MASSACHUSETTS INSTITUTE OF TECHNOLOGY

APOLLO

GUIDANCE, NAVIGATION AND CONTROL

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R-700

MIT's ROLE IN PROJECT APOLLO

FINAL REPORT ON CONTRACTS
NAS 9-153 AND NAS 9-4065

VOLUME I

PROJECT MANAGEMENT
SYSTEMS DEVELOPMENT
ABSTRACTS AND BIBLIOGRAPHY

edited by
James A. Hand

OCTOBER 1971

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FOREWORD

The title of these volumes, "MIT's Role in Project Apollo", provides but a modest hint of the enormous range of accomplishments by the staff of this Laboratory on behalf of the Apollo program. Man's rush into spaceflight during the 1960s demanded fertile imagination, bold pragmatism, and creative extensions of existing technologies in a myriad of fields, The achievements in guidance and control for space navigation, however, are second to none for their critical importance in the success of this nation's manned lunar-landing program, for while powerful space vehiclesand rockets provide the environment and thrust necessary for space flight, they are intrinsically incapable of controlling or guiding themselves on a mission as complicated and sophisticated as Apollo. The great achievement of this Laboratory was to supply the design for the primary hardware and software necessary to solve the Apollo guidance, navigation and control problem. It is to the credit of the entire team that this hardware and software have performed so dependably throughout the Apollo program.

The quantum leap in technology nurtured by the Apollo program has been and should continue to be of immensely significant benefit to this country--socially, economically and in terms of its national esteem. It is the responsibility of al! those who contributed to the proud achievements of Apollo to convince their countrymenof the directions thisnation ought to follow in implementing these newly gained -and hard fought for-advances.

C. Stark Draper, President
Charles Stark Draper Laboratory

R-700

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ABSTRACT

Seventy-six days after the President of the United States committed the nation to a manned lunar-landing program, the Charles Stark Draper (formerly Instrumentation) Laboratory of the Massachusetts Institute of Technology received the first major contract of the APOLLO program. The Laboratory was to design and implement the requisite hardware and software for the Guidance, Navigation and Control system of the APOLLO spacecraft, Chapter I of this volume of the Final Report discusses the Laboratory's management of the APOLLO project. Chapter II presents salient features in the development of the guidance, navigation and control system hardware. Appendix A contains abstracts of some research and engineering reports and theses prepared under Contracts NAS 9-153 and NAS 9-4065, and Appendix B is a bibliography of all such reports and theses prepared through June 1969.

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PREFACE

"I believe this nation should commit itself to achieving the goal before this decade is out of landing a man on the moon and returning him safely to earth." With these words, spoken on 25 May 1961, President John Fitzgerald Kennedy stated for all Americans the challenge of the APOLLO project. The Massachusetts Institute of Technology Instrumentation Laboratory* was selected to design and develop the hardware and software of the APOLLO Guidance, Navigation and Control system for safe and self-sufficient translunar flight, lunar landing, and return. This Finai Report describes that work -- a most demanding, innovative, and rewarding task.

This report presents the Draper Laboratory's efforts in Project APOLLO from the original contract award in mid-1961. through July 1969. The report is organized in five volumes:

VOLUME I: PROJECT MANAGEMENT AND SYSTEMS DEVELOPMENT

VOLUME II: OPTICAL, RADAR, AND CANDIDATE SUBSYSTEMS

VOLUME III: COMPUTER SUBSYSTEM

VOLUME IV: INERTIAL SUBSYSTEM

VOLUME V: THE SOFTWARE EFFORT

Volume I emphasizes what was done in terms of resource allocation and systems development and contains Appendices A and B. Volumes 11 through IV describe the hardware subsystems in detail, with emphasis on the final design configurations; Volume V fully treats the Laboratory's software effort. Appendix A presents abstracts of significant research and engineering reports and theses written under Contracts NAS 9-153 and NAS 9-4065. Appendix B is a bibliography of all such reports and theses prepared through June 1969. This date is also the cutoff for al! discussions within this report, except for APOLLO 11 -- the first manned lunar landing and return.

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*The Laboratory was renamed the Charles Stark Draper Laboratory in January 1970.

CHAPTER I

PROJECT MANAGEMENT

SECTION 1.0
OVERVIEW

This first chapter describes the APOLLO project management within the organizational framework of the Massachusetts Institute of Technology Instrumentation Laboratory.[4] The major observations which this chapter makes include the following:

1. The Laboratory and the APOLLO project operate in a unique academic, governmental and industrial environment.
2. The project operates within the framework of academic-rather than profit-motivated-objectives and policies; the entire Laboratory operation is conducted on a nonprofit basis.
3. The approach of augmenting Laboratory design groups with resident personnel from the industrial contractors provided a smooth transition from design by the Laboratory to manufacture by NASA contractors.
4. The project's focus evolved from system hardware development to system computer-program development.
5. Design-review and change-control boards were necessary and effective tools in the management of both the hardware and software development efforts.
6. The magnitude and complexity of the APOLLO project and the organization of the interfacing agencies required the establishment of a strong formal project organization.

The following sections discuss Laboratory objectives and policies, trends in staff levels, the fiscal background of the Laboratory's role in Project APOLLO, the original APOLLO organization, and changes which occurred during the period under discussion.

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* As explained in the Preface, this Laboratory was renamed the Charles Stark Draper Laboratory in Jaunary 1970; throughout this Final Keport, the use,of the present tense indicates facts current as of 30 June 1969.

SECTION 2.0
LABORATORY OBJECTIVES AND POLICIES

2.1 OBJECTIVES

The Laboratory's objectives and policies, which form the organizational framework for APOLLO, evolved under the direction of its founder, Dr. Charles Stark Draper.

The Instrumentation Laboratory has three basic objectives:

1. The advancement of education, particularly in the phases between formal academic courses and professional life
2. Contribution to knowledge and techniques of science, applied science and technology
3. Creation of operational components, subsystems and complete systems which can be produced by industry and with performance characteristics that represent pioneering capabilities for the nation.

2.2 POLICIES

Severai basic policies underlie all work undertaken by the Laboratory:

1. The Laboratory is an integral part of the academic structure of M.I.T., in general, and of the Department of Aeronautics and Astronautics, in particular.
2. Sponsored research projects are accepted only if they concern research and developments on the frontiers of applied science and technology in fields generally compatible with and important to progress in aeronautics and astronautics.
3. Projects must lie in fields in which the Laboratory possesses special capabilities as a result of past tasks successfully completed and the interests of ranking staff members. In addition, the Laboratory applies its expertise to new fields which potentially can benefit from the infusion of proven capability in related fields.

4. Work of a routine or "production-in-quantity" nature and activities directed toward refinement of design and manufacture that do not involve significant improvement in performance or size are accepted, but always in small quantities, and only when it is necessary to assist with testing or to stimulate industrial developments.

5. In general, Laboratory projects have been restricted to those dealing with the sensing, transmittal, processing and application of information for the purposes of complete operating systems. Special attention during the past two decades has been directed toward the generally neglected technology of dynamic geometry in stabilization, control, navigation and guidance of flightvehicles carrying out missions of accurate transportation to fixed or moving destinations.

6. Tasks of the scope and magnitude required to deal with complete subsystems and systems are selected for compatibility with the general pattern of pioneering research and development in the Laboratory.

7. Authority and responsibility are sought for each system project on terms that provide an environment of creative achievement for Laboratory staff and students.

8. The overriding policy is to work always for more effective use of this environment for an "internship" type of education which is aimed at the full realization of invididua! potential for professional careers that are contributions to society.

SECTION 3.0
LABORATORY STAFF

3.1 STAFFING LEVELS

The management of the Instrumentation Laboratory is concerned primarily with the allocation and direction of innovative technical personnel. Since this single resource is important for the entire Laboratory organization-and the APOLLO project in particular-the following paragraphs discuss overall staff characteristics,

Figure 3-l shows the growth of the Instrumentation Laboratory in numbers of personnel from 1946 to 1969. The figure depicts total numbers of Laboratory personnel, industrial-support residents, and subcontractors. Accelerated growth of the total personnel in the organization from 1961 is largely attributable to the APOLLO effort, which was contracted in that year.

Many of the industrial support personnel and subcontracted personnel shown in the figure have worked on the APOLLO project-notably those residents from such other APOLLO contractors as AC Electronics Division of General Motors Corporation, Raytheon Company, and Kollsman Instrument Corporation.

3.2 EDUCATION AT THE LABORATORY

Figure 3-2 shows that a large percentage of staff members hold M.I.T. degrees and that many staff members hold advanced degrees, as well. Continuation of formal education while working at the Laboratory contributes to the large number of advanced degrees. Undergraduate training of engineers, another educational focus of the Instrumentation Laboratory, is also illustrated. This is a major contribution of the Laboratory to the educational goals of M.I.T. and an important factor related to the technical level of performance for the APOLLO effort. Figure 3-3 shows the distribution of bachelor-degree disciplines among the Laboratory staff members and supervisors. A most important, but immeasurable, aspect of the Laboratory is the incentive for technical excellence and innovation given to individual staff members, industrial residents and students by the Laboratory's intimate academic ties and open working environment.

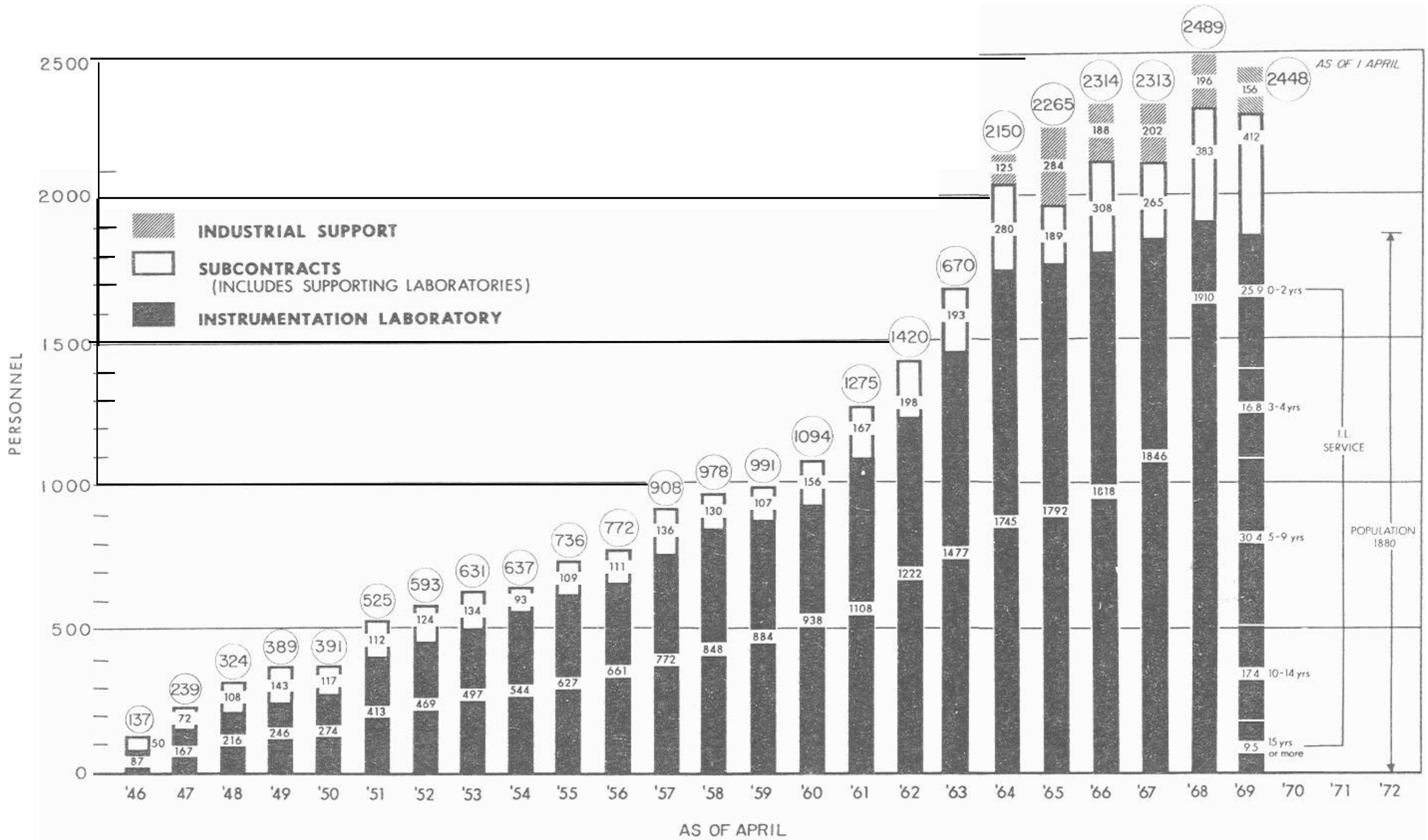


Fig. 3-1 Laboratory Personnel

BACKGROUND OF STAFF

HOLDING MIT DEGREES	28.6%
RATIO OF MASTERS TO DOCTORS	6.0
RATIO OF BACHELORS TO MASTERS	1.72

AVERAGE AGE – 35.7 YEARS

MILITARY SPONSORED VISITS BY ENGINEERS FROM INDUSTRY

NUMBER OF COMPANIES	SINCE 1/54	465
AVERAGE MAN-DAYS OF ENGINEERING VISITS	PER MONTH	2,726
MAN-DAYS OF ENGINEERING VISITS	SINCE 1/54	486,416

PRE-DEGREE TRAINING OF ENGINEERS

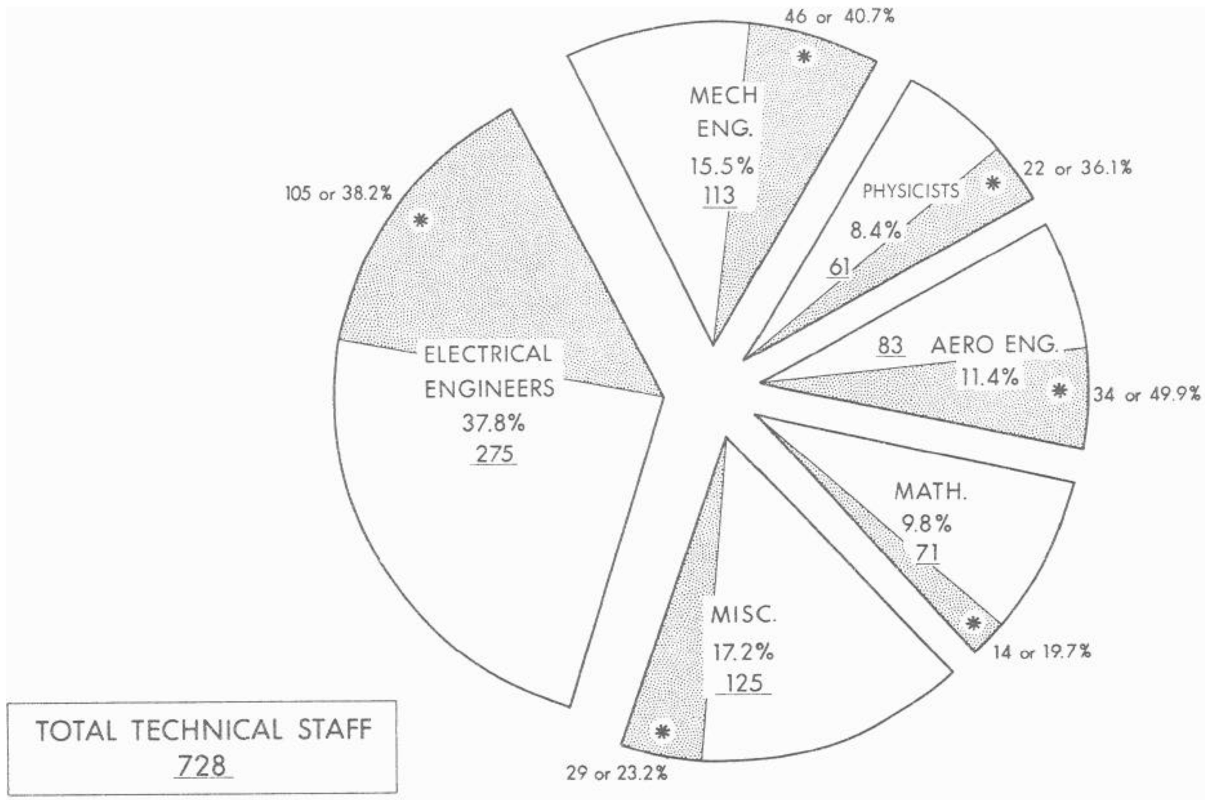
	SPRING	SUMMER	FALL
RESEARCH ASSISTANTS	23	11	18
STUDENT TECHNICIANS	76	200	74
STUDENT (COOPERATIVES)	76	(ALTERNATING EVERY 13 WEEKS TO 26 WEEKS)	

STAFF PERSONNEL TO INDUSTRY

YEAR	%	YEAR	%
1957	9.7	1963	5.7
58	11.8	64	5.5
59	17.3	65	4.7
60	10.3	66	11.6
61	6.4	67	12.6
62	5	68	12.8

Fig. 3-2 Staff Statistics

DISTRIBUTION BY BACHELOR DEGREE DISCIPLINES



* TECHNICAL SUPERVISORS (GROUP LEADERS & ABOVE)

Fig. 3-3 Technical Staff (Bachelor Degree

SECTION 4.0
FISCAL BACKGROUND

4.1 GENERAL FISCAL POLICY

Projects of the Instrumentation Laboratory traditionally are conducted on anonprofit basis. M.I.T. receives neither a fee nor a percentage profit for the Laboratory's efforts. Sponsoring agencies reimburse the Institute for the costs incurred in operating the Laboratory. As a national resource of talent and skill built up over the years at the request of sponsoring Federal agencies, M.I.T. makes the Laboratory available to the largest extent possible to help in the achievement of national goals. The benefits that accrue to M.I.T. rre the educational contributions the Laboratory makes to the undergraduate and graduate educational programs offered by the Institute.

4.2 FISCAL, BACKGROUND OF THE APOLLO PROJECT

The APOLLO project was negotiated for the effort required to perform the necessary research and development. From the inception of the project through 1966, the contract was negotiated on a yearly basis. Thereafter, a single contract covered the period from 1967 through mid-1970. In fiscal terms, the APOLLO effort is more than twice as large as any other project ever undertaken by the Instrumentation l-aboratcry. From 1962 on, APOLLO expenditure s and commitments have ranged from approximately one-third to one-half of the total Laboratory funding.

Figure 4-1 is a graph of the total MIT/IL expenditures and commitments on APOLLO through mid-1969. Quarterly and cumulative curves are given. All expenditure elements are included-salaries and wages, overhead, subcontracted work, materials and services, and travel. The quarterly breakdown of expenditures and commitments 'indicates generally the three phases of the contract to date: 1961 to 1965, when the primary concern was with systems hardware design and development; then a transitional phase into early 1967; and finally concentration of resources primarily on software development.

