R-447

MASSACHUSETTS INSTITUTE OF TECHNOLOGY

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APOLLO

GUIDANCE AND NAVIGATION

[redacted]

(NASA-CR-129903) NAVIGATION FOR THE
APOLLO PROGRAM J.M. Dahlen, et al
(Massachusetts Inst. of Tech.) May 1964
48 p
N73-70035
Unclas
00/99 39686

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NAVIGATION
FOR THE
APOLLO PROGRAM

by

J. M. Dahlen
and
J. L. Nevins

May 1964

MIT INSTRUMENTATION LABORATORY

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Approved: Milton B. Trageser Date: 6/18/64
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Presented at the National Space Meeting of The Institute of Navigation, St. Petersburg, Florida, April 30 - May 1, 1964.

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ACKNOWLEDGMENT

This report was prepared under DSR Project 55-191, sponsored by the Manned Spacecraft Center of the National Aeronautics and Space Administration through Contract NAS 9-153.

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NAVIGATION FOR THE APOLLO PROGRAM

Introduction

It is the intent of this paper to explain the basic navigation concepts and techniques used by the Instrumentation Laboratory in designing the Apollo Guidance and Navigation System. This system has the capability to control the spacecraft path throughout its mission which, for the basic lunar landing mission illustrated in Figure 1, contains fifteen distinct guidance and navigation phases. Also required, but not illustrated, is the capability to guide aborts from all phases prior to trans-earth injection. In order to perform these functions three distinct tasks must be accomplished.

1. Determine position and velocity on present spacecraft orbit.
2. Compute future spacecraft orbit or landing point and the initial conditions for the required maneuver.
3. Control application of thrust or lift so as to achieve the desired new orbit or landing point.

Tasks 1 and 2 are performed periodically during free fall phases- an activity we refer to as navigation. Task 3 is performed continuously during powered maneuvers - an activity we refer to as guidance. Guidance of the Apollo Spacecraft is inertial, i. e. applied force is sensed by accelerometers mounted on a gyroscopically stabilized platform and processed by a computer which generates steering and engine cut-off commands. Position and velocity are at least implicit in the guidance equations. The Lunar Excursion Module G&N system also utilizes radar and astronaut - visual inputs during the final approach to landing and therefore the LEM may be said to use radar-visual-inertial guidance. Also, supplemental data from the ground may be used advantageously by the Command Module G & N system during entry.

It has been a basic requirement that the Apollo G & N system be self-contained. The system is therefore immune from external interferences or jamming. A self-contained system is essential for performance of the guidance function. It is obvious, for example, that the powered lunar orbit insertion and trans-earth injection which take place on the back side of the moon, cannot be controlled from the earth. Trans-lunar injection and entry take place over ground tracks which in the worst cases may only be known several hours in advance. Provision of ground-based guidance facilities to actively control these maneuvers would, therefore, be costly.

The requirement for self-contained navigation capability is equally valid, but not immediately evident. Ground tracking and computation facilities are also capable of determining the spacecraft orbit and computing the initial conditions for maneuvers. Such facilities have demonstrated their navigation capability for Mercury, Ranger, Mariner and a number of earlier space vehicles, and they will undoubtedly be used to maximum advantage during Apollo missions.

Navigation for the Apollo mission is best performed using a combination of airborne and earth-based navigation, because in so doing we maximize:

1. Crew Safety
2. Spacecraft capability to complete mission
3. Mission flexibility

Crew safety is maximized by this approach primarily because it provides redundant sources of navigation during critical mission phases when delays in the acquisition of trajectory data and necessary corrections cannot be tolerated. Such a period is the last several hours before entry when navigation is essential to establish the final velocity correction required to hit the safe entry corridor. The crew is committed to this critical time interval at the time of trans-earth injection when they count on the availability of navigation data exactly X hours after injection cut-off. Another critical phase is the period after trans-earth injection when navigation data are required to establish the first midcourse correction on the return trip. A delay in the acquisition of this data will increase the size of the first correction when only a limited fuel supply remains in the tanks. Still another critical period immediately follows lunar orbit insertion when navigation is required to establish whether or not a safe orbit has been obtained. Airborne equipment is required to accomplish this task because a class of insertion guidance or propulsion malfunctions will cause impact before the spacecraft is visible from the earth.

Crew safety is also enhanced by the Apollo approach because the existence of two sources of navigation data permit a sophisticated system of monitoring guidance and navigation for the detection of subtle performance degradations as well as gross malfunctions.

Spacecraft capability to complete the mission is enhanced by the use of both airborne and ground-based navigation because the two sources of data complement each other to minimize errors and their associated velocity corrections. The airborne equipment provides strong angle data while the ground radars provide strong ranging data.

While the spacecraft has sufficient fuel to complete the mission using only airborne navigation equipment, the supplemental use of ground data reduces the velocity corrections required. The resulting fuel margin can be applied to compensate for excess weight or can be held in reserve to provide for degradation in spacecraft performance.

Finally, mission flexibility is enhanced by the dual approach which can be applied to circumvent constraints which are peculiar to one or the other navigation equipments. For example, the airborne tracking radars permit rendezvous on the back side of the moon which is required for direct, minimum time rendezvous.

In addition to serving the Apollo mission, development of a self-contained system provides a long term benefit by advancing the state-of-the-art in space navigation.

Navigation Sensing

Navigation angle data in cis-lunar space is obtained by a two line-of-sight instrument called a space sextant. This instrument is fundamentally designed to measure the angle between a selected star and an earth or lunar landmark. This choice results in the greatest accuracy obtainable within reasonable weight, power and development time constraints. The astronaut senses both the star and the landmark visually (refer to Figures 2 and 2A) and controls the instrument to track both with the aid of servo drives and spacecraft attitude control.

Additionally, the sextant contains photometric sensors for automatic star tracking and detection of light in the visual band radiated from the atmosphere at the earth's bright horizon. These features illustrated in Figure 3 permit acquisition of navigation data when earth landmarks are obscured by cloud cover or when a fully automatic guidance and navigation capability is desired. Single line-of-sight operation to track stars provides the orientation data required for alignment of the inertial platform.

Detection of visual radiation from the bright horizon as a navigation reference represents a basic advance in the state-of-the-art. Preliminary indications are that the correlation between brightness and altitude, which derives from the fact that density governs the amount of light scattering, is good enough to provide accuracy which approaches that obtainable using landmarks. It provides the basis for design of a fully-automatic instrument. The accuracy of this technique cannot be fully assessed, however, until the instrument is calibrated on the early Apollo missions.

Also under investigation are the techniques of detecting the occultation and refraction of star light by the atmosphere at the earth's horizon. The amount of attenuation and refraction are related to the atmospheric density at the altitude through which the light passes. Refraction would be detected visually (Figure 4) whereas attenuation would be detected by the photometric detector used in the automatic star tracking loop. (Figure 5). In all cases we look into the atmosphere at a high altitude to avoid the anomalies associated with clouds.

Infra-red techniques were investigated also, but were found not competitive with the photometric techniques. In order to obtain the desired accuracy, it is necessary to avoid the anomalies due to clouds and water vapor distribution. One can look into the atmosphere at a low altitude where there is ample IR radiation with a narrow band detector centered on the window in the water vapor emission spectrum, but this requires a very large aperture to get enough signal. If we look at high altitudes with a wide band detector, the signal strength is so low that a large aperture is still required. The net results is that the use of IR techniques would result in a heavier and larger instrument in order to achieve comparable accuracy.

Other self-contained navigation sensors that cannot compete with the space sextant are: radar, which requires excessive power and weight; and inertial components whose errors are excessive for the long duration of coasting flight.

The space sextant is a two line-of-sight instrument shown schematically in Figure 6 designed and used very much like the conventional mariner's sextant. It is operated to superimpose a star on a landmark at which time the angle is read out electronically into the computer. Pure stadiametric measurements are not used because they would complicate the design and because the stadiametric data is implicit in the sequence of star-landmark observations. Figure 7 illustrates the geometry of the navigation fix in space. Measurement of an angle, A, subtended by a star and a landmark locates the spacecraft on a locus which is a cone of semi-vertex angle, A, whose axis is in the direction of the star and whose vertex is centered on the landmark. Measurement of an angle, B, of a star above an illuminated horizon places the spacecraft on a locus which is a cone of semi-vertex angle, B, whose axis is in the direction of the star and which circumscribes the earth or moon. It is seen that three such measurements locate the spacecraft at one or the other of two points. In the Apollo navigation system such a "fix" is never actually made. Actually a sequence of angle measurements is made of which updates the present best estimate of position and velocity in a statistical sense.

In local orbit the star-landmark angle rate of change is too great for measurement by the sextant. In this case, a single line-of-sight, wide field instrument called the Scanning Telescope is used to track landmarks. (Figure 8). The direction of the tracking line with respect to the inertial platform is read into the computer which processes this data to update the local orbit ephemeris. Such a bearing "fix" locates the spacecraft on a line in the direction of the line-of-sight and terminating at the landmark. The Scanning Telescope is also required as a finder for the Sextant.

For rendezvous, navigation sensing is accomplished with a radar on the Lunar Excursion Module tracking a transponder on the mother ship. A back-up and monitor capability will be provided by a radar on the mother ship tracking a LEM transponder. For the rendezvous problem, single line-of-sight optical tracking could be used but radar tracking is better because it provides both range and angle data and is not subject to visibility constraints. Use of optical tracking data alone would force trajectory shaping to provide visibility and would result in fuel expenditures for correction which are comparable to the weight of the radar equipment.

Navigation and Guidance Techniques

Two general ground rules followed during the development of these techniques were:

a. Wherever possible, explicit guidance techniques are used. Reference trajectories are depended on as little as is practical and therefore there is little precomputed data to worry about in the Computer.

b. To minimize the number of special purpose programs that would be useful for only one phase of the mission, a unified approach to the problem has been taken. Thus, large blocks of computer programs can be shared for the many guidance phases.

There are three main programs; they are computation of position and velocity, powered flight steering, and entry.

Position and Velocity

Of all the main programs, this one is the most important and it is the only one that functions throughout the entire mission. Although it has to be done quite accurately, most of the time the computation is done on an open loop basis. The loop is closed whenever measurements are made, but this is a rather infrequent event. So whenever it is desired, the computer can provide knowledge of position and velocity simply by extrapolating information and integrating the equations of motion.

To preserve accuracy, the open loop integration uses the

Enke technique which integrates the deviation between a simple conic trajectory and the perturbation caused by the sun, the moon, and higher order terms of the earth's gravity field.

Closing the loop once a measurement has been made requires a comparison between the external measurement and an on-board prediction of this measurement This estimate of the angle to be measured, $\hat{A}_{SL}$ is computed on the basis of current estimated vehicle position and stored landmark coordinates. The actual angle measured, A_{SL}, is then compared with this estimate to establish the measurement deviation δA_{SL}. A statistical weighting vector, $\underline{W}$, is generated from a priori knowledge of nominal trajectory uncertainties, optical tracking performance, and a geometry vector $\underline{b}$ based on the type of measurement being made. This weighting vector is defined such that a statistically optimum linear estimate of the deviation of the vehicle position $\delta \underline{r}$, and velocity $\delta \underline{v}$, from the estimated orbit or trajectory is obtained when the weighting vector is multiplied by the measurement deviation δA_{SL}. The deviation of position ($\delta \underline{r}$), and velocity ($\delta \underline{v}$), are then added to the vehicle position and velocity estimates respectively to form a new orbit estimate. This procedure is repeated for each navigation measurement until orbital uncertainties are reduced to an acceptable level.

The general procedure shown in Figure 9 is used in all unpowered portions of the CSM and LEM mission phases. Any type of valid tracking data or measurement can be used such as range, range rate, optical or radar tracking angles.

The salient points of this technique are four:

a. This scheme is applicable to all phases of the mission for which there are only field forces. (approximately 99% of the time).

b. No dependence on a reference trajectory.

c. The total navigation program is about 1500 words of program.

d. Measurement data can be accepted from a variety of sources including ground based and vehicle based radar.

Powered Phases

The philosophy adopted for the powered phase is called the "velocity to be gained" concept. (Refer to Figure 10) The velocity to be gained ($\underline{V}_g$) results from a comparison of the actual velocity and the velocity required (assuming it could be attained impulsively). This vector cannot be acquired instantaneously but can only be acquired over a finite period of time. In order to guide in an efficient manner, a steering law is used which forces the $\underline{V}_g$ vector to zero without rotation. This steering law lines up the derivative of $\underline{V}_g$ with the velocity to be gained vector itself. The technique produces a closed form calculation.

Entry Phase

During the entry, a reference trajectory is used. However, rather than using stored reference trajectories, a nominal trajectory is computed on board just prior to the start of the actual entry phase. At that time, the range to the desired landing site as well as the particular entry conditions should be well known. Then by iteration techniques, a reference trajectory that satisfies these boundary conditions is generated. Thus, the need for stored nominals to span the entire spectrum is avoided.

The steering law used relates velocity, altitude rate, and drag acceleration to the computed nominal trajectory.

Apollo Guidance Computer

The AGC is the central processor for the guidance and navigation system. It is also the clock or basic time and frequency reference for the spacecraft. Figure 11 shows the interrelationship of the AGC to the various sensors and to the spacecraft control and propulsion system.

The AGC can communicate with the sextant and scanning telescope via the Coupling Data Units (CDUs). It can also communicate with the displays and it can receive inputs from the astronauts via the keyboard. In addition, the AGC can count pulses from the accelerometers, read gimbal angles and read and control radar angles. The AGC can send information to earth via telemetry and receive telemetry information on an uplink. During guidance modes of operation the AGC can control and stabilize the spacecraft and start and stop the engines.

