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SPARK PLUG—the Electronics Division of GENERAL MOTORS CORPORATION

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Houston, Texas

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EXTENDED APOLLO SYSTEMS UTILIZATION STUDY

Final Report

Volume 14

Guidance and Navigation

16 November 1964

"Available to U.S. Government Agencies and U. S. Government Contractors Only"

Prepared by

AC Spark Plug

The Electronics Division
of
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Milwaukee, Wisconsin 53201

~~Downgraded at 3-year intervals; declassified after 12 years; DOD DIR 5200.10.~~

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FOREWORD

This volume constitutes a portion of the Apollo X Study Final Report (SID 64-1860) prepared as part of the Extended Apollo Systems Utilization Study conducted by the Space and Information Systems Division of North American Aviation, Inc., for the National Aeronautics and Space Administration's Manned Spacecraft Center under Contract NAS9-3140, dated 6 July 1964. S & ID acknowledges the outstanding technical contributions made to the study by a number of companies; these organizations are identified below along with the title of the report for which they are responsible.

The final report has been prepared in the series of 23 volumes listed below:

1. Summary
2. Mission and Performance Analysis
3. Experimental Programs
4. Configurations, Structures, and Weights
5. Mission Plans and Functions
6. Environmental Control System (AiResearch)
7. Fuel Cells (Pratt & Whitney)
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20. Subsystems Supplement
21. Reliability and GSE
22. Development Planning
23. Condensed Summary

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PREFACE

The NASA Manned Spacecraft Center has for the past several years examined the application of the Apollo spacecraft to missions alternate to the basic lunar landing mission for which it is currently designed. The overall objective of these studies is an assessment of the advantages and disadvantages associated with the application of developed hardware to these other potential missions.

NASA studies of near-term applications, which represent possible initial extensions to current Apollo capabilities, include use of the Apollo spacecraft as (1) a space station logistics resupply vehicle (carrying up to six men), (2) an earth-orbital space station or experimental laboratory, (3) a lunar mapping and survey vehicle, etc. This report presents the results of the most recently concluded studies of the Extended-Mission Apollo, in which the application to a 45-day earth orbital laboratory role and to an extended lunar orbit mission have been examined in depth.

The initial Extended-Mission Apollo study, initiated in August 1963 under contract to NASA/MSC, examined the suitability of Apollo as an earth-orbital biomedical/ behavioral experimental laboratory. These experiments were to provide a basis from which man's suitability for protracted space missions could be determined. Three basic configuration concepts, indicated in Figure 1, were investigated throughout the study. Configuration Concept I utilizes only the Apollo command and service modules (CSM), with experimental in-orbit work space made available in the command module by the elimination of one crewman from the current crew size of three. The Apollo CSM subsystems were modified to sustain orbital operations for periods of up to 120 days without resupply. Configuration Concept II consisted of the Apollo CSM plus a 5600 cubic foot laboratory module built within the geometric limits of the LEM adapter. In this concept, the CSM subsystems support the laboratory module functions for the 120-day resupply period. Consequently, Concept II has a laboratory which is dependent on the DSM subsystems. The third configuration (Concept III) is similar to Concept II with respect to the addition of the separate laboratory module within the LEM adapter section; however, the Concept III laboratory was designed to carry its own subsystems; therefore, it is considered to be independent of the CSM subsystems. The subsystems in the CSM-independent laboratory module were designed for 1-year continuous operation with resupply of expendables by the Apollo CSM as required. Included as a part of the original Extended Mission Apollo study was the establishment of detailed development plans including costs, schedules and manufacturing and facilities plans. These development plans were based on an integrated but noninterference relationship with the Apollo program and on maximum utilization of Apollo technology, facilities, etc. The laboratory module, for example, utilized a large percentage of Apollo/Saturn interstage tooling.

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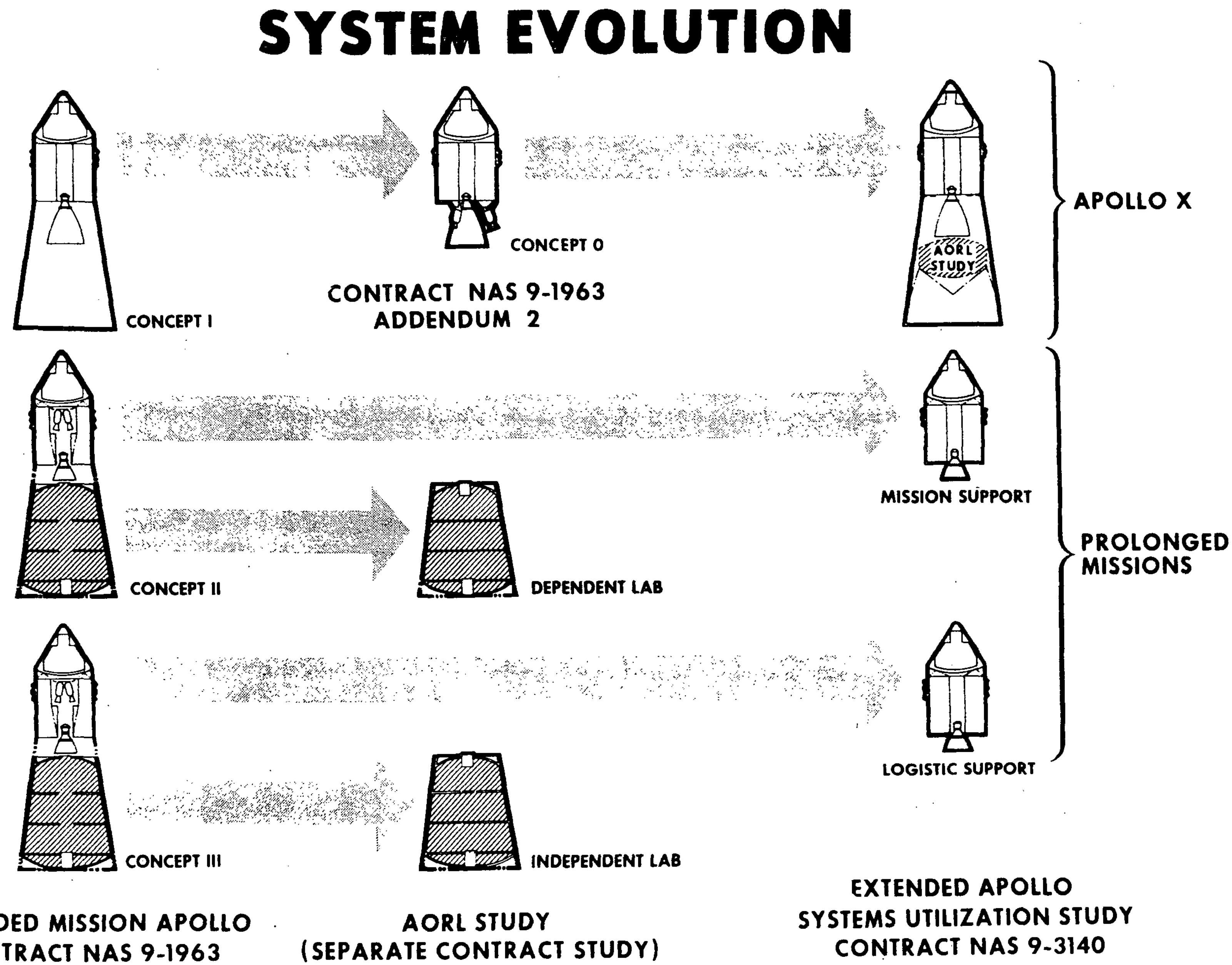

Figure 1. System Evolution

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Results of the initial Extended-Mission Apollo studies revealed several additional factors that warranted further investigation. The required 120-day resupply cycle dictated the use of advanced subsystem concepts in several areas. For example, the existing fuel cell electrical power system in the Apollo was replaced with a solar cell-battery power system to eliminate the excessive weight and volume penalties attributable to the fuel cells and associated expendables. Additionally, a molecular sieve was employed in the environmental control system to eliminate the large weight and volume attendant with using the existing Apollo lithium hydroxide CO_2 removal system for 120 days. Consequently, the limiting mission duration which might result from using only current Apollo subsystem concepts was not known. Also, although the feasibility of extending the life of the current Apollo subsystems had been established in the initial study, specific techniques for accomplishing this life extension had yet to be determined.

For these reasons, further studies were initiated as an addendum to the Extended-Mission Apollo contract. The purpose of these additional studies was to define the design characteristics and the maximum earth orbital mission duration of the Apollo CSM, assuming restriction to use of existing Apollo subsystem concepts. This configuration was identified as Concept 0. Included in this study was examination of each subsystem to determine the life extensions possible through the addition of spares and redundancies. The study concluded that the earth-orbital duration capability of Concept 0 was approximately 90 days (based on Saturn IB payload limits).

After review of these findings, NASA initiated the current Extended Apollo Systems Utilization Study, one part of which is to define in depth the design and operational characteristics of a vehicle based on the identical subsystems approach employed in Concept 0, but limited arbitrarily to a 45-day maximum mission duration. This vehicle, identified as Apollo X, appears to represent the most logical next-step in manned spacecraft duration capability beyond the current Apollo and Gemini programs.

As part of this same study program, the characteristics of the CSM configurations associated with the dependent and independent laboratory module (as in Concepts II and III) are being investigated relative to earth orbital and lunar missions of greatly extended durations. Under these Prolonged Mission studies, the Concept II CSM is identified as a Mission Support vehicle, since it supports the laboratory (AORL) by resupplying subsystems, crew members, and expendables. The Concept III CSM is identified as a Logistics Support vehicle since it resupplies only expendables and crewmen, with the CSM subsystems remaining in a quiescent state after docking to the laboratory module. The laboratory modules are being investigated separately (i.e., by another contractor) under a current MSC study program. The definition of the characteristics of the Apollo X laboratory modules is also included as part of this separate AORL study effort.

The primary objective of the Apollo X study has been to define a standard spacecraft design capable of alternately performing extended-duration lunar or earth-orbital missions of NASA near-term interest. Included in this primary objective were studies

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to determine specific modifications required for each subsystem to accomplish the extended missions. Subsystem qualification test programs, which would substantiate the analytically-derived extended-life subsystems, were also defined. In the experimental area, emphasis was placed on definition and integration of experimental packages (for earth-orbital missions) based on experimental lists provided by NASA at the initiation of the study. As an aid to NASA evaluation of the desirability of or requirement for, the laboratory module, studies were also performed defining development factors based on CSM changes attendant with and without the laboratory module. All development factors investigations were based on an integrated, but noninterference relationship with the current Apollo program.

The Apollo X study was initially focused toward consideration of earth-orbital experimental missions of up to 45 days' duration; spacecraft and subsystems requirements were to be developed through the determination of the experimental mission requirements. Secondary studies were aimed at determining the modifications required to the 45-day earth-orbital CSM design when applied to a lunar survey mission (including up to 28 days in lunar polar orbit). Several weeks after study initiation, however, NASA directed that equal emphasis be given to the lunar mapping mission, and requested that primary emphasis be placed on the definition of a standard CSM suitable for both the lunar and earth-orbital missions. Accordingly, a reevaluation was made of previously developed earth-orbital mission requirements. This reevaluation indicated that the lunar survey mission imposed overriding requirements in most subsystem areas.

As previously indicated, the overall study approach was based on derivation of a standard multimission CSM applied to different vehicle configuration arrangements, as indicated in Figure 2.

In the basic configuration, experimental work space is made available in the command module by elimination of the third crew member and couch, and by providing for in-orbit storage of the center couch under the pilot (left-hand) seat position. This configuration is adaptable to earth-orbital experimental missions, but is not suitable for the lunar missions since it would be required that the service module be completely filled (with propellant, fuel cells, cryogenic tankage, etc.) with no remaining volume available in the SM for mission-oriented equipment. The basic configuration has, however, the definite advantage of being able to provide an experimental laboratory for early flights (such as biomedical and human factors experiments) without schedule dependency on a separate laboratory module. Additionally, these early flights would result in a high-confidence base for performing subsequent missions with a laboratory or lunar survey mission module.

The laboratory configuration has increased volume availability as indicated in Figure 2, through the addition of a separate laboratory module carried within the LEM adapter section. The separate laboratory module (approximately 1200-1500 cubic feet volume) is sized by the provision of space inside the LEM adapter for a second module of identical geometry. This configuration can be applied to all of the earth orbital and lunar missions of current interest since mission-peculiar payloads can be installed

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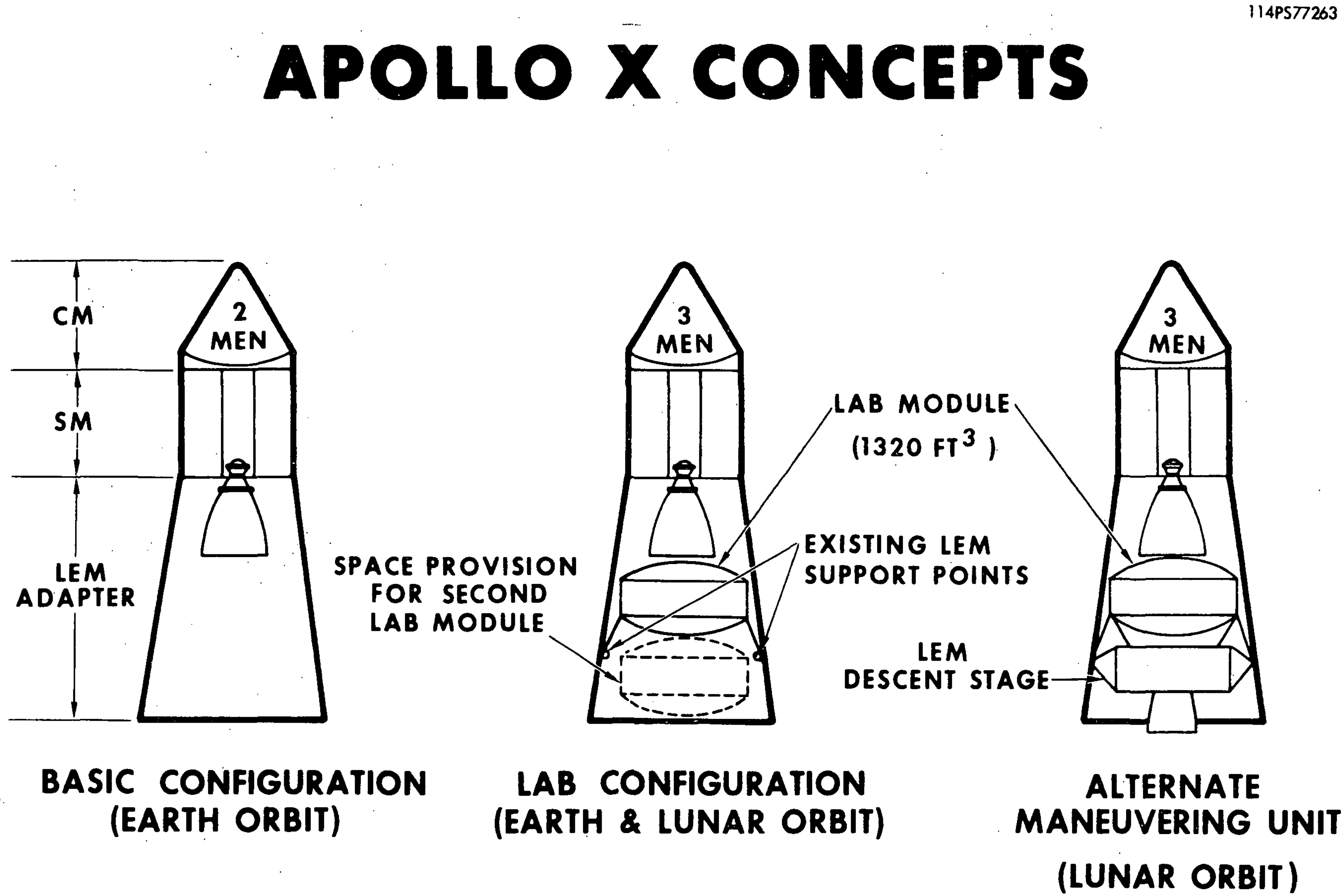

Figure 2. Apollo X Concepts

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within the laboratory module. The configuration is extremely versatile in that the experimental volume availability can be controlled by the addition of either one or two laboratory modules. In addition, a single module can be mounted on the LEM descent stage adapter, thereby permitting additional payload for lunar missions, orbital change capability, or for providing a lunar landing cargo module.

All technical studies were predicated upon the basis of the Apollo Block II configuration. Since past studies revealed that subsystems characteristics materially influenced configuration design, heavy emphasis was placed on subsystems definition as indicated in Figure 3. In this regard, not only primary subsystem components were examined in detail, but secondary components were also fully analyzed. In certain instances where it was found that the addition of spare or redundant Apollo subsystem components might not suffice for the extended-life application, product improvement areas were identified and tentative solutions were established.

Previous studies had identified in-flight maintenance as a major study problem area in view of the number of spares and redundant components used for subsystem life extension. Correspondingly, in-flight maintenance was examined with regard to possible methods of implementing these spares and redundancies as well as determining the overall demands on crew time. These requirements were integrated into crew and experimental station-keeping activities and served as a constraint in the experimental scheduling area. In the experiment design and packaging analyses, it was possible to examine the biomedical and behavioral measurements in great detail through the use of baseline data established in the initial Extended-Mission Apollo study. It is believed that the Apollo X biomedical and human factors experiments as now defined are sufficiently detailed to permit the establishment of equipment specifications.

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APOLLO X STUDY APPROACH

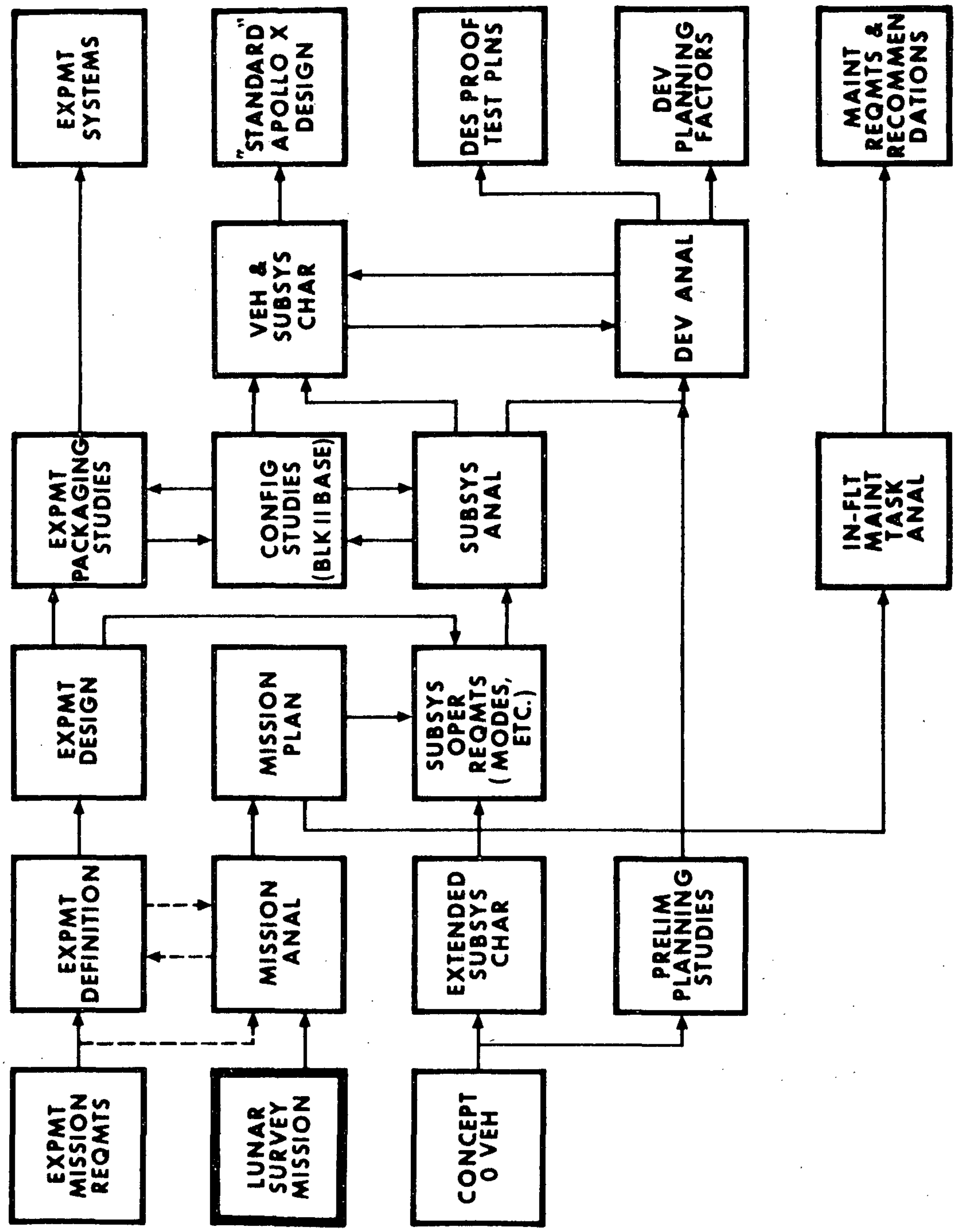

Figure 3. Apollo X Study Approach

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I. INTRODUCTION

PURPOSE

This report details the results obtained by AC Spark Plug for use by S & ID to select the optimum Guidance and Navigation (G & N) System for use during the extended Apollo (Apollo X) lunar polar orbit mapping mission. It also applies the defined G & N System to the earth polar and low inclination orbital Apollo X missions.

SCOPE

Constraints set by S & ID limited this study to the evaluation of the Apollo Block II Guidance and Navigation System applied to the extended Apollo X missions and to the definition of means of system life extension without major system modifications. This report describes the system in terms of equipment function, mission operation, reliability and maintenance, and physical parameters. Parametric studies have also been included for the defined missions. This study excludes consideration of the Apollo Guidance Computer, which is covered in a separate report.

CONCLUSIONS

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This study shows that the Block II Apollo Guidance and Navigation System can provide the lunar orbit navigation and local vertical stabilization requirements. Under the constraint of minor modification of the existing Block II G & N, it is also shown that the Apollo Block II Guidance and Navigation System can provide the apportioned reliability of .9975 for the defined lunar polar orbit mission when the minor system modifications are made and 91 lb of spares and redundancy are added.

The study also illustrates that the additional equipment required is very sensitive to the particular level of reliability assigned to this study. The present optics reliability barely exceeds that allocated to the G & N (less computer). Since the optics was not considered sparable (and not subject to major modification) the remainder of the G & N assembly reliabilities were required to approach unity. In addition, approximately 137.5 hours of a total 157 hours of mission usage time accumulated by the G & N Inertial Measurement unit (in the Apollo X mission) are for providing local vertical stabilization during lunar polar orbit as opposed to a total usage time of 33 hours for basic guidance in the present lunar orbiting and rendezvous mission. These combinations of requirements manifested themselves in two major areas; sparing of the IMU, and a redundant set of electronic coupling units. ~~Conf.~~

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The inherent reliability of the existing Block II G & N (less guidance computer) with minor modification and zero pounds of spares/redundancy in providing basic guidance from earth launch through establishment of lunar polar orbit and return to earth (with full backup autonomous navigation) is .990. Optimization of the various circuitry and reliability studies of the presently evolving Block II design are expected to improve this basic guidance capability further. Future tradeoffs between related C/M subsystems and the various backup and alternate capabilities they represent, such as the G & N and SCS, in providing vehicle stabilization may permit alleviation of apportionments and usage times in both areas. Conf.

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II. REQUIREMENTS AND DEFINITION

REQUIREMENTS ANALYSIS

MISSION PROFILES

The three basic mission profiles considered for the Apollo X study were the Lunar Polar Orbit Mapping Mission, the Earth Polar Orbit Mission, and the Low Inclination Earth Orbit Mission. Because the lunar polar orbit mapping mission is the longest and the most complex, it was used to establish the spares/redundancy requirements and to identify the critical components and subassemblies of the Apollo Block II G & N System. Then, the capability of this system design was used to determine its adequacy in the other two mission profiles.

The mission profiles are defined in Table 1, and the local vertical stabilization cycle to be used during the lunar and earth polar orbit missions is shown in Table 2. These mission profiles are further detailed in the Appendix.

RELIABILITY REQUIREMENTS

The required reliability of the Apollo G & N System, exclusive of the guidance computer, is listed in Table 3.

DESIGN CONSTRAINTS

Certain constraints have been imposed upon the G & N System design so that an adequate system may be developed for the defined Apollo X missions with a minimum of time and cost.

Mission

Since the lunar polar orbit mapping mission is the most severe application of the G & N System in the Apollo X study, the system configuration must meet all the reliability and performance requirements of this mission. This same configuration will be used for the two earth orbital missions.

The primary function of the G & N System in these missions is guidance. Navigation is the prime responsibility of ground tracking, except for initial lunar orbit determination where a combination of ground tracking and G & N System navigation can be used. None of the navigational capabilities of the G & N System will be deleted, however, and thus it will serve as a backup to the ground tracking system.

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Table 1. Apollo X Mapping Mission Profiles

Mission Phase	Time	G & N Functions
Lunar Polar Orbit Mapping (Profile 1)		
Launch and Parking Orbit Injection	12 minutes	Monitor launch, compute abort trajectories, and take over guidance, if necessary.
Earth Orbit	3 hours	Backup navigation.
Translunar Injection and Plane Change	5 minutes	Monitor thrusting, compute abort trajectories, and take over guidance if necessary.
Translunar Coast	72 hours	Midcourse ΔV corrections and backup navigation.
Lunar Polar Orbit	28 days	Orbital corrections during first few orbits and local vertical stabilization during mapping. Correct lunar polar orbit to 80 nm ± 1.1 nm (1σ). Stabilize CSM-Lab to track local vertical within ±0.125 degree deadband, all axes ±0.02 deg/sec drift rate within deadband.
Transearth Injection	3 minutes	Guide spacecraft.
Transearth Coast	1 hour	Backup navigation.
Plane Change	2 minutes	Guide spacecraft
Transearth Coast	71 hours	Midcourse ΔV corrections and backup navigation.
Entry	18 minutes	Guide spacecraft.
Earth Polar Orbit (Profile 2)		
Launch and Parking Orbit Injection	640 seconds	Monitor launch, compute abort trajectories, and take over guidance if necessary.
Parking Orbit (100 nm)	560 seconds	Backup navigation.
Transfer Injection	392 seconds	Monitor thrusting, compute abort trajectories, and take over guidance if necessary.
Transfer Trajectory	40 minutes	Backup navigation.
Earth Polar Orbit Injection	2 minutes	Guide spacecraft. Circularize orbit to 200 nm ± 10 nm.
Earth Polar Orbit	45 days max.	Orbital corrections, local vertical stabilization. Backup navigation.
Deboost	18 seconds	Guide spacecraft.
Entry	14 to 21 minutes	Guide spacecraft.
Low Inclination Earth Orbit (Profile 3)		
Launch and Parking Orbit Injection	623 seconds	Monitor launch, compute abort trajectories, and take over guidance if necessary.
Parking Orbit (80 nm)	90 minutes	Backup navigation.
Transfer Injection	10 seconds	Guide spacecraft.
Transfer Trajectory	40 minutes	Backup navigation.
Earth Orbit Injection	10 seconds	Guide spacecraft. Circularize orbit to 200 nm ± 10 nm.
Earth Orbit	45 days	Orbit correction and G & N checkout before entry. (Correct orbit after 22 days to 200 nm ± 10 nm.)
Deboost	18 seconds	Guide spacecraft.
Entry	14 to 21 minutes	Guide spacecraft.

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Table 2. Lunar and Earth Polar Orbit Mission Local Vertical Stabilization Cycle

Operation	Elapsed Time		Cumulative Time	
	Hours	Minutes	Hours	Minutes
Start at Pole	00	00	00	00
Mapping Interval	1	00	1	00
No Mapping	5	15	6	15
Mapping Interval	0	30	6	45
No Mapping	5	15	12	00
Mapping Interval	1	00	13	00
No Mapping	5	15	18	15
Mapping Interval	0	30	18	45
No Mapping	5	15	24	00

Cycle to 82.5 hours of mapping.

Table 3. Mission Probability of Success

Mission	Probability of Success	Remarks
Lunar Polar Orbit Mapping	.99779	This requirement was for the 28-day flight mission. A later requirement has reduced this requirement to a probability of success of .9975.
Earth Polar Orbit Mapping	.99908	It was desired that the maximum number of scheduled mapping cycles be determined, so that the reliability requirements would be met after entry. This requirement has been changed to the determination of reliability versus time for several different mapping cycles.
Low Inclination Earth Orbit	.99988	This requirement was for a 45-day mission. A later requirement was to determine what overall reliability could be obtained for the 45-day mission using the G & N System designed to meet the lunar polar orbit requirements.

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To account for the additional stresses experienced by the equipment, the equipment failure rates during mission thrusting phases will be multiplied by the following "K" factors.

Mission Phases	"K" Factor
Launch	3
Translunar Injection and Plane Change	2
ΔV Corrections	2
Deboost	2
Transearth Injection and Plane Change	2
Entry	2
Coast	1
Equipment Deenergized	0

G & N Modification

The G & N System will conform to the Apollo Block II System as closely as possible. Only minor system modifications are to be considered, and none will involve major G & N or spacecraft redesign.

Spares/Redundancy and Weight/Volumetric

System improvements that will not add weight or volume to the system will be explored before considering additional equipment. Spares and/or redundancy deemed necessary to meet the required mission reliability and performance will be kept to a minimum in order that the additional weight and volume penalties be minimized. Sparing is considered preferable to redundancy because of mechanization and packaging considerations. The final selection of spares/redundancy should include consideration of a maintenance analysis of the equipment.

ASSUMPTIONS

Certain mission and equipment assumptions are required to proceed with the study. Some of the more important ones are:

- On the earth orbital missions, the first orbit may be used for orbit verification and corrections.

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- Ground tracking and star occultation will be used for lunar orbit determination, and landmark tracking and star elevation measurements will only serve as backup.
- Minor computer program modifications will be made to enable the local vertical hold to be accomplished.
- The Apollo Block II G & N System configuration (presently still under design and development) will not change appreciably from the current configuration.
- The stressed module reliabilities, as defined by MIT/IL for Block I, are applicable to similar Block II modules. Reliabilities are based on average failure rates and parts counts where stressed failure rates do not exist.
- Launch time is time zero for all part and module reliabilities.
- Sparing cannot take place 1 hour before deboost to lunar orbit, 1 hour before entry, or during any thrusting phases. The times prior to all other thrusting phases were not considered critical since thrusting could probably be postponed if sparing were required.

ENVIRONMENTAL CONDITIONS

The basic environmental conditions for the lunar Apollo mission have been summarized in NAA Specification MC 999-0051, Apollo Environmental Design and Test Requirements, and in NASA Document ND 1002037, Apollo Airborne Guidance and Navigation Equipment Environmental Qualification Specification.

The Apollo X G & N equipment is expected to be subjected to the same conditions during a flight as shown in the above specifications, with the following exceptions.

- Time spans of exposure to various dynamic and climatic environments will be lengthened in some instances.
- The spacecraft crew compartment and various portions of the G & N equipment will be subjected to an atmosphere composed of 50 percent nitrogen and 50 percent oxygen at 7 psia and 70° F over the entire mission.

G & N equipment design is not expected to be affected by the above environmental changes.

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BLOCK II G & N SYSTEM DEFINITION

The Apollo G & N System is capable of independently performing two basic functions: guidance and navigation. To accomplish the guidance function, three accelerometers are orthogonally mounted on a gyroscope-stabilized platform that is isolated from spacecraft angular motion by a gimbal system. These accelerometers sense velocity changes, while the gyro-stabilized platform and gimbal system senses changes in the spacecraft attitude. Together, they provide steering and thrust control information. A sextant and a scanning telescope are provided to take sightings on celestial bodies and landmarks. These sightings are used to determine spacecraft position and velocity and to establish a precise alignment of the stable platform.

A digital computer serves as the primary data processing element of this system. It contains a catalog of celestial objects, and is programmed to calculate steering and thrust commands using information obtained from optical sightings or ground tracking stations. The G & N System together with the Reaction Control System form a loop that controls the spacecraft attitude and translation. A second loop, formed by the G & N System and the Service Propulsion System, provides control of spacecraft thrusting.

The Apollo Block II G & N System is divided into three major subsystems: inertial, optical, and computer. A functional diagram of this system is shown in Figure 4. The three subsystems, or combinations of subsystems, can perform the following functions:

- Calculate the position and velocity of the spacecraft by inertial measurements and by the following optical navigation sightings:

  Landmark tracking
  Star occultations
  Star — horizon sightings
  Star — landmark sightings

- Store and operate on data received from ground tracking,
- Align the stable member to an inertial reference, using precise optical sightings,
- Calculate necessary velocity corrections and provide control of spacecraft thrusting to effect the required trajectory,
- Control the spacecraft attitude with respect to an inertial reference,
- Provide backup to the Saturn IV B (SIVB) booster guidance system,
- Provide the astronaut with a data display indicating the status of the guidance and navigation problem.

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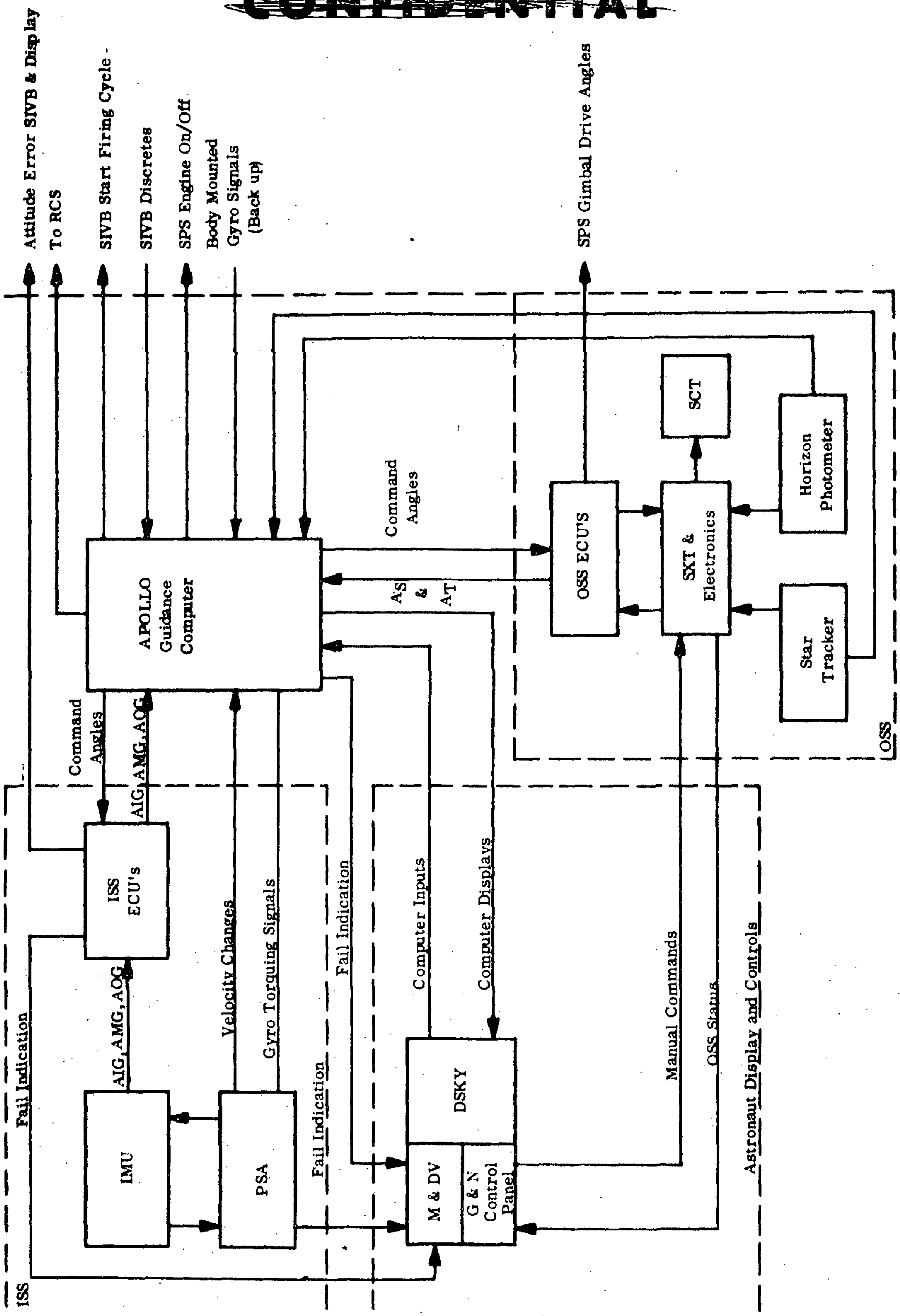

Figure 4. Apollo Block II Guidance and Navigation System

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INERTIAL SUBSYSTEM

The Inertial Subsystem (ISS) is used to establish and maintain an inertial reference that provides the capability to measure linear acceleration and spacecraft attitude with respect to inertial space.

