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GUIDANCE AND NAVIGATION SYSTEM  
FOR  
LUNAR EXCURSION MODULE

ABSTRACT

This report describes the Lunar Excursion Module (LEM) guidance equipment and concepts presently envisioned by the MIT Instrumentation Laboratory. They are subject to change as the LEM configuration and mission are more clearly defined.

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GUIDANCE AND NAVIGATION SYSTEM  
FOR  
LUNAR EXCURSION MODULE

I INTRODUCTION

The approach to Lunar Excursion Module guidance and navigation which appears most reasonable is described. The design criteria are as follows:

1. Provide capability to land at a preselected point on the lunar surface using only onboard guidance equipment.
2. Reduce the magnitude and variability of the required characteristic velocity by means of guidance precision.
3. Favor the use of equipment interchangeable with the command module equipment.
4. Provide ultimate back-up using rudimentary components.
5. Provide astronaut with command control.

Landing with the aid of a radar beacon on the lunar surface was investigated. This approach is unnecessary unless one must set down near a previously landed object which cannot readily be observed from the lunar parking orbit. However, the capability to use a surface beacon is inherent in the described system.

It is clear that accuracy in guidance pays off in terms of propellant requirements for the mission. Consequently, the use of a guidance system that has reasonably high performance is favored because it enables the saving of reserve fuel for some emergency use.

It is also intended to favor the use of equipment interchangeable with the command module equipment. The two-man LEM is thought of favorably in terms of extra living space, and might therefore be returned back to earth with the command module. If the equipment in the lunar excursion module is interchangeable

with command module equipment, we have in effect a spare inertial system. The circuits in the computers could be exchanged on a circuit-block basis. In short, there are available many spare parts, which is an important factor from the in-flight maintenance standpoint.

Finally, an end result desired is an over-all system that has back-up modes using a rather rudimentary attitude reference and accelerometer.

It should be noted here that a simpler system would be feasible under a different set of ground rules. Such a system might employ three body-mounted gyros, one accelerometer and an analog computer in place of the inertial platform and digital computer proposed in this report. A simpler radar system might also be preferable under slightly different conditions.

Design criteria consistent with such a simplified approach do not appear in our best judgement to be advisable at this time.

## II SUBSYSTEM DESCRIPTIONS

Command Module and Lunar Excursion Module systems are shown in Fig. 1 which gives an elementary idea of their locations in the spacecraft and a preliminary table of weights. Volumes are quoted below where appropriate.

### A. Inertial Measurement Unit (IMU)

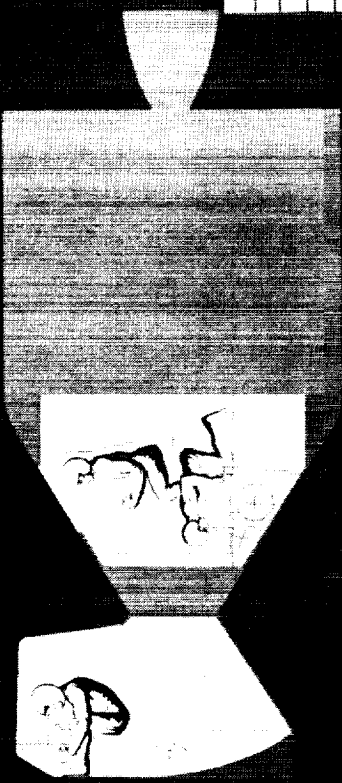
The Inertial Measurement Unit shown schematically in Fig. 2 is a three gimbal stabilized platform using three 25 Size Inertial Reference Integrating Gyroscopes (IRIG). The stable member of the IMU has three 16 Size Pulsed Integrating Pendulums (PIP). The axis orientation of the IMU is such that with all gimbal angles Zero (gimbal axes orthogonal) the inner axis will be aligned with the spacecraft pitch axis, and the outer gimbal axis along the spacecraft roll axis. Pitch and roll axes will have unlimited angular freedom. To avoid gimbal lock, the middle gimbal angle must be maintained less than  $65^{\circ}$ . The IMU will be space aligned



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# GUIDANCE EQUIPMENT



LEM EQUIPMENT	WEIGHT
IMU	60
PSA & CDU	30
AGC	100
RETICLE	20
SCT	7.5
TRACKING RADAR	45
ALTIMETER	10
D & C	55
BACK-UP COMPONENTS	20
CBL & JBX	11
NVB	
FLASHING LIGHT	15
RUNNING LIGHTS	5
TOTAL	378.5

CM EQUIPMENT	WEIGHT
IMU	60
PSA & CDU	30
AGC	100
SXT & SCT	34
TRANSPONDER	10
CBL & JBX	11
M & VD	8.5
D & C	69.5
NVB	45.8
FLASHING LIGHT	15
RUNNING LIGHTS	5
TOTAL	388.8

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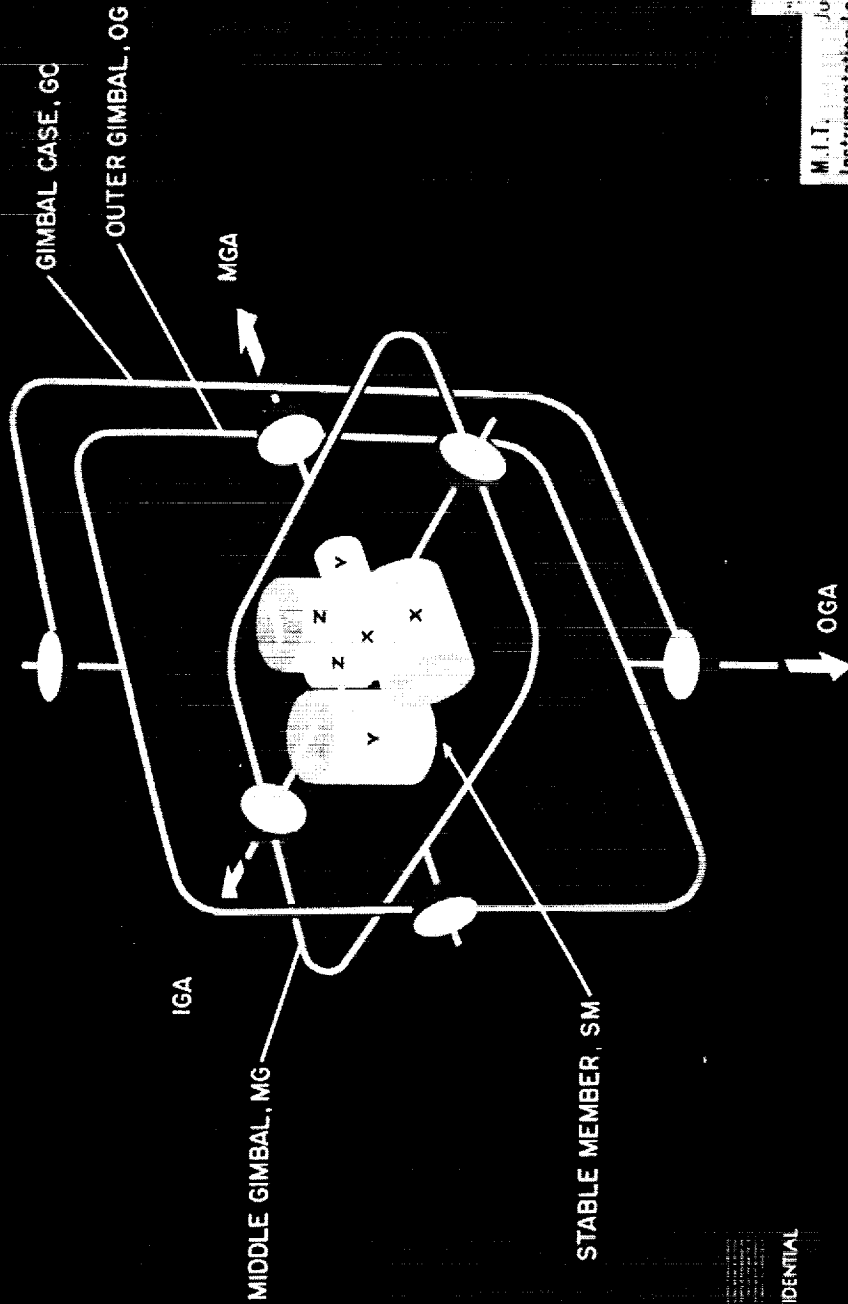
Figure 1

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## IMU SCHEMATIC



  
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Figure 2

by means of Scanning Telescope information and, because of this, the IMU and Scanning Telescope will be mounted in close proximity on a common base, called the Navigation Base.

The IRIG gyroscopes have an angular momentum of 450,000 gm cm<sup>2</sup>/sec. These are floated integrating gyroscopes and geometrically stabilized with respect to the case by the fluid and the magnetic suspension system (ducosyn). Signal information is transmitted by a microsyn signal generator. There is a microsyn torque generator which will be utilized for re-aligning the IMU in flight and for pre-flight erection and alignment. The gyroscope will be pulse torqued for these alignments.

The accelerometers are pulse torqued pendulums utilizing the 16 PIP as designed for the APOLLO mission. The accelerometers provide velocity increments of the integrated acceleration. The pendulums are geometrically stabilized with respect to the case by the flotation fluid and the magnetic suspension (ducosyn). The pendulum also contains a microsyn signal generator and torque generator.

It will be the function of the IMU to provide a space stabilized coordinate system and measure the specific force. The principal errors involved in LEM guidance are the gyro bias drift of 10 meru and the accelerometer bias error of 0.5 cm/sec<sup>2</sup>. The IMU is hermetically sealed in a spherical case of 14.25 in. diameter. There is a bulge at either end of the outer gimbal axis causing this dimension to increase to 18.75 in.