The AGC can be classified as a general purpose, parallel operation, fixed point digital machine having a large fixed wired core rope memory. It has an additional erasable ferrite core memory sufficient to meet the operational requirements of all mission phases. The basic word length of the machine is only 16 bits and the basic operation code is limited to eleven instructions. These apparent limitations are gotten around by having an interpreter routine. This interpreter first expands the basic operation code from 11 basic instructions to 72. This expanded set includes a variety of double precision operations, a small number of triple precision operations, and a set of double precision vector operations. Second, the interpreter is convenient for the programmer and it gives about a three to one saving in the fixed program storage required.

Another feature of the general programs is the ability to handle the many simultaneous jobs required for a mission as complex as this one. Among the simultaneous requirements are the asynchronous demands to display information, change of program, accept telemetry, uplink data, etc.; while at the same time performing real time integration, command engine firings, etc. This riddle is solved by two programs called executive and wait list. These programs have the capability of stacking up to seven program requests (in addition to the one being carried out) according to their assigned priorities, and of executing these jobs in the order of priority. This program priority is predetermined and included with the writing of the program. The executive programs are also able to stack up to six Tasks (these are time dependent operations to be initiated within the next two minutes) according to the time at which they have to be initiated, and of initiating these Tasks at those times. At the time of initiation, a Task causes the interruption of a less urgent Job and the execution of that Job. Thereafter, another Task, or less urgent Job, or another Job of higher priority can be executed. The Task is by definition of higher priority than a Job and thus causes an interruption to the Job to enable its execution. A Task may be repeated at some later time by making a request to the WAIT LIST during the interruption.

Man-Machine Interfaces

The usual discussions concerning the man-machine interface can be broken down into two categories; unfortunately, both cases usually represent extreme points of view. One point of view, illustrated by Figure 12, is the "fully automatic" system where the astronaut, wrapped in a life maintaining cocoon, is delivered to the lunar surface. The only real problem here is keeping him entertained during the mission. The other point of view, illustrated by Figure 13, is the "fully manual" system where the astronauts are given a rocket, a big window, a control stick, and appropriate charts and tables. This technique is certainly feasible in infinite energy type of vehicles, (an airplane with inflight fueling certainly falls within this classification) but becomes questionable for finite energy vehicles such as Apollo where highly accurate and complex navigation systems are needed to determine the most efficient path, or orbit, to the moon and back.

Instead of the two extremes quoted above, we would like to substitute a third category. This third category could be called "manually aided" systems and would combine the best features of both the man and the machine.

In order to illustrate this point of view, Figure 14 shows the functional relationship of the man to the spacecraft for a typical midcourse star-landmark angle measurement. For this task, the following things are expected of the man.

a. Acquisition and identification of a particular star and landmark. To do this, he must be able to maneuver the spacecraft via the control system. Also he must perform the pattern recognition problem of associating the desired star and landmark patterns from maps and charts to the real world beyond his optics.

b. He must be able to operate the displays and controls associated with the optics to position the desired landmark into the sextant field of view.

c. He performs the superposition of the star on top of the landmark, to the accuracy needed, and "marks" this event to the computer which notes the time of the mark and the angle.

Thus, this task requires man to do tasks which are not easily instrumented and fairly simple routine mechanical tasks such as pointing the optics, which could be instrumented by adding additional programs to the AGC.

Figure 15 illustrates the functional relationship of the man to the equipment for a manually aided star-horizon angle measurement. For this task, the following tasks are expected of the man.

a. Acquisition and identification of the star and proper horizon.

b. Establish the proper geometrical relationship of the star to the horizon.

c. Observe that the automatic star tracker locks on to star and that the AGC receives the automatic mark from the horizon photometer.

This task reduces the number of purely mechanical tasks, lets man perform the tasks for which he is unique; and allows the equipment to perform a measurement which, if the man were to do, would require additional electronics and indicators. The additional equipment would then allow man to perform the simple task of noting when the brightness displayed by an indicator passed through a certain level.

In summary then, manually aided systems make maximum use of the unique but distinctive abilities of man and equipment. This combination, we feel, minimizes the weight and complexity of the equipment and maximizes the reliability.

Another facet of this discussion is the question of control or sequence of operations. Here again, man possesses unique abilities in assessing the proper operation of his equipment and the optimum course of action. Again, the equipment can aid the man by doing a lot of routine sequencing associated with the many spacecraft tasks. At least it could check the sequencing to make sure that it had been performed and that it was done according to the check list.

On this level, the man and machine think exactly alike. They each need a predetermined check list, or logical path, and then a display, or signal, in order to confirm the event. If both perform the total sequence, the overall mission reliability goes up. At a minimum it allows man to sit back and modify the sequence, as necessary, to meet the myriad of possible contingencies. Only man is capable of executing the judgement necessary to perform a successful mission in the presence of unexpected and unplanned for difficulties.

Equipment Description

To sum up, navigation in deep space requires three things.

a. Optics to make sightings
b. A data processor
c. Guidance which requires:
    1. Gyros for attitude reference
    2. Specific force instruments for measuring non-field forces.
    3. Optics for aligning the gyros

Of course we require engines for making velocity changes and a vehicle stabilization system to neutralize vehicle dynamics. For rendezvous maneuvers, we also need radar in order to get range, range rate, and line-of-sight information.

The Apollo guidance and navigation system consists of three main units as follows.

a. The Apollo Guidance Computer - AGC
b. The Inertial Measurement Unit - IMU
c. The Optics
    Space Sextant - SXT
    Scanning Telescope - SCT

In support of these main units are the following.

d. The Coupling Data Unit - CDU
e. The Power and Servo Assembly - PSA
f. The Display and Controls - D & C
g. The Navigation Base

The AGC was described previously and its interrelationship to the system as a whole was shown in Figure 11.

IMU - Inertial Measurement Unit

The IMU is the primary inertial sensing element. It consists of three gyros and three accelerometers mounted on the innermost member of a three degree-of-freedom gimbal structure. Angular orientation of this inner platform is obtained from resolvers mounted on the gimbals. The information is then transmitted to the spacecraft attitude indicator and to the AGC via the CDUs. Non-field forces acting on the vehicle are sensed by the accelerometers which produce signals representing incremental change in vehicle velocity. These ΔV's are transmitted directly to the AGC.

CDU - Coupling Data Unit

The coupling data units are used to transfer angular information between the guidance computer, the IMU, the optics, the rendezvous radar, and the vehicle stabilization and control system. The CDU is essentially an analog-digital conversion device. There are three CDUs for the IMU and two CDUs for the optics and radar.

Optics

There are two optical units, the scanning telescope (SCT) and the sextant (SXT). These two units are rigidly mounted and aligned to the same mounting structure as the IMU. This mounting structure is called the navigation base.

The SCT is a single line-of-sight, wide angle, unity power instrument used for acquisition and general viewing of stars and earth or moon based landmarks.

The SXT is a two line-of-sight, narrow field of view, high power instrument used for making precise midcourse sightings and for aligning the IMU during the mission.

PSA - Power and Servo Assembly

The PSA is a support item and is used in all operations involving the system. It provides various levels and kinds of power to the rest of the system. In addition, it serves as a location for the support electronics for the system such as the servo control amplifiers for the IMU and optics drives.

The equipment is mounted in the spacecraft, as shown in Figure 16, on what is known as the Lower Equipment Bay (LEB). As mentioned previously, the IMU and optics are rigidly mounted to the navigation base. This assembly is located behind the G & N display panels. Figures 14 and 15 show schematically this location whereas Figures 17, 18, and 19 show an actual system undergoing functional testing at the Instrumentation Lab.

D & C - Displays and Controls

Figure 20 is a photograph of the D & C showing its location in the LEB of the spacecraft. Figure 21 is a close-up of the same mockup.

At the top center is a map and data viewer (M & DV). This unit is a small film projector using film cartridges which can be driven either manually or with a motor. Star charts, maps, operating procedures, both normal and emergency, and general information are stored in the film cartridges associated with this unit. Below the M & DV is the optics panel with its attached eyepieces. The SXT is on the left and the SCT is on the right. Below the optics panel is the main G & N display panel. This panel called the I & C (indicator and control) panel contains the controls and displays for the optics modes, the IMU temperature control modes, the M & DV controls, the optics controller, and the vehicle attitude impulse controller. The attitude impulse controller is used to make small changes in vehicle rate during optical measurements. Just below this panel is the PSA and below that the AGC.

To the left of the M & DV is the IMU control panel which controls and displays the IMU modes. Under the IMU control panel are the displays associated with the CDUs.

To the right of the M & DV is the display and keyboard for the computer. A similar display unit, with a rectangular mounting configuration, is part of the displays on the main display panel. The AGC displays on the main display panel are used during high g maneuvers (boost and entry) and for all thrust maneuvers. The main display panel is shown in Figure 16.

Summary

Figures 22 and 23 summarize the various navigation and guidance activities required for a complete lunar mission including landing on the moon.

The system described uses one computational technique to determine position and velocity; but this technique accepts, with equal ease, optical landmark tracking data in close orbits, optical angle measurements in midcourse, and radar data both on board and ground.

Additionally, the on-board capability also reaps a long-term benefit beyond the immediate goal of landing men on the moon. The development of this equipment represents a substantial advance in the state-of-the-art in space navigation. To our knowledge, the Apollo G & N system is without precedent - it provides the crew with a total capability independent of reference paths or cooperation from the earth. It is our belief that as spacecrafts grow in payload and range capability, greater reliance will be made on self-contained guidance and navigation equipment, while reasons of economy and convenience will dictate that earth-based facilities be used for monitor, back-up and near-earth operations.

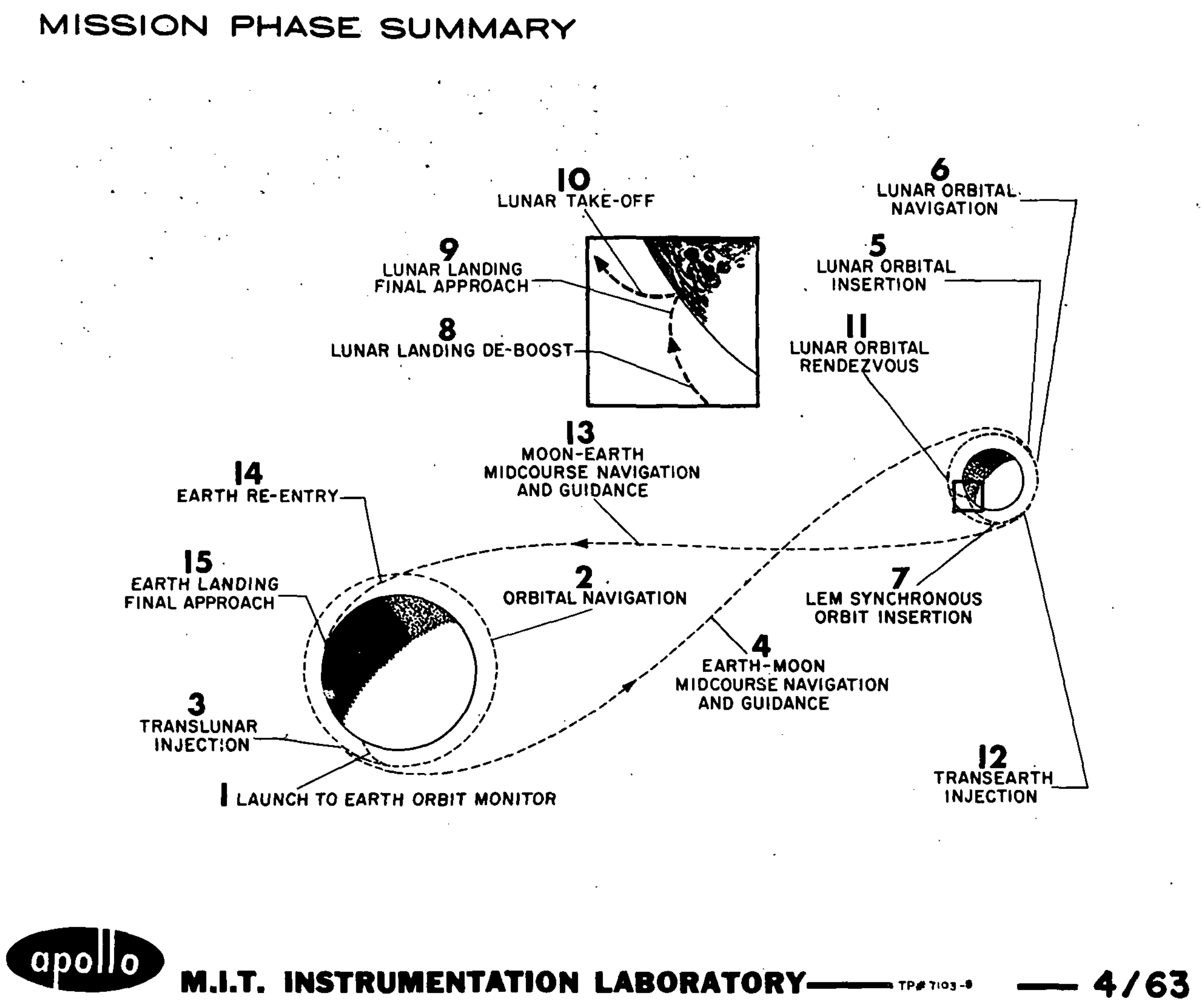


Fig. 1 Mission phase summary.