Inertial Subsystem Equipment Definition

This subsystem consists of an inertial measurement unit (IMU), three electronic coupling units (ECU), and the major portion of the power and servo assembly (PSA). The IMU contains a three-gimbal inertial platform containing three stabilization gyros (25 IRIG Mod 2), three accelerometers (16 PIPA), and the associated electronics. The IMU also contains a simple on-off temperature controller, which is used to maintain the 16 PIPA's and the 25 IRIG's at their proper operating temperatures. A thermostat is used to monitor the temperature of the stable member. When a high- or low-temperature condition exists, the thermostat will issue an alarm signal, which is displayed by means of a light.

The ISS portion of the PSA contains the electronics and power supplies necessary to support the IMU. Figure 5 indicates the major elements comprising the inertial attitude reference portion of the ISS.

Linear accelerations along three orthogonal reference axes are sensed by the three accelerometers. Using time reference pulses from the computer, the sensed acceleration is integrated, and information in the form of velocity changes is sent to the computer for use in determining spacecraft velocity and position. A block diagram of a 16 PIPA loop is shown in Figure 6.

The ECU's have two basic functions: (1) converting analog signals representing the IMU gimbal angles to digital signals for the guidance computer, and (2) converting digital signals from the computer to analog commands to coarse align the IMU gimbals. In addition, the ECU's can also provide analog attitude error signals to the SIVB as a backup to the SIVB guidance and attitude error information to the astronaut. A block diagram of one ECU is shown in Figure 7.

Inertial Subsystem Modes of Operation

The Inertial Subsystem has two basic states, standby and operate. In the standby state, only temperature control and suspension for the inertial components are maintained. In the operate state, the inertial subsystem is in one of the following four modes, which are selected by the guidance computer.

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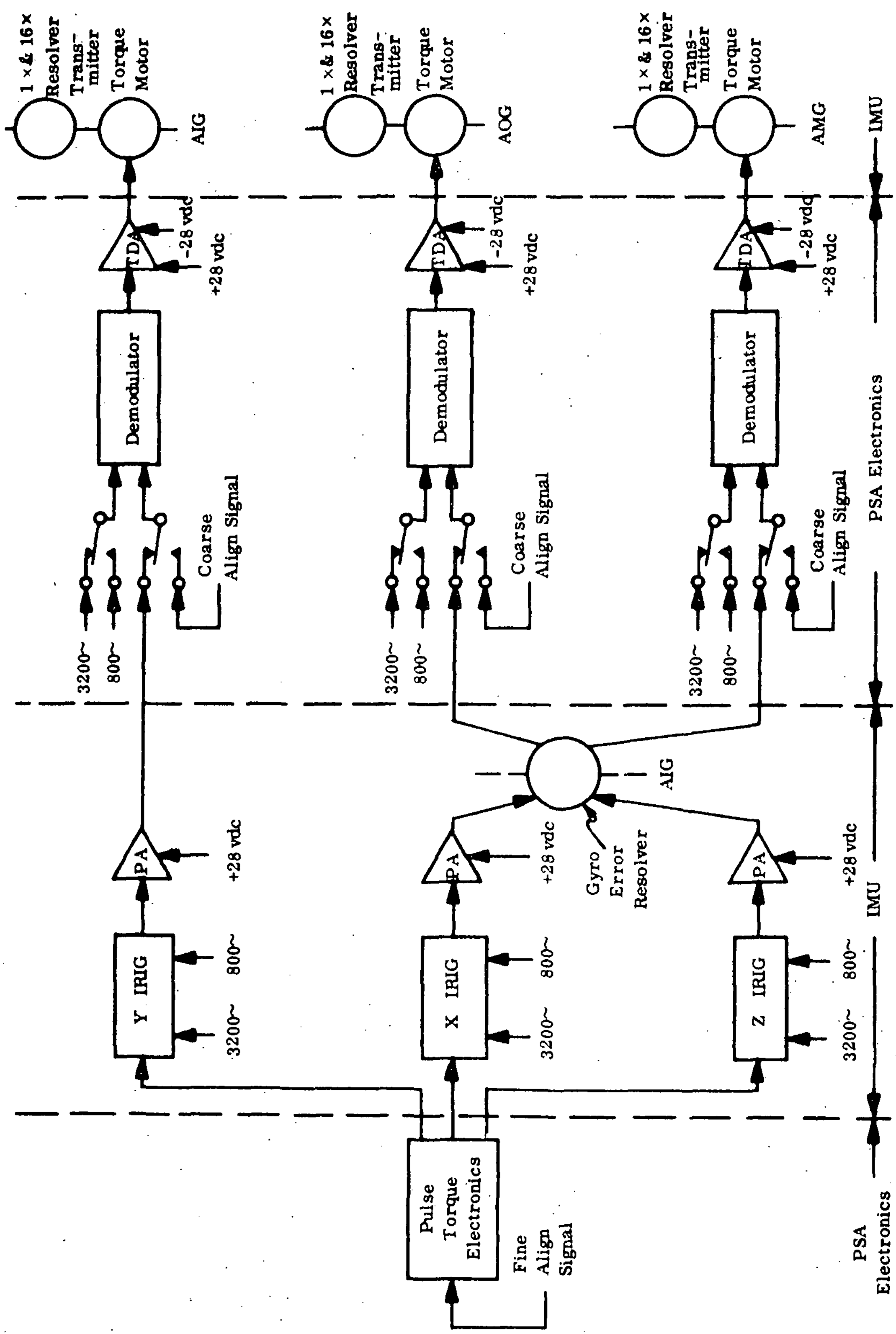

Figure 5. Inertial Attitude Reference

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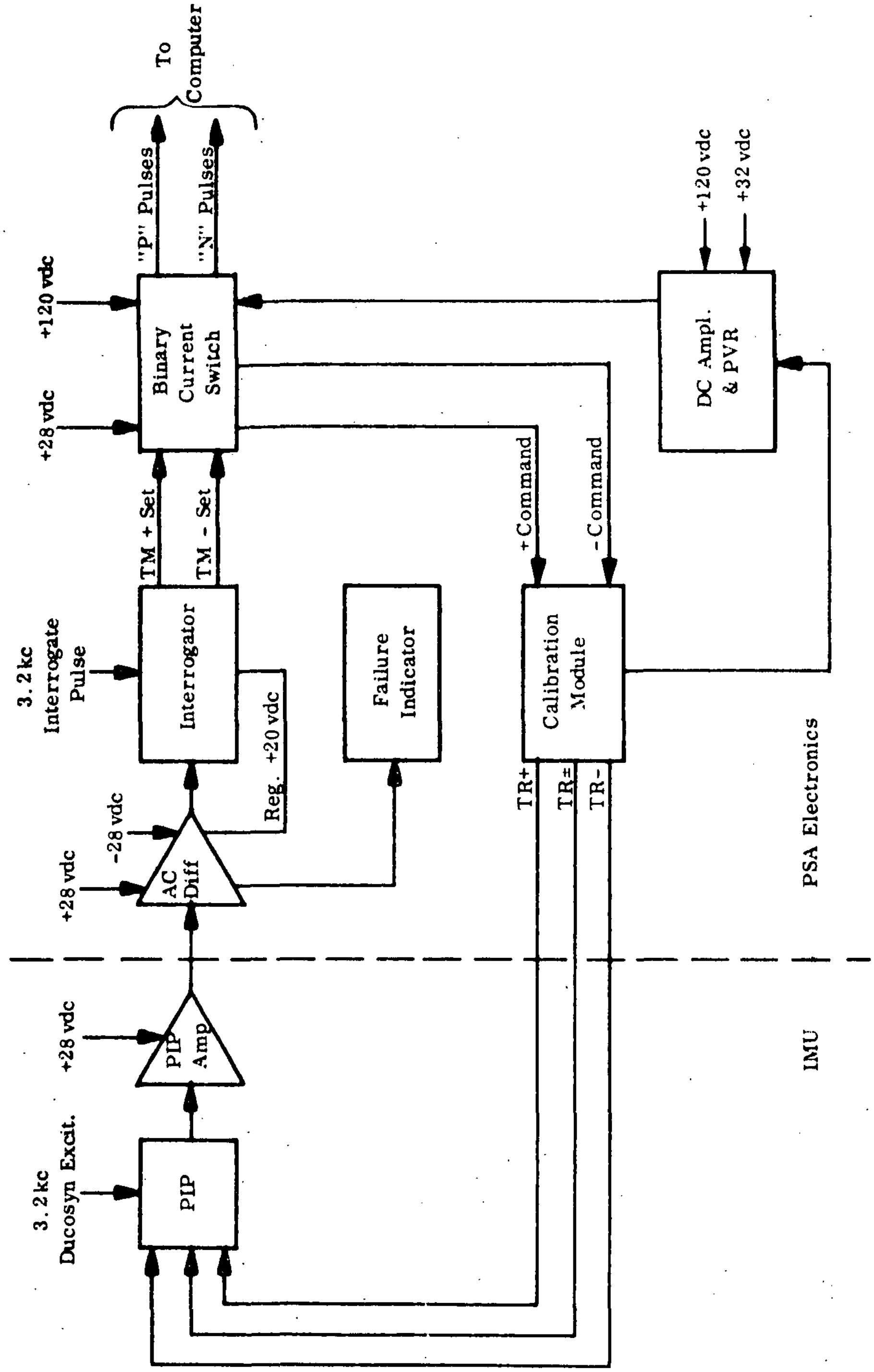

Figure 6. 16 PIPA Loop

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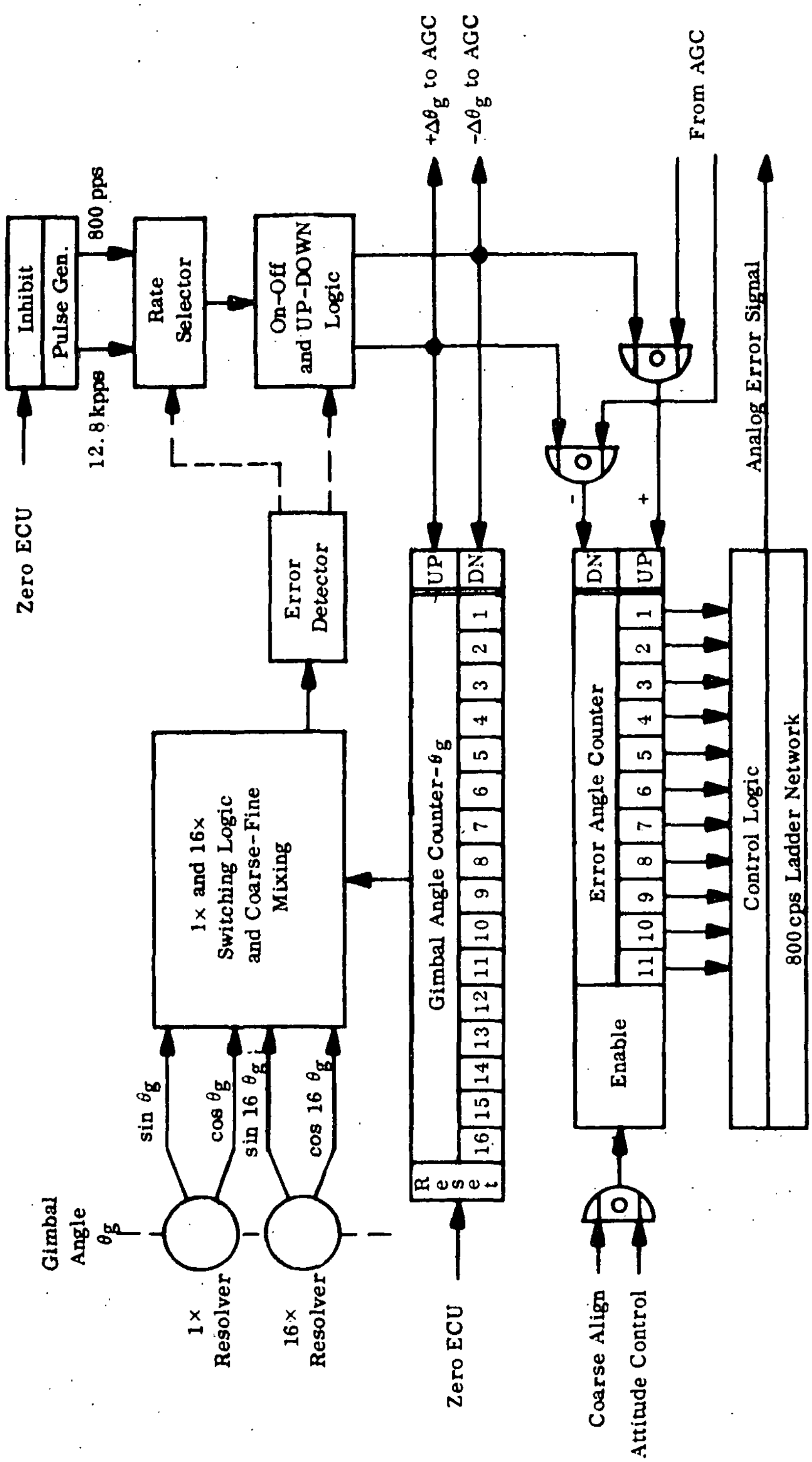

Figure 7. Electronic Coupling Unit

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Zero ECU

In the Zero ECU mode, the gimbal angle counters in each of the three ECU's and in the computer are reset to zero and $\Delta\theta_g$ pulses are inhibited from being transmitted to the computer. The stable member attitude is maintained by the 25 IRIG error signals.

Coarse Align

In the Coarse Align mode, the guidance computer commands the IMU gimbal to a desired orientation. The computer calculates the differences between the present and the desired gimbal angles. These differences are transmitted to the ECU's, where they are converted to analog error signals. The gimbal servos use these error signals to drive the gimbals to the desired orientation. When a gimbal angle changes by an increment of 40 arc-seconds, a pulse is transmitted to the computer by the ECU. A functional diagram of the Coarse Align mode is shown in Figure 8.

Fine Align

In the Fine Align mode, the computer commands a "fine" change in the inertial orientation of the stable platform. The guidance computer commands the pulse torque electronics to torque the floats of the 25 IRIG's from null through a small correction angle. This produces an error signal to the gimbal servos, which change the inertial orientation of the stable platform such that the 25 IRIG floats are again at a null. Fine Align is the only mode in which the IRIG floats can be torqued since in the other three ISS modes, the pulse torque electronics are deactivated. A functional diagram of the Fine Align mode is shown in Figure 9.

Attitude Control

The Attitude Control mode is identical, as far as the ISS is concerned, to the Fine Align mode with the exception that the pulse torque electronics are deactivated. Since the platform is inertially stabilized and the ECU's transmit changes in gimbal angles to the guidance computer, the computer can calculate the attitude of the spacecraft with respect to the inertial reference. The PIPA loops are in operation whenever the ISS is in the operate state. However, under normal inflight conditions, the outputs from the PIPA loops to the computer will only be meaningful when the ISS is in the Attitude Control mode.

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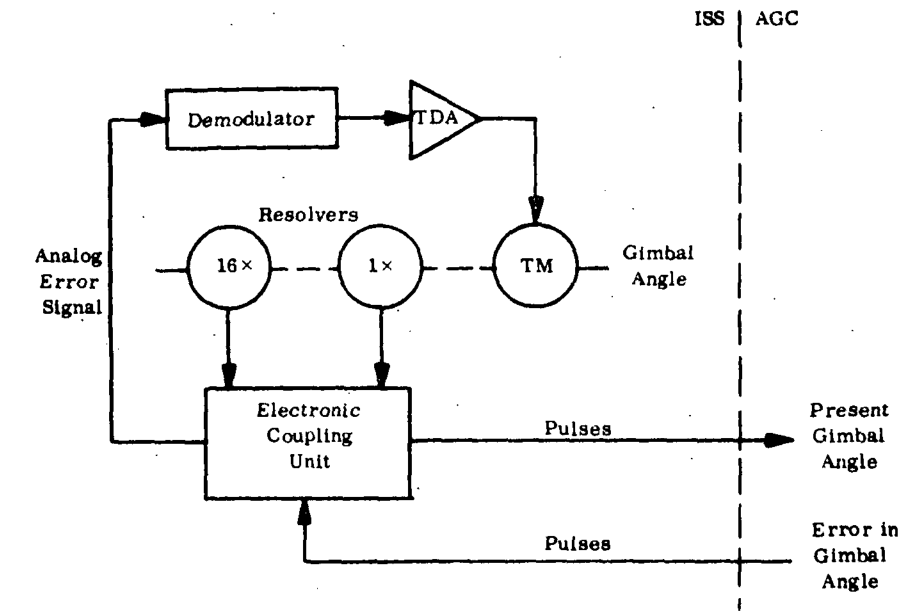

Figure 8. ISS Coarse Align Mode

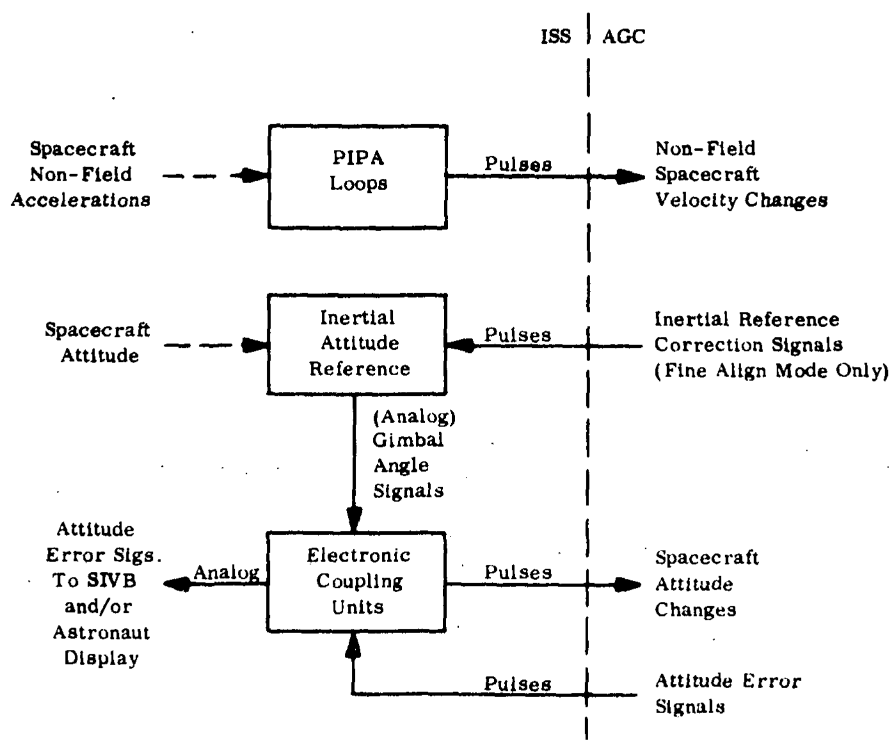

Figure 9. ISS Attitude Control and Fine Align Modes

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OPTICAL SUBSYSTEM

The purpose of the Optical Subsystem (OSS) is to provide instruments with which the astronaut can measure angles between: (1) a star and the horizon, (2) a star and a landmark, (3) a star and the navigation base, and (4) a landmark and the navigation base. Since the IMU is precision mounted to the navigation base, the guidance computer can combine the information received from both the ISS and OSS to perform operations necessary to the mission success.

Optical Subsystem Equipment Definition

The OSS consists of a scanning telescope, a sextant with a star tracker and a horizon photometer, and their associated electronics and controls. The sextant is a dual line-of-sight, 28-power, 2-degree field-of-view instrument. The star line-of-sight (StLOS) can be articulated 55 degrees from the landmark line-of-sight (LLOS) about the trunnion axis, which in turn, can rotate ±270 degrees about a shaft axis, which is coincident with the LLOS. The star tracker is used to keep the StLOS pointed at a preselected star while the horizon photometer, which is parallel to the LLOS, is pointed at the "blue line" of the earth by controlling the attitude of the spacecraft. The scanning telescope is a single line-of-sight, 1-power, 60-degree field-of-view instrument. Its LOS is fully articulated about the same polar coordinates as the sextant StLOS. The available view is limited to approximately a 55-degree, half-angle cone due to spacecraft structure.

The StLOS can be controlled by the astronaut's positioning of the hand controller stick. The hand controller supplies a voltage to an integrator that drives the sextant and the scanning telescope, which is slaved to the sextant. Angular position of the StLOS is transmitted to the guidance computer by means of electronic coupling units (identical to the inertial subsystem electronic coupling units), which encode the angular rotations of the 1× and 64× resolvers on the trunnion axis, and the 1× and 16× resolvers on the shaft axis. The StLOS can also be controlled by the computer via the electronic coupling units.

Optical Subsystem Modes of Operation

The Optical Subsystem has three prime modes of operation: Zero Optics, Manual, and Computer. These modes and their submodes are discussed below.

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Zero Optics Mode

In the Zero Optics mode error signals are generated in the 64× and 1× sextant trunnion resolvers and in the 16× and 1/2× sextant shaft resolvers. These signals are connected to their respective motor drive amplifiers via two-speed switches. In addition, there is scanning telescope trunnion resolver follow-up between the two 1× trunnion resolvers, and as a result, all four servo channels are driven to zero. When switched to the Zero Optics mode all optics ECU counters are cleared. See Figure 10.

Manual Mode

In the Manual Direct mode the sextant trunnion and shaft angles are driven directly by the hand controller. Generally, the scanning telescope follows the sextant position. The Manual mode contains three scanning telescope options (0-Degree Offset, 25-Degree Offset, and Star Line), two sextant options (Direct and Resolved), and an Automatic Star Tracking Option.

<u>0-Degree Offset</u>. In this mode, the 800 cps reference is placed upon the cosine winding of the scanning telescope trunnion 1× resolver. This constrains the scanning telescope to shaft motion only.

<u>25-Degree Offset</u>. The 25-Degree Offset mode is the same as the 0-Degree Offset, except that the 800 cps reference is run through a 25-degree offset transformer to the sine winding of the scanning telescope trunnion 1× resolver. This offsets the scanning telescope LOS 25 degrees with respect to its zero position, thereby providing the capability for visually scanning a 110-degree cone by rotating the shaft. This is shown in Figure 11.

<u>Star Line</u>. In the Star Line mode, the scanning telescope trunnion is slaved directly to the sextant trunnion by means of both resolver and tachometer follow-up.

<u>Direct</u>. Up-down and left-right motion of the hand controller results in polar coordinate image motion. Moving the hand controller to the UP position results in image motion radially inward (that is, trunnion angle increasing). Moving the hand controller to the right results in counterclockwise image motion (that is, shaft angle increasing). The Direct mode is to be used when incrementing the counters to a specified value determined from the Map and Data Viewer.

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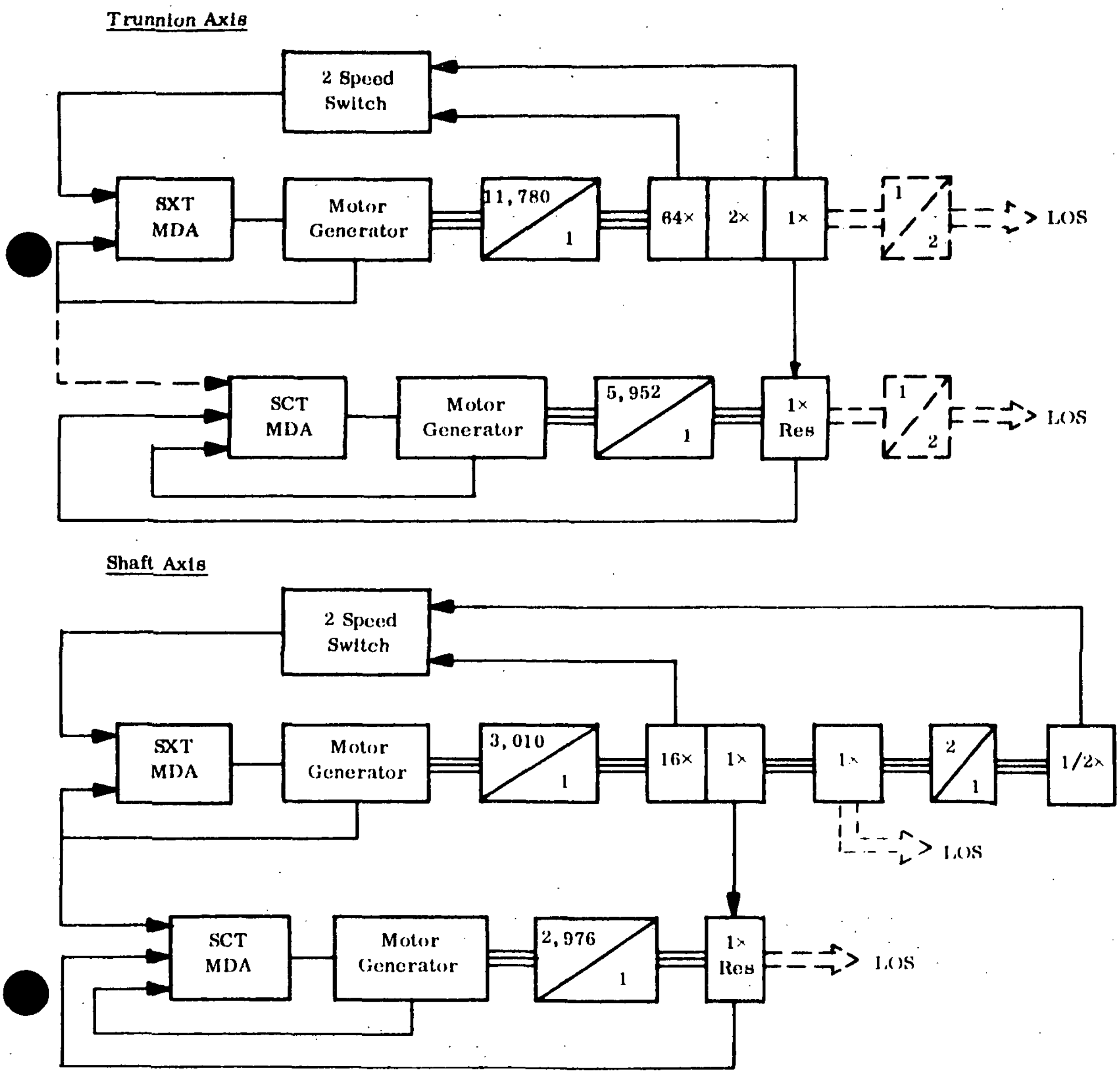

Figure 10. Optical Subsystem — Zero Optics Mode

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Resolved. Up-down and left-right motion of the hand controller results in up-down and left-right image motion. The resolved mode is to be used when attempting to superimpose a landmark and star. See Figure 12.

Automatic Star Tracking. In this mode, a star tracker keeps the sextant StLOS aligned to a star. This mode is unique in that it is the only mode in which the hand controller has the capability of directly driving the scanning telescope trunnion motor drive amplifier. The shaft error goes to the sextant shaft motor drive amplifier via the cosecant amplifier. See Figure 13. The Manual Track mode is to be used when taking star-horizon measurements.

Computer Mode

In the Computer mode the scanning telescope trunnion servo channel follows the sextant trunnion by means of 1× resolver and tachometer follow-up. The Computer mode contains three modes, which are described below.

Computer Zero Optics

The Computer Zero Optics mode is the same as the normal Zero Optics mode with the one exception noted above, that is, tachometer follow-up in the trunnion servo channel.

Computer Operate

In the Computer Operate mode the digital-to-analog converter output commands the sextant in both trunnion and shaft.

Tracker Operate

The star tracker output commands the sextant trunnion and shaft; the trunnion is controlled directly, and the shaft is controlled via the cosecant amplifier.

COMPUTER SUBSYSTEM

The Computer Subsystem, discussed in detail in Volume 16 of this study, consists of the Apollo Guidance Computer (AGC) and two display and keyboard (DSKY) units mounted on the D & C panel. The two DSKY's enable the astronauts to communicate with and control the computer. The DSKY's can enter information into the computer's erasable memory, display information present in the computer, or initiate programs.

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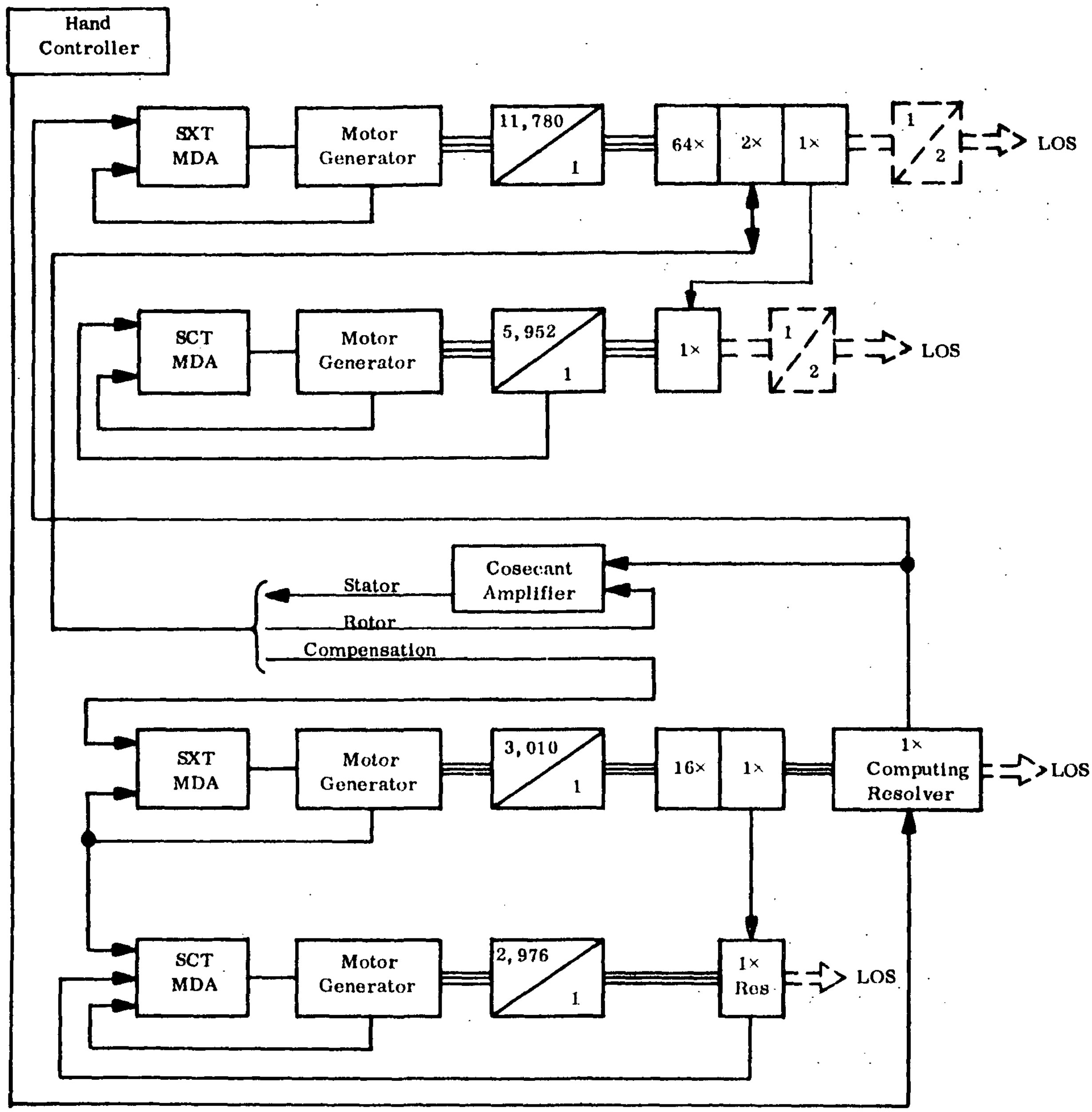

Figure 12. Optical Subsystem — Manual Resolved Mode with Scanning Telescope Following St LOS

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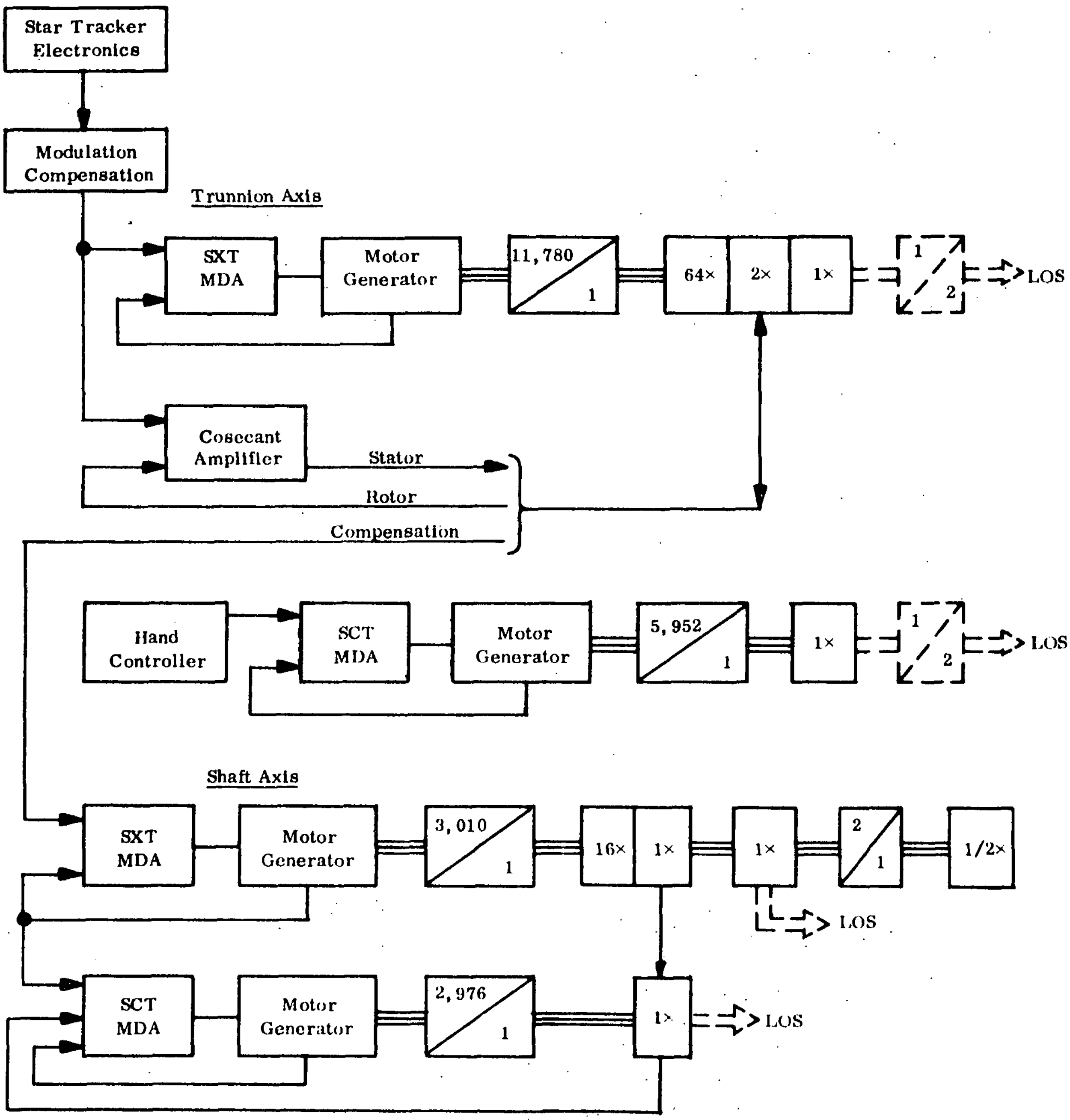

Figure 13. Optical Subsystem — Tracker Operate — Manual Optical Subsystem

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The AGC is the central processor for the G & N System. It is also the clock or basic time and frequency reference of the spacecraft. The AGC can communicate with the sextant and scanning telescope via the ECU's and can also communicate with the astronaut via the DSKY's. In addition, the AGC can count pulses from the PIPA loops and read the IMU gimbal angles. The AGC can send information to earth via telemetry, and receive telemetry information on an uplink. During guidance modes, the AGC can control and stabilize the spacecraft and start and stop the service module engine.