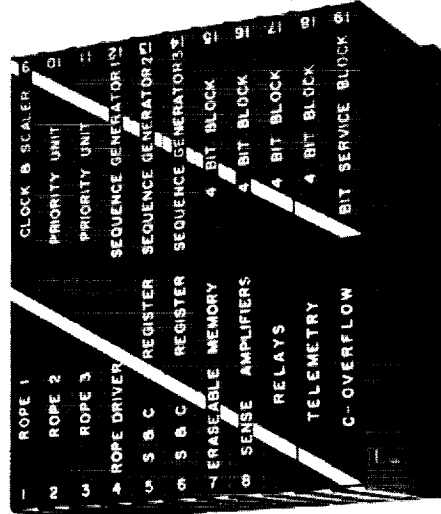
#### B. Apollo Guidance Computer (AGC)

The following is a preliminary description of the computer that is being developed for the guidance system. The computer requirements for the LEM mission have not been completely defined; however, the computer is designed to provide flexibility in memory size and interface characteristics. Fig. 3 shows the AGC and PSA mock-ups for the command module. Some of the

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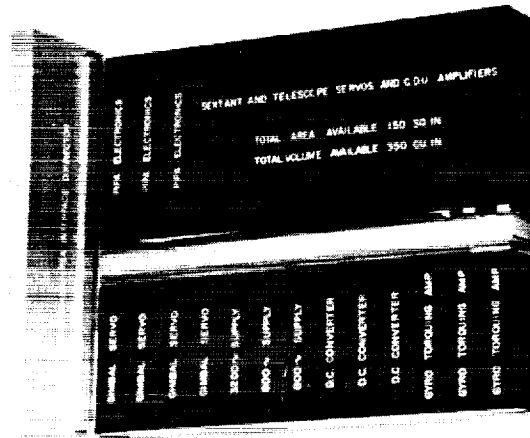
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MODEL COMPUTER



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POWER SERVO ASSEMBLY



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Figure 3

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interfaces that the computer has include IMU angle pickoffs as whole numbers, accelerometer readouts as increments of velocity, telescope angle readouts, radar drive, information for pilot display, clock signals for guidance system, GSE and signals between computer and spacecraft such as, autopilot, communications, and telemetry.

As an integral part of the computer subassembly there is the necessary control equipment to operate and check out the computer. The control equipment is made up of command buttons, mode selection switches, and a keyboard; as output devices there are meters, decimal number lights, and discrete lights. The computer subassembly contains provision for heat transfer to a cold plate and cabling between computer blocks.

The computer is constructed using Weld Pack techniques for the computer blocks. The blocks sizes and form factor are chosen with consideration of repairability and the possible requirement for in-flight replacement of blocks. Volume is 3070 cu in. Dimensions are 21.5 × 7.5 × 19.0 in.

### C. Scanning Telescope (SCT)

This instrument is essentially an articulated telescope having two degrees of freedom and a variable focal length eyepiece which adjusts the magnification and field continuously between 1 and 3 power and 60° and 20°, respectively. An optical schematic is given in Fig. 4. The outer prism should extend just beyond the vehicle skin, to provide a hemispherical viewing field. The SCT is equipped with both manual and servo drives. Electrical gear tooth counter encoders and mechanical angle counters are provided for automatic and visual angle readout, respectively.

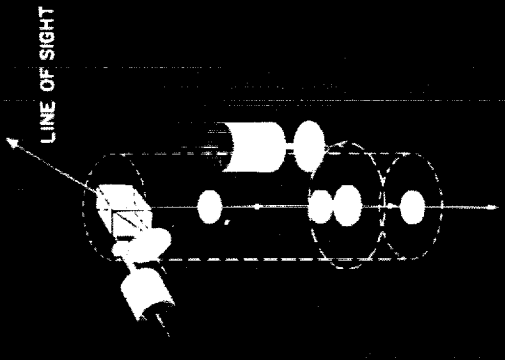
The command module SCT is used for acquisition and identification of celestial bodies and for tracking of landmarks which have higher angular rates with respect to the spacecraft. It is also

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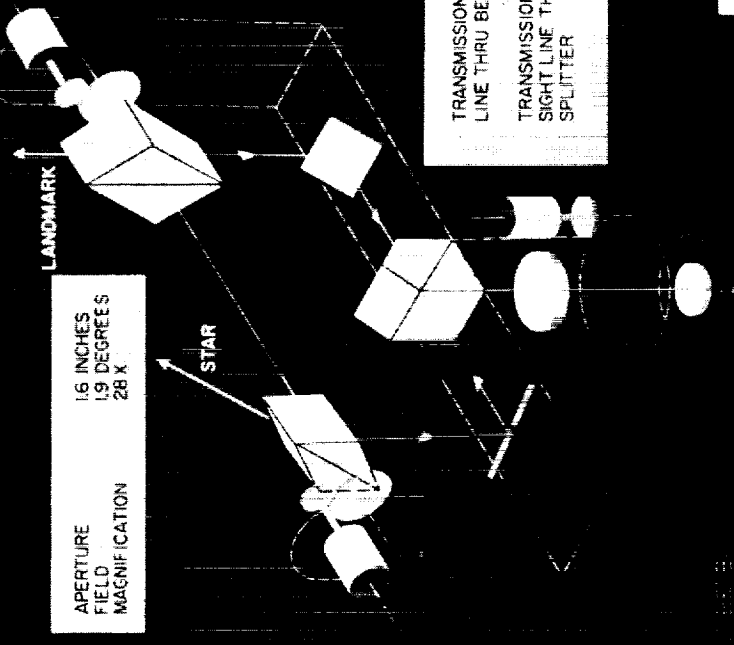
## OPTICAL SCHEMATICS

SCANNING TELESCOPE



EFFECTIVE APERTURE 0.2 IN. 0.4 IN.  
 FIELD 60 DEG. 20 DEG  
 MAGNIFICATION 1x ; 3x  
 NOTE:  
 RANGE OF MAGNIFICATION WILL BE  
 CONTINUOUSLY VARIABLE FROM 1x TO 3x

SEXTANT



APERTURE 1.6 INCHES  
 FIELD 1.9 DEGREES  
 MAGNIFICATION 28 X

TRANSMISSION OF STAR SIGHT  
 LINE THRU BEAM SPLITTER 80%  
 TRANSMISSION OF PLANET  
 SIGHT LINE THRU BEAM  
 SPLITTER 10%

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Figure 4

used as a back-up to the space sextant for star tracking during alignment of the IMU.

In the LEM, the scanning telescope will be used to track stars for the purpose of aligning the IMU. It must therefore have a stable orientation with respect to the IMU, which is the function of the navigation base to maintain. The instrument envelope can be described as a 4 in. diameter cylinder 15 in. long whose axis runs normal to the skin of the vehicle. A small drive pack will be attached at the inside end.

D. Navigation Base (NVB)

The navigation base is the common mounting structure for the IMU and the optical instruments. In the command module, this fixture also provides for retraction of the SCT and Space Sextant. In the LEM, the navigation base will be considerably simpler because no retraction of the SCT is required and because more uncertainty in the angular relationships can be tolerated. Because the design of the NVB is such a strong function of the installation details, it is not possible to make a reasonable estimate of the weight of this fixture at the present time.

E. Coupling Display Units (CDU's)

The CDU is essentially an assembly of electromagnetic transducers and gears with a servo motor drive and display readout. It provides the interface between the IMU, AGC display panel and spacecraft autopilot. One CDU is required for each of the three IMU gimbals. Dimensions are 1.75 x 3.5 x 7.5 in. each. They will probably be mounted in the control panel.

F. Power and Servo Assembly (PSA)

The PSA contains the equipment which converts 28 v d-c spacecraft power to the various forms required by the guidance equipment. It also contains much of the electronics associated with the IMU and optical instruments. The estimated power drawn from the spacecraft supply exclusive of radar requirements

is: 1) Average (in flight) 500 w, 2) Average (on surface) 200 w, 3) Peak (Turn-on) 875 w. A mockup is shown in Fig. 3. Volume of this subassembly is 1023 cu in. Dimensions are 21.5 x 2.5 x 19.0 in.

#### G. Reticle

Visual sensing of the landing site and the mother spacecraft during rendezvous is accomplished by means of the reticle illustrated in Fig. 5. The reticle projects bright cross hairs onto the scene as viewed through a large field of view window. During landing, the reticle is slaved to the landing point for which the guidance system is aiming.

The reticle also provides the pilot with the means of putting commands into the guidance system while still fairly far out from the landing site. The pilot can steer the reticle in his field of view to an alternate landing site. The guidance system will then respond and generate the proper heading and throttle settings to land at the new point. Another mode is to track some rock or other object during the actual vertical descent to trim up the horizontal velocity. During rendezvous, the reticle is slaved to the radar dish which is tracking the transponder on the mother spacecraft. A visual check on radar tracking performance is thereby obtained.

The reticle design assumes that the astronaut's cockpit permits a direct view "aft" through the exhaust plume from the thrust axis to a direction perhaps  $60^{\circ}$  away from the thrust axis. If this field of view is not available, a serious design penalty will result, because alternate means of viewing the landing area are certain to entail heavy optical gear or less efficient landing trajectories.

#### H. Tracking Radar

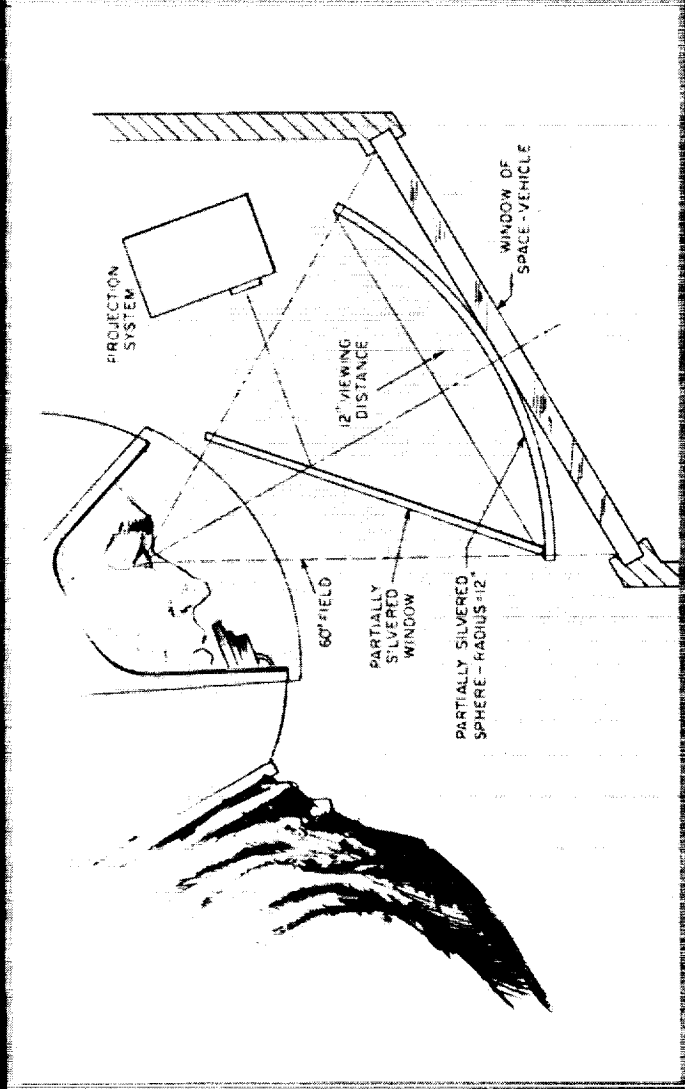
The tracking radar envisioned for lunar landing and orbital rendezvous is a monopulse system having four radiation lobes (beams). These lobes overlap each other to some extent and are symmetrically located about the antenna boresight axis. The



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# RETICLE PROJECTION SYSTEM



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Figure 5

frequency of operation is x-band. Two gimbals and two gyros are used. The antenna is an 18 in. paraboloid with a beamwidth of  $4.6^{\circ}$ . The estimated volume and power consumption are 1 cu ft and 60 w, respectively.