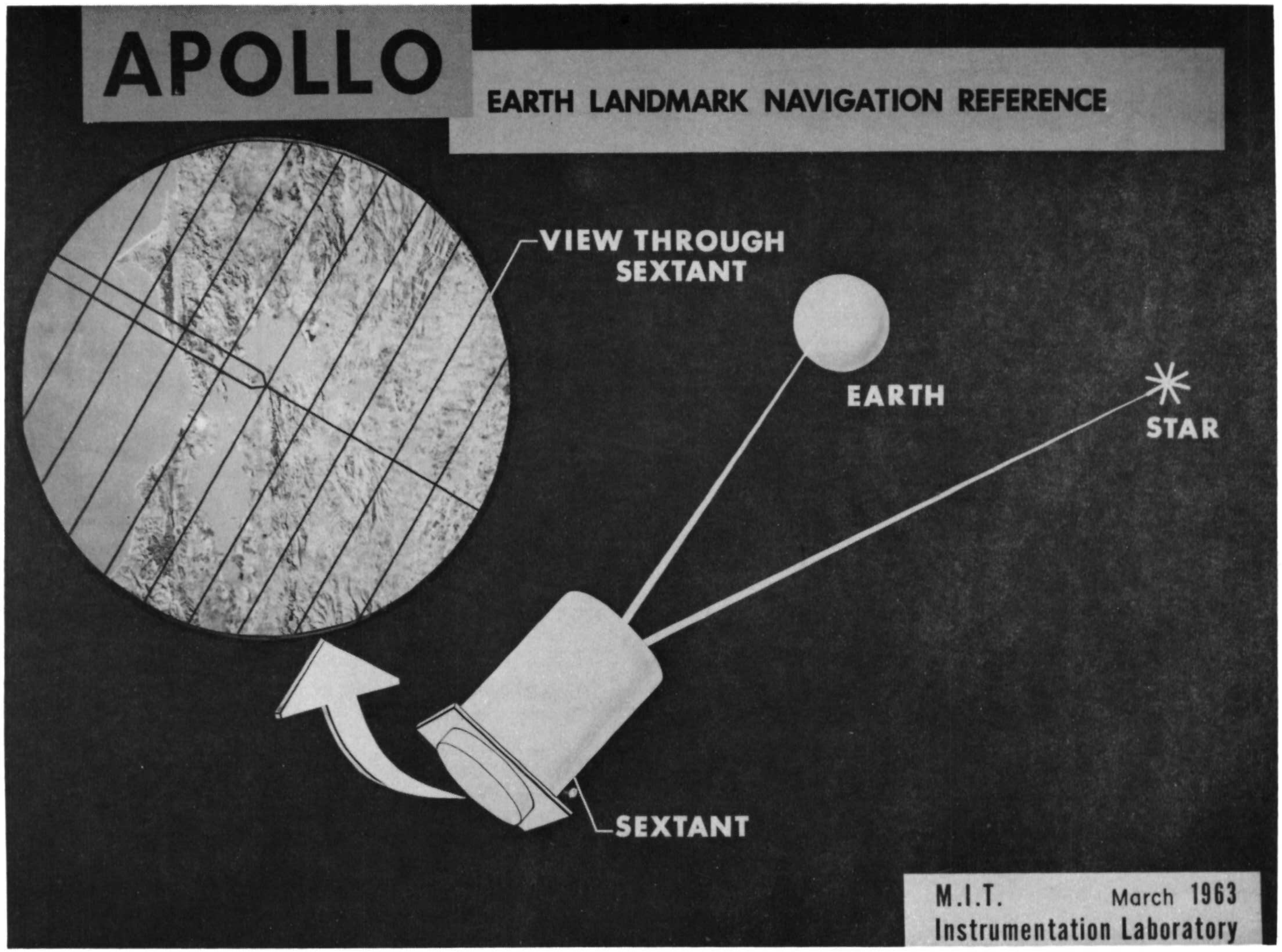


Fig. 2 Earth landmark navigation reference.

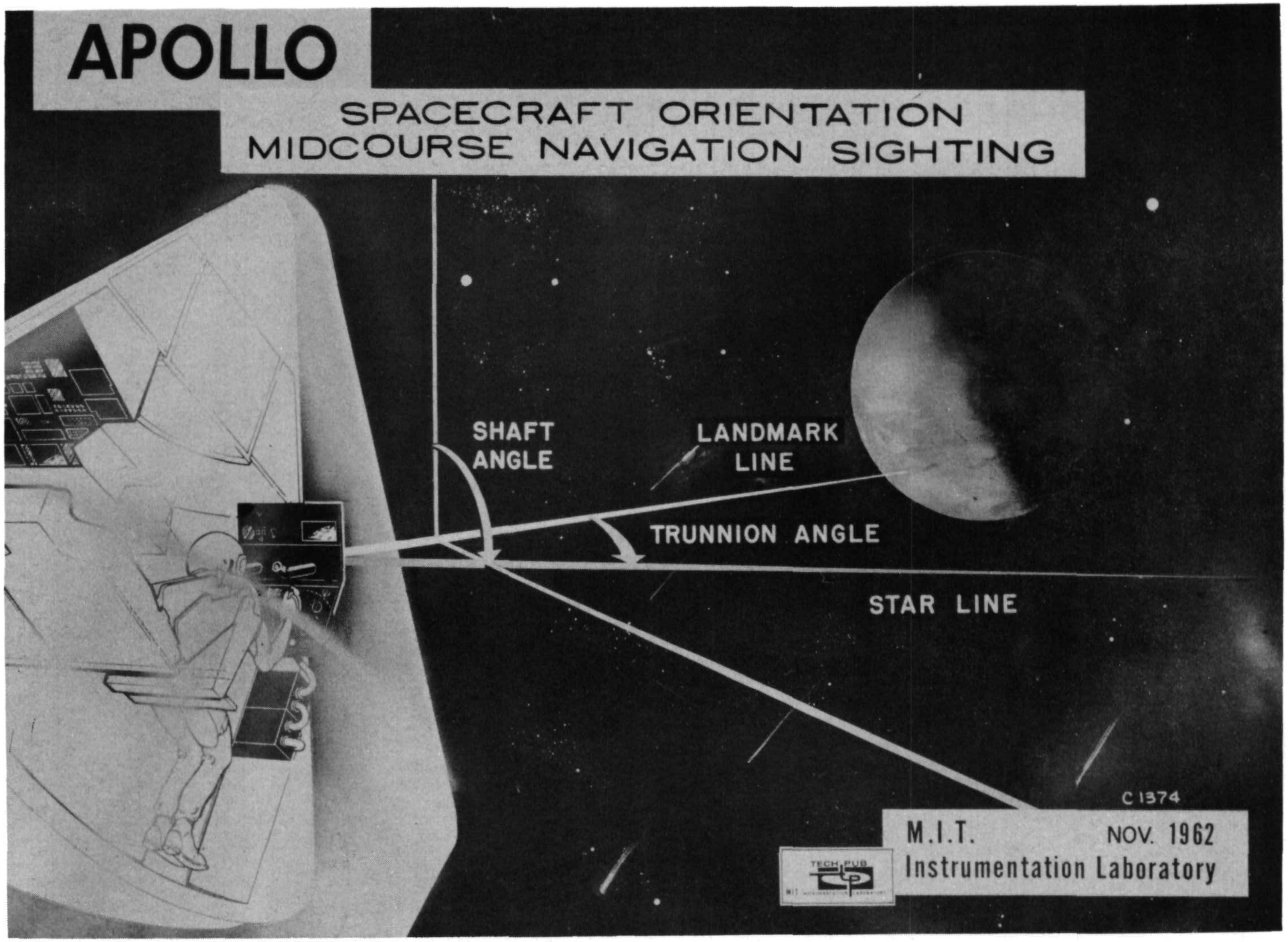


Fig. 2A Spacecraft orientation midcourse navigation sighting.

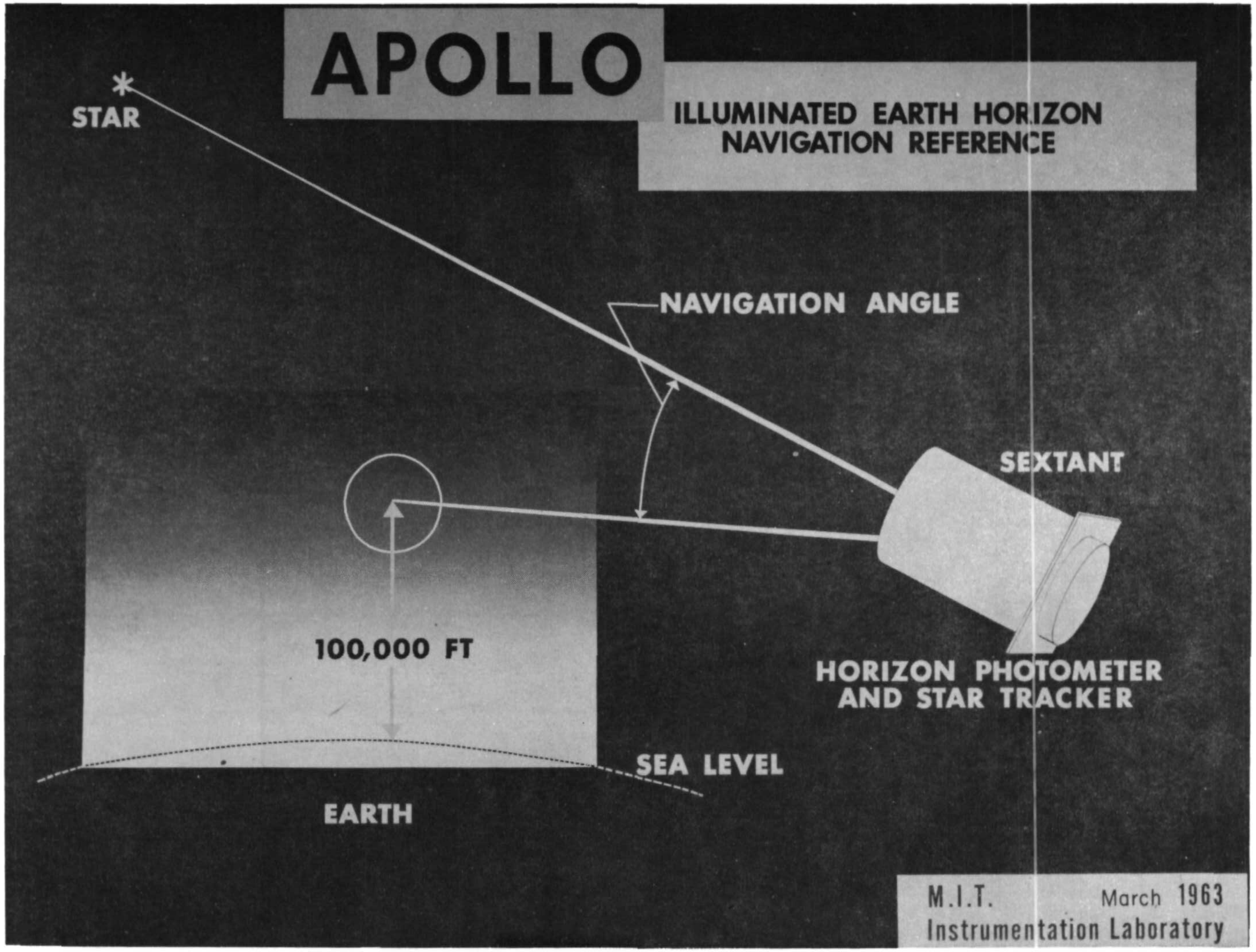


Fig. 3 Illuminated earth horizon navigation reference.

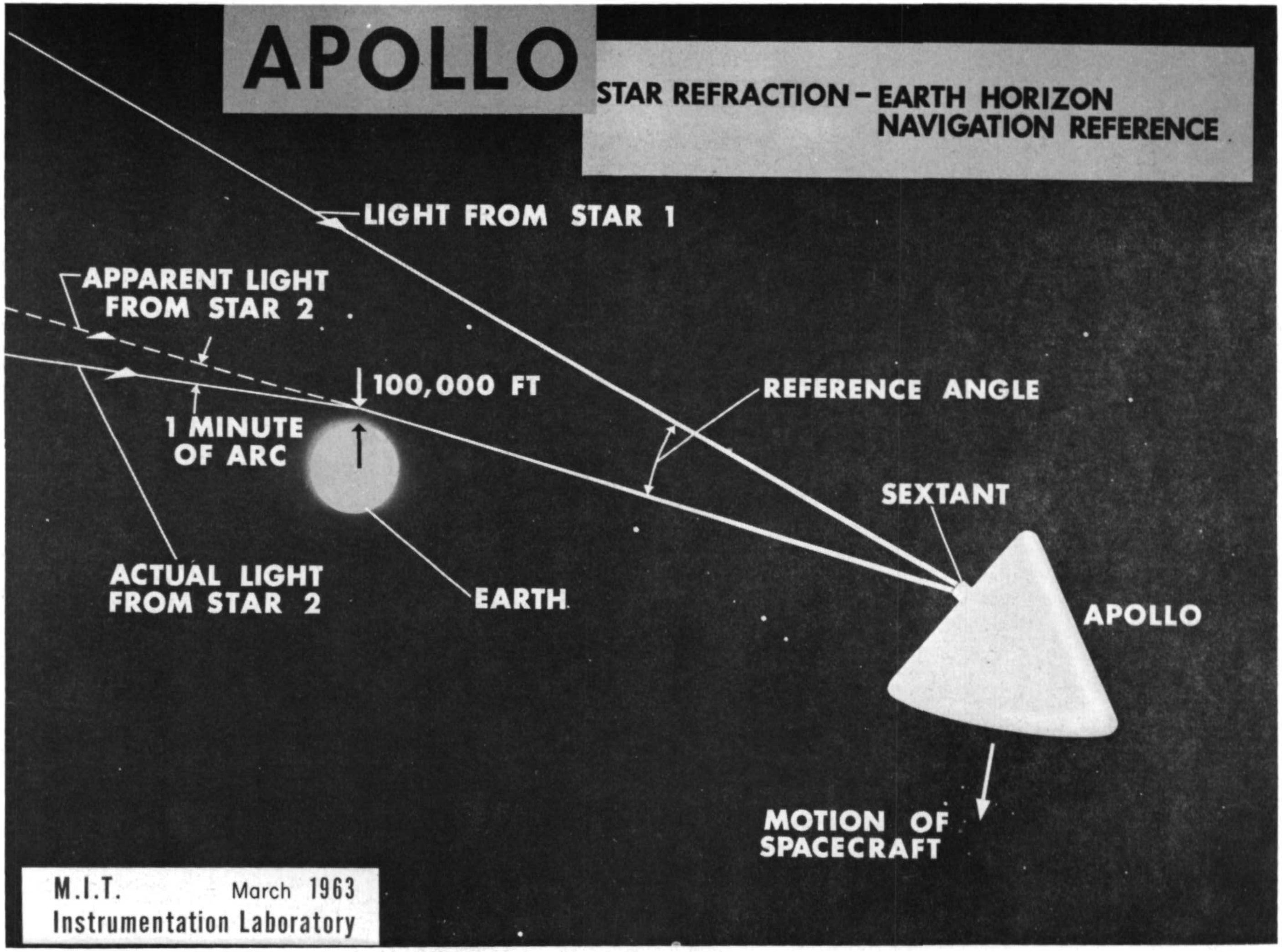


Fig. 4 Star refraction - earth horizon navigation reference.