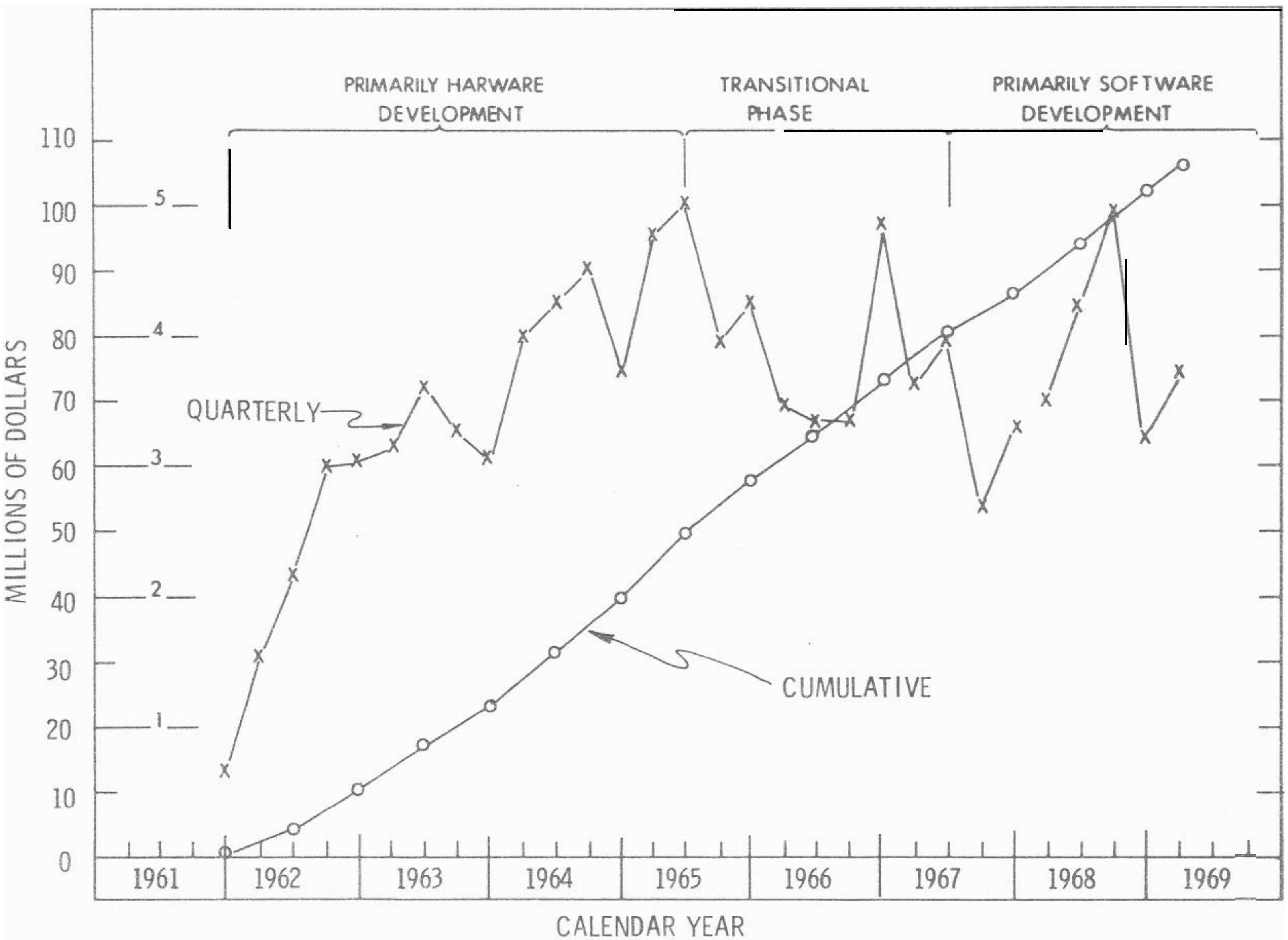


Fig. 4-1 MIT/IL and on APOLLO (total)

SECTION 5.0
APOLLO ORGANIZATION

5.1 BACKGROUND

Traditionally, the Laboratory has operated on a mission-oriented basis for the execution of its major projects, Rather than organize a very large group which might possess many kinds of technological expertise, the Laboratory assigns responsibilities for a large contract to smaller groups, each of which functions within a delimited technological area.

5.2 APOLLO PROGRAM ORGANIZATION

The evolution of the APOLLO organization at the Laboratory was closely reiated to the program phases and followed closely the evolution of the NASA APOLLO organization.

During the initial phase of the program, until June 1963, the APOLLO organization at the Laboratory consisted of a small team which served as the system focal point, integrating the ideas and design layouts from many groups in the Laboratory. These other groups, formed under previous ongoing programs, contained the expertise and experience which had to be applied to the development of the complex multidisciplinary system of APOLLO Guidance, Navigation and Control.

The Laboratory's responsibilities during this period included the design of the Primary Guidance, Navigation and Control System hardware, assistance to NASA in selecting contractors to build the equipment under M.I.T. direction, and discussions with NASA and the spacecraft contractors for the development of interface specifications. Also during this early period, lines of communication were established among Laboratory engineers and their peers at the spacecraft contractors for the interchange of data needed for design decisions; this informal communication network worked more or less successfully to supply data before formal release of data could be obtained. Laboratory engineers were deeply interested in obtaining fast and accurate interchange of up-to-date data to aid in the design of the PGNCS.

The NASA interface was changing during this period from the relatively informally organized Langley Study Group to the new and highly structured Manned Spacecraft Center in Houston,

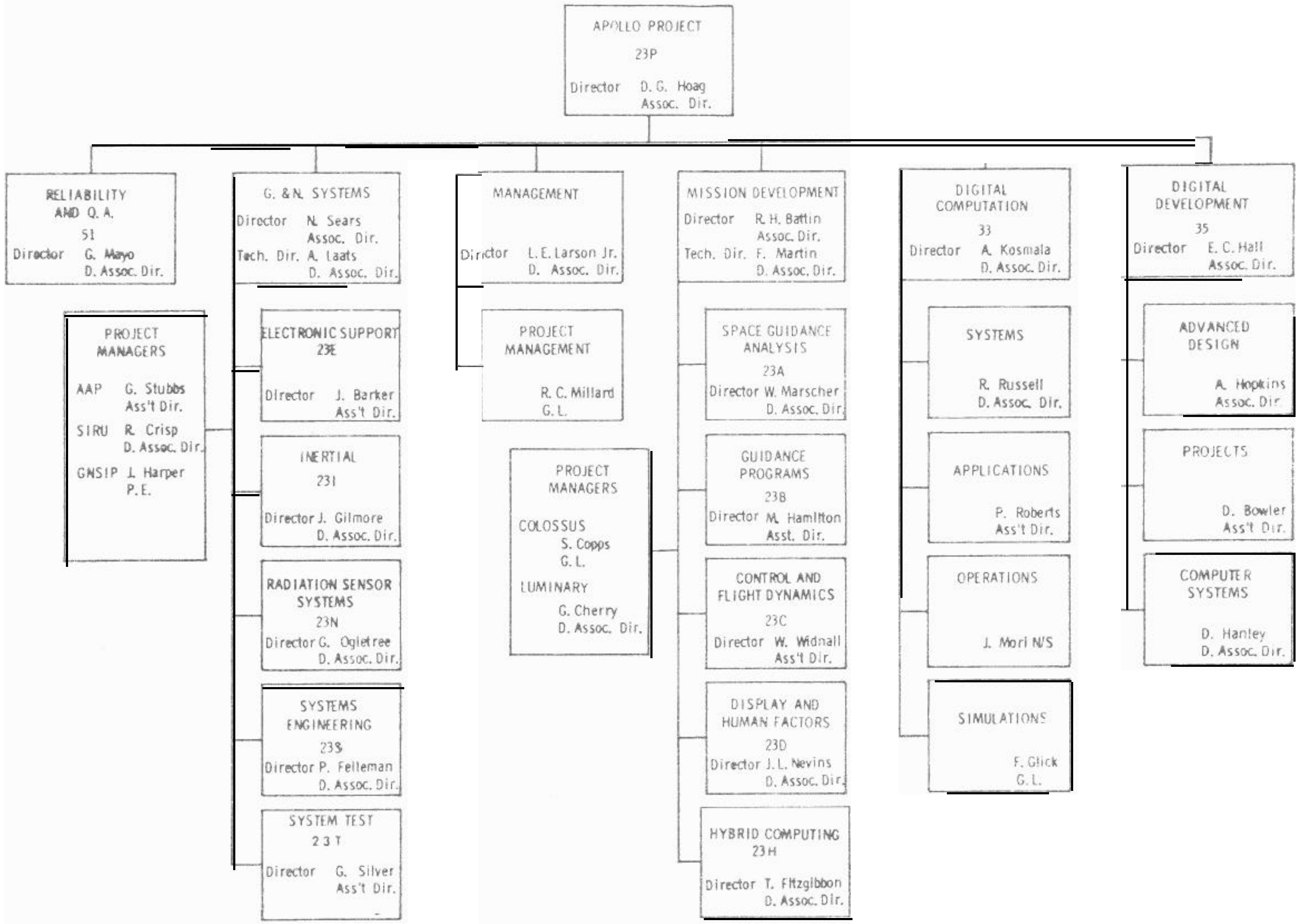


Fig. 5-1 APOLLO Project

Although none of the technical aspects of the APOLLO program presented problems new to the Laboratory, the very magnitude of the problems and the complexity of the system did require new solutions. Once the industrial contractors who would build the PGNCS under M.I.T. direction were chosen, and because of the maturation of the MSC organization, a more formal project-management organization was required at MIT/IL.

With the selection of the industrial contractors, M.I.T. instituted the use of resident engineers to work at the Laboratory, assisting in the design production and thereby acquiring the first-hand knowledge necessary for eventual manufacture. This partnership proved highly successful in obtaining a producible design in an innovative and advanced system.

The evolution of the Laboratory's APOLLO organization was basically completed with the establishment of many APOLLO groups integrated formally in July 1963. This change brought the necessary technical skills into a direct-line function under a Program Manager, thus ensuring design control and providing NASA with definitive engineering interfaces. Figure 5-1 illustrates M.I.T.'s APOLLO organization for this and subsequent periods,

The sheer size of the program, the tight schedules, the volume of required documentation and the resultant necessary emphasis on management organization to carry out the complex technical tasks were all unusual in Laboratory experience. The Laboratory's evolving APOLLO organization did cope with these problems and, during this period, produced the hardware designs for the Command Module (CM) Block I series 100, CM Block II, and the Lunar Module (LM) systems. In addition, all the facilities, procedures and special tools for testing the systems were developed. Software to operate with the system followed on a schedule parallel with that of the hardware; development of equations, specifications, simulators and other necessary parts of the software effort proceeded.

After three years in this relatively stable organizational mode, the APOLLO groups were reoriented to accommodate problems developing in the software and in recognition of the advanced status of the hardware. By July 1966, the hardware design on the last configuration was completed and the industrial contractors' resident effort had been generally phased out. The M.I.T. hardware role had moved into reviews of changes, field-site operations, and monitoring of system-performance data. The hardware organizational entities remained the same, but the levels were reduced to meet only operational requirements.

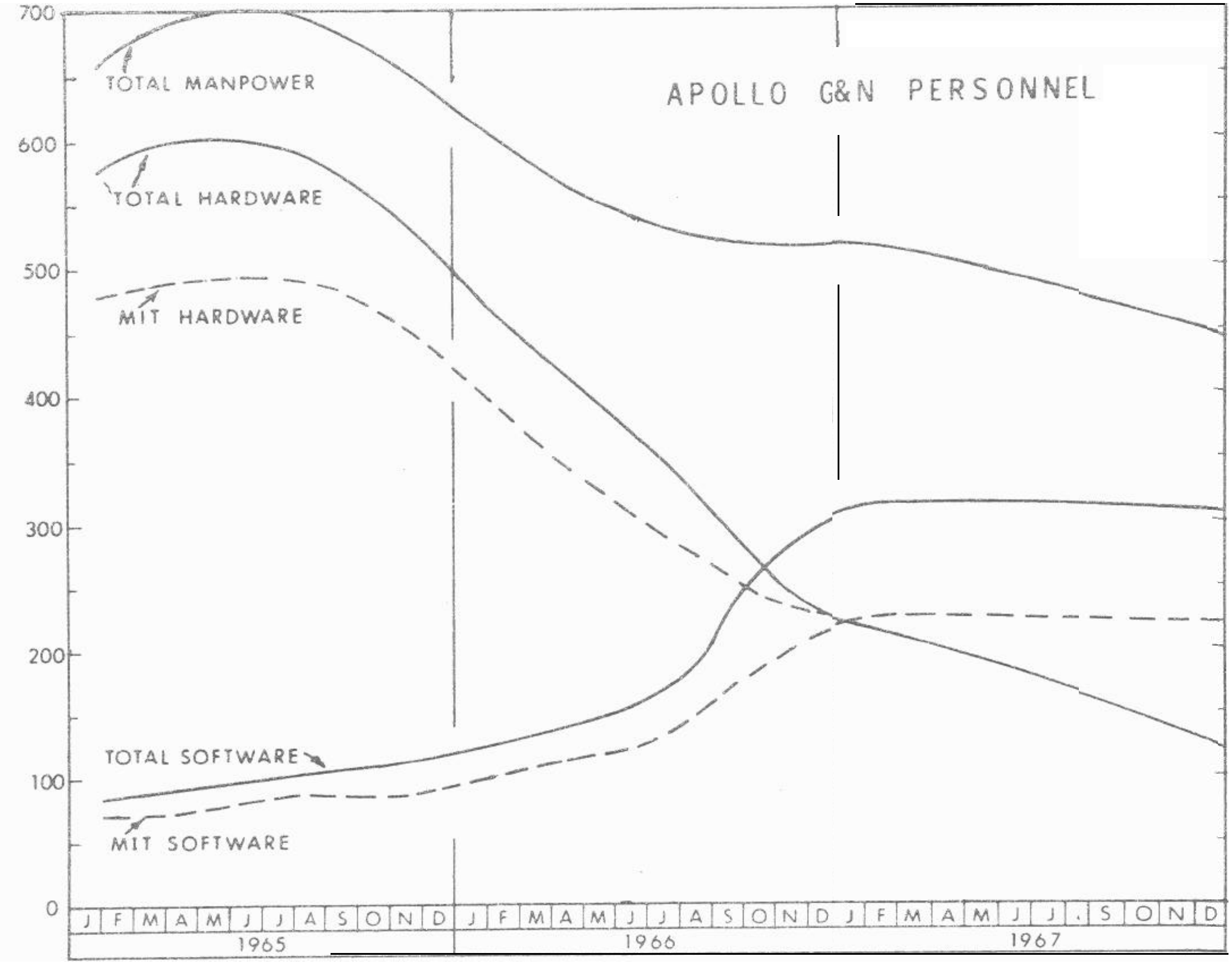


Fig. 5-2 APOLLO G&N Personnel

At this same time, however, the need for a vastly expanded software organization became evident, The early software development had consisted of a small group that was highly motivated, innovative and experienced. As the schedule developed, program requirements demanded production software, which required extensive testing. The magnitude of the effort dictated the organization of groups functionally oriented to handle phases of the "production" job, with control procedures to ensure quality and schedule.

During this change in organization, the basic line structure remained the same; the thrust at this point was in applying the managerial controls and techniques learned on the hardware phase to the software. A new position created at this time was the Project Manager-one responsible for the CM software programs and one for the LM software programs. These positions were created to match NASA and contractor interfaces and to provide a focal point at the Laboratory for coordinating requirements and the use of resources (manpower and computers) in developing the two distinct deliverable end items.

5.3 EVOLUTIONARY CHANGES IN PROJECT EFFORTS

The transition in focus during Project APOLLO's lifespan is shown in Table 5-I, which lists in chronological order the APOLLO missions and major milestones in the APOLLO effort at MIT/IL,.

5.3.1 Engineering Resource Allocation

From 1961 through 1965, the project manpower resources were concentrated on developing system hardware. In 1966, this hardware development effort tapered off, and the requirements for designing and developing the mission computer programs increased. Later in 1966, software development took precedence as the primary task. This resource allocation trend is demonstrated by Figure 5-2, which details APOLLO personnel assignments at MIT/IL from 1965 through 1967. This time period corresponds with the transitional phase of the project, illustrated by the changing slopes of the manpower curves and by the crossover from hardware to software predominance. Figure 5-3 shows the trends for the total project life through mid-1969.

The allocation shift in engineering resources from hardware to software efforts did not result in sudden personnel changeover. To the contrary, total project staff gradualiy decreased from a peak of 700 in mid-1965 to the 1969 level of about 475. This total includes M.I.T. personnel, resident staff from the industrial contractors,

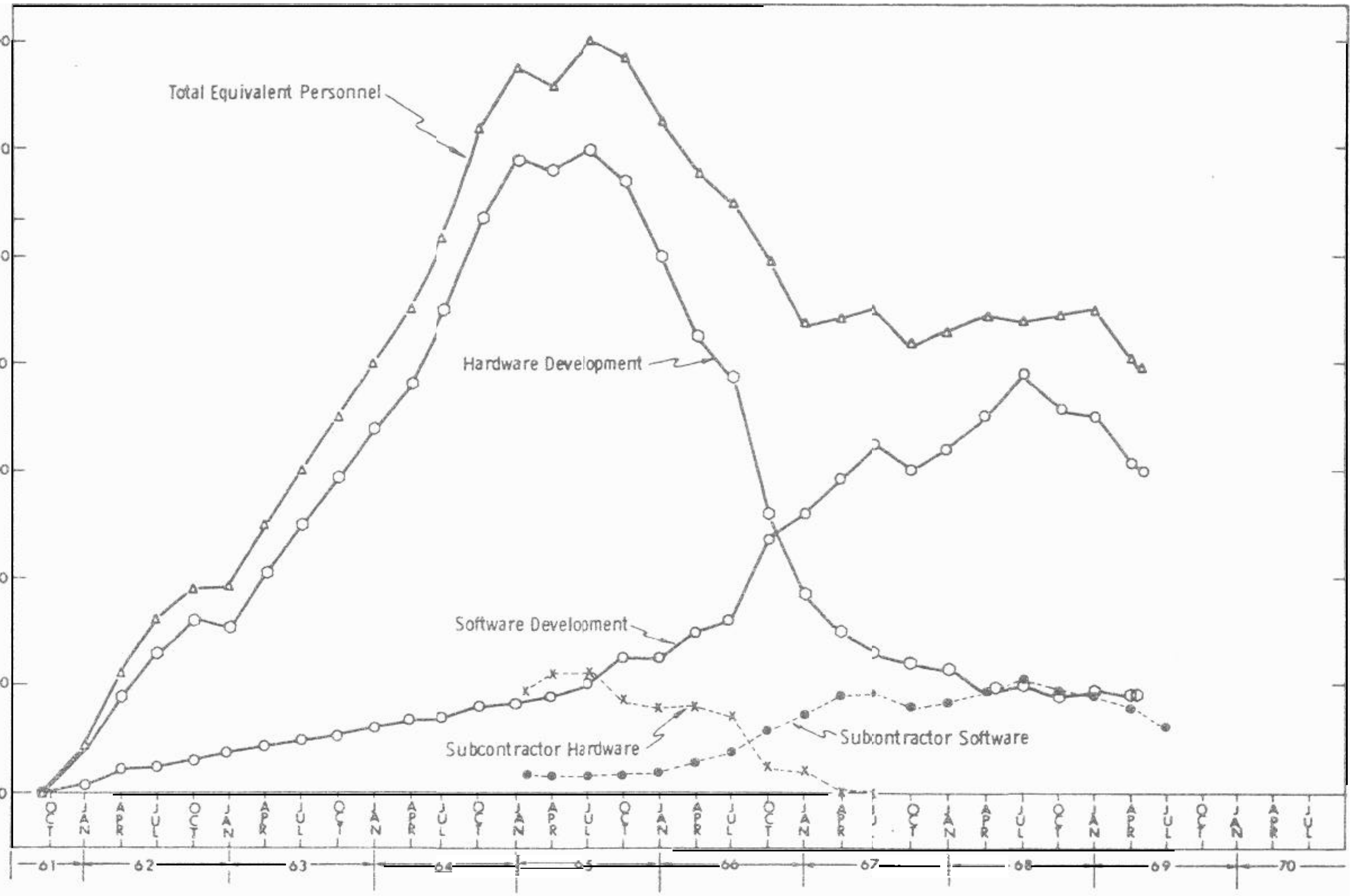


Fig. 5-3 APOLLO Personnel (All Phases)

Table 5-I

Missions and Milestones

February 1961	MIT/IL study contract (six-months' duration) for APOLLO GN&C system received from NASA.
August 1961	NASA selected MIT/IL to develop the GN&C system for APOLLO.
	C. Stark Draper, Director of MIT/IL, proposed in a meeting with NASA Administrator James E. Webb, Deputy Administrator Hugh L. Dryden and Associate Administrator Robert C. Seamans that at least one astronaut should be scientifically trained, since it would be easier to train a scientist to perform the pilot's function than vice versa. In a letter to Robert C. Seamans in November 1961, Dr. Draper further suggested himself as that individual.
Summer 1961	Proposal made to NASA for GN&C study; completed work statements for the Instrumentation Laboratory's APOLLO Project.
October 1961	M. Trageser appointed Program Director and D. Hoag appointed Technical Director of APOLLO Project at MIT/IL,
	GN&C system design started.
November 1961	NASA representatives from MSC and Headquarters visited MIT/IL to further discuss the APOLLO contract changes and hear presentations on technical progress to date,
February 1962	First contractor briefing on GN&C system.
	MIT/IL released requests for quotation to industry for major components of the GN&C system-computer, inertial, and optical subsystems.
March 1962	Industrial contracts for pulsed integrating accelerometer and inertial reference integrating gyro.
May 1962	AC Electronics Division of General Motors Corporation was selected by NASA for fabrication of the inertial platform, for systems integration, and for test and assembly of the GN&C system. Kollsman Instrument Corporation was selected for the fabrication of the optical subsystem. Raytheon Company was selected to manufacture the computer subsystem.
July 1962	NASA assigned MIT/IL to report on a simulated lunar-landing trainer utilizing the GN&C instrumentation and other displays as necessary or proposed.
Summer 1962	MSC reported that the Lunar Module GN&C systems should have as many components as possible identical to those in the command module. MIT/IL concurred.
November 1962	Decision by NASA and MIT/IL to use micrologic circuitry in the prototype onboard computer.