During the initial portion of the landing maneuver, the radar is used as an altimeter. When the vehicle has descended to a point which is about 20 miles from the landing site, the radar is pointed at the site and proceeds to measure the slant range for the remainder of the maneuver. To measure slant range, the radar uses only the two lobes in the elevation plane. The mono-pulse technique is ideally suited for this application and is the main reason for its selection. Note that the radar is not used for angle tracking at this time so that it must be continually pointed by visual and/or inertial means so that the antenna boresight axis intersects the lunar surface at the selected site.

The direct measurement of slant range instead of altitude during the latter portion of the descent is desired because it is considered to be a more accurate and useful quantity. The slant range represents the range-to-go and accurate measurement of it insures the most efficient use of propellant in controlling the flight path to the desired landing site. The slant range measurement also provides a more accurate means for determining the altitude with respect to the landing site since it is not subject to terrain slope as is the direct altimeter measurement. For example, a constant terrain slope of 3 degrees would cause an altimeter measurement to differ from the desired altitude by 15 percent.

The tracking radar is capable of measuring the slant range to within 1 percent. If accurate knowledge of the elevation angle between the antenna boresight axis and the local horizontal were available, the desired altitude could be determined to within 1 percent. If the radar is physically separated from the astronaut's visual means for sighting the landing area, there will be some error in our knowledge of this elevation angle because of

spacecraft flexure and vibration. It is tentatively desired that this angular transmission error be no larger than  $0.25^\circ$  which results in slant range and altitude errors of approximately 1.6%.

During lunar orbit rendezvous the radar is used in co-operation with a transponder on the mother spacecraft. Prior to take-off the radar acquires and locks onto the target and proceeds to track in range and angle throughout the ascent and rendezvous phases. The primary reason for the radar in this application is to provide range data. An adequate source of target bearing data is already available in the reticle which can be pointed at the easily visible mother ship by the astronaut. The main reasons for requiring the radar to angle track are to:

1. Eliminate the need for continuous visual tracking by the astronaut merely to keep the target within the radar beam.
2. Permit the vehicle to have certain thrust attitudes in which visual contact may not be practical but continued radar operation is desired.

The following is a summary of the desired ranges and accuracies ( $3\sigma$ ) of the tracking radar:

1. Lunar Orbital Rendezvous -- cooperative rendezvous with Apollo spacecraft carrying a transponder.

Operating range:	200 nm to 200 ft
Range rate limits:	$\pm 5000$ fps to 0 fps
Range accuracy:	$\pm 0.5\% \pm 20$ ft
Angle tracking accuracy:	$\pm 5$ mr

2. Lunar Landing

- a. Slant range mode of operation

Operating range:	20 nm to 500 ft
Range rate limits:	-2500 fps to 0 fps
Range accuracy:	$\pm 1\% \pm 20$ ft

- b. Altimeter mode of operation -- this mode is not necessarily required but is available with very little, if any, penalties in weight, etc.

Desired operating altitude: ~100 nm to 500 ft

Altitude accuracy:  $\pm 1\% \pm 20$  ft

I. Altimeter

A small FM/CW altimeter is used during the vertical descent phase of the lunar landing. The altimeter will be body-fixed and will probably use an x-band frequency. The desired altitude of operation is 1000 ft to zero with an error less than  $\pm 1\% \pm 5$  ft ( $3\sigma$ ). The volume and power consumption are expected to be 200 cu in. and 15 w, respectively.

J. Displays and Controls (D & C)

This subassembly is largely undefined for the LEM application. The following D and C groups, also found in the command module, will be required:

1. Computer Group
2. Scanning Telescope Controls from Sextant and Scanning Telescope Group
3. Inertial Measuring Unit Group
4. Coupling Display Unit Group
5. Gyro Torquer Group
6. Clock Group
7. G and N Power Controls
8. Conditional, Caution and Abort Lights
9. IMU Gimbal Angles Display
10. Attitude Error Signals

In addition there will be the following new groups:

1. Reticle Control Group
2. Tracking Radar Group
3. Altimeter Group
4. Back-up Attitude Reference and Program
5. Back-up Accelerometer Display and Program

Fig. 6 is a very preliminary version of the cockpit showing the reticle, an 8-ball attitude reference, a body-fixed accelerometer indicator, radar indicators, reticle control stick, condition lights, etc. Estimated D & C panel area is 260 in<sup>2</sup>. Required depth varies from 2 to 8 in.

K. Lights

Both the mother ship and the LEM will have a flashing light and running lights for use in rendezvous and docking.

L. Back-up Inertial Components

These units have not been adequately defined to permit description here. The time and method of their use is discussed briefly in the following section which describes system operation.

M. Sextant (SXT) (Command Module Equipment Only)

This instrument, similar in principle to the mariner's sextant, is used for midcourse angle measurements, star elevation measurements and star tracking for IMU alignment.

N. Map and Visual Display Unit (MVD) (Command Module Equipment Only)

This unit provides the navigator with instructions, maps, instrument set-up data, etc.

O. Transponder (Command Module Equipment Only)

This transponder is tracked by the LEM radar during lunar orbit rendezvous.

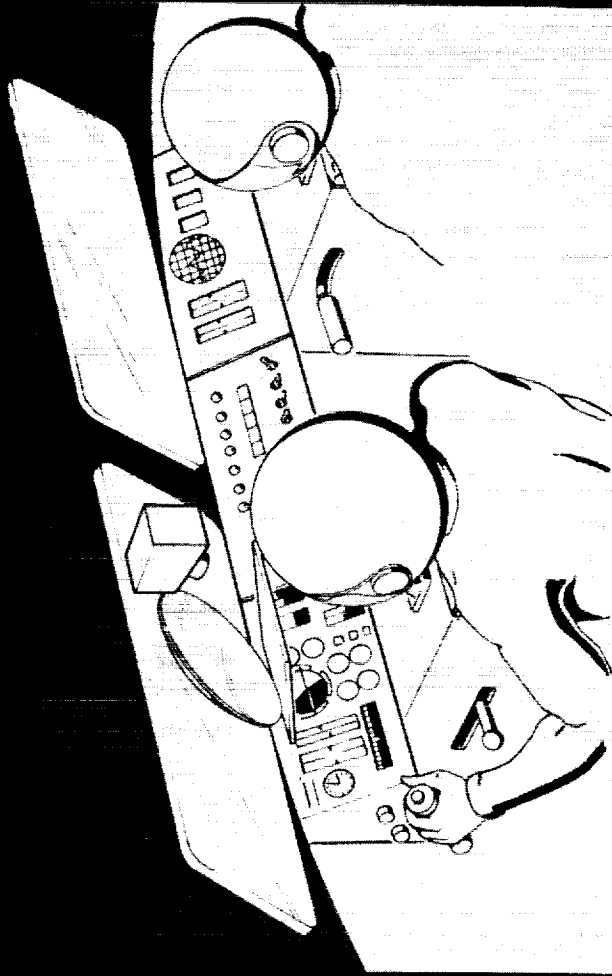
III. SYSTEM OPERATION

A. Lunar Orbit Injection

Fig. 7 illustrates the lunar landing mission. The command module equipment has guided the spacecraft into an approach trajectory which nearly passes through the desired orbital injection point 100 nm above the lunar surface. Fig. 8 illustrates the fourth and last velocity correction made to the translunar trajectory

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Figure 6

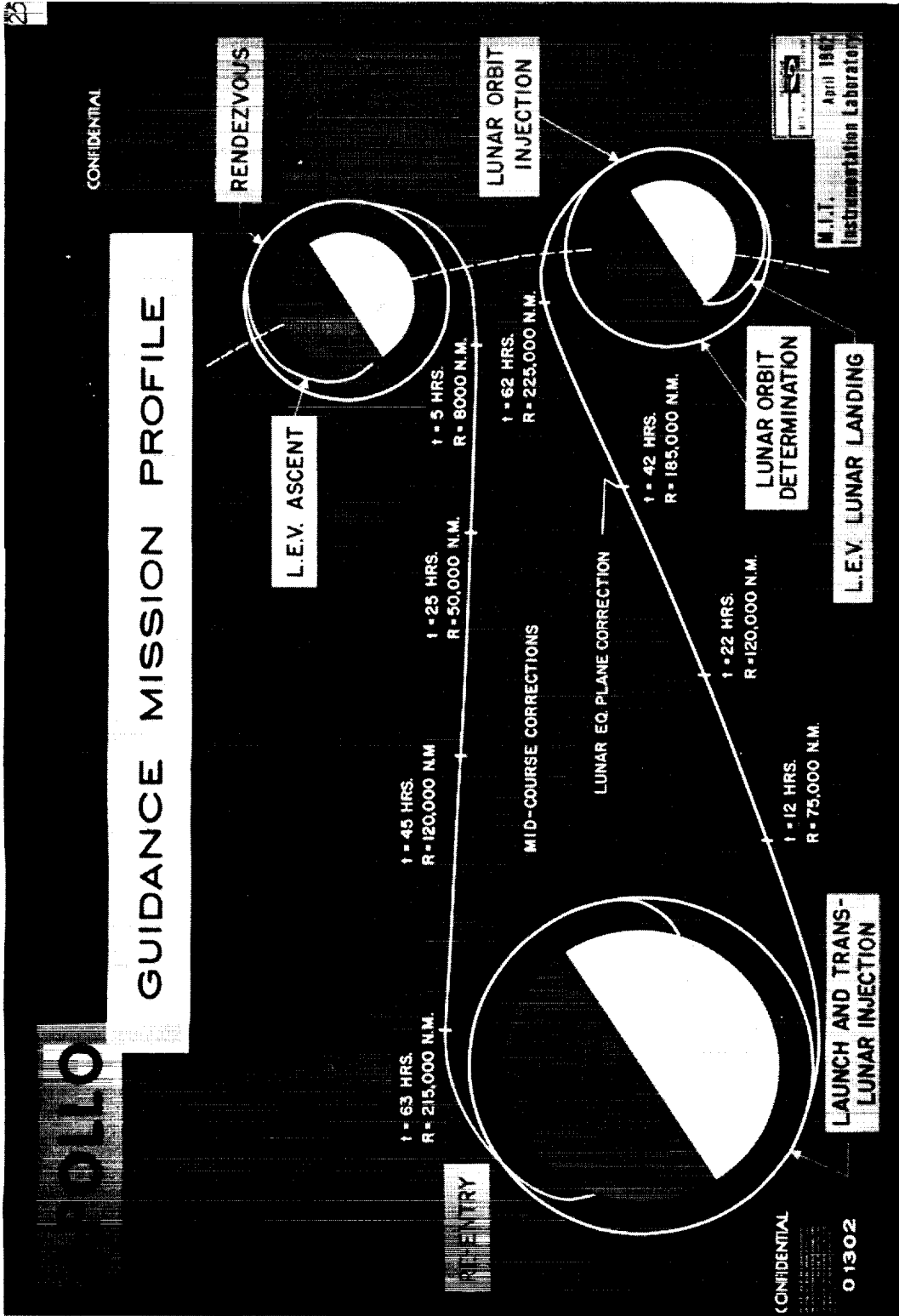


Figure 7

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## LUNAR ORBIT INJECTION

POSITION DETERMINATION POINTS

LAST MID-COURSE CORRECTION  
TIME FROM PERILUNE = 0.8 HRS.  
RANGE TO LUNAR SURFACE = 1850 N.M.