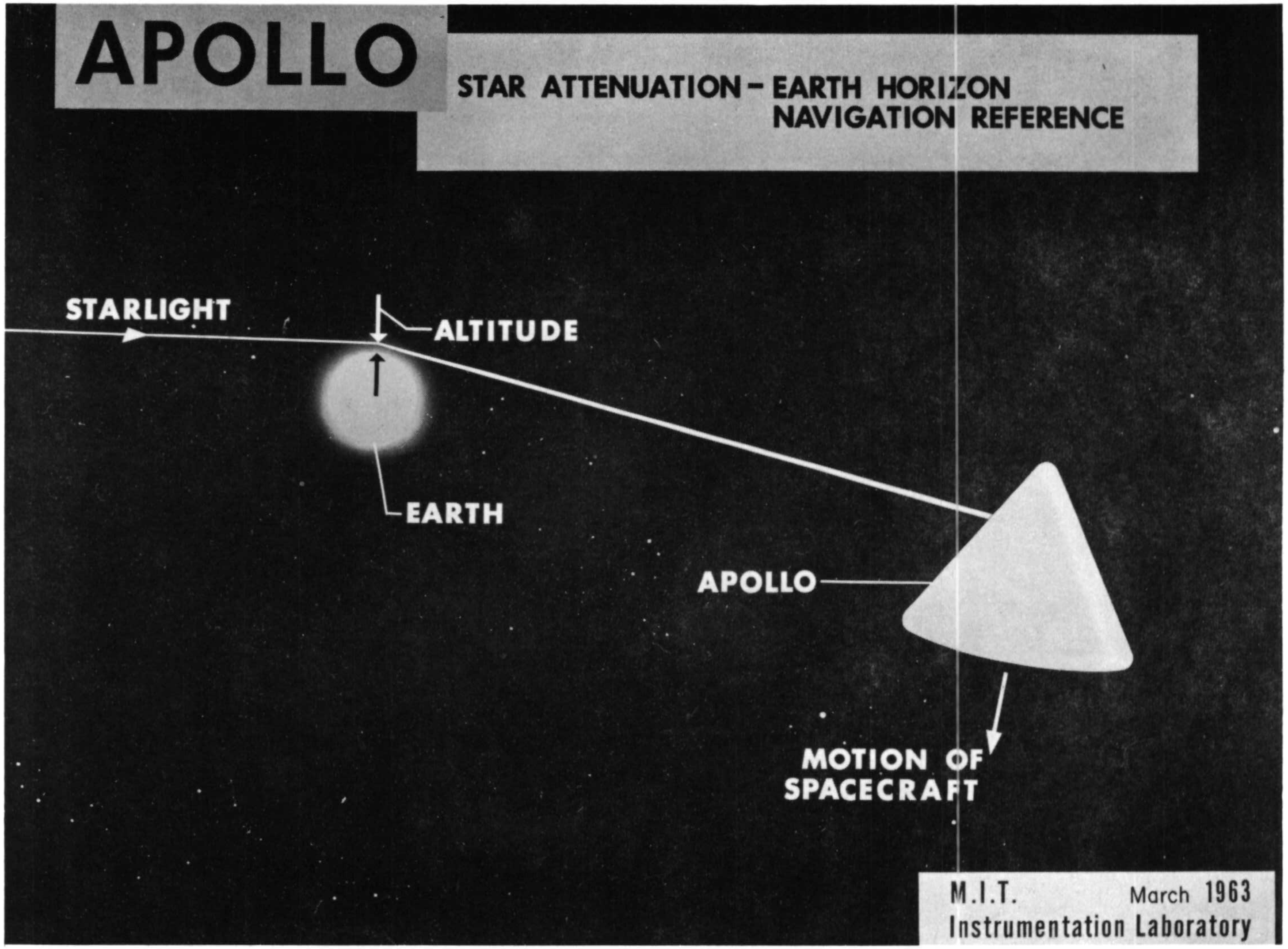


Fig. 5 Star attenuation - earth horizon navigation reference.

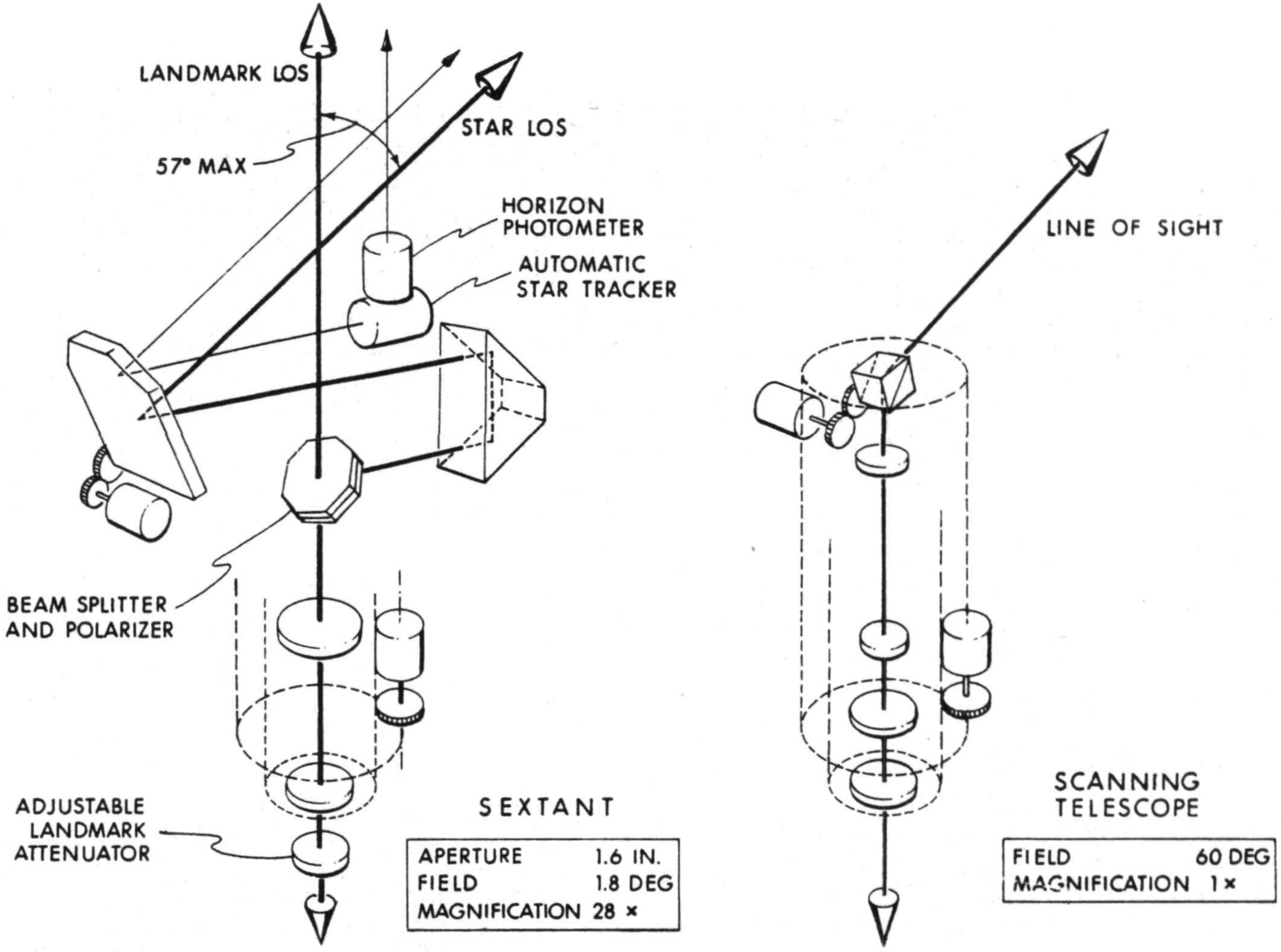


Fig. 6 Optical schematics.

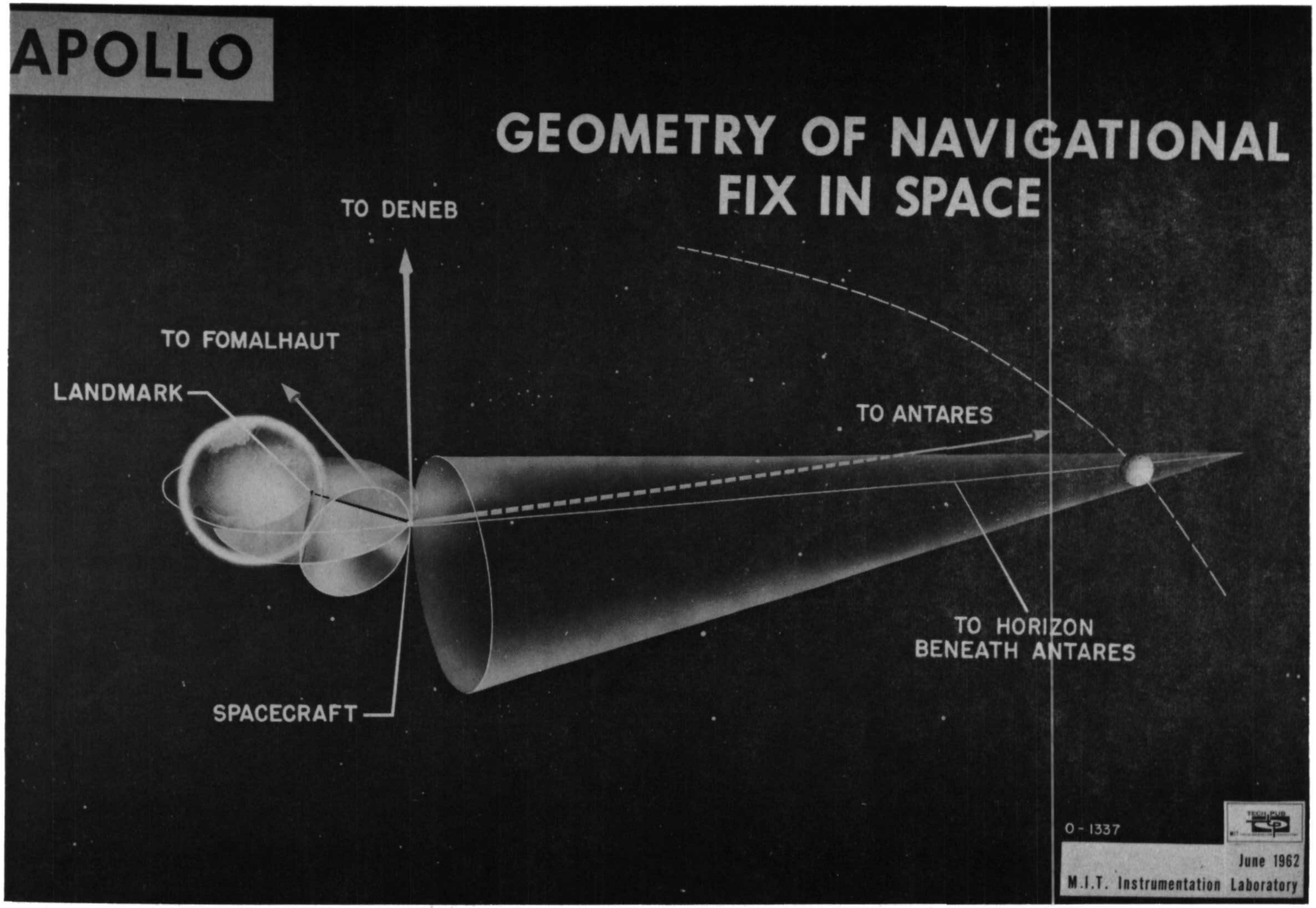


Fig. 7 Geometry of navigational fix in space.