June 1964	First GN&C system delivered to North American Rockwell.
	Digital-autopilot development started.
	Start of Block II GN&C system release for command module.
	Lunar Module GN&C system design release started. Map and data viewer was deleted from the Command Module GN&C system during the same month.
	Lunar Module Optical Rendezvous System (LORS) and optics protective door deleted.
October 1964	Block I series 50 released with inflight-repair capability deleted.
November 1964	Block II series 100 released.
September 1965	SUNRISE CM test program released.
December 1965	Critical design review on GN&C systems was held at MIT/IL.
	Deletion of star tracker and horizon photometer from optical subsystem.
January 1966	First GN&C system for the Lunar Module delivered to Grumman.
February 1966	R. Ragan appointed Deputy Director for NASA programs and D. Hoag appointed Program Director of APOLLO project at MIT/IL. R. Battin became Technical Director, Mission Development, and J. Miller became Technical Director, Systems Development. M. Trageser became Director of Advanced Technology for the Laboratory.
25 August 1966	APOLLO 3, ballistic suborbital unmanned CSM.
October 1966	Software-program effort separated into subroutines.
November 1966	SOLARIUM 54 mission program released for APOLLO 4 (CM).
	SOLARIUM 55 mission program released for APOLLO 6 (CM).
March 1967	SUNBURST mission program released for APOLLO 5 (LM).
9 November 1967	APOLLO 4, SATURN V launch into high-apogee orbit, simulated lunar return.
22 January 1968	APOLLO 5, unmanned Lunar Module, earth orbit.
4 April 1968	APOLLO 6, SATURN V launch into high apogee orbit, unmanned simulated lunar return.
September 1968	SUNDISK 282 mission program released for APOLLO 7 (CM).
	COLOSSUS I (237) mission program released for APOLLO 8 (CM).
October 1968	SUNDANCE 306 and COLOSSUS IA (249) mission programs released for APOLLO 9 (LM and CM, respectively).

11 October 1968	APOLLO 7, first manned earth orbit in the Command Service Module, rendezvous to spent booster.
21 December 1968	APOLLO 8, first manned lunar-orbital mrssion,
March 1969	LUMINARY I (69/2) and COLOSSUS II (45/2) mission programs released for APOLLO 10 (LM) and CM, respectively),
3 March 1969	APOLLO 9, first manned Lunar Module flight, earth orbit, rendezvous exercises to spent booster.
April 1969	N. Sears appointed Technical Director, Systems Development.
	LUMINARY IA (99/1) and COLOSSUS IIA (55)missionprograms released for APOLLO 11 (LM) and CM, respectively),
18 May 1969	APOLLO 10, manned Lunar Module operations in lunar orbit, descent to 50,000 ft.
16 July 1969	Launch of APOLLO 11.
20-21 July 1969	First manned lunar landing.
24 July 1969	Safe return to earth by APOLLO 11 astronauts.

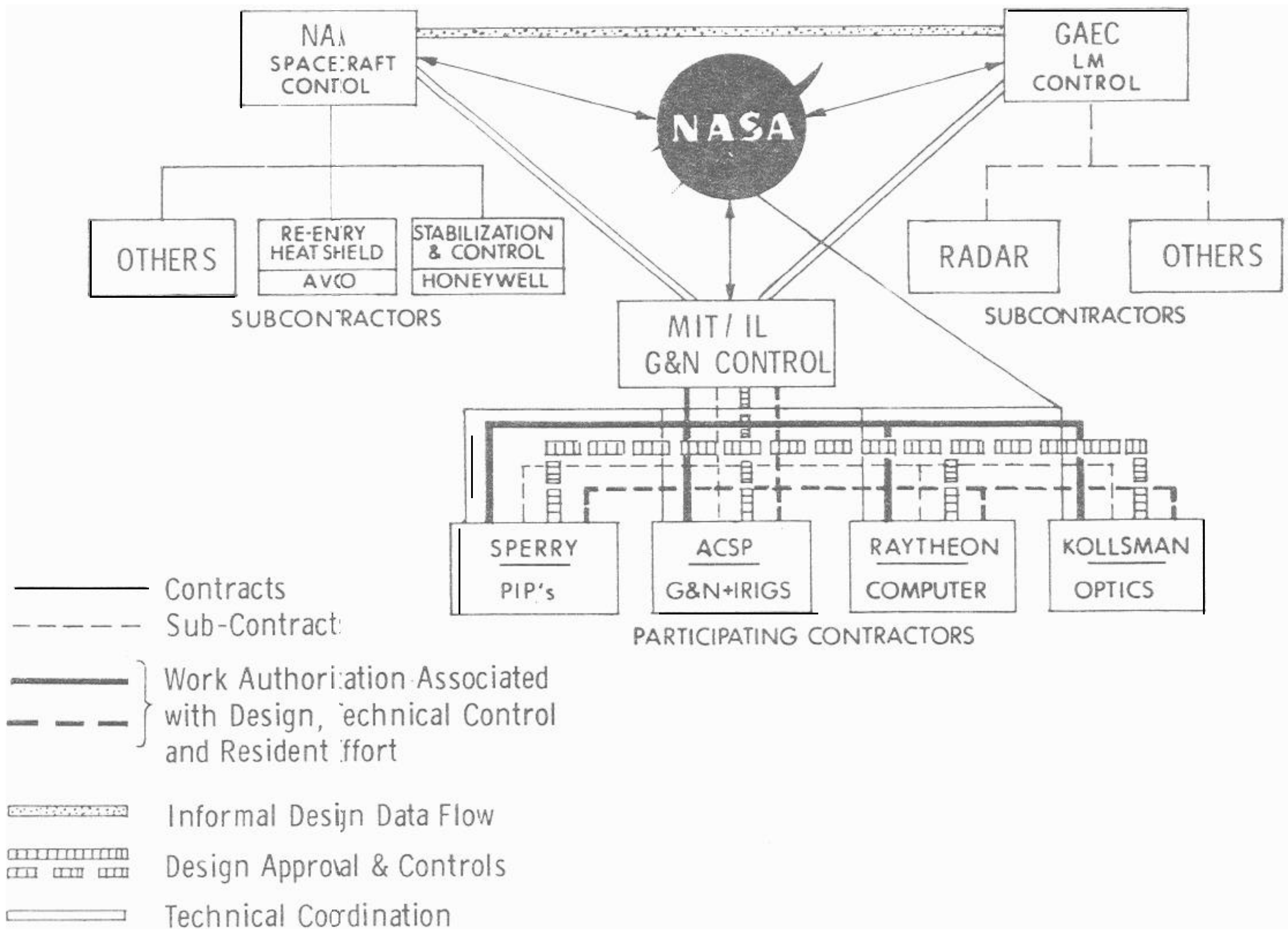


Fig. 5-4 Early APOLLO Design Coordination

and personnel subcontracted directly to M.I.T. The transition involved several factors: returning industrial resident staff to their respective organizations (AC Electronics, Raytheon, KIC), increasing the number of subcontracted engineers and programers on the software effort, changing the assignments of M.I.T. personnel, and permitting attrition and reassignment of staff members to other projects. The manpower curves of Figure 5-2 reflect the changing work requirements, while maintaining a reasonably stable Laboratory staffing level. Rapid changes in "Subcontracted Hardware" and "Subcontracted Software" curves indicate the use of this temporary engineering resource as a buffer in overall staff reallocation. (This practice is similar to the general practice in industry of using "job shoppers" to help during peak work periods.)

Another buffer is the employment of part-time students from the Institute and from other metropolitan Boston universities. Short-term or part-time analyses and studies by these students serve to accomplish work for which hiring a full-time staff member is not feasible. In addition, employment of students is extremely important from theviewpoint of the Laboratory's educational objectives-giving the student opportunities to work on "real world" technical appiications-and as a future source of engineering talent. Illustrative of the students' contributions on such problems are the approximately 60 theses prepared during the contract period of APOLLO research. A number of them are cited as T-Reports in Appendixes A and B.

5.3.2 Hardware Developmental Tasks

Early APOLLO work was concerned with conception, design, and development of the GN&C system hardware and included four major engineering tasks. The first task was technical direction of the APOLLO GN&C associate contractors. AC Electronics, Raytheon, and Kollsman were associate contractors with the Instrumentation Laboratory until mid- 1964. They were directed administratively by NASA and technically by the Laboratory. In this arrangement, their technical management was implemented from MIT/IL through a series of Technical Directives (TDs), defining their work and the reporting requirements, and placing funding limits in terms of maximum man-months of work for specified tasks.

Figure 5-4 is a diagram of this working arrangement prior to the contract novation. In the illustration, the double lines indicate the technical coordination exercised by MIT /IL among all of the participating contractors who were concerned with GN&C systems development or usage, The other lines designate separate functions: contracts, design approval/control and work authorization. Design approval/control was exercised through the release of drawings from MIT/ IL, and work authorization

HISTORY OF TDRR's (HARDWARE)
PROCESSED THROUGH APOLLO CCB

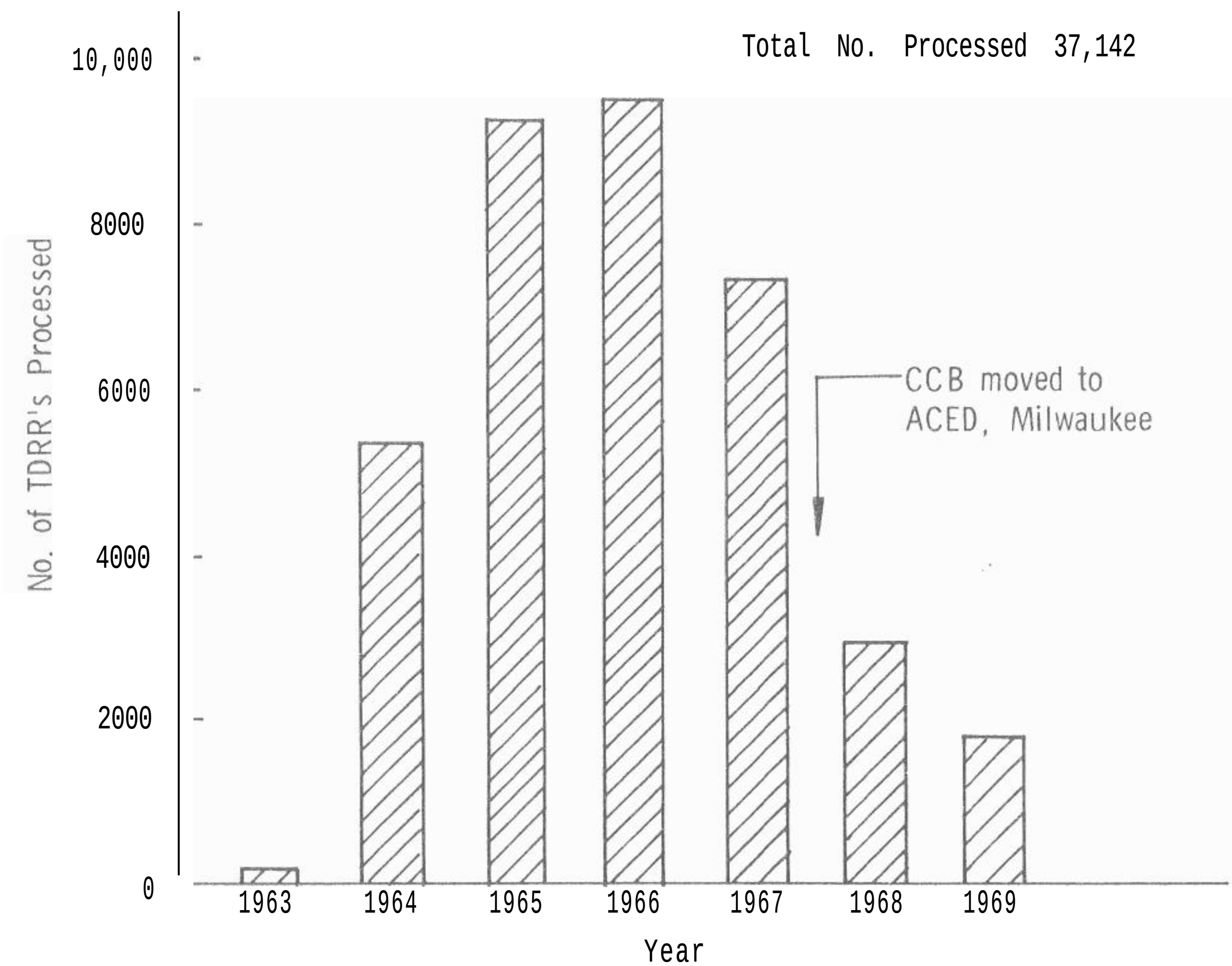


Fig. 5-5 Technical Data Release or Revisions (TDRRs) Processed through APOLLO CCB

was via the technical directives. The thick, solid lines designate that contractual relationships existed between NASA and all the participating contractors,

A second task was systems design and testing at the Laboratory. The Design Review Board (DRB) and Change Control Board (CCB) became major management tools in the control of this work, The change control board was chaired by NASA and met at the Laboratory until the time when AC Electronics became responsible for GN&C systems integration. Some 37 000 Technical Data Release or Revisions (TDRRs) have been processed using this single approval and drawing release approach (Figure 5-5). The design- review and change-control functions are discussed in more detail shortly.

The Laboratory also supported NASA by monitoring systems development from production through testing, spacecraft installation, and Cape Kennedy launch to ensure mission flight objectives. This was done by assigning systems experts from the Laboratory to work with staff engineers at APOLLO field sites—North American Rockwell, Grumman Aerospace Corporation, Manned Spaceflight Center, and Kennedy Space Center. The operation of these field sites is treated more fully in Chapter IT, but it is worth noting here that these teams were very important in interface control and in assuring the implementation of subsystem and system test philosophy developed at the Instrumentation Laboratory, The series of interface Control Drawings (ICDs) and process specifications evolved, in part, because of these teams. The excellent communication between the Laboratory and the spacecraft contractors, which resulted, in part, because of these fieid-site teams, is exemplified by the present high level of understanding of system performance characteristics (see APOLLO Operations Handbooks which are generated by- the spacecraft contractors as an example),

The fourth Laboratory responsibility was generation of GN&C systems test procedures for checkout and prelaunch countdown de-monstration at Kennedy Space Center, including the generation of data for computer initialization prior to launch. Comprehensive testing of system-level operational integrity prior to launch was of major importance in the hardware development phase. Section I of the APOLLO Guidance System Operations Plans exemplifies this work (see H-567 and R-577).

5.3.3 The Software Developmental 'Tasks

As the GN&C systems were designed and tested, developing computer programs for inflight implementation of the guidance, navigation, and control techniques became necessary. Conceiving, designing, and developing these programs for both the command and lunar module computers resulted in 12 computer programs from well over 900 man-years of work. (The overall software development effort is described fully in Part 3.) Verification-test series, including All-Digital, hybrid, and all-hardware tests, had to be developed, Six distinct levels of exhaustive testing were done -from math-model checkout, through pre-release testing, to post-release checkout continuing to the launch day. Both nominal timeline and overstress types of tests were conducted. Typically, several hundred revisions of each program were assembled, as all the capabilities for a specific mission were designed and integrated, and as the tes-t series revealed pre-release program anomalies, or "bugs," requiring removal.

Guidance System Operations Plans (GSOPs) were written for each mission. These documents contained seven major sections: prelaunch testing, data links (up and down telemetry), digital autopilots, operations modes, guidance equations, control data, and error analyses. The GSOP became the governing specification for the systems' software design and use; thereby representing a major systems management tool in the development effort.

The Laboratory also supported flight controllers at Cape Kennedy and at Houston with engineers at the two locations and at the Laboratory in contact with the control centers via. restricted telephone lines. This real-time consulting support has continued for all APOLLO missions. Post-flight data were evaluated to aid in planning subsequent missions.

APOLLO crews were trained by MIT/IL in the general principles of space guidance, navigation, and control, in the use of the GN&C systems for each spacecraft, and in simulations for specific mission segments, Instrumentation Laboratory engineers also supported mission simulations at Cape Kennedy and Houston just prior to each flight, These simulations confirmed final flight plans and detailed timelines. Also, the Laboratory developed a series of simulated-flight computer programs for training both the astronauts and ground crews in operating the actual mission programs.

5.4 COMMUNICATIONS

5.4.1 Technical Memoranda

The technical memorandum served as an important medium of information dissemination both within the Laboratory and throughout the APOLLO industrial and Governmental community. For the APOLLO Project, memoranda in the order of 10 OOC have been published by MIL/IL over the past eight years, They are the ongoing operational documents, although they are not contractually recognized and have loosely controlled distribution requirements. A listing of the various types of memoranda follows, In retrospect, the publication of the memoranda and distribution to the various specialists in the Laboratory, industry, and NASA appears as one of the more important methods of communication, primarily because of the speed of information dissemination. Memoranda of general interest are also often published as Engineering or Research Reports,

System Test Group (STG)—These documents are published within the GN&C Systems Test Organization and are concerned primarily with hardware development and systems testing. This type of memorandum reports on the major outputs of the hardware- related engineering groups and the activities centered in the Systems Test Laboratory. For example, the major effort of the System Test Group in verifying the compatibility of the mission programs and the hardware has been documented via the group's memoranda.

Optics and Navigation (O&N)--This type of memorandum is published by Group 23N of the GN &C Systems organization. It is concerned with design, development, testing and flight use of optical subsystems.