1 - 0.5 HR  
R - 940 N.M.

1 - 0.3 HR  
R - 220 N.M.

INITIAL ERRORS (RMS)		
POSITION	UNCERTAINTY	DEVIATION
ALTITUDE	0.8 N.M.	2.5 N.M.
RANGE	0.4 N.M.	3.5 N.M.
TRACK	0.4 N.M.	0.5 N.M.
VELOCITY		
ALTITUDE	5 fps	105 fps
RANGE	1 fps	37 fps
TRACK	1 fps	2 fps

LUNAR ORBIT INJECTION (100 N.M.)

POST-INJECTION ERRORS (RMS)		
	UNCERTAINTY	
POSITION		
ALTITUDE	0.85 N.M.	
RANGE	0.5 N.M.	
TRACK	0.5 N.M.	
VELOCITY		
ALTITUDE	5.3 fps	
RANGE	2.1 fps	
TRACK	2.1 fps	

CIRCULAR ORBIT DEVIATIONS (RMS)

ALTITUDE	4.5 N.M.
PLANE	0.4 DEG.
PERIOD	28 SEC.

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Figure 8

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0.8 hr prior to injection. Also shown are the final two angle measurements made between lunar landmarks and reference stars. These were preceded by perhaps forty observations during the lunar passage, all of which have contributed to the computer's knowledge of the trajectory.

The position at lunar orbit injection is known with an accuracy of better than a mile, velocity to better than 5 fps in altitude and 1 fps in the other components, as shown in Fig. 8. The last velocity correction was made some time earlier, bending the trajectory very close to the desired point with some slight deviations from the location initially intended. The velocity deviations were not reduced during midcourse corrections even though the computer was aware of them because they are more efficiently washed out by the retro-thrust during the injection into lunar orbit. The position deviations are of little consequence because of the fairly high satellite orbit. Therefore, propellants were saved by not removing them. The application by the inertial system of deboost velocity of several thousand feet per second has increased the errors a small amount in position, but a more significant amount in velocity. The resulting orbit has the errors shown in Fig. 8; some  $\pm 4 \frac{1}{2}$  nm error in altitude from the desired 100-mile circular orbit. The orbital plane is established with an accuracy of  $0.04^\circ$  or better than one mile.

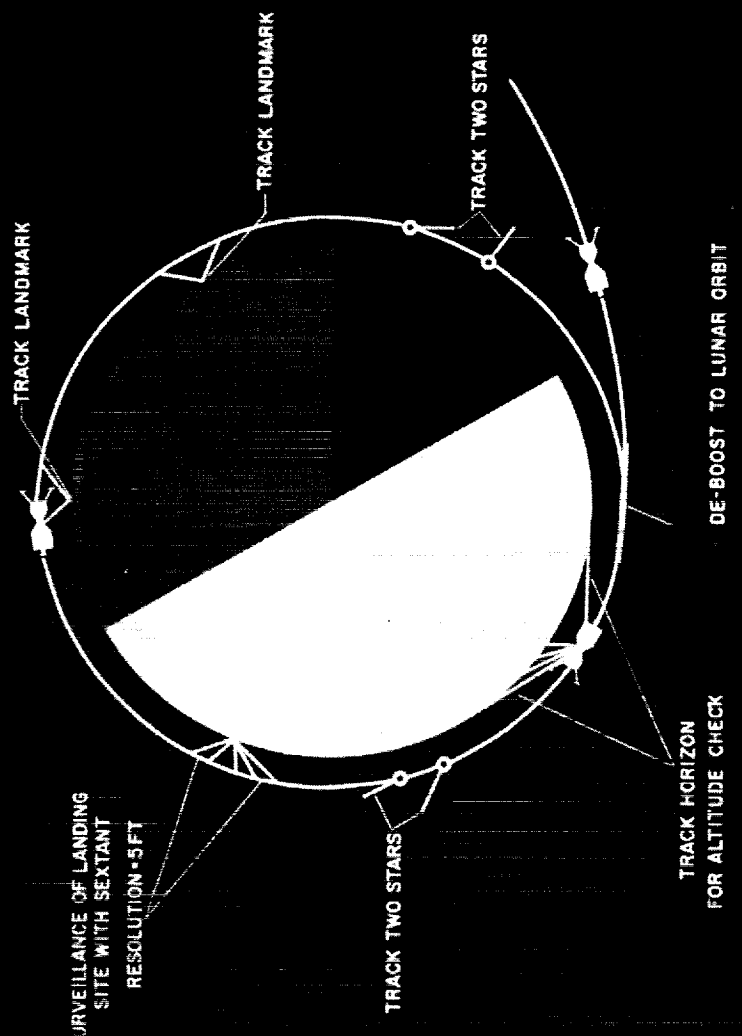
#### B. Orbital Parameter Determination

The flight plan envisioned calls for spending one, or possibly two, orbits at the 100-mile altitude as indicated in Fig. 9. The parking orbit can be better determined by making observations during this period of orbital flight. One method of observation is sighting at the lunar horizon. This gives the direction of local vertical which yields position when measured with respect to the directions to two stars. The sextant in the command module makes these observations by measuring the elevation of certain stars above the lunar horizon.

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# LUNAR ORBIT DETERMINATION

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SURVEILLANCE OF LANDING SITE WITH SEXTANT RESOLUTION ± 5 FT

ORBIT UNCERTAINTIES

	ALT	RANGE	TRACK
POSITION	1500'	1500	1500
FT			
VELOCITY	2	2	2
fps			

\*ALT ABOVE TERRAIN FEATURE GOOD TO 300'

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Figure 9

A second kind of observation is the measurement of the bearings of a landmark, one that has been mapped so that it has a known relationship to a coordinate system on the moon. The landmark is tracked from the command module of the spacecraft and several discrete measurements of its bearings are made as the spacecraft passes nearby. This gives information on the position of the spacecraft in terms of the geography of the landmark. The position of the spacecraft with respect to the landmark can be found by this method with errors of only a few hundred feet. The scanning telescope and the inertial system in the command module are used in this measurement. The shadow area on Fig. 9 indicates the region in which landmarks are expected to be successfully tracked, even when the sun is not illuminating the scene. In this case, the earth is above the figure and is at somewhat more than half full phase. Sunlight reflected from the earth to the moon is bright enough to distinguish landmarks; it is some 30 to 60 times brighter than moonlight here on earth. In the remaining sector nothing is visible; it is totally dark.

The landing site is in the area of one of the objects which is either selected or identified during this low-orbit passage. Other checkpoints on the approach to the landing site can also be surveyed, accurately establishing the relationship between several points underneath the orbit. The computation of position and velocity which follows the orbital measurements is correct within about 1500 ft and 2 fps, which is a direct result of the present uncertainty in lunar maps of approximately 1500 ft. If lunar maps are improved, this accuracy will improve. These errors also correspond to approximately a 600-foot terrain height uncertainty in using the horizon for star elevation measurements.

C. LEM De-Boost to Perilune

The present concept calls for a two-phase landing maneuver. The first maneuver, carried out by the LEM after the orbital parameters have been accurately determined is a 460 fps thrust

toward the center of the moon. The inertial system in the lunar excursion vehicle is aligned by tracking two stars with the scanning telescope aboard the LEM. The inertial system is then used to control the application of a 460 ft/sec velocity change normal to the velocity with an accuracy of about 0.3 fps. The result of this is synchronous orbit having the same size and, therefore, the same period as the orbit in which the mother spacecraft is parked. One period after this maneuver if no further thrust is applied, the two vehicles will approach each other with a close passing distance. If the LEM engine failed to start at perilune or something else went wrong, rendezvous is possible using the mother ship to maneuver to the LEM.

As shown in Fig. 10, under these conditions the passing distance is 1.5 miles. This error is predominantly in range. A small midcourse correction of several feet per second by either the command/service module or the LEM, approximately a quarter of an hour before the time of closest approach, would remove this small error.

This synchronous orbit has a perilune of 50,000 ft, which is reached about  $90^{\circ}$  after the de-boost. At perilune, if all goes well, the second phase of the landing maneuver is started. This next phase is a continuous thrust maneuver which eases the vehicle down a shallow arc to the surface.

Also under consideration for the landing operation is a single phase landing maneuver which is a continuous thrust descent from a 20 mile parking orbit. This concept has the principal advantages of better viewing of the landing site from parking orbit and a single thrust phase landing maneuver. It also involves less total characteristic velocity. On the other hand, it requires greater confidence in the injection guidance precision. At this time the 100 mile parking orbit approach seems appropriate.

