined, designed, built, and tested. This effort proceeded with the realization that hindsight would reveal desirable design improvements. It was decided not to let these improvements retard the progress of the Block I design. Instead, these changes would be saved to make a Block II improved design. During the first two years of the development program the complicated mission became much better defined. There were a number of decisions and some alterations in course. The most notable of these changes occurred in mid 1962. At that time it was determined that the most practical method of making the manned exploration was to use a smaller vehicle designed especially for the landing. This vehicle would make the descent to the moon while the larger portion of the spacecraft stayed in lunar orbit. Prior to this decision to use the Lunar Orbit Rendezvous method, it was envisioned that the whole Command Module which is designed especially for atmospheric re-entry, the service module, and a large landing propulsion module would be landed together on the moon.

The flexibility inherent in the design of the system to be installed in the command module enabled its application also in the new smaller vehicle, the Lunar Excursion Module.

The Laboratory had the first of its two prototype guidance systems operating before the middle of 1964. The second of these is shown in figure 2. This figure shows the Block I design in the test stand used to check out the system at each field site. This system consists of the same functional parts as the Block II system. Systems of this type will undergo extensive ground testing and some flight tests in Apollo vehicles.

Figure 1 shows that the study and experimentation phases for the Block II design were much shorter than for Block I. This is because the bulk of the guidance and navigation problem had been defined by the execution of the Block I design. Except for this shorter study and experimentation phase, the development cycle for Block II has phases similar to those which were experienced in Block I. At the present stage, mock-ups of the systems exist and operating hardware subsystems exist.

Figures 3 and 4 show the guidance system location in each of the spacecraft. In figure 3 the installation in the Command Module is shown. Figure 4 shows the installation in the Lunar Excursion Module. The elements of these systems are of identical design except for some features that pertain to the details of the installation.

Summary of Block II Changes.

It is intended that this paper shall make the initial presentation of the Block II Apollo Guidance and Navigation System. A discussion of the whole guidance system in a paper of reasonable length will necessarily be somewhat general. Many of the features of Block II have evolved from those of Block I.[4, 5] Of the many changes between the Block I and Block II, only six are substantial. For one, we have been able to incorporate a smaller and lighter instrument for remembering spatial orientation and measuring acceleration. Physically the computer is substantially smaller but substantially more powerful. The third change is the incorporation of the star tracker and horizon photometer in the Block II command module optics to enable the use of the illuminated earth horizon as a navigational phenomenon. The fourth major change is the use of an electronic coupling data unit to provide the interface between the geometrical reference, the computer, and the spacecraft control systems and indicators. Formerly, a less flexible mechanical unit was used. For another, a far reaching packaging change resulted from an intensive effort to moisture-proof all electronic modules and connections against the humid saline environment of the cabin. Sixth, a digital autopilot is incorporated in the Block II guidance system. This gives a better integrated and more efficient overall spacecraft steering and attitude control system in both the Command Module and the Lunar Excursion Module.

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SECTION 2

Guidance and Navigation Task of the Apollo Mission

The Apollo mission consists of a complicated series of intervals of free-fall flight alternated with intervals of powered flight. The tasks of the guidance system during these two types of flight are somewhat different. During periods of free fall, knowledge of the position and velocity of the vehicle is maintained. This activity is referred to as navigation. Figure 5 shows that for navigation, data are used from MSFN (Manned Spaceflight Network for spacecraft tracking), optical measurements between stars and the earth and the moon, and radar. These operations are generally under the control of the astronaut. These data when processed enable the display of position, velocity, and the required maneuver to reach the proposed destination. During these free-flight phases and especially when navigational sensing is in process, attitude control is frequently necessary.

Guidance occurs during powered-flight phases. Guidance is the measurement of vehicle velocity changes and the control of vehicle attitude to produce the required changes in course and speed. During guidance a spatial reference is necessary. For this purpose a gyroscopically stabilized platform (Inertial Measurement Unit) is employed. This unit also measures acceleration and delivers this information to the guidance computer for processing. Radar is still another sensor used in guidance during the lunar landing phase.

Guidance requires the initial conditions of velocity and position. These initial conditions are generated during the navigation phases. Similarly, velocity changes during thrusting phases are a necessary type of information for maintaining accurate navigation.

Figure 6 shows each of the navigation mission phases and Figure 7 shows each of the guidance phases in a normal mission. These figures will be used in explaining briefly what is to be accomplished by the guidance and navigation system during each of these phases.

Launch to Earth Orbit

During the operation of the launch booster, guidance is provided by a system in the forward section of the booster, during this powered flight, however, the command module guidance system is operating in a guidance mode. During most of the flight it is generating steering signals which can be used by the booster should there be a failure in its guidance system. The command module guidance system data are also useful in showing the crew that the flight is being guided on approximately the right course. It is the objective of this mission phase to achieve an efficient but safe earth orbit.

The Earth Orbit Navigation Phase

The spacecraft and the final stage of the booster remain in earth orbit for one or several periods. Navigational measurements are taken on-board during this orbiting primarily to verify the proper operation of equipment. The optics portion of the command module guidance system is used to better define position and velocity. The on-board data are combined with information sent up from the ground position tracking network. Finally, the gyroscopic reference in the command module guidance system is accurately realigned to the stars, by once more using the optical device.

Translunar Injection Guidance Phase

During this phase the system in the command module is active in supplying the crew with monitoring data and in providing steering signals which are available as a back-up for the booster guidance system. The measurements made of the accelerations during this phase will be used to generate initial conditions for the translunar navigation phase.

Translunar Navigation Phase

The spacecraft separates from the booster and begins a long coasting flight to the moon shortly after translunar injection. The small error in spacecraft velocity at translunar injection would result in a gross error in the arrival conditions of the spacecraft at the moon if left uncorrected. Upon injection, a series of navigation measurements using both the on-board optical system and the ground tracking system is initiated to better determine the position and velocity of the spacecraft. As these measurements proceed, the position and velocity of the spacecraft become more accurately defined. When sufficient accuracy is obtained, a mid-course maneuver is made to improve the accuracy of the course. Three such maneuvers will generally occur on the trip to the moon. Each of these maneuvers has the order of magnitude of several or ten feet per second. The first may occur as early as an hour or two after injection.

LEM Injection Guidance Phase

The first application of the Excursion Module guidance system[6] is to accurately apply a several-hundred-foot-per-second maneuver. The maneuver is intended to produce an excursion module trajectory which has a minimum altitude of ten miles in the vicinity of the designated landing site. The initial conditions for this maneuver are obtained from the navigation phase in the command module. The data are entered into the excursion module guidance system manually by the crew using the computer display and control unit.

Lunar Landing Guidance Phase

At the proper time, as determined by the Excursion Module guidance system, the landing engine is ignited to remove velocity from the LEM. Inertial measurements of acceleration and orientation are used in the early steering of the vehicle. After a fairly low velocity and altitude have been obtained, altitude and velocity data from the landing doppler radar are used in the guidance system to obtain a more accurate set of guidance conditions. Approximately at the same time the programmed destination is displayed to the crew by means of some window markings and the Excursion Module guidance computer display. The guidance objective during this phase is to efficiently use the propellant to rapidly obtain a good landing at the desired location.

Surface Navigation Phase

While the Excursion Module is on the lunar surface, its rendezvous radar is used to track the Command Module to determine more accurately the location of the Command Module orbit relative to the landing site.

Ascent Guidance Phase

The initial conditions obtained while on the surface are used to calculate the rendezvous trajectory in the Excursion Module computer. Inertial measurements are used for the control of steering and rocket cut-off during the ascent. Thrust termination initiates a coasting trajectory which is a collision course with the command module.

Mid-Course Rendezvous Navigation

The Rendezvous Radar on the Excursion Module is used to supply its guidance system with information on the relative positions of the Excursion and the Command Modules. This tracking and range data enable a successively more accurate determination of the relationship between trajectories. As this datum improves, several rendezvous mid-course corrections establish a more accurate collision course. These are applied in the same manner as translunar mid-course corrections.

Rendezvous Guidance Phase

When the vehicles are close together, radar and inertial measurements are again used to steer and control thrust termination to remove most of the hundred or so feet per second of relative velocity between the two vehicles. Final control of the rendezvous at close ranges is strictly a piloted manual function.

Return to Earth

The return trip of the Command and Service modules to the earth is similar to the trip from the earth to the moon. The homeward trip involves a transearth injection guidance phase, coasting mid-course navigation phases, and mid-course correction guidance phases. The object during the last mid-course navigation phase is to determine accurately the initial conditions for re-entry guidance.

Re-Entry Guidance Phase

In appearance the Command Module is a body of symmetry. However, the mass distribution within it is concentrated on one side so that the re-entry body trims asymmetrically, thus providing a modest lift-to-drag ratio of three tenths. During atmospheric entry the guidance system makes inertial measurements of acceleration and orientation and provides control of the re-entry body in roll. The first objective in the exercise of roll control[7] is achieving the correct solution to the altitude problem. Thus, by adjusting the amount of lift in the vertical plane, the desired re-entry range may be established. Trajectories having ranges of several thousand miles may be controlled in this manner. During the later phases of the re-entry this roll control is exercised to adjust the transverse aim and to establish the fine aim in range to bring the Command Module accurately to the designated landing site.

Mission Abort Paths

This section so far has summarized very briefly the main features of a normal lunar exploration mission. The total of such features presents a complex guidance problem having many phases. It does not, however, give the full picture of the flexibility and capability which the guidance system must have. Figure 8 suggests the variety of conditions which may be encountered in various kinds of abort circumstances. This abort logic will not be belabored in this paper other than the suggestion of its complexity given by the rather overwhelming set of options shown in Figure 8.

Section 3.

Inertial Measurement Unit And Its Uses

Function

The function of the Inertial Measurement Unit in the Apollo Guidance and Navigation System is 1) to provide an accurate memory of spatial orientation and 2) to provide an accurate measurement of spacecraft acceleration. Its functions are the same in both the Command Module and the Lunar Excursion Module. A three-gimbal gyroscopically stabilized platform was chosen to implement these functions.

Figure 9 shows a schematic of the chosen arrangement. In the middle of the figure is a stable platform which has mounted on it three gyroscopes represented by the larger size cylinders and three accelerometers represented by the smaller size cylinders. This stable platform is supported by a gimbal system which provides three degrees of freedom between the structure of the spacecraft and the inner member. In both vehicles the outer gimbal axis is aligned approximately along the thrust direction of the vehicle.

Prior to any maneuver the inertial space alignment of the inner axis is chosen. The selected direction is always chosen so that the inner gimbal axis lies approximately normal to a plane that more or less contains the planned maneuver. Since the thrust axis of the vehicle remains approximately in this plane, such a selection of inner axis orientation results in the direction of the outer and inner gimbals being widely separated. It can be seen that if these two axes were parallel or nearly parallel the loss of one degree of gimbal freedom would occur. Thus the inner member could no longer be stabilized. However, in any reasonable maneuver this geometry can always be easily avoided by the proper selection of inner axis orientation as described. Alignment along a preferential axis for each specific mission phase does not represent any penalty. As this portion of the system is unpowered during coasting phases of the mission to conserve electrical energy, it must be realigned prior to use during an accelerated maneuver in any case.

For the stabilization of the inner member, motors are provided on each of the three axes. They provide the necessary torques between pairs of gimbals to accomplish the stabilization. The gyroscopes are only called upon to provide signals. One gyroscope is sensitive to rotations which occur about the inner gimbal axis only. A resolver is used on the inner gimbal axis to operate on the signals of the remaining two gyros to develop a signal to use for the middle-gimbal and outer-axis stabilization motors. With this arrangement the gyro signals are held to very small values. Thus the inner member provides an accurate memory of inertial space orientation since the gyroscopes have very low unbalance drift. Excellent servo performance is maintained even while the inner and outer axes are as close together as 15 or 20 degrees. Thus all attitude maneuver capability nearly exists without the added weight, complexity, and dynamic problems associated with a fourth degree of gimbal freedom.

The stable platform is used to provide an inertially fixed coordinate system for the measurement of accelerations. This is accomplished by mounting the three accelerometers on the platform. Each accelerometer is sensitive to acceleration along one direction. Thus the accelerometer set delivers signals to the computer giving the three components of acceleration resolved along the stabilized platform axes

For the accelerometer coordinate system to have meaning, a method must be provided to accurately establish the orientation of the stabilized platform. The orientation method consists of sensing, transmitting, and acting.

Sensing is accomplished by the use of Command Module and Excursion Module optical devices for determining orientation from the stars. This will be described in the next sections. The optics and the inertial measurement unit are mounted on a common rigid structure so that orientation information obtained from the optics can be transmitted to the inertial unit.

The transmission of this information requires accurately machined gimbals and a precision resolver between each pair of gimbals. Each precision resolver has two windings. A coarse winding generates sine and cosine functions of the respective gimbal angle. The fine windings generate these functions for an angle which equals 16 times the respective gimbal angle. Thus these resolvers provide the means for measuring the geometry within the gimbals. Processing of their data by the computer enables the determination of inner member orientation. Thus transmission of orientation is accomplished.

Action requires accurate and sensitive changes in inner member orientation. To accomplish this each gyroscope has magnetic components for producing a precision torque. These components are excited with pulses directly under the control of the computer so that each pulse produces an increment of angle change.

An operation mode is present so that the converse transmission of geometry is possible. Thus the indication of spacecraft orientation relative to the inner member is provided. This information is used to generate spacecraft steering signals in the guidance computer.

Gyroscope

The three gyroscopes used on the Apollo Block II Guidance and Navigation System are designated 25 Inertial Reference Integrating Gyro Apollo Block II. This gyroscope is a relatively minor modification of Apollo Block I gyro. The designation "25" is associated with a family of inertial components which has a body diameter of 2.5 inches. A cutaway of this gyroscope is shown in figure 10. The drawing shows a spherical inner member which contains a gyro rotor supported on ball bearings. This rotor is driven at 24,000 rpm by 800-cycle-per-second excitation of its hysteresis motor. The motor has an angular momentum of roughly one-half-million dyne-centimeter-seconds.

To minimize disturbing torques the mass of this inner member is buoyantly supported by a dense viscous fluid in the thin gap between the surface of the spherical float and the gyro housing. In addition to the support provided by buoyancy, either of two features provide supplementary suspension.

When operating, a magnetic suspension is used to supply this supplementary support. At each end of the spherical float there is a magnetic device called a ducosyn. The ducosyn has eight magnetic poles and numerous coils. By providing excitation on the proper combination of coils, each ducosyn has the means of translating an end of the gyro float in two transverse degrees of freedom. Furthermore, the ducosyn pole faces are slightly tapered which gives a capability for applying end to end forces on the float. Thus the pair of ducosyns provide small magnetic forces to supplement the buoyancy of the float in the dense fluid. Thus the float is suspended in five degrees of freedom.

The remaining degree of freedom allows the float to rotate when the gyroscope is rotated about a direction transverse to both the rotor and the pivots. Other combinations of coils in the ducosyn sense the small angular excursions. The resulting signals are used in the stabilization servos. Torque to process the gyro for fine alignment of a stable member is also provided by this arrangement of coils.

When not operating, the supplementary support is provided by pivots. The sole purpose of the pivots shown in the cutaway is to keep the float approximately centered during periods of storage.

Ducosyn improvements and the arrangements in the expansion bellows are the primary changes in the Block II over the Block I gyro.

Accelerometer

Three size-16 Pulsed Integrating Pendulum units are mounted on the stable member. Figure 11 shows one of these sensors. Again "16" stands for an inertial component family having a 1.6-inch case diameter.

The suspension scheme and internal arrangements of the Pulse Integrating Pendulum unit are generally similar to those of the gyroscope which have just been described. The internal float arrangement of the Pulse Integrating Pendulum is, of course, simpler than that of the gyroscope. An off-center mass distribution is substituted for the rotor assembly. In the accelerometer small angular displacements cause the ducosyn signals used to develop error pulses. These error pulses are applied to the torquing ducosyn as precision pulses of current to restore the Pulse Integrating Pendulum assembly to its reference position. The computer uses these pulses as data representing increments of the velocity change along the respective accelerometer's sensitive direction.

Inertial Unit Assembly

Figure 12 is a photograph of the Inertial Measurement Unit for system 600F, the first Block II prototype system. This Inertial Measurement Unit is designated as "12.5", which is the number of inches corresponding to its spherical diameter. In Block I the corresponding dimension is 14 inches. The Block II design has a correspondingly lighter weight. The change in size and weight resulted largely from improvement in the available state of the art in electromagnetic components, especially the precision resolvers.

In the photograph, the inner member is exposed by the removal of two gimbal hats. The outer of these has been removed from the gimbal case which contains hydroformed cooling passages. Spacecraft Cooling fluid is circulated through these passages.

On the inner gimbal the large shiny disc on the axis houses a direct-drive torque motor, resolvers, and slip rings. The use of direct-drive torque motors enables the achievement of especially excellent servo performance. In this arrangement a permanent magnet stator is mounted in one gimbal. The armature is mounted on the other gimbal without gearing. The armature is excited with the direct current and has an air gap 3.8 inches in diameter. Each axis has an unlimited degree of freedom. The absence of gearing enables the inertias of the gimbals to aid in their own stabilization.

At each end of the middle axis is an assembly containing 40 slip rings each. Visible in the photograph are the ends of the gyro and accelerometer with associated electronics attached. Some of the remaining electronics on the inner member is associated with its temperature control heaters and sensors. The temperature of the inertial components is maintained with a much greater accuracy than one degree fahrenheit.

SECTION 4.

The Command Module Optics and Its Use

Function

Unlike other portions of the guidance and navigation system, the design of the Command Module optics is not employed in the system aboard the Excursion Module. The Command Module optics has three functions. The function which determines its characteristics the most is that of navigation. Another function for which it also is well qualified is the measurement of the inertial platform orientation. The Command Module optics are also of use for general viewing since it consists of a pair of articulating telescopes, one of high power and the other of low power.

The principle of navigation which the Command Module optics is designed to implement is illustrated in figure 13. This figure shows a spacecraft on a lunar trip and the directions to some stars. From the spacecraft, an earth landmark can be seen. The cone in the figure having an axis parallel to the direction of the star Fomalhult is a locus of points from which a certain angle can be measured between the direction of the star and the landmark. The precise measurement of such an angle establishes a cone of position in space. A second cone, very flat in the figure, with its axis parallel to the direction of Deneb, intersects the first cone in a line of position. To obtain a measurement which determines the spacecraft position along this line, a third object must be used. This object could be a second landmark or it could be the moon as shown in the figure. In the figure the angle between the moon horizon and the star Antares is employed to obtain the location along the line. Thus, the principle of navigation to be implemented by the Command Module optics is illustrated.

The illustration above assumes the simultaneous measurement of three angles between the stars and two landmarks. Simpler hardware results if time-sequencing single measurements of such angles is done. It is this type of measurement which the Command

Module optics is designed to implement. The theory used in the reduction of such data is given in references 8 and 9.

Mechanization

The Command Module optics consists of two telescopes as shown in figure 14. The telescope on the left is the space sextant and the one on the right is the scanning telescope. The sextant is a powerful instrument having a magnification of 28. The scanning telescope has unity power. Both telescopes are fitted with eye pieces which provide eye relief in excess of two inches so that they can be used with space helmets on if necessary. Both head assemblies containing the deflecting optics are each driven by motors and gearing about parallel axes called the "shaft axis."

The scanning telescope has only one line of sight which is deflected normal to the shaft axis by a dove prism which is driven by a second motor and gearing.

The sextant's 1.6-inch objective has in front of it a partially reflecting beam-splitter mirror. Thus the view through the beam splitter is fixed in the spacecraft. The beam splitter enables the deflection of the second telescope line of sight by means of other mirrors including one which is precisely driven. This articulated line of sight is used for seeing stars and thus has been provided the greater efficiency. The body-fixed line of sight used for observing landmarks is given poor efficiency to make up for the difference of brightness of the targets.

The sextant is used during mid-course navigation in a very similar manner to a marine sextant of great power. To make a mid-course navigation sight, the crew first uses the scanning telescope as a finder. The spacecraft is maneuvered to guide the desired earth or lunar landmark onto the shaft axis so that it can be seen with the narrow field of view of the sextant. By applying very small control impulses to the spacecraft it is made to come almost to rest with the landmark in the middle of the sextant field of view. Now a search is made for a suitable star with the scanning telescope. Once acquired with the scanning telescope, the star can be observed through the sextant. The crew now obtains by means of the two degrees of freedom of the star line of sight the juxtaposition of the star on the landmark as shown in figure 16. It should also be noted that this juxtaposition need not occur in the center of the field to yield an accurate measurement.

When the measurement geometry is correctly obtained, the observer depresses a button which enables the computer to mark the time and the angle of the accurate trunnion drive of the sextant. This angle is measured by means of a single- and 64-speed precision resolver mounted directly on the trunnion of the drive axis. The accuracy of such measurements has an order of magnitude of ten seconds of arc. Thus, sensitivity to changes of position when midway between the earth and the moon is several miles.