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MISSION OPERATION

Operation of the standard Apollo Block II G & N System may be best illustrated by describing its use in the Lunar Mapping Apollo X mission. This description mainly treats the functional capabilities of the G & N System that are planned to be used. The major phases in the mission are:

- Launch and Parking Orbit Injection,
- Parking Orbit,
- Translunar Inject,
- Transposition and Docking,
- Translunar,
- Deboost to Lunar Orbit,
- Lunar Orbital Correction,
- Lunar Mapping,
- Transearth Inject,
- Transearth,
- Entry.

LAUNCH AND PARKING ORBIT INJECTION

Prior to launch, the IMU is aligned to an earth coordinate system by the guidance computer. The guidance computer fine aligns the IMU stable member such that two of the PIPA loops sense no acceleration due to gravity, and the 25 IRIG's sense a predetermined earth rate. At lift-off the guidance computer automatically switches to a boost monitor program when a discrete is received from the booster (SIVB) guidance system. During boost monitor, the ISS is in the Attitude Control mode and transmits attitude and velocity information to the guidance computer. The computer then calculates the trajectory and attitude error of the spacecraft. The total attitude and the attitude errors are displayed to the astronaut. In case the actual trajectory deviates too far from that desired, either an abort sequence or a takeover of the SIVB guidance can be initiated by the astronaut, by automatic onboard malfunction detection, or by a ground station. When the parking orbit has been achieved, the SIVB engines are shut down.

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PARKING ORBIT

During the earth parking orbit phase, earth based ground tracking is used to accurately determine the spacecraft position and velocity as a function of time. This information is transmitted from the ground to both the G & N and SIVB Guidance Systems. During this phase, the Inertial Subsystem remains in the operate state. Also during the parking orbit phase, the IMU alignment will be updated in the following manner. The ISS will be placed in the Zero ECU mode by the computer, so that the gimbal angle counters in both the computer and ECU's will be reset to zero. The computer then places the ISS in the Attitude Control mode. Using the gimbal angles, the computer calculates the attitude of the spacecraft with respect to the degraded inertial reference defined by the IMU. The OSS is now turned on in the Computer mode. The computer places the OSS in the Zero Optics mode, and then commands the sextant star line-of-sight (StLOS) to an attitude at which a predetermined star should be located. The computer controls the attitude of the spacecraft during this time to allow accurate measurements. The astronaut assures that the proper star is in the sextant's field-of-view by switching to the Manual Resolved SCT Follow StLOS mode, and if necessary, slews the StLOS such that the star is centered in the scanning telescope field-of-view. The astronaut then energizes the star tracker, which "locks on" to the star. The computer records the IMU gimbal angles, the optics shaft angles, and the trunnion angles. In a similar manner, a second star-sighting is performed. The computer calculates the alignment error of the stable member about its three axes, switches the ISS to the Fine Align mode, and pulse torques the IRIG's to correct for the alignment error.

TRANSLUNAR INJECT

The inertial reference will be updated prior to translunar inject. The G & N System will perform the same monitor function as it did for the launch to parking orbit phase of the mission.

TRANSPOSITION AND DOCKING

After translunar inject, the Command and Service Modules are disconnected from the SIVB structure, rotated through 180 degrees, and then docked to the Laboratory Module. The Laboratory Module is then disconnected from the SIVB structure, which is no longer needed. During this phase, the G & N System "holds" the attitude of the spacecraft when the astronaut is not commanding the spacecraft with the Rotational — Translational hand controller. When the hand controller is in its center position, the computer controls the Reaction Control System jets to keep the IMU gimbal angles from changing. Since the stable member is fixed with respect to inertial space, any change in gimbal angle will correspond to a change in spacecraft attitude.

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TRANSLUNAR

During the major portion of the translunar phase, the navigation function will be performed by ground tracking, and position and velocity data as a function of time will be transmitted to the computer, so that, in the event of failure of the communications link, the spacecraft and crew can continue the mission or return to earth by using the navigation capability of the G & N System.

Three velocity corrections are nominally planned during the translunar phase. Prior to these corrections, an alignment of the stable platform will be performed in the following manner: The ISS will be switched to the operate state early in the procedure to allow adequate time for thermodynamic transients on the inertial components to die out. The ISS is switched to the ECU Zero mode and then to the Attitude Control mode.

The OSS is turned on and placed in the Zero Optics mode. The astronaut examines the star field in the scanning telescope for a star that is contained in the computer star catalog. If a catalog star is not found in the 60-degree field-of-view, the astronaut switches to the Manual Direct 25-Degree Offset mode and slews the sextant and scanning telescope about the shaft axis. This enables the astronaut to search a 110-degree field-of-view without moving the spacecraft. Once a catalog star is found, the scanning telescope and sextant are slewed until the star is located on the telescope center line. The Manual Direct St LOS mode is selected, causing the scanning telescope to repeat the sextant St LOS.

The hand controller is operated until the star is located at the center of the telescope field-of-view. In addition, the star will be in the sextant field-of-view. The star tracker is turned on and the sextant "locks on" to the star. The shaft, trunnion, and gimbal angles are recorded in the computer.

A second star is sighted in a similar manner, and the shaft, trunnion, and gimbal angles are again recorded. The computer calculates the stable member alignment error and the gimbal angle errors. The computer switches the ISS to the Coarse Align mode and commands the gimbals, via the ECU's, to the proper angles. The ISS is then switched back to the Attitude Control mode.

Two more star sightings are performed in a similar manner. The computer calculates the error in the stable member alignment with respect to inertial space. The ISS is switched to the Fine Align mode and the IRIG's are torqued to correct for the alignment error.

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Once an accurate attitude reference is established by the ISS, the guidance computer commands the spacecraft, via the Reaction Control System, to the proper attitude for thrusting with the Service Module engines. Since thrusting will only last a few seconds, no thrust vector control is programmed by the computer.

DEBOOST TO LUNAR ORBIT

Deboost to lunar orbit is accomplished in a manner similar to mid-course velocity corrections. Since deboost to lunar orbit will last a few minutes, the computer, through the digital-to-analog portion of the optics ECU's, will control the gimbals of the Service Module thrust engine to provide steering as well as thrusting control.

LUNAR ORBIT CORRECTIONS

Verification of the lunar orbit will be performed by ground tracking from the earth and/or by onboard star occultation measurements. Orbital corrections will be performed in the same manner as mid-course velocity corrections.

LUNAR MAPPING

The inertial reference will be accurately aligned to inertial space, and local vertical will be determined and maintained by the G & N System with the aid of the Reaction Control System. The computer will: (1) calculate the desired attitude of the spacecraft with respect to inertial space as a function of orbital position, (2) compare the desired attitude with the attitude defined by the IMU inertial reference, and (3) correct the spacecraft attitude to the desired attitude.

TRANSEARTH INJECTION

Inertial guidance of the spacecraft is performed by the G & N System in the same manner as deboost to lunar orbit except that 1 hour after injection, a trajectory plane change is performed.

TRANSEARTH

The transearth phase is conducted in the same manner as the translunar phase.

ENTRY

Shortly before entry, the stable platform is aligned using optical sightings. During entry, the G & N System, by means of inertial guidance, controls the direction of the lift and drag forces by rolling the spacecraft. The G & N System is capable of landing the spacecraft within a predetermined area on the earth to facilitate recovery of the crew and Command Module.

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III. TECHNICAL ANALYSES

BLOCK II PERFORMANCE ANALYSIS

Performance of the Block II G & N System was investigated in three main areas: the first was a consideration of the navigational accuracy; the second, the ability of the system to maintain local vertical; and finally, system reliability degradation with time.

The navigational and local vertical stabilization requirements are considered only during the lunar polar orbit phase. The remaining mission phases are considered to be sufficiently similar to those existing in the lunar orbital rendezvous mission, and therefore the G & N System guidance monitor, control, and autonomous navigation capabilities are within the performance requirements specified.* The capability of the G & N guidance and monitor functions necessary for injection into 200 nm circular earth orbits from initial parking orbits (Profiles 2 and 3) has been previously evaluated in the XMAS and MODAP studies.** For the loose tolerance required on orbit circularization, the G & N is considered more than adequate.

The G & N System ability to provide local vertical stabilization signals for the specified periods during lunar mapping operations involves the establishment of an inertial reference with a known angular relationship to the initial mapping point local vertical. The computer (with minor program revisions) can then compute the required vehicle attitude with respect to the inertial reference as a function of time (based on a knowledge of orbital position and velocity), and can issue the appropriate attitude commands to maintain the vehicle within the desired attitude deadband and rate tolerances.

The basic Apollo Block II G & N System reliability analysis considers the application of the equipment as used in all the mission phases of the lunar polar orbit mission profiles. The applicable Apollo Block II G & N System error sources are listed in Table 4.

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* MIT/IL Performance Specifications, Guidance and Navigation System — Block II, 10 June 1964 (Confidential).

**AC Spark Plug: Modified Apollo Logistics Spacecraft (MODAP) and Extended Mission Apollo Space Station (XMAS) Missions Final Study Report, 15 December 1963.

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Table 4. Apollo Block II G & N System Error Sources

Sensor Error Source	1σ Value
Gyro	
Bias Drift	1 meru
SA Unbalance	10 meru/g
IA Unbalance	10 meru/g
Compliance	1 meru/g^2
Accelerometer	
Bias	1.5×10^{-4} g
Scale Factor	6×10^{-5} g/g
Nonlinearity	2×10^{-5} g/g^2
Sensitive Axis Misalignment	20 arc-seconds
Initial Alignment or Realignment	
Prelaunch	40 arc-seconds level 6 arc-minutes azimuth
In Space	60 arc-seconds all axes
Optics	
Star-to-Horizon "blue" line (Sextant Tracker-Horizon Photometer combination for orbit navigation)	6 arc-minutes
Star-Landmark (Sextant — for translunar navigation)	10 arc-seconds
Landmark Track	6 arc-minutes
Star Occultation (An error source not related to the system)*	2 to 6 arc-minutes

*Baker, D. S., et al: Lunar Orbit Determination by Star Occultations and MSFN Tracking, MIT Report E-1429, September 1963.

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LUNAR ORBIT NAVIGATION

For lunar orbit navigation, it is required that altitude information be obtained with a standard deviation of less than the orbit tolerance of ±1.1 nm. Ground tracking or onboard navigation (star occultation or landmark tracking) are considered as possible navigation modes. Parameters considered are measurement uncertainty, sampling time, and initial orbit injection errors.

Using the scanning telescope for star occultation or landmark tracking, measurement uncertainty standard deviations of 2 to 6 arc-minutes were assumed. For ground tracking, range accuracy standard deviations of 50, 500, and 5,000 feet were considered. Reports indicate that range accuracies of 50 feet are attainable.* It is assumed that no correlation exists between the errors in consecutive measurements.

The time between consecutive measurements strongly influences the time necessary to obtain the necessary accuracy. Conservative estimates concerning the sampling rate of the ground tracking stations were made. This was further enhanced by the assumption of a worst case as far as visibility of the space ship from ground is concerned, for example, it was assumed that the space ship is only visible for a half revolution. An analog philosophy was applied to onboard navigation.

Position and velocity errors at the time of injection into lunar orbit were chosen according to the performance specifications. However, to obtain a more conservative estimate of the performance, errors twice as large were also considered. In addition, it was assumed that the data is processed in an optimum linear filter.**

The results, as shown in a series of plots (Figures 14 through 22) of position and altitude accuracy versus the number of orbital revolutions, indicate that the required altitude accuracy of 1.1 nm can be obtained after one revolution. From these curves, the following additional conclusions were drawn:

- Ground tracking is superior to onboard navigation,
- Changes in the injection errors have minimal effect on the measurement uncertainty after one revolution,
- If the onboard instrument accuracy for star occultations is sufficiently good, the interval between observations can be increased after one revolution,
- The main position error orientation is more tangential than normal with respect to the orbit.

*Baker, D. S. (op. cit.)
Smith, L G. and E. V. Harper: Midcourse Guidance Using Radar Tracking and On-Board Observation Data, NASA TN D-2238.

**Sorenson, H. W., F. Salomon, T. Kido: Application of Wiener-Kalman Filter Theory to Orbit Determination, AC Spark Plug ACRD Report, August 1963.

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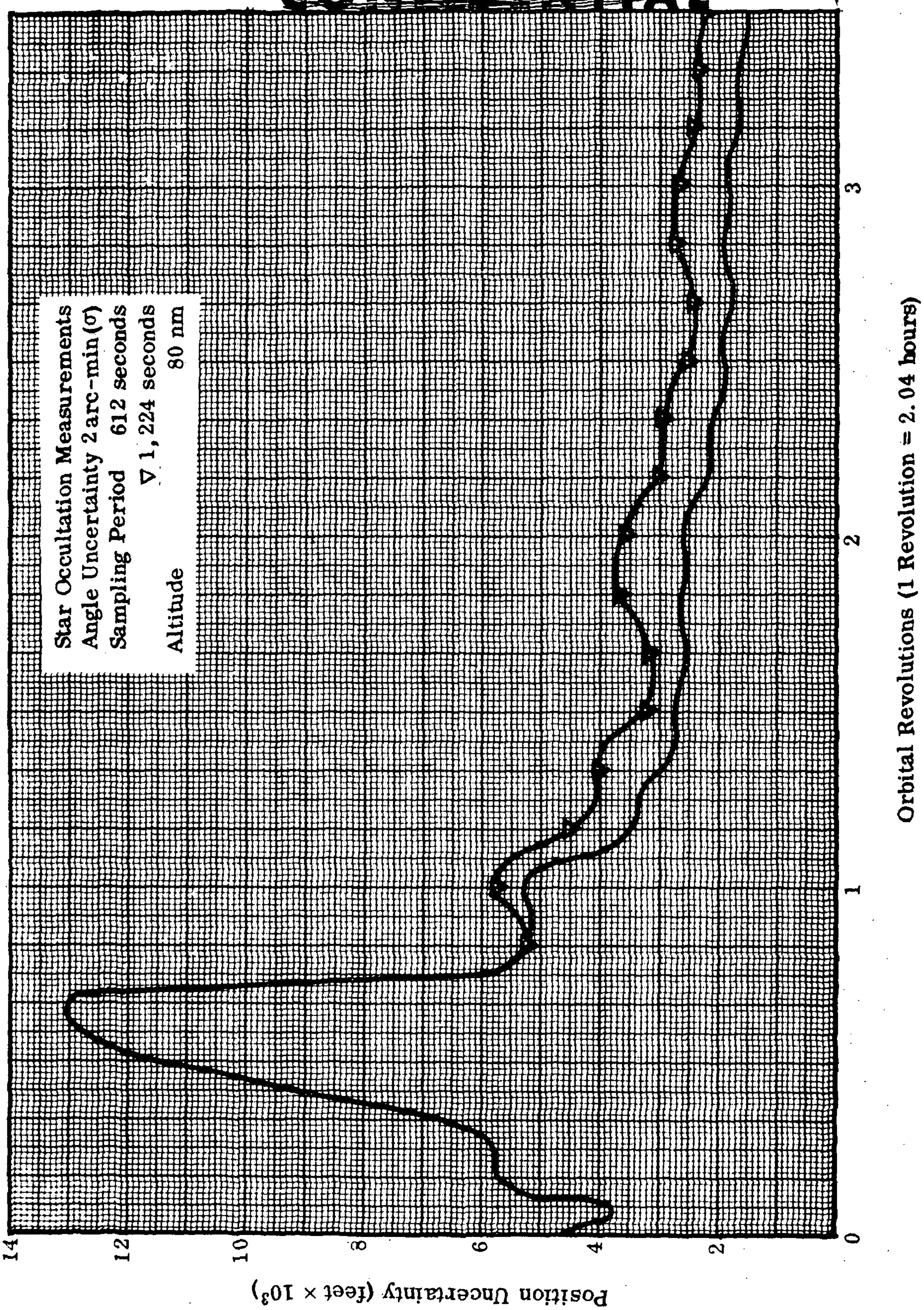

Figure 14. Maximum Position Uncertainty (1 σ)

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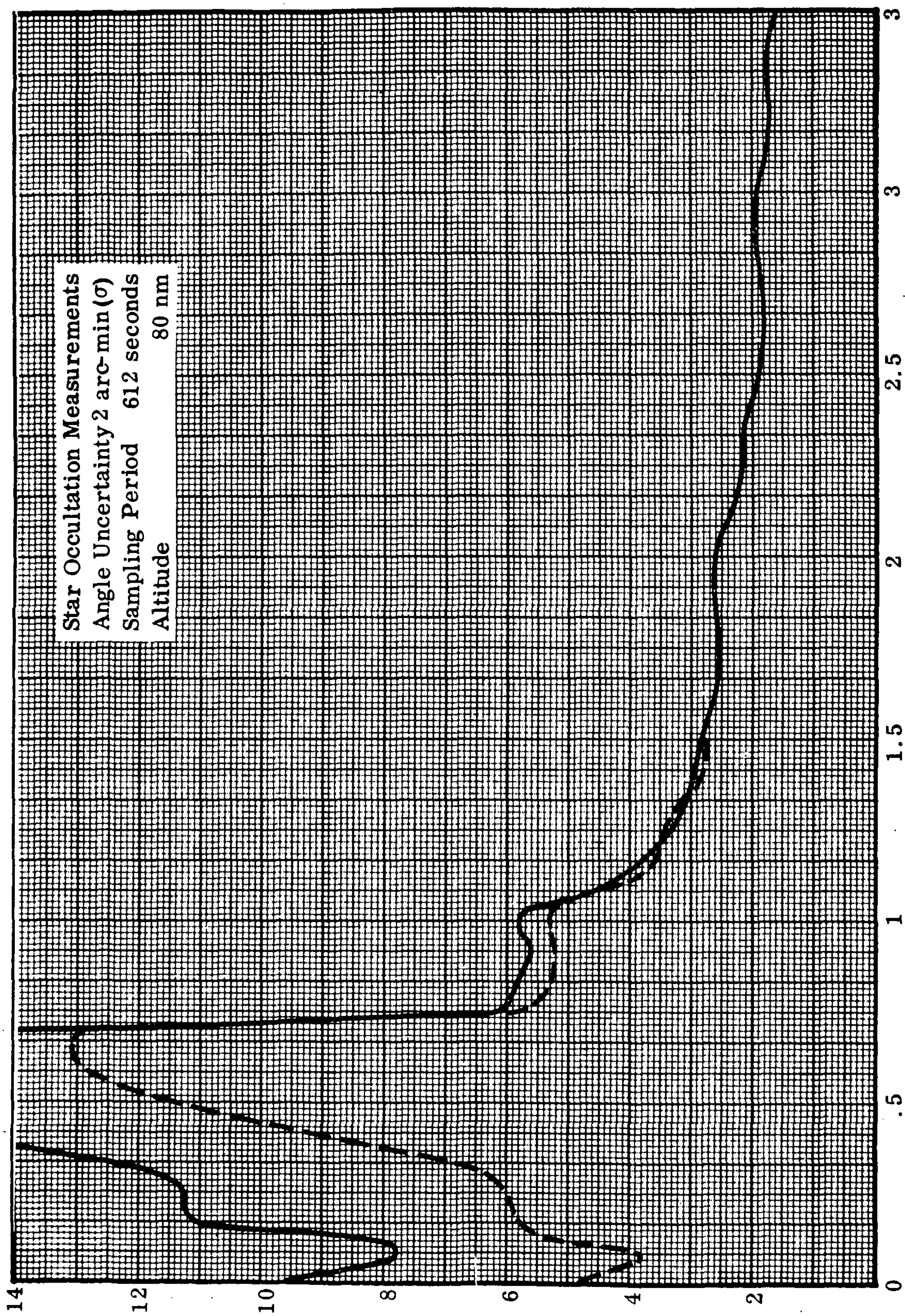

Figure 15. Maximum Position Uncertainty (1 σ) Versus Initial Accuracies

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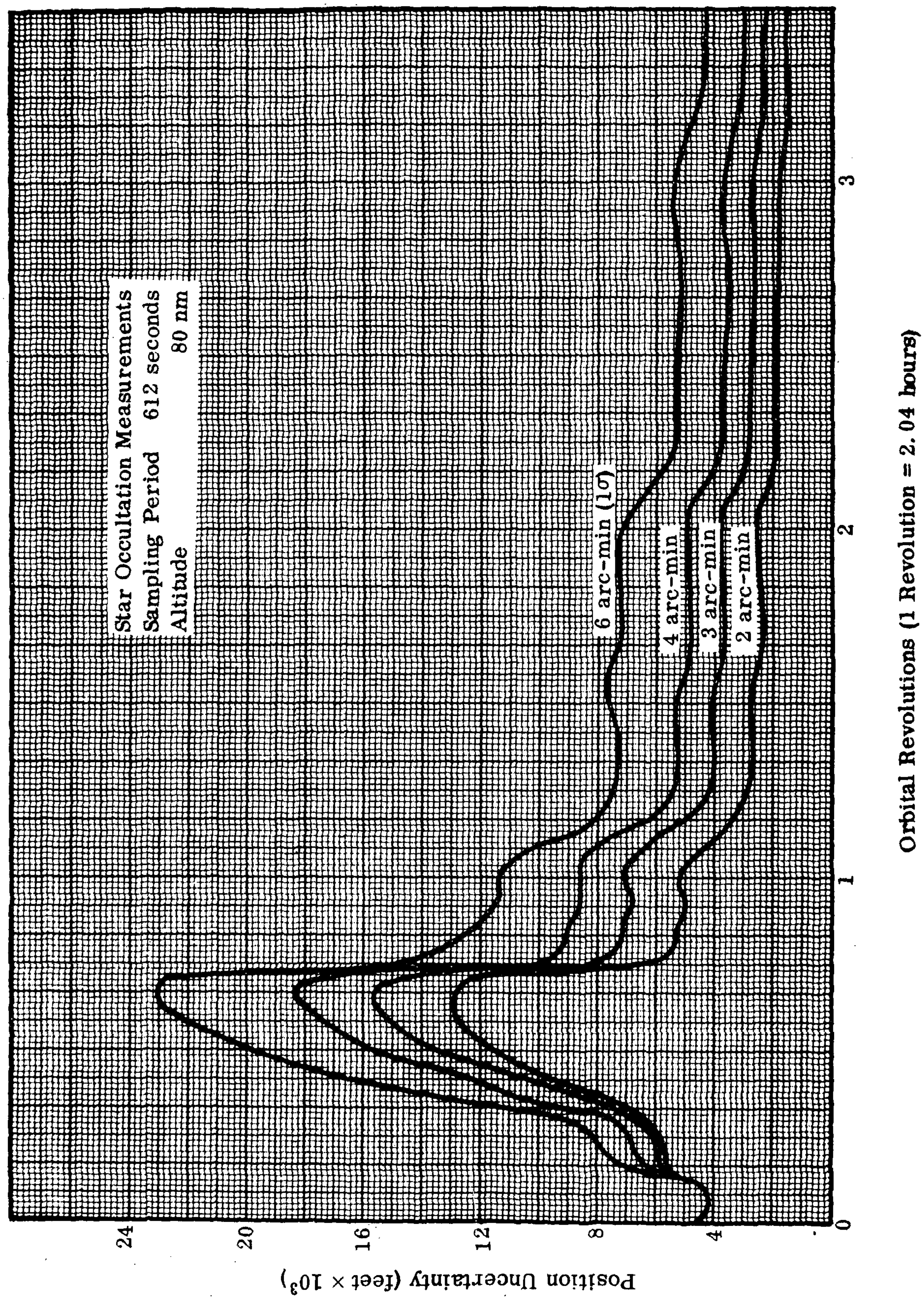

Figure 16. Maximum Position Uncertainty (1 σ) Versus Angular Accuracies

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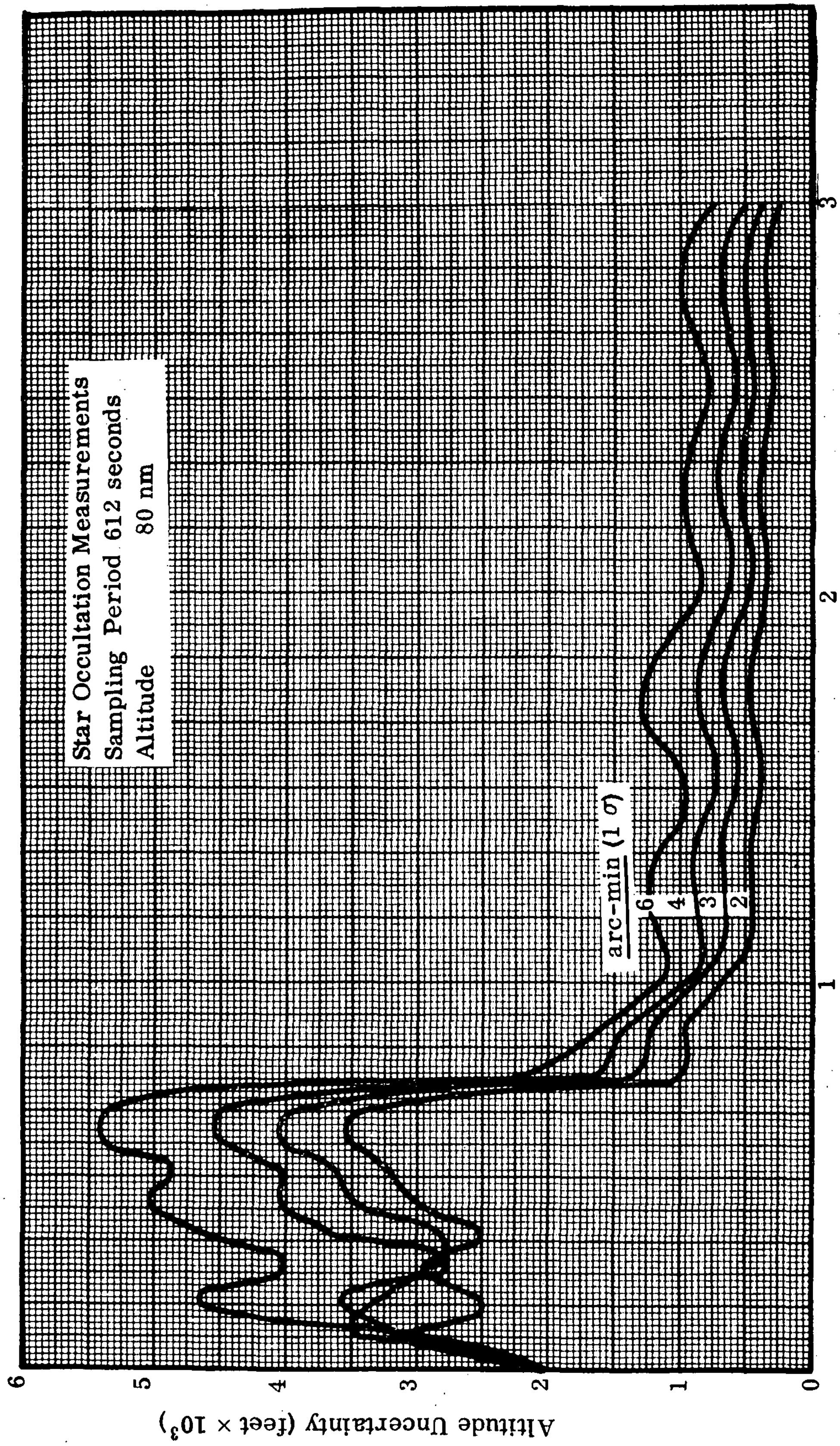

Figure 17. Altitude Uncertainty (1 σ) Versus Angular Accuracies

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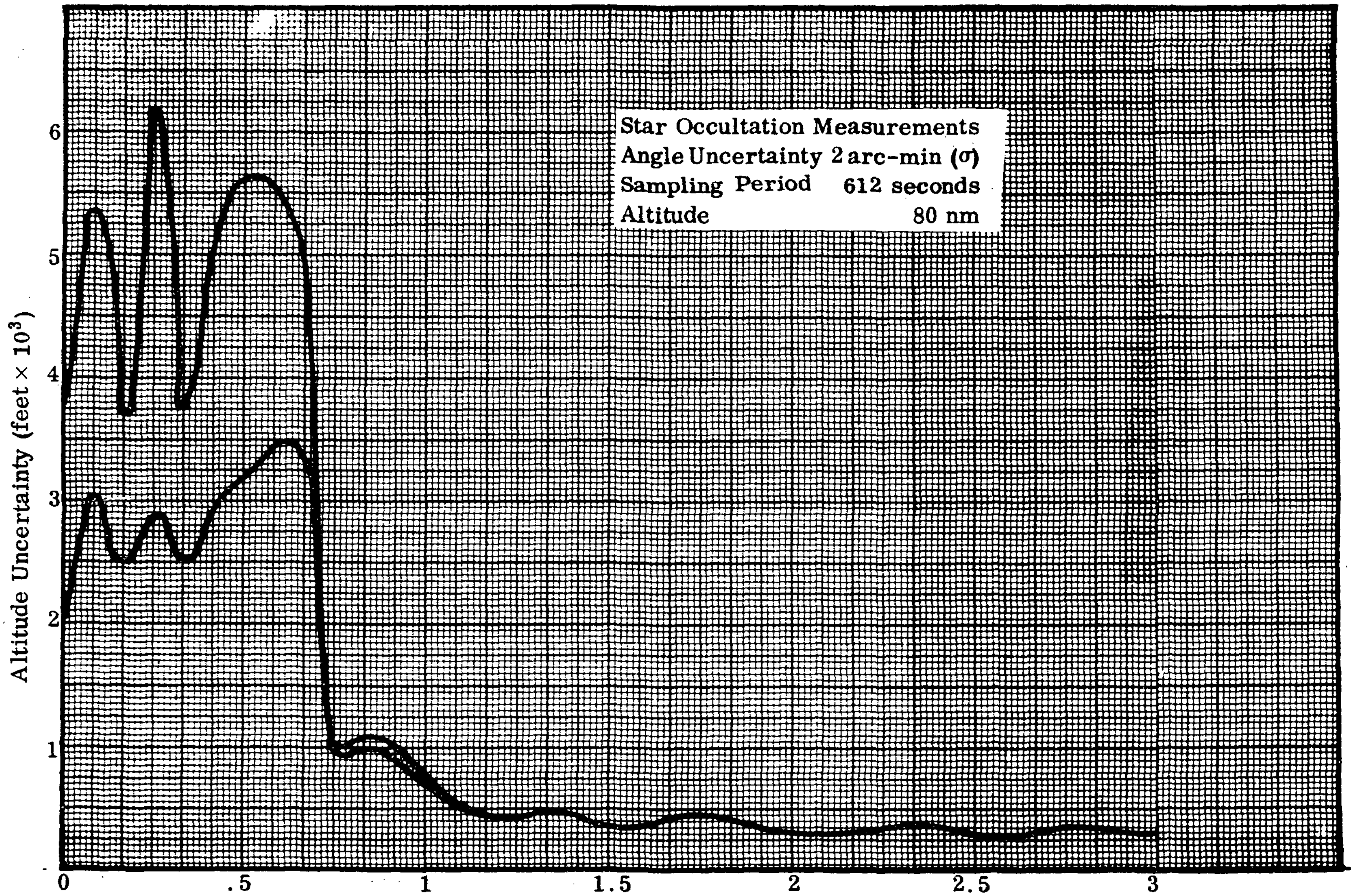

Figure 18. Altitude Uncertainty (1σ) Versus Initial Accuracies

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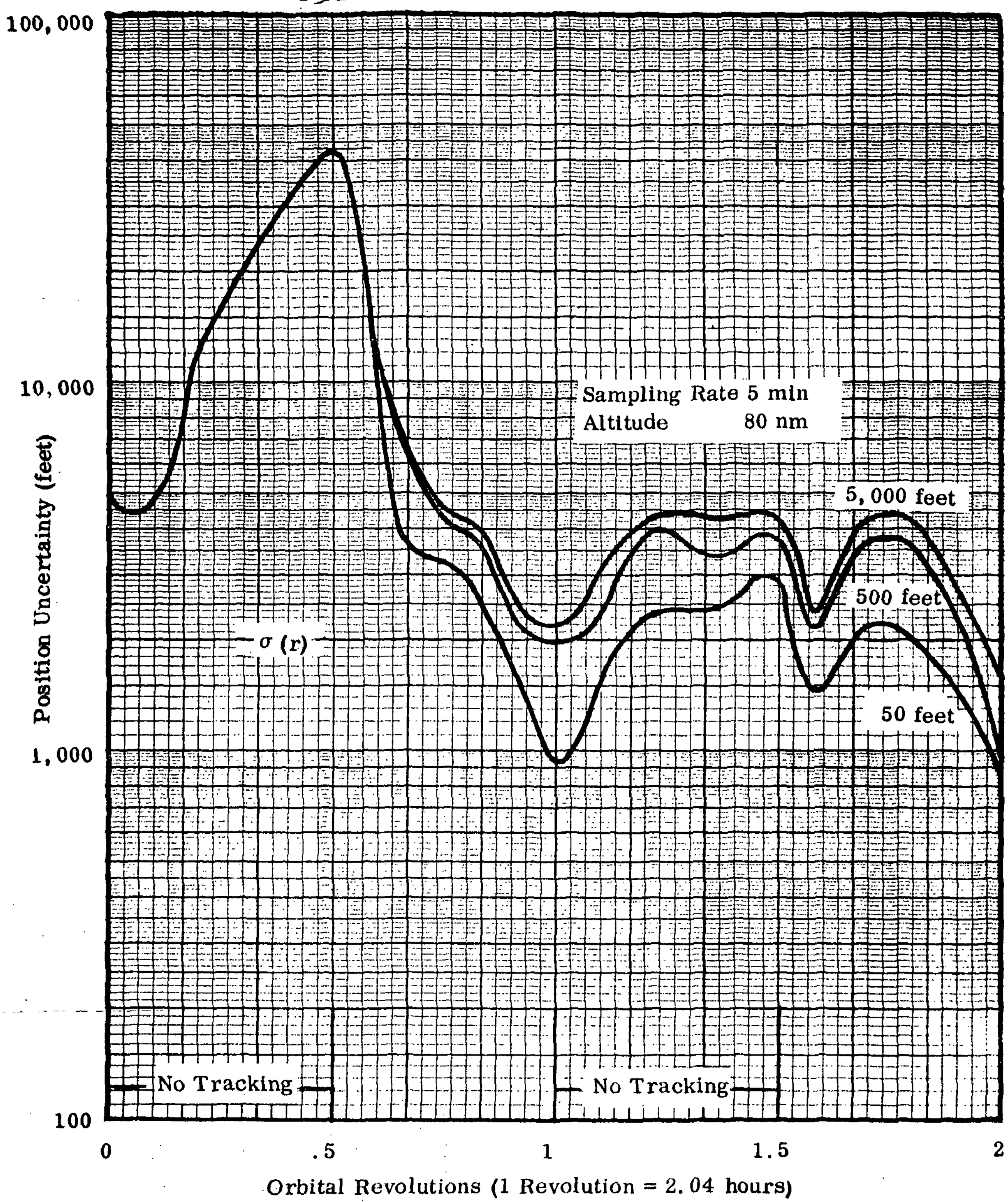