Inertial Subsystem (ISS)—Almost all the work performed by Group 231 is documented by this type of memorandum. Technical matters concerned with both the inertial subsystem and interfacing subsystems may be published also in the form of System Test Group memoranda.

Digital Development (DD)—Computer-subsystem hardware development testing is documented through these memoranda, published by the Digital Development Group,

Display Group (DG)—The Display and Human Factors Division of the Mission Development organization was focused during the initial project phase on the design and testing of displays and controls. Its work has shifted toward inflight equipment usage, and more recent memoranda have thus been concerned with software and flight procedures,

Digital Computation Group (DCG)—This group's memoranda are concerned with development of the computer programs and with the All-Digital simulations for testing them.

Space Guidance Analysis (SGA)—This series deals with the derivation of the mathematical. and physical techniques for thenaviga.tion and guidance programs and is published by the Space Guidance Analysis Group in the Mission Development organization.

COLOSSUS, SUNDANCE, and LUMINARY—These three series of memoranda deal specifically with the computer programs of the same names. They are published by the Mission Program Development Division of the Mission Development organization and serve to describe topics such as differences and similarities between programs, the quality of a given program revision, and problems needing resolution.

Mission Techniques-This series is published by the Technical Staff of the Mission Development organization and deals with special techniques and procedures for using the GN&C systems. It assures consideration in systems utilization of crew preferences and techniques refinement from one mission to the next.

Post-Flight, Data Analysis--The performance of the GN&C systems on each mission is evaluated in detail via this series of memoranda, They are based on crew debriefings, downlink telemetry listings, and the impressions of Instrumentation Laboratory staff members monitoring the missions.

5.4.2 Administrative Letters and Memoranda

Four types of administrative letters and memoranda provide for formal communication among the general staff of the Laboratory, and between the Laboratory and NASA.

Contract Letters, published by the Laboratory Business Office, involve the agreements between the Institute and the various governmental agencies.

Administrative Memoranda (A-Memos) are published by the general staff of the Laboratory and deal with such topics as personnel promotions and policy changes. These memoranda are not formally distributed outside M.I.T., although they may 'be posted on bulletin boards as unrestricted information.

APOLLO Project Memoranda deal with project-wide matters of either an administrative or technical nature. They are published by the managerial staff of the APOLLO Project.

APOLLO Guidance Letters (AG-Letters) are formal communications from the Project to NASA. They are contractually recognized documents of either administrative or technical content and are normally signed by Project staff members at the division-director level or above.

5.5 DESIGN REVIEW BOARD

The Design Review Board (DRB) was a significant medium of communication and design management established to review and evaluate all technical aspects of the GN&C systems, including interface problems, procurement logistics, suitability for fabrication, assembly, and test or use. The review was intended to provide early consideration of system interface integration capability and reliability factors. The APOLLO Project Director or his designated alternate acted as chairman of the board. Membership included design-group heads, a reliability-group representative, and a systems-group representative. Technical specialists were called in as required. MIT/IL, Report R-383 contains the detailed design-review procedures.

Design -review deliberations were conducted at two distinct levels. At the systems- and subsystems-interface levels, usually presided over by the Project Director, overall performance, design concepts, configuration, and interfaces within and external to the GN&C system were evaluated, and broad requirements and ground rules were established.

At the subsystem level, usually presided over by a division director, there were three separate design-review boards. The Director of Digital Development chaired the board for the computer subsystem and also acted as the technical monitor for Raytheon production of the subsystem. The Director of Mechanical Design chaired the board for the optical subsystem and acted as technical monitor for production by Kollsman. This board was also responsible for the Block I, Block II, and Lunar Module navigation bases, the Block I coupling-data-unit mechanical design, and the map and data viewer—all fabricated by AC Electronics. The third board, headed by the Inertial Subsystem Design Group Director, was responsible for the rest of the system. It oversaw AC Electronics production of the inertial subsystem, all the electronic packaging, displays and controls, cable harnessing, signal conditioner, and system integration.

Design reviews had been conducted continuously from the very beginning of the program. The basic configuration design, performance requirements, ground rules, and major program aspects of GN&C systems development evolved from those reviews. When designs matured to the level where they were ready for production release, formal design review boards were instituted. The system-level board met weekly and reviewed in detail the candidate-design drawings and specifications. The rationale and compromises were evaluated down to the component level. After board approval, the design drawings went to the change control board for formal release. As the work tempo increased, board meetings were scheduled more frequently. However, the volume of drawings soon reached such proportions that the single board could not handle it. The subsystem-level boards were then established. Design review and change control board activity peaked in 1965 (see Figure 5-5). Thereafter, when review requirements became less voluminous, the three boards at the Laboratory were consolidated into one group.

The board's control of GN&C system evolution is demonstrated by the 37 000 drawings and specifications released by the change control board. To meet analogous needs in computer program development, a Mission Design Review Board (MDRB) and a Change Control Board were formed, as discussed below.

5.6 CHANGE CONTROL BOARD

The Change Control Board implemented the decisions of the Design Review Board regarding hardware by the formal release of the system drawings and test specifications. Formal release procedures were agreed upon by the associate GN&C contractors as described in MIT/IL document E-1166, Revision 5, "Technical Data Release Procedures."

From a managerial viewpoint the guidelines for controlling drawing release were important with respect to mission accomplishment, schedule compliance, and cost impact in that order. By December 1963 such guidelines had been formulated, tested in practice, and agreed to by NASA, MIT/ IL, AC Electronics, Raytheon, and Kollsman. Because these guidelines became sovaluable in the control of systems design release, they are reiterated here.

It was agreed that the change control board at MIT/IL would recognize two levels of changes and operate with priority assigned. accordingly. These still-valid guidelines and levels evolved to the approach described below.

In addition to the two classes of document release, revision to either class of document is divided into two broad categories, Class 1 and Class 2, as defined below:

CLASS 1 CHANGE. (Ref. ANA 445) Any proposed engineering change to (1) an accepted or unaccepted end item (system, equipment, unit, or subassembly) or (2) an accepted or unaccepted part listed on a provisioning list(s), whether such part is to be incorporated in the end item or stocked in the supply system, is designated a Class 1 change when one or more of the following is affected:

1. Contract specification, contract price or fee with increase or decrease, contract weight, contract guarantees, contract delivery, or contract test schedules
2. Contract-specified reliability and/or contract-specified maintainability
3. Performance as stated either in definite terms, or goals, or as experienced in items in service use
4. Interchangeability, or a change in category regarding substitutability or replaceability
5. safety
6. Electrical interference to communications electronic equipment or electromagnetic radiation hazards
7. Aerospace Ground Equipment/Support Equipment (AGE-SE), trainers, training devices, or Government Furnished Equipment (GFE)
8. Preset adjustments or preset schedules to the extent that (a) new identification must be assigned or (b) operating limits are affected
9. Systems, equipments, or facilities produced by other contractor(s) to the extent that the affected other contractor(s) must accomplish an engineering change to maintain compatibility at the interface(s)
10. Operational computer programs
11. Any proposed change involving retrofit of an accepted item
12. Interface activities of logistics, training, and reliability.

Class 1 changes must be approved by the MIT/IL design review board or AC E iectronics board depending on design responsibility prior to the change control board. 'The effectivity for a Class 1 change must be specified prior to the review by the design review board. Any change made to the effectivity at the change control. board requires approval of the design review board and/or must be in accordance with NASA Contract Change Authorization. The effectivity stated at time of change control board approval is mandatory.

Any proposed changes to NASA documents (NDs) which MIT/IL considers mandatory and that affect manufacturing processes, workmanship requirements, inspection criteria, and flight processing specifications are transmitted to NASA for review and technical concurrence before being transmitted to the industrial contractor for any action on his part. Such concurrence is obtained in writing from NASA before any further action is taken. Technical explanation and justification of necessity are furnished in writing as part of the submission to NASA.

All proposed Class 1 changes are prepared as complete package changes. The changes are defined in all areas of the drawing structure through the highest assembly affected including ATP, PS, FTM, and ND. The effect on ground-support equipment is defined in the package,

CLASS 2 CHANGE. (Ref. ANA 445) Any engineering change not falling within Class 1 as defined above is designated as a Class 2 change. Generally, Class 2 changes are those changes which are desirable but not technically necessary from a system function standpoint. For instance, changes required to comply with documentation format specifications would be in this class. A Class 2 change cannot change form, fit, function, or reliability so as to affect interchangeability. No effectivity is specified and the change is incorporated on the basis of no change in contract cost and no schedule impact.

When a technical data release is prepared to incorporate a Class 1 change in a document, Class 2 changes are sometimes incorporated on the same release. Class 2 changes released in this manner automatically become Class 1 changes and are subject to all the requirements imposed for a Class 1 change, including design review board review and approval prior to the change control board. Care is therefore exercised that true Class 2 changes processed by this method do not produce a cost or schedule impact or result in nonessential changes to hardware.

If any change on the technical data release is considered by the change control board to be Class 1 or if any doubt should arise concerning the Class 2 designation for a change, the entire data release is submitted to the design review board for evaluation and approval.

Some documents are processed through the change board for record purposes only and to ensure distribution throughout the system. Documents falling rnto this category are interface control documents and interface revision notices. When documents of this type are submitted to the change board, the technical release should be boldly marked in the “Description of Changes” column “For Information Only,” thus indicating that the signatures of the contractor and NASA are not required.

The action on these changes is normally one of the following:

1. Immediate authorization of the change:
   (a) Considered to be within scope of the contract and, therefore, no added fee
   (b) Considered to be out-of-scope of the contract
2. Request for a NASA/contractor MIT/IL study of alternative solutions
3. Authorization of the change with a different effectivity or class of change.

Other agreements reached are concerned with the method to be followed in determining the cost and schedule impact of change.

5.7 THE ROLE OF THE INDUSTRIAL RESIDEKTS

Residents from thevarious APOLLO contractors provided a most important source of design and development support to the Instrumentation Laboratory. They brought with them specialized skills to deal with the inertial, optical, and computer subsystems. During their stays, often for a year or longer, they typically reported directly to their lead engineer who in turn reported to a Laboratory staff member. In some cases individual residents reported to individual staff members.

The industrial support residents helped in virtually all APOLLO groups. For example, during the peak year of effort, 1965, residents from AC Electronics were located in the Systems Test Group, Inertial Subsystem, Optical Subsystem, Display and Controls, Reliability, Ground Support Equipment, and Systems Operations. At various times during that year there were from 50 to 60 engineers in residence at the Laboratory from AC Electronics.

Raytheon residents aided MIT/IL, in all the groups generally concerned with design, development, and test of the computer subsystems and in the Space Guidance Analysis group. Kesidents, during 1965, were located in the Analysis Group and in coupling data unit design, Systems Test Group, Digital Development, Ground-Support Equipment, Reliability, and Packaging groups. In 1965 there were about 40 engineers in residence from Raytheon Comnany. The previously shown APOLLO manpower resource allocation chart graphs the totals of all residents at the Laboratory from the beginning of the contract through mid-1969.

The resident engineers, designers, and draftsmen from Kollsman Instrument Corporation supported the design, development, and test of the optical subsystems, Primarily they were located in the Mechanical Design, Ground-Support Equipment, and Systems Test groups. Approximately 17 of these personnel were in residence during 1965.

Technical liaison between the Laboratory and their own firms was an important function of the residents. Technical information was disseminated quickly and accurately throughout a very complex set of organizations. Figure 5-6 demonstrates this by showing the organization and flow of communications between the Instrumentation Laboratory and the APOLLO contractors after the contract amendment in 1964. Many types of directions and information-technical coordination, design approval

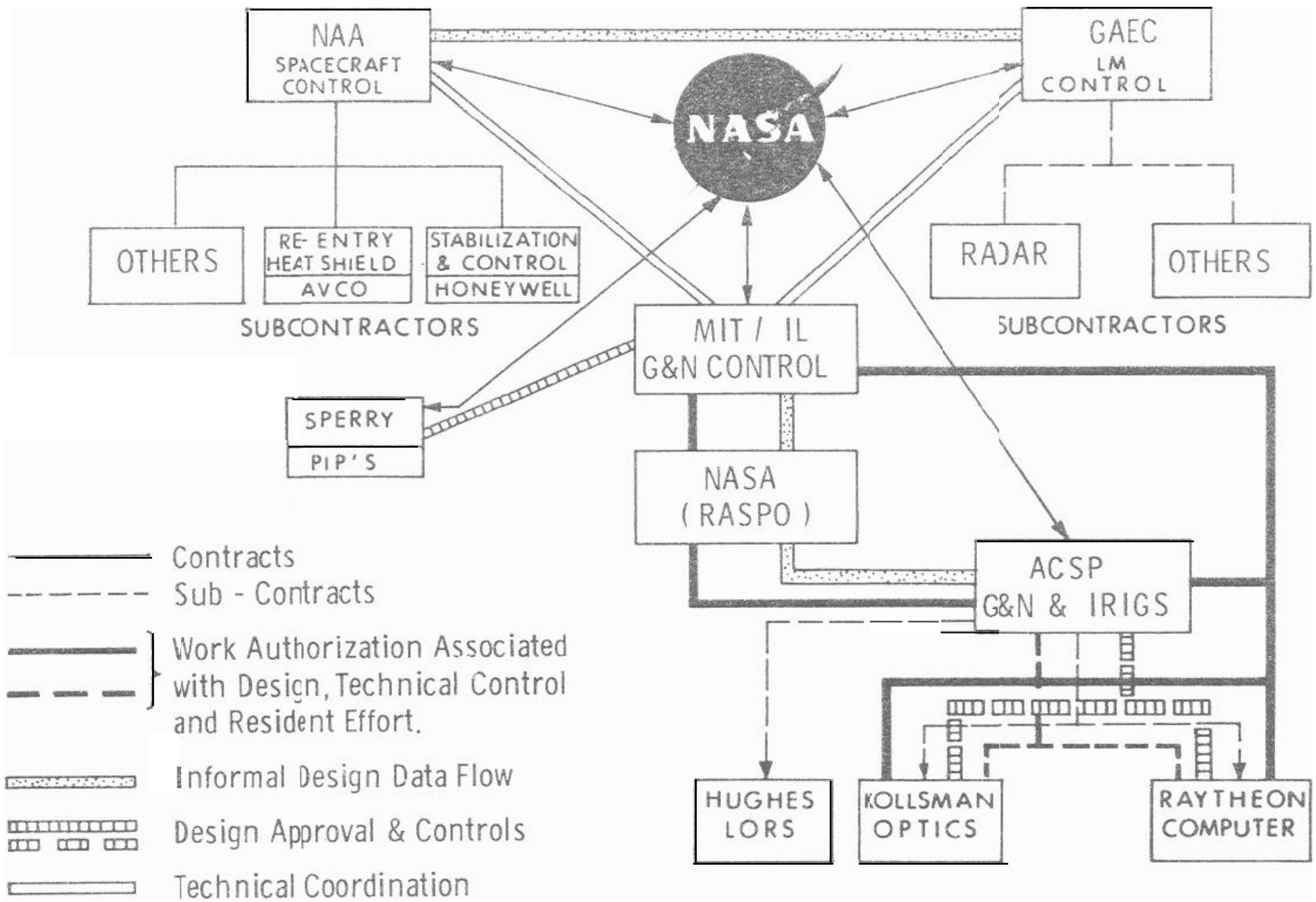


Fig. 5-6 Organization and Contractors

IL and APOLLO

and control, and "level-of-effort" work authorization directives (Technical Directives)—were flowing from the Laboratory even after 1964. The industrial residents performed this technical liaison function primarily through the "Informal Design Data Flow" channel.

Design drawings were submitted by the resident representatives for design board approval, thus giving them a direct role in amendments and modifications. The drawings were then forwarded to the change control board as technical data releases. All the formal documentation affecting GN&C system development was released through this technical management channel.

Residents also supported field site activities as field engineers and as participants in various coordination meetings. Finally, as Appendix A shows, resident staff have worked jointly with the M.I.T. staff to do research and analyses leading to E- and R -series reports.

SECTION 6.0
PROJECT AND MISSION DEVELOPMENT REORGANIZATION

6.1 RECOGNITION OF NEED FOR CHANGES

From 1961 through 1965, the project organization was intentionally kept informal to maintain a creative atmosphere. It became clear in viewing the scope and complexity of the programing work, however, that increased staff and a more effective organization would be necessary. One important evolutionary series of decisions supporting this realization was the increase in the APOLLO Guidance Computor memory size from 4000 to approximately 39 000 words to meet programing needs. Other decisions involved the requirement for a digital (computer-controlled) autopilot as an added function of the system, thereby increasing the complexity of the software effort significantly. There was also intense pressure to develop SUNSPOT, the first manned orbital-flight program, very quickly after it was defined. By the end of 1965, the development of the computer programs became the central problem of the entire APOLLO project at the Instrumentation Laboratory.

One overall problem was the shortage of experts to design, code and test the computer programs. Documentation of the routines fell far behind, because only a small nucleus of engineers familiar with the programs were able to translate them for the rest of the APOLLO community. These engineers were so busy developing the programs that they had little or no opportunity to document them.

Another problem area involved the need for more stringent change control, to limit iterations, control the MSC / MIT interface, and to meet required schedules.

6.2 REORGANIZATION APPROACH

It was decided in November 1965 that a new organizational structure was necessary to meet the evolving requirements for computer-program development. Three actions resulted from this decision: by early 1966 the first formal organizational charts were developed and formally documented; in mid-1967, new groups were defined and established under Mission Development; and, in Autumn 1967, the project-manager control technique was added wherein a specific individual would have total responsibility for a specific mission program. In the previously shown APOLLO organization chart, the Project Managers for the computer programs COLOSSUS and LUMINARY refer to the control technique added.

Under this new organization structure, project chains of command were clearly defined. Organizational charts and status were updated monthiy. One aim of the early reorganization was simply to break the software development tasks into more manageable components. Another objective was to relieve some of the excessive workload on the Director of Mission Development by reducing the number of supervisorypersons reporting to him. Formal organization also reduced the number of supervisory staff reporting directly to the APOLLO Project Director.

6.2.1 Group Definition and Tasks

Group 23A, Space Guidance Analysis Division, was established with the broad tasks of conceiving and developing the mathematical and physical approaches to the mission (e.g., the navigational targeting techniques and cross-product steering techniques in guidance) and defining the mission phases in terms of specific computer programs,

Group 23B, Mission Program Development Division, was chartered to implement the guidance, navigation and control techniques in specific mission programs and to document the programs in the form of flow charts and computer routine descriptions. The documentation also included system integration-the control of fixed and erasable memory allocations, change control, assembly control, program integration, and telemetry and program service.

The Display and Human Factors Division, Group 23D, was made responsible for mission-operations support, design of flight procedures, astronaut training in the use of GN&C systems, and a variety of mission-simulation tasks. The part of Section 4 of the Guidance System Operations Plan discussing interfaces among the flight crew, crew decisions, computer processes, and ground support, is one example of the work of Group 23D.

The Systems Engineering Division, Group 23S, was assigned a set of analytical tasks, including, for example, the GN&C system error analysis (formerly GSOP Section 7) performed before each APOLLO flight, the mission planning and flight support given to NASA, and the mission task operations used in simulated flight programs. The overall. design and coordination of Sections 2 and 4 of the GSOP were the responsibilities of Group 23S.

6.2.2 Project Management Approach Adopted

By the end of 1966, the first computer program for Command Module manned earth-orbital flight had been developed at the Laboratory and had been manufactured by Raytheon. When the APOLLO program was delayed in January 1967 by a tragic fire, software development turned first toward reviewing again the computer routines from acrew-safety standpoint and then to simultaneous development of the programs needed for the Command and Lunar Modules in manned lunar orbit and lunar landing, The increased pressures on the organizational structure led to the establishment of the Project Manager function for each of the required computer programs. With Project Managers now responsible explicitly for the major computer programs, line managers were able to devote more of their time to innovative activities, obviating the need for still additional staff to take on the responsibility for innovation.