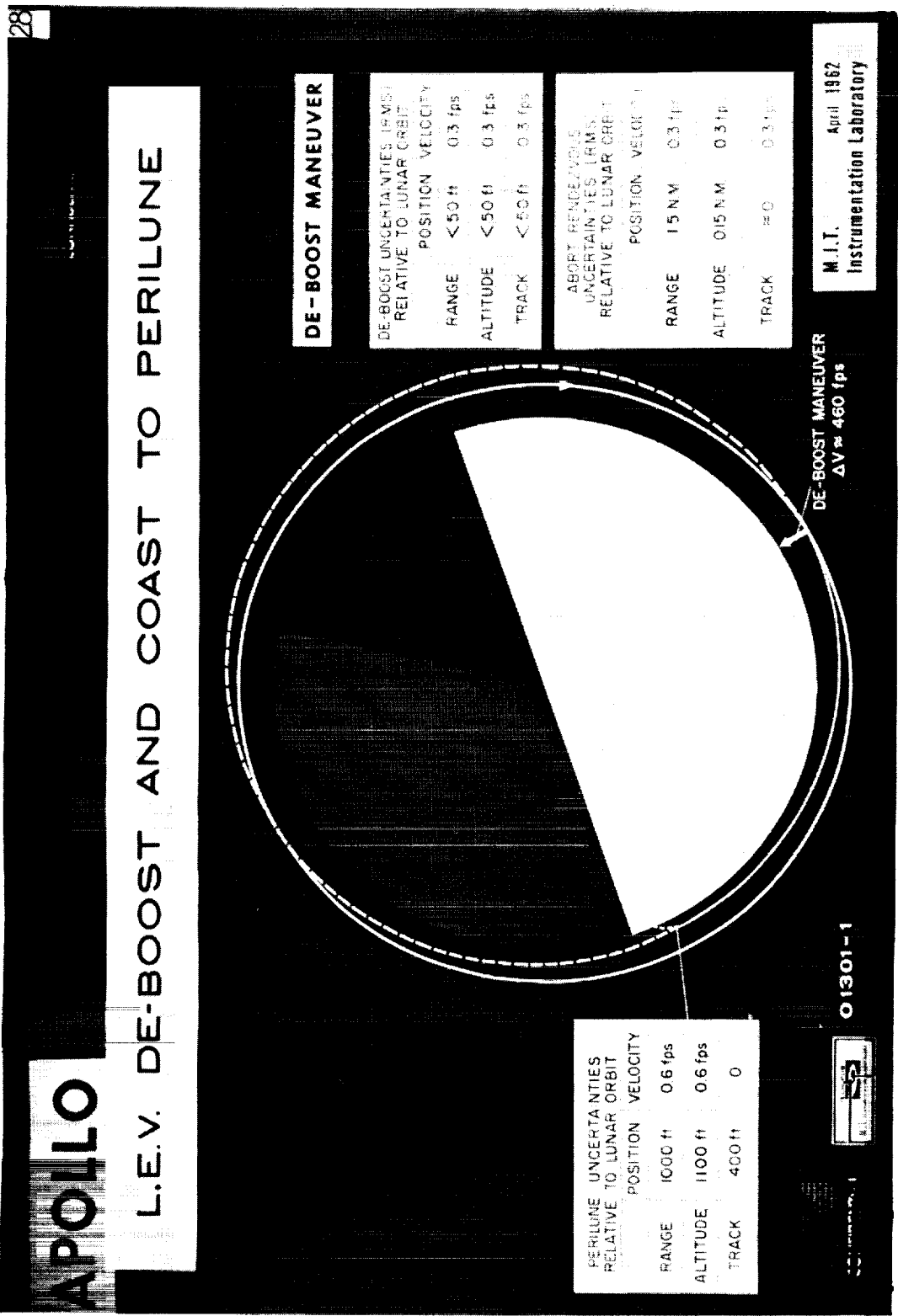


Figure 10

D. Descent Trajectory to the Lunar Surface

In the normal mission without abort, the error of 0.3 fps discussed above results in the largest uncertainty of initial conditions for the landing maneuver. At the beginning of the landing maneuver, the LEM has a position error of approximately one fifth of a mile and a velocity error of less than 1 fps relative to the orbit of the command/service module. Since the observation of the landing site during the prior orbit of the command/service module established its position relative to the landing site with an error of several hundred feet, the predominant position error of the LEM relative to the landing site is a result of de-boost guidance inaccuracy. The predominant velocity error, however, results from the 2 fps uncertainty from lunar mapping errors. Small additional position errors and comparable additional velocity errors are introduced from inertial system errors during the landing maneuver. Thus, without extra information the lunar descent can be made using only inertial data with an accuracy of about 1500 ft in altitude and range and 1000 ft in track from the landing site and approximately 5 fps in each component of velocity.

The descent trajectory is illustrated in Fig. 11. The first phase has a horizontal or slightly flame-up attitude for the first 160 sec. The second phase of the descent has a  $5^{\circ}$  flight-path angle with a  $14^{\circ}$  flame-down attitude. With a vehicle of good visibility the target area itself can be seen some  $10^{\circ}$  off axis throughout this phase of the flight. As the vehicle approaches from a point about 60 miles away, visual inspection of the target area gives improved knowledge of the conditions into which the vehicle will settle. This is not a constant flight path but rather still a nearly full-throttle path. About 10 miles from the landing site and an altitude of about 12,000 ft, the vehicle will throttle back to hold a constant acceleration of about  $21 \text{ ft/sec}^2$ . During this final approach, the selection of the actual landing site is made. The scale drawing at the bottom of Fig. 11 shows that the landing site is not at first in direct view; it is over the horizon. Finally,

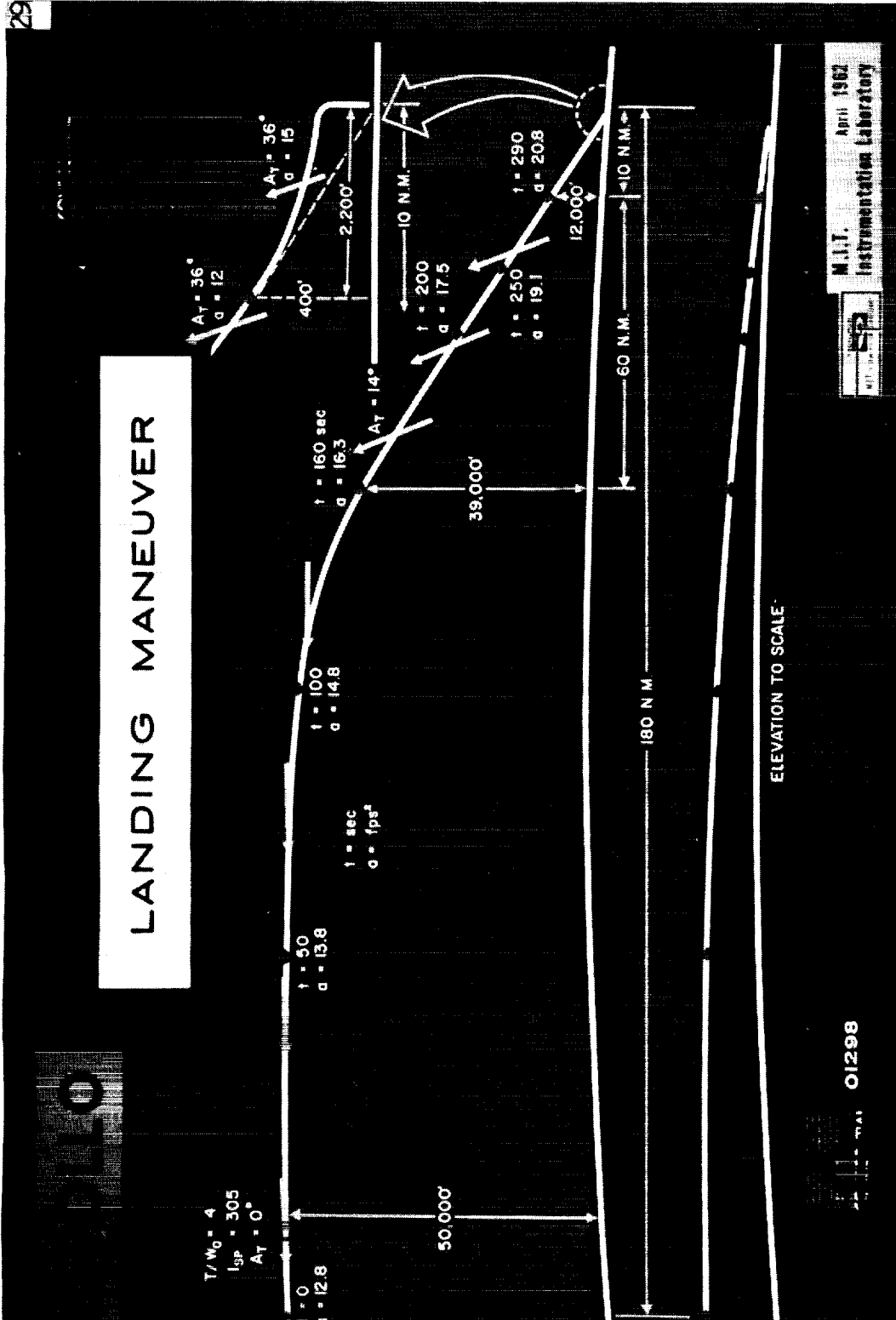


Figure 11

the tail end of the maneuver is completed with the flare-out and vertical descent.

The most efficient maneuver to get onto the surface of the moon involves the latest flare-out trajectories of Fig. 12. The advantage of a late flare-out is that little time is spent in hovering, hence a maximum amount of fuel is saved for the return trip back into orbit. The disadvantages of earlier flare-outs are somewhat longer hover time and a little more propellant used. It would be the pilot's option, based on his visual contact with the landing area, to determine what the proper trade-off would be between length of time hovering at the landing site and the saving of fuel for the return trip. Sometime during the descent he initiates this flare-out, which brings the vehicle to low vertical velocity. The flare-out can be continued in fairly straightforward movement to the ground or in hovering phases and translational movements at the initiation of the pilot.

E. Abort From Descent Trajectory

If, in descending to the moon during this powered flight, something should go wrong with the guidance system, with the life support system, or with some other system on the lunar excursion vehicle, there are several abort trajectories available as shown in Fig. 13. The first is an abort immediately after initiating the landing maneuver; the second is about halfway through the descent flight after the first LEM orientation change; the third is just at the end of the descent flight; and the fourth is after two minutes of hovering. All of these occur in rather natural sequence, which is an advantage of the synchronous orbit.