Figure 19. Maximum Position Uncertainty (1σ) Versus Ground Tracking Accuracies

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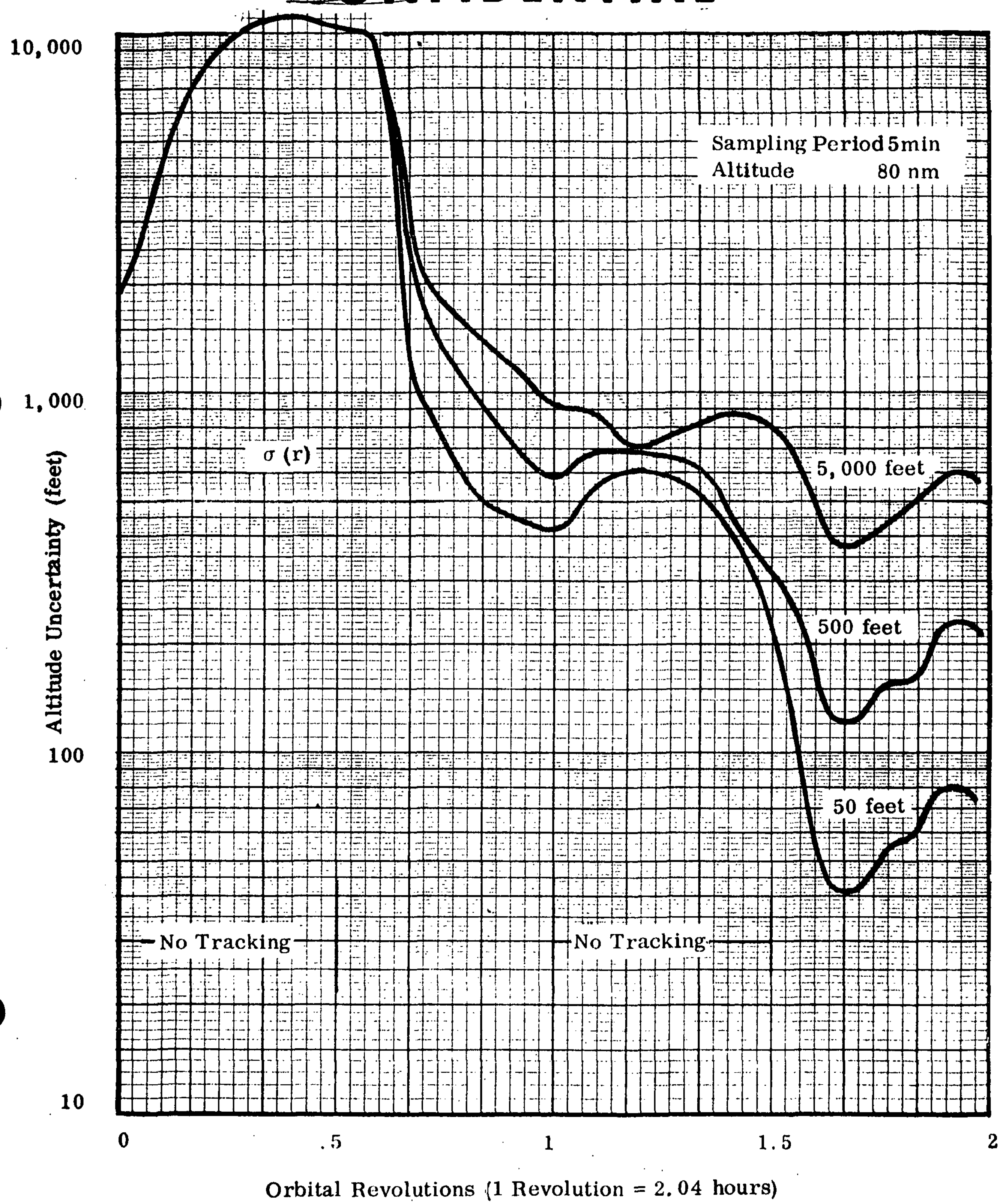

Figure 20. Altitude Uncertainty (1 σ) Versus Ground Tracking Accuracies

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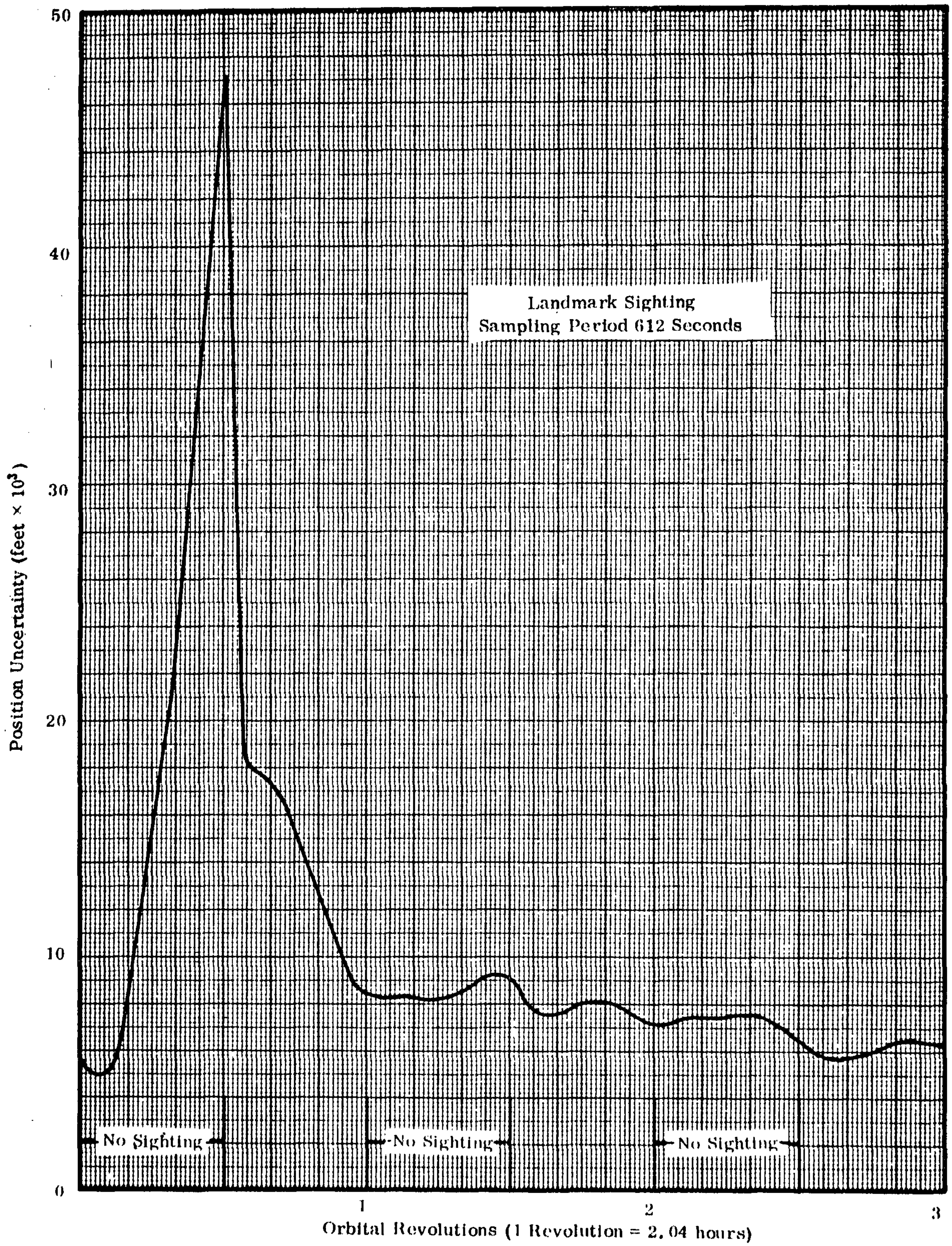

Figure 21. Maximum Position Uncertainty (1 σ)

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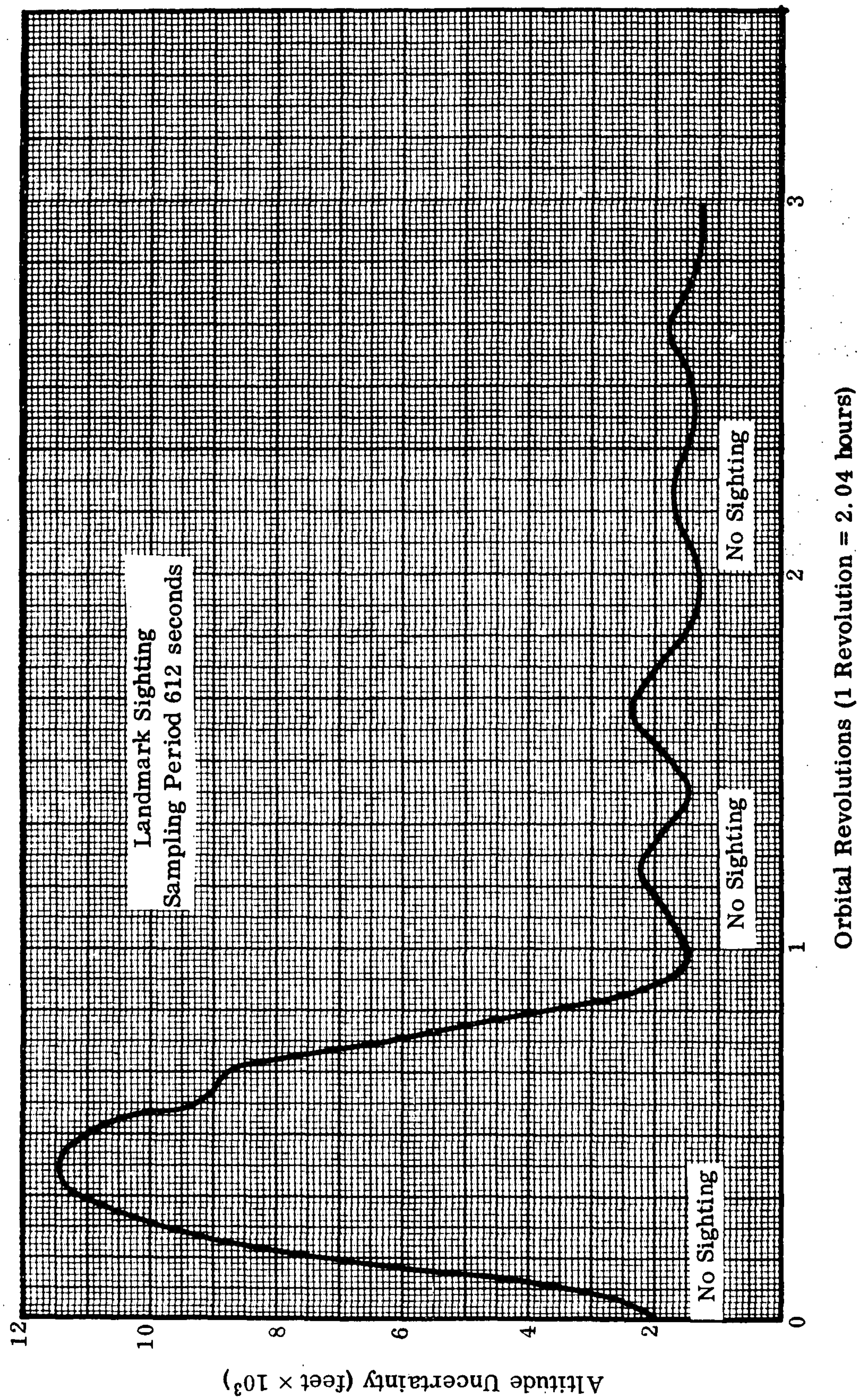

Figure 22. Altitude Uncertainty (1σ)

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LOCAL VERTICAL STABILIZATION

The local vertical attitude deadband requirement for the lunar mapping mission is given as ±450 arc-seconds about all axes. Contributing factors to the error in local vertical attitude include the initial space alignment of the inertial platform, the gyro bias drifts, the spacecraft positional uncertainty in orbit, and errors in the vehicle attitude control system. The alignment uncertainty between the platform axes and the star lines-of-sight has a standard deviation of 60 arc-seconds about any axis. The gyro bias drift has a standard deviation of 1 meru (0.9 arc-sec/min). The spacecraft positional uncertainty at the start of mapping is assumed to have a standard deviation of 0.5 nm. This positional uncertainty yields an angular error of 110 arc-seconds in an 80 nm lunar orbit. Vehicle control system errors are not considered here.

Since the above error sources were independent and random in nature, they were combined by the root-sum-square technique to determine an overall standard deviation. See Table 5. Tabulating these error sources versus time, it is seen that the attitude deadband requirements are met with a 0.5 nm position uncertainty. The additional vehicle control system uncertainty allowable lies in the range of 50 to 60 arc-seconds.

Table 5. Local Vertical Stabilization Error (1σ)

Time (min)	Alignment Uncertainty (arc-sec)	Gyro Bias Drift (arc-sec)	S/C Position Uncertainty (arc-sec)	Local Vertical Uncertainty (arc-sec)
0	60	0	110	125
15	60	13.5	110	126
30	60	27.0	110	128
45	60	40.5	110	132
60	60	54.0	110	136.5
75	60	67.5	110	142

The longest scheduled mapping interval is 1 hour; however, the time considered has been extended beyond this to allow for alignment and memory time before the start of mapping.

In all of the above, it is assumed that a computer program will be written that will enable the computer to command desired changes in attitude based upon the spacecraft position in orbit, knowledge of the orbit parameters, and time.

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The attitude rate also has a requirement imposed upon it during mapping. It must be kept within ±0.02 deg/sec of zero on two spacecraft axes, and of 3 deg/min on the third axis. The ECU's supply attitude information to the computer in the form of pulses with a scale factor of 39.55 arc-sec/pulse. The computer can detect the spacing between pulses to better than 10 ms. A rate of 0.02 deg/sec corresponds to 1.82 pulses/sec, or a spacing of 0.55 second between consecutive pulses. Therefore, the computer can detect the maximum desired rate about zero on two axes to within 2 percent. The maximum allowable rate of 4.2 deg/min about the third axis produces a spacing of 0.156 seconds between consecutive pulses, and therefore the computer can detect the maximum desired rate about 3 deg/min to within 6.5 percent. It is assumed that the computer can be programmed to issue correction commands when the desired rate inputs are exceeded.

BLOCK II RELIABILITY VERSUS TIME

The Apollo Block II G & N System (with no modifications, spares, or redundancy), when applied to the lunar polar orbit mapping mission, will yield an overall success probability through entry of .91112. This number was arrived at by combining module reliabilities into major subsystem reliabilities, and combining these, in turn, to yield the overall G & N System reliability. These reliability values are listed below, and the system reliability versus mission time is shown in Figure 23.

Subassembly	Probability of Success
IMU	.96962
PSA	.97648
ECU	.96387
Optics	.99886
D & C	.99955
Total G & N	.91112

Module excitation times were determined from the times assigned to complete the various functions in the time line. "K" factors, as defined by NAA, have been applied to the equipment operating times to account for more severe environments encountered during the thrusting phases.

Module failure rates, where applicable, were based upon the Block I module stressed failure rates defined in MIT Report R-429, "Reliability and Quality Assurance Progress Report," dated December 1963. Some module failure rates have been updated for Block II, where equipment design has solidified. In both of these areas, the stressed

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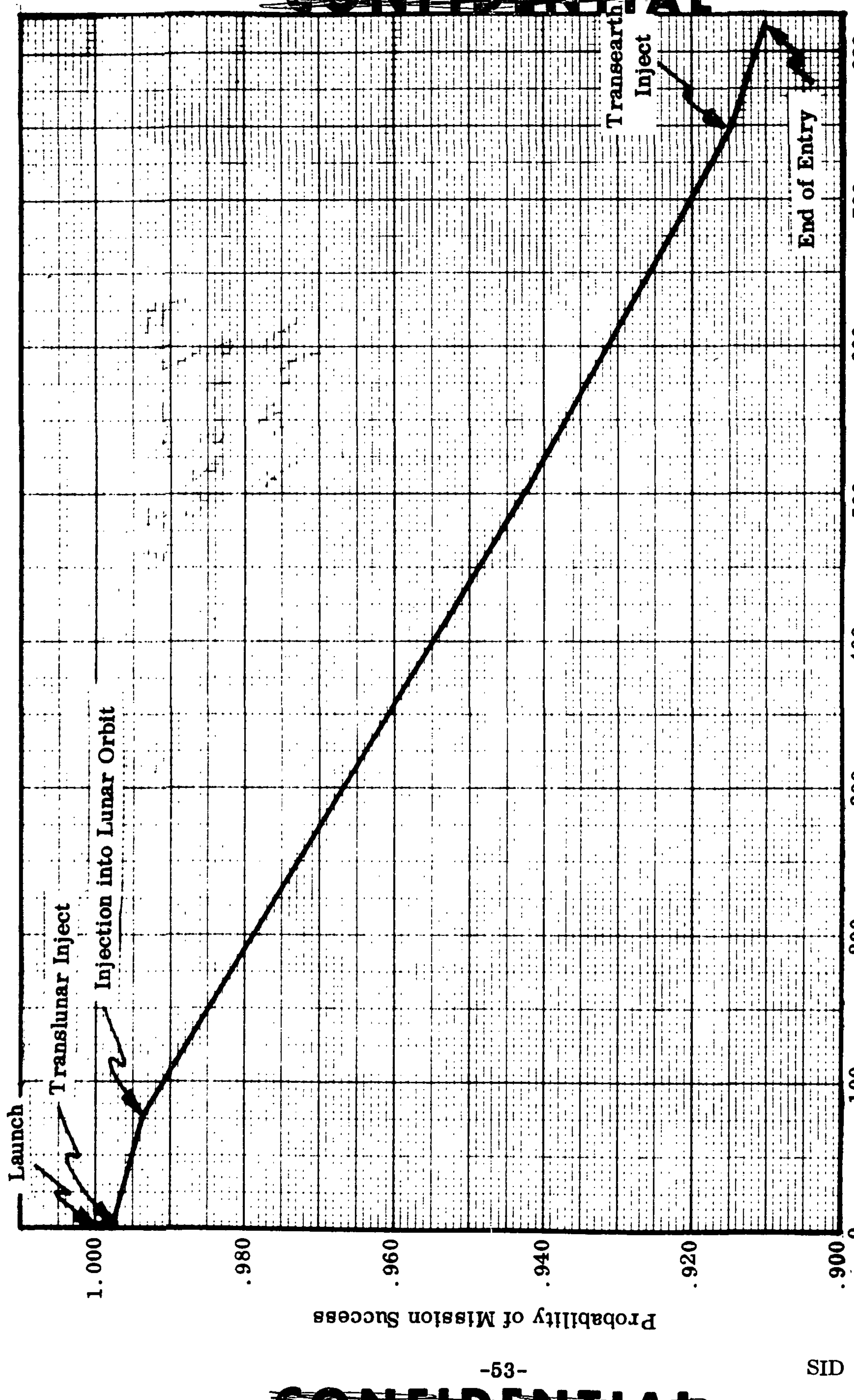

Figure 23. Reliability Versus Time per Profile 1 Apollo X Mission (Block II G & N System Without Modification or Spares)

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failure rates were developed from the basic part failure rates as modified by a stress factor based on the temperature environment and the ratio of operating to rated power or voltage. In the case of the electronic coupling units, only a parts count and an average failure rate have been used without any attention being paid to circuit applications since these units are still in the design development stage. The failure rates used for all parts and modules are found in the Appendix.

Employing these usage times and failure rates, module reliabilities were calculated by assuming an exponential distribution of times to failure. Certain equipment reliabilities were omitted from the calculations because they are not used in the defined mission, or they are not considered essential and the mission can be adequately performed without them.* Therefore, their failure could not cause a basic G & N System failure. All other module reliabilities were combined as though all modules were connected serially, and that any module failure would produce a G & N failure in the mission. A failure effects analysis would also eliminate certain equipment from the reliability calculations; however, this was not done because it was found that the results of a "quick look" failure effects analysis are useless.

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*The following equipment was deemed nonessential:

IMU Temperature Alarm Module,
Optics Two-Speed Switch,
Tracker X Channel,
Tracker Y Channel
Modulator and Loop Compensation
Star Presence
Cosecant Amplifier
Photometer Electronics (PSA)
G & N Panel Brightness Control
Optics Head Tracker Equipment
Optics Head Photometer Equipment
Radar Mode Module
Sextant Trunnion 2x Resolver
Sextant Shaft Computing Resolver
Sextant Shaft 1/2x Resolver
Trunnion Angle Counter
Shaft Angle Counter
Map and Data Viewer

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PROBLEM AREAS

PERFORMANCE

There are no performance problem areas. The G & N System is expected to adequately meet all the performance requirements of lunar orbit navigation and local vertical stabilization.

RELIABILITY

As shown in the analysis, the reliability apportioned to the G & N System in the lunar polar orbit mission (.9975) is not met. Also, it is shown that the inertial measurement unit (IMU), the power and servo assembly (PSA), and the electronic coupling units (ECU) are each below the required overall reliability at the end of the mission, while the sextant and scanning telescope head assembly reliability is marginally above the overall G & N requirement.

Each of these subassemblies contains certain low-reliability or life-limited pieces of equipment to which their overall low reliabilities can be attributed. The troublesome elements are:

- Inertial Measurement Unit

  Slip rings
  Bearings
  Resolvers
  Torque motors
  PIP suspensions
  IRIG suspensions
  IRIG and preamplifier assemblies
  PIP
  PIPA preamplifiers

- Power and Servo Assembly

  Gimbal servo amplifiers
  Fail indicator module
  3200 cps, 1 percent amplifier
  800 cps, 5 percent amplifier
  3200 cps AAC filter
  800 cps AAC filter
  800 cps, 1 percent amplifier
  -28 vdc power supply
  PIPA calibration module
  Pulse torque power supply
  Binary current switch
  DC differential amplifier and precision voltage reference

In the ECU, all modules have too low a reliability factor except possibly the 4-volt power supply and the clock and mode logic modules, which are marginal. The optics subsystem failure rate is due mainly to bearings, resolvers, and motor-tachometers.

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From the above, it is seen that the types of modules that require reliability improvement are electronic modules and life-limited mechanical or electromechanical modules.

The designs on most Apollo Block II electronic modules have not been optimized as yet. This is especially true on the ECU's. Effort has just begun to determine stress factors, based on circuit applications, to update the module failure rates and to determine where the design may be changed to improve performance and reliability. Preliminary indications show that the ECU reliability is better than the figures obtained on a parts count and average failure rate basis. These investigations will be continuing for some time, and therefore it cannot be predicted exactly which modules can be improved and by how much.

The problem areas in the mechanical and electromechanical life-limited equipment all seem to be due to slip rings, bearings, and brushes. Slip rings tend to become noisy due to sliding contact wear. This wear is a function of the materials used, the quality of fabrication, temperature, and lubrication. Also, to a lesser extent, contamination, either external or caused by chemical action of the parts themselves, can cause degradation of the slip rings.

Brushes appear to be affected in general by the same causes. In addition, brush wear is approximately proportional to shaft speed, and therefore particular applications become important.

On the other hand, bearing failures are due mainly to lubrication problems because of size, temperature, and humidity restrictions. They are also affected by fabrication quality and application.

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LIFE EXTENSION ANALYSIS

Several approaches have been investigated to bring the G & N System reliability up to the required level. System reliability can be increased by improvement of equipment, modification of equipment usage, and the addition of spare and/or redundant equipment. It should not be overlooked, however, that once the complete Apollo Block II System has been defined, a detailed failure effects analysis can be performed and all modules can be assigned stressed failure rates based upon their component applications. This will certainly improve the accuracy of the reliability predicted for the basic Apollo Block II G & N System.

To complete this study, life extension will be discussed on the basis of the present reliability predictions.

AREAS OF IMPROVEMENT

Modules

Since the design of much of the Block II electronic equipment has not been completed or optimized, it is not possible to define those areas where modules have been improved. Similarly, performance of Block II mechanical and electromechanical modules is still being defined, so here also it is not possible to say which modules can be improved and to what degree. Life tests under various environments are continuously being conducted on all Block II equipment in order to determine life limiting causes and methods of improvement.

Alternate G & N Approaches

To reduce certain module usage times, and thereby improve their predicted reliabilities, power for inertial component suspension and for the PIPA loops can be switched off until just prior to their use. Previously, suspension power was applied throughout the entire mission and the PIPA loops would operate whenever the ISS was operating.

Using the mechanization shown in Figure 24 to switch on the suspension power 15 minutes before ISS turn-on, the usage time is cut to one-fourth of the original on the following modules.

3200 cps AAC filter
3200 cps, 1 percent amplifier
Ducosyn transformer
PIP suspension
IRIG suspension

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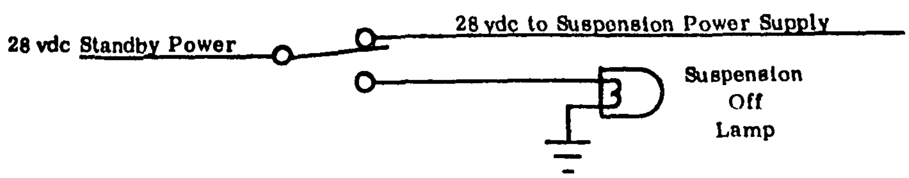

Figure 24. Mechanization to Allow Suspension Turn-Off

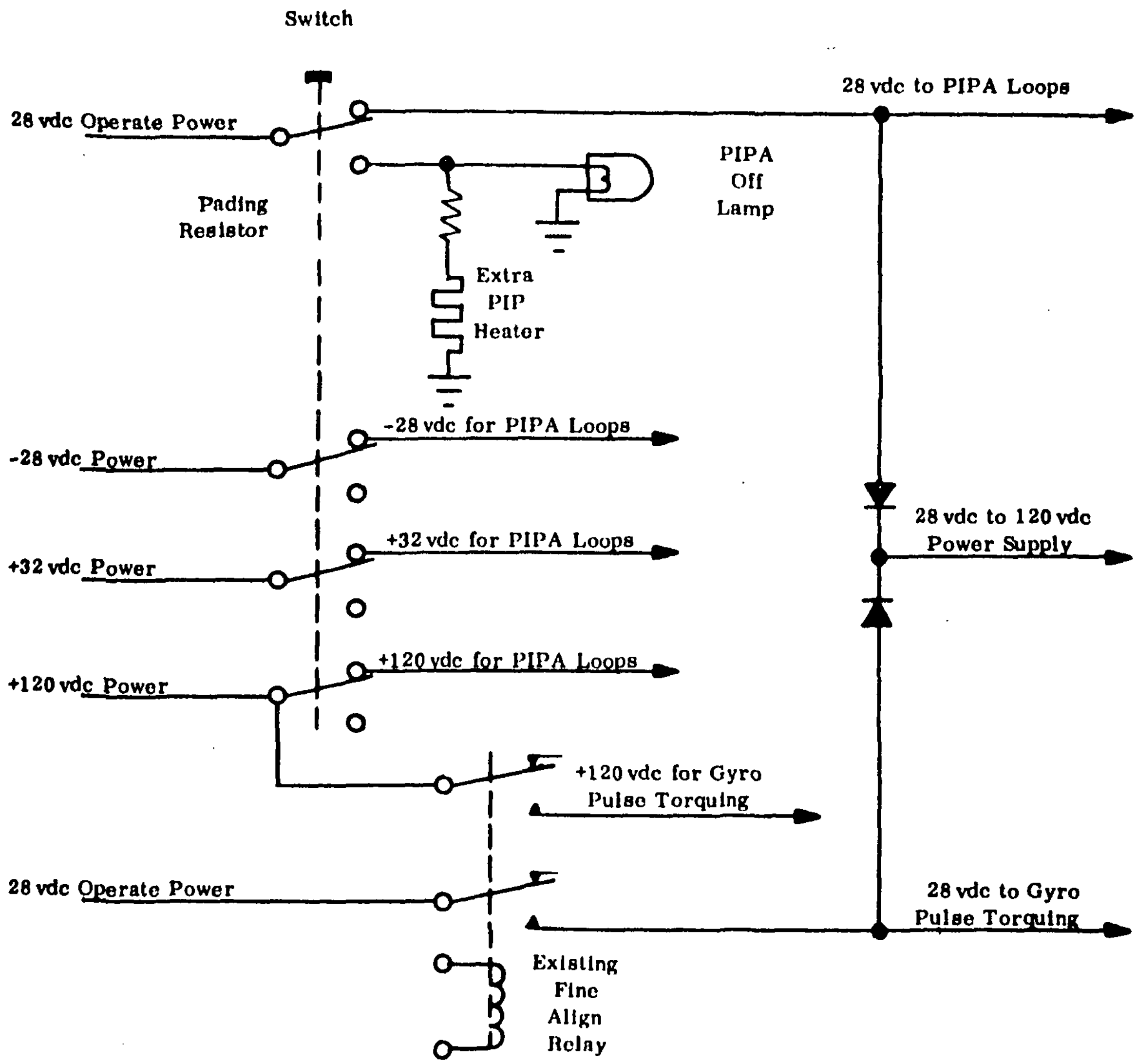

Figure 25. Mechanization to Allow PIPA Loop Turn-Off

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Using the mechanization shown in Figure 25 to switch on the PIPA loop excitation 5 minutes before thrusting, the usage time on the following modules can be radically reduced.

PIPA calibration
AC differential amplifier and interrogator
Binary current switch
DC differential amplifier and precision voltage reference
PIP
PIP preamplifier
Pulse torquing power supply

With the above modifications, the G & N probability of success in the lunar mapping mission is .936. The new subassembly reliability breakdown is as follows.

Subassembly	Probability of Success
IMU	.982
PSA	.9905
ECU	.96387
Optics	.99886
D & C	.99955
Total G & N	.936

At present, the reliability effects due to switching power on and off are not known. For the above modifications, PIPA loop switching has been reduced while that for the suspension was increased.

There are a limited number of built-in alternates for performing the required G & N functions as shown in the Appendix. In many instances, the same modules are used, but in alternate modes. A detailed failure effects analysis would indicate whether an alternate approach is available after a module failure has occurred and what the resultant improvement in reliability would be.

SPARES/REDUNDANCY

Based upon G & N System reliability after modifications, the use of spares and/or redundancy was investigated to increase the overall success probability to the required value of .9975 for Profile 1 (lunar polar orbit). The effects of sparing or using redundancy were investigated at the module, major assembly, and system levels. Based

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upon the reliability gained, the additional weight penalty, the malfunction isolation capability, packaging, the effects on G & N accuracy, the feasibility of sparing under space conditions, and time periods when sparing is not possible, a suggested combination of modules, assemblies, spares, and redundancy was developed. These are as follows:

- Redundant heater power slip rings in the IMU,
- One spare IMU and spare inertial component calibration modules in the PSA (44.6 pounds),
- Five redundant ECU's, one redundant 4-volt power supply and clock and mode modules (23 pounds),
- One spare D and C panel (13 pounds),
- One spare each of the following PSA modules (10.4 pounds)

Gimbal servo and align amplifier
Fail indicator and warning
3200 cps AAC filter
3200 cps, 1 percent amplifier
-28 vdc power supply
Sextant motor drive amplifier
Scanning telescope motor drive amplifier
800 cps optics compensation
Servo integrate relay
Anti-creep circuit
Scanning telescope trunnion transformer and relay
Tracker mode relay
Scanning telescope mode relay
G & N subsystem filter
800 cps AAC filter
800 cps, 1 percent amplifier
800 cps, 5 percent amplifier

It is seen that 91 pounds of additional equipment, consisting of 68 pounds of spares and 23 pounds of redundancy, are needed. With these, the G & N probability of success in the lunar polar orbit mapping mission was calculated to be .997512.

Subassembly	Probability of Success
IMU	.999445
PSA	.9996
ECU	.999622
Optics	.99886
D & C	.99983
Total G & N	.997512

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In all the above calculations, it was assumed that the optics head could not be spared or made redundant since this would constitute a major G & N or spacecraft redesign. With this constraint, the weight and volume penalties become critical if more of the other equipment is added to increase the overall G & N success probability.

Some form of redundancy was required in the case of ECU's to improve their overall reliability during critical periods. Critical periods when sparing could not be performed are: entry, all thrusting periods, 1 hour prior to deboost to lunar orbit, and 1 hour prior to entry. It was considered that all other thrusting periods could be postponed if sparing was necessary. For the reliability calculations, each of the five ECU's and the 4-volt power supply and clock and mode logic modules were considered to be paralleled by identical units. An alternate approach would be to design redundant parts into the existing modules.

A possible implementation for paralleling entire coupling units is shown in Figure 26. The redundant ECU operates independently and in parallel with the primary unit, with the outputs controlled by switches. Normally, the outputs of the primary ECU would be used and those of the redundant ECU inhibited by the output switches. The switching function for the output switches is provided by the failure detector, which monitors six selected signals. It is conservatively estimated that 90 percent of all possible failures in the ECU would be detected by the monitoring of these six signals. If a failure should occur in the primary system, the signal outputs used by other subsystems would be switched from the primary ECU to the redundant ECU.

The six primary ECU signals selected for monitoring are: +14vdc, +4vdc, 800cps reference, digital-to-analog converter failure, fine error, and coarse error. The +14vdc would be monitored in the analog mode module and would serve as a test for the operation of the 14vdc supply, as well as the 28vdc source, which the 14v supply uses for primary power. The +4vdc supply would be monitored at the error angle counter module, and would serve as a check on digital logic power. Both the 14v and 4v signals could be monitored by a device that would give a failure indication (change of voltage level) if the signal was not within a specified tolerance band. The 800 cycle reference could also be monitored by a similar device after initial detection (rectification) and integration (filtering) to a dc level. The 800 cycle reference signal would be monitored in the digital-to-analog converter and coarse error loop. This would check the operation of circuits in both modules.