6.3 SOFTWARE MANAGEMENT AND ORGANIZATION

The Laboratory's approach to management and organization of its software effort for Project APOLLO is discussed fully in Volume VI of this Final Report.

SECTION 7.0
CONCLUSIONS

Given the present state-of-the-art in management theory, no completely proven rules were available for general applicationto the APOLLO project management; furthermore, no hard and fast rules have been derived from managing the APOLLO project. Based on the unprecedented APOLLO experience, however, i-t seems useful to conclude by enumerating some of the guidelines and lessons learned, These may prove valuable in future efforts of similar complexity and scale.

1. An informal organization was workable in the five earliest, innovative years of APOLLO. Thereafter, this approach exhibited shortcomings related to problems of communication, documentation, and the changing external environment as the project grew larger and the type of work changed significantly. A more formal structural approach was viable.
2. It was practicable to effect a complete change through the same general project organization in shifting the type of developmental. work from hardware to software without affecting the nucleus of the permanent staff. This capability depended to a large extent on the existence of a "buffer" of industrial residents and subcontracted support personnel and students; this was a rather unique situation benefitting the Instrumentation Laboratory.
3. The transfer of technology from MIT/IL as the GN&C systems design agency to the APOLLO industrial community was efficiently accomplished through the practice of having industrial support engineers in residence at the Laboratory. This accomplishment was exemplified especially by the residents' participation with the Design Review Board during systems development and design release.
4. The Project Manager authority/ responsibility assignment proved a workabie solution to the general problem of developing two specific types of mission programs with one group of engineers on a parallel work schedule.
5. In-depth managerial visibility was particularly crucial to the software development effort, especially in the area of change control. The use of the Project Manager approach to this area of measurement and control proved workable.

The APOLLO Project has been one of the most innovative, demanding and rewarding efforts ever undertaken by the M.I.T. Instrumentation Laboratory. At this writing, it is fully expected that its satisfactory culmination will be demonstrated by the success of all future APOLLO missions,

The national goal of landing American astronauts on the moon before 1970 and returning them safely has been met with scientific and technological rewards of unprecedented significance. As a real-world problem for furthering the educational and research aims at M.I.T., the project met all organizational objectives. Representing a solved problem in large-scale systems engineering and management, the project can be a new model for application to the future space and social goals of our nation.

CHAPTER II

GN&C SYSTEM DEVELOPMENT (HARDWARE)

This chapter presents the salient features of the APOLLO guidance, navigation, and control (GN&C) system development (hardware) at M.I.T. It begins with a description of the XPOLLO mission in ***some*** detail and continues with a discussion of flight experience. Design objectives and limiting factors are then outlined preparatory to descriptions of the system hardware and an outline of interfaces with the spacecraft. The techniques developed in generating the prototype GN&C systems, and problem-solution combinations are described, as are the spacecraft installation methods, probiems, and solutions, Finally, computer programs for system testing (pre- and post- spacecraft installation) and ground support equipment are described.

SECTION 1.0

APOLLO CONFIGURATION

The goal of the Apollo Project is to place human exploration teams onto the moon and return them safely to earth. A spaceship consisting of three modules is launched on a trajectory to the moon by a Saturn V launch vehicle. The Command Module (CM) is designed for atmospheric re-entry and is to be the home for the three-man crew during most of the trip. The Service Module (SM) provides maneuver propulsion, power and expendable supplies. The Lunar Module (LM) is the vehicle which actually makes the lunar descent. It carries two of the three-man crew to the lunar surface while the other two modules remain in lunar orbit, The Apollo Guidance and Navigation System is the primary onboard equipment used for determination of the position and velocity of the lunar module and for control of its maneuvers. Similar guidance equipment is contained in both the Command Module and the Lunar Module. Each vehicle is equipped with a device for remembering spatial orientation and measuring acceleration, an optical angle device for angle measurements, displays and controls, means to interface with a spacecraft control system and indicators, and a central digital processor,

SECTION 2.0

THE APOLLO MISSION

A brief synopsis of atypical Apollo lunar landing mission is presented in this section. While there are many detailed variations from flight to flight, the guidance, navigation and control systems' functions are generally discussed here,

To prepare and verify the equipment for flight during the prelaunch phase, automatic programmed checkout equipment performs exhaustive tests of the major subassemblies. Two operating sets of guidance equipment are prepared for launch: the SATURN and APOLLO command module guidance systems both continuously measure vehicle motion and compute position and velocity. The SATURN guidance equipment in the SATURN Instrument Control Unit controls the launch vehicle, while the Apollo guidance equipment in the command module (CM) provides a monitor of SATURN guidance during launch. The lunar module GN&C system, after prelaunch testing, is normally powered down for the launch phase of the mission.

Shortly after the firing of second stage thrust, the launch escape tower is jettisoned and the vehicle passes out of the atmosphere. When orbit is achieved, the main S-IVB propulsion is shut down, the APOLLO spacecraft configuration remaining attached to the SATURN S-IVB.

During second and third stage boost flight, the APOLLO command module is configured to allow the crew to take over the S-TVB steering function manually if the SATURN guidance system indicates failure. Ground tracking navigation data telemetered from the manned space flight network (MSFN) stations are available to correct the position and velocity of the SATURN navigation system and to provide navigation data of the GN &C system. The inertial subsystem alignment in the command module may also be updated by star sightings with the optical subsystem. For these measurements, the crew exercises manual command control of vehicle attitude through the SATURN attitude control system. Translunar injection is performed using a second burn of the SATURN S-IVB propulsion. The spacecraft configuration injected into the translunar free- fall path must now be assembled for the remaining operations. The astronauts separate the command and service modules (CSM) from the lunar module (which is housed inside the adapter in front of the S-TVB stage), and then turn the command and service modules around for docking with the lunar module,

Very soon after injection into the translunar free-fall coast phase, MSFN-computed navigation measurements are examined to determine whether there is need for an early midcourse maneuver to correct error in the flight path. This first correction — if it is needed — is made perhaps a few hours after injection.

Mission control on the ground periodically examines the ground-based radar data for uncertainty in position and velocity and the estimate of indicated velocity correction required to improve the present trajectory. If the indicated position and velocity uncertainties are suitably small and the indicated correction is large enought to be worth making, then the crew may execute the telemetered midcourse correction. If the required correction happens to be very small, it is made by using the small reaction control thrusters. Larger corrections are made with a short burn of the main service propulsion rocket.

For lunar orbit insertion maneuvers, as with all normal thrusting with the service propulsion of the spacecraft, the inertial subsystem is first aligned using star sightings. The guidance initiates engine turn-on, controls the direction of the acceleration, and signals engine shutdown when the maneuver is complete. The lunar orbit insertion maneuver puts the spacecraft in an orbit approximately 60 nautical miles above the lunar surface. During. the lunar orbit, navigation measurements may be made to update the knowledge of the actual orbital motions.

Before initiation of the lunar module descent orbit injection maneuver, the vehicles are separated, the lunar module inertial subsystem receives final realignment from star sightings, and the maneuver attitude is assumed. Before separation, however, the lunar module GN&C system is turned on and receives a checkout and initial conditions. During the free-fall phases of the lunar module descent, the command module can make optical tracking and VHF range-only measurements of the lunar module for confirmation of its relative orbit. After the descent coast, the descent engine is re-ignited; the velocity and altitude reducing maneuver is controlled via the lunar module inertial subsystem and autopilot calculations in the computer.

The controlled trajectory of the powered-descent final approach is selected to provide the lunar module crew with a view of the landing area. The vehicle attitude, descent rate, and direction of flight are all essentially constant so that the landing point being controlled by the guidance appears fixed with relation to the window. The astronaut may observe that the landing point being indicated is in an area of unsatisfactory surface features; he can then select a new landing point for the computer-controlled landing. At any point in the landing, the astronaut can elect to take over partial or complete control of the vehicle.

The final approach phase ends as the spacecraft enters a hover phase near the lunar surface. The crew makes the final selection of the landing point and maneuvers to it either by tilting the vehicle or by operating the reaction jets for translation. Touchdown is made with the spacecraft near vertical and at avelocity of approximately 4 ft/sec, or less. The period on the moon includes considerable activity in exploration, equipment deployment, experimentation and sample gatherings. Also during this time, spacecraft systems are checked and prepared for the return. For example, the launches from the lunar surface can be initiated over a range of time by entering a holding orbit at low altitude until the phasing is proper for transfer to the command module. During ascent, the rendezvous radar makes direction and range measurements to the command module for refinement of the navigation data in the lunar module computer. The phasing of motion between the two vehicles eventually reaches a specific point from which a standard transfer burn will put the lunar module on an ascending trajectory to intercept the orbiting command module, During this period, radar measurements provide data for the lunar module computer's small velocity corrections needed to establish more accurately the intercept trajectory. The coasting continues between and during these corrections until the range to the command module is reduced to a few miles,

The terminal rendezvous phase — a series of braking thrust maneuvers under control of the lunar module GN&C systems and the astronauts — uses data from its inertial sensors and the rendezvous radar. This operation reduces the lunar module velocity relative to the command module to zero at a point near the CM. This leaves the lunar module pilot in a position to initiate a manual docking maneuver using the translation and rotation control of the reaction jets. After a final docking, the lunar module crew transfers to the command module, and the lunar module is jettisoned and abandoned (or made to impact the lunar surface on command from earth).

The guided transearth injection is made under the control of the primary GN&C system to put the spacecraft on a free-fall coast to satisfactory entry conditions near earth.

Ground-based or onboard navigation measurements may indicate possible midcourse correction maneuvers during transearth flight. The aimpoint of these corrections is the center of the safe earth-entry corridor suitable for the desired landing area.

An entry too high could lead to an uncontrolled skipout of the atmosphere; an entry too low could lead to atmospheric drag accelerations exceeding crew tolerances,

After the final safe-entry conditions are confirmed by navigation, the inertial platform is aligned or realigned, the service module is jettisoned, and the inertial entry attitude of the command module is achieved by GN&C system commands to the twelve reaction jets on the command module surface.

Safe reduction of highvelocity to suborbital conditions through the energy dissipation effect of the atmospheric drag forces is the first concern of the entry guidance. At lower velocity, controlling to the earth recovery landing area is included in the automatic guidance; manual entry maneuvers can also be used in a backup mode. This continues until velocity is reduced and position achieved for deployment of the drogue parachutes, Final letdown is normally by three parachutes to a water landing.

SECTION 3.0

FLIGHT EXPERIENCE SUMMARY

3.1 INTRODUCTION

As the culmination of eight years of intensive effort, two astronauts walked the "magnificent desolation" of the moon's Sea of Tranquility the night of Monday July 20, 1969. This first manned lunar landing of project APOLLO successfully accomplished an almost insurmountable technical and managerial task. Even the byproducts of this monumental project are considered significant advances in many technological disciplines. This section examines generally the onboard guidance, navigation, and control systems of the APOLLO command module and the lunar module spacecraft with emphasis on in-flight usage during the developmentalmissions leading to and including the first manned lunar landing. Later sections describe the systems from a design standpoint, and in more detail.

3.2 COMMAND MODULE SYSTEM

The guidance, navigation, and control (GN&C) system in the command module (CM) is shown in Figure 3-1. The onboard digital computer plays a central role in the system operation. It receives and transmits data and commands appropriately from and to the other components and subsystems shown. Starting clockwise at the bottom of Figure 3-1, the computer receives data and instructions from the ground by radio telemetry and sends back to the earth formats of data of interest for mission control. The astronaut with his hand controllers can command the computer to execute rotational and translational maneuvers. The inertial measurement unit (IMU) provides a measure of spacecraft attitude with respect to an inertial frame defined by the alignment of the inner gyro stabilized member. In addition, accelerometers on this stabilized member measure the linear- acceleration components being experienced by the spacecraft due to engine thrust and aerodynamic drag. Rigidly mounted to the base of the inertial measurement unit is the articulating optical subsystem that the astronautsusevisually to measure direction to stars for inertial unit alignment and to measure the present direction from the spacecraft tonavigation features of the earth and moon for determining spacecraft position and velocity.

During rendezvous exercises with the lunar module (LM) when the module is returning to the orbiting command module from the lunar surface, the communication system

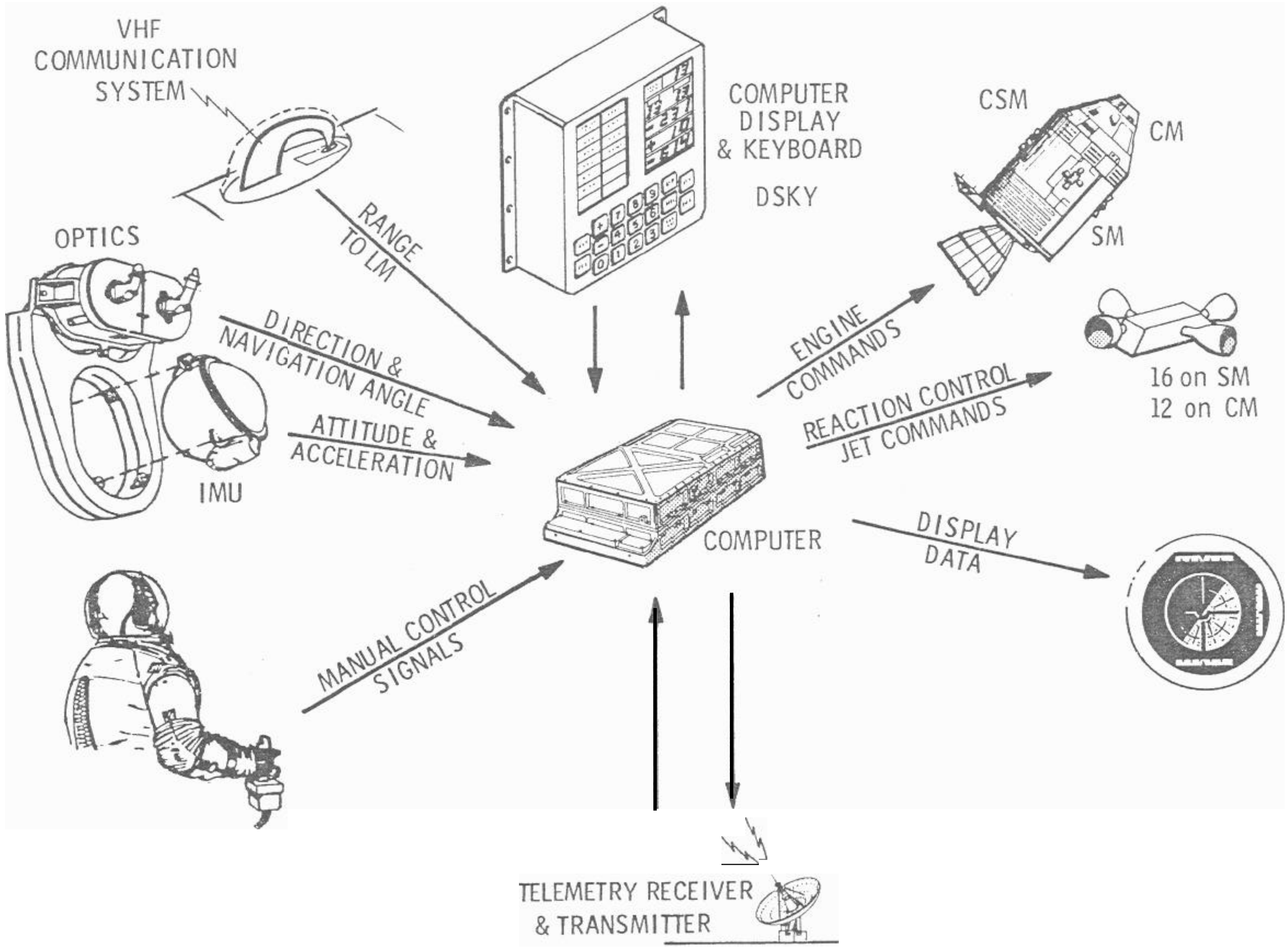


Fig. 3-1 , Navigation and

between the spacecrafts measures the range of the lunar module from the command module to help in the command module backup of the rendezvous.

The computer display and keyboard (DSKY) is the primary communication interface between the astronauts and the computer. By use of the keyboard the astronauts can call up the programs, routines, and displays desired and insert the data the computer needs. The numerical display presents data of interest so that the astronauts can monitor the progress and results of the computations.

During powered maneuvers the computer sends the main propulsion engine on and off signals, and commands the angles of the engine swivel gimbals. Each of the 16 reaction control jets on the service module (SM) and the 12 jets on the command module can be commanded on and off separately by signals from the computer so as to achieve spacecraft torque and translation force as required.

In the command module the only display from the system not on the DSKY is the total attitude and attitude error appearing on the main panel.

3.3 LUNAR MODULE SYSTEM

The system in the lunar module is shown in Figure 3-2. The inertial measurement unit, computer, and DSKY in the lunar module are identical to those in the command module except for the accelerometer scaling in the inertial measurement unit and the flight program in the computer. The lunar module optical system is a simple periscope for measuring star direction and thereby providing inertial alignment or realignment data.

During the lunar landing, the landing radar measures local altitude above the lunar terrain, altitude rate, and components of horizontal velocity for the computer's use in navigating and guiding the landing maneuvers. On the lunar surface and during rendezvous in orbit, the rendezvous radar tracks the command module orbiting above and provides the lunar module computer with direction, range, and range rate.