F. Sensing of the Landing Site

During the final approach to the landing site range and bearing data are required to correct the errors which have accumulated in the inertial system. Small throttle adjustments are made when slant range data first become available at about 20 miles uprange. The AGC directs the radar at the landing site indicated



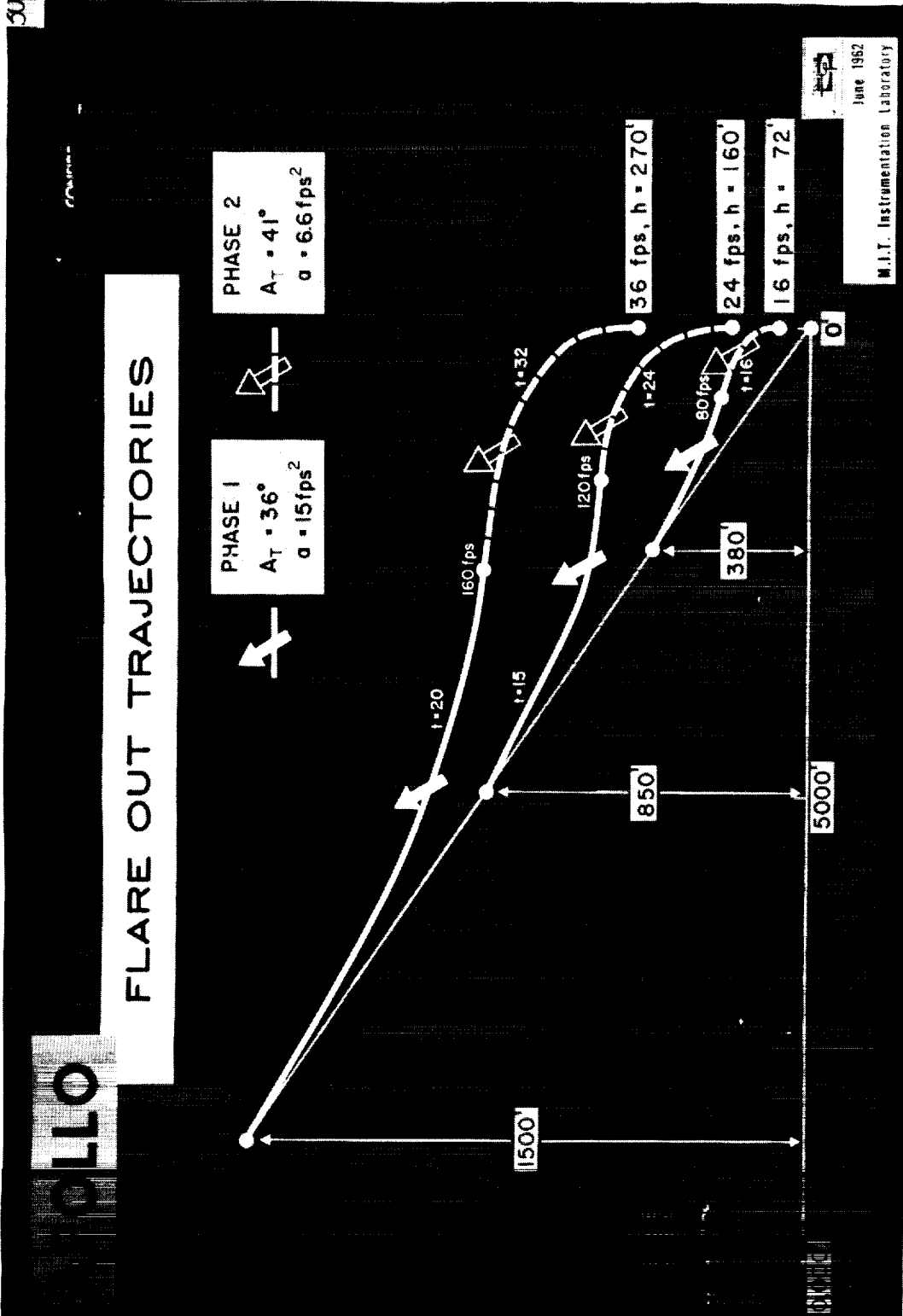


Figure 12

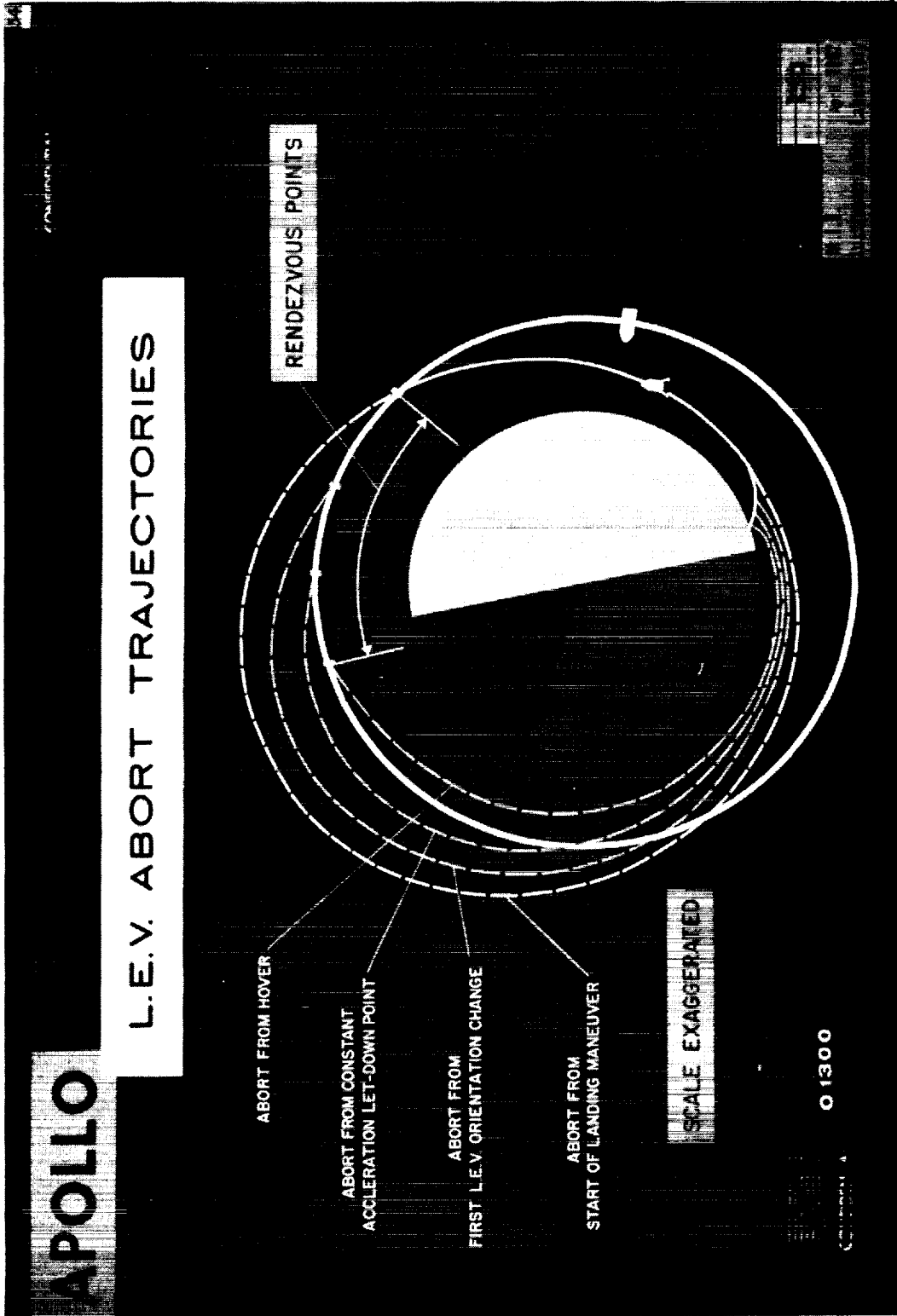


Figure 13

by the inertial system. The reticle is pointed at the same spot, also by AGC command. At a range of about 10 miles the pilot can begin to assess the landing area. At any time he is not satisfied with the area to which the system is aiming the vehicle, he can take over control of the reticle from the AGC, direct it to a more suitable area and return control to the AGC. This action will cause the computer to direct attitude and throttle changes as required to adjust the path toward the new landing site. This operation is illustrated in Fig. 14.

If the LEM were required to land adjacent to a beacon, the radar would be used in its beacon tracking mode. The AGC would then aim the vehicle and reticle at the desired landing site which would be offset from the beacon by some prescribed amount.

The range and bearing data acquired during the final approach will update the inertial system providing the computer with position and velocity information good to about 20 ft and a few feet per second, respectively, when the LEM is within 500 ft of the landing site where the radar is no longer useable. From this point on to touch-down (30 sec plus hover time) guidance is basically inertial. However, if no further external data is acquired the errors at touch-down will be uncomfortably close to their allowable limits. It is therefore advisable during the vertical descent phase, after completion of the flare maneuver, to use a low altitude radar altimeter to up-date the inertial system vertical coordinate channel. The resulting altitude error at cutoff should be less than 5 ft. Similarly, it appears necessary to rely on the astronaut's visual sensing of translational velocities during the vertical descent phase. A dust cloud sufficiently dense to obscure the surface is expected to arise when the altitude has dropped to about 50 ft or less. The IMU can be relied upon to detect velocity changes during the remaining descent. The altimeter is expected to function even when the vehicle is immersed in dust. If this should not turn out to be the case, however, the inertial system can be