The digital-to-analog converter (DAC) failure signal could conceivably involve computer programming. The error counter of the ECU that controls the DAC accepts computer pulses as its driving function. A method of checking DAC operation would be to match the solution determined by the DAC to that of the computer, and give a failure indication should a mismatch occur.

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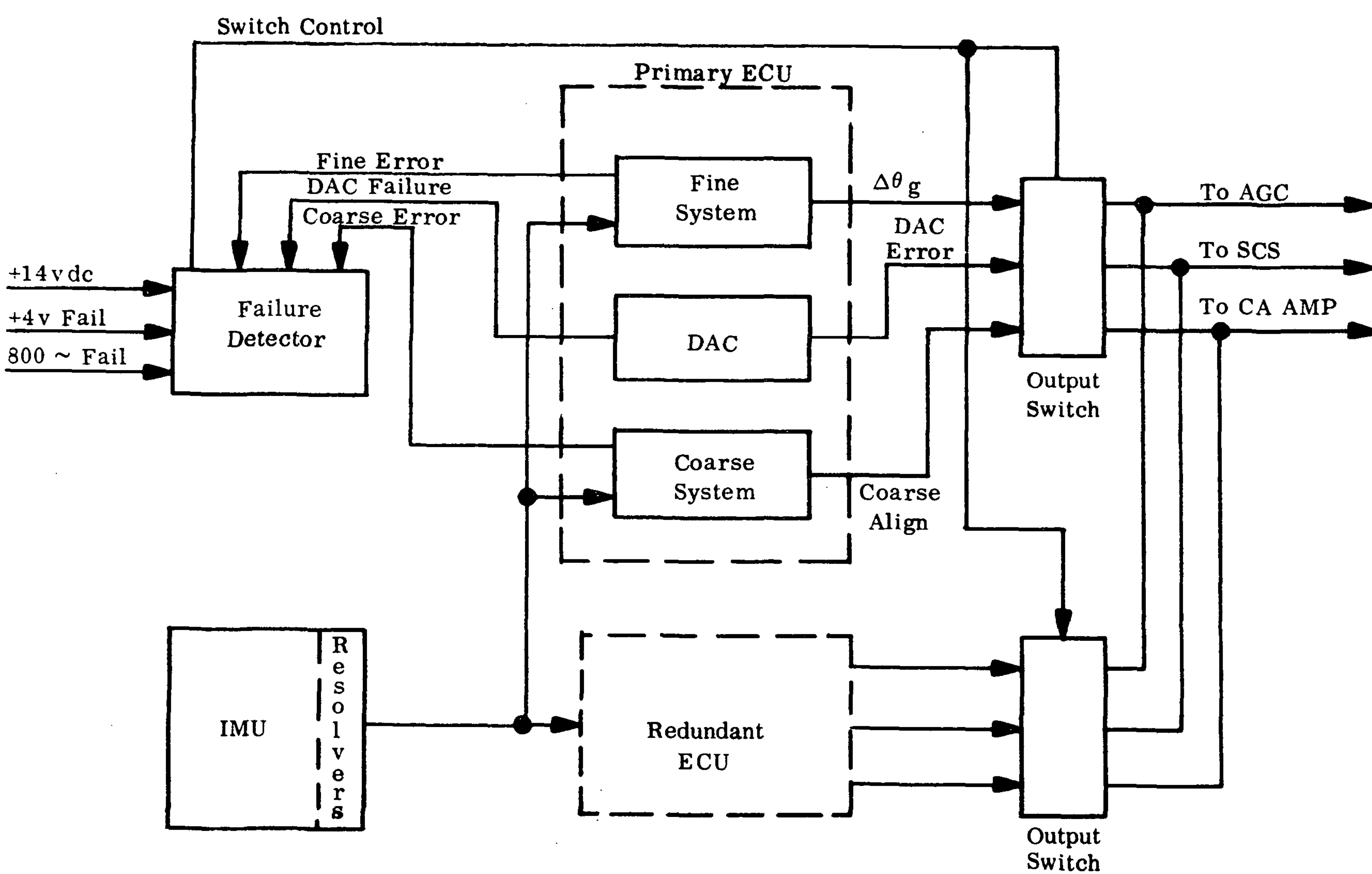

Figure 26. Proposed Redundant Electronic Coupling Unit (ECU) Diagram

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- Inertial Measurement Unit
  - Total weight is 42 pounds.
  - Four captive socket head screws will be used to secure the IMU.
  - Three PIPA calibration modules and one IRIG calibration module will be matched to the spare IMU, and installed with it.
  - The coolant connectors on the IMU will accommodate a connection and disconnection of the supply hoses in an atmosphere of 7 psia without loss of coolant.
  - Replacement of the IMU can be accomplished with the G & N Indicator Control Panel removed. In addition, the optical shroud may require removal. Some mechanical redesign of panels may be required to minimize G & N disassembly.
  - Appropriate grips or handles on the IMU for zero-g handling will be required.
- Power and Servo Assembly
  - Average weight of the modules is 0.68 pound.
  - All spared modules are accessable (as presently configured) with only the G & N Indicator Control Panel removed.
  - Moisture proofing is accomplished at the module level.
  - Captive screws are used to hold the modules to the header and coldplate.
  - All receptacles will have guide pins to aid in replacements.
  - One size socket head common to all modules.

The onboard storage of the listed spares is assumed to utilize space in the laboratory Module associated with mission Profile 1. The volume requirements are tabulated, and of special significance is the heat requirement of the IMU. The requirements are as follows.

Inertial Measurement Unit

The volume required to store the spare IMU is approximately 1,440 cubic inches (14-inch clearance sphere). The spared IMU, filled with coolant fluid, will require an expansion bellows plug to relieve high pressure differentials in the coolant lines caused by temperature extremes. The IMU will also require shock mounting while in storage. This can be accomplished by providing a shock-mounted plate fitted with mounting pads to which the IMU can be mounted in the storage area. Since the IMU will require heat during the storage period, a 28 vdc source (approximately 2 watts average) must be available in the storage area to operate the heaters built into the IMU.

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Power and Servo Assembly

The average volume of a PSA module is 11 cubic inches. Each module will be packaged separately in a moistureproof bag and will require an outer container for shock and vibration protection. It is anticipated that the average module, with packaging, will require approximately 15 cubic inches. The suggested spares list for Apollo X contains 21 PSA modules and will require approximately 315 cubic inches of storage space.

Display and Control Group

The D & C Group will be packaged in moistureproof bags, and will require an outer container for shock and vibration isolation. The D & C Group will require approximately 1,560 cubic inches of storage space.

Malfunction Isolation Considerations

The time required to isolate, remove, and replace a malfunctioned component is based on an average time of 2.5 hours. It can be seen that (based on the probability of failure for a G & N System with no spares and Malina's Tables of Poisson's Exponential Binomial Limits as applied to probability of failures) the probability of having more than two failures is insignificant. Therefore, the total maximum maintenance downtime for the total mission will be considered as twice the average downtime for one failure, or 5 hours.

The average downtime was arrived at by sampling probable failures and estimating individual times required to isolate the failure. The 2.5-hour average downtime consists of:

- Data accumulation — 1/4 hour
- Communications with earth — 1/4 hour
- Planning and interpretation of data — 1 hour
- Removal and replacement — 1/2 hour
- Verification of repair — 1/2 hour

The fault isolating procedure that will be used is intended to proceed through the following steps.

- Detect failure.
- Switch computers. (If failure clears, computer failure can be checked out with self-check program; if not, system diagnostics can be initiated.)

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- Switch ECU's.
- Localize failures to greatest practical extent via downlink. Supplement standard downlink with system diagnostic programs in the computer.
- If possible, find mode changes, backup methods, or other procedural means for bypassing the problem.
- Determine the failed component via telemetry data,and replace.
- Assist astronaut in manual checkout of the system until failed component is isolated.

These steps are intended to be run in the order listed. At any time, restoration of satisfactory operation will make further progress through the list unnecessary.

The above fault isolation procedure assumes the primary role for isolation of malfunctions on the downlink telemetry with onboard capability as a backup tool. Table 6 shows a list of downlink telemetry requirements (based on Block I planning), which appears to provide minimal needs for isolation to the recommended spares level. Figure 27 illustrates a procedure that can be implemented on a ground computer (or any combination of automatic and manual procedures) to provide the ground malfunction isolation capability. Implied within this procedure is a variable format telemetry scheme (not presently implemented) that permits expanded sampling and sampling rates in the computer and PIPA loop areas when a subsystem malfunction occurs. This type of system has been successfully used on other guidance systems. It essentially permits expanded looks at malfunctional areas by temporarily reducing monitoring of other areas so as not to exceed overall allocated bandwidth. The variable format for the computer and PIPA loops could be implemented by uplink command, astronaut operated switch, or both.

Because of the extremely integrated nature of the Computer Subsystem with the Inertial and Optical Subsystems, diagnostic subroutines in the computer can be implemented on a stimuli-response basis to assist in localizing to IMU or PSA electronics. These tests would be called up by astronaut operation of the Display and Keyboard. A ground based program, coupled with system diagnostic routines, can be made adequate for establishing malfunction isolation, spares, and/or alternate procedures.

The onboard test point availability is expected to be adequate for isolating to the spares listed. If special onboard instrumentation is provided, manual isolation of failures by use of an ac-dc VTVM and an oscilloscope could be accomplished. However, this type of approach would be recommended only in the event of dual failure of G & N and telemetry. Possible consideration should be given to simplified manual isolation by block replacement of modules (at some penalty in reliability however) to minimize the specialized knowledge and tedious procedure and intepretation that this type of approach applies.

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Signal Name	Samples (per sec)	Bits (per sample)	Comment
-28 vdc supply	1	8	Adequate for level checks. Will not pick up transients.
IMU 28 v, 800 cps, 1 percent, 0-phase supply	1	8	Adequate for level checks. Will not pick up transients.
Optics 28 v, 800 cps, 1 percent, 0-phase supply	1	8	Adequate for level checks. Will not pick up transients.
2 v, 3200 cps supply	1	8	Adequate for level checks. Will not pick up transients.
28 vdc Bus A or B	FM 1kc		On OR of Bus A and Bus B to check for noise.
IMU 28 vdc operate	1	1	
IMU 28 vdc standby			
Optics 28 vdc	1	1	Assuming TD with level check capability — fine.
IGA torque motor input	10	8	Check of TDA output — includes ADA — will not follow.
IGA 1× resolver sine output	10	8	Should follow most resolver operation — IMU interface signal.
IGA 1× resolver cosine output	10	8	Should follow most resolver operation — IMU interface signal.
IGA servo error	100	8	Primary dynamic check of stabilization loop performance — should track.
MGA servo error	100	8	Primary dynamic check of stabilization loop performance — should track.
MGA torque motor input	10	8	Check of TDA output — includes ADA — will not follow.
MGA 1× resolver sine output	10	8	Should follow most resolver operations — IMU interface signal.
MGA 1× resolver cosine output	10	8	Should follow most resolver operations — IMU interface signal.
OGA torque motor input	10	8	Check of TDA output
OGA 1× resolver sine output	10	8	Should follow most resolver operations — IMU interface signal.
OGA 1× resolver cosine output	10	8	Should follow most resolver operations — IMU interface signal.
OGA servo error	100	8	Primary dynamic check of stabilization loop performance — track.
Trunnion ECU output	10	8	Should enable thorough dynamic analysis.
Sextant trunnion motor drive amplifier output	10	8	Should enable thorough dynamic analysis.
Shaft ECU output	10	8	Should enable thorough dynamic analysis.
Sextant shaft motor drive amplifier output	10	8	Should enable thorough dynamic analysis.
Optics direct trunnion control	1	8	Should enable thorough dynamic analysis.
Optics direct shaft control	1	8	Should enable thorough dynamic analysis.
Zero encode	1	1	Available in AGC if desired.
Coarse align	1	1	Available in AGC if desired.
Fine align	1	1	Available in AGC if desired.
X tracker output	10	8	As above — optics signals adequate for analysis.
Y tracker output	10	8	As above — optics signals adequate for analysis.
Sextant trunnion tachometer feedback	10	8	As above — optics signals adequate for analysis.
Sextant shaft tachometer feedback	10	8	As above — optics signals adequate for analysis.
Scanning telescope trunnion tachometer F/B	1	8	As above — optics signals adequate for analysis.
Scanning telescope shaft tachometer F/B	1	8	As above — optics signals adequate for analysis.
Scanning telescope trunnion motor drive amplifier output	1	8	As above — optics signals adequate for analysis.
Scanning telescope shaft motor drive amplifier output	1	8	As above — optics signals adequate for analysis.
Tracker ON/OFF	1	1	
25-degree Offset			
0-degree Offset			
Optics manual/computer	1	1	

Table 6. Downlink Telemetry Signals

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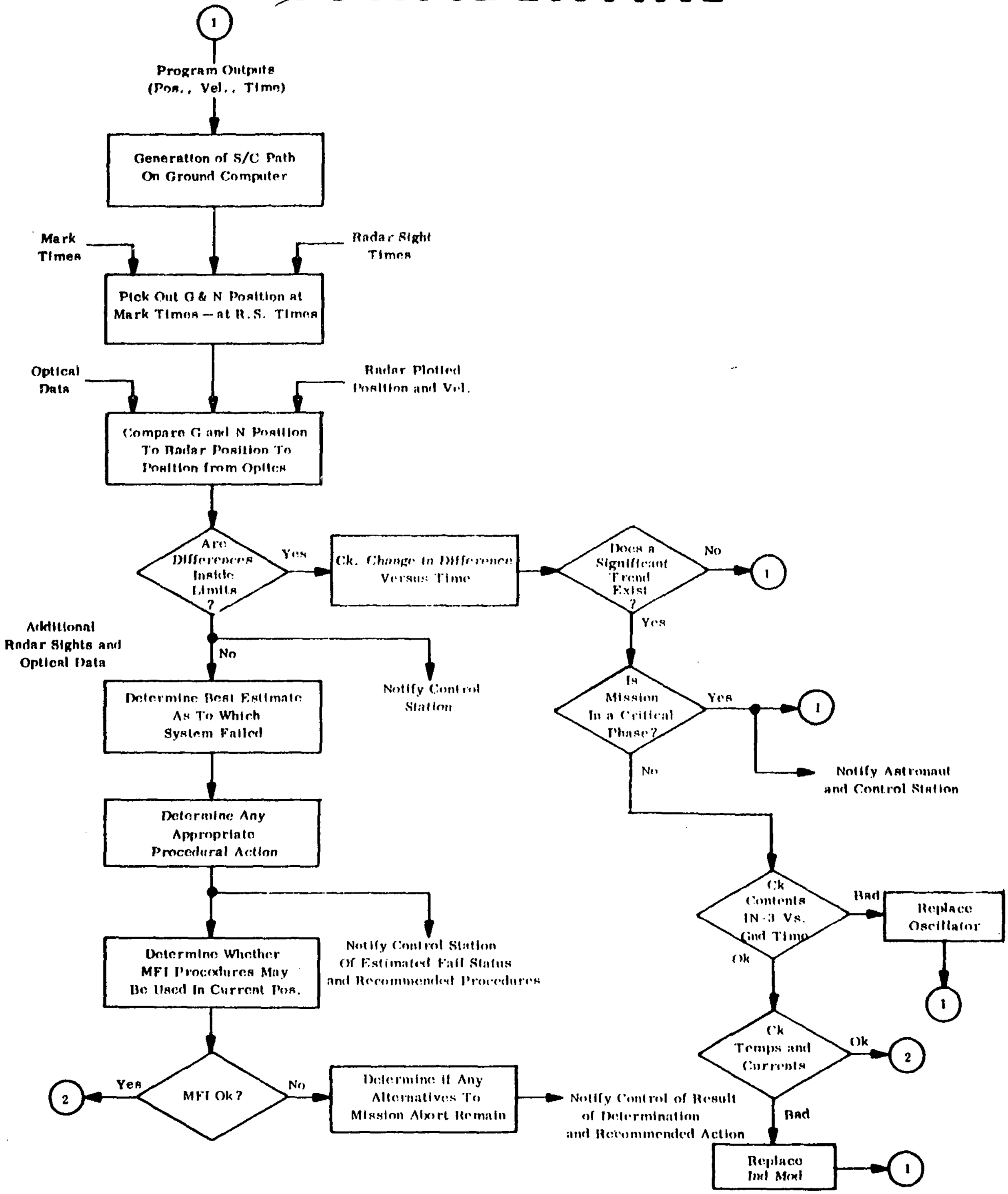

Figure 27. Malfunction Isolation Procedures (Sheet 1 of 6)

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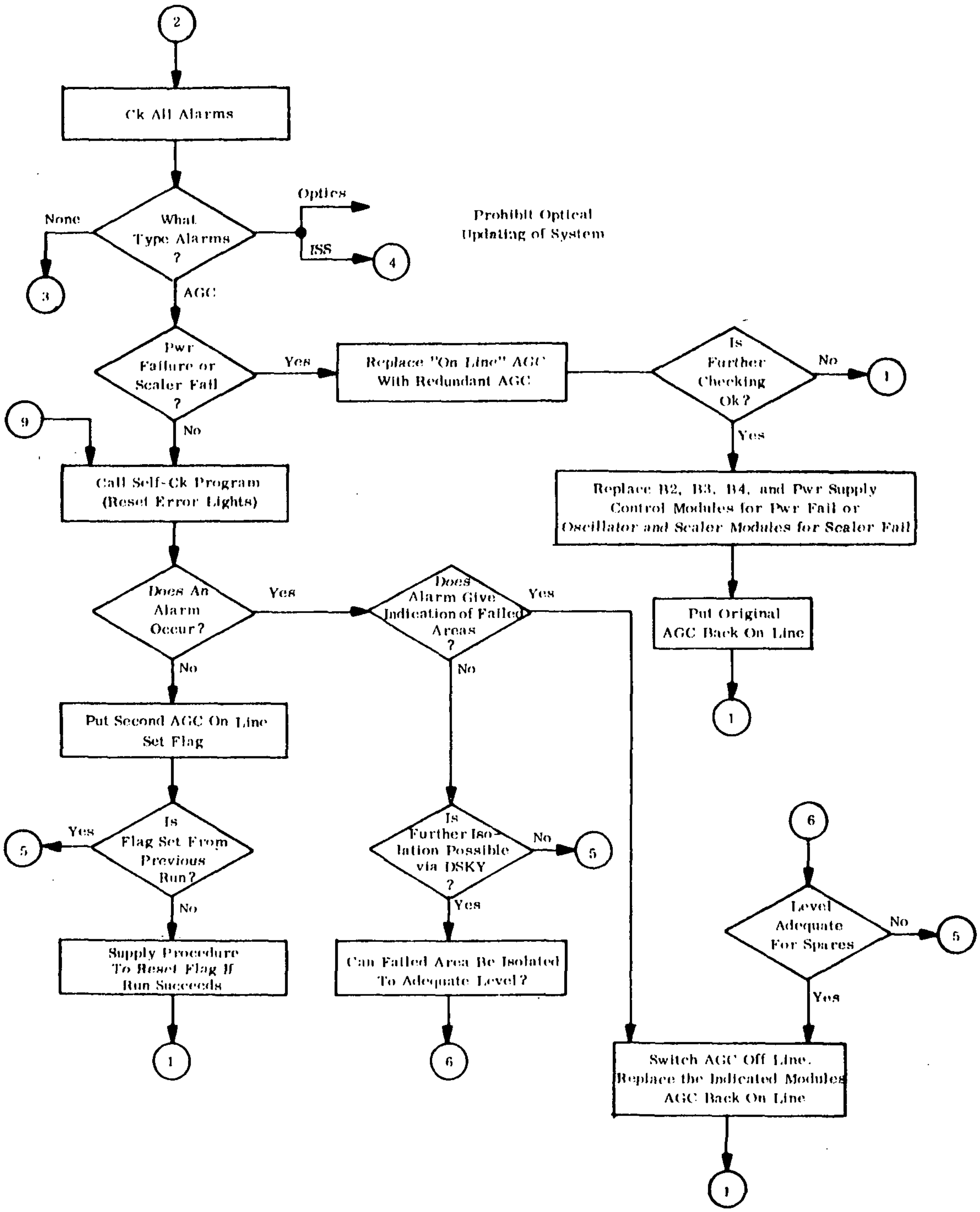

Figure 27. Malfunction Isolation Procedures (Sheet 2 of 6)

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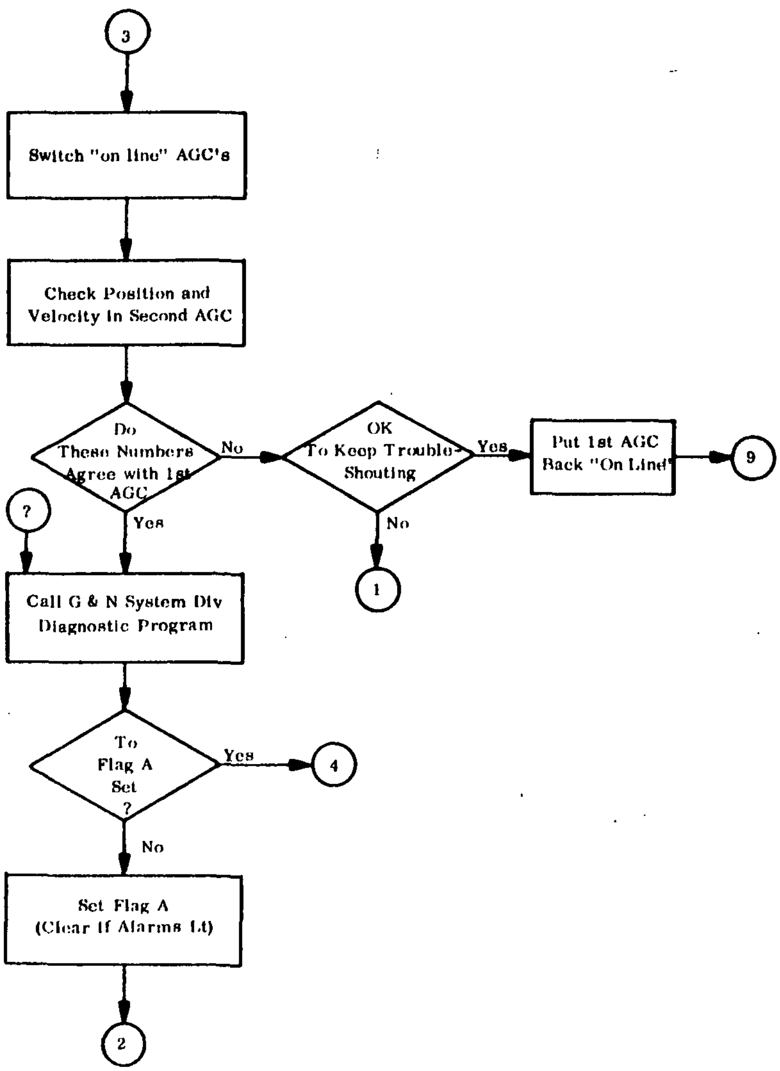

Figure 27. Malfunction Isolation Procedures (Sheet 3 of 6)

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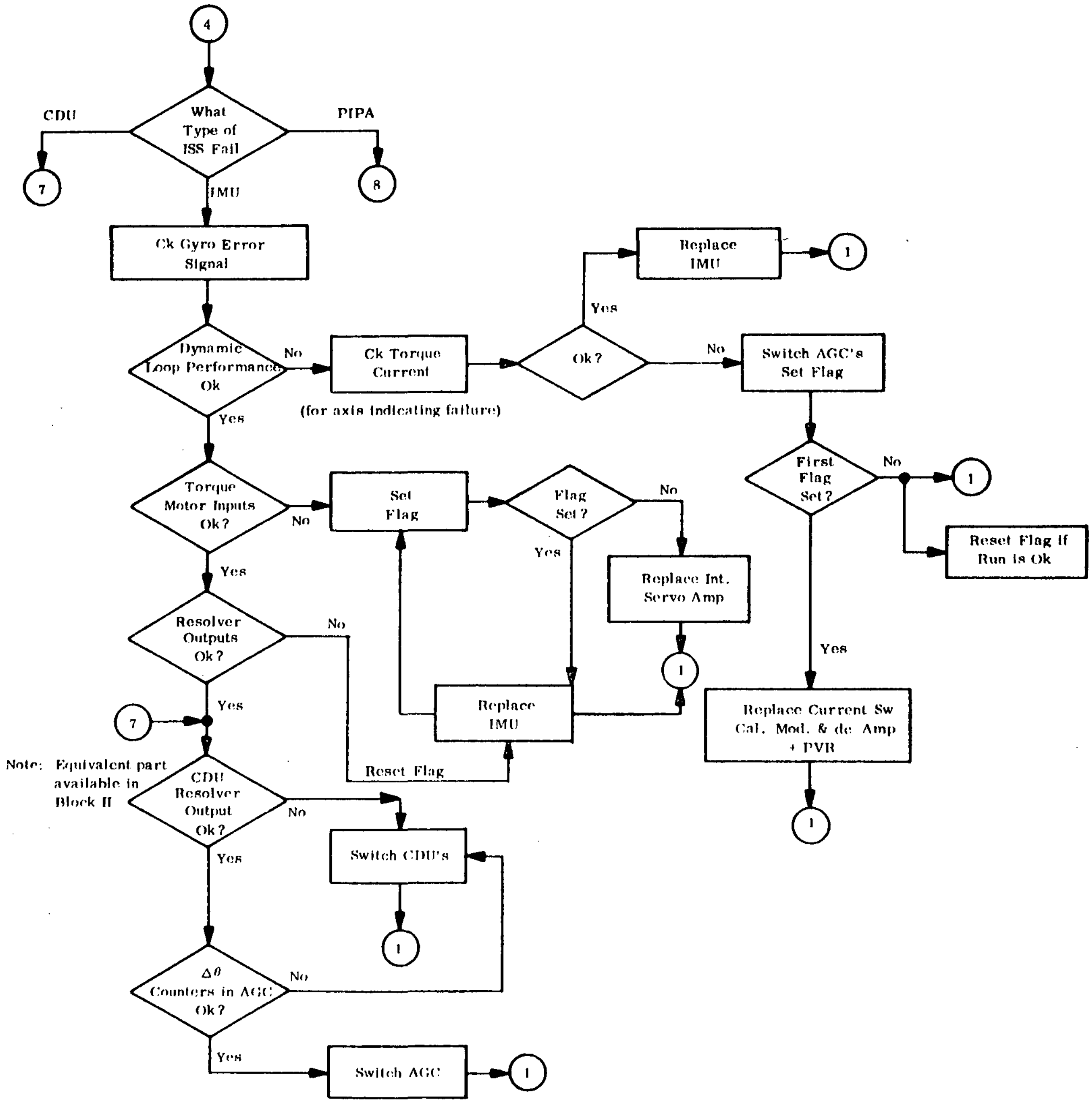

Figure 27. Malfunction Isolation Procedures (Sheet 4 of 6)

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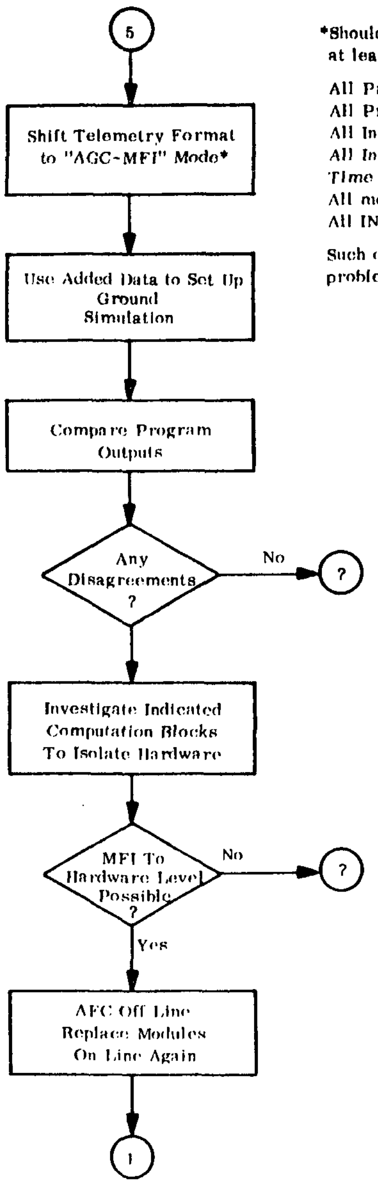

*Should allow the following data at least to go on downlink.

All Program Inputs
All Program Outputs
All Interrupts Ind.
All Increments Ind.
Time of all of above
All moding commands (DSKY)
All IN & OUT Registers
+
Such other data as analysis of problem may indicate.

Figure 27. Malfunction Isolation Procedures (Sheet 5 of 6)

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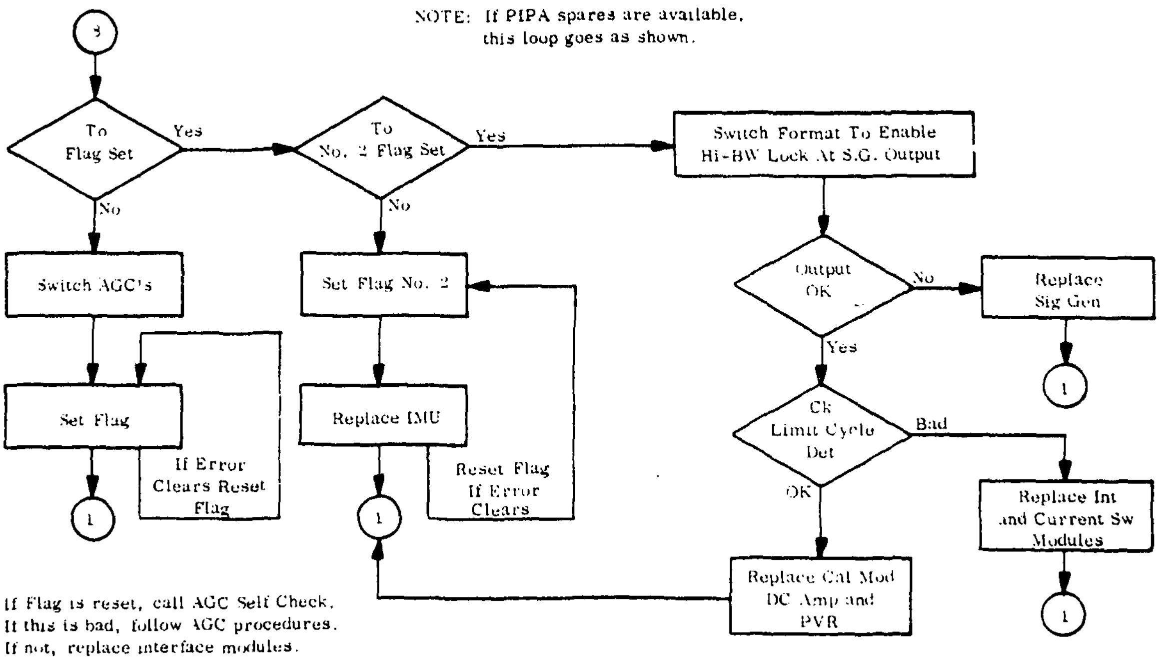

Figure 27. Malfunction Isolation Procedures (Sheet 6 of 6)

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PARAMETRIC STUDIES

Using the modifications, spares, and redundancy as defined for the lunar polar orbit mapping mission, parametric curves of probability of success versus time are shown in Figures 28, 29, and 30 for Profiles 1, 2, and 3, respectively. Curves of probability of success versus weight are shown in Figures 31 and 32 for Profiles 1 and 3, respectively. Since Profile 2 does not have a definite operating length, end-point reliabilities could not be calculated for a curve of reliability versus weight.