The earth's horizon is also a promising phenomenon for navigation. Its use is illustrated in figure 17. The figure shows an illuminated horizon. Near sea level the atmospheric scattering is so extreme that the appearance of the horizon is cloud white. However, if the horizon is observed through a column of air passing roughly 100,000 feet above sea level, it has the same scattering power as the zenith sky. This results from the column having the same total mass of air as a vertical column. As the altitude of observation of the column changes by 2,500 feet the scattering power should change by 10 percent. Experiments currently under way show promise of good accuracy.

The star tracker and horizon photometer shown on the sextant head in figure 14 are for making such measurements. The body-fixed horizon photometer line is scanned across the horizon by means of slow rotation of the vehicle. The photometer circuitry provides the computer with a mark at the moment the column being scanned has a certain ratio of brightness to the peak brightness observed near sea level. Since a star is being tracked automatically while the scan occurs, the mark enables the reading of the precision angle.

Figure 18 shows a Block II sextant and scanning telescope. The predominant difference between this and the Block I instrument is the inclusion of the star tracker/photometer. The frame of this addition is shown on the left hand side of the sextant head.

Other Uses

The Command Module optics is well qualified to make navigational measurements in earth or lunar orbit. To do this the Inertial Measurement Unit is referenced to the stars. The scanning telescope is used to observe the bearing of a mapped landmark. By means of the resolvers on the Inertial Unit gimbals and on the scanning telescope drives, these bearings are referenced to the stabilized platform. Several such bearings give sufficient information to derive the characteristics of the spacecraft's orbit.

The inertial platform is aligned to the stars with the Command Module optics. This is accomplished by first approximately aligning the stable platform. Then the direction of a suitable star is tracked with the sextant and marked. The direction of a second star is marked The computer processes these data to obtain the directions of the two stars in the inertial reference coordinate system. This defines the orientation of the stable platform relative to the stars. Using the capability to process the gyros by torques, the stable platform can now be aligned more accurately to some desired orientation.

SECTION 5.

The Lunar Excursion Module Optics and Its Use

The Optics on the Lunar Excursion Module is very much simpler than that on the Command Module. It consists of a single unit called the optical alignment telescope. The single unit has no mechanical drives but does have two manual degrees of freedom. The sole purpose of this telescope is to provide stable-member orientation information to the computer. It does this by means of the crew tracking two stars. The same principles that were discussed for command-module Inertial Measurement Unit alignment apply.

Figure 19 shows a wooden mockup of the Excursion Module Optics. This photograph has been marked up to show the outlines of the LEM vehicle. Also shown is the line of sight of the telescope. As the markings show, this instrument has a single line of sight. This line of sight makes a 45-degree angle with the instrument's tube which is mounted vertically in the Excursion Module. The field of view is 60 degrees. The field of view is aimed upward so that the sunlit moon will not be present to obscure the visibility of the stars being used for prelaunch alignment. The line of sight has several positions. The position is selected by the upper knob on the side of the instrument. The forward position is shown in the drawing. Alternative operating positions are 60 degrees to the left and 60 degrees to the right. These positions enable the crew to see roughly one-half of the overhead sky. In addition to these three operating positions there is a storage position to prevent damage to the Optics from lunar dust during the landing.

In order to make an observation on a star, a suitable detent position is chosen so the star may be seen. The lower knob on the mockup rotates the position of the reticle in the instrument. The reticle has a straight marking and a spiral marking as shown in Figure 20. By using the knob to rotate the reticle the direction of the radius from the center of the field of view to the star is measured. When the alignment is correctly made the observer marks with the buttons. The reticle angle is given to the computer manually by means of a counter and data entry into the computer keyboard. A second measurement must now be made to the same star. To do this the spiral is rotated to overlay the star. Note how the direction of the straight line and the direction of the spiral in the neighborhood of the star are nearly at right angles. Thus the two observations correspond to measuring both degrees of freedom of the bearing of the star.

Similar measurements on a second star are made. The computer has now obtained data sufficient to determine the orientation of the stable platform.

TABLE I

APOLLO GUIDANCE COMPUTER CHARACTERISTICS - BLOCK II

WORD LENGTH	15 Data Bits plus one Parity Bit
NUMBER SYSTEM	One's Complement
MEMORY CYCLE TIME	11.7 sec
FIXED MEMORY REGISTERS	36,864 Words
ERASABLE MEMORY REGISTERS	2,048 Words
NUMBER OF NORMAL INSTRUCTIONS	34
NUMBER OF INVOLUNTARY INSTRUCTIONS (Interrupt, Increment, etc.)	10
INTERRUPT OPTIONS	10
ADDITION TIME	23.4 sec
MULTIPLICATION TIME	46.8 sec
DOUBLE PRECISION ADDITION TIME	35.1 sec
DOUBLE PRECISION MULTIPLICATION SUBROUTINE TIME	575 sec
INCREMENT TIME	11.7 sec
NUMBER OF COUNTERS	29
POWER CONSUMPTION	100 Watts (AGC + DSKY's)
WEIGHT	58 pounds (computer only)
SIZE	1.0 Cubic Foot (computer only)

TABLE II

APOLLO GUIDANCE COMPUTER INTERFACES-BLOCK II

NUMBER OF INPUT DISCRETES	73
NUMBER OF INPUT PULSES (Serial and Incremental)	33
NUMBER OF D.C. OUTPUT DISCRETES	68
NUMBER OF VARIABLE PULSED OUTPUTS (Serial, Incremental, and Discrete)	43
NUMBER OF FIXED PULSED OUTPUTS	10
NUMBER OF CONNECTOR WIRES	365

SECTION 6.

Computer

Since another paper in this symposium[10] treats the organization of computation and control in the Apollo Guidance Computer, this section will emphasize the physical nature of the Block II computer rather than the organizational aspect of it. Figure 21 shows a simplified block diagram that is applicable to both the Block I[11, 12] and Block II computers. In this section it will be observed that there is a very extensive fixed memory consisting of core ropes. There is a very limited erasable memory, primarily for data. This consists of a ferrite coincident-current memory plane. There is extensive logic for the performance of arithmetic, sequence generation, computer control, etc.

Tables I and II summarize the Apollo Block II computer characteristics and interfaces. Block II contains twice the erasable memory of Block I. There is an increase of 50 percent in the fixed memory. There are considerable speed increases resulting from extra operation instructions. The interface is considerably expanded to handle additional functions in the control of spacecraft.

In spite of increased capacity and power, the Block II computer is smaller physically. The two reasons for this are shown in figures 22 and 23. Figure 22 is an unpotted computer logic module. The computer contains 24 of these modules. They differ only in the wiring pattern on the matrix located in the middle of the module. On the surface of this module can be seen the active components. Each of the small rectangular elements contains two micrologic nor gates in a flat package. In Block I single nor gates were contained in small To-47 transistors cans.[13] The Block II high-density packaging results in a greatly reduced thickness for each logic module. Consequently, a reduction in the overall size of the computer is achieved.

Smaller physical size also results from a higher density in the fixed memory. In Block II, 24 sixteen-bit words are stored per core. An unpotted Block II module is shown in figure 23. In a Block

I rope module of of similar volume, there were 16 such words per core. The core rope is an extremely dense technique for storing information. It also possesses considerable reliability advantage since physical destruction of the core or wire is necessary for loss of information

The mockup of the Block II computer is shown in figure 24. Except for the core rope memory module plugs, the computer is physically sealed against moisture. The internal layout of the computer is shown in figure 25. Basically, it consists of two planes of plug-in modules.

SECTION 7.

Power Servo Assembly Characteristics

The power servo assembly is a collection of electronics in a common package which supports the operation of the Inertial Measurement Unit, the Optics Unit, and other parts of the system with power supplies, servo amplifier, etc. These miscellaneous circuits are packaged in modules of various sizes.

Figure 26 shows several typical Power Servo Assembly modules before potting. The frames of these modules are made of magnesium for light weight and good thermal distribution. These modules plug into a single header which makes the necessary inner modular and system connections. The construction of the Power Servo Assembly is shown in the cutaway of figure 27. It can be seen that the modules of various sizes screw upward into this header in the Command Module. Each one presents the header with a flat area to establish good thermal contact. The header is in turn screwed to the coldplate which is a part of the spacecraft. This coldplate has spacecraft coolant fluid circulated within it, although this feature is not shown in the cutaway.

After the modules are plugged into the header, a cover is placed over the frame to complete the seal of the power servo assembly's modules and plugs. This assembly is installed as a unit onto the spacecraft coldplate. In all there are 44 modules that comprise a set for the Command Module Power Servo Assembly.

The wooden mockup of the Power Servo Assembly for the Command Module is shown in figure 28. The screw holes are for fastening the power servo assembly to its coldplate in the spacecraft. The raised rectangles represent plugs which the Power Servo Assembly presents to the Guidance and Navigation system harness. The cylinders in the foreground represent test connectors which are accessible after spacecraft installation.

SECTION 8.

Control Interfaces

The control interfaces between the guidance and navigation system and the spacecraft consist of signals to and from the computer and of signals to the spacecraft from the Coupling Data Units. In addition, the Coupling Data Units provide interfaces within the Guidance and Navigation system. Before discussing the overall control configuration, the Coupling Data Unit will be briefly examined.

Figure 29 shows a very simplified schematic of an electronic Coupling Data Unit. The first function shown in the schematic is that of converting the mechanical angle data into digital quantities which the computer can process. The schematic shows that each of these mechanical angles is converted to sine and cosine signals by resolvers. In the Coupling Data Unit it is desired to have a solid-state digital counter contain a word which represents the value of the mechanical angle. The contents of this counter represent the indicated value of the angle. The most significant four binary digits of this word are used for initial processing of the resolver signals. These four bits θ_1, represent the large part of theta. The least significant bit corresponds to 22-1/2 degrees. These bits are used in the sine and cosine function multipliers. These multipliers consist of solid state switches controlled by the bits through logic to switch appropriate combinations of the sine and cosine of 22-1/2 degrees, 45 degrees, 67-1/2 degrees......., to form a sum of the resolver signal times the sine and cosine, respectively, of the large part of theta.

The less significant figures in the counter indicating theta are summed in a similar linear function multiplier operating on the excitation signal. Examination of the trigonometric forms indicated shows that, to a good approximation, the sum as indicated in the drawing of these signals should be zero or of very small value when the counter correctly indicates the value of theta. When the value of theta is not correctly indicated, the error detector generates pulses which are delivered to the computer to give it knowledge of changes in the value of theta indicated. These pulses are, of course, also used to add or subtract increments from the counter.

The discussion above is extremely simplified and illustrates the principle used in analog-to-digital conversion in the electronic CDU's. In actual fact, each mechanical angle to be encoded has a coarse and a fine resolver. The operations described above are performed on both resolver signals with a nonlinear circuit to select between the coarse and fine resolvers, depending on the size of the error signals.

A second function of the electronic CDU is to generate an analog signal from digital increments delivered for steering by the computer. These increments go into a counter which controls a similar linear function multiplier. An analog 800-cps signal proportional to the argument in the counter is generated. In some cases this is an engine gimbal command. In other cases this represents attitude errors, etc.

A complete discussion of the CDU is complicated because many of the operations are on multiple-speed signals. It is further complicated because the CDU's have within them logic to select a number of modes which are used for various purposes. Examples of modes are Inertial Measurement Unit Coarse Alignment, Zero Optics, Steering, etc.

The wooden mockup shown in Figure 30 illustrates the physical characteristics of the electronic Coupling Data Unit package. In Block II this sealed container has five CDU's plus associated logic and power supplies as shown in the cutaway in figure 31. The packaging is very similar to that found in the computer. There are two layers of modules plugging into a wire-wrapped header, all of which is contained within a seal.

Figure 32 is an overall block diagram of the guidance system operating as a digital autopilot in steering the spacecraft. It can be seen that during powered flight two of the analog signals from the Coupling Data Units go directly to the gimbal engines[14] on the spacecraft. Similarly, during free-fall flight, signals go directly from the computer to the solenoid-actuator amplifiers[14] in the Reaction Control System. Flexibility is the advantage of the electronic Coupling Data Units in Block II over the mechanical units used in Block I. The arrangement gives the computer knowledge of Inertial Measurement Unit gimbal angles while allowing it to generate independent steering commands. These can then bear fairly sophisticated computational relationships to each other.

SECTION 9.

Installation and Displays and Controls

The appearance of the Apollo Guidance and Navigation System installed in the Lower Equipment Bay of the Command Module is shown in figure 33.

The displays are somewhat abbreviated in Block II compared wth Block I. The control stick on the left is for the Optics. The control stick on the right is for changing spacecraft attitude by means of very small impulses of thrust.[14] This stick provides a sensitive means of steering the landmark line of sight by turning the spacecraft. Above the left-hand control stick are switches which establish the proper mode of Optics system operation. The buttons next to the right-hand stick are for marking the navigational sighting time to the Computer. The numeric panel with keys and display lights to the right is the Computer display and keyboard unit. An identical unit exists on the main panel of the Command Module where the astronauts can reach it during power and re-entry flight. The optics eye pieces are not shown in this mock-up.

The Lunar Excursion Module displays are more abbreviated. An identical Computer display and keyboard is provided on the LEM main panel. Several of the meters, attitude displays, and markings on the Excursion Module window, all display guidance quantities but are not considered a part of the guidance system.

Command Module installation features can also be noted in figure 33. The figure shows the panel installation. The computer and CDU's are the lowest member of the guidance system shown in the photo. They occupy a bay which is displaced to the left of the remaining equipment. Above them is the Power Servo Assembly shown attached to the coldplate with the coolant-carrying flex lines above it. The bellows which isolate the optical unit from structural stress are barely visible near the top of the photograph.

The optics bellows are better visible in figure 34 which shows the behind-the-panel installation. Below the bellows is a spherical member which is the Inertial Measurement Unit. The Inertial

Measurement Unit housing is mounted on precise surfaces of the oval-like structure. This oval structure is called the Navigation Base and is mounted with three strain-isolation mounts to spacecraft structure. The Navigation Base also carries the optics unit thus giving the right mechanical connection necessary for Inertial Measurement Unit alignment.

The Excursion Module Guidance and Navigation system installation has two locations. Shown in figure 35 is the front location. This is outside the vehicle pressure hull and above the crew's heads. In the foreground is the Optical Alignment Telescope. Behind it a toroidal member with the protruding struts is a Navigation Base that connects this Telescope to the spherical Inertial Measurement Unit. Behind the Inertial Measurement Unit is that portion of the electronics which could not easily tolerate a cable run having a length of the order of 20 feet. The remainder of electronics, consisting of most of the PSA, Coupling Data Units, and the computer, is near the very back of the vehicle.

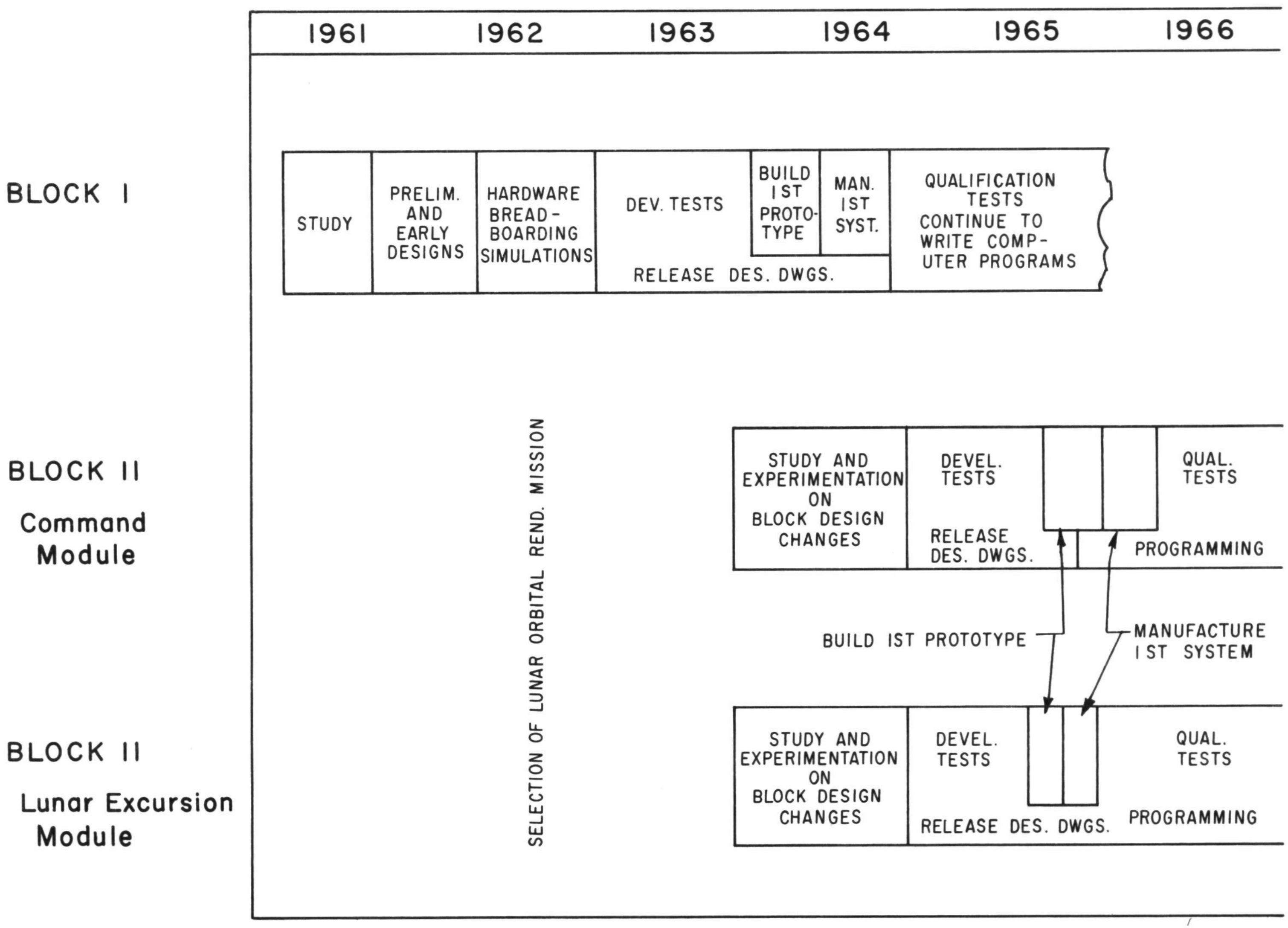

Fig. 1 Apollo G&N Program Milestones

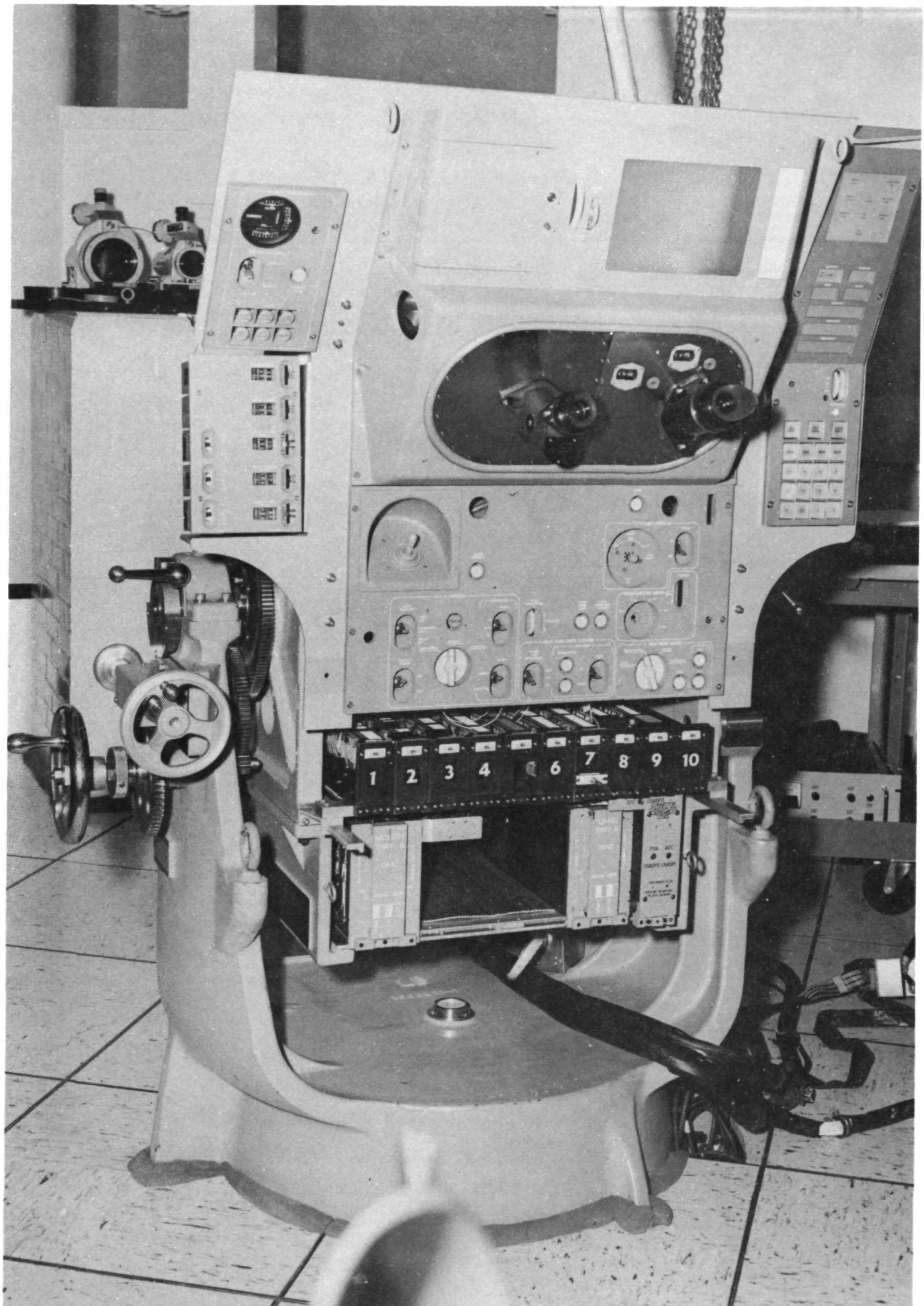

Fig. 2 First Prototype (System Number 5) On Test at Instrumentation Laboratory of MIT