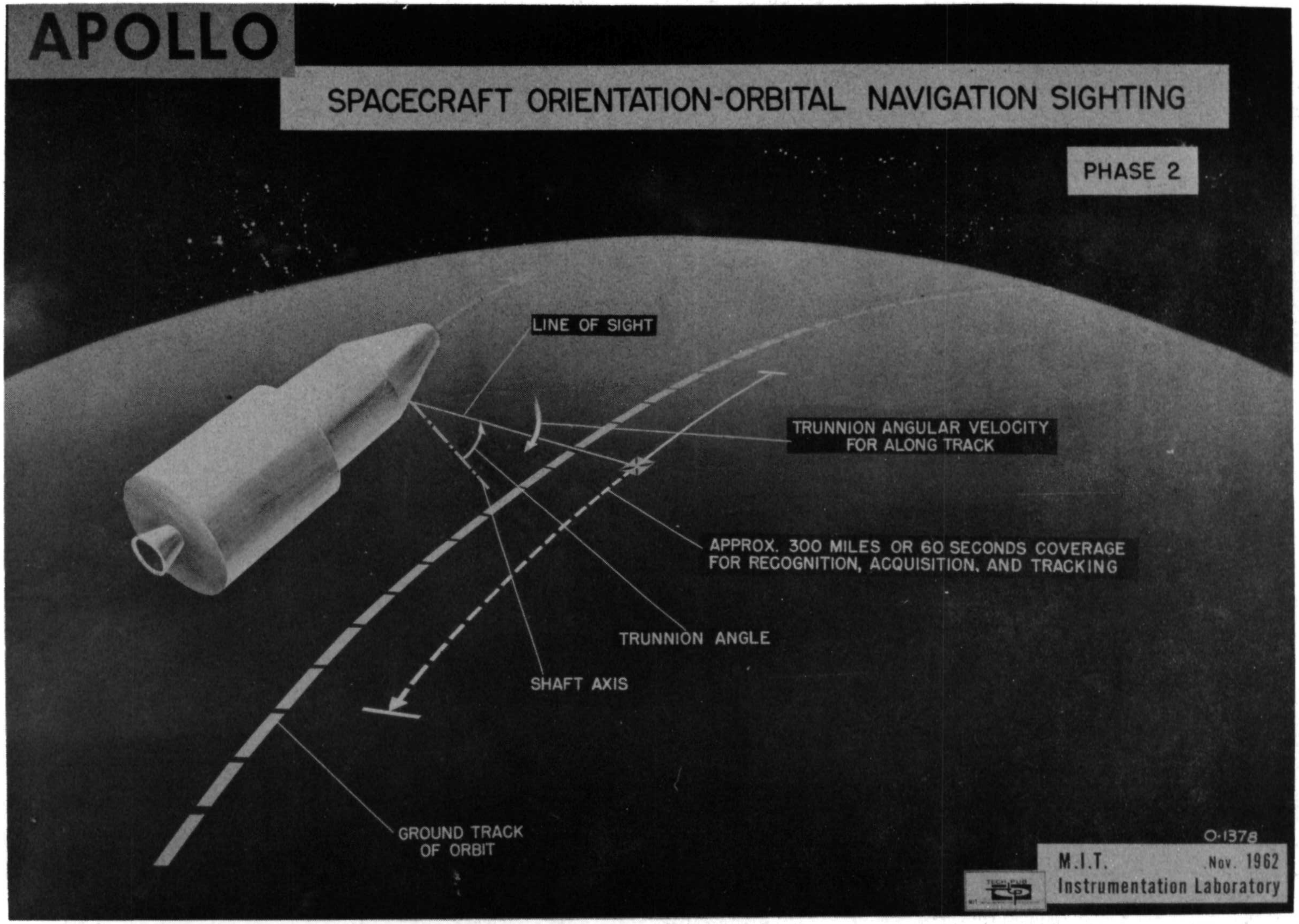
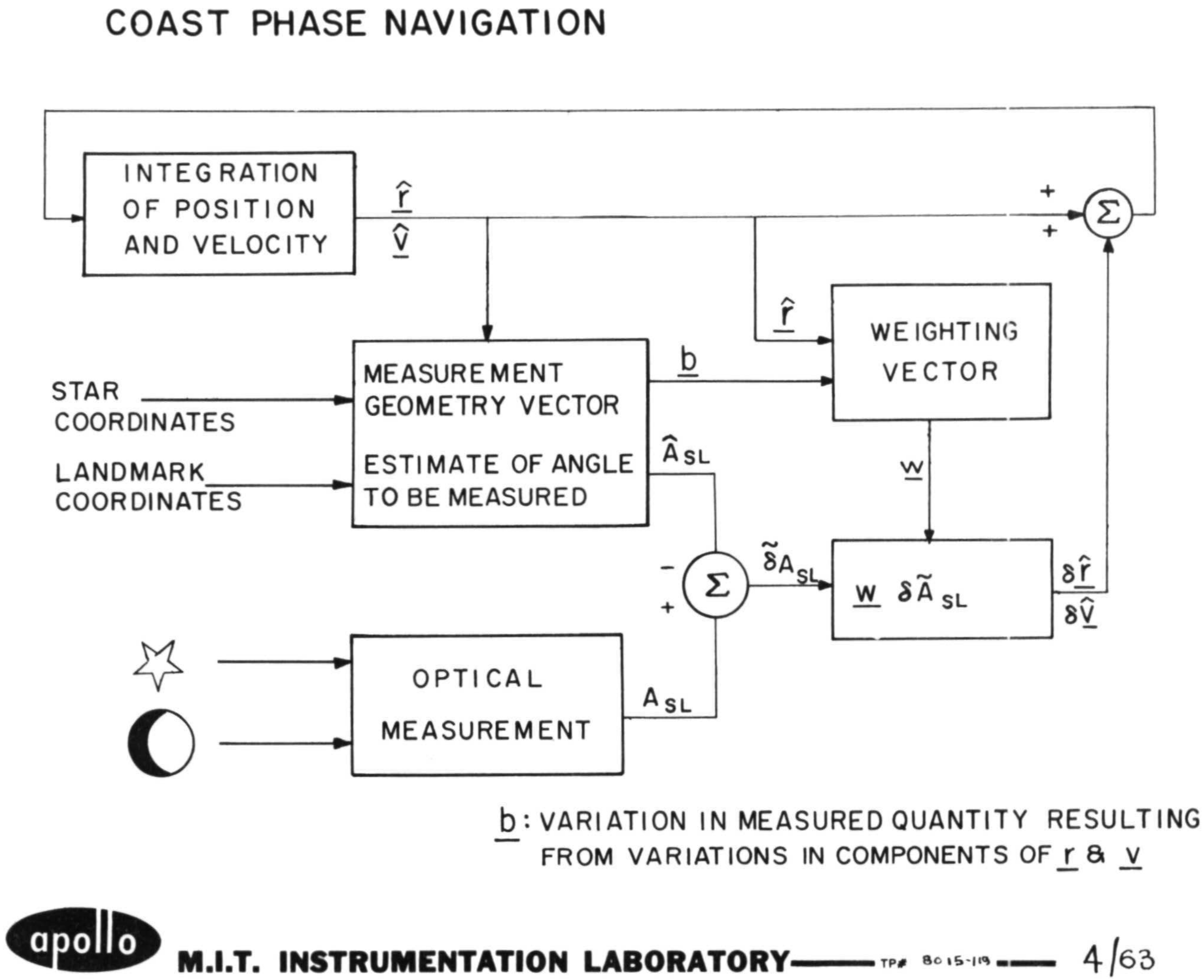
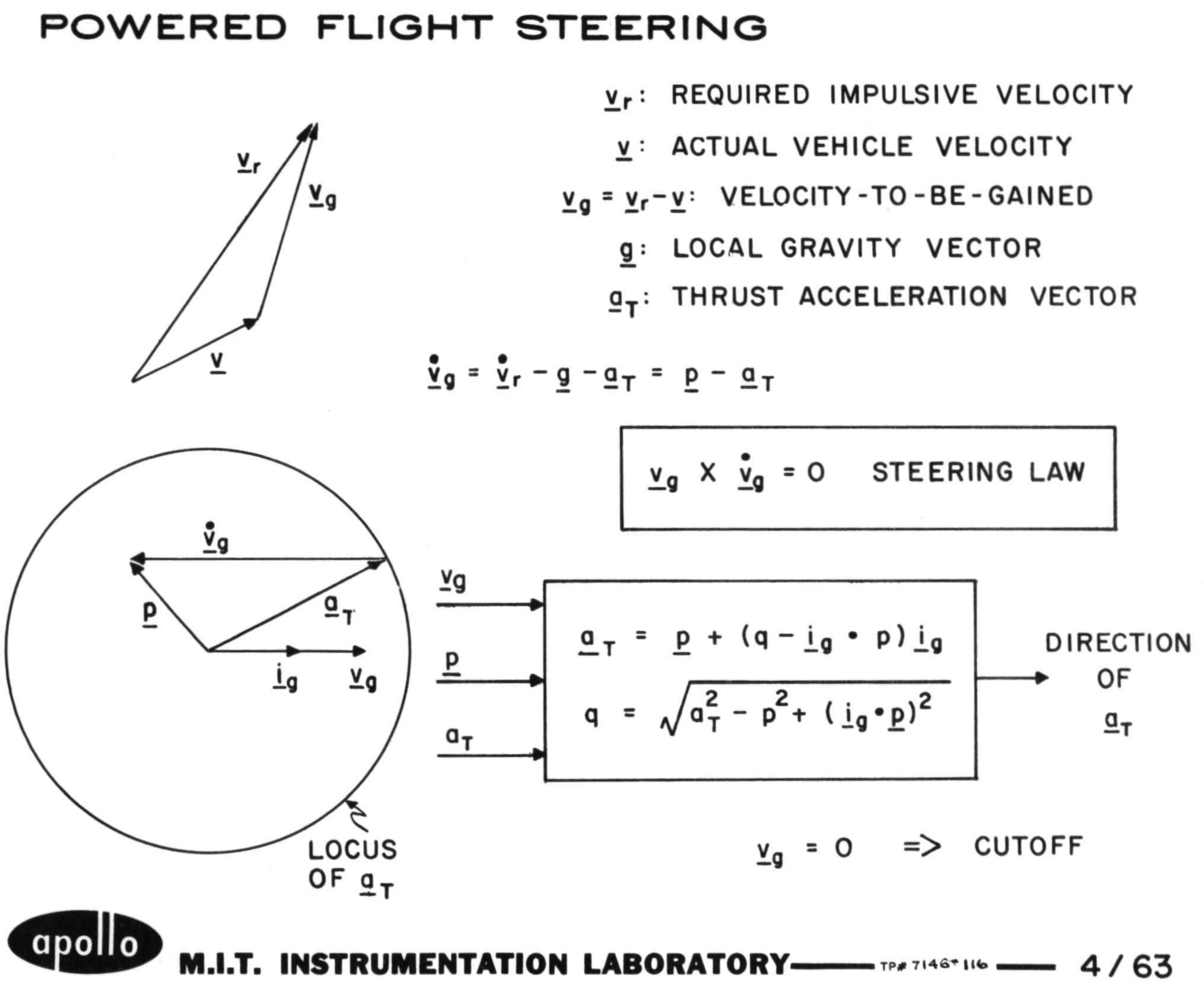


Fig. 8 Spacecraft orientation - orbital navigation sighting, phase 2.



$\underline{b}$: VARIATION IN MEASURED QUANTITY RESULTING FROM VARIATIONS IN COMPONENTS OF $\underline{r}$ & $\underline{v}$

Fig. 9 Coast phase navigation.



POWERED FLIGHT STEERING

$\underline{v}_r$: REQUIRED IMPULSIVE VELOCITY

$\underline{v}$: ACTUAL VEHICLE VELOCITY

$\underline{v}_g = \underline{v}_r - \underline{v}$: VELOCITY-TO-BE-GAINED

$\underline{g}$: LOCAL GRAVITY VECTOR

$\underline{a}_T$: THRUST ACCELERATION VECTOR

$$\dot{\underline{v}}_g = \dot{\underline{v}}_r - \underline{g} - \underline{a}_T = \underline{p} - \underline{a}_T$$

$$\boxed{\underline{v}_g \times \dot{\underline{v}}_g = 0 \quad \text{STEERING LAW}}$$

$$\underline{a}_T = \underline{p} + (q - \underline{i}_g \cdot p)\, \underline{i}_g$$

$$q = \sqrt{a_T^2 - p^2 + (\underline{i}_g \cdot \underline{p})^2}$$

$\underline{v}_g = 0 \Rightarrow$ CUTOFF

Fig. 10 Powered flight steering.