Besides commanding lunar module descent engine gimbal angle and engine thrust on and off signals, the computer also commands the thrust level to this throttleable engine in accordance with the guidance laws being used in the landing. The fixed thrust ascent engine is not gimbaled. Control torques during both the descent and ascent powered flight are provided by the 16 reaction control jets on the ascent stage in all three axes in addition to their use during non-powered flight for rotational and small translational maneuvers,

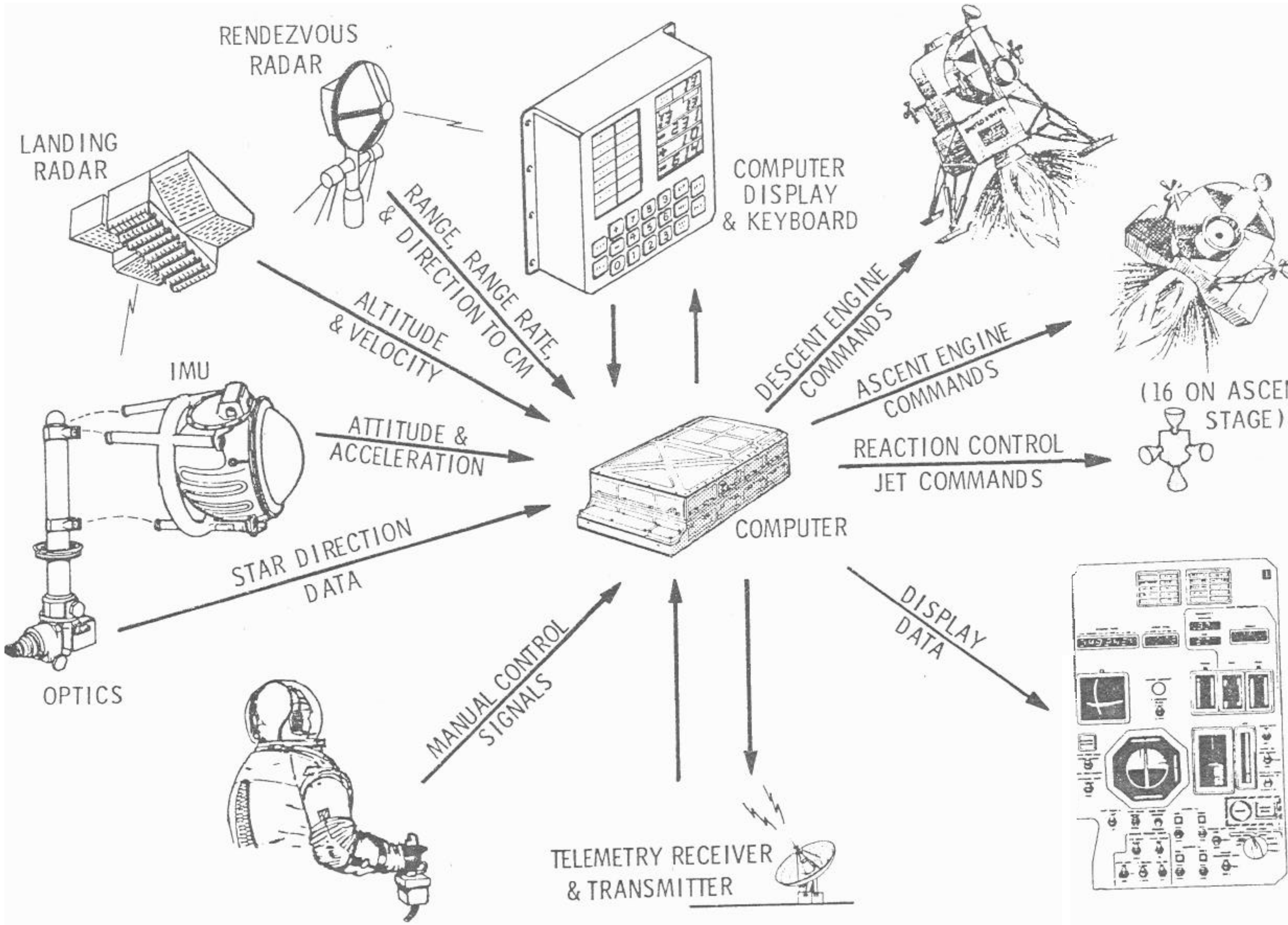


Fig. 3-2 Lunar Module Navigation and

System-driven displays in the lunar module include the keyboard display and the attitude and attitude error display similar to that in the command module. In addition, the lunar module computer drives displays of altitude, altitude rate, and horizontal velocity components during the landing, and displays of range and range rate to the command module during the ascent and rendezvous.

Through July 1969 the GN&C systems have supported nine flights of the command module and four flights of the lunar module as summarized in Table 3-I. In over 850 hours (35 days) of activeoperating experiencein space, the systems have never failed to support the flight plan.

3.4 INERTIAL MEASUREMENT UNIT

The inertialmeasurementunit (IMU) is a three-degree-of-freedom gimbaled platform isolating three single-degree-of-freedom gyros and three single-axis accelerometers from the spacecraft rotational motion. The orientation of the platform and the directions of the sensitive axes of the accelerometers are held non-rotating by the gyro error signals feeding the platform drive servos. The orientation is held to the attitude determined earlier by alignment to the stars. The inertial measurement unit provides the computer with information of spacecraft attitude by signal readout of the unit gimbal angles and accelerometer measurements of the linear motion arising from rocket propulsion or aerodynamic forces.

3.4.1 Gyro or Accelerometer Failure Prediction

Performance of the APOLLO inertial measurement unit is perhaps more accurate than needed to support the missions. The use of highly accurate gyros and accelerometers does result in efficient maneuvers and savings in propellant, but the more significant purpose for high-performance gyros and accelerometers is that they are more reliable, can degrade without severe mission penalty, and their degradation can be a forecast of outright failure before the system is committed to a mission.

During subsystem and spacecraft testing over many months prior to launch, the various parameters indicating the performance of the gyros and accelerometers are measured and their signature of normal performance noted. Based on these data, the flight computer is loaded with compensation values for 15 coefficients describing the non-perfect behavior of the inertial measurement unit. These coefficients are quite stable and repeatable in a good unit.

TABLE 3-I

PRIMARY NAVIGATION, GUIDANCE, AND CONTROL SYSTEMS (APOLLO FLIGHTS)

	Launch Date	Launch Vehicle		Mission
Apollo 3 CSM	25 Aug. 1966	AS-202	Unmanned	Ballistic Suborbital
Apollo 4 CSM	9 Nov. 1967	AS-501	Unmanned	Orbital, High Apogee, Simulated Lunar Return
Apollo 5 LM	22 Jan. 1968	AS-204	Unmanned	Lunar Module Earth Orbital
Apollo 6 CSM	4 Apr. 1968	AS-502	Unmanned	Orbital, High Apogee, Simulated Lunar Return
Apollo 7 CSM	11 Oct. 1968	AS-205	Schirra, Eisele, Cunningham	First Manned Earth Orbit Rendezvous to Booster
Apollo 8 CSM	21 Dec. 1968	AS-503	Borman, Lovell, Anders	Manned Lunar Orbit
Apollo 9 CSM/LM	3 Mar. 1968	AS-504	McDivitt, Scott, Schweickart	First Manned LM Earth Orbit Rendezvous to Booster
Apollo 10 CSM/LM	18 May 1969	AS-505	Stafford, Young, Cernan	LM Operations in Lunar Orbit Descent to 50 000 ft, No Landing
Apollo 11 CSM/LM	16 July 1969	AS-506	Armstrong, Collins, Aldrin	First Lunar Landing

3.4.2 Accelerometer Performance

During the long periods of free fall with no maneuvers, the output of the accelerometers is an excellent measure of their indication bias; i.e., the accelerometer output with zero input.

The flight of APOLLO 8 to the moon gave an excellent opportunity to watch this zero-input accelerometer performance because the inertial measurement unit was left running the whole 147 hours and, unlike earth orbital flights, the spacecraft stayed in sight of the telemetry receiver stations for much of the time. The data obtained are shown in Figure 3-3 as a function of the time when the check was made. In addition, the compensationvalues determined from prelaunch testing which were loaded into the computer are indicated. The data show the predictability and consistency of this accelerometer performance term.

3.4.3 Gyro Performance

The inertial measurement unit gyro-drift term under near- zero- acceleration environment can also be measured during flight by dividing the angle change that the unit's star-referenced realignment process requires by the time period since the last alignment. These results are displayed in Figure 3-4 for the APOLLO 8 flight. Also shown in this figure are the time intervals between the alignments upon which the drift computation is based. If this time period is short, the error in measured unit alignment angle change dominates and adds significant error to the indicated drift measurement. The roughness in drift data around 80 hours is due to the shorter period between alignment (once per lunar orbit or about 2 hours) so that alignment errors become significant. Assuming the true drift curves would be smooth, it is possible to infer the' actual alignment error that would cause the indicated roughness. In this case, we compute that inertial measurement unit star-alignment uncertainty during the 10 lunar orbits of APOLLO 8 had RMS values of 41, 31, and 58 arcseconds about the unit's X, Y, and Z axes respectively.

3.5 OPTICAL SUBSYSTEM

The optical subsystem in the command module consists of a two-line-of-sight, 28-power, narrow field-of-view sextant and a single-line-of-sight, unity-power, wide field-of-view scanning telescope both mounted on a rigid navigation base with the inertial measurement unit. The lunar module optical subsystem is a unity-power periscope mounted on the module's navigation base that supports the module's inertial measurement unit.

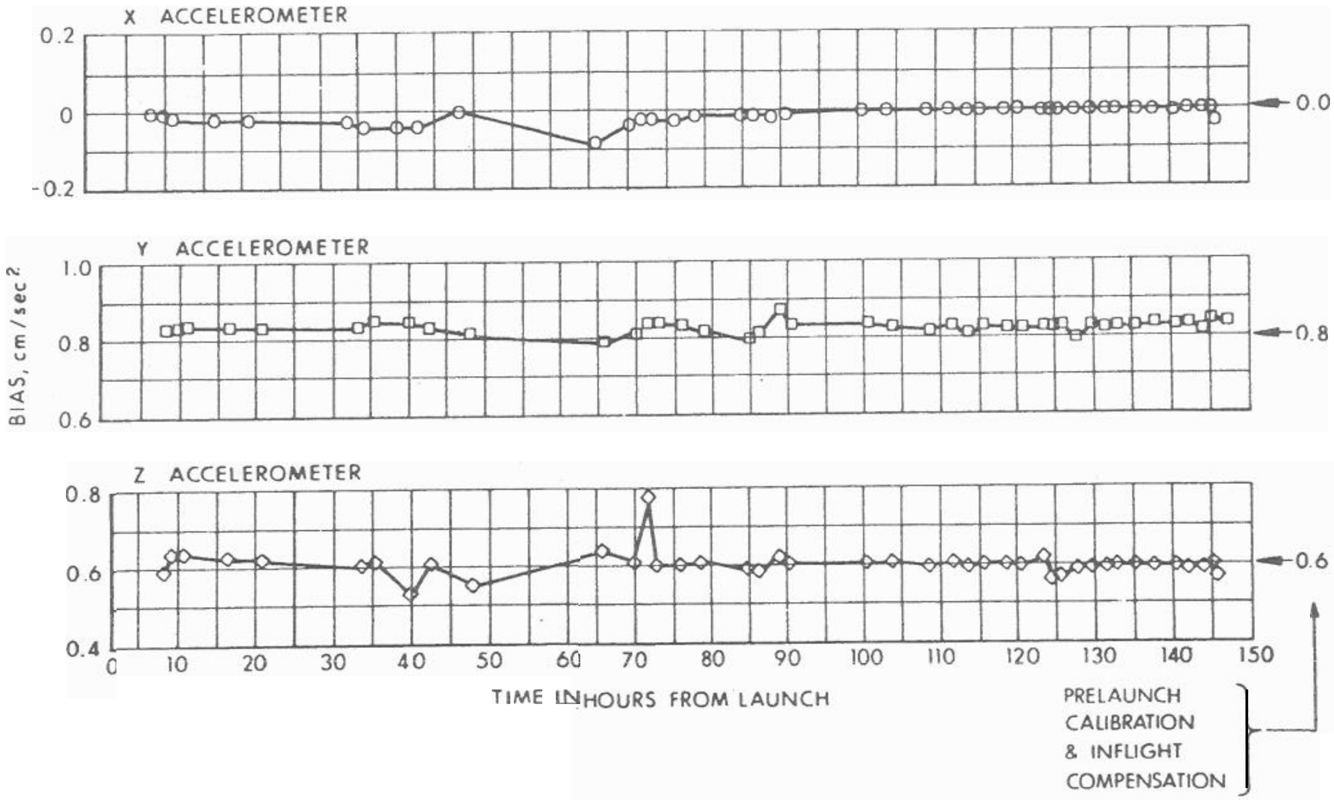


Fig. 3-3 Bias — Measured During 8

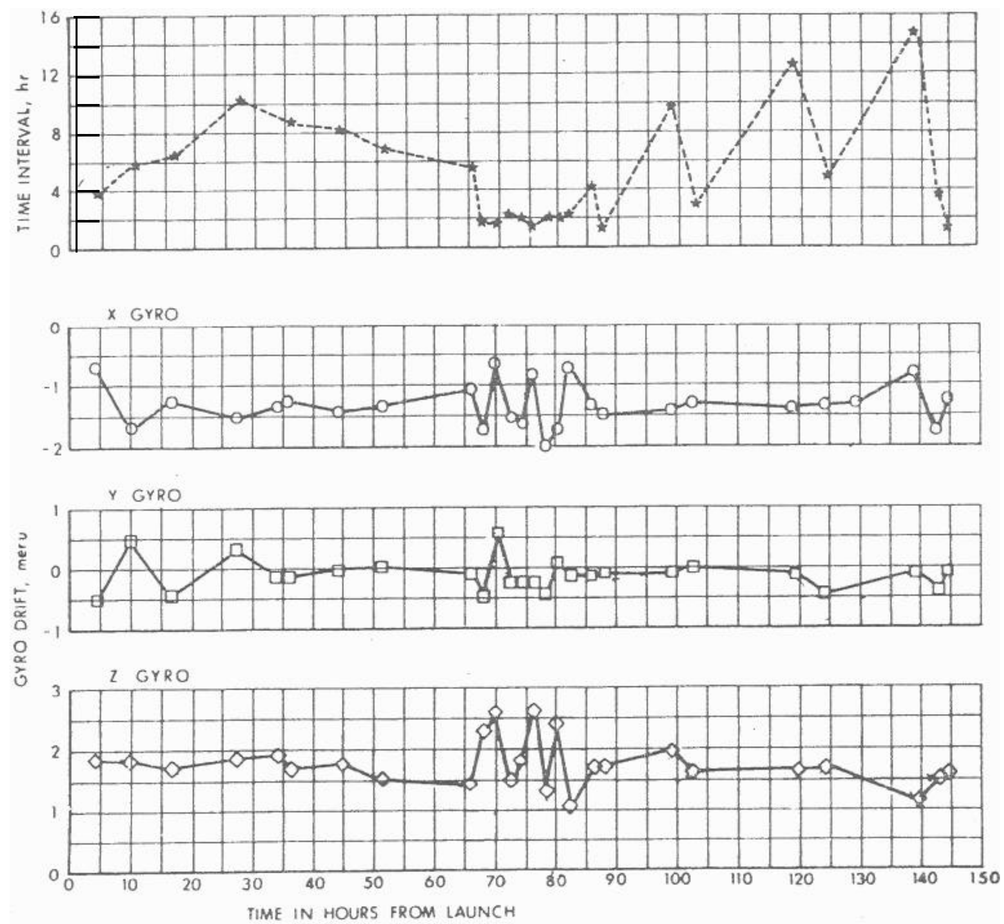


Fig. 3-4 Compensated Gyro Drift Measured During APOLLO 8

3.5.1 Star Visibility in Scanning Telescope

A number of physical constraints are imposed by the spacecraft that limit design freedom. Due to these constraints, the scanning telescope suffers both from light loss and a propensity to undesirable light scatter with associated washout of the background when the sun or other bright sources outside the field of view illuminate the objective. In addition, the necessarily wide field of view means, of course, that a random aim of the instrument is likely to find the sun, earth, moon, or another part of the spacecraft in view. These bright objects in view either prohibit sighting or severely degrade the eye's accommodation and ability to see stars.

Analysis and test forecast that even the brighter navigation stars would not bevisible from space through the scanning telescope within perhaps 70 degrees of the sun. Star visibility and constellation recognition would only be possible with the spacecraft oriented such that the optics were pointed away from the sun, shaded by the spacecraft, and with the sunlit earth or moon out of the field of view. The lunar trip of APOLLO 8 bore out this forecast. With dark adaption, it was always possible to find and identify constellations and stars in the scanning telescope as would be required in the initial step in an alignment of the inertial measurement unit from a random orientation. An unplanned test of this occurred. The loss of the inertial alignment on the way back from the moon due to a procedural error required recourse to the initial alignment computer program numbered P51. The scanning telescope was used to identify and acquire the needed stars and the inertial alignment was quickly accomplished.

APOLLO 8 flew to the moon without a lunar module and did not answer the question of star-visibility degradation caused by the lunar module docked to the command module. The lunar module would be a bright sunlit source partially in the field of view of the telescope. When the lunar module was taken to the moon on APOLLO 10, a favorable attitude with respect to the sun was accidentally discovered. Apparently, with the sun in line with the long axis of the docked configuration (and behind the lunar module), the lunar module shades the command module optics and puts the visible part of the lunar module in shadow. Constellations for star identification could be recognized in this configuration.

Of course, light scatter in low orbit is not a problem in the use of the scanning telescope with the spacecraft in the nighttime shadow of the earth or moon. Abundant stars are clearly visible. The other major use of the scanning telescope in earth or lunar orbit to track landmarks for navigation data does not suffer from problems of target visibility.

3.5.2 Star Recognition in Sextant and IMU Alignment to Stars

Besides its use in navigation, the sextant-articulating star line of sight makes the precision measurement of star direction for inertial measurement unit alignment. For each of the lunar missions, the unit operated continuously. Periodic realignment is performed with the sextant using the automatic star pointing acquisition of the computer realignment program, P52. Thenavigators have never reported any doubt that they had the correct star in the two-degree field of view, The proper star is the only bright star near the center of the field. For added confidence, after the astronaut centers and marks on the needed two stars in sequence, the computer program provides a check of the star-angle difference displayed to tine operator. This is the difference between the measured angle separating the two stars used for the alignment and that angle calculated from the computer's catalog of star coordinates for the identified stars. The average difference displayed was 0.007 degree (24 arcseconds) for the 30 alignments of the APOLLO 8 mission. Once the two star-direction measurements are made, the needed correction is applied to the inertialmeasurement unit orientation by the computer. Following this it is customary to make a further check by asking the computer to point the sextant to a particular third star. The requested third star came up in the center of the crosshairs to the satisfaction of the navigator every time,

The sextant does not suffer from the low light transmission and the light scatter problem of the scanning telescope. This was as expected. The narrow field of view, the use of a simple mirror rather than a complex prism to point the line of sight, the better light shielding possible, and the larger collecting aperture make the sextant's visibility of stars superior. It can be used for third magnitude stars within 15 degrees of the sun.

3.5.3 Particles and Deposits

Not forecast, however, was the loss in function caused by the debris cloud from the landing vehicle after separation. Confusion between local particles and stars has been so bad that the optics are unusable for star sighting until this debris cloud has dispersed.

During the design of the APOLLO optical-subsystems, there was much concern that the potential problem of deposits on the external glass surfaces would degrade optical performance. The threat would be from particles or film arising from the residue of reaction- control-jet firing or waste dumping into space. As flight time progressed, no degradation of visibility was observed in any instrument on any mission. A special examination was made late in the flight of APOLLO 7 by removing the eyepiece so that scatter or glare on the objective system from deposits would be visible. No glare existed. It appears that these surfaces will remain sufficiently clean in space for short missions at least.

3.5.4 Lunar Module Alignment Optical Telescope

The lunar module alignment telescope is provided with an exterior sun shield and baffles that reduce light scatter problems significantly. Lunar module inertial measurement unit alignment with stars has been demonstrated using this instrument both in orbit and while on the moon's surface. The design of user procedures for this instrument while on the surface had to account for executing sequentially the two degrees of freedom of star direction such that error from the moon's inertial rotation component was minimized.

3.5.5 Other Features of the Optics

The APOLLO 9 flight demonstrated other methods of inertial unit realignment using the optics. Data in the form of tables were carried aboard with which the celestial coordinates of planets could be obtained as a function of time in the mission. Command module inertial realignment was exercised using the coordinates of Jupiter as a non-catalog star manually inserted into the computer's memory. Automatic optics pointing was called and Jupiter appeared in view with a fine display of her moons. The navigator centered on Jupiter and then another star, with the program P52, in a demonstration of the use of a planet for inertial realignment.

A sun filter is provided to make it possible to use the sun as an alignment or realignment target and was exercised for the first time on APOLLO 9. The filter worked well functionally-even sunspots could be counted-but the actual realignment was not performed.

The automatic pointing capability of these optics to celestial, surface, or orbiting targets suggests other uses. A 16-mm movie camera was adapted to the sextant and carried on APOLLO 8. Motion picture film of the lunar surface was obtained with excellent resolution from orbit in this manner; however, program constraints in supporting the lunar landing have not yet allowed full exploitation of cameras on the optics.

Orbital navigation techniques which involve multiple sightings to landmarks of known or unknown coordinates were included in the flight program for use with the scanning telescope and/or sextant (Ref. E-1261, E-2262, R-482). The automatic optics pointing routine has proven accurate enough to maintain the selected landmark within the 0.9-degree radius of the sextant field of view. However, for low earth orbits, the apparent target rate was high enough to be troublesome in making the transition from automatic to manual modes for the fine sighting marks. A computer-contained, rate-aided pointing routine was recently added to facilitate the necessary transition.