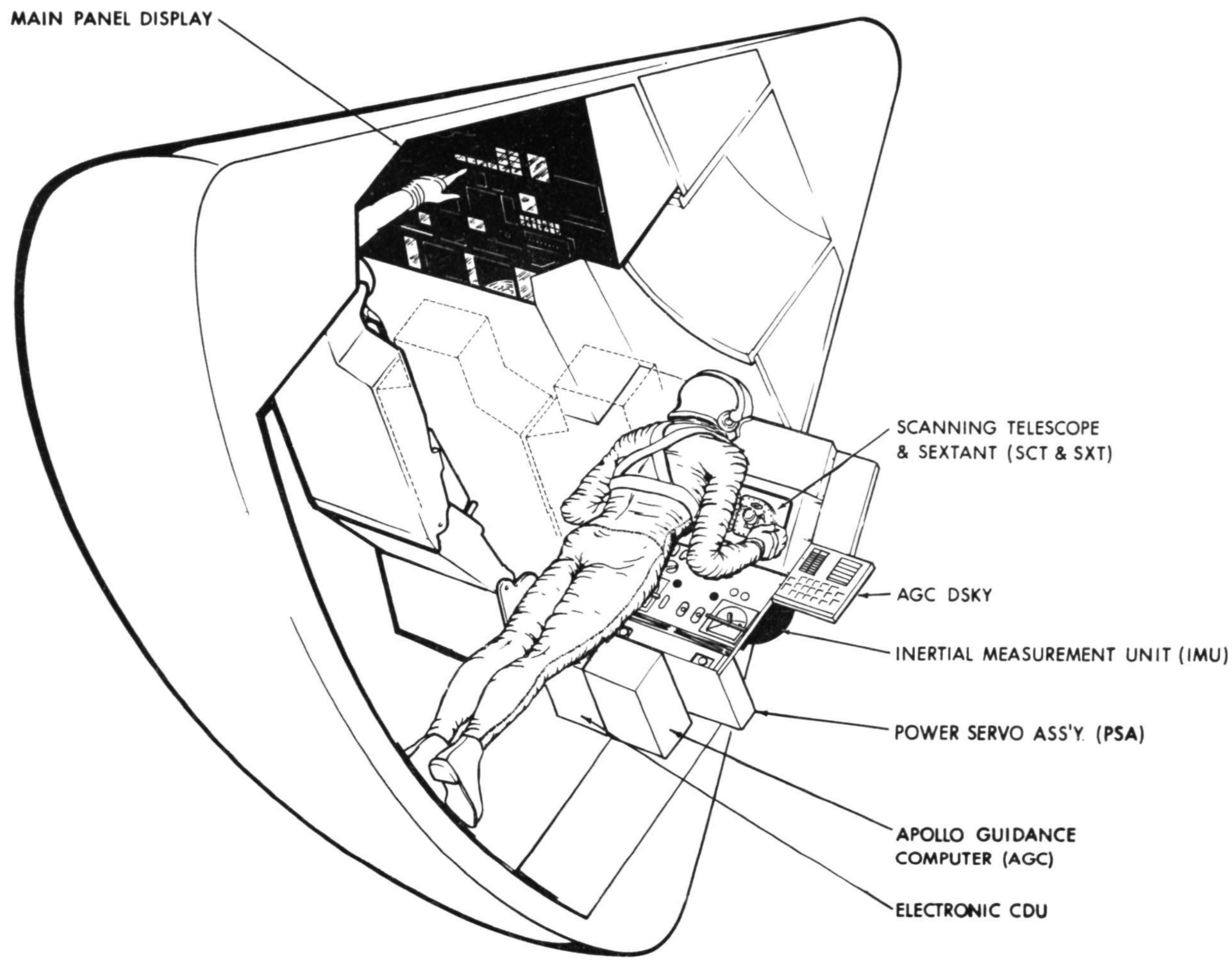

Fig. 3 Location of the Guidance and Navigation System in the Command Module

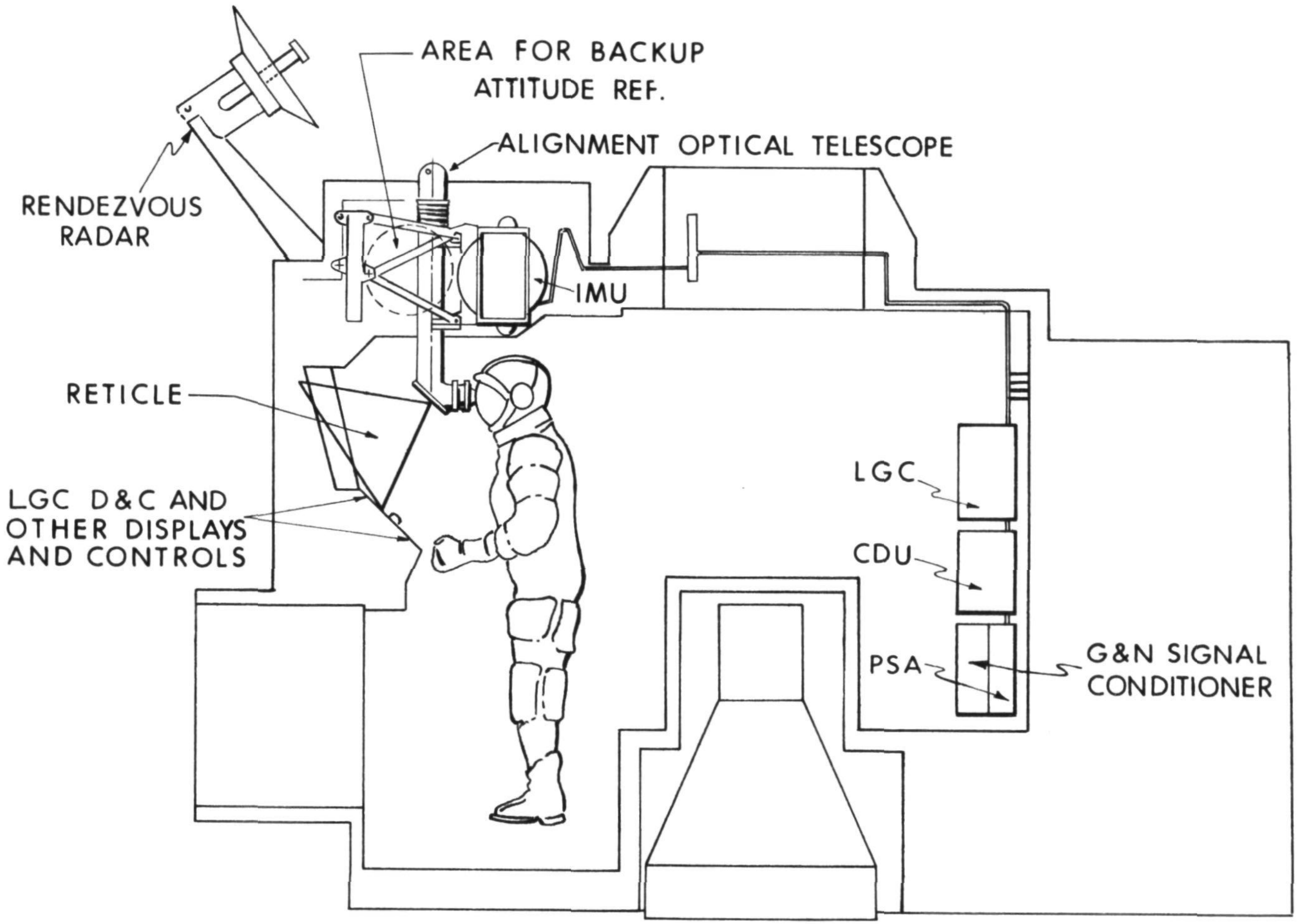

Fig. 4 Location of the Guidance and Navigation system in the Lunar Excursion Module