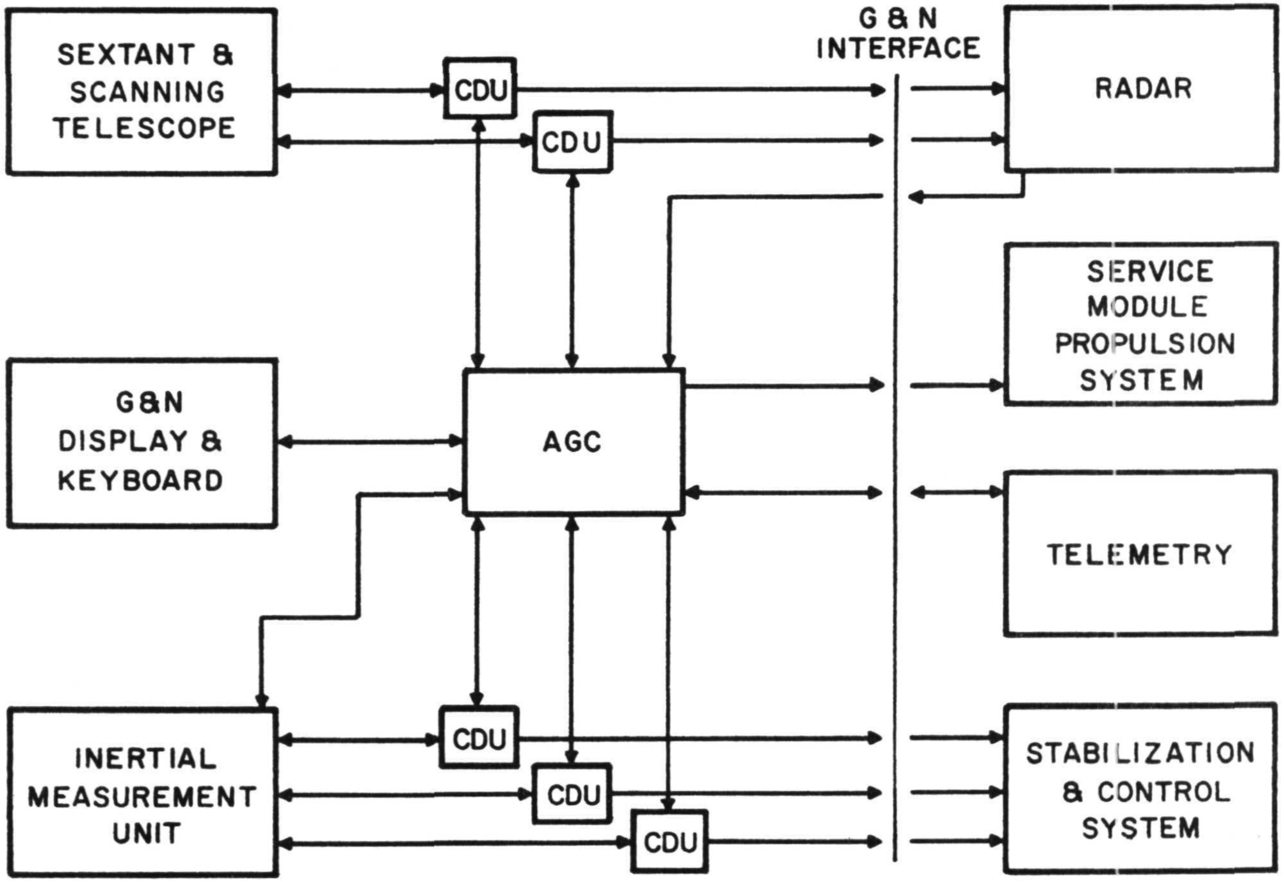


Fig. 11 AGC interfaces

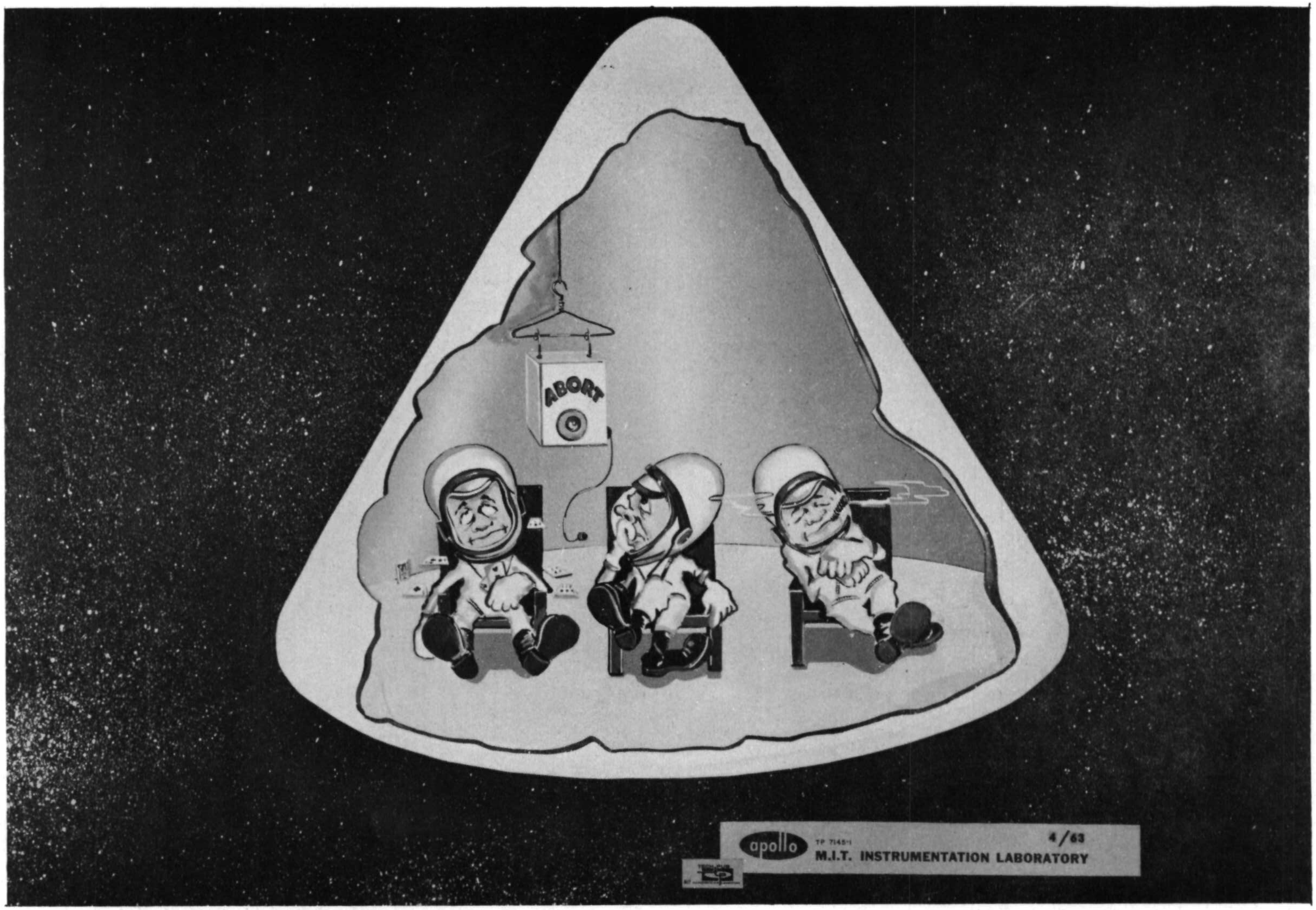


Fig. 12 "Fully automatic" system.

Fig. 13 "Fully manual" system.

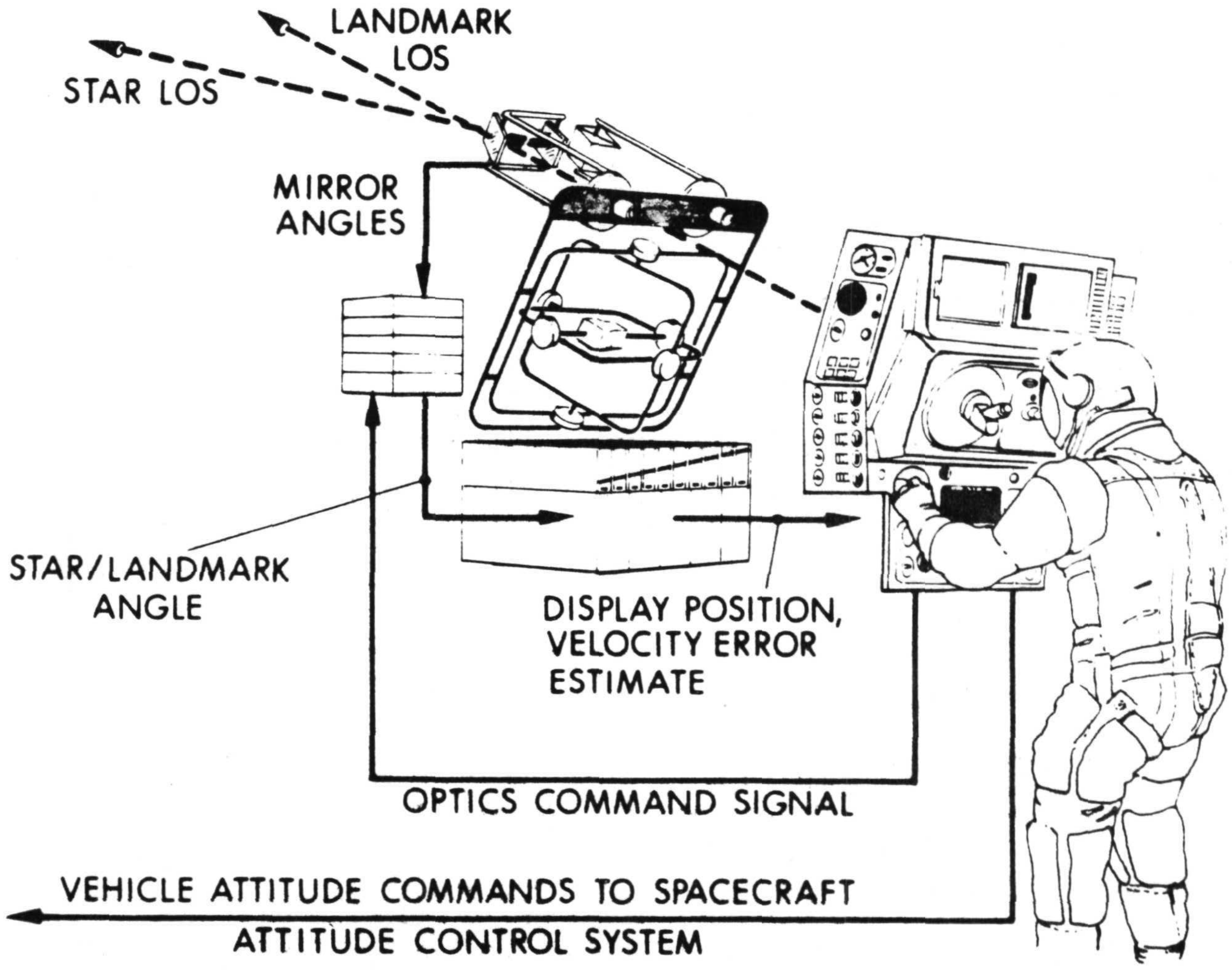


Fig. 14 Midcourse navigation - manual star/landmark angle measurement.

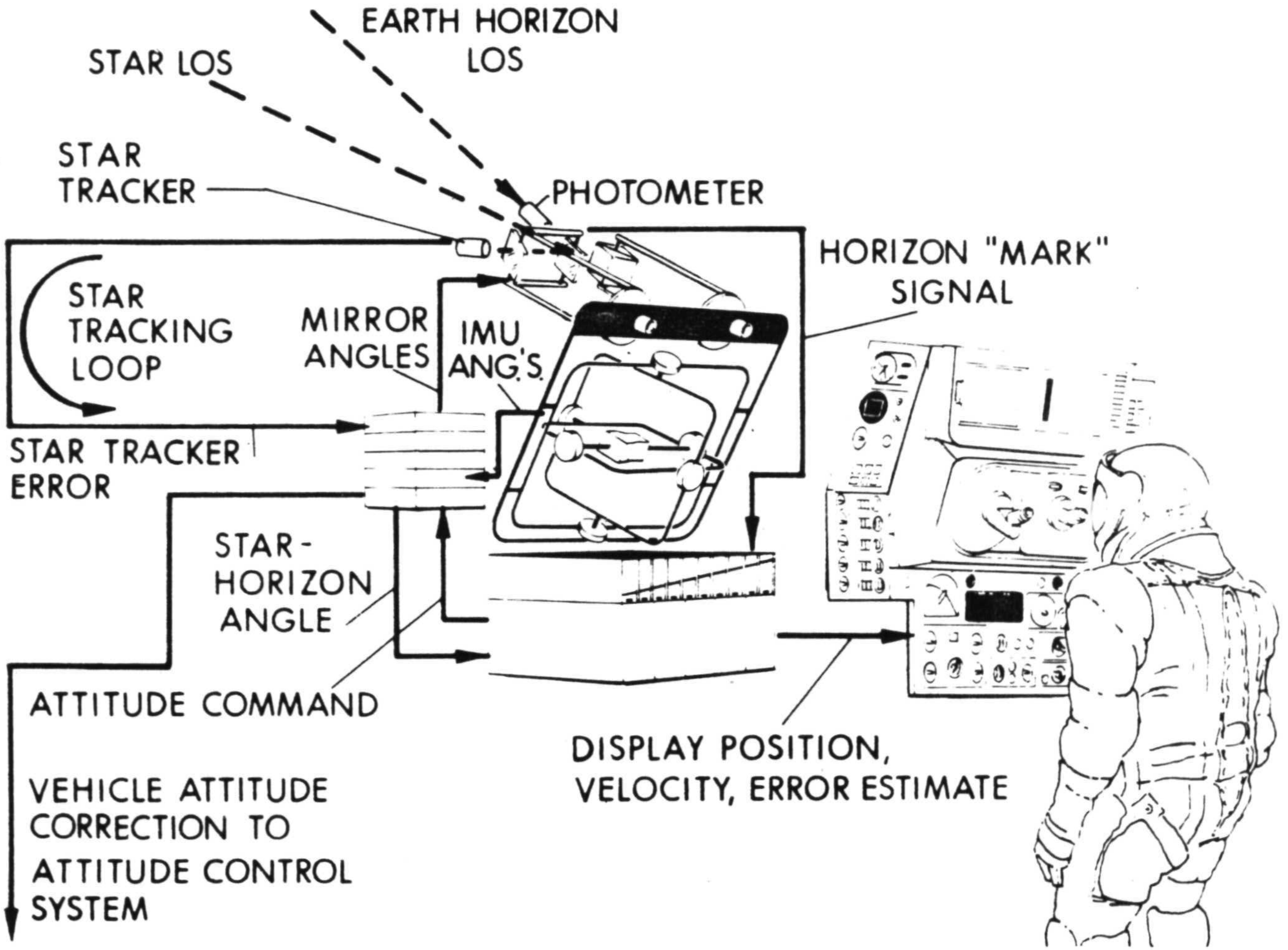


Fig. 15 Midcourse navigation - automatic star/earth horizon angle measurement.