3.6 COMPUTER SUBSYSTEM

Two identical APOLLO flight computers are used, one each in the command and lunar modules. The computer is the nucleus of the system in each spacecraft. Its computational capabilities are enormous for its size of only about one cubic foot and 70 pounds and for its operating power of approximately 70 Watts. Its necessary features include the ability to handle a number of different computational problems simultaneously in real time interleaved in a single central processor on a priority basis. Also of special note is the large number of signal interfaces with which it communicates with other systems throughout the spacecraft.

The crew displayed considerable skill in operating the computer using the display and keyboard. The dialog between astronaut and computer involves in each spacecraft some 30 to 40 major programs, approximately 25 routines, and a grammar involving approximately 28 regular verbs, 57 extendedverbs, and 93 nouns. Extensive exercises of these, in five manned missions, have had remarkably few procedural problems. In the few cases- of mistakes, the system has been reasonably tolerant and the crew have recovered without help.

The use of the system in supporting a manned or unmanned flight, an earth orbital or lunar flight, and a command or lunar module, is determined by the computer program written into its hard-wired nondestructible memory of over half a million bits. A total of nine significantly different flight programs have been developed. The specification, formulation, design, coding, testing, and documentation of these programs have been a major undertaking. The change-control activity alone has seen over 750 program change requests processed. Extensive testing of the programs is carried out at various facilities to certify that the program is sufficiently error-free for flight and that logical curiosities and anomalies not fixed have suitable workaround procedures designed, advertised, and that appear in the crew flight checklist when appropriate.

Flight experience with the computer has shown the value of the error detection and alarms built into the hardware and software programs and the memory protection offered by these features.

For many situations, the computer program associated with the keyboard is able to detect illogical inputs to the display due to mispunched keys. The display signals “operator error” so that the astronaut can try again.

Other error detection has been designed and experienced in flight that causes the computer to “restart” automatically. Restart causes the program to go back a few steps to a point where the computation state was saved in memory and then start fresh from that point. This is done so quickly that the astronaut is only aware that it has happened by the fact that the restart light or other alarm light comes on.

The most dramatic experience of this sort occurred during the later phases of the lunar landing of APOLLO 11. A heavy load was imposed on the computer due to an unexpected arrangement of mode switches associated with the rendezvous radar that is not used during this mission phase. Five times the computer displayed alarms indicating overload, causing restart and necessarily meaning that service to low-priority tasks was temporarily suspended. The important and necessary tasks were protected by the designed logic and the landing proceeded to the final semi-automatic maneuver by astronaut Armstrong away from a boulder strewn crater and to an excellent touchdown at Tranquility Base.

Serious internal logic problems or procedural errors during less critical times have raised alarms requiring the astronaut to manually recycle back to the start of the program in progress. In the few cases experienced, the crew have been able to recover without help. In such situations the Mission Control Center has examined the state of the memory to verify that memory corrections do not need to be telemeteredup. The important point is that the computer and its associated operating procedure are generally forgiving of the errors of the type experienced.

The use of a digital computer to tie together the measuring, processing, and commanding functions of the system has provided design flexibility of enormous value. Computer program changes, although often appearing trivial, can be unsafe without the time-consuming retesting to uncover any new errors introduced by the change. Changing the program to accommodate hardware or operational problems has nevertheless saved considerable time, effort, and expense.

3.7 CONTROL SYSTEMS-DIGITAL AUTOPILOTS

The attitude control systems of the APOLLO spacecrafts presented a design challenge. These autopilots had to consider many permutations of a number of variables. Among these are the variations in spacecraft configuration arising from combinations of docked and undocked flight of the command module, the service module, and the two stages of the lunar module. Next are the variations provided for achieving control torque with the arrays of ***small*** reaction control thrusters on the command module, the service module, and the ascent stage of the lunar module, and the gimbaled drive of the service and lunar modules descent stage main engines. Then there are the variations in the flight regimen from free-fall. coasting flight, rocket-powered accelerated flight, and aerodynamically influenced atmospheric entry. Finally, there are the widevariations in dynamic properties of each of the spacecraft configurations as fuel is expended, mass and inertia vary, bending frequencies and their damping vary, and fuel slosh modes and coupling vary.

The original configuration for these autopilots was a conventional analog system approach. But in 1964, the decision was made to incorporate these autopilots into the command and lunar module digital computers. A direct digital equivalent of the signal processing of the analog autopilot design candidates would not work. Sampling rates would have to be too high for the dataprocessing speed of the computer. Success depended upon new design approaches that capitalized upon the flexibility and nonlinear computations directly available in a digital computer. About 10 percent of the memory in the command and lunar module computers is devoted to autopilots. During times of high activity, only 20 to 30 percent of the available computation time is used in autopilot data processing.

3.7.1 Rate Derivation for Control

Essential to control systems is some form of stabilization signal. For analog autopilots this is obtained typically from angular rate-indicating gyroscopes mounted to the vehicle structure or other specialized sensors. The APOLLO primary system does not have sensors to measure angular rate directly. The digital autopilots derive spacecraft-attitude rate by processing available attitude signals. The simple ratio of inertial measurement unit-indicated attitude difference divided by time difference is grossly inadequate for the necessarily more complicated wide bandwidth control situations. The attitude signals come from the unit in quanta steps of about 0.01 degree, and at low rates no new information is available for rate indication until the next angle increment occurs. The system circumvents this problem by providing in the computer a model of the spacecraft response to applied torques. This model includes the torque level obtained from firing reaction control jets, the torque obtained from the thrust acting on the engine at the existing engine gimbal angle, and the presently existing spacecraft moments of inertia. As attitude jets are fired and engine gimbal angles change, this model can immediately change the estimated angular rate. Then, periodically, the integral of this rate estimation can be compared with the inertial unit orientation angles and the weighted difference applied to the state of the model to bring it back to consistency with the actual indicated time history of spacecraft orientation. This provides a fast-responding, low-noise indication of spacecraft rate without the need for special gyroscopes with their associated weight, power, and reliability penalties.

3.7.2 Autopilot Gain Scheduling

The computation capabilities of the digital computer have made feasible a very effective gain scheduling for the autopilots. The acceleration measured by the inertial unit during powered flight allows the computer to estimate spacecraft mass loss as propellant is expended. With this knowledge good estimates of spacecraft moments of inertia can be computed that give directly a good prediction of angular acceleration produced by a torque command. By using this to adjust the gains within the autopilot, the widevariations in dynamic characteristics of the spacecraft can be accommodated with near optimum response.

3.7.3 Lunar Module Descent Powered Flight Control

Of the various configurations, the lunar module powered flight autopilot during descent is the most interesting. It was originally intended to use a slow responding engine swivel to put the thrust vector through the center of gravity with a slow Computation loop. Dynamic control torques about all axes were to be provided by reaction-control-jet firings. Although the engine trim gimbal could be commanded at the fixed rate of only 0.2 deg/sec on each axis in response to discrete signals from the computer, the challenge to make this provide most, if not all, of the dynamic control was motivated by the reaction-control-jet fuel savings and the much smaller number of firings that would result. The challenge was ambitious because of the very slow speed with which the engine swivel could be made to effect changes in torque. A minimum impulse torque from the small jets is obtained within 15 milliseconds of the command while the same impulse requires 400 milliseconds with the engine gimbal, Nonlinear control laws were examined and a formulation for a third-order minimum-time control law was achieved (Ref. E-2450). This autopilot uses computer-generated estimates of attitude, attitude rate, and attitude acceleration every 0.1 second upon which to base a policy for commanding the appropriate axis of the trim gimbal in a time sequence of its three possible states: plus 0.2 deg/sec, zero, and minus 0.2 deg/ sec. This time sequence is such that the spacecraft angular acceleration, angular velocity, and angle error would theoretically all be brought to zero simultaneously in the fastest possible time. Provision was made to fire the reaction jets appropriately if the angle error got past a threshold, but in simulations this logic was rarely exercised.

The benefits of this nonlinear control law became particularly evident when it developed that the total time the downward firing jets could operate in the descent configuration was severely limited because of the danger of burning of the descent stage by the jet plume. Although using the lunar module descent engine to push the docked command module is considered as a backup for service-propulsion failure, this configuration was the only safe way of achieving the long descent-engine burn in APOLLO 9 to qualify the engine for manned operation. However, with the lunar module pushing the command module, another problem appeared. The lunar module forward firing jets were now severely limited due to their impingement on the docked command module. The solution was to have the computer cease all X-axis jet activity forward or aft when instructed and fly the lunar module control system pushing the command module entirely with the slow moving trim gimbal. This was how the descent engine docked burn of over 6 minutes duration and over 1700 ft/sec was controlled in the APOLLO 9 flight. The residual cross-axis velocity error at the end of the burn indicated only 0.1 ft/sec. Attitude error during the burn remained small and no large bending or slosh modes were seen.

3.8 RENDEZVOUS NAVIGATION

A particularly critical phase of the APOLLO mission is the rendezvous of the lunar module with the command module following the lunar module's ascent from the lunar surface. The navigation problem of the rendezvous is to measure the positions and velocities of the two spacecraft in a common frame and from this to determine the maneuvers required to bring the two spacecraft together in a prescribed fashion (Ref. E-21 76).

3.8.1 Rendezvous Measurements and Calculations

The active partner in the lunar landing mission rendezvous, the lunar module, makes its navigation measurements with the inertial measurement unit during maneuvers and with the rendezvous radar during coasting phases. The rendezvous radar, when successfully tracking the command module, measures directly the range and range rate for the computations. The radar antenna gimbal angles with respect to the spacecraft are sent to the computer, along with the spacecraft orientation measured by the inertial unit from which the computer can determine the direction to the command module in the stable coordinate frame, The estimates of radar angular misalignment and the position of one of the two vehicles are updated recursively in a Kalman optimum filter as radar measurements are incorporated periodically. The navigation states are used in several programs, in the computer to produce the targeting parameters for the various rendezvous maneuvers that are executed with the inertial guidance steering.

3.8.2 Lunar Module Active Rendezvous in APOLLO 9

The lunar module was active with the command module target in the earth orbital rendezvous on APOLLO 9. At least three major independent navigation processes were underway simultaneously: 1) in the lunar module, McDivitt and Schweickart wereusing the rendezvous radar, inertial unit, and the computer: 2) in the command module, Scott was using the optics, inertial unit, and computer: and 3) on the ground, mission control was using the tracking network to feed data to the Mission Space Center real-time computer complex. Initial separation of the two vehicles was followed by several maneuvers that brought the lunar module to the desired state: behind, below, and now catching up with the command module. The final critical transfer phase initiate (TPI) maneuver to put the lunar module on an intercept trajectory with the command module was then determined. The lunar module navigation and computation indicated (+19.4, +0.4, -9.7) ft/sec.* The command module navigation and computation indicated (+19.5, +0.5, -9.0). The ground tracking solution was (+19.6, +0.1, -10.5).

The transfer phase initiate maneuver was made by the lunar module with its own solution using the reaction control jets. During the 20-minute coast toward the command module, the intercept trajectory was improved by two small midcourse corrections determined onboard. The line-of-sight rates as the lunar module approached the command module indicated zero. The braking maneuver was done manually to the prescribed velocity step-versus-range schedule.

3.9 MIDCOURSE NAVIGATION

For onboard midcourse navigation between the earth and moon, the apparent positions of these bodies against the background of stars as seen from the spacecraft is measured by superimposing images in the dual line-of-sight sextant in the command module. The output of the sextant is the measured angle between a known reference star and a visual navigation feature of the earth or moon.

To make a navigation sighting, the midcourse navigation program in the computer, P23, can point the sextant's two lines of sight at the reference star and a landmark identified by the navigator. It is then the navigator's task to center the superimposed star image onto the landmark, if landmarks are being used, or onto the near or far substellar point of the horizon, if the horizon target is being used. When superposition

*X, Y, and Z velocity change components; X horizontal in orbital plane in direction of orbital velocity. Z vertical down, and Y completes the right-handed set.

is achieved, the navigator pushes the MARK button which signals the computer to record the navigation angle and the time of mark. Using a Kalman optimum recursive filter formulation, the computer then determines the state-vector change this measurement would cause if incorporated and displays the resultant position and velocity change magnitudes, If the navigator is satisfied with the display and is satisfied subjectively with his mark, he allows the computer to incorporate the state-vector change.

The first manned trip to the vicinity of the moon of APOLLO 8 during Christmas 1968 gave an excellent test of the APOLLO system's onboard navigation capability. Although ground tracking navigation was the primary system, the onboard navigation system had the task of confirming a safe trajectory and providing a backup for return to earth in the remote chance that ground assistance became unavailable for onboard use.

The navigator made 51 sightings in four groups using the earth's horizon, followed by 30 sightings in two groups to the moon's horizon. The period spanning the last group of earth sightings and the first group of lunar sightings is most interesting. Figure 3-5 shows the variation in predicted perilune passage altitude for this period. The relatively poor performance for the earth sightings arises from the great distance -about 140 000 nautical miles at this time. The effect on the quality of the perilune prediction caused by incorporating the first few lunar sightings is dramatically evident. Also shown are two displays available to the navigator. The NOUN 99 display gives the computer's estimate of the RMS error in the present state vector estimate. It is seen that when the perilune estimate was not too good, the computer was indicating its uncertainty. The NOUN 49 display gives the magnitude of the change in present state vector that is caused by incorporating the sighting. As would be expected, the first few of these for the lunar sightings resulted in fairly large changes in the estimated state vector while the remaining had a very small effect. At the end of this group of measurements, the indicated perilune was 67.1 nautical miles.

The final set of 15 translunar sightings was made about 30 000 nautical miles from the moon with little additional effect on the peril.une estimation since it was now quite accurate, The final indication was 67.5 nautical miles or about 1.3 nautical miles lower than a value later reconstructed from ground tracking data.

3.10 CONCLUSION

Only some of the salient aspects of the GN&C systems performance on the manned APOLLO missions were emphasized in discussing flight experience to date. This

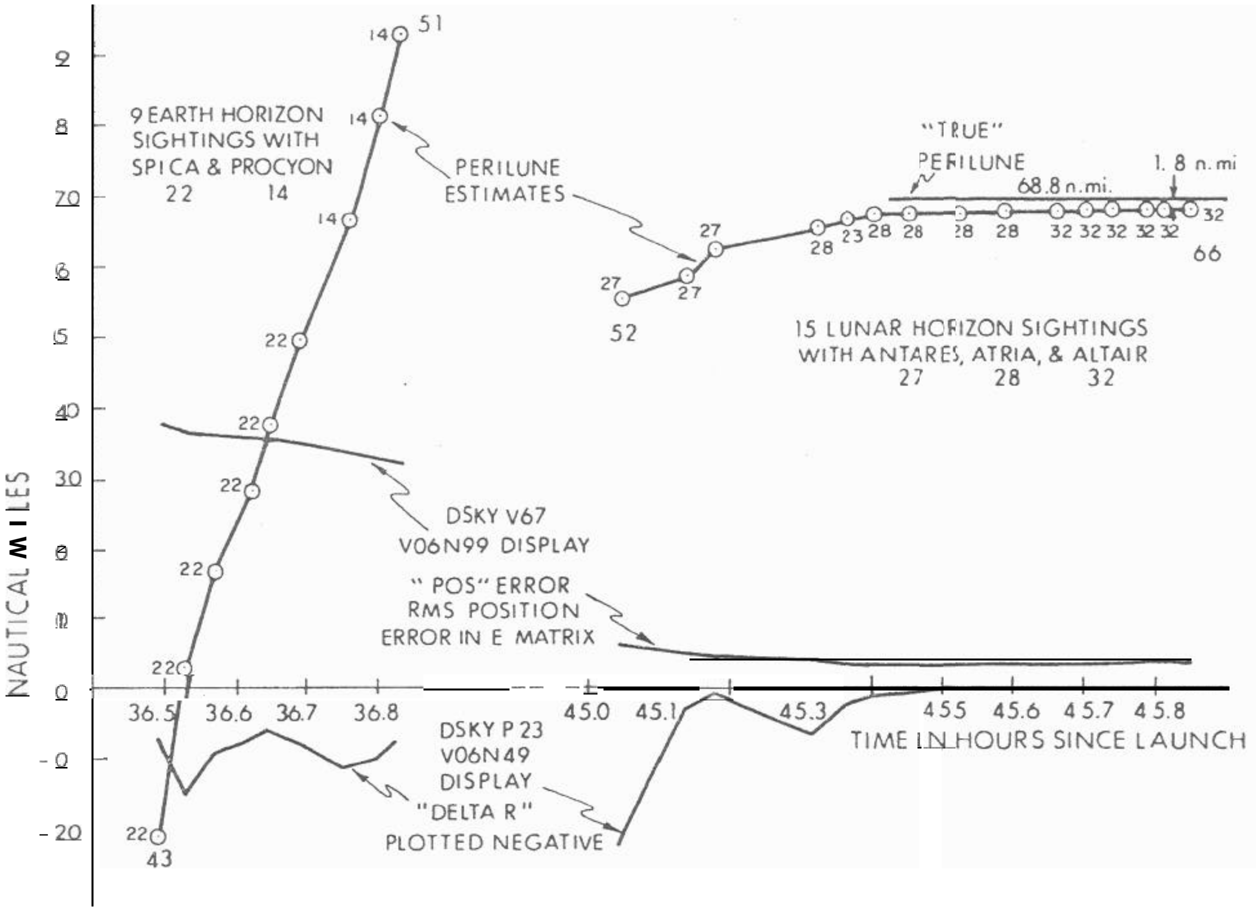


Fig. 3-5 APOLLO 8 Midcourse Navigation

is just one part of the total technical challenge that has been met; the first manned landing on the moon. Unprecedented opportunities for scientific exploration of the moon are now available.

Section 4 which follows reveals some of the background that led to the configuration of the GN&C system as used on APOLLO missions.

SECTION 4.0
DESIGN OBJECTIVES AND LIMITING FACTORS

4.1 BACKGROUND INFORMATION

Thus far Chapter II has dealt with the general problem of spacecraft guidance, navigation, and control followed by a synopsis of the APOLLO mission requirements, and then the flight experiences gained up to and including the first lunar landing mission. The remainder of this chapter describes development of the GN&C systems to implement the APOLLO missions from inception to the successful first lunar landing.

This section treats three related topics at a systems level: the character of the early GN&C systems design objectives;[1] the limiting factors of the design which evolved,[2] and the factors that had a limiting influence on the developmental efforts. These system level topics are discussed before the hardware in order to provide increased insight for the reader as the level of descriptive detail increases.

4.1.1 Scope of the Original Project Objectives

The original design objectives centered on the concept that the primary capability for spacecraft guidance and navigation would be provided by a self-contained onboard system. Ground-based facilities would provide backup support or a secondary capability. Hence, the first objective was to design a reliable, autonomous system that depended on neither the deep-space radar technology nor the telemetry communications capabilities of the early 60's.

The scope of the original subset of goals to meet the primary objective above ranged from developing a basic understanding of the phenomena used in onboard guidance and navigation methods to providing the equipment for space flight use. Between these limits were many areas of guidance concept development, error analyses, technique development, component design, testing, and so forth, It was recognized

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1. MIT/IL Letter and Enclosure, C-577, Robert G. Chilton, NASA Space Task Group, from Milton B. Trageser, 27 July 1961.

2. MIT/IL Document E-2119, by APOLLO Staff, "Critique of the APOLLO Guidance System Design," May 1967.

that the responsibilities would require coordination with the various NASA centers and with APOLLO contractors who had not yet been selected when the MIT/ IL proposal was written in mid-1961.