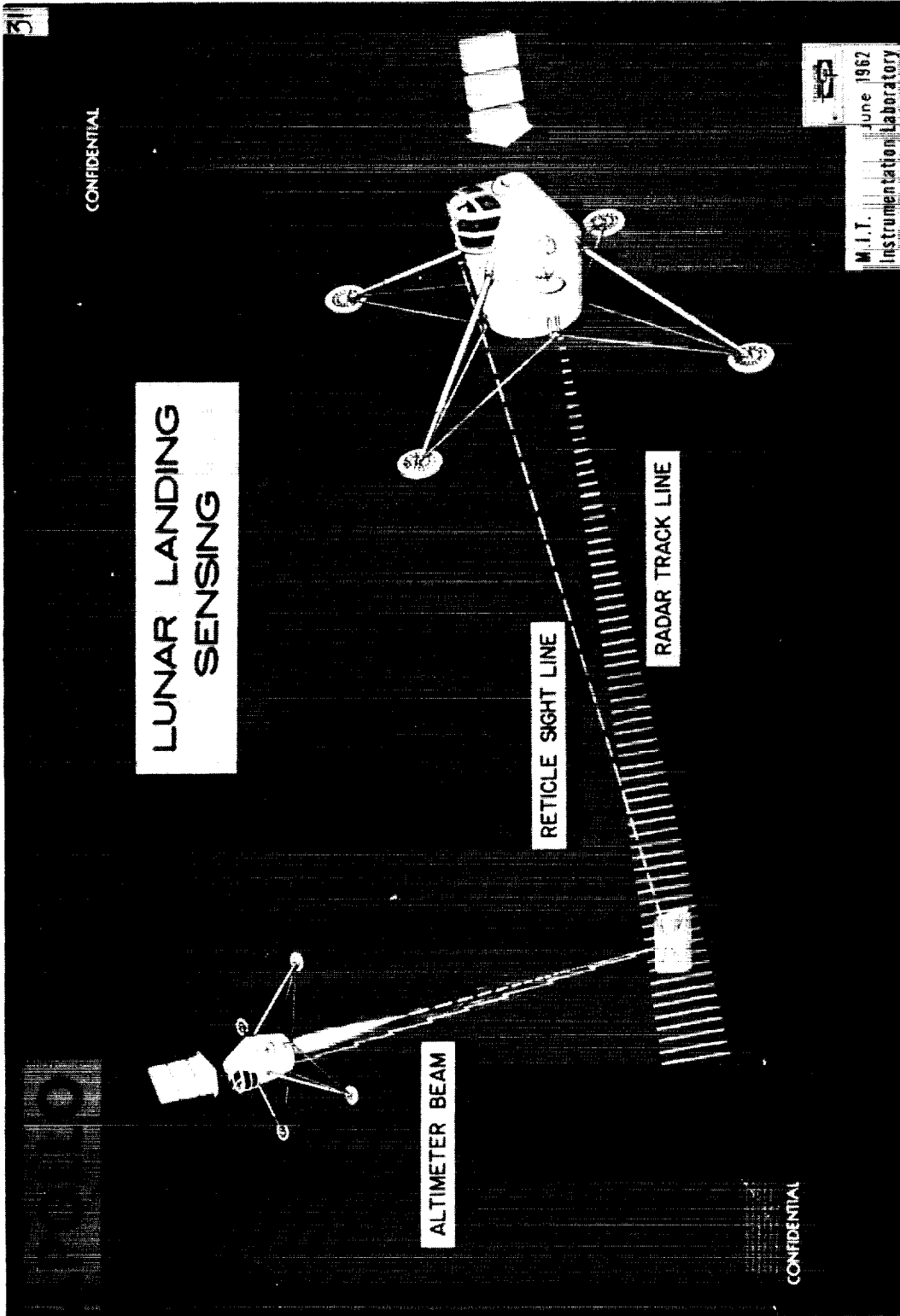


Figure 14

used for altitude information for a short period (perhaps 30 sec) after the last good bit of altimetry is received.

It has been assumed throughout this report that the astronaut has a direct view "aft" through the exhaust plume from the thrust axis to a direction perhaps  $60^{\circ}$  away from the thrust axis. If this field of view is not available, a serious design penalty will result, because alternate means of viewing the landing area are certain to entail heavy optical gear or less efficient landing trajectories.

#### G. Lunar Surface Operation

The operation of the system while on the surface consists of aligning it shortly prior to flight by tracking two stars with the scanning telescope. This function might be visual or automatic, depending what the time study of the countdown suggests. The determining of the location of the mother spaceship is another function which occurs either over communication links, by radar tracking, or by simply relying on a computer computation based upon orbit data stored before separation from the mother ship. Initial conditions would be set into the computer and the equipment would be checked out. This operation is illustrated in Fig. 15.

#### H. Lunar Launch

During the 5 min interval before lift-off, when the orbiting spacecraft is within 200 nm slant range, the tracking radar can lock on the transponder in the mother spacecraft. Tracking can be continuous from this time until rendezvous is complete. However, because the launch will be guided inertially, radar data is not required until the rendezvous maneuver is started. Perhaps continuous tracking from before launch would be advisable as a performance test.

The take-off envisioned is shown in Fig. 16, with about 137 sec of burning to get about 5,600 fps of velocity at cut-off. The position errors at burn-out are negligible; the velocity errors are very small, about 1.5 fps.

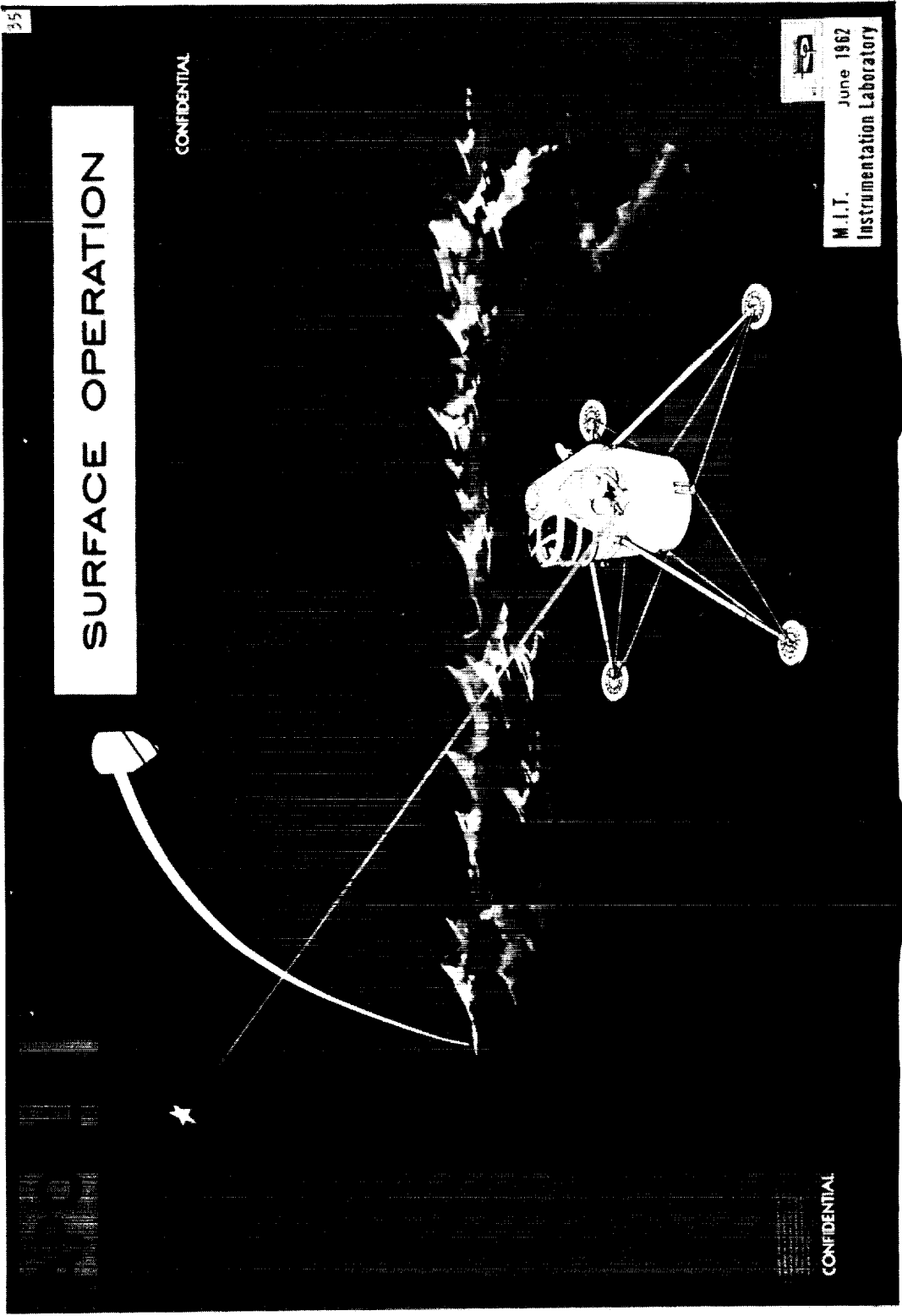
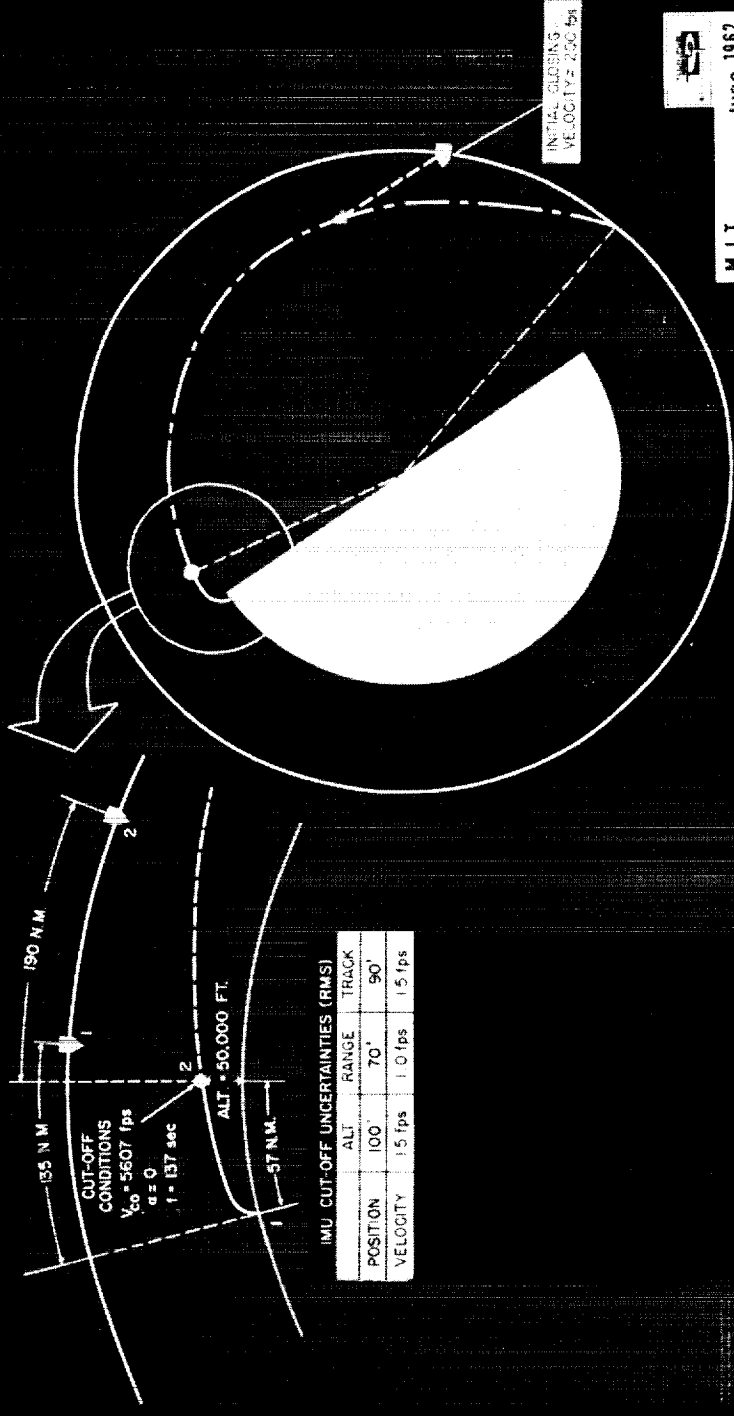


Figure 15

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# L.E.V. LAUNCH & ASCENT TRAJECTORY