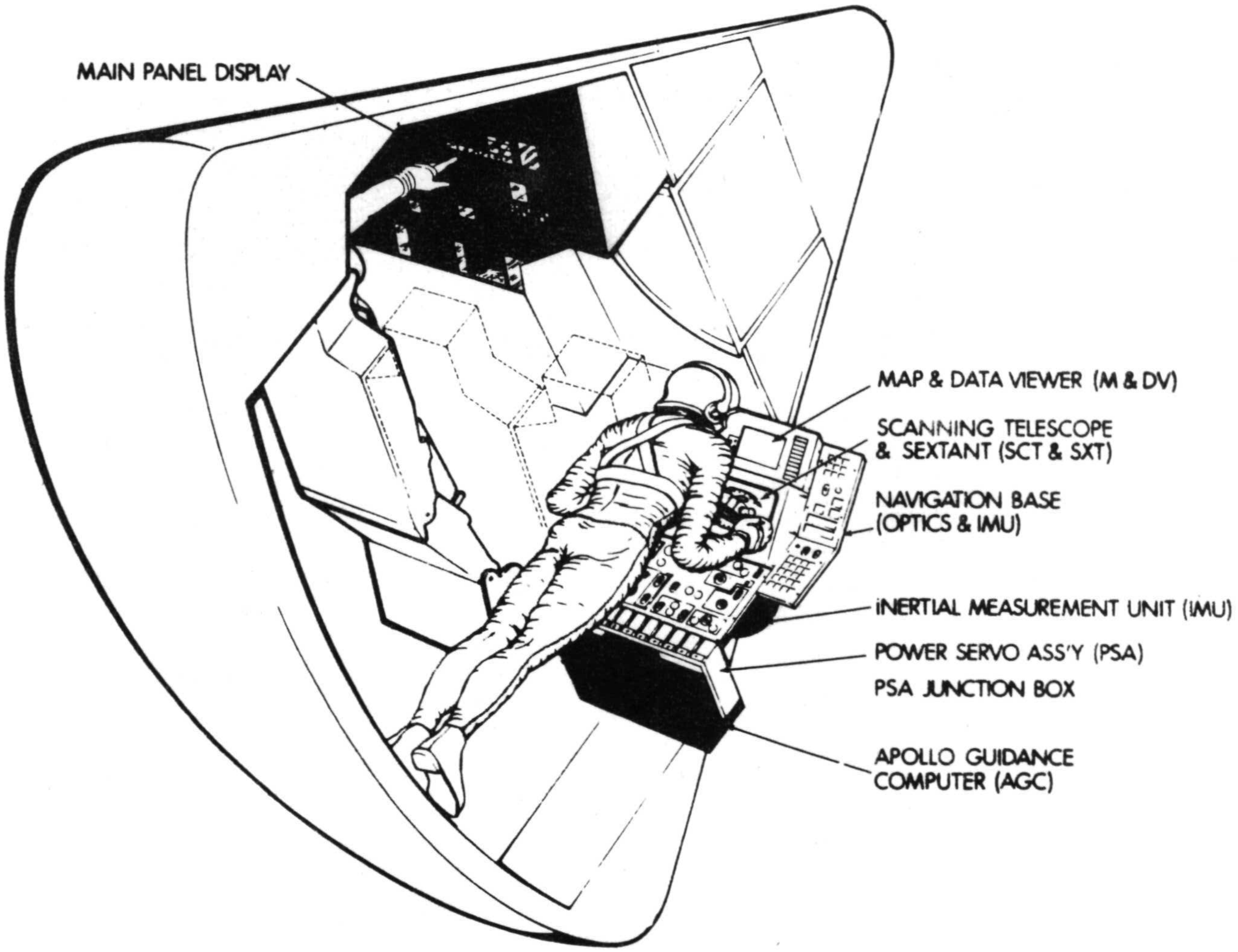


Fig. 16 Apollo G and N - spacecraft location.

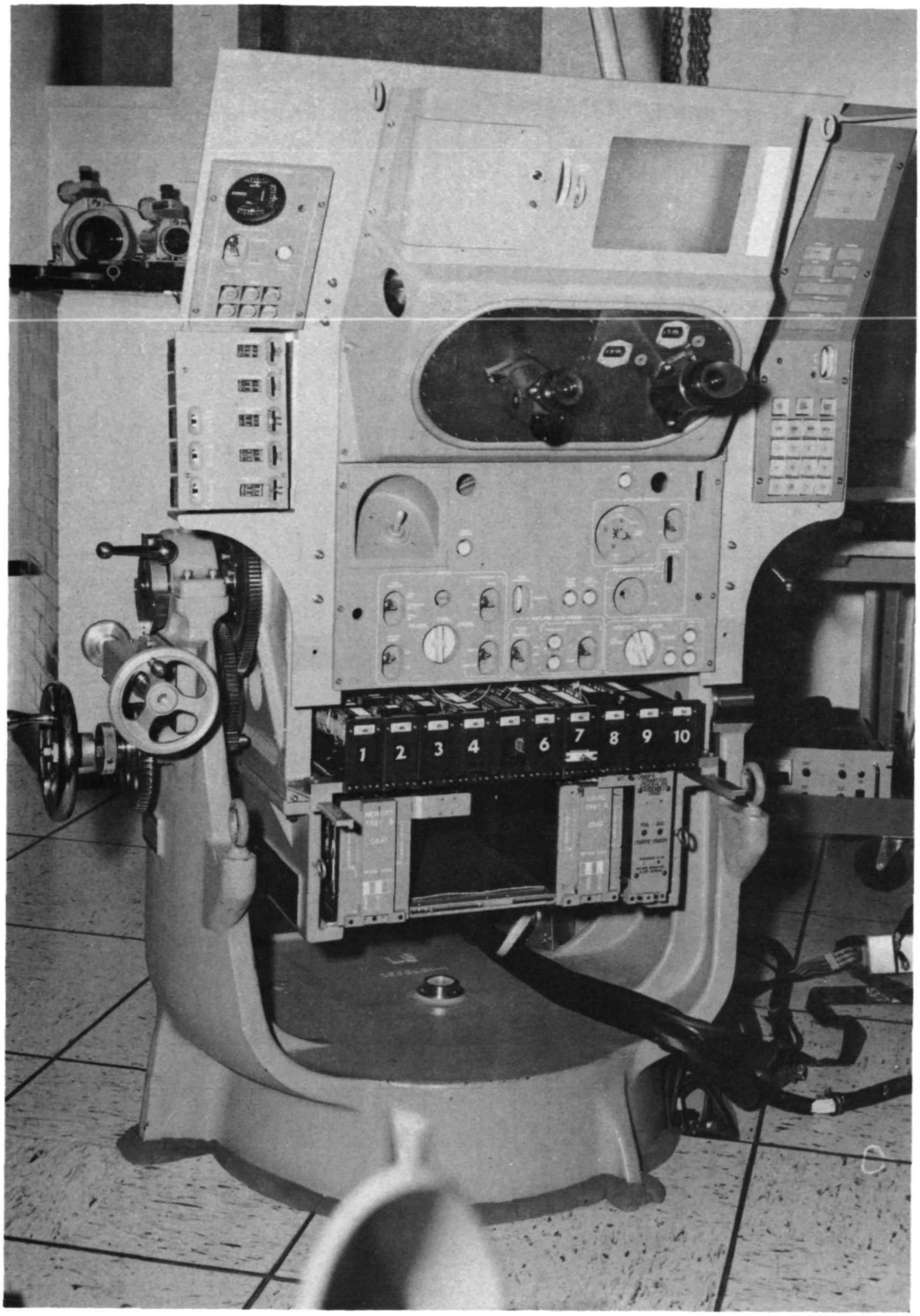


Fig. 17 Actual system undergoing functional testing.

Fig. 18 Actual system undergoing functional testing.

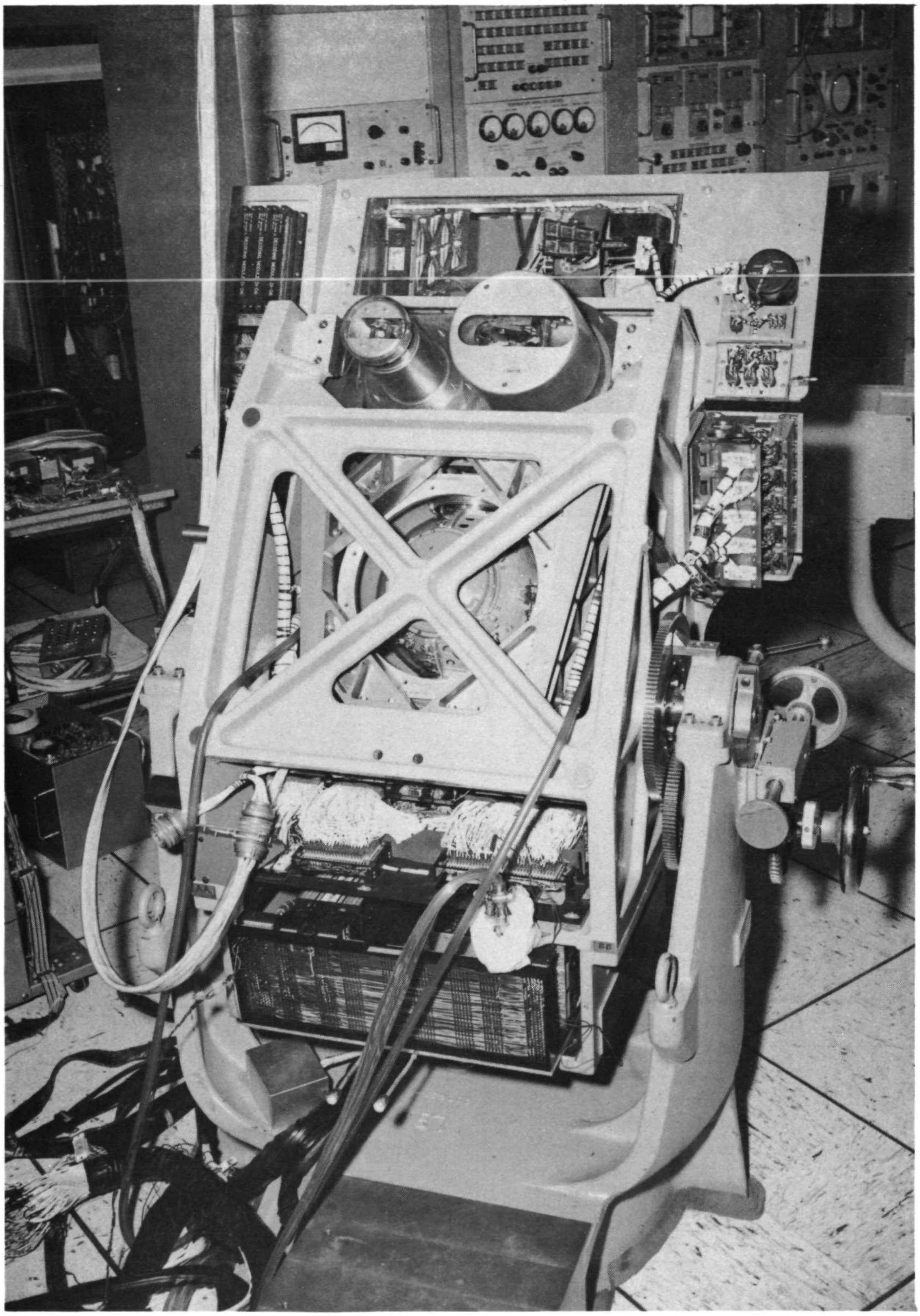


Fig. 19 Actual system undergoing functional testing.