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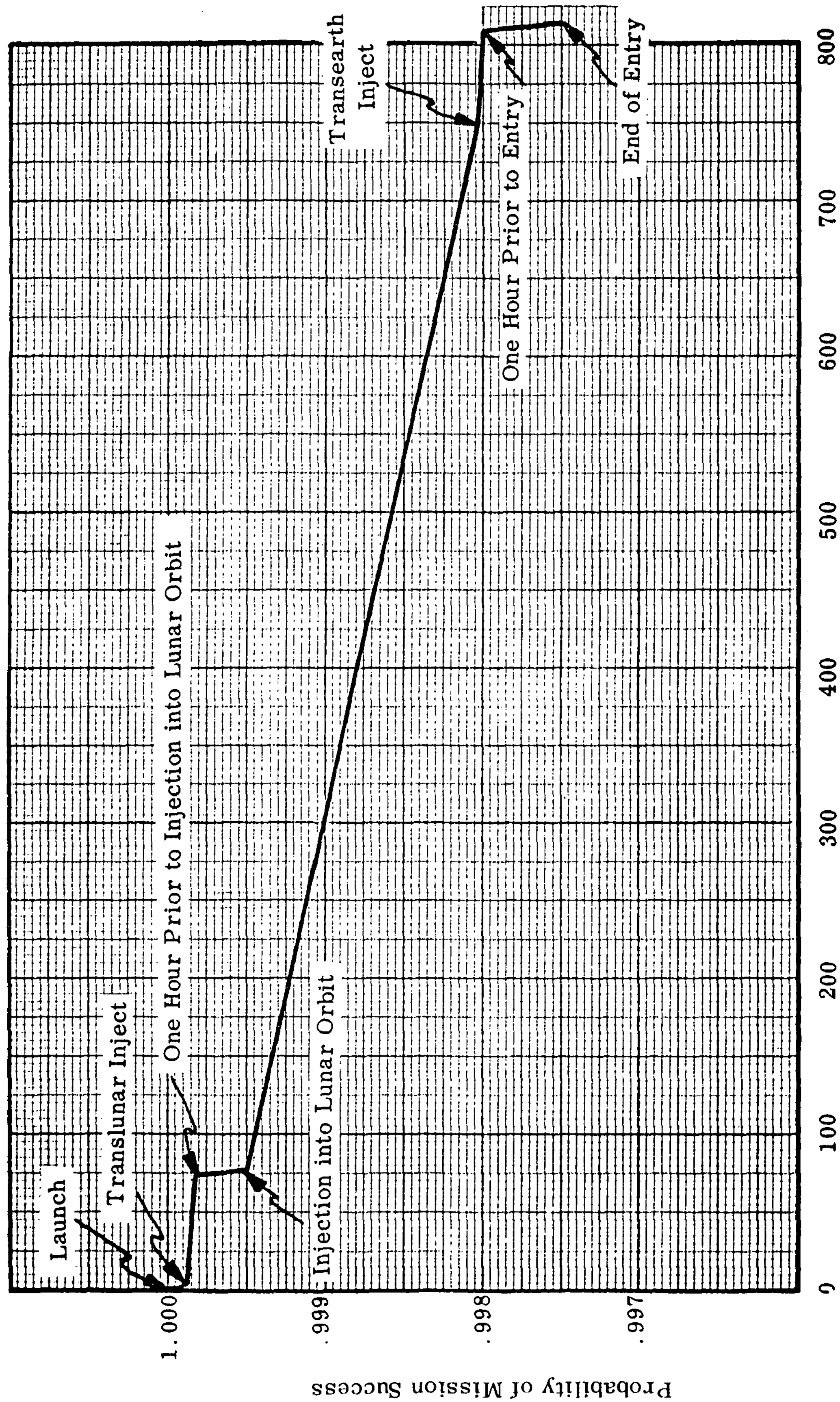

Figure 28. Reliability Versus Time, Profile 1, Apollo X Mission

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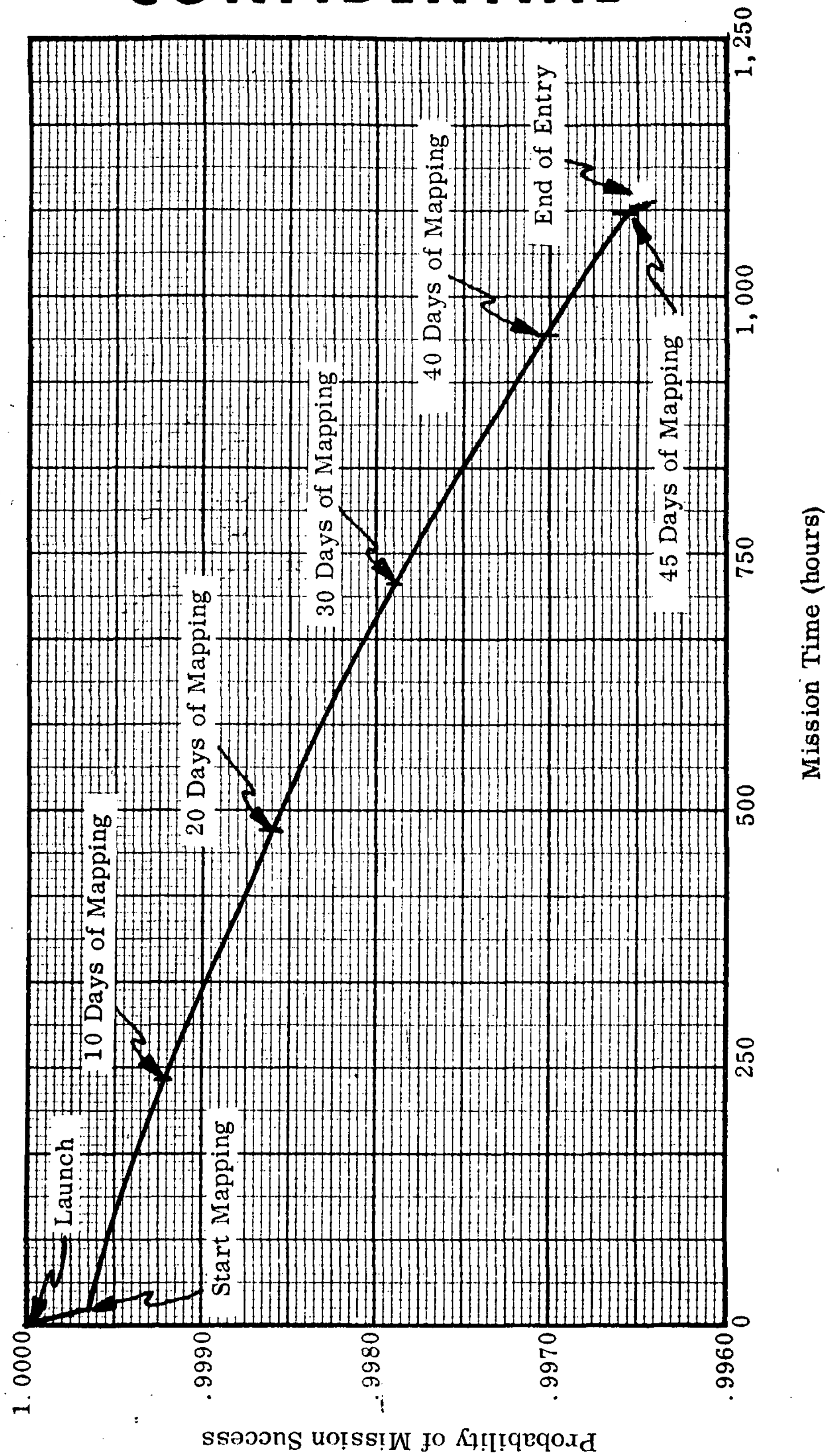

Figure 29. Reliability Versus Time, Profile 2, Apollo X Mission

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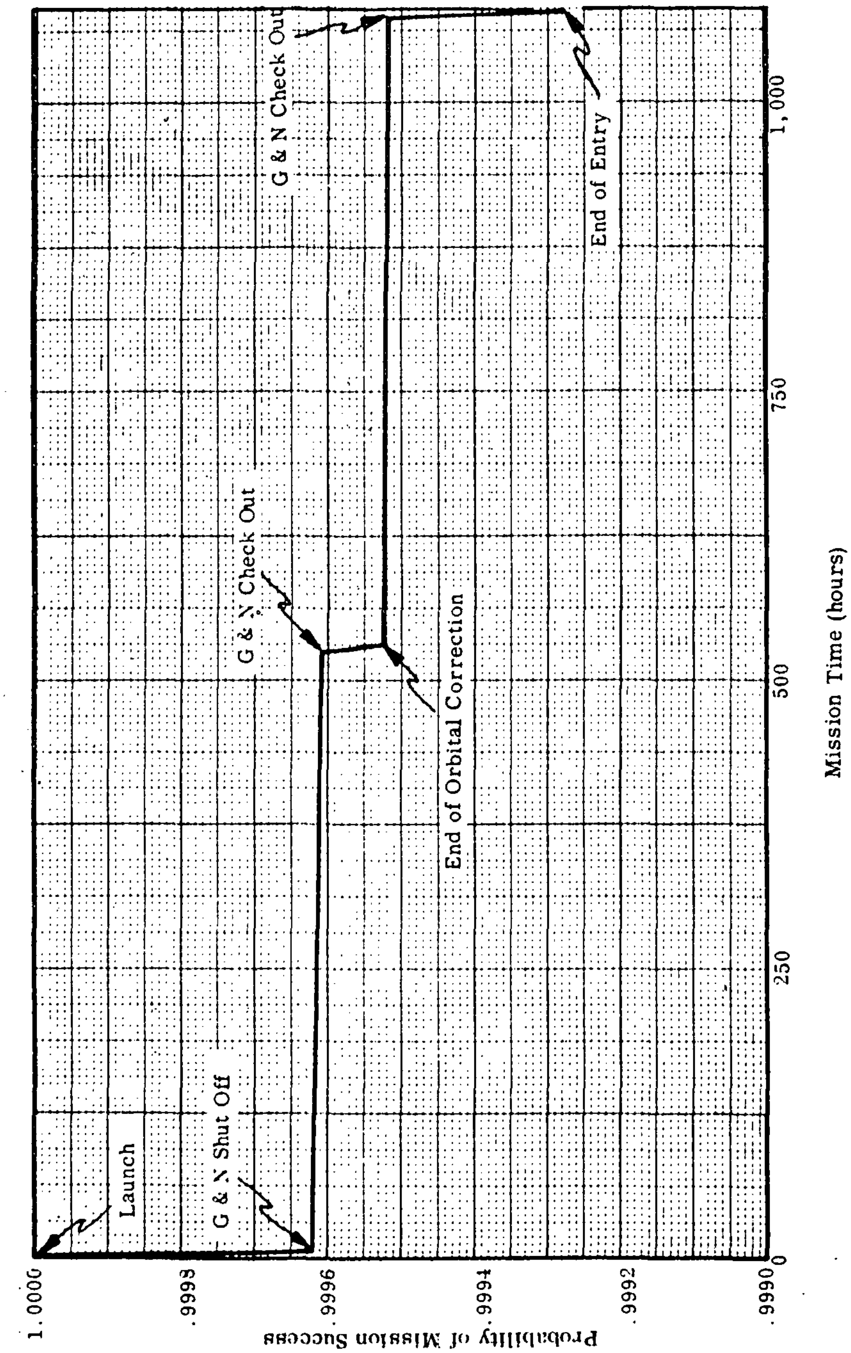

Figure 30. Reliability Versus Time, Profile 3, Apollo X Mission

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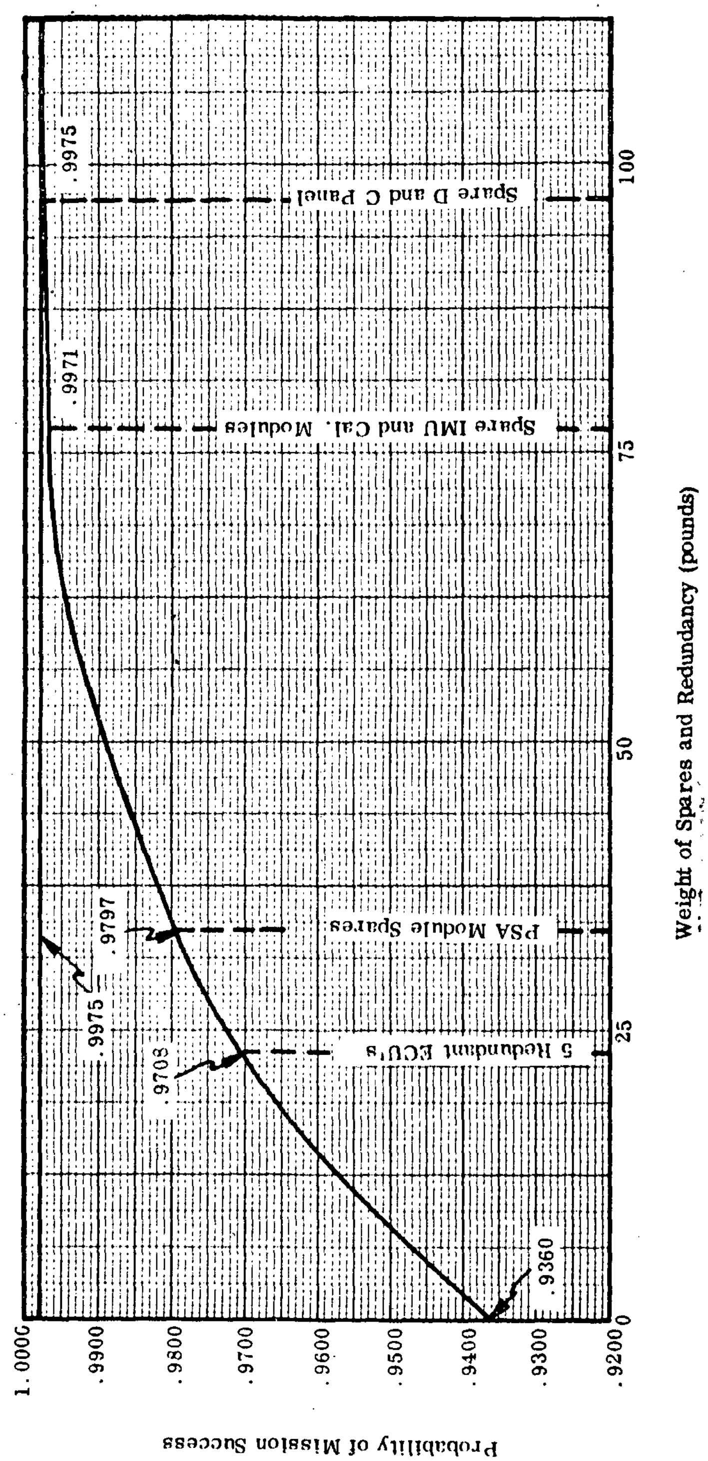

Figure 31. Reliability Versus Weight, Profile 1,Apollo X Mission

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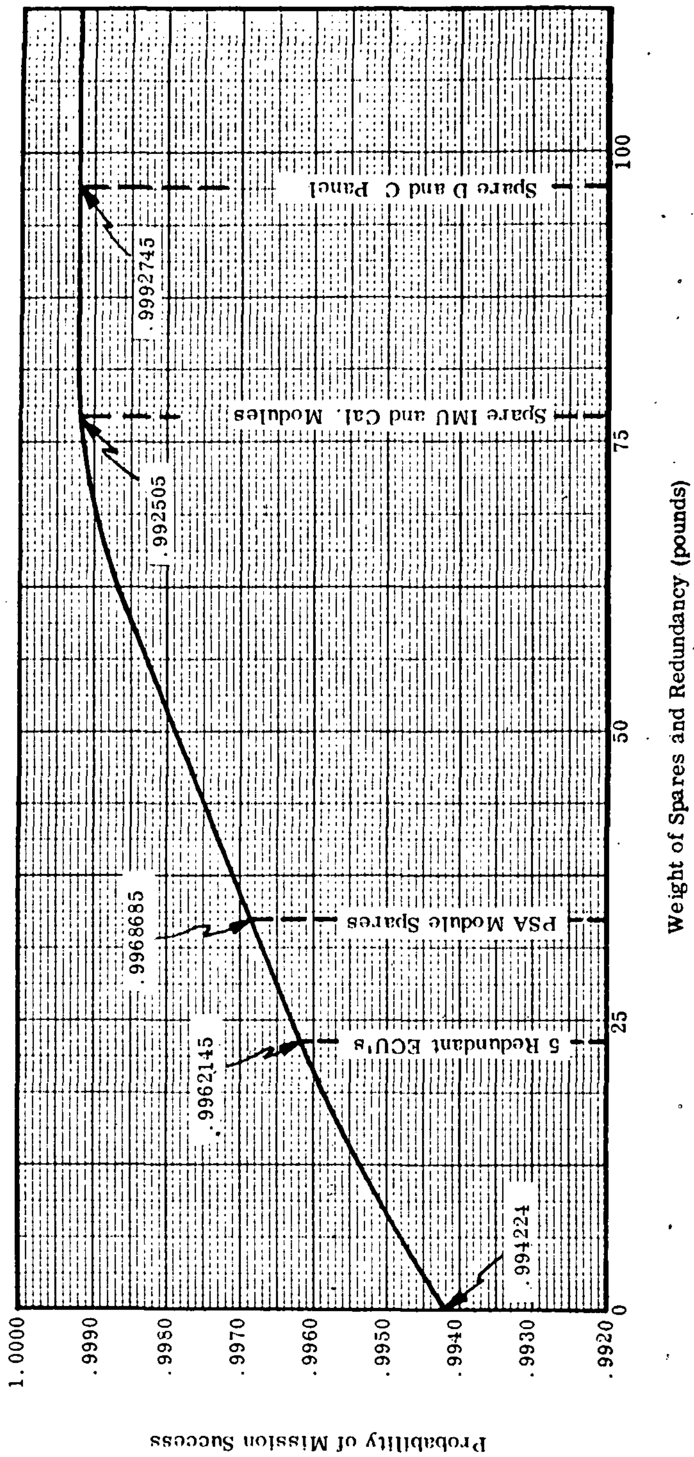

Figure 32. Reliability Versus Weight, Profile 3, Apollo X Mission

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CHANGES FROM BLOCK II SYSTEM

The recommended system at present would consist of the Apollo Block II G & N System with the modifications, spares, and redundancy previously discussed. However, dependent upon certain areas where further definition and study is necessary, the recommended spares and redundancy requirements may be lessened.

With a reduction in the overall G & N System reliability requirement to .9965, or with further engineering development, the following spares configuration applied to the modified Block II System appears ultimately feasible.

- One spare IMU and inertial component calibration modules in the PSA (44.6 pounds),
- Two spare ECU's and one spare each of the 4-volt power supply and the clock and mode logic modules (10 pounds),
- One spare each of the previously spared PSA modules (10.4 pounds).

The G & N System additional weight requirement would then be reduced to 65 pounds of spares and no redundant modules, except possibly some IMU slip rings. The additional packaging and storage requirements would then also be reduced commensurably.

AREAS REQUIRING FURTHER ENGINEERING DEVELOPMENT

Stressed Failure Rates and Failure Effects Analysis

A significant quantity of the reliability predictions have been based on the average failure rates of the parts and modules. A study of electrical stresses on the individual parts in each module will yield higher reliabilities than previously assumed. Specific examples where minor changes in modules and subassemblies can increase reliability will also become apparent. This technique was proven successful in the ECU assembly, where a preliminary parts stress investigation allowed an increased probability of success to be predicted.

Once the actual stressed failure rates are determined, a detailed failure effects analysis can be used to separate anticipated failures into:

- Major G & N failures for a specific mission,
- Minor performance degradation.

From past experience, it is known that without a failure effects analysis, failures that result in minor performance degradation are often erroneously predicted as mission failures.

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Component and Development Testing

The Apollo G & N equipment approaches the state of the art in the areas of component usage and safety factor design. However, some of the remaining failure causes have been traced only to major categories. This is particularly true of slip rings bearings (gimbal, motor-tachometer, and gyro), and torque motor brushes. Testing under conditions that simulate actual usage should be conducted on these components to obtain specific examples of failure. In this manner unique failure modes peculiar only to the G & N equipment can be identified.

Switching Stresses

Power turn-on, turn-off, and switching effects on failure rates and modes of the G & N equipment have not been fully investigated. It is recommended that this be done.

Optics Equipment Improvements

The optics head had a probability of success that barely exceeded the G & N design goal. Therefore the remaining portions of the G & N equipment would have had to achieve probabilities of success approaching unity. A detailed investigation of the components of the optics head should therefore be conducted. If this investigation is followed by a more accurate trade-off and apportionment study for the remainder of the G & N equipment, there is a good chance of eliminating some of the on-board spares presently recommended.

Alternate Procedures

An engineering investigation of the procedures used to perform G & N mission functions is needed. One such investigation resulted in a new procedure for alignment of the inertial reference, which reduced the operating time and simplified the effort required of the astronaut. Another preliminary study on the failure of components used in the optics head revealed that alternate procedures can be established which, with some loss in convenience and accuracy, will allow the astronaut to perform the G & N functions necessary to mission success. These investigations should be continued and expanded into other G & N areas. Alternate procedures, to be used in the event of component failures within the G & N systems, should be established to increase crew safety and the probability of mission success.

Redesign Summary

It is believed that the majority of G & N system components are "state of the art." However, redesign may be worthwhile in the light of technological advances and the studies of stressed failure rates, failure effects, and alternate procedure outlined above.

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IV. TEST PLANS

INTRODUCTION

The purpose of these test plans for the Apollo X Guidance and Navigation (G & N) equipment is to assure that the G & N is developed, manufactured, and tested as a qualified subsystem for the Apollo X mission. The sequence of the Apollo X program progression is shown in Figure 33. The proposed sequence is intended to yield the following advantages:

- Design goals and achievements are subjected to continuous scrutiny with respect to critical mission requirements,
- Manufacturing and inspection integrity are continuously evaluated,
- Potential problem areas are screened several times to eliminate human error,
- Trouble spots are exposed early in the sequence when corrective action can most judiciously be initiated,
- Some measure of reliability and predicted performance has been established before the program of flight tested reliability is started,
- Additional information is available for establishing reliable operation and maintenance requirements.

The G & N System development schedule is integrated with the Lunar Apollo Program (see pages 114 and 115) and represents the best estimate to date of:

- Least conflict of critical facility usage,
- Maximum utilization of present and projected field operations,
- Minimum necessary research and development activity,
- Maximum usage of equipment qualified by the Lunar Apollo Program.

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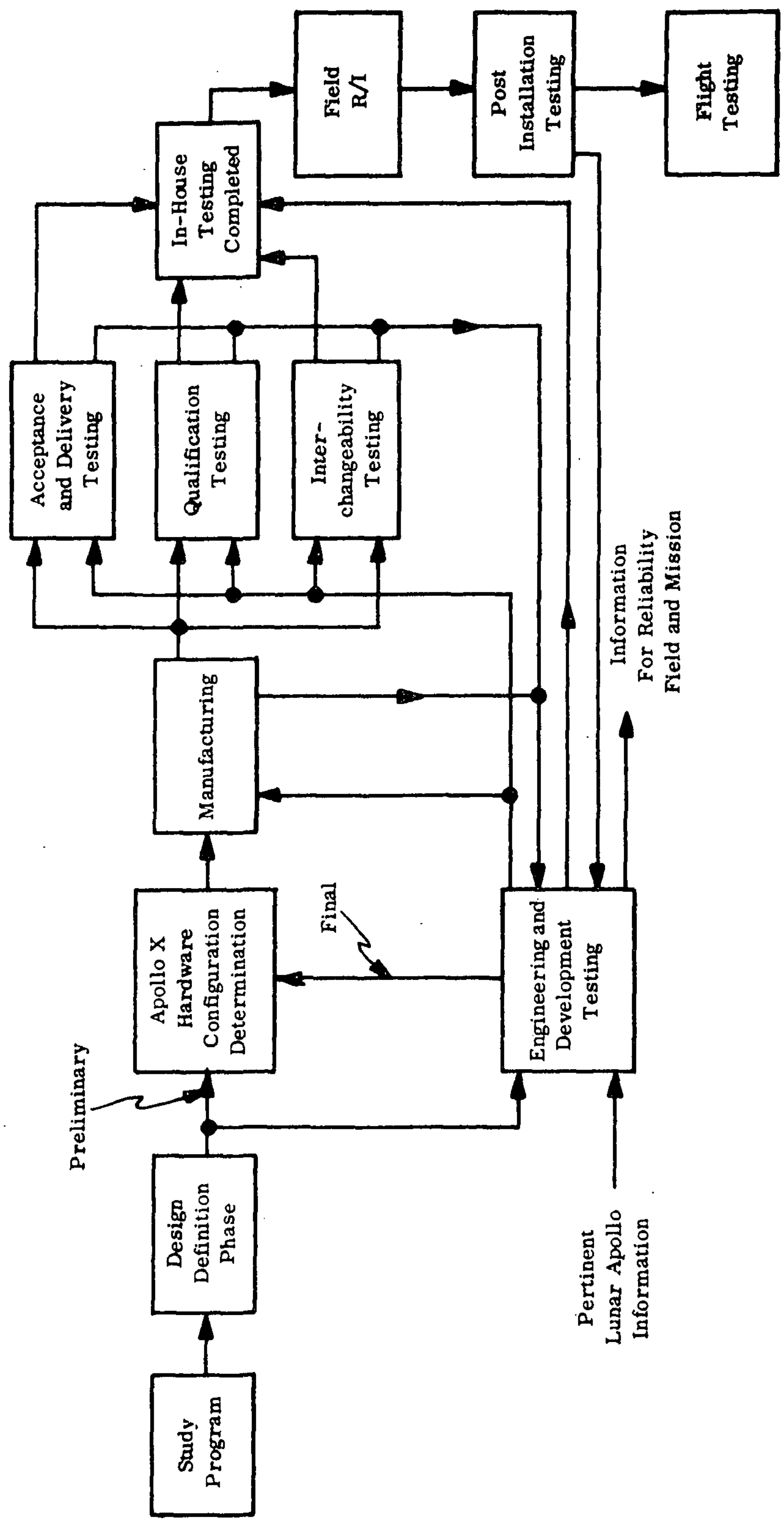

Figure 33. Apollo X G & N Program Progression

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SCOPE

The planning for this study will be presented in four major test areas:

- Design and development,
- Acceptance and delivery,
- Qualification,
- Interchangeability.

Each of the areas will be expanded upon separately and recommendations made for Apollo X. Because of the similarity of Apollo X to the Lunar Apollo equipment and mission, the Lunar Apollo will be considered as a base guideline. Therefore, only additional or unique requirements for Apollo X testing will be fully investigated. The additional constraint is that no planning has been considered for formal reliability demonstrations in compliance with a NAA request.

SUMMARY

All reliability assessments, quality assurance, manufacturing techniques, hardware and design change control, program control and review, liaison procedures, field support, and cost accounting methods presently in existence on the Lunar Apollo Program are recommended to be adapted to the Apollo X Program.

The modifications proposed for Apollo X and the life limited equipment will undergo a reliability and research and development testing program, starting with the definition phase of the project. This program will continue until no further modifications to the G & N can be justifiably proposed. Wherever possible, the results of bench test programs of the Lunar Apollo will be utilized.

An interchangeability and design review testing program will be conducted with a G & N subsystem to evaluate the effects on G & N parameters and performance of onboard spares, repair, and redundancy.

It is recommended that no testing should be done on the electromagnetic interference and susceptibility of the Apollo X G & N equipment. Results from the Lunar Apollo Program can be considered as typical performance of the Apollo X G & N.

The Apollo X G & N is considered qualified for mission vibration, acceleration, and shock by the Lunar Apollo qualification program and normal manufacturing controls, with the exception of the Apollo X PSA package, redundant ECU package, modified D & C group, and on board spare IMU. These units will be qualified for dynamic environments.

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One Apollo X G & N will be subjected to the temperature-altitude-humidity testing requirements of ND 1002037, Apollo Airborne Guidance and Navigation Equipment Environmental Qualification Specification, with the inclusion of the 50 percent nitrogen and 50 percent oxygen atmosphere.

One of each type of Apollo X modules will be exposed to the two-gas atmosphere under temperature-altitude conditions.

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TEST PLANNING

DESIGN AND DEVELOPMENT TESTING

General

During the program definition phase of the Apollo X Program at AC Spark Plug, a milestone will be reached at which time an integrated design and development bench test program will be justified, initiated, and supported. Sufficient detail information will be available to preschedule specific development test tasks between AC Spark Plug, Kollsman, and Raytheon.

All design and development testing considered herein presupposes that in some specific instances:

- Apollo X requirements are not the same as the G & N Block II,
- Lunar Apollo requirements are modified and anticipated similarity of dual functions cease to exist,
- To proceed on a noninterference basis with Lunar Apollo schedules parallel efforts may be required.

Design and Development Testing Objectives

The objectives of the design and development activities and testing for the Apollo X G & N System are to assure the following.

- Preliminary designs, mechanizations, and goals established by the study program are optimized and integrated with a realizable equipment and testing schedule.
- Reliability of the G & N spacecraft equipment is maintained, improved, and definitized throughout the program.
- Complete manufacturing definitions, test plans, test requirements, and adequate support equipment are specified, evaluated, proof tested, and supported throughout the program. This includes both inhouse, subcontractor, and field locations.
- Adequate electrical and mechanical G & N interfaces with the spacecraft are determined and maintained through sufficient liaison with NASA and NAA.
- Necessary spacecraft crew utilization, maintenance, and emergency repair capabilities have been reflected into the design of the G & N equipment.

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Design and Development Testing Plan Implementation

During the definition phase of the program, several areas of design improvements and test results required to obtain increased reliability will have been investigated and changes defined or anticipated. These are expected to include requirements for:

- Redesigned PSA module packaging for flight replacement and maintainability,
- Unique packaging and test requirements of redundant ECU's,
- Mechanizations, circuits, and procedures for accelerometer loop turn-off and inertial component suspension turn-off,
- Redundant slip ring usage where indicated by reliability studies for temperature control, attitude control, PIPA loops, and IMU power interfaces,
- Minimization of vibration transmissibilities to critically aligned components such as navigation base — IMU — optics alignments and inertial components,
- Reduction of electrical stress factors within modules by partial component redundancy and upgraded components,
- Mechanization simplifications for the reduction of possible failures and malfunctions,
- Specific recommendations for engineering test evaluations of life-limited rotating components, environmentally affected sealing, and pressurization materials and components,
- Integrated proof testing of Apollo X G & N equipment change impact on support and handling equipment and associated test procedures,
- Verification of any new guidance computer flight, ground test, field test, and inhouse test routines,
- Establishment of initial electrical and mechanical spacecraft interfaces,
- Integrated task analysis with respect to inflight maintenance and onboard spares replacement,
- Confidence checks of failure effects analysis, error analyses, and test tolerance studies,
- Thermal evaluation of proposed mechanizations.

All of the above anticipated developments will generate logical demands for initial circuit breadboards and testing, both for flight and ground support equipment. These should be followed up with engineering developmental proof models as warranted by the particular test results and analysis. In addition, ground support equipment and Apollo X G & N flight hardware compatibility should reasonably be expected to be established and maintained. To adequately define spacecraft mounting configurations, a mechanical gage or mockup Apollo X G & N model would be required.

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Design and Development Data Requirements

All test data from the Lunar Apollo Program will be reviewed. In addition, current results from the Apollo X testing will be reviewed. Reliability assessments of modules, major units, and components must be available and augmented by failure rates unique to Apollo X G & N equipment. A moderate amount of digital computer machine time will be required for servo loop analysis and new guidance computer routine operation. Adequate analog computer time will be used to accomplish the design objectives.

Design and Development Equipment Required

To support the design and development objectives will require:

- One G & N Block II System, updated with developmental proof models to an Apollo X configuration. This includes requirements for developmental proof models of the Kollsman optics and the Raytheon computer with two display keyboards.
- One G & N Block II GSE and universal test site support and calibration equipment, updated to Apollo X configuration, with developmental proof models.
- Flight hardware and GSE breadboards.
- One accelerometer
- One gyroscope

  (One accelerometer and one gyroscope:) Status and evaluation test area. To be determined at time of receipt
- Miscellaneous breadboard parts.
- One mechanical gage system.

Design and Development Facilities

Approximately 30 × 30 feet of floor test space with a 13.5 × 13.5 foot vibration isolated pad is required. In addition, approximately 15 × 15 feet will be required for the mechanical gage system, and approximately 20 × 20 feet of circuit breadboard area will be required Usage of all areas will be integrated with Lunar Apollo where possible on a noninterference basis

ACCEPTANCE AND DELIVERY TESTING

Acceptance and Delivery Testing Objective

The objective of acceptance and delivery testing is to maintain manufacturing and inspection integrity with respect to material, component, module, subassembly, subsystem, and G & N performance.

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Acceptance and Delivery Testing Plan Implementation

To accomplish the testing objective, the test flowgram of Figure 34 is proposed for the Apollo X program. The flowgram sequence is very similar to the type employed on the Lunar Apollo flowgram. The main areas of difference are: an additional spare IMU will be tested, a PSA-ECU combination with redundant ECU's, a minimally modified G & N Indicator panel, and the additional onboard spare subassemblies and modules testing.

Testing of the spare IMU will be accomplished by: serial testing of a G & N with two IMU's, parallel testing two ISS configurations, or by the use of a standard ISS configuration for testing all spare IMU's. The method of testing will be determined by the integrated schedule of equipment usage. A preliminary study indicates that a standard ISS configuration for testing all spare IMU's represents a minimal equipment usage. The onboard spares can be tested at the subassembly or module level; however, the interchangeability testing program will verify the assumption. Should the interchangeability program prove this assumption incorrect to any extent, the integrated testing of spares at the ISS, OSS, or G & N level can be integrated without any additional facilities or test equipment.

Checkout and initial testing of the optics and map and data viewer is proposed to be done at Kollsman Instrument Corporation. Checkout and initial testing of the guidance computer and display and keyboard is proposed to be done at Raytheon. However, AC Spark Plug will perform receiving inspection tests on these components prior to acceptance.

The following areas of activity will be necessary, starting with the program definition phase and continuing through to the end of the delivery program to assure timely delivery.

- Initiation and continuing monitor of PERT planning,
- Documentation of manufacturing effort, that is, manufacturing progress and documents defining equipment and building techniques,
- Procurement of material, components, and vendor selection,
- Monitor vendor and subcontractor scheduling with respect to delivery schedule,
- Determination of plant layout and integrated usage of manufacturing facilities,
- Determination, design, evaluation, procurement, maintenance, and calibration of test equipment,
- Development and implementation of fabrication and assembly techniques,
- Monitor and implementation of adequate manufacturing personnel training,

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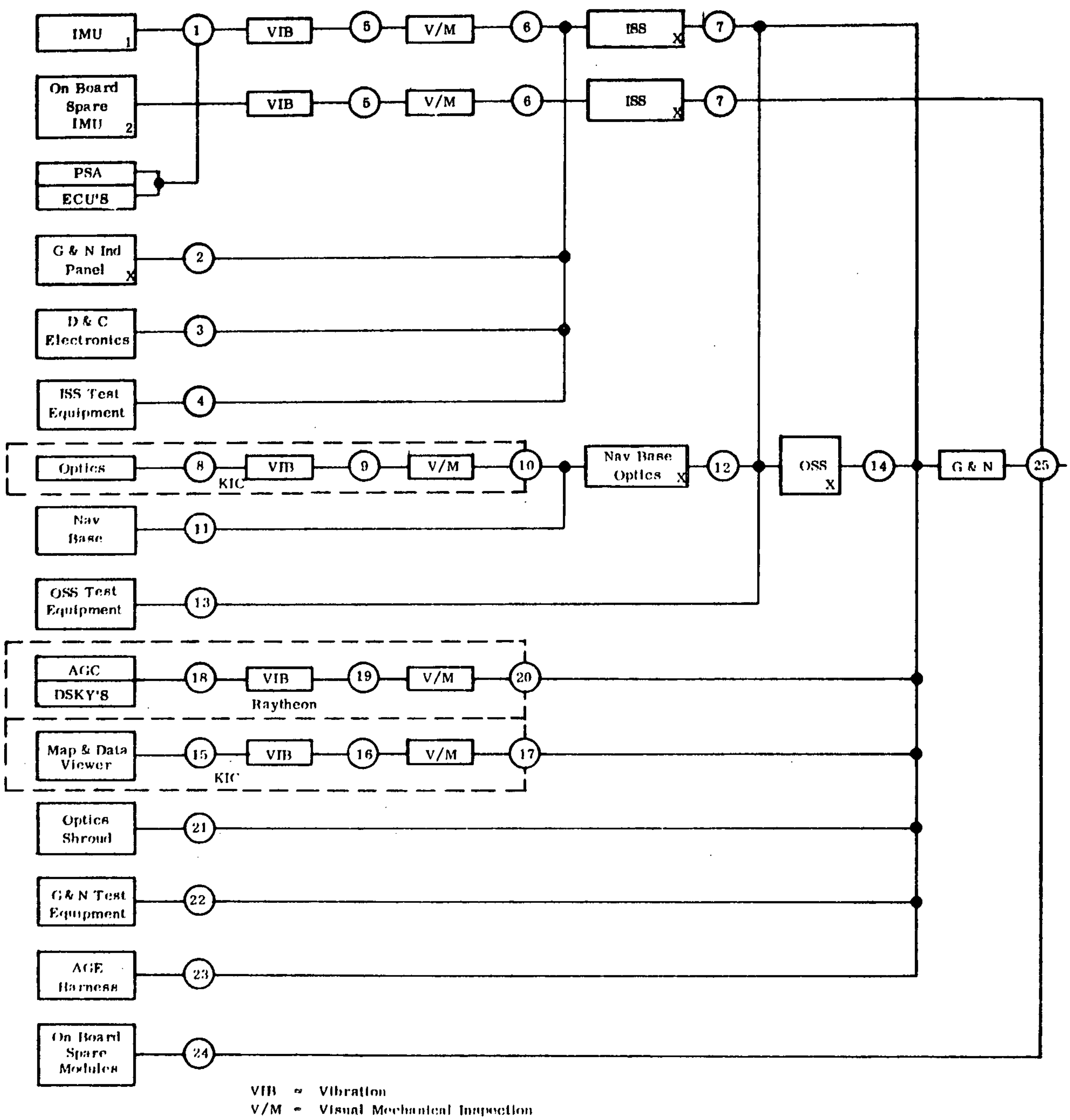

Figure 34. Manufacturing Flowgram

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- Institution of corrective action and resolution of manufacturing and testing problems,
- Assistance and liaison with Engineering during initial breadboard phase,
- Determination of adequate lead times for critical components, subassemblies, and processes for timely manufacturing and delivery,
- Determination and maintenance of adequate quality assurance and test records.

The activities previously discussed will rely heavily on experience gained from the Lunar Apollo Program, and will employ the same techniques.

Data Handling

Complete data history and lot control of incoming materials, parts, modules, subassemblies, and major units will be maintained. These records will contain status, location, and disposition of all articles. In addition, all testing data required by FTM, ATP, and JDC must adequately be monitored, prepared, reproduced, distributed, and kept on file for evaluation. PERT planning and summary sheets will periodically be required. Failure reporting and corrective action summaries must be periodically distributed.

Equipment Required

The acceptance and testing program will require a complete GSE to equip a universal G & N ISS-OSS test site, a complete GSE to equip an ISS-OSS test site, and a complete set of GFP, GSE, and GOE auxiliary equipment, and site equipment to support both test sites. This equipment's integrated usage with the Lunar Apollo Program is to be determined.