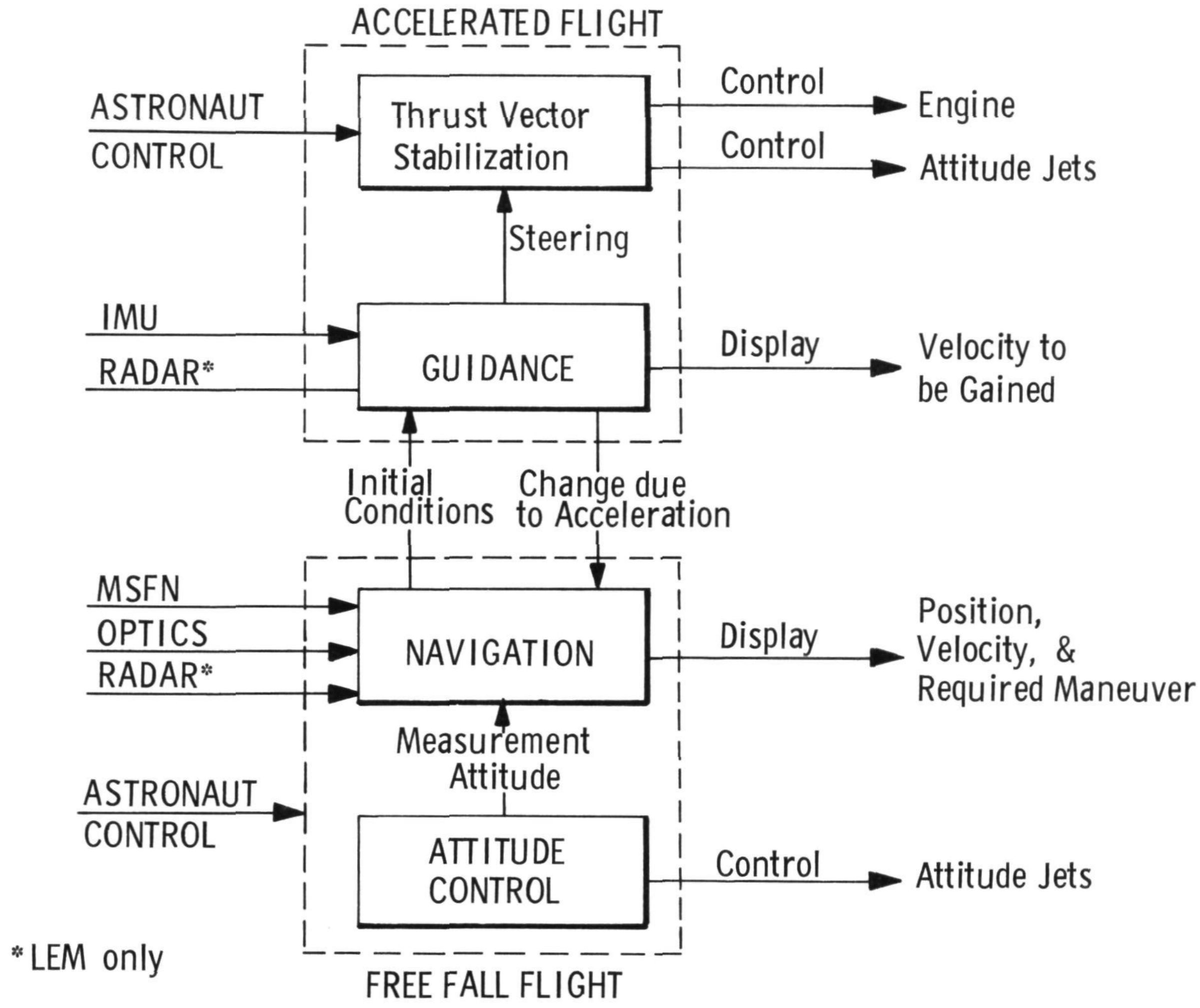

Fig. 5 Apollo Guidance Navigation - Function Flow

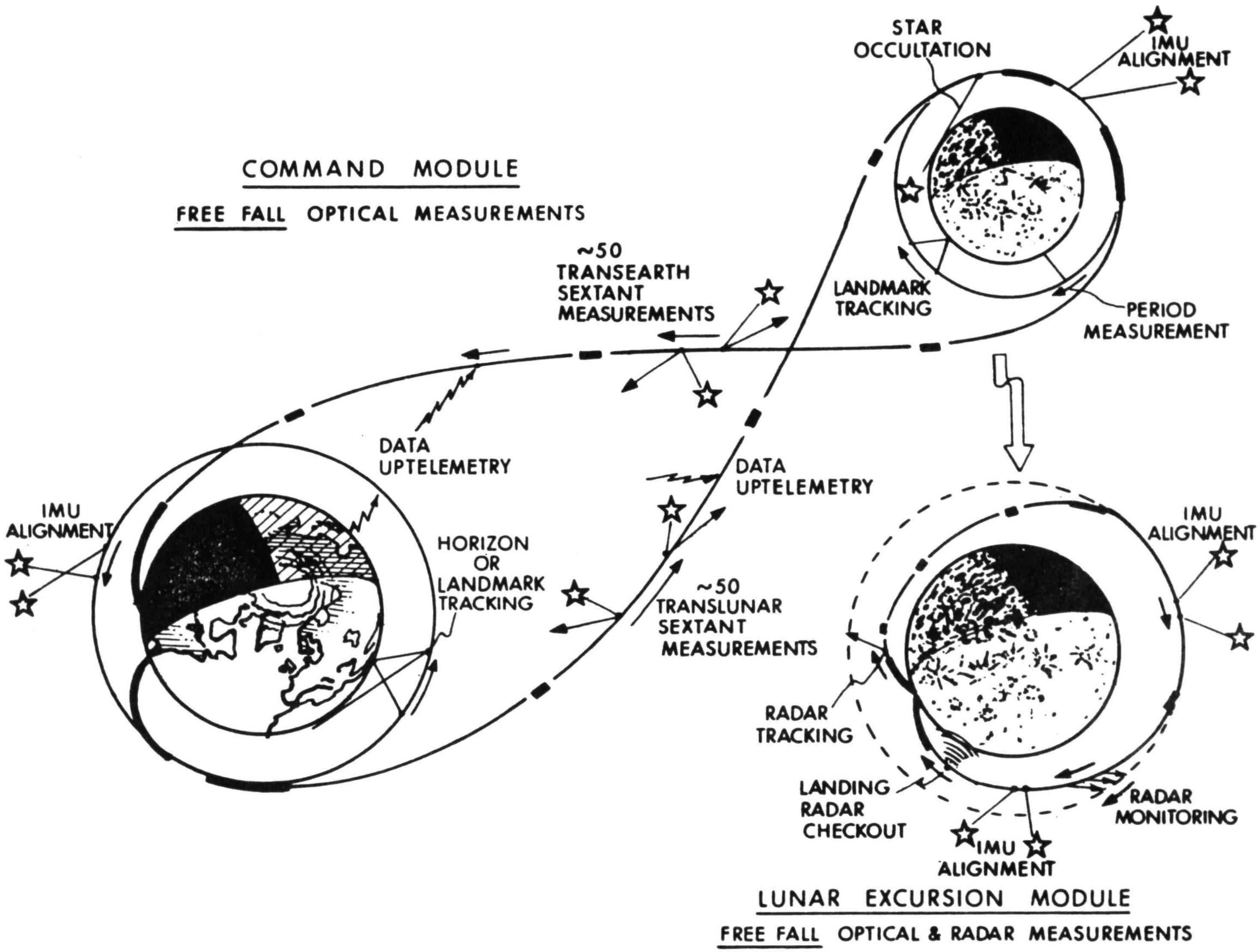

Fig. 6 Navigation Mission Phases

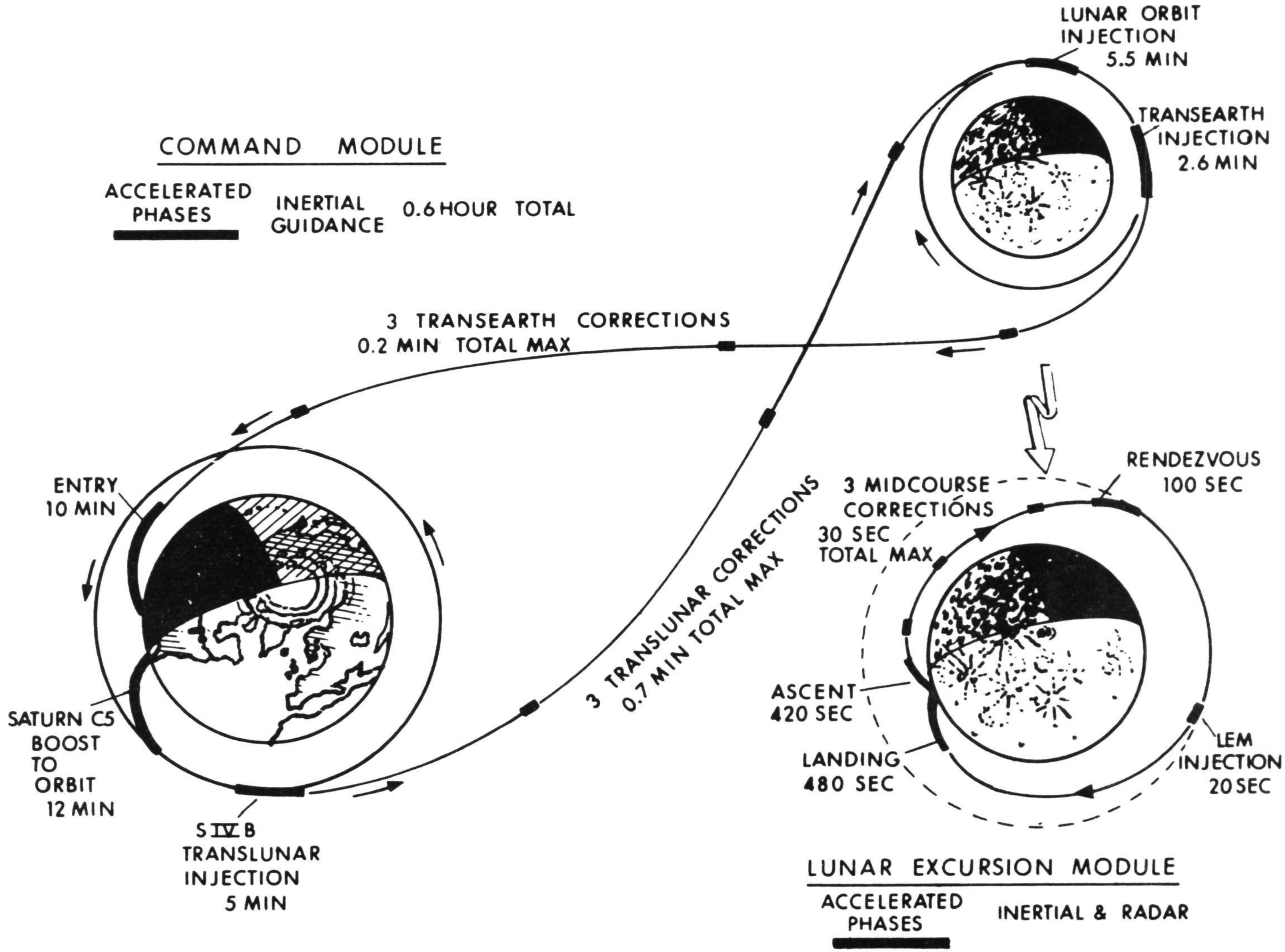

Fig. 7 Guidance Mission Phases

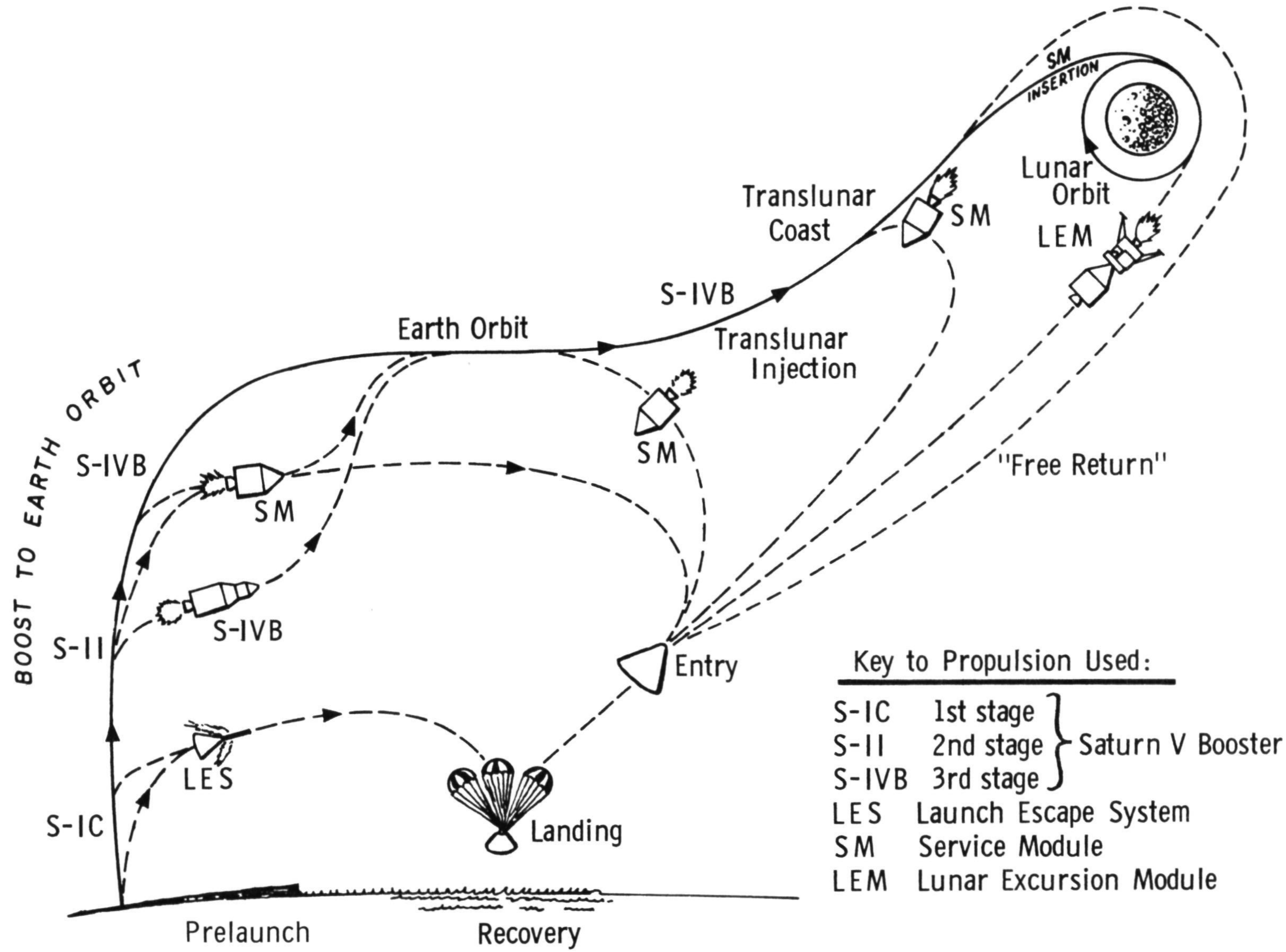

Fig. 8 Mission Abort Paths

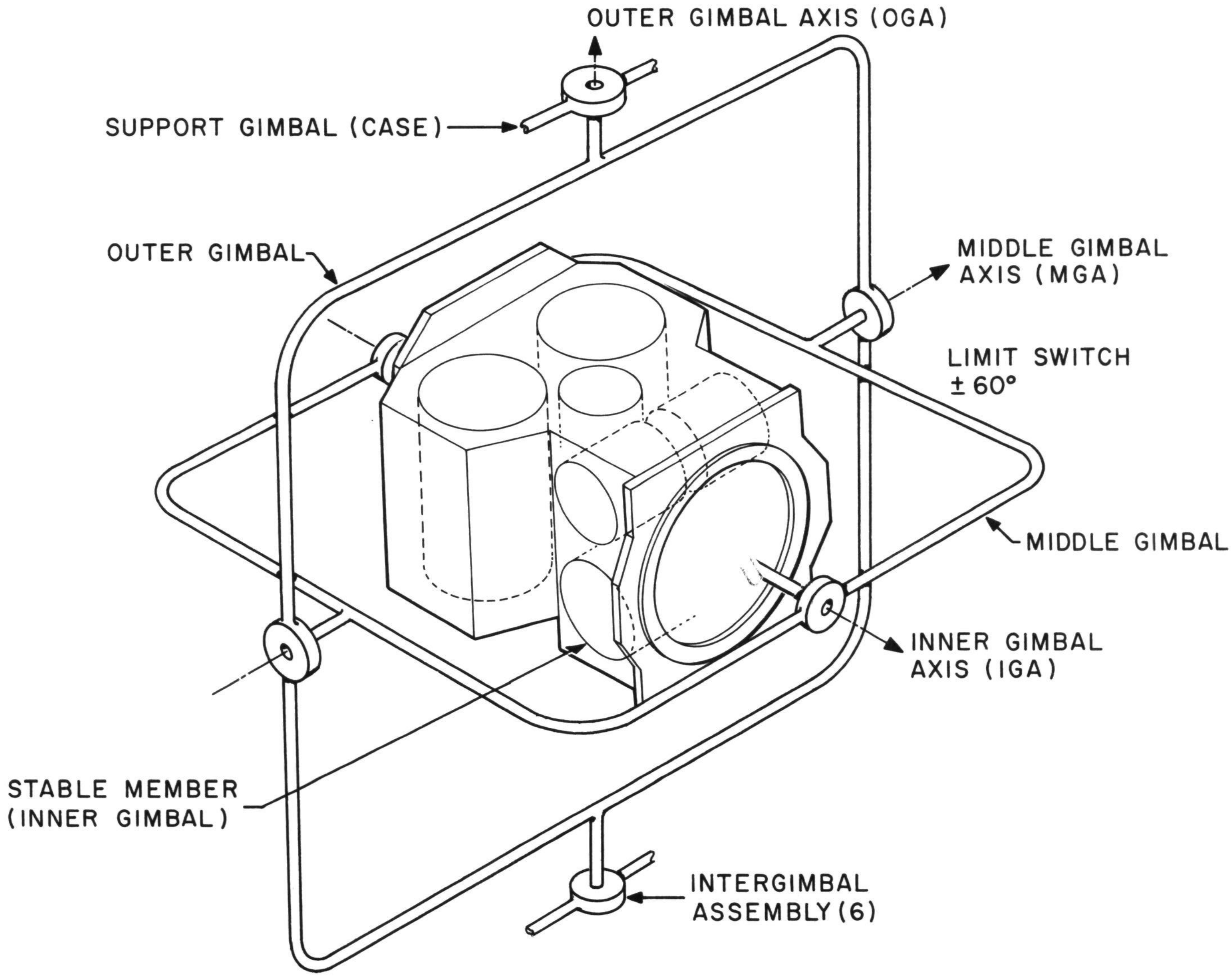

Fig. 9 Schematic of the Inertial Measurement Unit

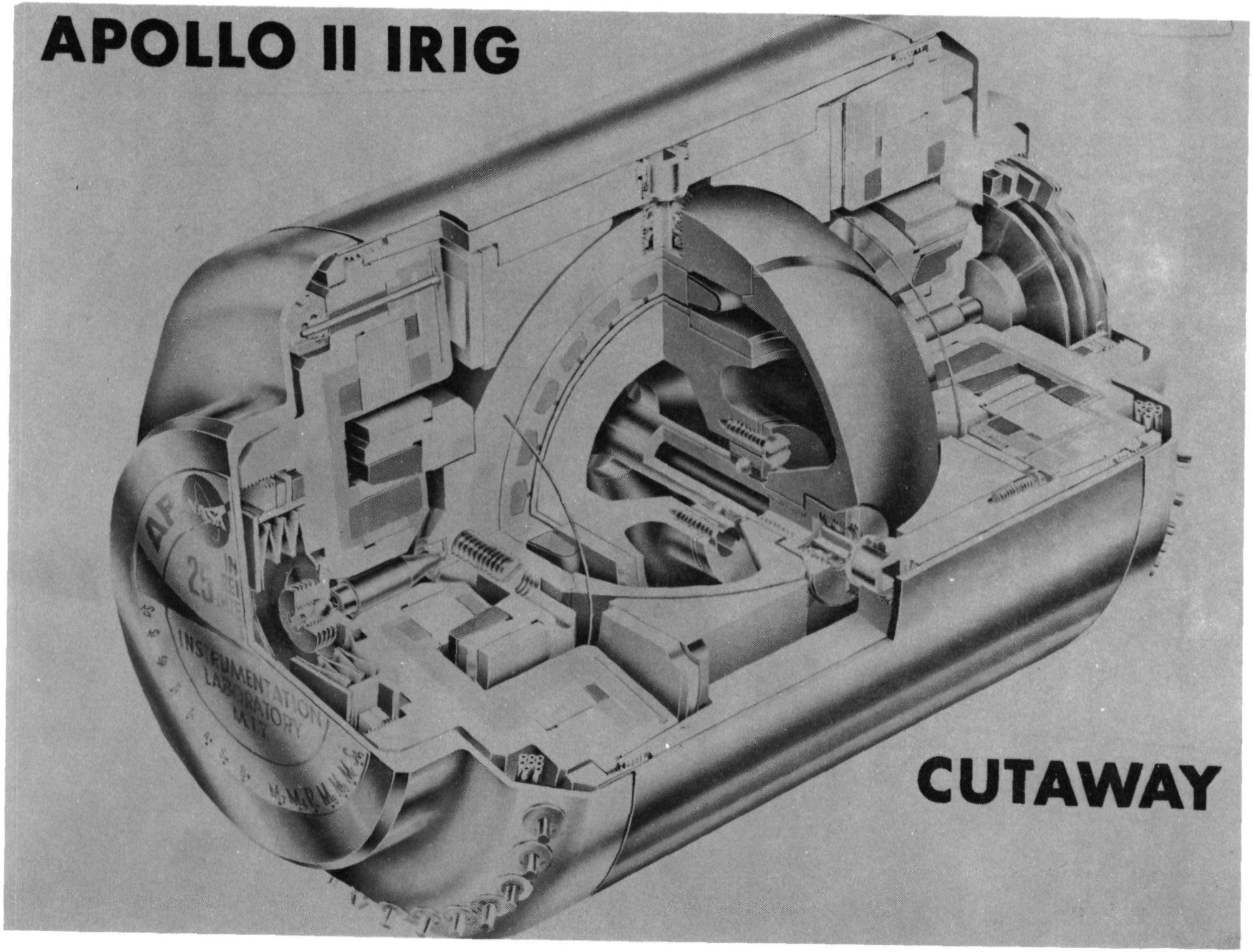

Fig. 10 Cutaway of Block II 25 IRIG (25 Inertial Reference Integrating Gyro)

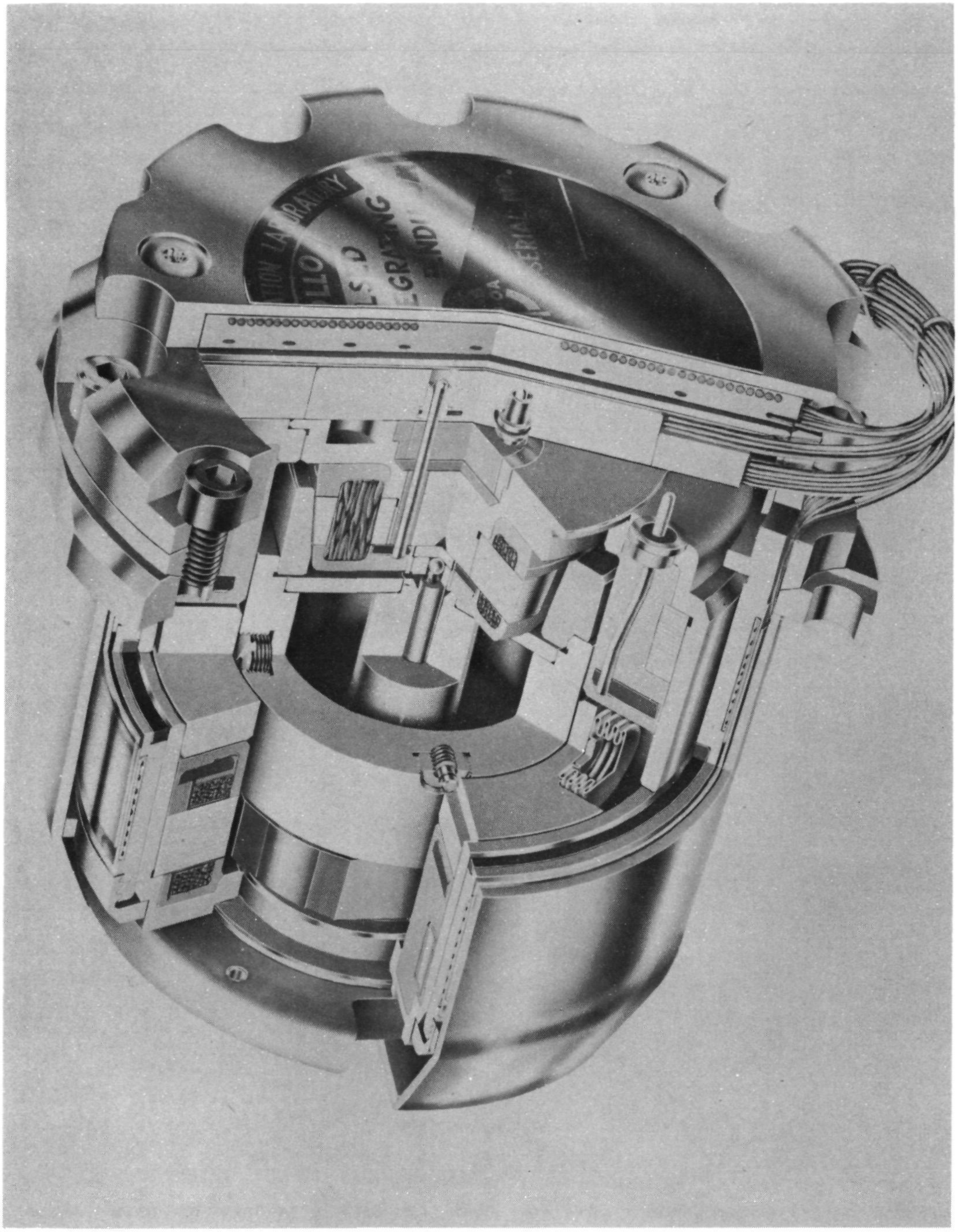

Fig. 11 Photograph of 16 PIP (16 Pulse Integrating Pendulum)

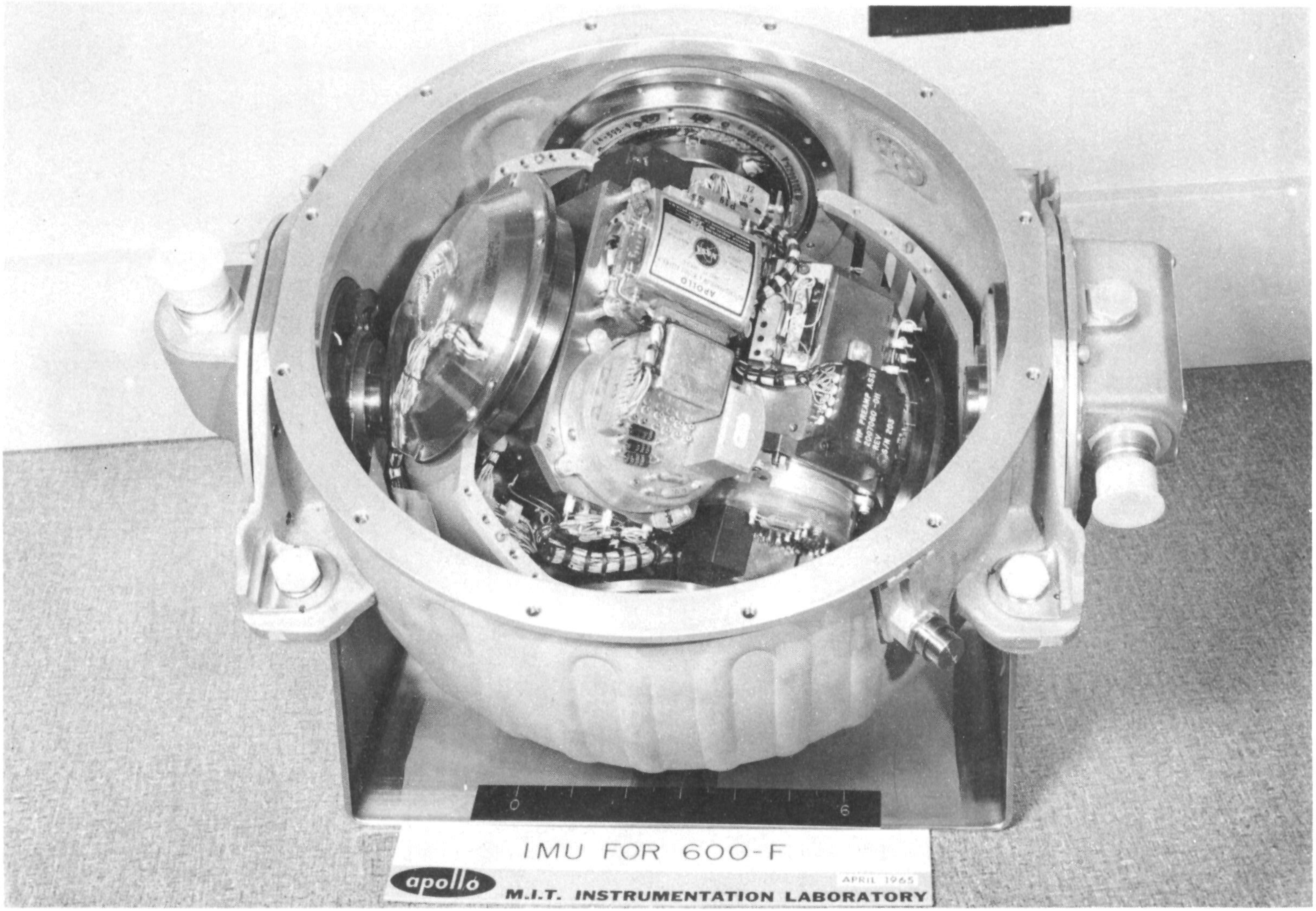

Fig. 12 Inertial Measurement Unit of System 600F (LEM Functional)