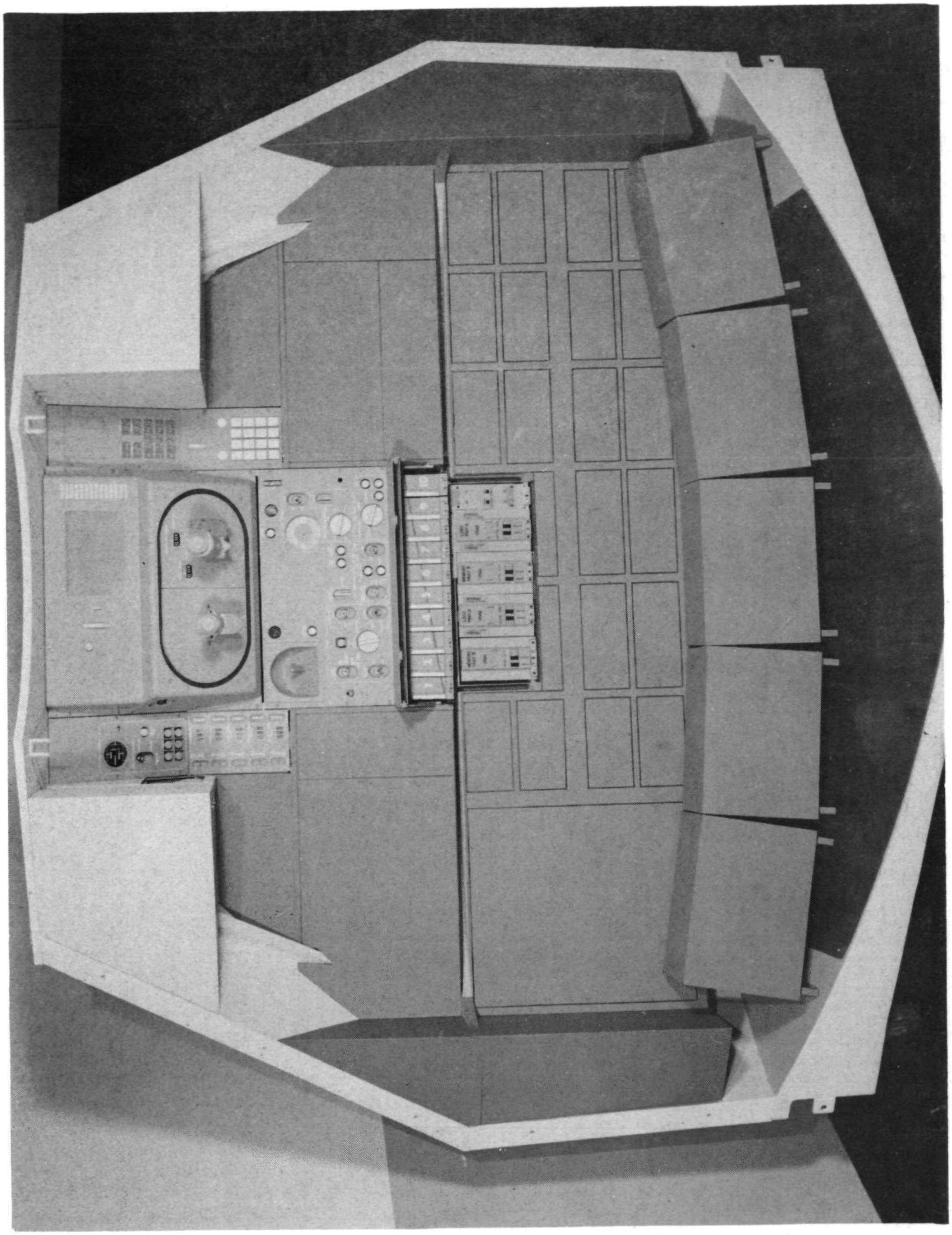


Fig. 20 D and C mockup.

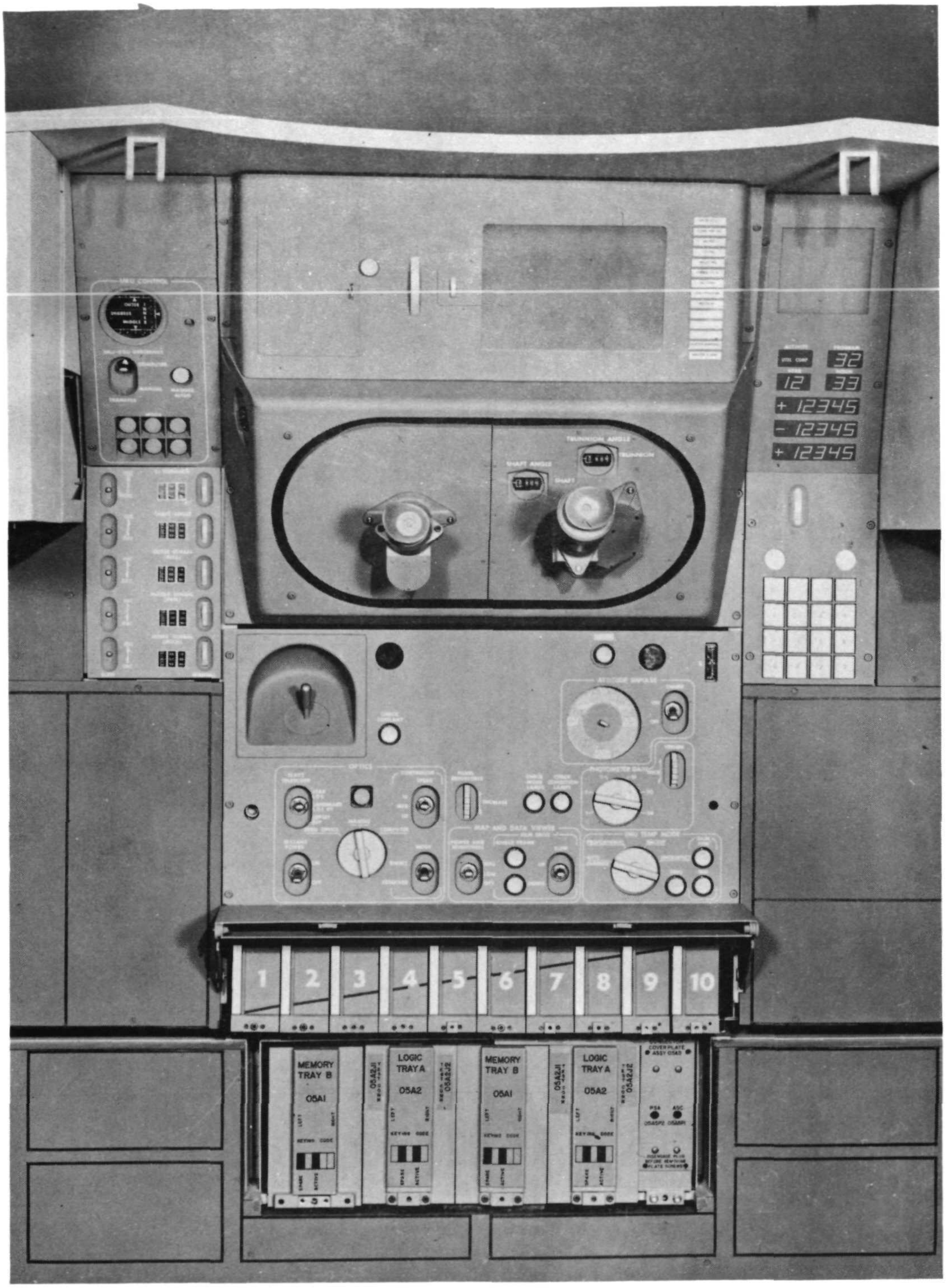


Fig. 21 Close-up of D and C mockup.

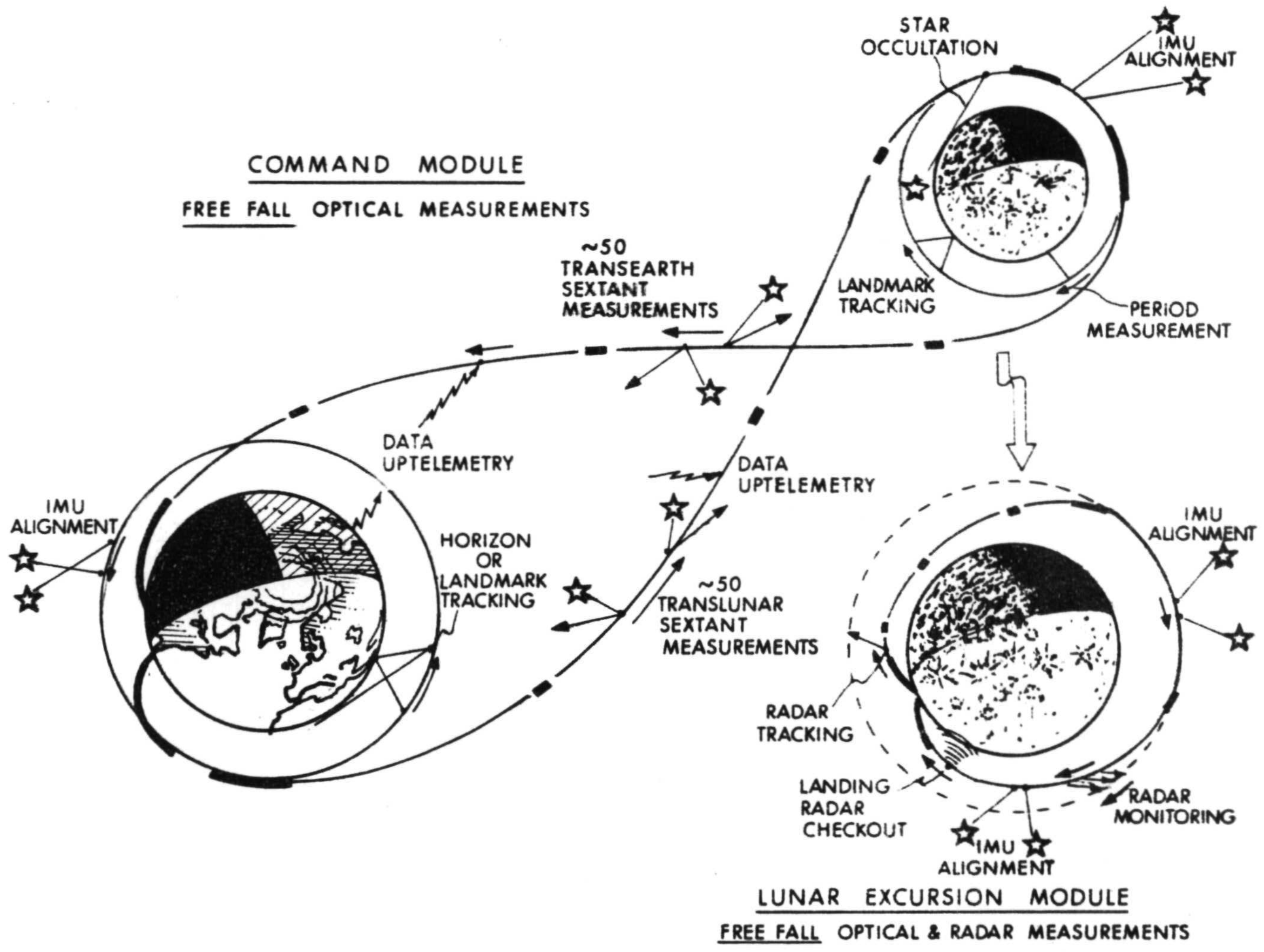


Fig. 22 Navigation mission phases.

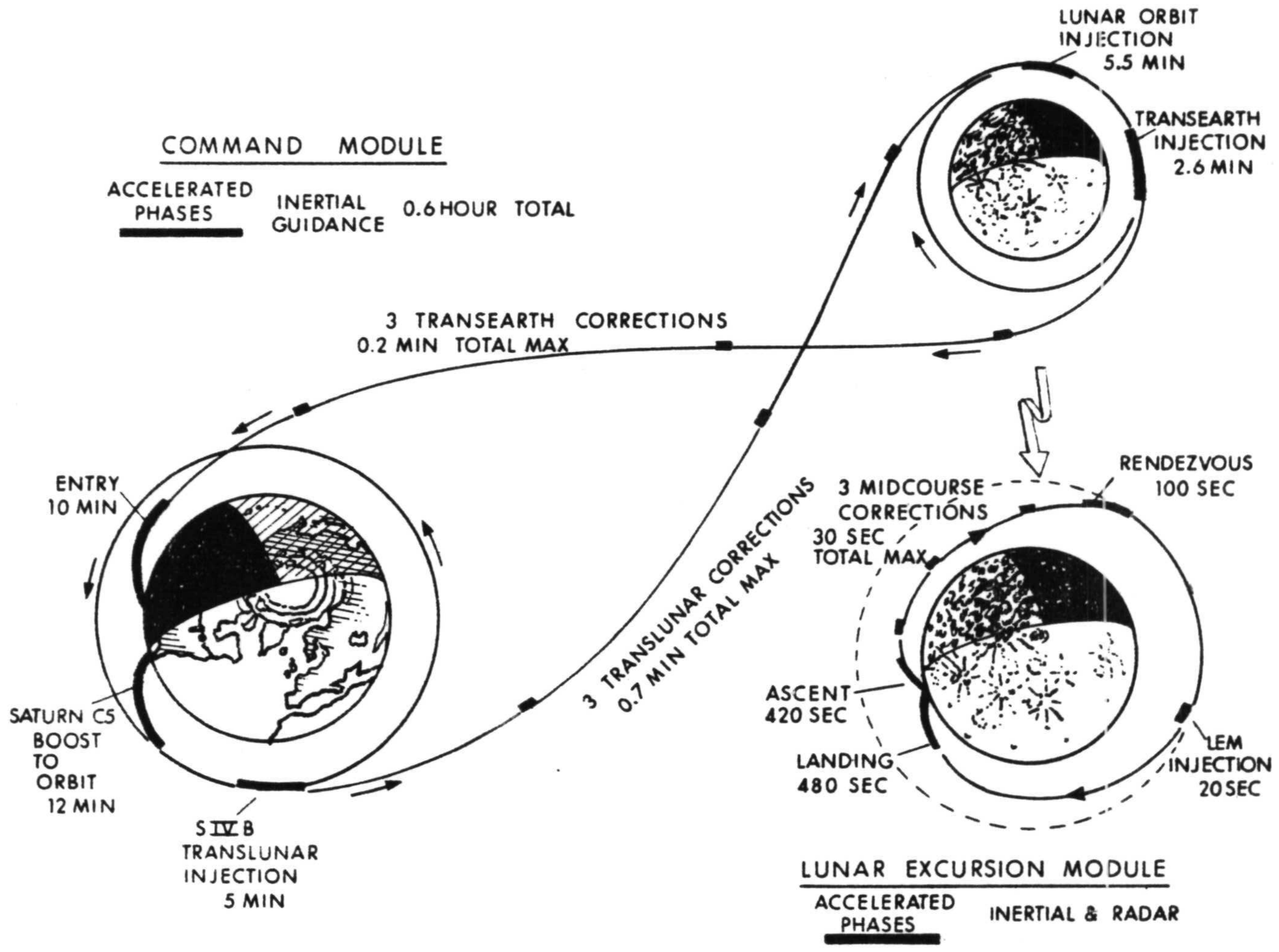


Fig. 23 Guidance mission phases.

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