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Figure 16

## I. Rendezvous Maneuver

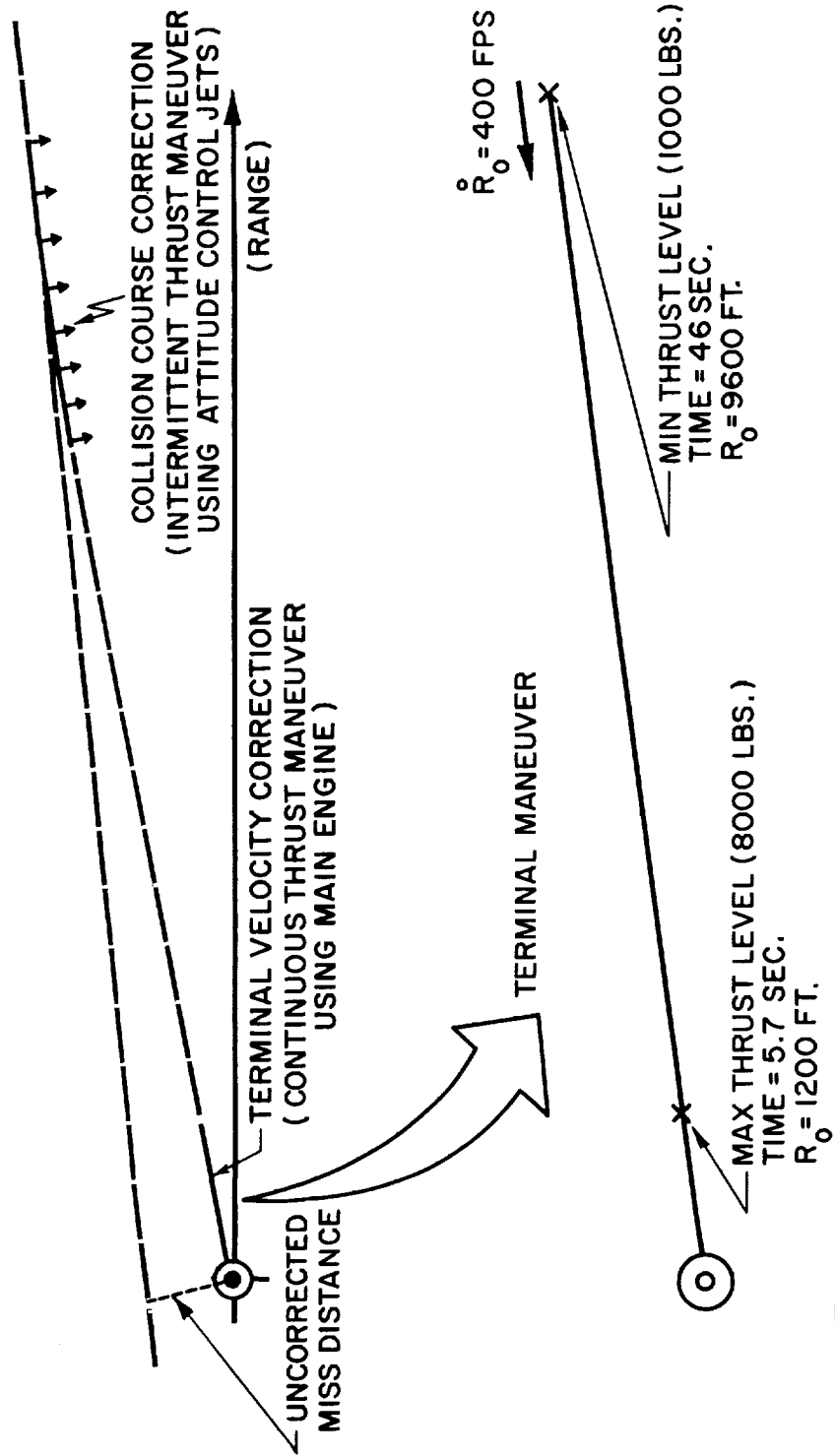
Figure 16 shows the type of rendezvous orbit that is attractive. This transfer trajectory is  $160^\circ$ , which is far enough from the Hohmann path so that a fairly efficient means of maneuvering out of the plane is possible. At the same time, it is close enough so that not very much extra energy is consumed. The result of the launch burn-out errors is equivalent to missing the target by about 0.1 mile in track, 1 mile in altitude, and 3 miles in range. This is a very close and very easy rendezvous as concerns acquisition and the amount of maneuver required. The closing velocity is about 200 fps, in contrast to earth orbital rendezvous, which is about 600 fps.

In the normal case, where one has only a mile error, normal to the path, a velocity correction such as is shown in Fig. 17, is employed. At a substantial distance, in the vicinity of about 60 miles, the 100-pound thrust attitude control jets are used to apply approximately a 5 fps velocity correction, which establishes a collision course. The terminal velocity correction is applied with the main engine at close range as shown in Fig. 17 also.

It is interesting to consider that, if the guidance system should have failed while on the lunar surface, and launch were accomplished using the back-up attitude reference and body-fixed accelerometer with one percent errors, the burn-out errors would increase about 50 times. The result would be about a 50-mile miss in the rendezvous region. But recovery is still possible in a number of ways. Because of inefficient manual steering, an excessive amount of fuel would probably have been consumed during launch. Hence there might not be sufficient fuel available to complete the rendezvous maneuver. In that event the service module could be maneuvered back into the rendezvous, but at the expense of reducing the safety margin by using up some of its capability for returning to the earth. One option in such a case is to leave the LEM behind after transferring the men to the



# RENDEZVOUS MANEUVER



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Figure 17

command module, thereby reducing the fuel requirement to get back to earth by foregoing the comfort of the extra living space of the LEM.

J. Rendezvous Sensing

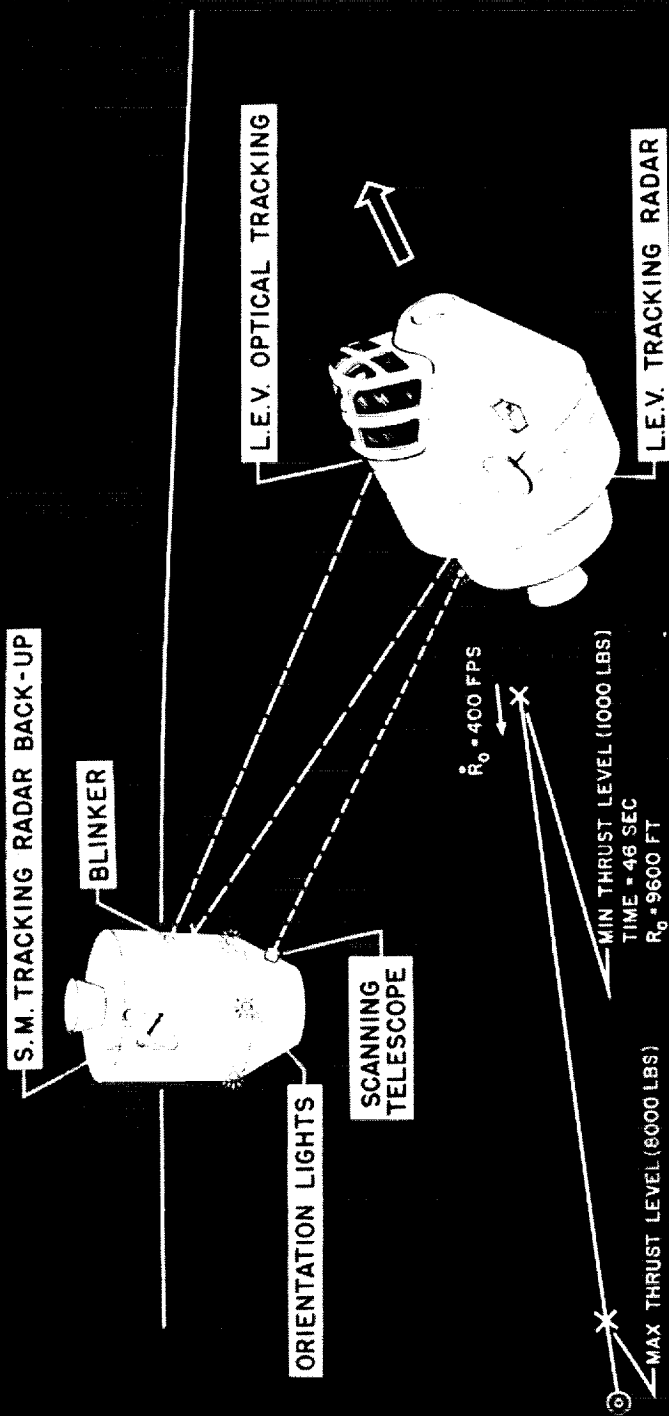
Figure 18 shows the means of sensing for rendezvous. Under normal circumstances the tracking radar provides the range and angle data required by the AGC to compute the rendezvous maneuvers. The thrust maneuvers are controlled inertially by the IMU and AGC, but the radar will probably be employed simultaneously in a monitoring function.

The mother ship will be visible to the LEM crew at all times due to reflected sunlight or its own flashing light. A 400-w light flashing once per second on the command module at a distance of 65 miles from the LEM would have the brightness of Sirius. At a distance of 100 miles, it would have the brightness of Vega, which is still the second brightest star. At 250 miles, which is the minimum distance to line up for a rendezvous, if there is a possibility of a miss by 50 miles, it would have the brightness of Polaris, which is still a bright object. Sunlight will cause the mother ship to appear equally bright.

Several alternate modes are therefore possible. During radar tracking, if the reticle is slaved to the radar boresight axis, the astronaut can monitor tracking performance. If radar fails, angle tracking is possible by manual control of the reticle. Perhaps a mode is feasible wherein the radar is slaved to the reticle to permit ranging when tracking loops fail. The ultimate back-up for angle tracking from the LEM appears to be the technique of judging apparent motion of the mother ship against the starry background. Langley Research Center has been working on a rendezvous technique whereby the LEM would be maneuvered to zero the apparent mother ship motion. This appears to be a feasible method of establishing a collision course at some penalty in terms of maneuver fuel.

# RENDEZVOUS PHASE

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Figure 18

Sensing for the terminal rendezvous maneuver without radar would be accomplished using the running lights on the command/service module. The pilot would wait until he could see the running lights of the mother ship from which he could judge the size of the vehicle. He would then make a short rendezvous maneuver involving only a couple of city blocks but, having rather poor velocity sensing capability, he would probably require several maneuver commands to effect a rendezvous.

Since rendezvous may have to be completed by command/service module maneuvers, the LEM is also equipped with a flashing light and a set of running lights. Equipping the mother ship with a tracking radar is also under consideration. Visual tracking from the command module is readily accomplished using the scanning telescope.

The possible recovery from a variety of failures is of deep concern when contemplating the Lunar Orbital Rendezvous Mission. If everything is functioning fine, the rendezvous and the other maneuvers by the excursion vehicle apparently are made with comfort. If not, the system provides ways of somewhat uncomfortably recovering the LEM and successfully getting the crew back to earth.