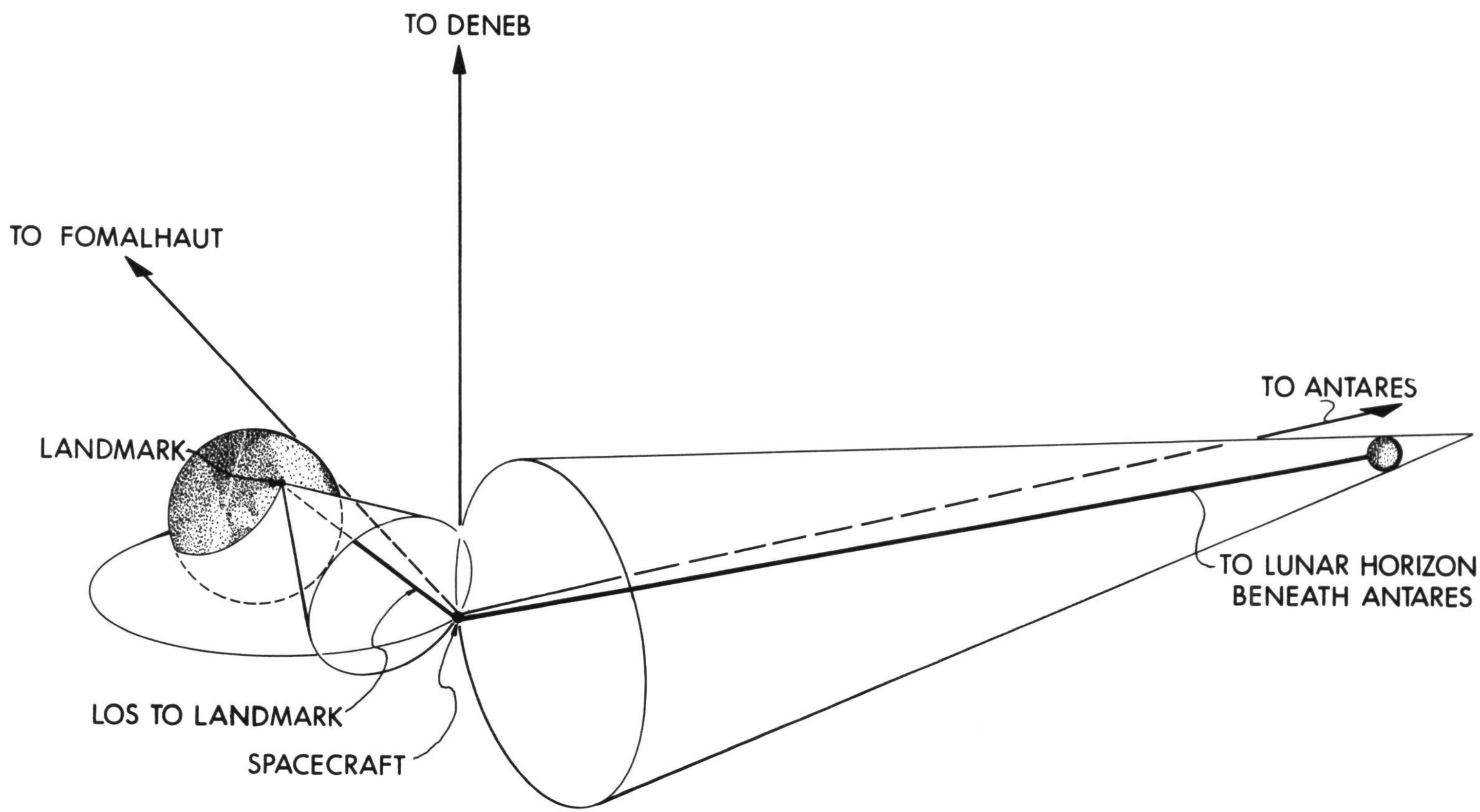

Fig. 13 Geometry of a Navigation Fix in Space

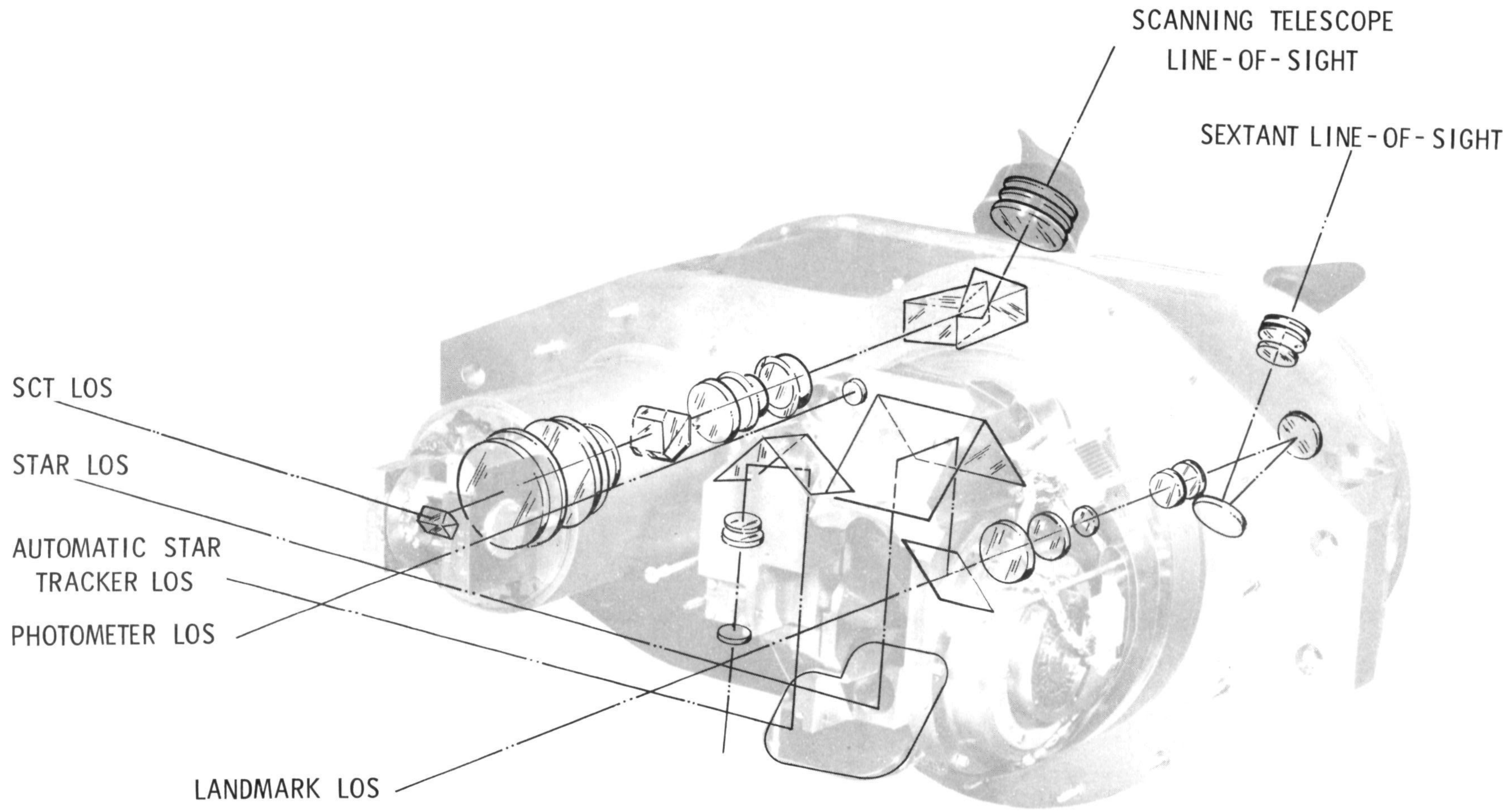

Fig. 14 Simplified Cutaway of the Command Module Optics

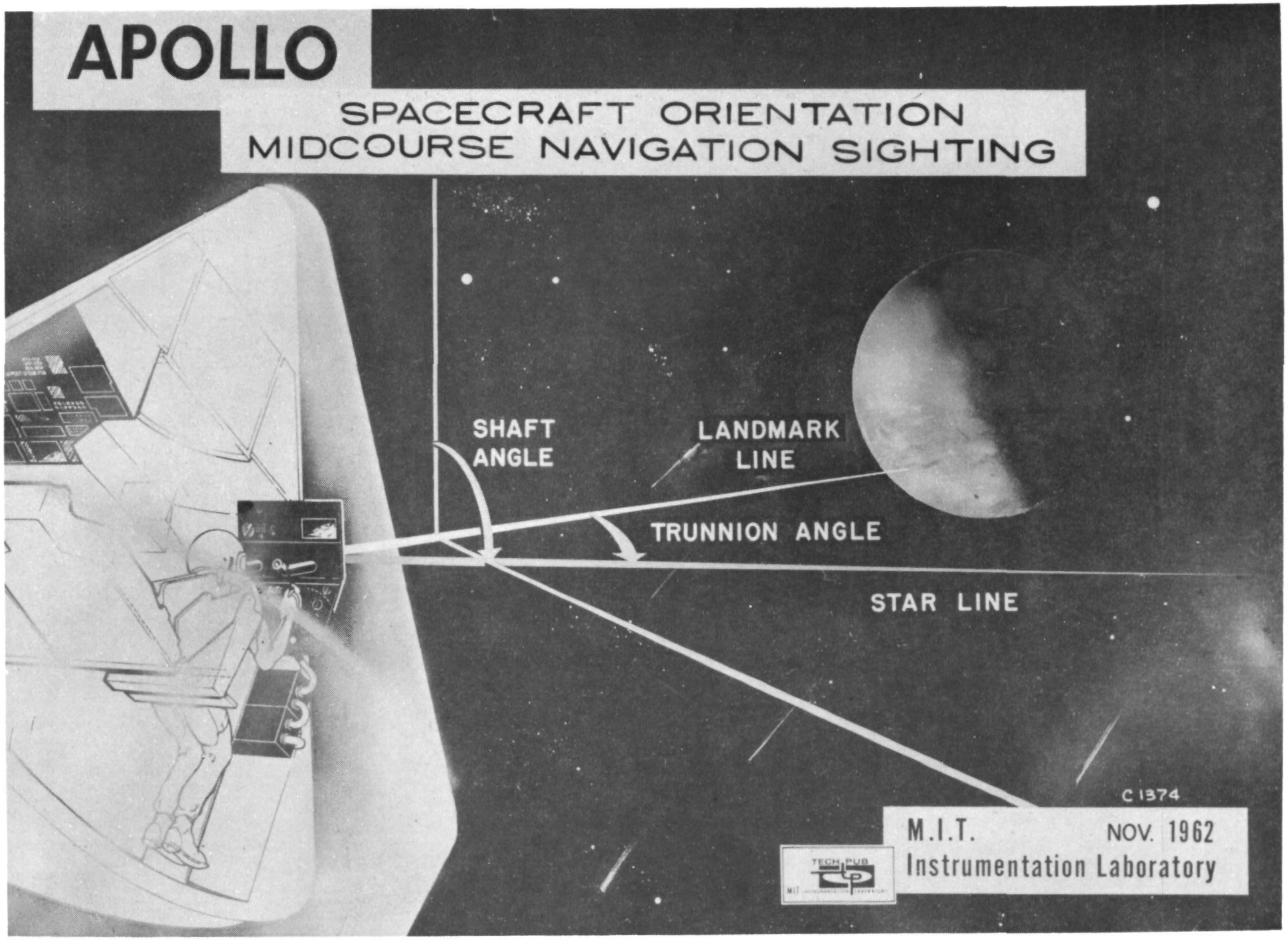

Fig. 15 Spacecraft Orientation-Midcourse Navigation Sighting

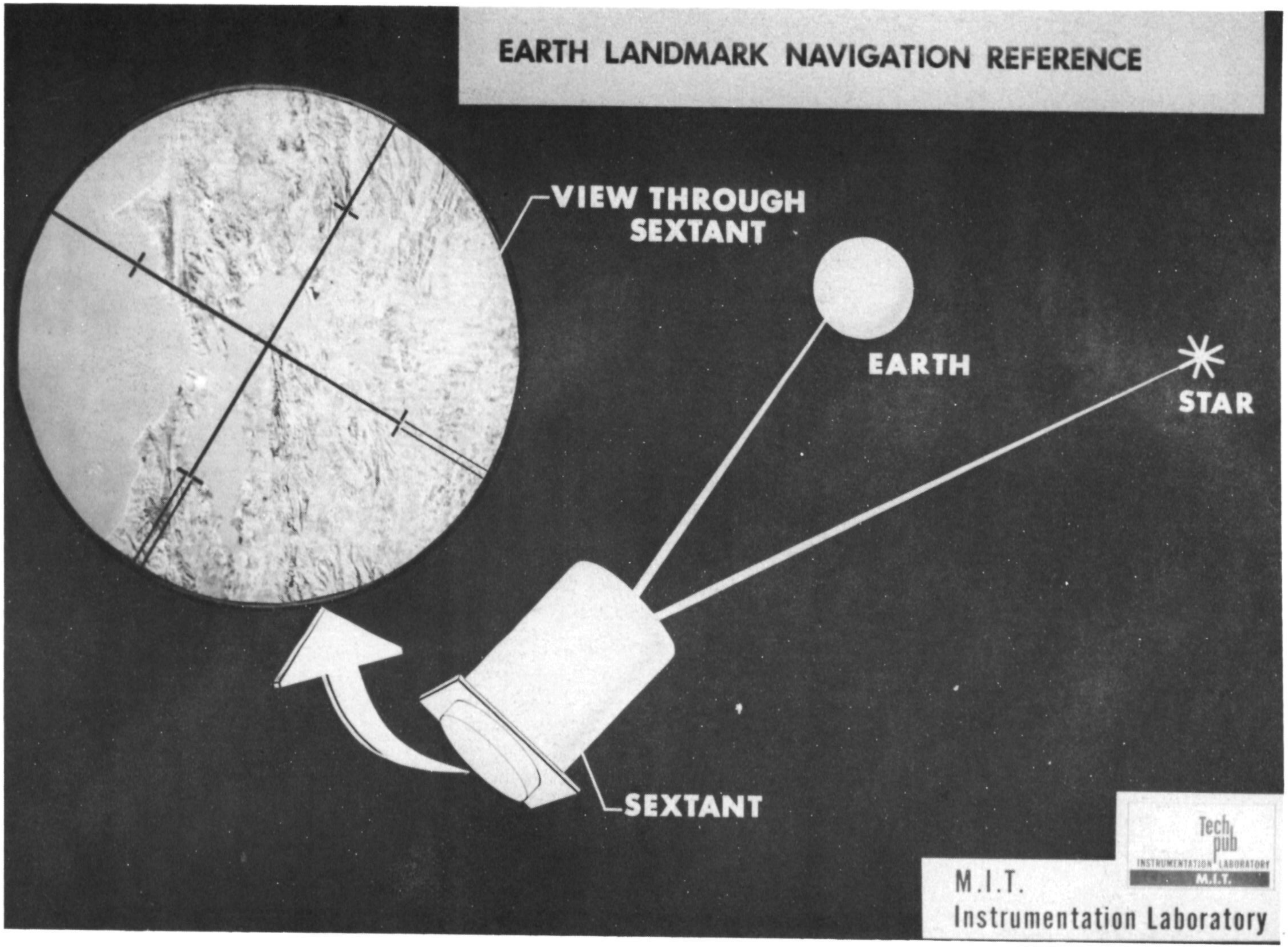

Fig. 16 Earth Landmark Sighting

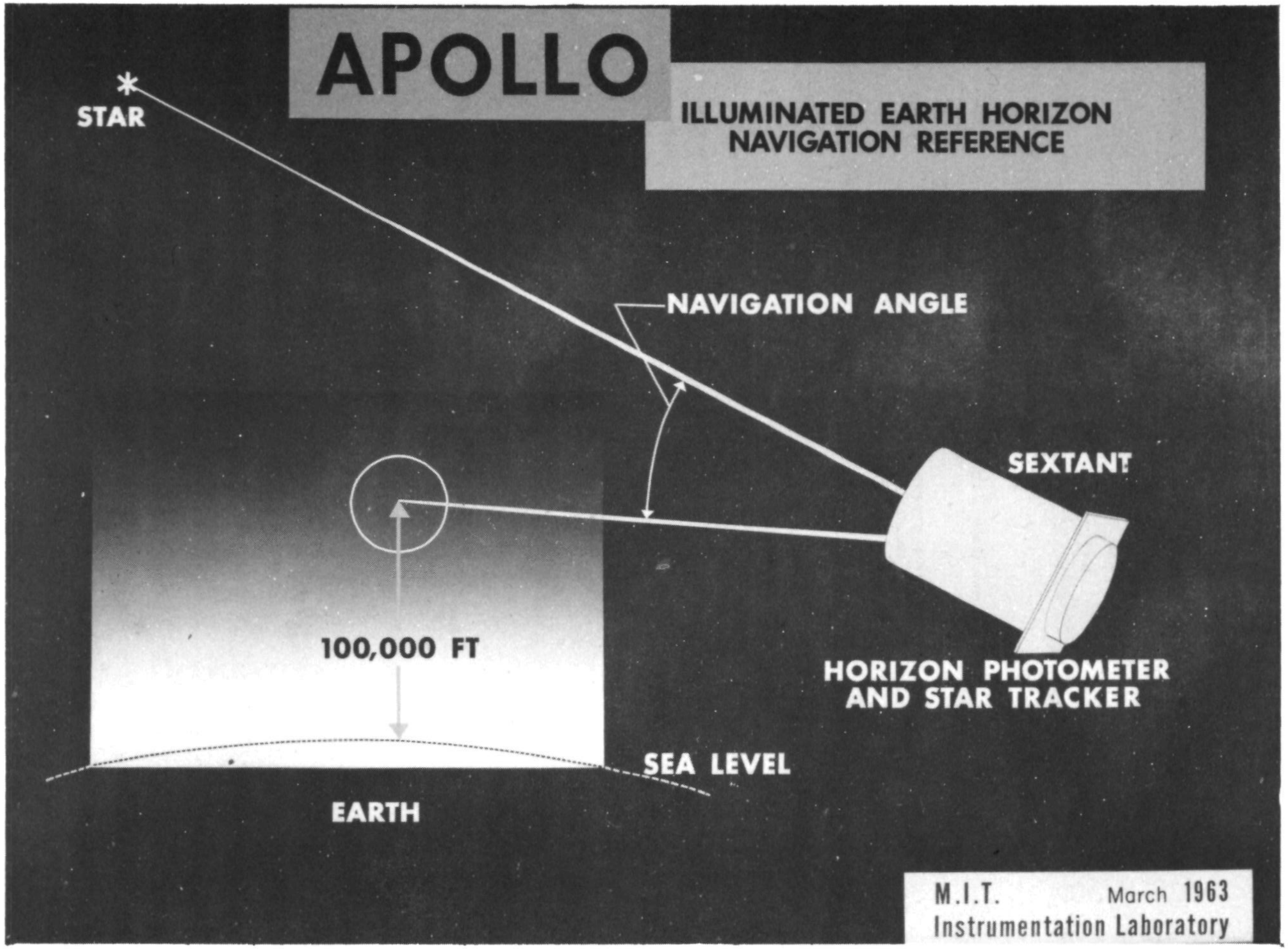

Fig. 17 Earth Horizon Sighting

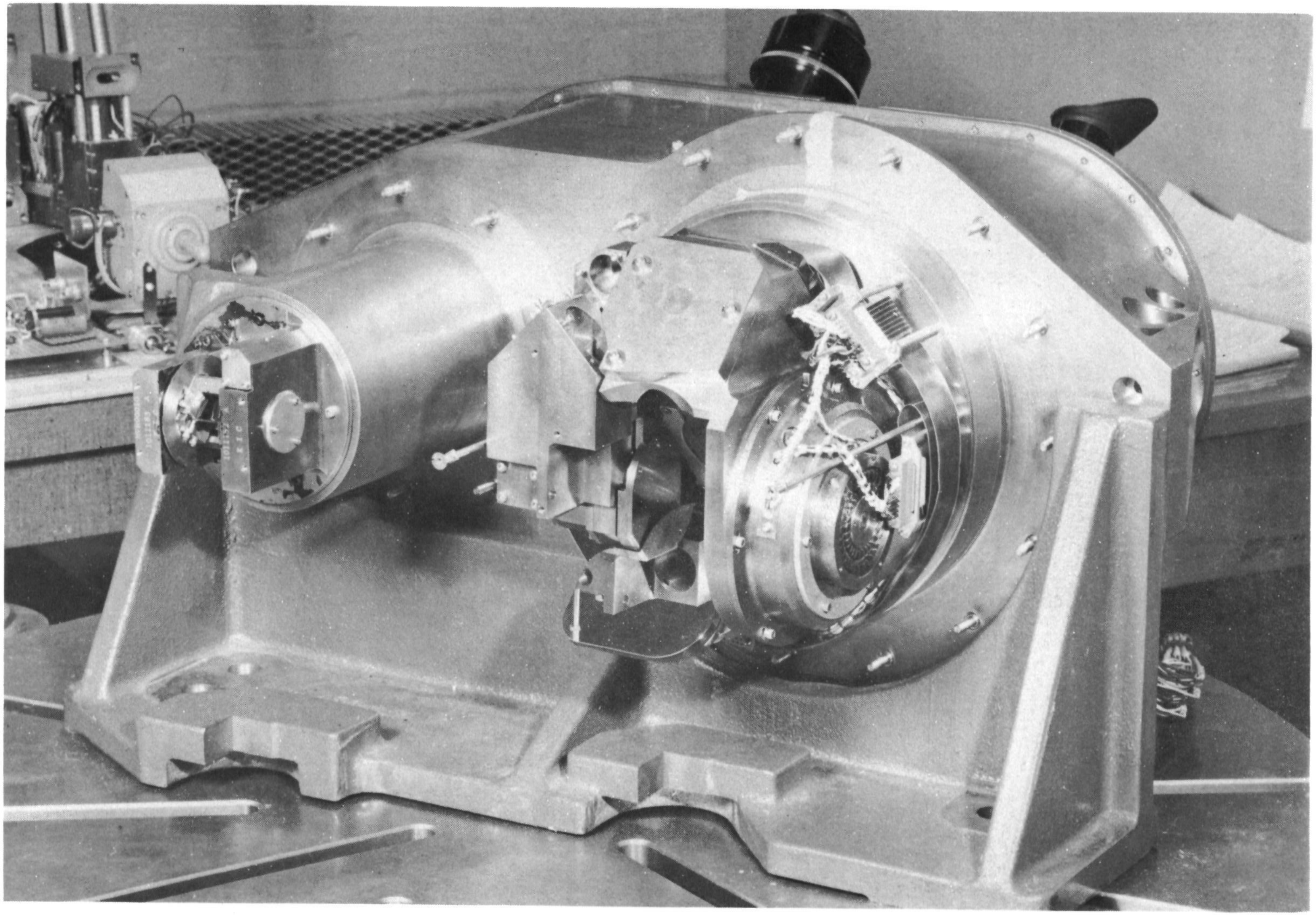