A 16 PIPA test console (GFP) and support equipment will be required, which includes as major items a precision rotary head (GFP) and a 36 × 36 inch leveled surface plate. In addition, modification kits for present inhouse test fixtures will be updated for Apollo X G & N equipment. The integrated usage of this equipment with the Lunar Apollo Program is also to be determined.

Acceptance and Delivery Testing Facilities Requirement

To support the acceptance and delivery testing, a minimum of 50 × 50 feet of floor space of test area is required. This test area should contain a 13.5 × 13.5 foot isolated pad and four precision aligned optical targets. In addition, the environment of the test area should be as follows.

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- Air conditioned (temperature 75° ± 4° F, relative humidity 50 percent or less)
- Pressure differential (internal positive pressure of 0.05" H_2O)
- Filtering method (final filter HEPA type)
- Particle count tolerance (determined by activity, size, and product through empirical analysis)
- Particle size and count (adjacent to optical unit). A count of 100,000 particles per cubic foot of 5.0 micron or larger. This requirement applies only to periods when the sextant and scanning telescope optics heads are uncovered.
- Hoods (when required to contain dust sources or vapors)
- Volatile vapors (toxic vapors vented through hoods and air conditioning)
- Floors (smooth waxed tile)
- Interior finish (hard finish paint)
- Utilities and fixtures (standard benches — open storage)
- Cleanliness (layout with consideration to cleaning access — cleaning weekly or more frequently, as required)
- Cover garments (caps, gowns, and boots)
- Entrance chamber (required)
- Additional floor space is needed to support inertial component prealignment

INTERCHANGEABILITY TESTING

The Apollo X is predicated on onboard flight-replaceable modules, subassemblies, and components of the G & N System. A formal program of interchangeability testing is an important part of an integrated test plan to demonstrate the integrity of G & N spacecraft interfaces when spares are inserted in the G & N. The need for tested interchangeability is a unique requirement for Apollo X. No similar-type test is conducted on the Lunar Apollo Program, although the design itself reflects interchangeability to the presently scheduled field sparing levels.

Interchangeability Testing Objectives

The objectives for interchangeability testing are as follows.

- A formal demonstration that modules, subassemblies, and major portions can be replaced in the G & N without recalibration and without degrading the performance.
- To establish the relationship between G & N parameters and spares replacements.

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Interchangeability Testing Plan Implementation

Interchangeability testing will be conducted at AC Spark Plug by Manufacturing and monitored by Quality Assurance. Initial data analyses will be the joint responsibility of Quality Assurance and Engineering; final data analyses will be the responsibility of Engineering. The G & N will be installed in a universal test station area capable of conducting tests with both ISS-OSS and G & N configurations. After an initial G & N acceptance and delivery test to establish a reference for further data evaluation, each module, subassembly, or major component of the onboard spares will be substituted into the G & N. Testing will then be conducted to evaluate the changes in G & N parameters. The process will then be repeated until every piece of the spares has been evaluated in the G & N. Where necessary, the G & N will be rearranged into an Inertial Subsystem or Optical Subsystem configuration to accomplish the evaluation of the spare replacements. This interchangeability test flowgram is shown in Figure 35. All testing will be conducted at normal room ambient environmental conditions. Upon completion of the interchangeability testing, necessary refurbishment and formal testing can be accomplished on the G & N and used to update or support as a spare the NAA environmental test vehicle, ground test house spacecraft, or the NAA G & N Block II house spacecraft. Except for the refurbishment, this disposition is most feasibly accomplished on site.

Proof of Accomplishment

The satisfactory completion of those tests in the final test method (FTM) that bear a significant relationship to the module, subassembly, or spare unit replaced in the G & N will constitute an adequate demonstration of interchangeability. The tests conducted will be of the following, but not necessarily limited to the list.

- Standby Power-On Test
- Operate Power-On Test
- Failure Indicating Circuitry Test
- IMU Temperature Control
- G & N System Power Supply Test
- Map and Data Viewer Test
- G & N Panel Brightness and Lamp Test
- Guidance Computer Operational Test
- Zero Optics Test
- Optics Slew Rate Test

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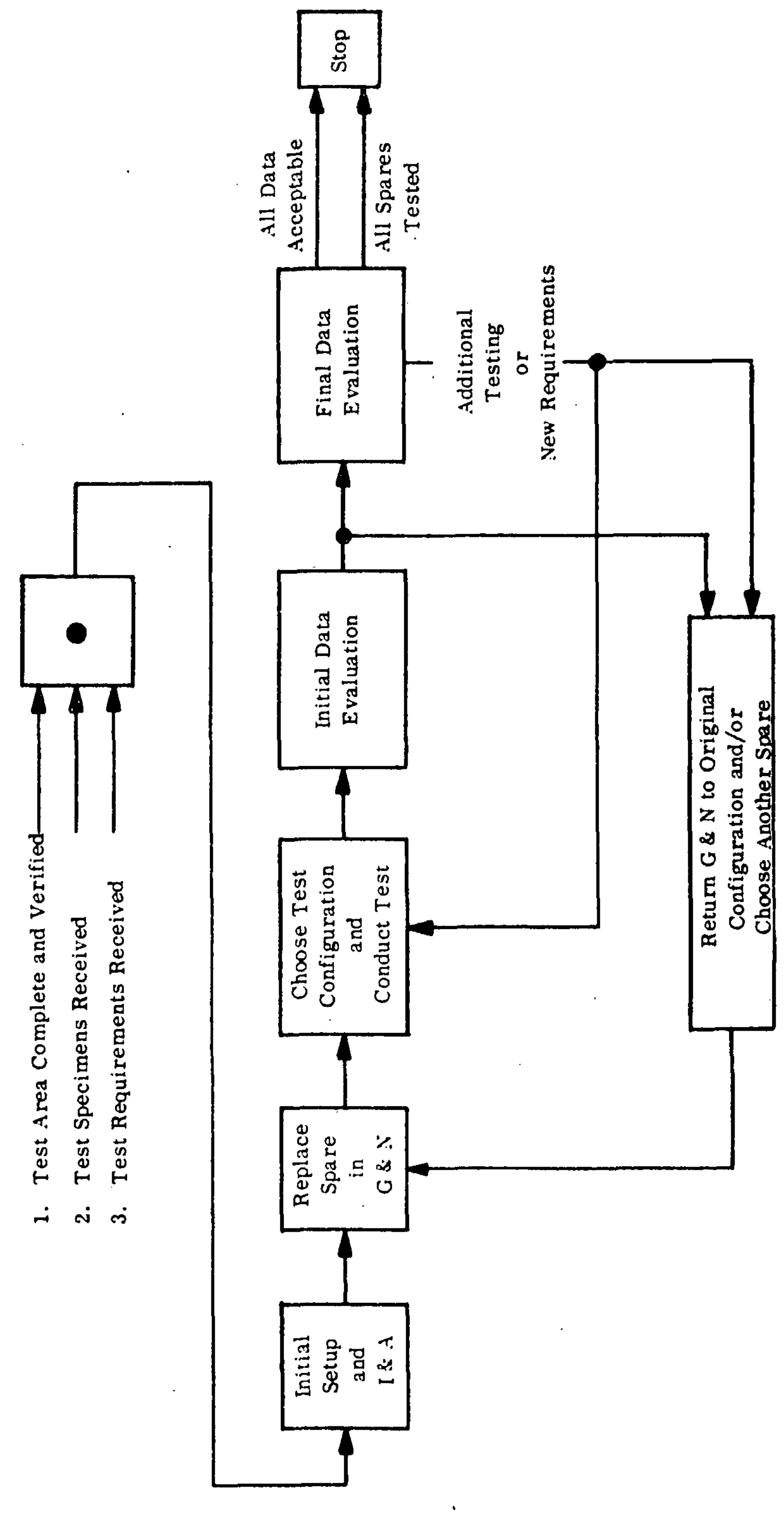

Figure 35. Interchangeability Test Flowgram

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- Optics Resolution Test
- Guidance Computer Mode Control Test
- Manual Mode Control Test
- Minimum Impulse Controller Test
- IRIG Scale Factor and Gimbal Torque Test
- Gyrocompassing Test
- IRIG Coefficient Determination and Simulated Space Fine Align
- PIPA Scale Factor Test

The final data evaluation will determine the completion and adequacy of the demonstration.

Interchangeability Testing Data Requirements

Data Acquired

Data to be analyzed and reviewed will be specified by the particular job description cards (JDC) performed. This will constitute the main bulk of data. Additional data required will be:

- Gyro, accelerometer, and optics characteristics at periodic intervals,
- Inertial components temperature log,
- G & N System test environments and operating time,
- G & N test configuration,
- Sufficient information of subsystem, system, and assembly serial numbers to enable accurate time correlation techniques.

Data Analysis

The data analysis of the interchangeability testing can be divided into three main categories: JDC required analysis, Engineering analysis and justifications, and computer parameter variations. JDC required analysis, essentially a "go," "no-go" basis, will be used as a criterion for proceeding with the testing. The final data evaluation will use any combination of correlation and regression, statistical variance, difference equations, or analog simulation to effectively analyze the signal or parameter variations. This type of data analysis will be on an Engineering basis. Analysis and justification

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will also be performed for any necessary adjustments which may have to be made to maintain G & N tolerances. Where allowable "end-to-end" performance requirements are not met due to interchanging components of the Apollo X G & N, the parameter variations will be used with a G & N error model and computer analyzed for mission performance degradation.

Interchangeability Equipment Required

A complete Apollo X G & N System, including onboard spares and redundant units, will be required. The units of the G & N will be of the QA Class A Manufacturing status.

The support equipment required includes a complete GSE to equip a universal G & N, ISS-OSS, and test site, and one complete set of GFP, GSE, GOE auxiliary equipment, and site equipment to support a universal test site. Integrated usage of infrequently used calibration equipment with acceptance and delivery testing may be feasible.

Interchangeability Testing Facilities Requirement

To support the interchangeability testing a minimum of 20 × 20 feet of floor space of test area is required. Other requirements are identical to those previously listed for the Acceptance and Delivery Testing Facilities.

QUALIFICATION TESTING

Qualification testing of the Apollo X G & N is based on the philosophy of "similarity" with the Lunar Apollo command module equipment. A comparison of the Apollo X and Lunar Apollo mission operating environments and stresses reveals that they are basically the same except for flight duration. Therefore, the extent of testing has been limited to:

- Unique environments of Apollo X,
- Dynamic testing of significant structural or mounting variations from the Lunar Apollo equipment.

The entire philosophy of the qualification testing is to use the most severe environment or stress where a choice between similar environments exists. Per NAA request, no provisions have been made for formal G & N reliability, life, or design safety factor testing.

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Qualification Objectives

The main goal of the qualification program for the Apollo X G & N is to assure that the environmental design considerations of the G & N have been met with respect to the mission requirements. The main objectives are as follows.

- Demonstration that the unique structural configurations of Apollo X G & N will withstand the dynamic stresses of ND 1002037 without causing equipment degradation.
- Evaluation of the possible effects of material contamination or material interaction with the two-gas atmosphere reflecting into adverse equipment performance.
- Demonstration of the capabilities of Apollo X modules and the full G & N to perform within specified tolerances when subjected to the temperature, temperature-altitude, and humidity exposures as detailed in ND 1002037 with the addition of a 50-percent oxygen, 50-percent nitrogen atmosphere during the exposures.

At present, NASA utilizes ambient earth atmosphere as a gas environment. If the Lunar Apollo qualification is eventually modified to reflect 100-percent oxygen, testing in the two-gas atmosphere exposure proposed herein could be extensively reduced.

Qualification Testing Plan Implementation

Table 7 lists the applicable paragraphs of ND 1002037 that will be performed on Apollo X G & N equipment. The Apollo X qualification program was derived with the following assumptions.

- Results from the Lunar Apollo Program concerning electromagnetic interference and susceptibility will serve to qualify the Apollo X.
- With the exceptions of the onboard spare IMU, PSA, and ECU packages, the Apollo X G & N equipment will be qualified for mission vibration, acceleration, and shock by the Lunar Apollo qualification program and normal manufacturing controls.

The entire Apollo X G & N qualification program will take place at AC Spark Plug. The individual flowgrams, Figures 36 through 40, show the testing to be accomplished. The arrangement is only tentative, should facility usage cause conflicts, the order of testing may be rearranged.

Due to the length of time before qualification testing is to begin, it does not appear feasible to detail the performance and test configurations of the PSA, ECU, D & C group, and spare IMU. Rather, it is proposed to submit detail test planning and procedures for each unit prior to conducting the qualification test for that unit. A sufficient time in the scheduling has been allowed for this activity.

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Table 7. Apollo X Qualification Summary

Environmental Exposures	Status	Test Paragraph ND-1002037	IMU/NB ACSP	Spare IMU ACSP	PSA ACSP	ECU ACSP	D & C Group ACSP	MDV KIC	SCT-SXT KIC	AGC RAY	Modules ACSP	Apollo X G & N ACSP
Temperature-Altitude	OP	4.3.1									1*	1*
Temperature	NOP	4.3.2									2*	2*
EMI and Susceptibility	OP	4.3.3										
Humidity	OP	4.3.4									3*	3*
Explosion	OP	4.3.6					3					
Acceleration-Launch	NOP	4.3.7.1		4								
Acceleration-Entry	OP	4.3.7.2			3†	4	4					
Vibration-Launch	OP	4.3.8.1			1	2	1					
Vibration-Launch	NOP	4.3.8.1		1								
Vibration-Flight	OP	4.3.8.2				3						
Vibration-Abort	OP	4.3.8.4			2	1	2					
Vibration-Abort	NOP	4.3.8.4		3								
Vibration-Entry	OP	4.3.8.3		2								
Vibration-Entry	NOP	4.3.8.3										
Shock-Flight	OP	4.3.9.1		5	4	5	5					
Shock-Earth Landing	NOP	4.3.9.2		6**	5**	6*	6**					
Overstress	NOP/OP	4.3.10										
Salt Spray	NOP/OP	4.3.5										

*Test condition to include atmosphere of 50-percent nitrogen, 50-percent oxygen, at 7 psia.

†To be operated essentially NOP except for all available relay continuity.

** Simulated hardware may be used during this phase.

NOTE: The numerical ranking of tests is shown to indicate the desired or logical sequence for testing. This sequence is compatible with other Apollo facility scheduling.

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[1] → (1) → [2] → (2) → [3] → (3) → [4]

[1] Initial inspection and acceptance test per FTM

(1) Temperature altiude exposure in two-gas atmosphere approximately 1 per day partial functional

[2] Post-temperature — altitude inspection and acceptance

(2) High and low temperature exposure in two-gas exposure equipment nonoperating

[3] Post-temperature inspection and acceptance per FTM

(3) Humidity exposure in a two-gas atmosphere approximately 1 per day partial functional

[4] Post-humidity inspection and acceptance test per FTM

Figure 36. Apollo X G & N and Module Qualification Flowgram Test Sequence

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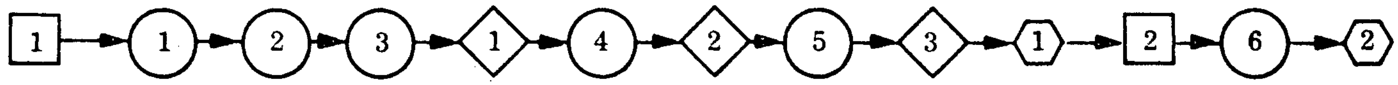

1	Complete initial inspection and acceptance test
1	Launch vibration exposure — equipment nonoperating except for normal heat
2	Entry vibration exposure — equipment operating
3	Abort vibration exposure — equipment nonoperating
1	Post-vibration acceptance test
4	Entry acceleration exposure — equipment nonoperating except for normal heat
2	Post-acceleration acceptance test
5	Flight shock exposure — equipment operating
3	Post-flight shock acceptance test
1	Complete visual mechanical including inside IMU
2	Complete inspection and acceptance test
6	Earth landing shock exposure
2	Post-earth landing shock visual mechanical inspection

Figure 37. Apollo X Spare IMU Functional and Environmental Qualification Test Flowgram

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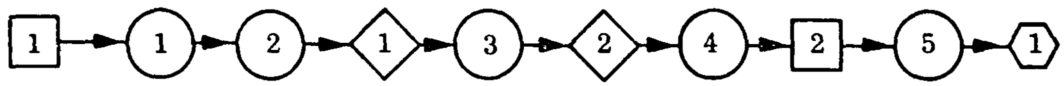

□ 1 Initial inspection and acceptance test per FTM

○ 1 Launch vibration exposure — equipment operating

○ 2 Abort vibration exposure — equipment operating

◇ 1 Post-vibration acceptance

○ 3 Entry acceleration — with exception equipment nonoperating due to centrifuge slip ring limitations

◇ 2 Post-acceleration acceptance test

○ 4 Flight shock exposure — equipment operating

□ 2 Post-flight shock acceptance test — complete per FTM

○ 5 Earth landing shock exposure

⬡ 1 Post-landing shock visual mechanical

Figure 38. PSA Functional and Environmental Qualification Test Flowgram

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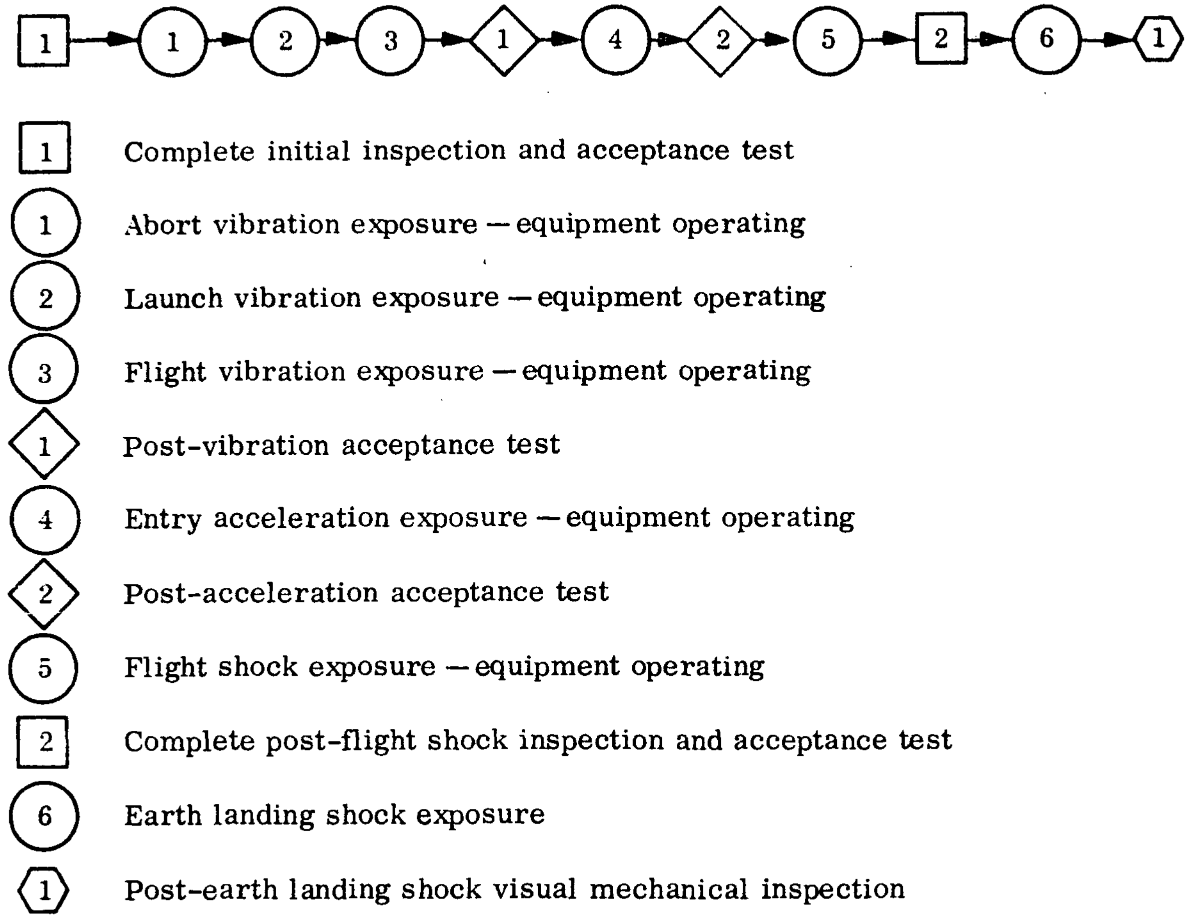

Figure 39. ECU Functional and Environmental Qualification Test Flowgram

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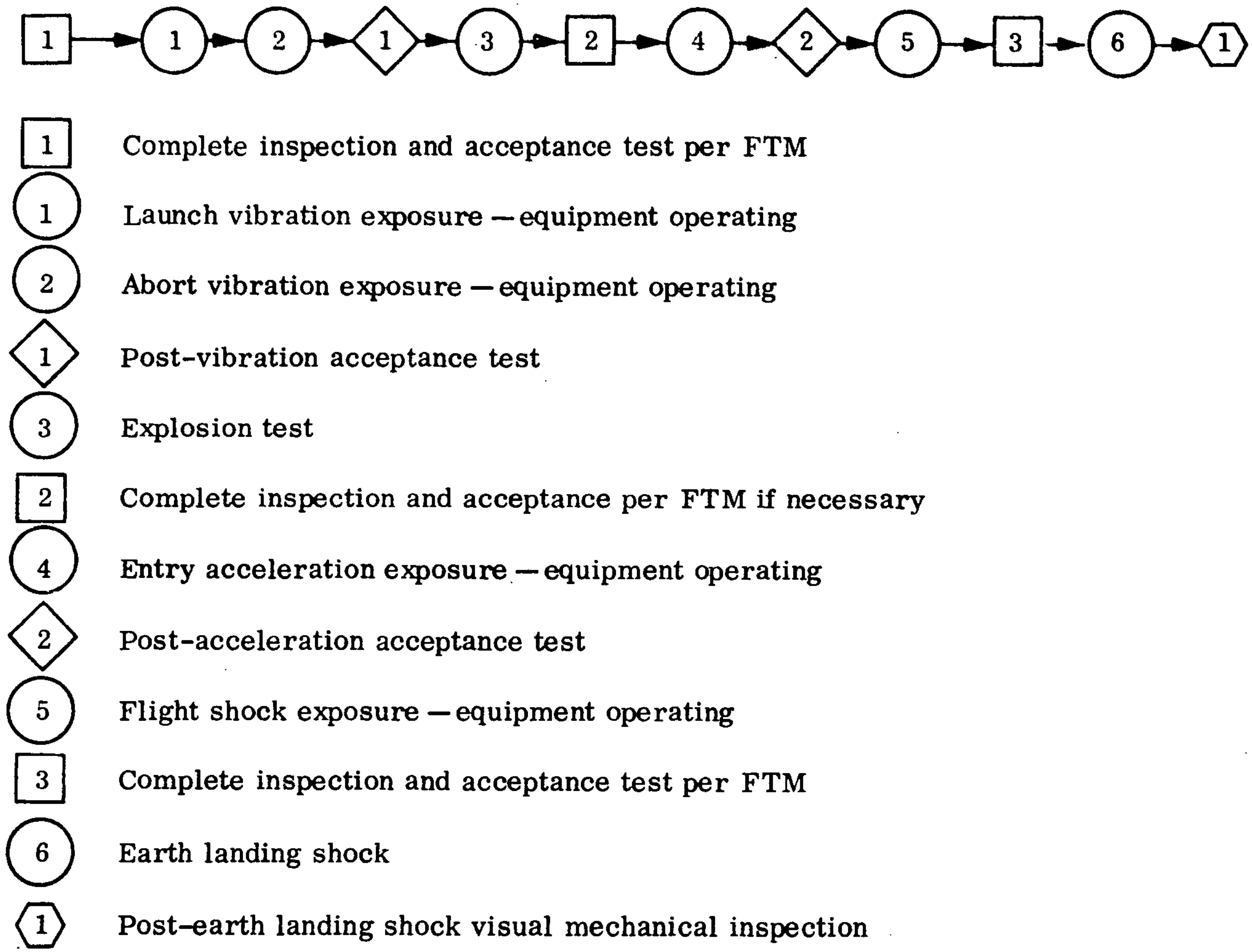

Figure 40. Apollo X D & C Group Functional and Environmental Qualification Test Flowgram

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General Equipment Performance

- Before and after environmental exposures, the G & N equipment will operate as specified by applicable FTM and JDC.
- During the environmental exposure those parameters and signals monitored will be within the JDC tolerance except where requirements are specifically modified by engineering analysis to account for instrumentation limitations.
- No servo loop will exhibit any tendency to oscillate or lose control.
- During dynamic environments, all switches and relays will remain in their intended position and not produce intermittents.
- The dynamic environmental exposures will not destroy or impair the ability of the G & N equipment to retain repeatable alignments, gimbal drift rates, gimbal transient responses, and inertial component characteristics and coefficients. The degree of repeatability will be determined by engineering analysis.
- During all qualification testing, the G & N equipment will operate to keep the temperature of inertial components between 120° and 140° F.

Specific Equipment Performance and Test Configurations

Module Performance. Before and after environmental exposures, the modules will operate as specified by their applicable FTM's. However, during the exposure, signals monitored will be within the FTM or engineering-specified tolerances.

Module Test Configuration. Each module will be operating or nonoperating during the exposure, as specified. Upon completion of the environmental exposure, an abbreviated functional test and visual inspection will be conducted before the module is removed from the test area for complete inspection and acceptance.

G & N System Performance. Before and after test exposures, the G & N will satisfactorily complete:

- Standby Power-On Test,
- Operate Power-On Test,
- Failure Indicating Circuitry Test,
- Temperature Control Test,
- G & N System Power Supplies Test,
- Guidance Computer Operational Test,

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- Optics Positional Accuracy Test (modified),
- Guidance Computer Mode Control,
- Frequency and Step Response Test (modified),
- IRIG Scale Factor and Gimbal Torque Test,
- PIPA Scale Factor and Bias Test,
- IRIG Coefficient Determination Test and Simulated Space Fine Align,
- Gyrocompassing Test.

During the exposure, as many tests as practical will be completed from the above list.

G & N Test Configuration. The Apollo X G & N will be installed in the applicable test chamber, with the optics aligned to a precision heading. Exercise of the G & N equipment will be accomplished by remote means to conduct the testing during the test exposures. This will be accomplished by control with a remote control display keyboard, operation through the computer test set, and remote control switch actuators.

Qualification Data Requirements

Environmental Data

- Temperature — chamber walls
- Pressure
- Coolant temperatures and flow rates
- Two-gas atmosphere composition — 50-percent oxygen and 50-percent nitrogen
- Humidity
- Random vibration inputs
- Acceleration inputs

G & N Equipment Signal List

The signals listed in Table 8 do not necessarily constitute the normally available signals, but are only a preliminary estimate of qualification testing monitored signals at various times during the test cycle.

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Table 8. G & N Equipment Signal List

Data List

+28 vdc Standby
+28 vdc Optics and IMU Operate
+120 vdc
+32 vdc
+20 vdc
-20 vdc
-28 vdc
28 vac, 3200 cps
4 vac, 3200 cps
2 vac, 3200 cps
28 vac, 2-phase, 800 cps Wheel and Blower Voltages
28 v, 1-percent, 800 cps
28 v, 5-percent, 800 cps

PIPA Loop

X, Y, Z Velocity Increment Pulses to Guidance Computer
X, Y, Z Torquing Currents
X, Y, Z Precision Voltage References
X, Y, Z Scale Factor Voltages
PIPA Average Temperature
X, Y, Z G/S Output
X, Y, Z Preamplifier Output

Stabilization Loops

X, Y, Z Gyro Preamplifier Outputs
X, Y, Z Gyro Error Resolver Outputs
IG, MG, OG Torque Motor Currents
IG, MG, OG Coarse Align Signal from ECU's
IG, MG, OG 1× and 16× Resolver Outputs to ECU
X, Y, Z $\Delta\theta$ Pulses from Guidance Computer
Common Gyro PVR
X, Y, Z Gyro Torquing Current
Gyro Average Temperature

ECU

4 vdc
$\Delta\theta$ Pulses from Guidance Computer (2 lines)
$\Delta\theta$ Pulses to Guidance Computer (2 lines)
Coarse Align Signals to Gimbal Servo Amplifier
Signal to FDAI
SIVB Interface
Guidance Computer ECU Enable
Guidance Computer Coarse Align Enable
Guidance Computer Error Angle Counter Enable

Optics Loops

Motor Drive Amplifier Outputs (4)
Tachometer Outputs (4)
X and Y Star Tracker Servo Outputs (2)
Photometer Preamplifier Outputs (1)
Photometer Output (1)
Trunnion 2× Resolver Signals
Shaft and Trunnion Command Signals
Star Presence to Computer
Hand Controller Outputs
Optics Moding Switches Outputs
Computing Resolvers Outputs

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Data Handling

Signals monitored will be recorded by one of the following:

- Photoelectric recorders,
- Pen and ink recorders,
- Magnetic tape recorders,
- Paper printouts from data processing equipment,
- Logbooks and data sheets.

The photoelectric, pen and ink, and written records will be reviewed, analyzed, and summarized by Engineering. Magnetic tapes will be processed, and some signals played back through G & N analog loop simulations to obtain gimbal and stable member transmissibilities. All random vibration inputs to equipment will be processed through appropriate filters and graphical presentations prepared.

Qualification Equipment Required

- Test Specimens — Class A manufacturing status Apollo X configuration:
  - One Apollo X G & N system less onboard spares, including a Guidance Computer and two display keyboards,
  - One each of every type module in the PSA (not including ECU modules),
  - One Apollo X spare IMU,
  - One Apollo X PSA with all modules installed,
  - One Apollo X redundant ECU package,
  - One Apollo X D & C Group.
- Support Equipment
  - One Full Universal Test station GSE complement
- Instrumentation Requirements
  - Thermocouple temperature recorders,
  - Chamber pressure gages,
  - Wet and dry bulb temperature recorders,

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- Mass spectrometer,
- Magnetic tape data acquisition equipment,
- Random noise analyzer,
- Signal conditioning circuitry,
- X-Y plotting equipment,
- CEC recorders and amplifiers,
- Sanborn recorders and amplifiers.

- Special Fixturing and Cabling:
  - Environmental GSE-PSA simulated junction box,
  - Remote switch actuators,
  - Vibration and shock holding fixtures (4),
  - Cabling to mate with temperature-altitude feedthrough connectors and test specimens and GSE.
- Facility Requirements:
  - Temperature-vacuum chamber — NRC space simulator,
  - Temperature-vacuum chamber with simulated star and solar sources,
  - Explosion chamber,
  - Vibration exciter,
  - Shock exciter,
  - Approximately 50 × 50 feet of test area,
  - Gas handling facilities.

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V. DEVELOPMENT PLANNING

OBJECTIVE

This section presents a brief and comprehensive description of the proposed Apollo X Program. This effort includes design determination, design support, manufacture, test, and field support of the Apollo X G & N hardware as described and implied in the technical portion of this study.

A major objective of the work effort will be to utilize as much as possible, without impairment of mission objectives and noninterferences with the Lunar Apollo Program, the design already completed for use in the Lunar Apollo Program. The major stress on design changes will be to increase and sustain reliability of intended Apollo X missions.

SCOPE

AC Spark Plug will determine the Apollo X hardware design, and will complete the development of the equipment to be defined during the Program Definition Phase of the project. Continued support will be given to the items of the Apollo X Program common to the Lunar Apollo Program.

The contractor will procure, fabricate, and deliver hardware to be defined by negotiated work statements. Required testing during the various stages of manufacturing will be accomplished.

Due to the developmental nature and the preliminary planning stage of the Apollo X Program, the schedules and general tasks presented constitute projections and are considered to be beyond the control of AC Spark Plug. Specific control and definition is expected to be established via a firm proposal during the Program Definition Phase.

The above activities will be supported by such groups as Management, Reliability and Quality Control, Engineering, Documentation, and Field Operations. The general scope of activities for these groups follows.

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MANAGEMENT

The Management function will include such items as program technical control, issuance of proper directives, utilization of PERT and hardware delivery schedule controls, handling general program financial control, and maintaining liaison with NASA, NAA, and other agencies and subcontractors.

AC Spark Plug will provide and control documentation as specified in anticipated negotiated work statements as a portion of the overall management effort.

RELIABILITY AND QUALITY CONTROL

Reliability and Quality Control have as a basic responsibility the continuation of reliability and quality efforts already established during the Lunar Apollo Program. These are in general conformance with NASA Reliability Publication NPC 250-1 (July 1963) and MIT/IL Reliability Plan R-349 (Revision B). The quality control program will be in general conformance with NASA Quality Assurance Publication NPC 200-2 (April 1962) and MIT/IL Quality Assurance Plan R-396 (Revision A).

An additional responsibility is to adapt the above techniques and guideline documents to unique areas of the Apollo X Program.

ENGINEERING

Engineering will undertake as its portion of the overall program function such tasks as follow.

- Sufficient and necessary design analysis of the Lunar Apollo equipment to enable a firm and detail determination of common design usage with Apollo X configuration.
- Complete design analysis and verification of performance of the Apollo X equipment.
- Detail and sustaining design effort comprising engineering and technical support required to enable the selection of engineering models (assemblies, parts, and components) that will meet the requirements of the design analysis and lend themselves to production. This effort includes engineering support to production and packaging design.
- Complete performance, procurement, assembly and test, and operating procedures for the Apollo X G & N equipment.
- Performance of system tests and checkouts, development test support, development of necessary test setups, and assembly and test of the complete Apollo X G & N equipment.

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FIELD OPERATIONS

In addition to the tasks specified above, AC Spark Plug will conduct a Field Operations Program for the Apollo X G & N equipment. This program will provide supervision, administration, and planning for field site activities, support activities, spare parts, associated test equipment, as well as site operation, personnel for the operation and support of the contractor-furnished equipment from factory acceptance through post-flight analysis.

TEST SCHEDULES AND MILESTONES

To accomplish the general objectives outlined on an integrated but noninterference basis with the Lunar Apollo Program, the Apollo X major milestones and the Apollo X integrated equipment and test schedules, presented in Figures 41 and 42, were developed.

These schedules and milestones are set forth as an integral effort on unique Apollo X hardware configurations and basic Apollo configurations. Alteration of delivery configurations would required re-appraisal of the milestones and the schedules.