Fig. 18 Outboard End of the Command Module Optics, Cover Removed

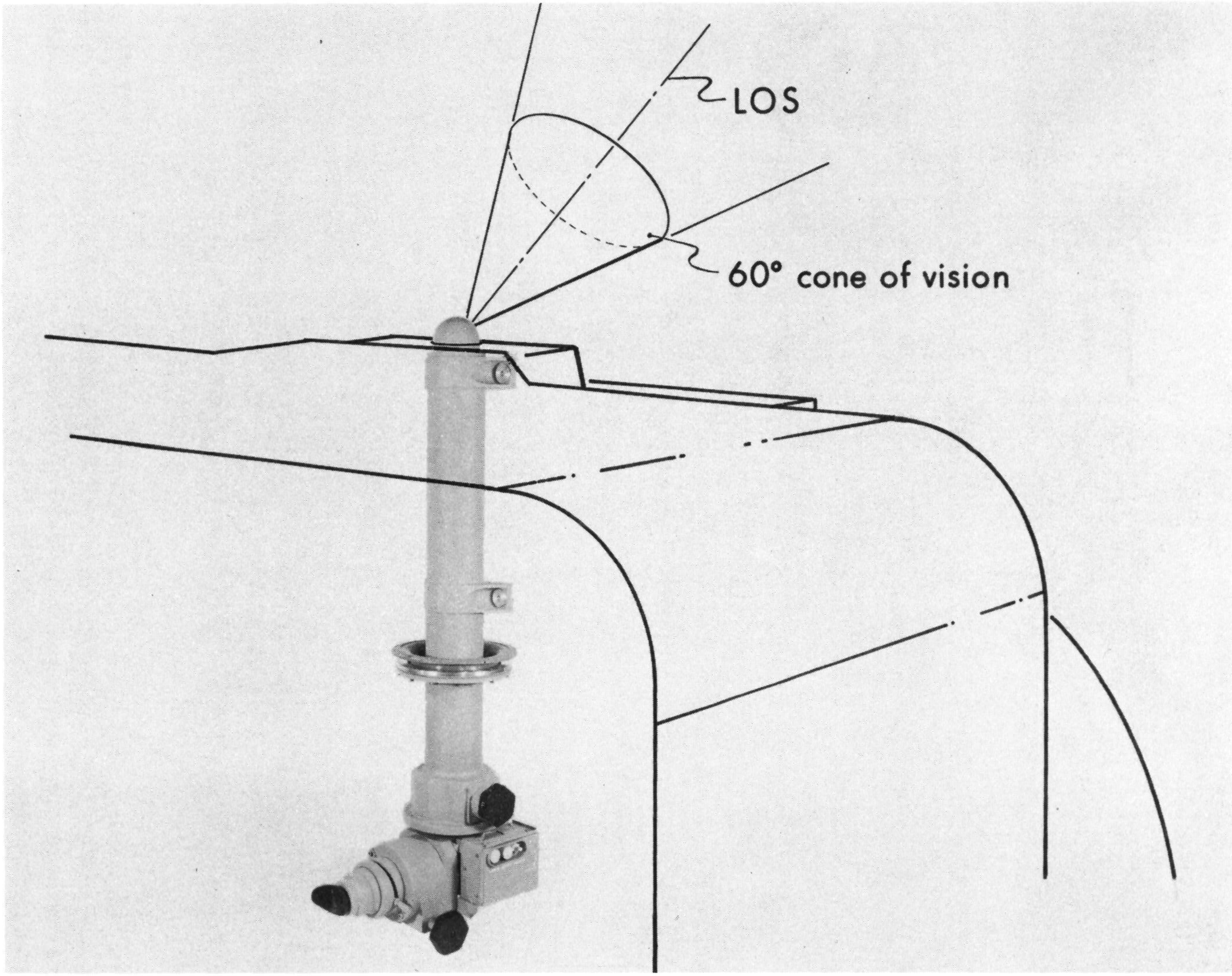

Fig. 19 Mockup of the Lunar Excursion Module Optics

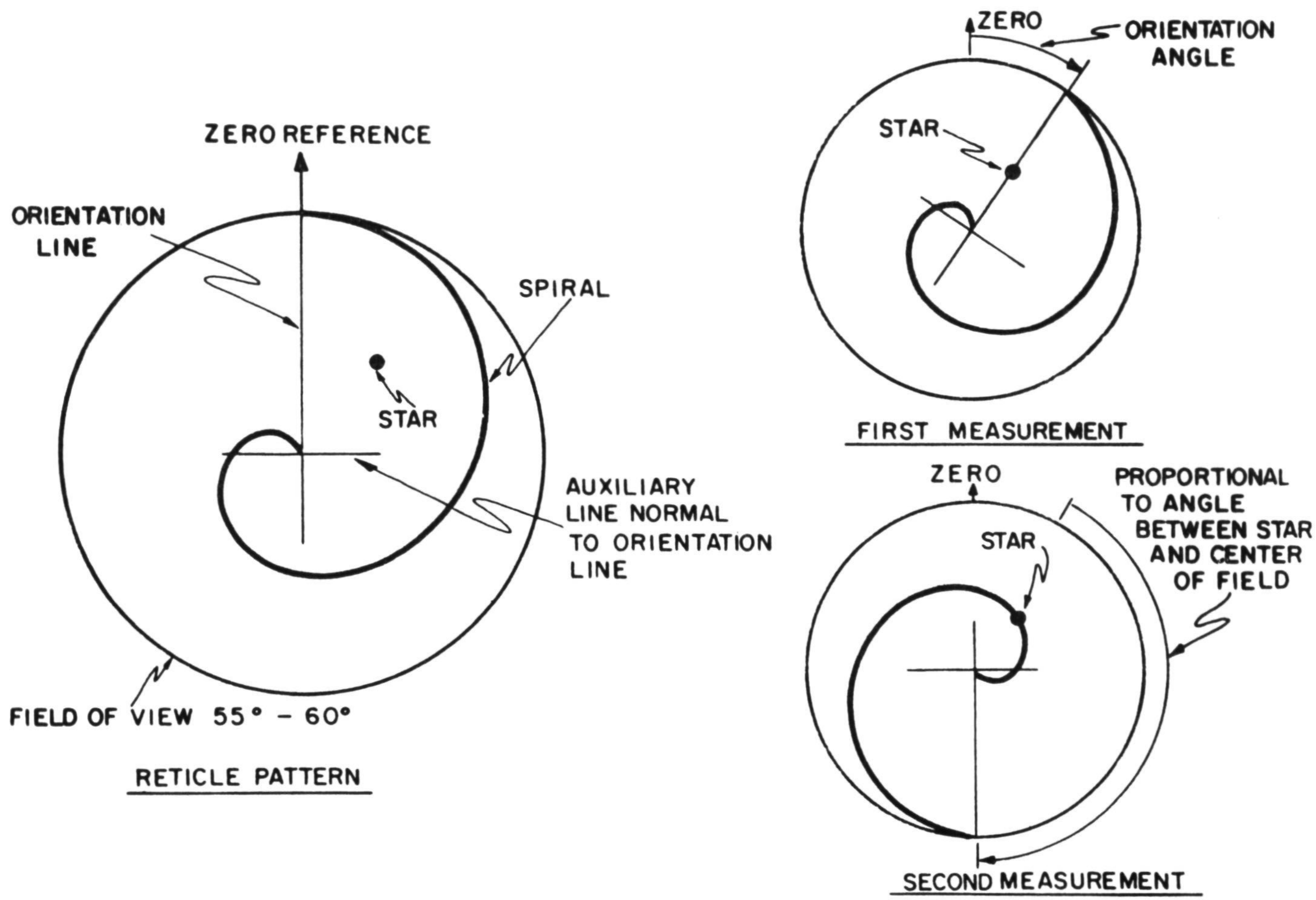

Fig. 20 LEM Optics Reticle

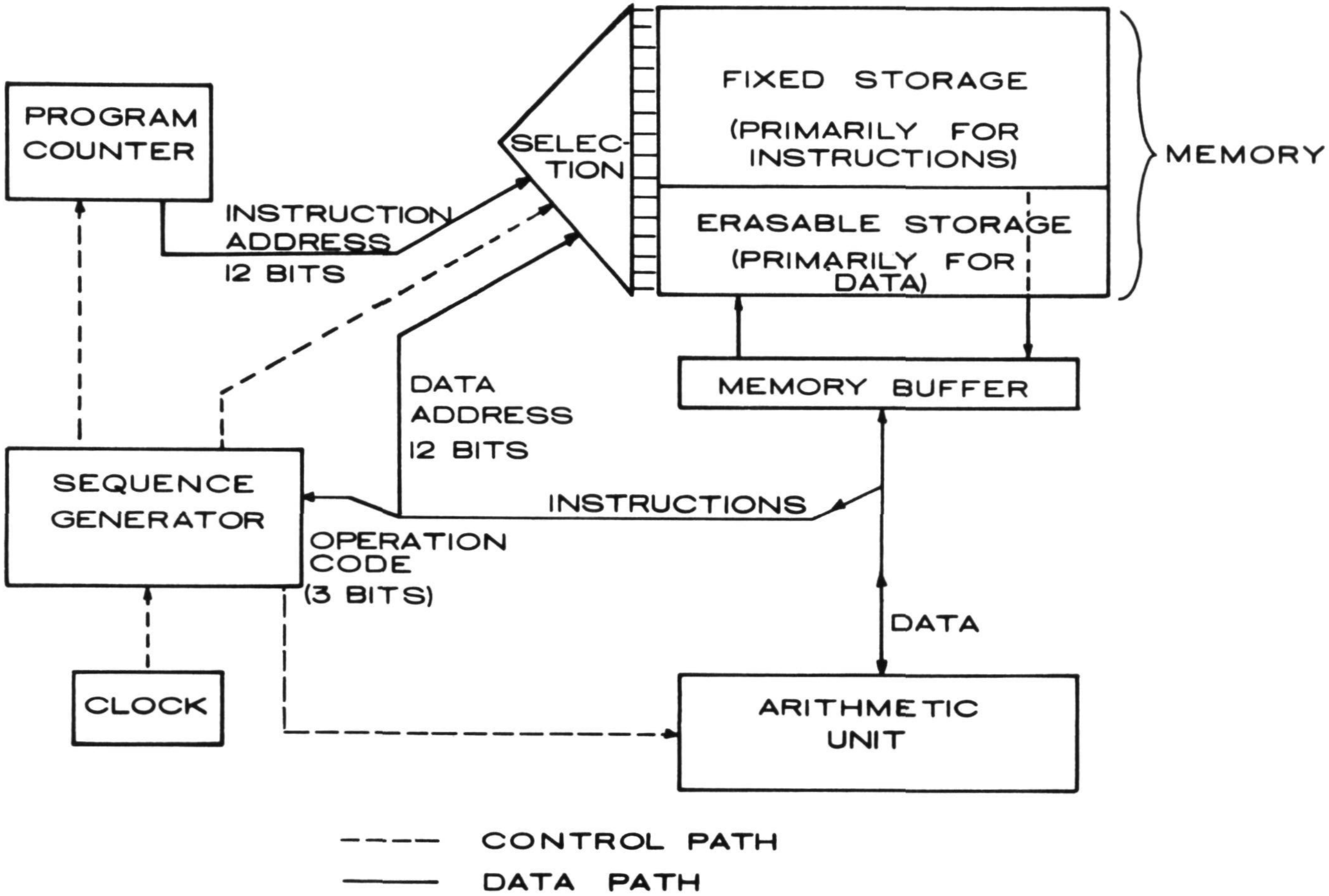

Fig. 21 Simplified Block Diagram of the AGC

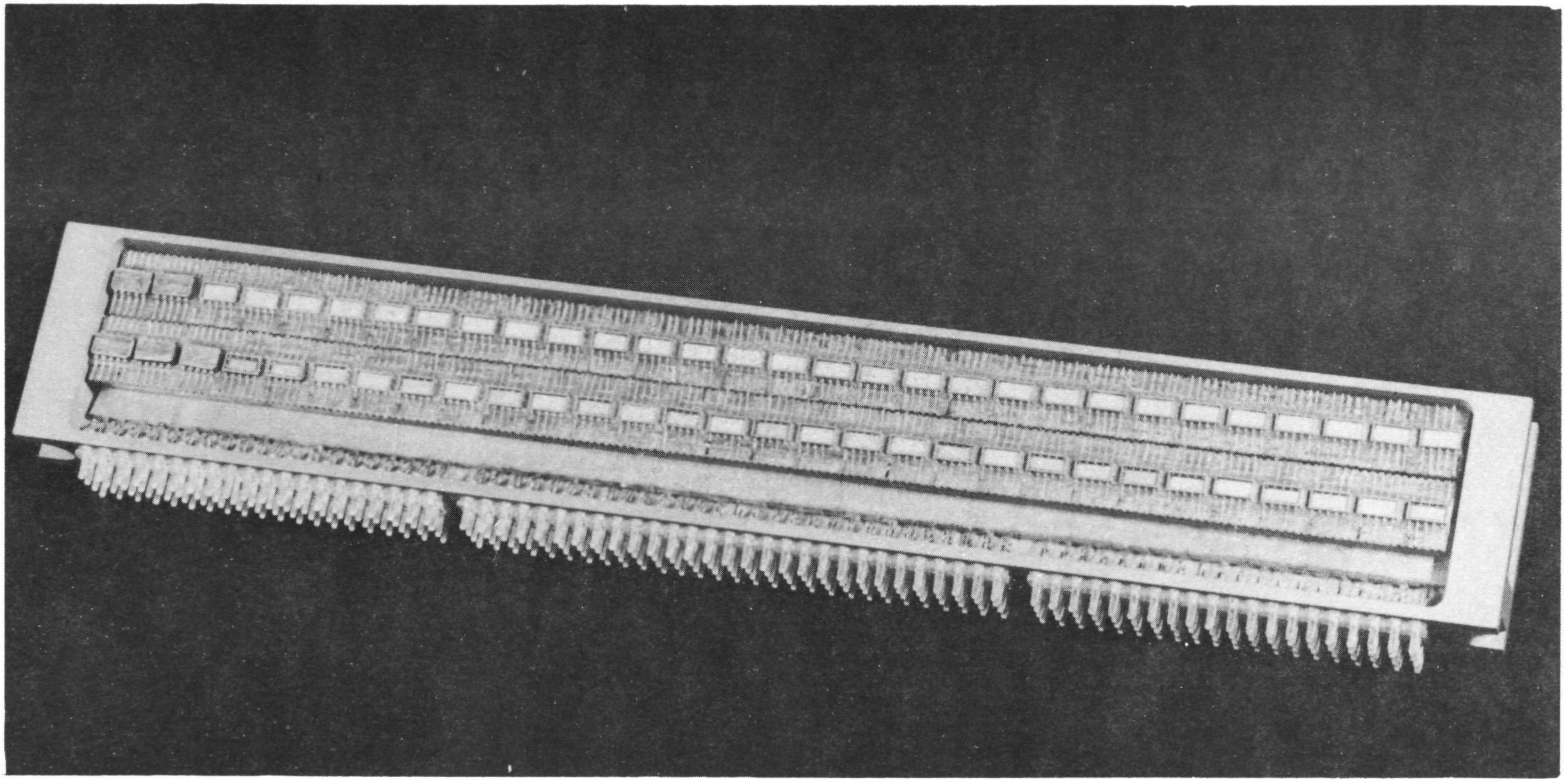

Fig. 22 Unpotted Computer Logic Module, Block II

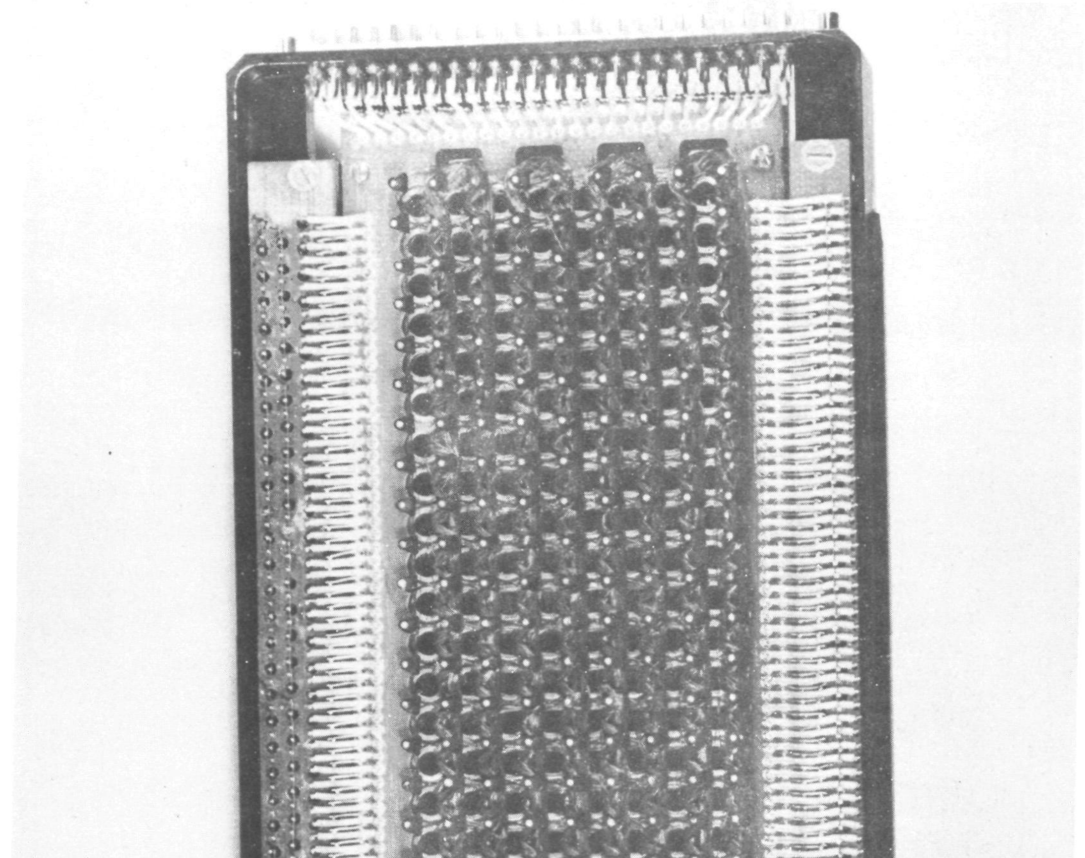

Fig. 23 Unpotted Core Rope Memory Module, Block II

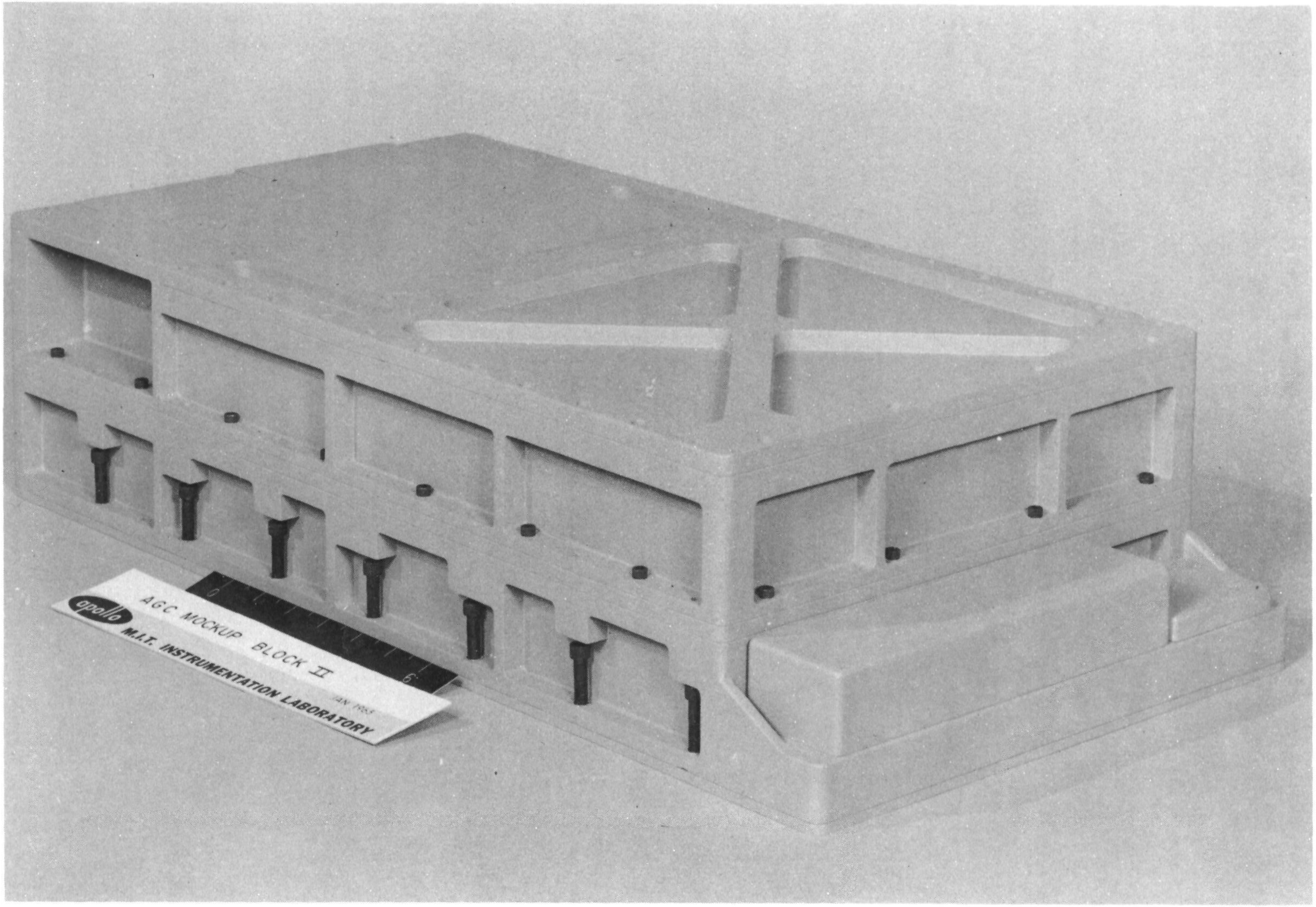

Fig. 24 Computer Mockup, Block II

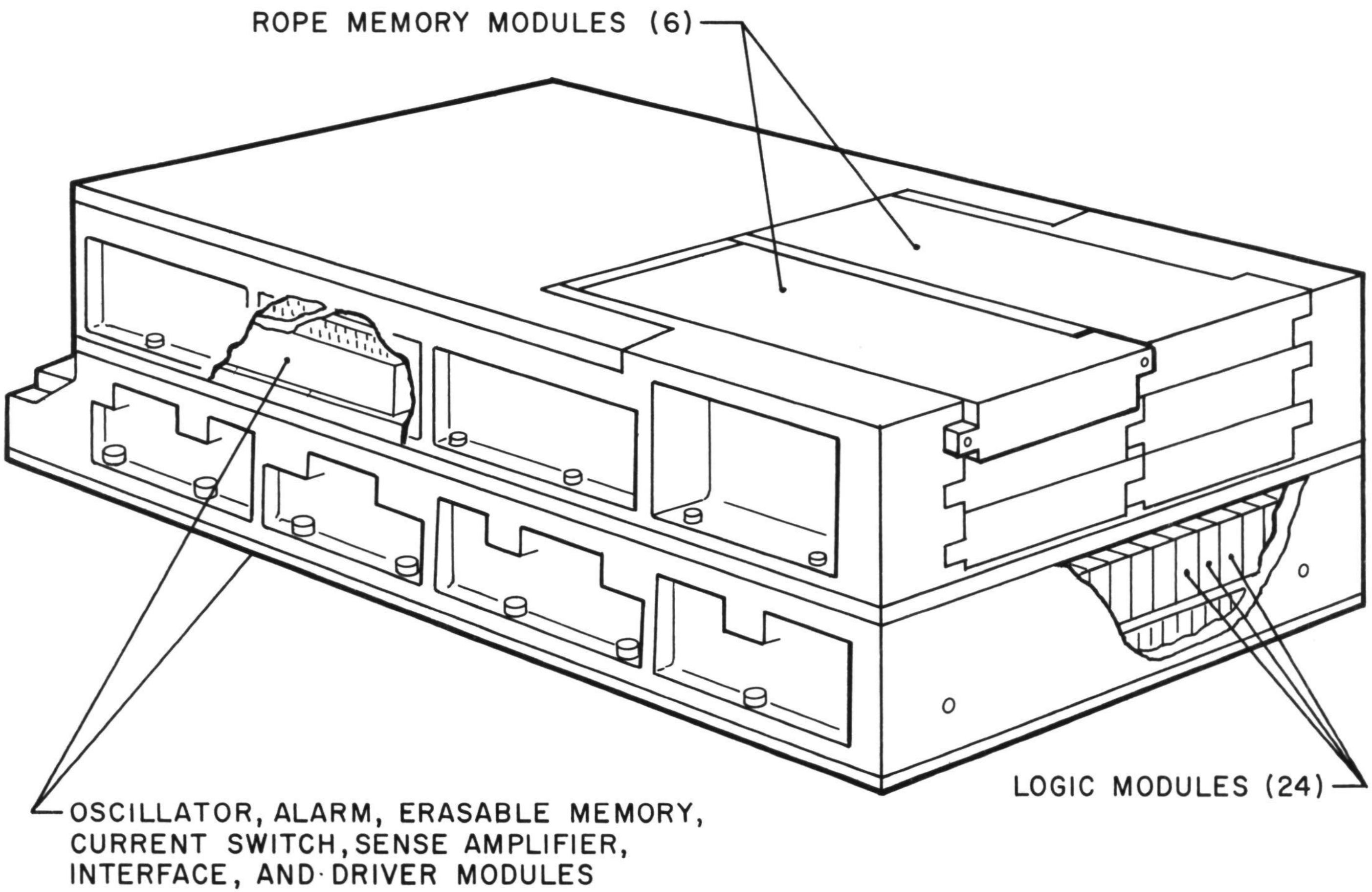

Fig. 25 Computer Cutaway

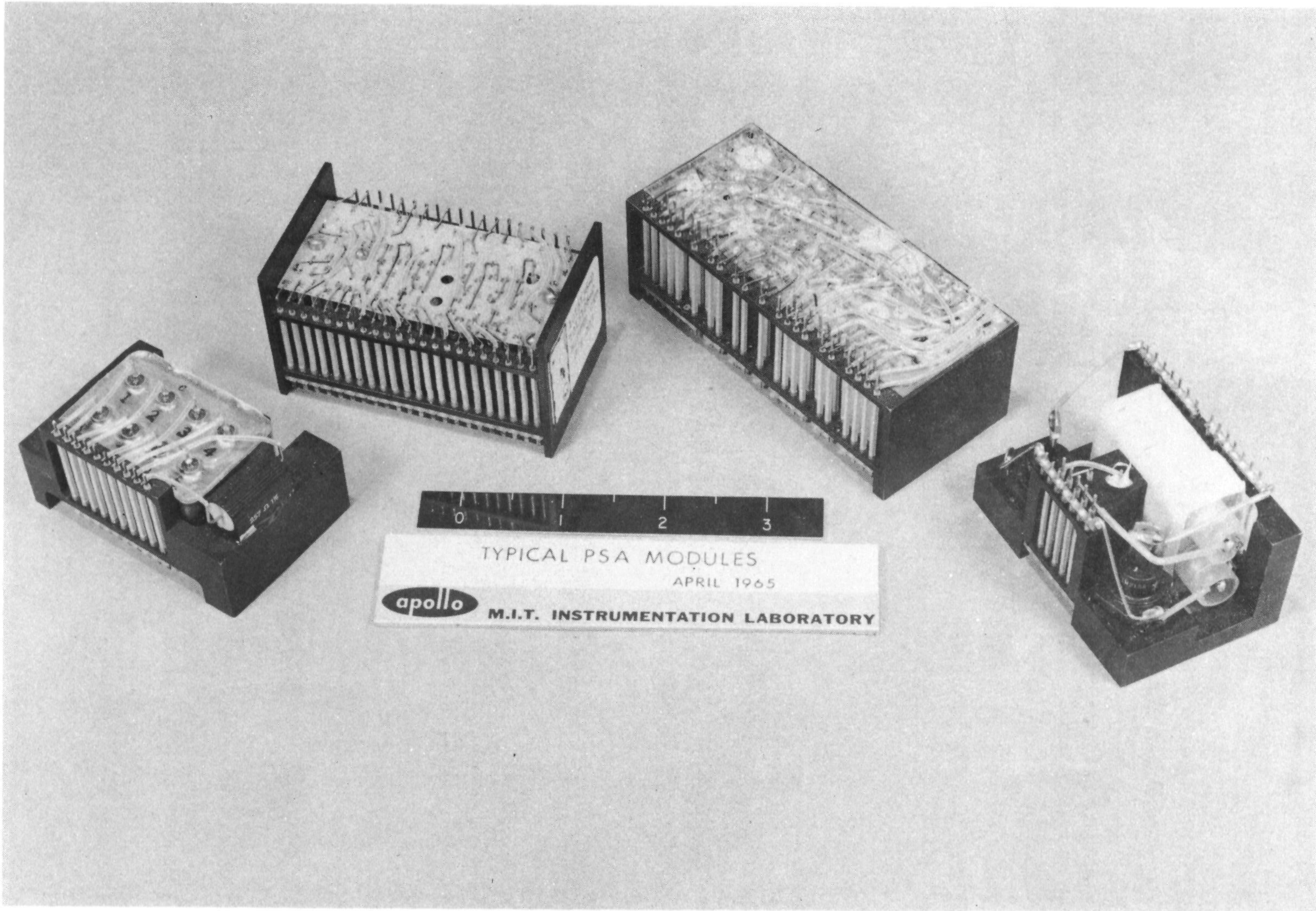

Fig. 26 Typical Power Servo Modules

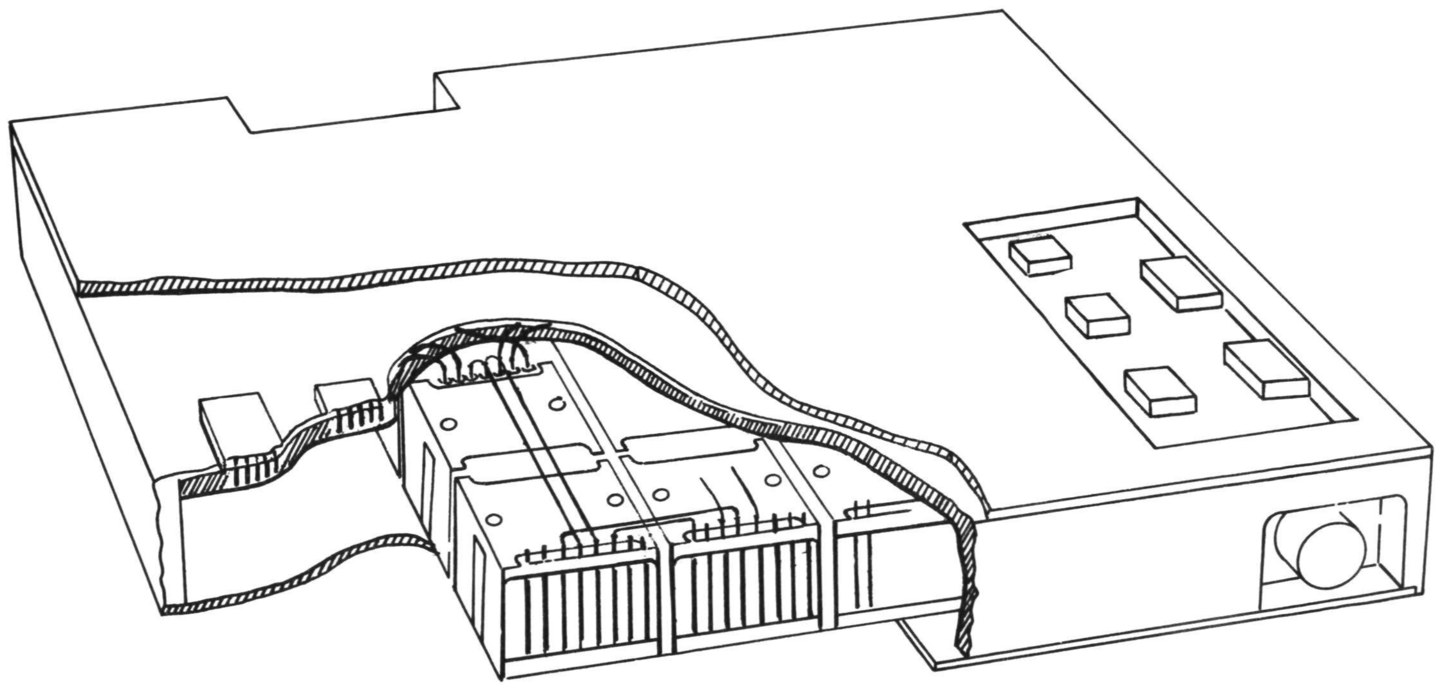

Fig. 27 Power Servo Assembly Cutaway

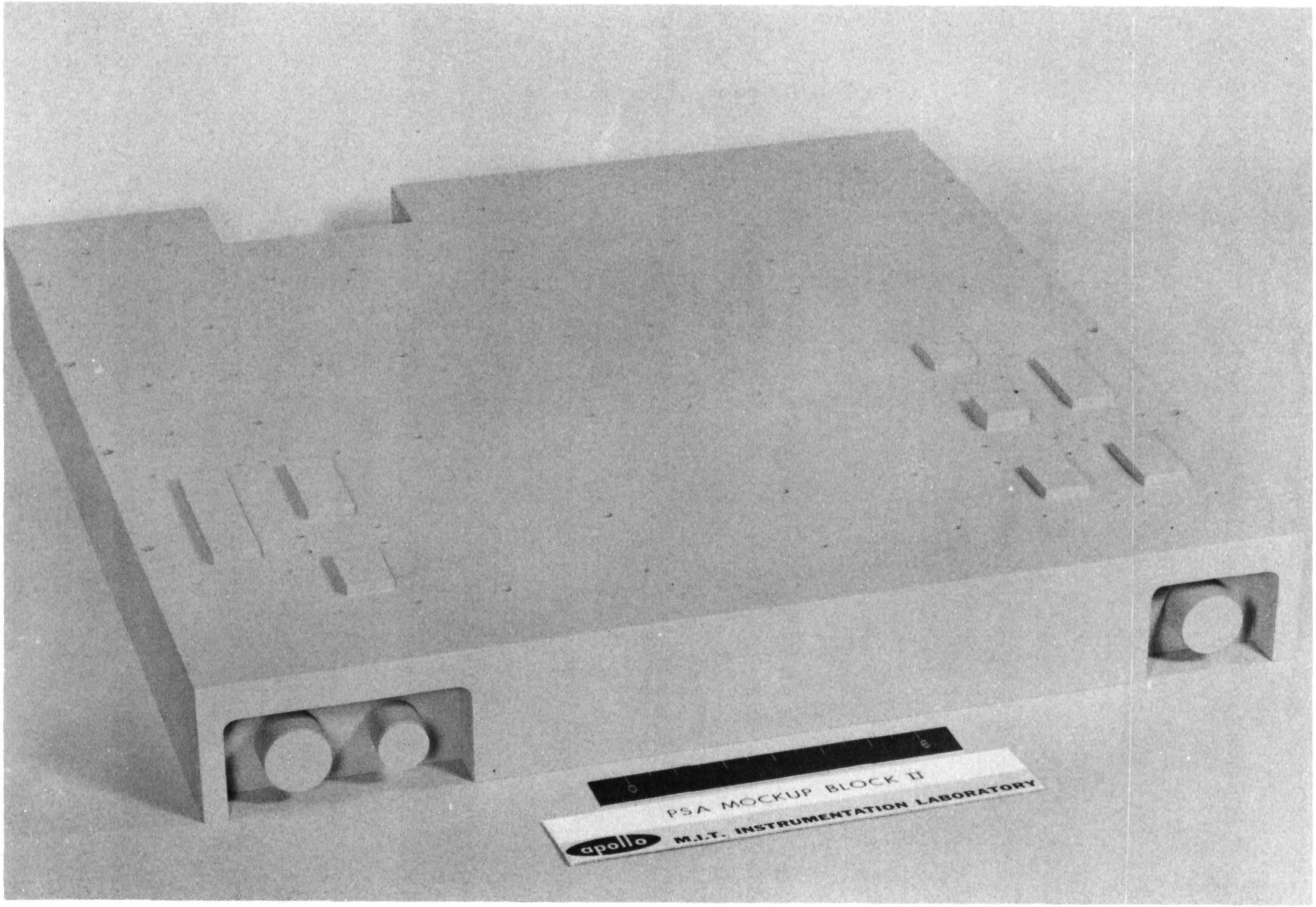

Fig. 28 Power Servo Assembly Mockup, Block II

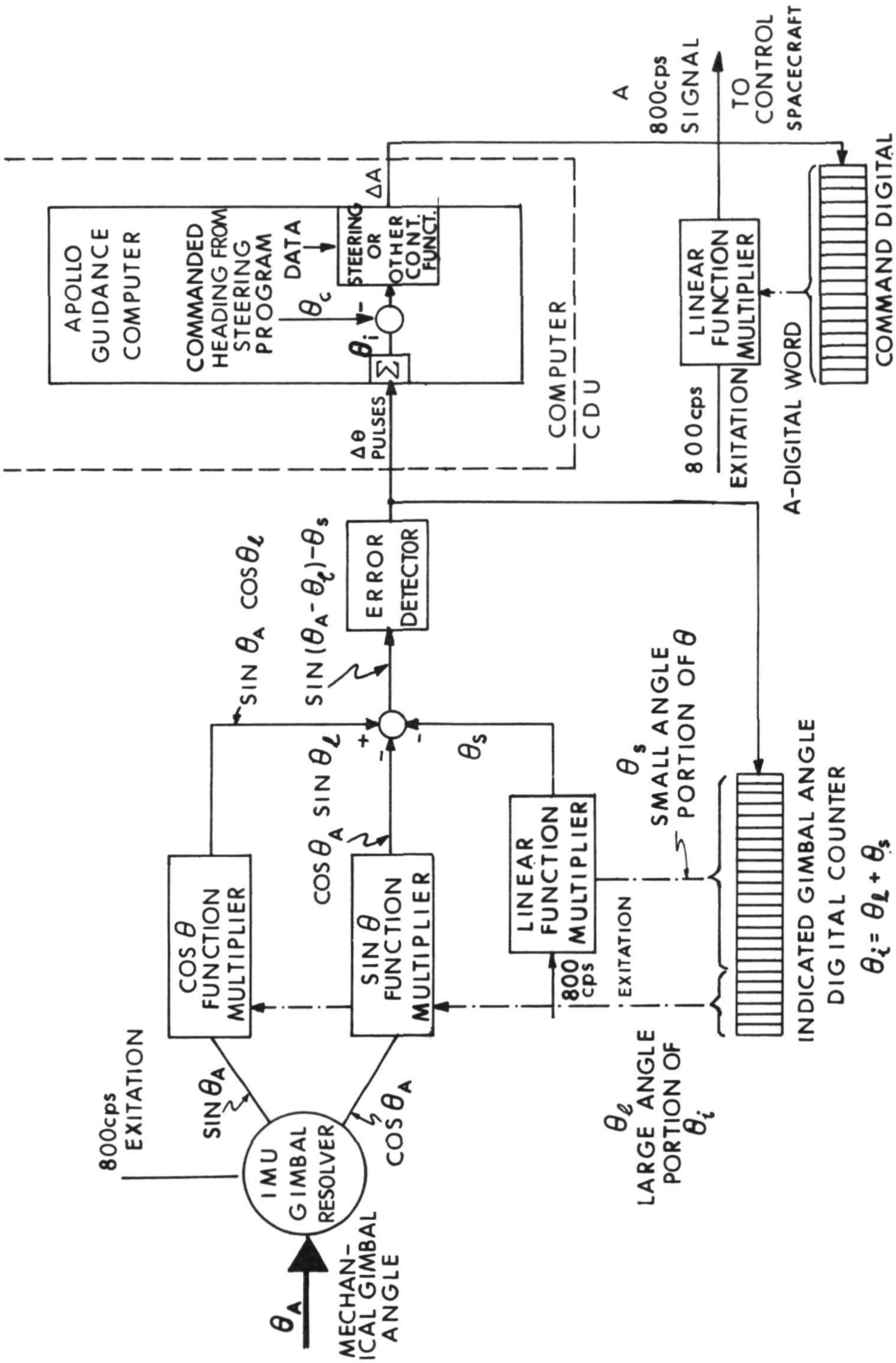

Fig. 29 Electronic Coupling Data Unit Schematic

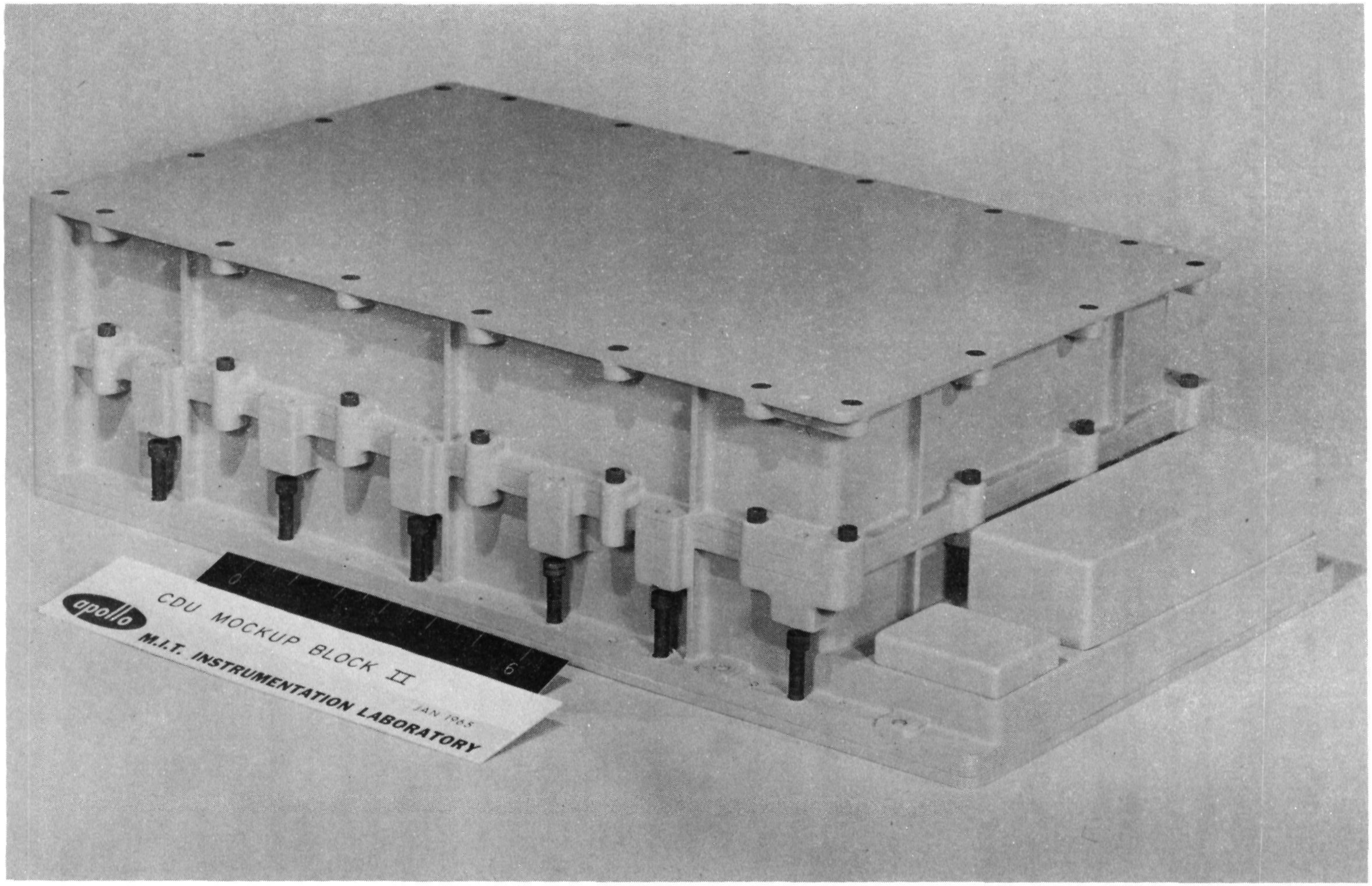

Fig. 30 Electronic Coupling Data Unit Mockup