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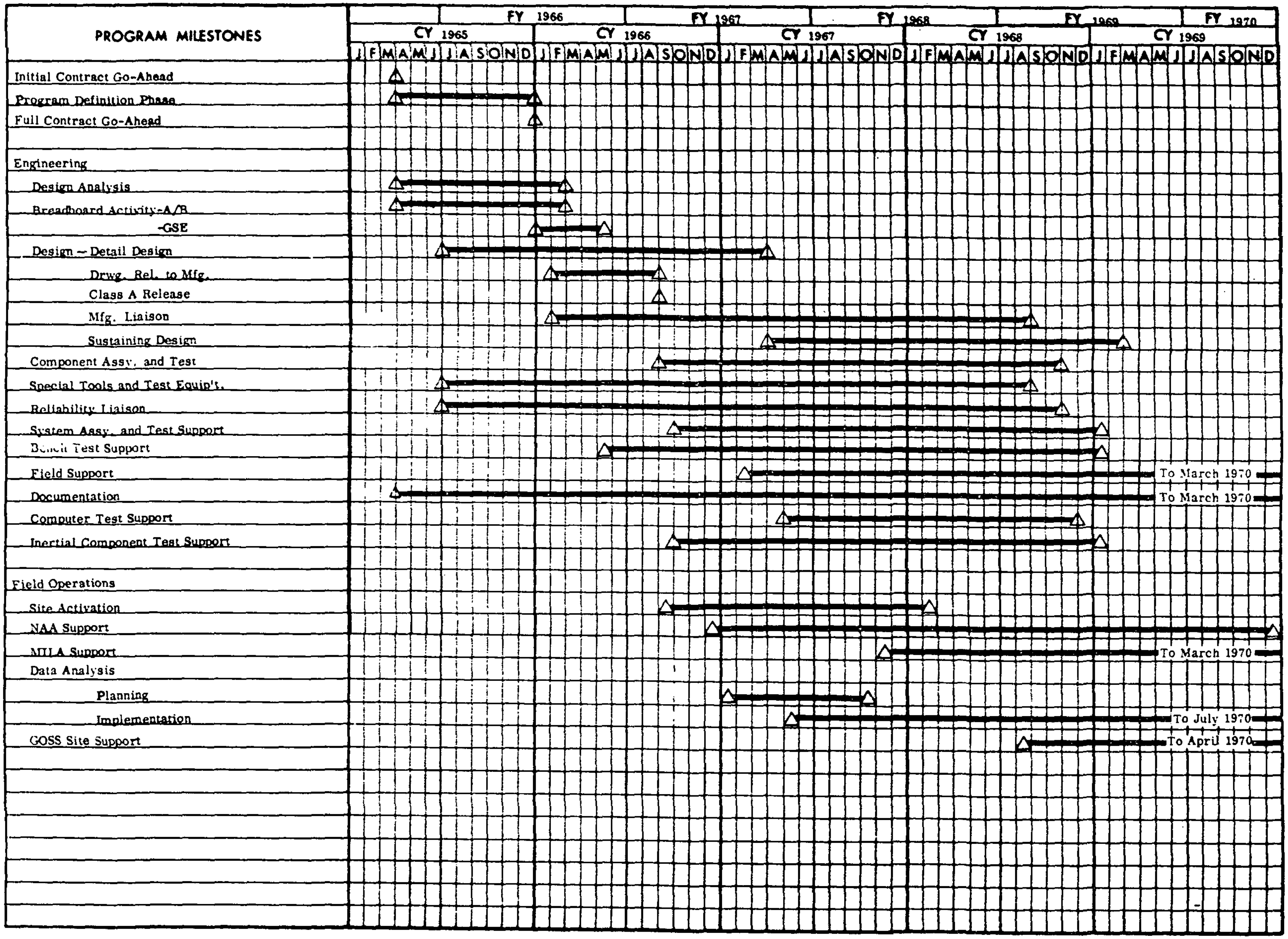

Figure 44. Apollo X — Major Milestones

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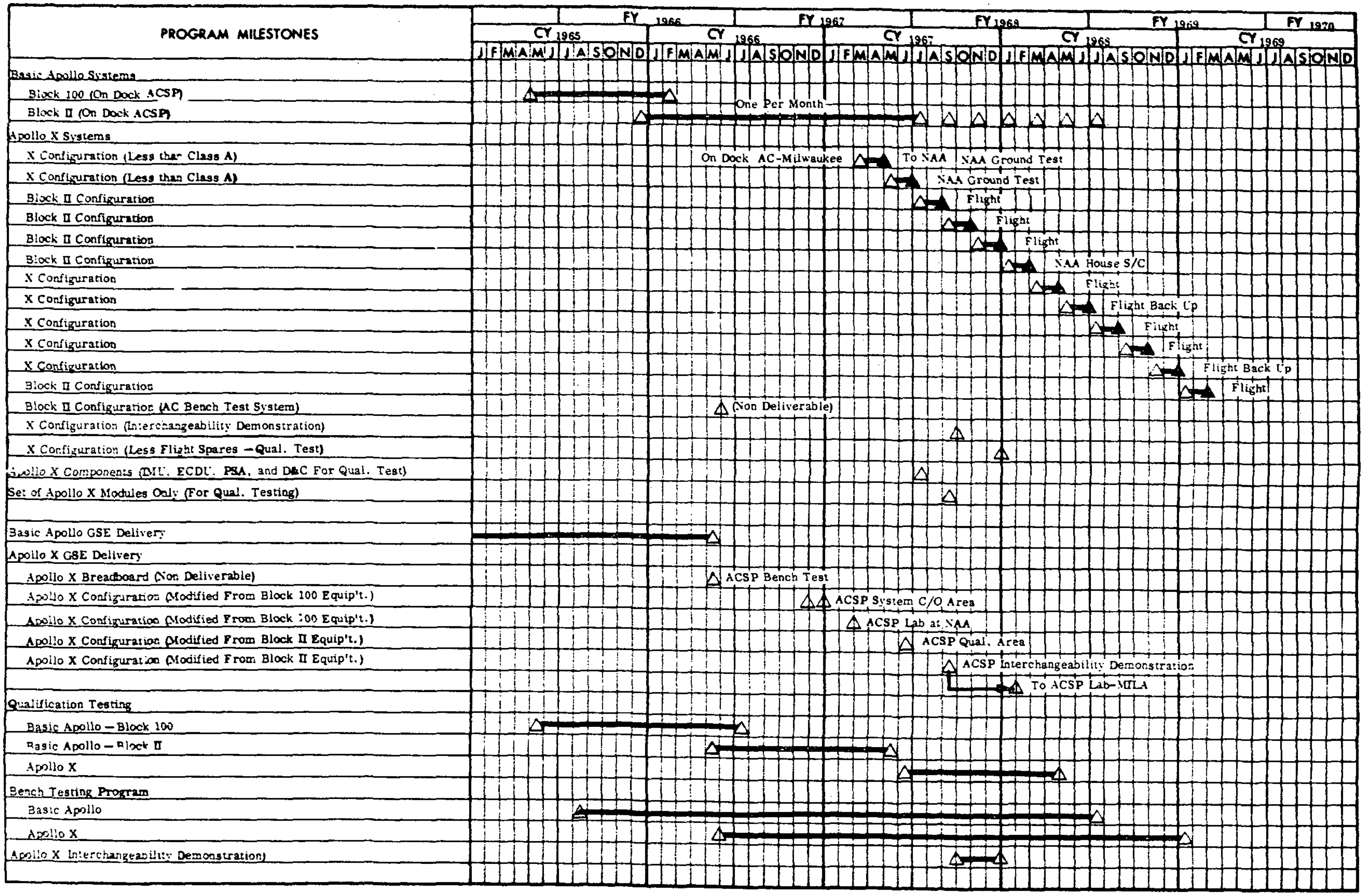

Figure 45. Apollo X — Integrated Equipment and Test Schedules

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VI. CONCLUSIONS AND RECOMMENDATIONS

CONCLUSIONS

The Apollo Block II G & N System can meet all the LPO mission performance requirements with the addition of the recommended modifications, spares, and redundancy.

The LPO measurement uncertainty in Altitude after one orbit is well within the orbital altitude accuracy requirements of 1.1 nm using either ground tracking, star occultation, or landmark tracking.

The G & N System can provide a reference to maintain the spacecraft attitude to within 0.125 degree of local vertical and the spacecraft rate to within 0.02 degree/sec. of the desired rate about all axes.

The G & N System, with the recommended modifications can meet an overall reliability of 0.9975 for the LPO mission with 91 pounds of spares and redundancy. A reliability of 0.9965 can be attained with 65 pounds of spares only. Further improvements can be expected after completion of those areas where further engineering development is necessary.

RECOMMENDATIONS

When Block II design is completed, perform a complete parts stress analysis based on circuit application of parts.

Implement life testing of Block II mechanical and electromechanical components under conditions similar to their Apollo X applications to determine life and to obtain realistic stress factor ratings as recommended in the test plans.

Update the Block II failure effects analysis, presently under study, to the Apollo X configuration to determine which part and module failures can cause a G & N System failure in the mission.

Define alternate G & N capabilities for performing required functions after the failure effects analysis has shown that a module failure can produce an adverse G & N effect.

Investigate the effects on Apollo X system reliability of repeated on-off power switching .

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VII. APPENDIX

SUPPORTING DATA

Table A.1 is a detailed Apollo X G & N Time Line for the lunar polar orbit mapping mission, earth polar orbit mapping mission, and the low-inclination earth orbit mission phases. This table is a further breakdown of Table 1 of this report.

Table A.2 lists the component and subassembly reliability failure rates for the Apollo Block II G & N. These values were used to compute the probability of success values given in the Technical Analysis portion of this report. An asterisk (*) is used to denote the main contributors to the failure rate, and a dagger (†) is used to denote the secondary contributors.

There are a limited number of built-in alternates for performing the required G & N functions as shown in Figures A.1 through A.11. In many instances, the same modules are used, but in alternate modes. A detailed failure effects analysis would indicate whether an alternate approach is available after a module failure has occurred and what the resultant improvement in reliability would be.

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Table A.1. Apollo X—G & N Time Line

Time (minutes)	Event	Mission Phase	Inertial Subsystem	Optical Subsystem	Guidance Computer
		PROFILE I			
0-12	Launch into 100 nm Parking Orbit Guidance Monitor	Boost S-V			X
12-20					
20-25	Alignment Update	Earth Orbit		X	X
25-175					
175-180	Alignment Update			X	X
180-192					
192-197	Translunar Inject Guidance Monitor	Translunar Inject			X
197-227					
227-257	Dock CSM to Lab				X
257-615					
615-630	Alignment			X	X
630-647					
647	Mid-Course Correction				X
647-3215					
3215-3230	Alignment			X	X
3230-3247					
3247	Mid-Course Correction				X
3247-4340		Translunar			
4340-4355	Alignment			X	X
4355-4442					
4442-4447	Alignment Update			X	X
4447-4457					
4457	Mid-Course Correction				X
4457-4500					
4500-4505	Alignment Update			X	X
4505-4517					
4517-4521	Deboost to Lunar Orbit				X
4521-4641	10-Star Occultation Measurements	First Lunar Orbit			X
4641-4646	Alignment Update	Second Orbit		X	X
4646-4656					
4656-4657	Orbit Correction				X
4657-4761	9-Star Occultation Measurement				X
4761-4766	Alignment Update	Third Orbit		X	X
4766-4776					
4776-4777	Orbit Correction				X
4777-4870	8-Star Occultation Measurement				X
4870-4875	Alignment Update			X	X
4875-4881					
4881-4941	Local Vertical Map	One-Half Orbit			X
4941-5226					
5226-5241	Alignment			X	X
5241-5256					
5256-5286	Local Vertical Map	One-Fourth Orbit			X
5286-5571					
5571-5586	Alignment			X	X
5586-5601					
5601-5661	Local Vertical Map	One-Half Orbit			X
5661-5961					
5946-5961	Alignment			X	X
5961-5976					
5976-6006	Local Vertical Map	One-Fourth Orbit			X
6006-6291					
6291-44,166	Repeat above mapping cycle for a total of 55-half orbit maps and 55-quarter orbit maps except a full alignment will precede the first local vertical map of each cycle.				

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Table A.1. Apollo X—G & N Time Line (cont)

Time (minutes)	Event	Mission Phase	Inertial Subsystem	Optical Subsystem	Guidance Computer
PROFILE I (cont)					
44,166-44,466					
44,451-44,466	Alignment			X	X
44,466-44,481					
44,481-44,484	Injection	Transearth Inject			X
44,484-44,526					
44,526-44,531	Alignment Update			X	X
44,531-44,541					X
44,541-44,543	Trajectory Plane Change				
44,543-45,510					
45,510-45,525	Alignment			X	X
45,525-45,540					
45,540	Mid-Course Correction				X
45,540-47,110					
47,110-47,125	Alignment	Transearth		X	X
47,125-47,140					
47,140	Mid-Course Correction				X
47,140-48,670					
48,690-48,705	Alignment			X	X
48,705-48,720	------				
48,720-48,725	Alignment Update			X	X
48,725-48,740	------				
48,740	Mid-Course Correction				X
48,740-48,785	------				
48,785-48,790	Alignment Update			X	X
48,790-48,804	------				
48,804-48,822	Entry	Entry			X
PROFILE II					
0-10	Monitor Launch to 100 mm Parking Orbit	Boost S-V and SIVB			X
10-20		Parking Orbit			
20-27	SIVB Relight (392 seconds)				X
27-55	Transpose, Dock and Coast	Transfer Trajectory			X
55-60	Alignment Update			X	X
60-66	------				
66-68	T. V. C.	Earth Orbit Injection			X
68-128	Orbit Verification and Corrections	First Orbit			X
128-143	Alignment-Update			X	X
143-158	------				
158-218	Local Vertical	Second Orbit			X
218-503	------				
503-518	Alignment			X	X
518-533					
533-563	Local Vertical				X
563-848	------				
848-863	Alignment		X	X	X
863-878					
878-T	Recycle 158 to 878 as often as rel. allows		X		
T	T. V. C. (18 seconds)	Deboost			X
(T+14) - (T+35)	Entry T = 720C + 158; C = Total Cycles	Entry			X

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Table A.1. Apollo X — G & N Time Line (concluded)

Time (minutes)	Event	Mission Phase	Inertial Subsystem	Optical Subsystem	Guidance Computer
		PROFILE III			
0-10	Monitor Launch into 80 mm Parking Orbit	Boost SIVB			X
10-20	------				
20-50	Dock CSM to Lab				X
50-85	------	80 mm			
85-90	Alignment Update	Orbit		X	X
90-100	------				
100	SM Light for 10 seconds				X
100-125					
125-130	Alignment Update	Orbit Transfers		X	X
130-140	------				
140	SM Light for 10 seconds				X
140-200	Orbit Verification and Corrections				X
200-31,370	------	200 mm Orbit			
31,370-31,430	G & N Checkout		X	X	X
31,430-31,790	------				
31,790-31,805	Alignment			X	X
31,805-31,820	------				
31,820	SM Light	Orbit Correction			X
31,820-31,880	Orbit Verification				X
31,880-64,490	------				
64,490-64,550	G & N Checkout		X	X	X
64,550-64,910	------				
64,910-64,925	Alignment			X	X
64,925-64,940	------				
64,940	SM Light for 18 seconds	Deboost			X
64,940-64,954	------				
64,954-64,975	Entry	Entry			X

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Table A. 2. Block II Module Failure Rates

Assembly	Failures per 10^6 Hours	Usage
INERTIAL MEASUREMENT UNIT		
Electrical Connector	0.02	16
Slip Ring*	3.0	6
Temperature Alarm	0.1	1
Temperature Control	0.25	1
Ducosyn Transformer	0.25	1
Thermostat	0.06	4
Padding Resistors	0.05	3
Heaters	0.1	4
Resolver*	5.0	4
Bearing*	0.6	6
Torque Motor*	5.0	3
16 PIPA Preamplifier*	1.55	3
16 PIP*	4.2	3
PIP Suspension*	2.6	3
IRIG and Preamplifier*	12.0	3
IRIG Suspension*	2.0	3
Precision Resolver Adjustment	1.44	1
Blowers and Controls	2.4	2
Water Connector	0.2	2
POWER AND SERVO ASSEMBLY		
PIPA Calibration*	2.2	3
Gyro dc Calibration	6.6	1
Pulse Torque Power Supply	8.8	1
AC Differential Amplifier and Integrator	7.3	3
Gimbal Servo and Alignment Amplifier*	6.0	3
Fail Indicator and Warning*	7.3	1
3200 cps AAC Filter†	2.8	1
3200 cps, 1-percent Amplifier*	6.5	1
-28 vdc Power Supply†	3.9	1
Sextant Motor Drive Amplifier	2.2	2
Scanning Telescope Motor Drive Amplifier	2.2	2
800 cps Optics Comparator	0.8	1
Servo Integrate Relay	12.0	1
Anti-Creep	3.6	1
Scanning Telescope Trunnion Transformer and Relay	6.2	1
POWER AND SERVO ASSEMBLY (cont)		
Tracker Mode Relay	12.0	1
Sextant Mode Relay	6.0	1
DC Differential Amplifier and Precision Voltage Reference*	2.7	4
Binary Current Switch*	2.6	4
G & N Subsystem Filter	0.2	3
800 cps AAC Filter†	2.7	2
800 cps, 1-percent Amplifier†	3.5	2
800cps, 5-percent Amplifier*	3.7	3
ELECTRONIC COUPLING UNIT		
Quadrant Selector*	17.7	5
Cos (θ - ψ) Generator*	17.1	5
Coarse System Electronics*	8.2	5
Main Sum Amplifier and Quadrature*	11.8	5
Digital-to-Analog Converter*	4.5	5
Gimbal Logic*	5.9	5
Error Angle Count and Logic	6.0	5
Clock and Mode Logic†	5.6	1
4-Volt Power Supply†	1.7	1
DISPLAYS AND CONTROLS		
Control Assembly	4.26	1
G & N Indicator Control Panel		
Attitude Impulse Switch	6.0	7
Sextant Hand Controller	12.4	1
Assembly Parts	13.24	1
OPTICS HEAD		
Resolver*	5.0	5
Motor Generator*	5.0	4
Electrical Connector	0.2	10
Bearing	0.6	51
Transformer	0.2	1
Resistor	0.1	2

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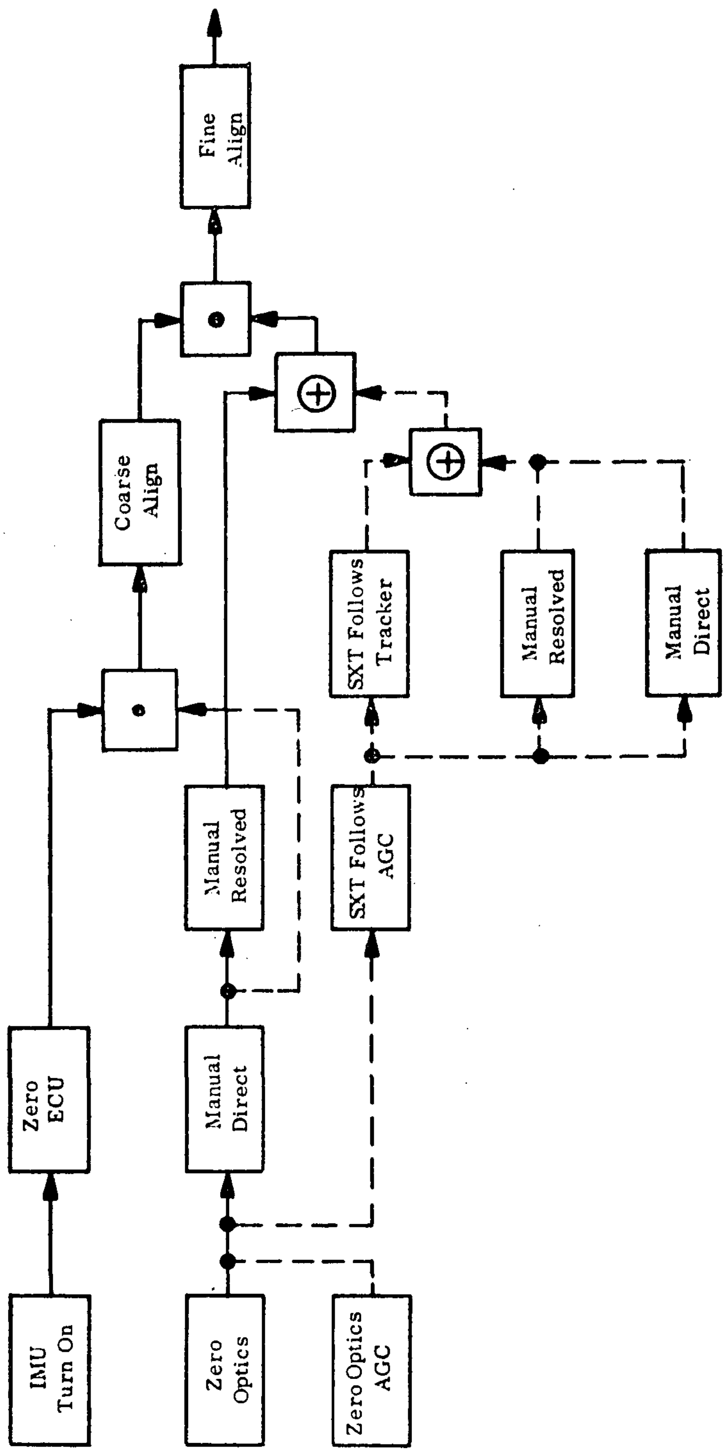

Figure A.1. IMU Alignment

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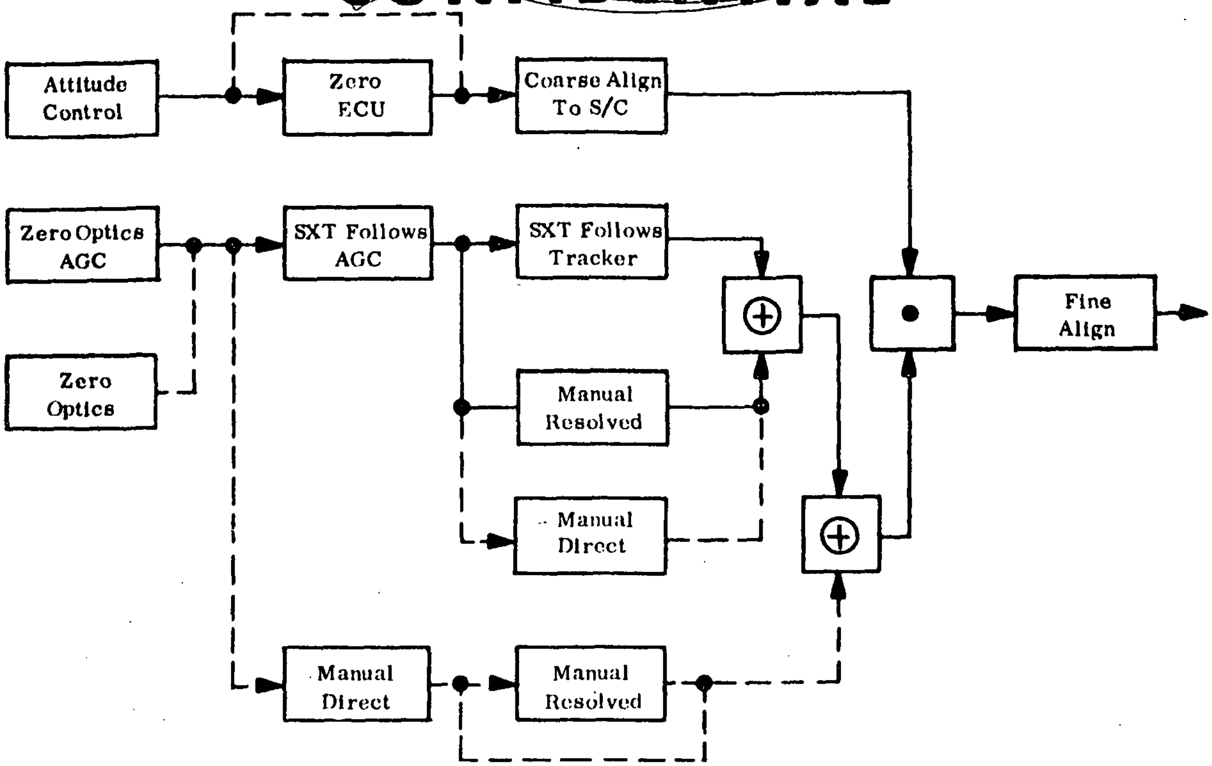

Figure A.2. IMU Alignment — Update

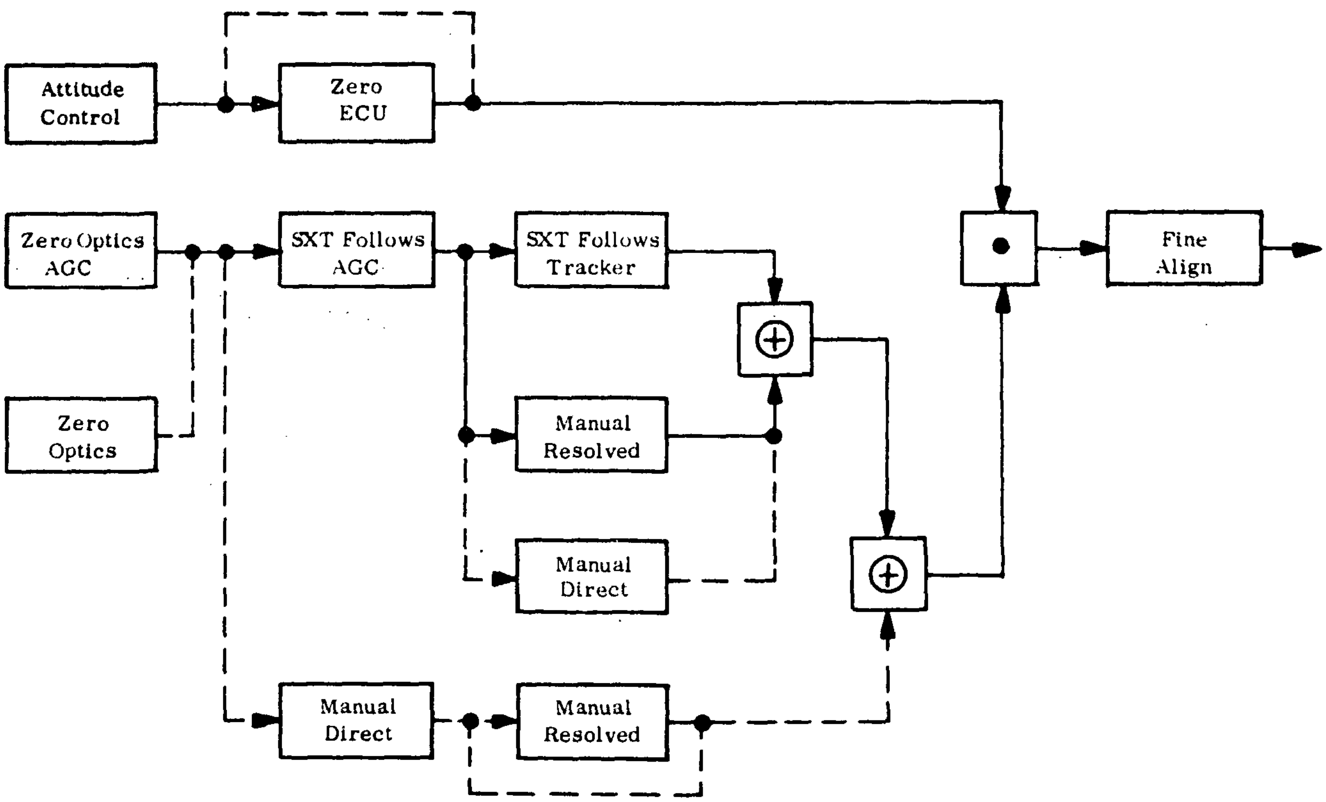

Figure A. 3. IMU Alignment — Update

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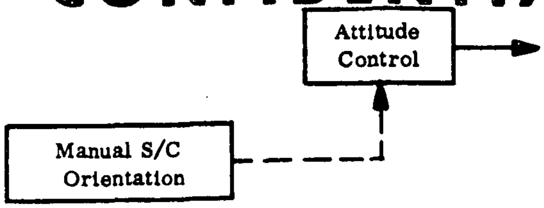

Figure A.4. Thrust Vector Control

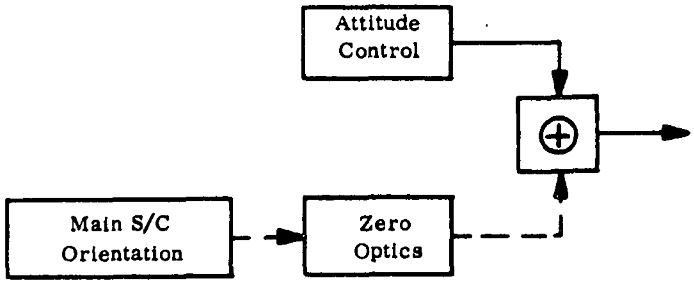

Figure A.5. Midcourse Correction — Orbit Correction

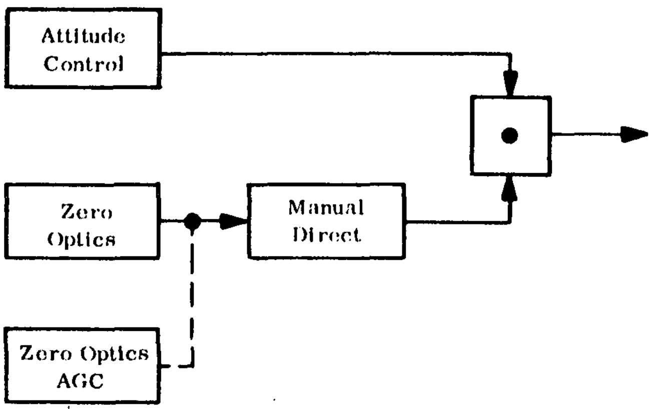

Figure A.6. Landmark Tracking

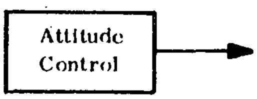

Figure A.7. Attitude Control

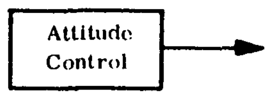

Figure A.8. Boost Monitor

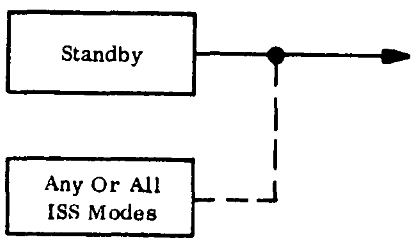

Figure A.9. Nothing

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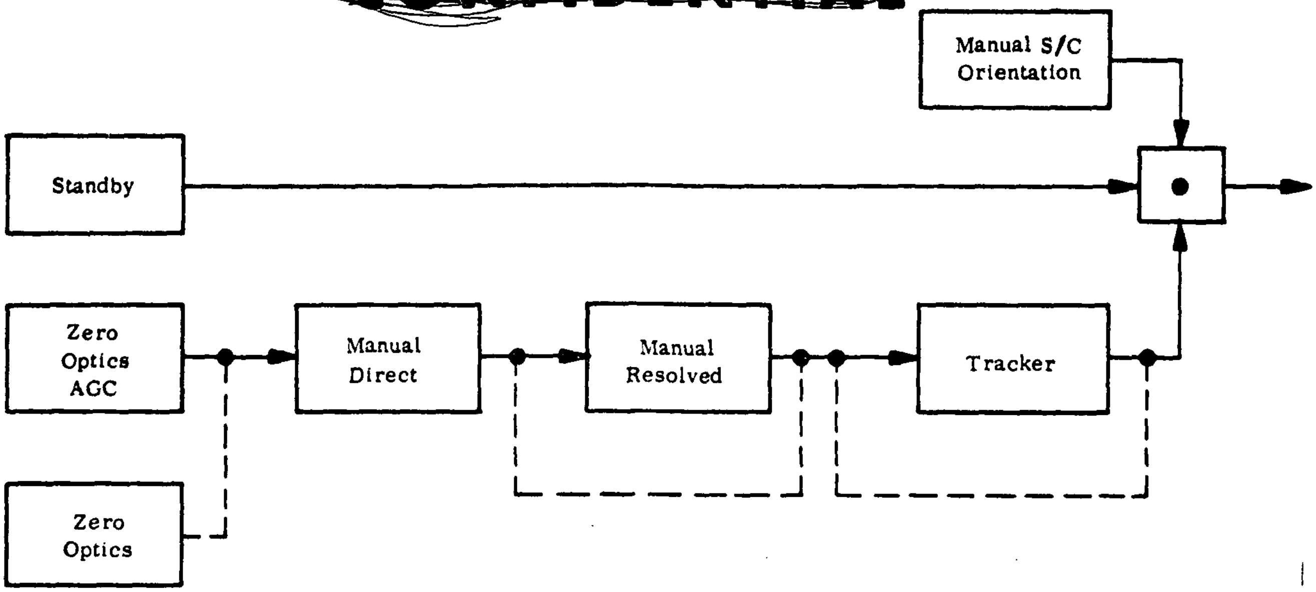

Figure A-10. Star-Landmark Measurement

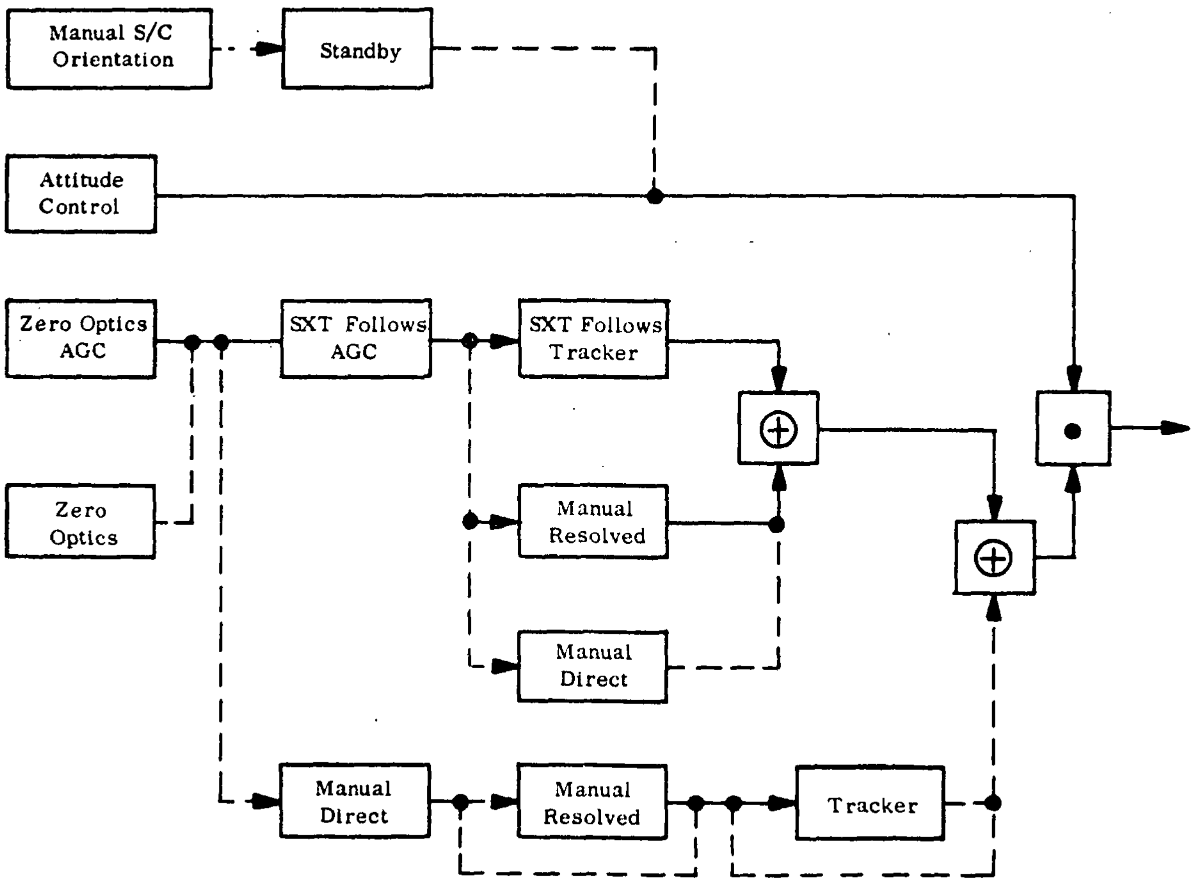

Figure A. 11. Star-Landmark Measurement

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