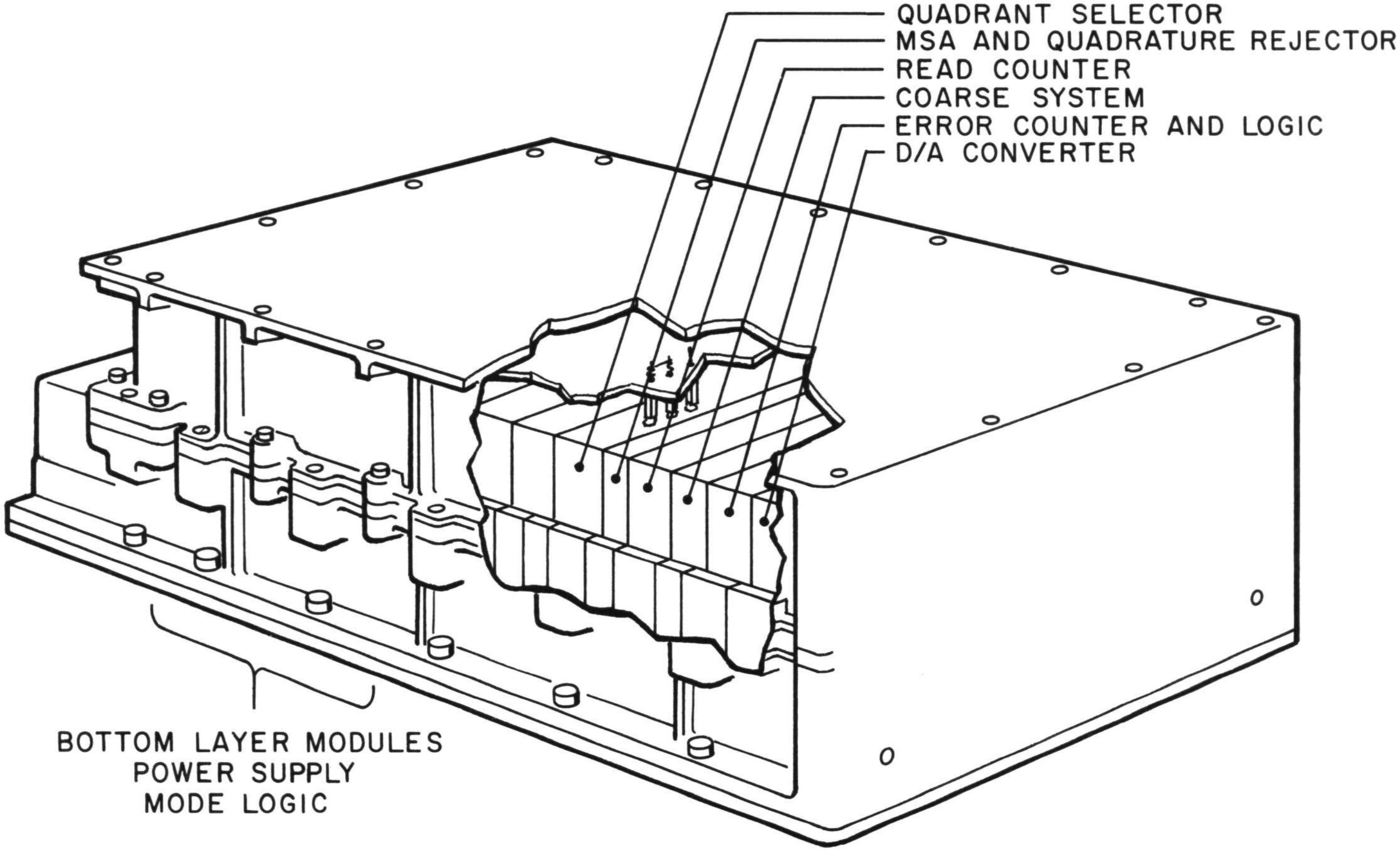

Fig. 31 Coupling Data Unit Cutaway

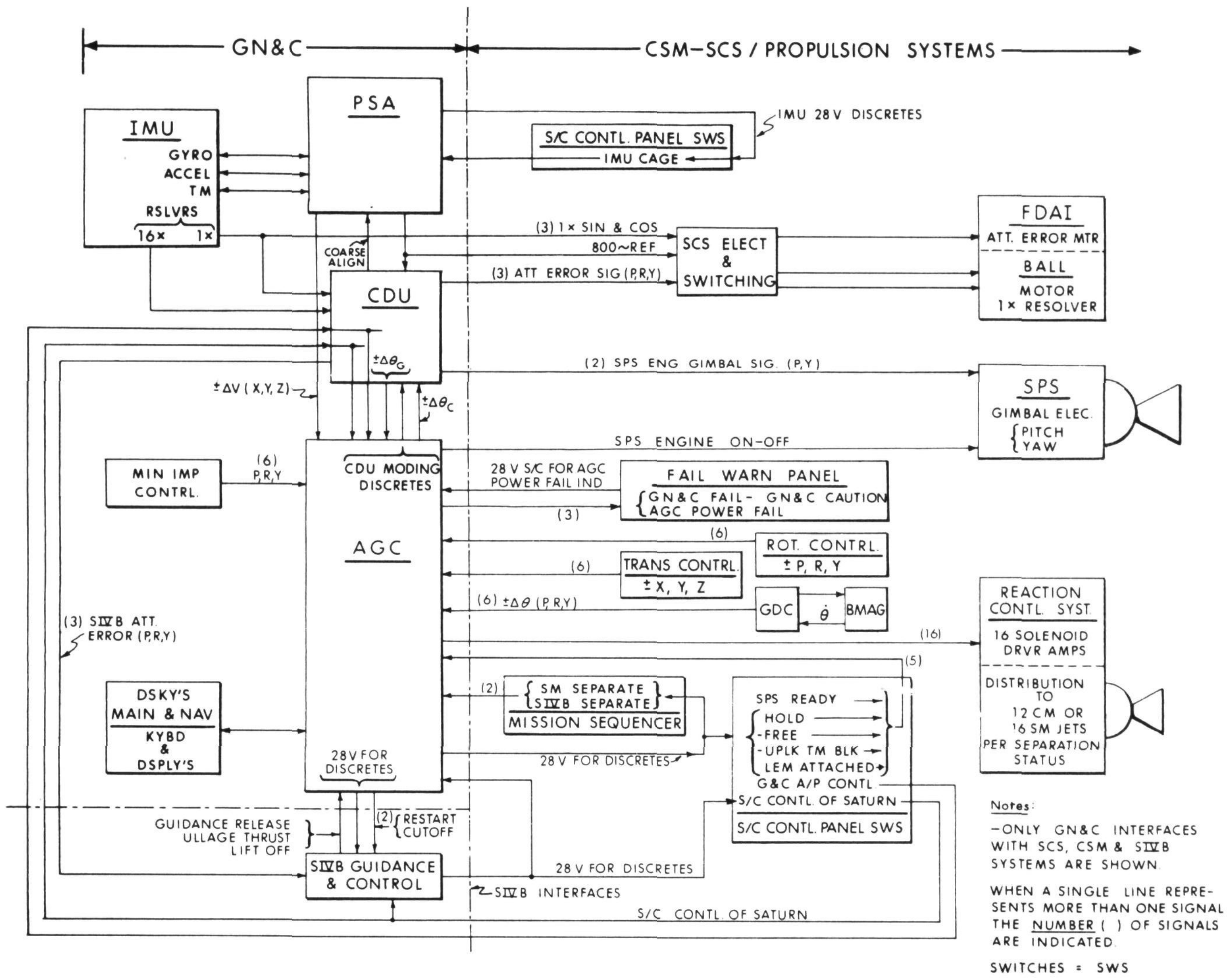

Fig. 32 Digital Autopilot Block Diagram

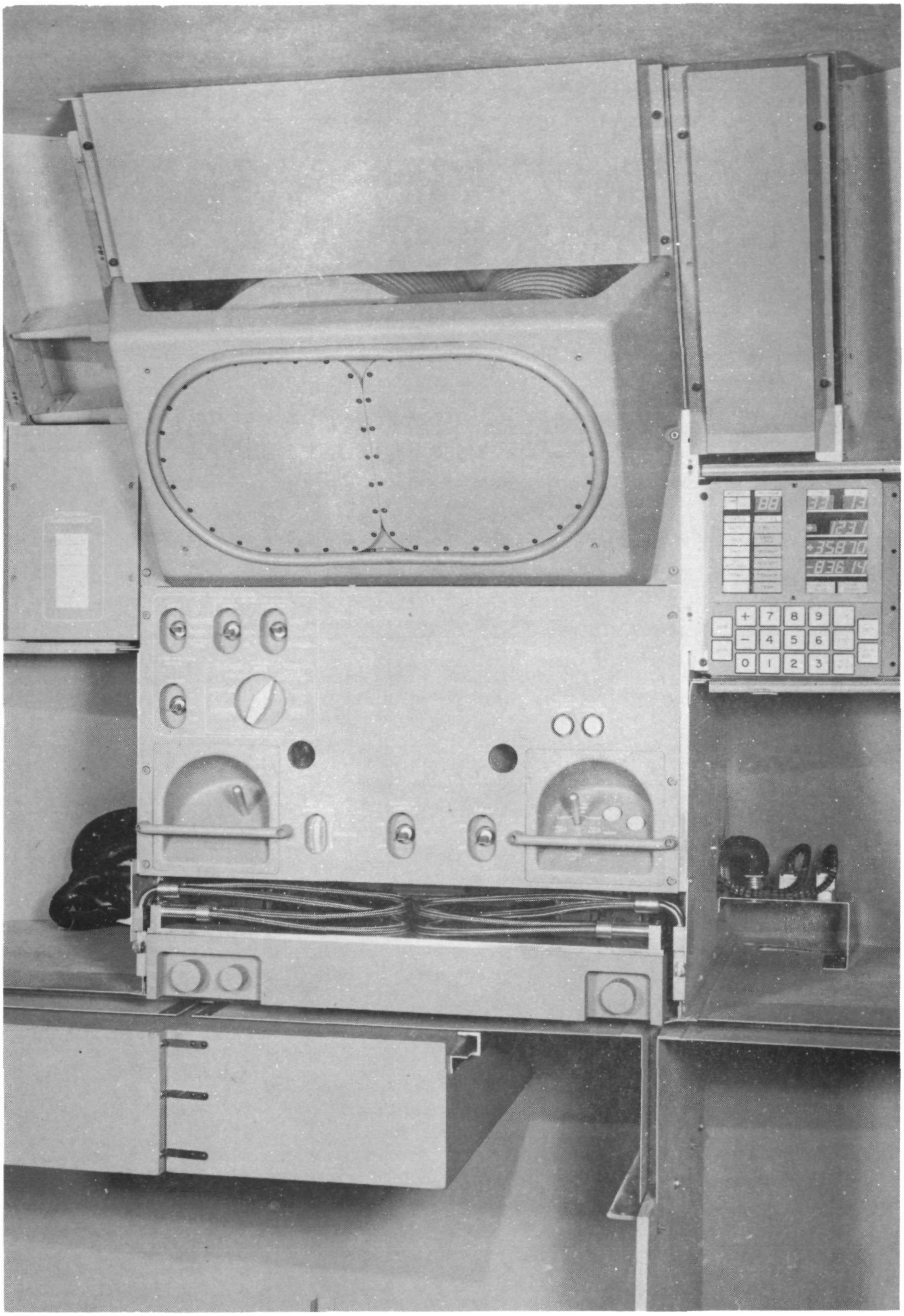

Fig. 33 Command Module Displays and Controls

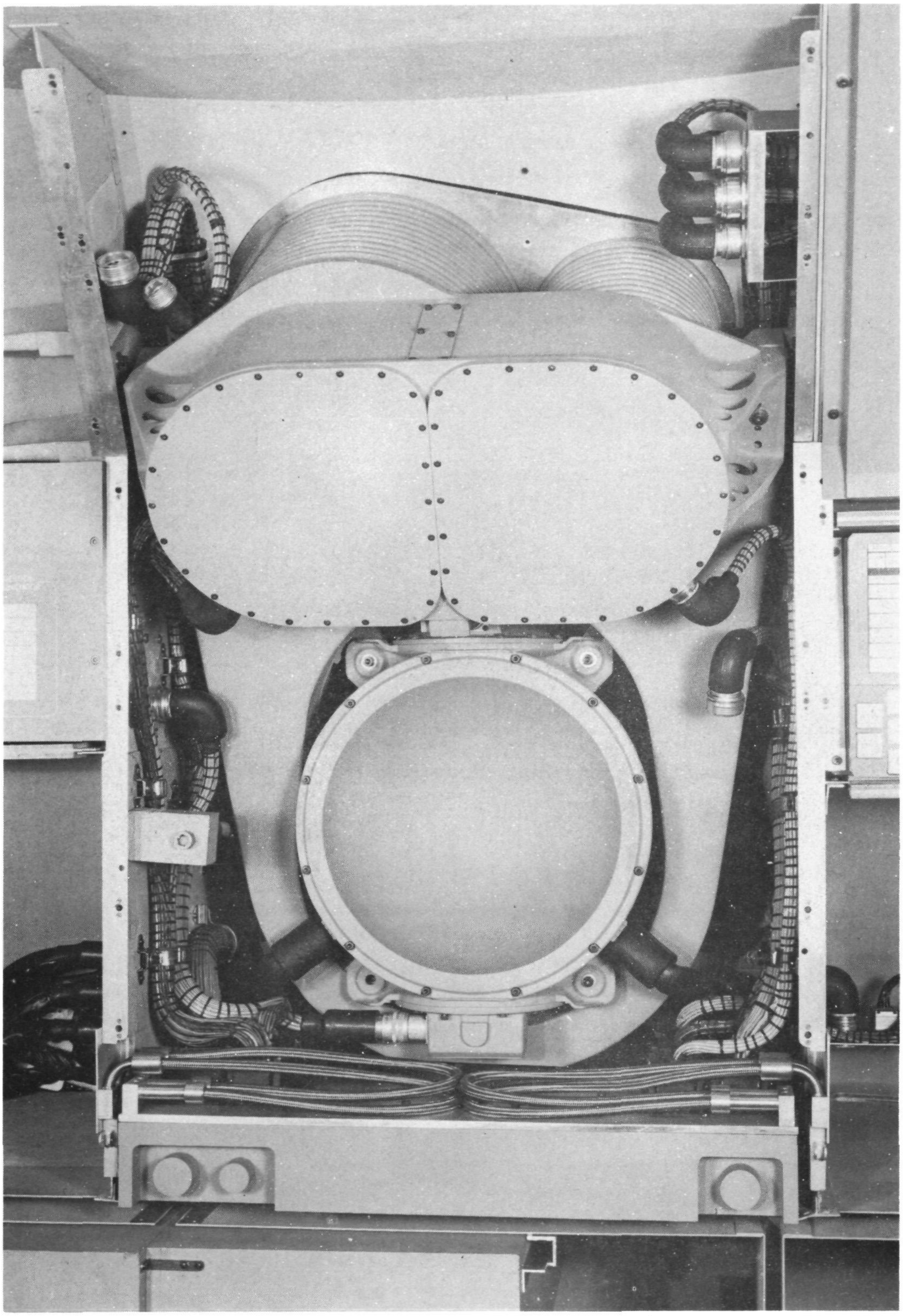

Fig. 34 Command Module Installation Mockup, Block II

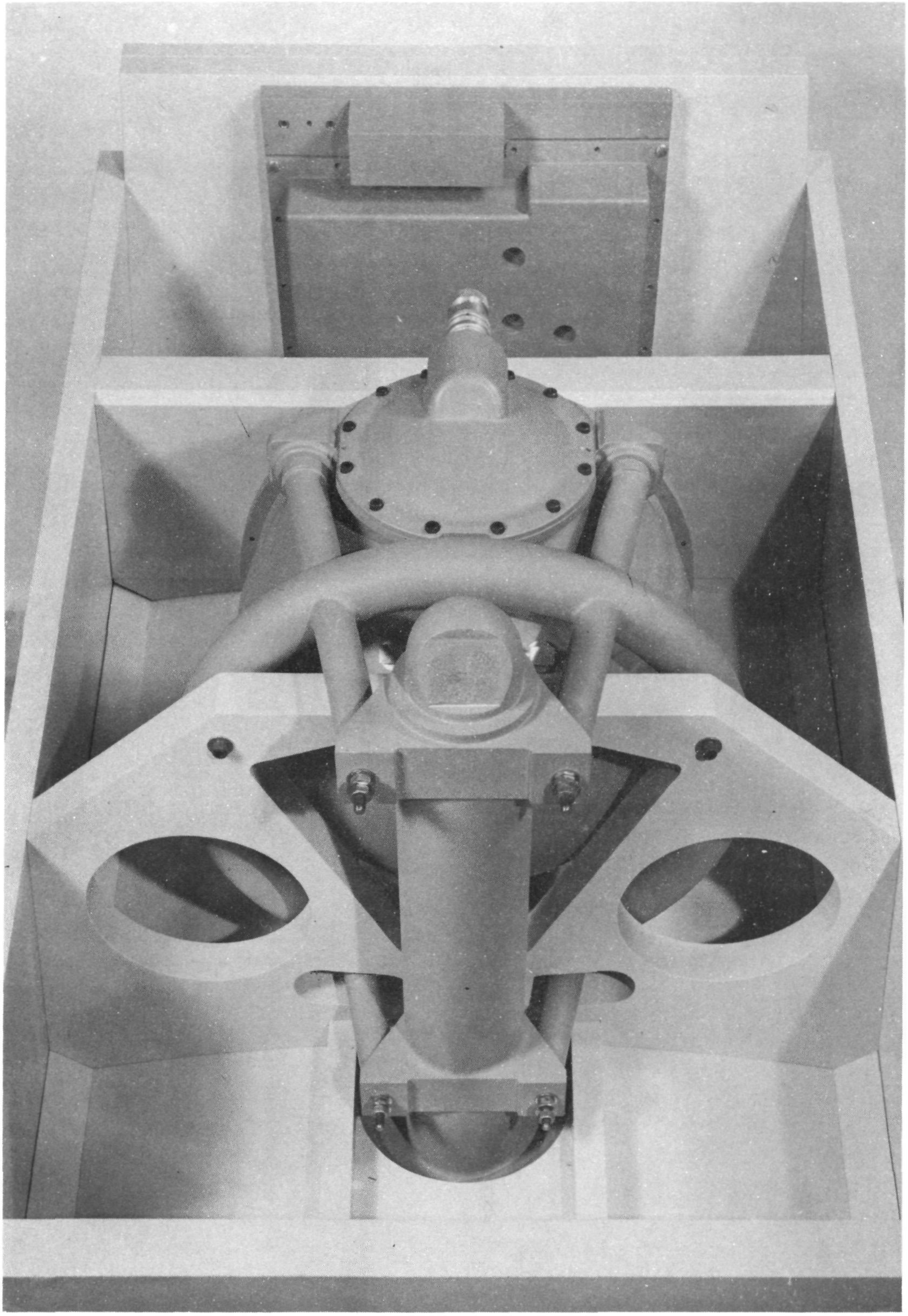

Fig. 35 Lunar Excusion Module Installation, Front

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