4.2 TECHNICAL APPROACH

4.2.1 Midcourse Guidance, Navigation, and Control

The attainment of autonomy for onboard guidance and navigation, and the concept that it would be primary for both functions, necessitated a major effort by MIT/IL to develop the methods and instrumentation with which star-to-landmark and star-to-horizon angle measurements could be made. A sextant to provide this precision measurement to an accuracy of 10 arcseconds was envisioned. Parenthetically, however, the earth-based facilities became primary for the navigation functions, circa 1965, with the onboard system as backup. By this time, however, the onboard sensor capability had been developed and the onboard navigation capability was well along in design.

The early approach considered that star-to-landmark angle measurements would provide more accurate data for midcourse navigational computations than would star-to-horizon measurements. Later, studies by MIT/IL demonstrated that a dearth of usable landmark targets could occur during cis-lunar flight due to combined factors of cloud cover, stringent landmark lighting requirements, limitations on slant angle to targets, and so on. The star-horizon method did prove usable in midcourse flight as the increased range to the horizon made haziness of the horizon less significant in the measurement of subtended angle and as sighting techniques were improved through flight experience. (Refer to Section 3 of this chapter.)

An early objective to automate the star-to-horizon angle measurements through a star tracker and horizon photometer was not implemented on APOLLO because of cost and schedule limitations. The concept, however, does have operational merit and should be considered for future space missions. (See Volume II on Candidate Subsystems.)

It was recognized from the start that the mechanization of three simultaneous star near-body sightings for unambiguous position determination would be impractical. A technique was therefore developed whereby the onboard guidance computer utilizes sequential sextant star-to-landmark measurements in the navigational process. Optimumuse of the data is made by the appropriate weighting of each measurement with regard to its accuracy and effect on the predicted position uncertainty of the spacecraft.

A single general purpose digital computer was designed to provide the variety of functions that were required for the overall mission. The construction of a computer of reasonable size and complexity was considered feasible through the use of a compact non-erasable core rope memory unit. The qualitative criteria demanded of the computer development were high reliability, low power consumption, reasonably high speed, versatility, and an input-output suitable for interfacing directly with spacecraft and other guidance and control subsystems. Originally the computer used magnetic cores for the logic and the erasable storage. Later the direction of development was changed to the standardized micrologic NOR gate implemented in the present computers. (See Volume III on Computer Subsystem.)

In 1961 the scope of the software development effort was not understood well enough to determine the flight computer memory needs. The storage requirements were accordingly grossly underestimated. The total memory of about 4000 words fixed and 250 words erasable in the early computers had been deemed adequate for the simple tasks envisioned. As the GN&C system development progressed, the versatility and capabilities inherent in the computer design were exploited, Additional tasks were heaped upon the computer seemingly without end. It soon became evident that the storage capacity of the BLOCK I computer was woefully inadequate.

The Block II computer design was begun with a clearer appreciation of the magnitude of the programing requirements. Many diagnostic, computational, and moding functions, which had previously been done by the crew or other parts of the GN&C, could be done more efficiently by the computer. A major challenge was undertaken to design a digital autopilot into the flight program assembly. For these tasks, the Block II memory was sized at about 39,000 words. At that, the Software Control Board, which was established to direct the design and allocate resources, was plagued by continual trade-offs required to keep from exceeding the capacity of the machine. Future computer design efforts could well draw on the experience gained.

The feasibility of a self- contained guidance and navigational capability was predicated on the availability of a precision onboard inertial sensor with which to measure spacecraft motion against a known reference frame.

In late 1961, plans were to fly an APOLLO Spacecraft in the fall of 1963 with an MIT/IL guidance and navigation system onboard. The NASA direction was to build that which we knew how to build in the time allotted, and to use only components proven by production and operational techniques. This, along with the efforts to maximize reliability and operational flexibility while minimizing weight, power requirements, cost, etc., and above all the overriding considerations of schedule, forced many far-reaching decisions.

A fundamental decision was the selection of a three-gimbal, three-degree-of-freedom design for the inertial measurement unit. The operational convenience of an all-attitude inertial unit favored the choice of a four-gimbal or body-mounted (strapdown) inertial unit to avoid gimbal lock. However the flight plans at the time and the fact that inflight platform realignment would be done in any event made the avoidance of gimbal lock straightforward. The three-gimbal inertial unit would allow a simpler, smaller, lighter, more reliable, and less power-consuming system.

To conserve electrical power, the computer, inertial, and optical subsystems could be placed in standby or shut down completely when not in use.

The power conservation problem was not nearly as limiting a factor on spacecraft operations as originally thought and it became possible to leave the inertial measurement unit on and aligned throughout an entire mission. Thus, spacecraft control functions such as continual attitude maneuvering for passive thermal control and wide-band attitude stabilization could be implemented during crew sleep periods.

4.2.2 Lunar Landing

In 1961 the methods which would be used to acquire navigational data, especially during the lunar descent and landing, were at best speculative. Schemes to utilize optical range finders, visual tracking of beacons, inertial measurements, and radars were considered as candidates. To support the development schedule, the GN&C system was designed to be capable of interfacing with a largevariety of data sources: thus, when the spacecraft designs were finally chosen, the impact on GN&C was minimal.

A few examples of the importance of the original objective of designing a flexible guidance and navigation approach are:

1. Very accurate inertial measurement unit realignment inputs preparatory for inertial guidance to the preselected landing site.
2. The ability to accept state vector updates from earth-based radar and computers that have propagated the data several orbits ahead even with significant orbit perturbations caused by the very recently discovered lunar gravitational anomalies.
3. The ability to retarget the onboard guidance from up-telemetry with new earth-based computational tools such as the Lear Processor.
4. The ability to accept final, partial or complete guidance commands from the astronaut.

4.2.3 Lunar Launch, Transearth Injection, Atmospheric Entry Guidance

A major objective of the APOLLO GN&C developmentwas to achieve a proper balance between the fully automatic and the fully manual system (Ref. R-411). This man-in-the-loop concept was aimed at careful integration of the man-machine interface in order to maximize the performance of the total system. The functions which the man performs best-interpretation and decision making-are assigned to him while the high speed iterative capabilities of the computer are exploited for the required repetitive calculations.

4.3 LIMITING DESIGN FACTORS

4.3.1 GN&C System

The following paragraphs describe some of the limiting factors of the design based on experience gained and with recognition that the experience is valuable for design of similar but more advanced guidance, navigation, and control systems for future space missions.

It should be emphasized that most design decisions are made as an engineering judgment compromise among conflicting constraints. Besides the principal motive of achieving required performance within the operational environment, the impact of schedule, weight, reliability, cost, power drain, producibility, etc., were all considered.

The one constraint which weighed most heavily in the design of the subject equipment was that of schedule-the urgent quick release of the design to production. Now in retrospect, this constraint does not appear so compelling. More time should have been allowed in the design schedule to allow for the test of breadboard and prototype models for their performance in the expected environment. Unfortunately, much of this testing occurred after production was too far along to make indicated design changes without too much expense. For this reason the changes were not made unless it could be demonstrated that the existing design simply would not do the intended job.

Many of the criticisms listed herein are, then, of the category of those which have been known for some time based upon test and early experience, but discovered too late to affect the designs already in production. They are no less instructive, though, in guiding the effort of design on future systems, Other criticisms may relate to some characteristic of the design that suffered in the design compromise approach necessary. The other characteristics which would suffer by fixing the indicated feature criticized are generally evident and are not pointed out. It is deliberately not the purpose of this section however, to relate the causitive factors or rationale in each of the features criticized.

Restrictions on vehicle motion to avoid gimbal lock resulted in increased reaction control system (RCS) propellant consumption during some attitude maneuvers, and, more particularly, increased the amount of software in an already crowded computer. In this later technological time frame, all-attitude inertial subsystems have become a requirement for future spacecraft with both strapdown and four-gimbal types under active consideration.

Difficulties from interference were experienced with internal electrical interfaces in pulse circuits without transformer coupling at both ends of the line. Similar problems were experienced with line switching at low logic levels. Ground interference pickup could have been reduced by the use of isolated ground planes. Some power supply specifications might have been tightened; for example, gyro power wheel supply and microsyn excitation. A more efficient power distribution system could have been devised using pre-regulators. These regulators could have provided more suitable and varied voltage levels that would, in turn, require fewer total component parts and less total power.

There were internal mechanical difficulties experienced with the system of assembly in the spacecraft. Precision alignment has to be maintained under conditions of limited accessibility. It would have been preferable not to have a separate structural navigation base, but to design the system as an integral structure.

It wasnecessary to dissassemble parts of the spacecraft (CSM) to remove the main display and keyboard. Access to units within the spacecraft is a general external mechanical problem. In retrospect, partial retainment of the early design objective for inflight maintenance of the system could have resulted in easier replacement of subsystem elements at least during preflight spacecraft testing.

Integral cooling of the electronic packages would have simplified the thermal design problem. Separate cold plates created packaging design inefficiencies and resulted in structural mounting problems. The cold plate interface restriction and spacecraft mounting geography resulted in cable routing and design problems as well as poor connection accessibility. The lack of accessibility resulted in a connector hazard during spacecraft installation. A possible alternative approach might be to define a gross electronic package envelope interface in the spacecraft with primary structure tie-down points. Packaging allocations, structure, and interconnections could then be defined to minimize external 'harnessing, accessibility, and cooling problems. This would also minimize package scattering and associated weight penalties.

4.3.2 System Fabrication

4.3.2.1 Power Supply Protection

There is no internal protection of power supplies against failure conditions. Although operating experience on the system did not indicate that the present protection is inadequate, internal protection of power supplies would enhance reliability.

4.3.2.2 Signal Conditioning

Signal conditioning should be internal to subsystems. Timely decisions on the measurement requirements would have allowed incorporation of signal conditioning in the power and servo assembly without the schedule difficulties experienced with the present units.

4.3.3 Testing

4.3.3.1 Coupling Data Unit

In order to meet the needs of inertial component performance and alignment testing, angular readout with higher accuracy and smaller quantization would be desirable.

4.3.3.2 Inertial Components

To meet the needs of testing, reduced command module accelerometer quantization would be desirable, Better signal generator output test points should be available. Better gyro drift performance would permit more time between alignment and use of the inertial measurement unit which would benefit the crew significantly. Smaller quantization would also permit steering via the reaction control system and efficient bias pulse rejection during free fall.

4.3.3.3 Axis Offset

The offset of the X-IMU axis from the spacecraft axis in the Block I design approach was desirable from the point of view of testing, The ability to put each of the platform axes vertical in the Fine Align mode made inertial component testing in the spacecraft possible. A fourth degree of gimbal freedom would have also provided this capability.

4.3.3.4 Diagnostics

The Block I failure indicator lights should have been retained or a substitute provided in Block 11 to aid in quick diagnosis of problems during checkout. Additional status bits for failures in the computer would have been useful.

4.3.3.5 Prototypes

The use of prototype (and mock-up) hardware prior to actual commitment to a production configuration would have appreciably lessened the production changes. Prototype hardware should be fabricated and tested prior to production hardware.

4.3.4 Crew Displays and Controls

4.3.4.1 Interfaces

The computer function and interface inmodingvarious subsystems should be enlarged. Crew activity should be limited to program selection rather than the effort in actually setting devices for GN&C system activities. (Displays and controls for manual backup modes would remain.)

4.3.4.2 Repetitive Functions

Mechanization with a single button would be appropriate for repetitive functions such as program selection, instructions to proceed, etc.

4.3.4.3 Inputs

The input data program should assume that all data are positive decimal unless the data are preceded by an identifier for octal data or a negative sign for negative decimal data. Displays should contain a floating decimal point identifier to simplify their use.

4.3.4.4 Identifiers

The capacity of data identifiers (nouns) needs to be greatly enlarged. Also, special identifiers are needed for such things as checklist codes. (Block II uses a verb-noun combination plus an R1 code that usurps the R1 register.)

4.3.4.5 DSKY Pushbuttons

Limitations on presently available hermetically sealed switching elements have resulted in pushbuttons with excessive actuation forces. Solid state switches need to be developed to eliminate this design limitation.

4.3.4.6 Lights

Optical reticle lights should be controlled by their own unique controller-not like Block II where the dimmer dims the optical angle readouts, as well as the reticles.

4.3.4.7 Data Package

The onboard data package is very large, even for a simple earth orbital mission. More complex missions will multiply the carry-on data package many-fold. It therefore appears that a low weight, medium access speed, bulk data storage device is needed.

4.3.5 Crew Training

Experience has shown that a need exists for the formal training of APOLLO crew members on a systematic basis. The significance of a good GN&C systems background for crew members highlights the need for recognizing training as a specific contract task. Inherent in including a training task is the coordination of GN&C training with all other crew training.

4.3.6 Reliability

The use of solid state switching for optics moding, certain moding in the inertial subsystem, and in the computer DSKY would have enhanced system reliability.

(Other reliability considerations are delineated within the following specific subsystem sections of this report.)

4.3.7 Computer Subsystem

4.3.7.1 Memory

Perhaps the most obvious limitation of the computer with respect to its employment in lunar missions is the lack of greater program memory capacity. Program memory usage tends to consume all that is available to it, however, and it would be only a matter of time before the same lack would be felt. Nevertheless, this overall problem is one that will need to be solved in the next generation of design.

Another important item is the amount of time and cost required to produce program ropes. Turnaround time is being shortened, but faster and less expensive manufacture will be an important consideration in the future. Meanwhile, the rope memory offers high density and reliability of program storage.

The rope's long memory cycle time is the chief limitation of the computer's speed. Although there is no serious speed deficiency, there have been many instances where greater speed would have simplified programing; hence a faster fixed memory would be desirable.

4.3.7.2 Logic

Several of the logical features of the machine, designed to minimize its physical size, have been deleterious to the programing effort. A word length of 20 bits or more instead of the present 16 would have simplified the address structure, scaling, and certain interfaces, and would also have given greater precision in arithmetic operations. This might have cost an extra memory cycle time in multiplication but would probably still be desirable. The primary conclusion here is to furnish adequate address fields and arithmetic precision even at a sacrifice in size. The apparent disadvantage in speed will be outweighed by the advantages.

Scaling of problem variables has been a burdensome analytic chore for mission programers. A floating point arithmetic unit would be well worth its cost.

A larger, more straightforward instruction set would be possible with increased word length, and would permit new programers to learn the order code faster. The guidance computer instructions are tricky and meant to be used by experts, which has been bad economy. The independent sign representation used in double precision operations has been awkward for programers. The interrupt structure would benefit from a revision giving fast service to certain critical interrupt functions such as RCS jet shutoff and down-telemetry data. Provision for interrupt specification as part of the program memory load or under dynamic program control rather than in the hardware would enhance the usefulness of the interrupt mechanism. More attention should be given to the ground operations dependent on the guidance computer. For example, addition of a few extra bits to each memory register in core rope simulators with corresponding additions to the logic could provide an extremely valuable diagnostic aid for program debugging.

4.3.7.3 Input-Output

There has been some feeling that the computer's interfaces have lacked flexibility. They were designed for specific interface functions with some slight excess of inbits and outbits. Actually, it would be extremely difficult to generalize the computer's interface, but it is a valid goal for the next generation.

In retrospect it seems that the computer would have benefited from greater built-in manual intervention and diagnosis capability. Replacing the separate alarm indicators of Block I with a single indicator in Block II was a step in the wrong direction from the point of view of diagnostics. There should at least be provision for malfunction diagnosis on board the spacecraft, perhaps via the test connector with carry-on equipment. The Block II AGC test connector is inaccessible in the command module.

4.3.7.4 Software

A more sophisticated computer-aided programing package might have lessened the mission programing burden. This has always been considered, but requires a magnitude of effort that has never been deemed worth its cost, particularly because of the storage inefficiencies that would result. This is an important area of development for the future.

4.3.7.5 Power Supply

Standby switching in the Block 11 power supply is done by relays to avoid the power dissipation of the electronic equivalent. Interference caused by uncontrolled voltage and current discontinuities have been enough of a problem to warrant using electronic switching.

4.3.7.6 Logic Electrical Design

The Block II NOR gate achieves low power dissipation at the expense of speed. A few instances have been uncovered in which the timing margin is not as large as desired. This is largely due to the fact that the problem occurred only in potted production computers available for test after design changes become difficult to phase in. This points up the need for fast turnaround between design and production and for a comprehensive test program for early production computers.

Signal interference has been observed as a result of wire and module placement in the trays. More careful attention will have to be paid in the future to this problem.

Interface and power signals are the worst offenders, and might well be isolated. The transmission characteristics of interconnections will need better control, as is obtainable in multilayer boards as opposed to wire wrap. The computer aspects of electromagnetic compatibility practice are in need of r-e-examination. Case grounding should be used where appropriate, for example.

4.3.7.7 Memory Electrical Design

Rope memory electronics have been troublesome in Block II, having necessitated several design changes. This type of specialized and complex circuit should have had more comprehensive development design.

4.3.7.8 DSKY Electrical Design

The electroluminescent display lights and the relays that drive them have been troublesome out of proportion to their legibility and power saving advantages. Reliability is the main problem, notably in vibration and probably in the zero-g environment. Advances in incandescent segmented displays driven by semiconductor elements make them look attractive for this application.

4.3.8 Optics Subsystem

4.3.8.1 Automatic Sensors

During the return from the moon, it is difficult to find suitable earth landmarks. The earth's horizon provides a suitable reference. The horizon photometer permits an accurate determination of the horizon reference attitude, which is several times better than can be done manually. The photometer would also permit onboard earth reference during unmanned flights.

The star tracker would be a considerable aid to the astronaut during navigation measurements, During unmanned flights it would permit automatic onboard navigation and automatic inertial measurement unit realignment.

4.3.8.2 Optics Control System Design

Residual torques in optics unit assembly components (flex-print, resolver leads, etc.) have caused optics assembly problems by moving the line-of-sights when no input commands have been applied to the optics servos. An increase in friction load should eliminate this kind of problem.

4.3.8.3 Optical Design

A factor of three improvement in the telescope resolution can be realized by optimal optical design. The present design fulfills APOLLO requirements, but the resolution improvement would also improve light transmission and light scatter characteristics.

4.3.8.4. Mechanical Design

An optics door on the spacecraft would improve optics performance, reliability, and durability. The door should be designed as an integral part of the optics subsystem to maximize its performance in conjunction with the optics.

4.3.8.5 Light Scatter

The general utility of the optics as a navigation tool is somewhat limited by the scattered light environment.

4.3.9 Inertial Subsystem

4.3.9.1 Accelerometer Design

Relative to the availability of current technology, had the phase-modulated accelerometer development maturity been attained at the date of APOLLO design commitment, it would have been an obvious instrument choice, A single large stainless steel bellows would have proved more desirable. The improved sealing techniques incorporated in the phase-modulated accelerometer design would have eliminated a major APOLLO accelerometer failure cause.

4.3.9.2 Electronic Design

The multiplex configuration represents the next obvious step in design maturity of the coupling data unit.

4.3.9.3 Thermal Control System

The inertial measurement unit thermal control system would have been enhanced if a helium fill and coolant bypass could have been developed. The use of helium would have reduced the blower duty cycle and perhaps even have eliminated the requirement for the blowers. A coolant bypass for use in standby operation modes would have significantly reduced the standby power requirements. A fail-safe bypass technique could have been developed.

SECTION 5.0
GN&C SYSTEM DESCRIPTION-DESIGN DECISIONS

Having now discussed the guidance, navigation, and control systems from several viewpoints, the general problem, the specific APOLLO missions problem, the early objectives and later critique, this section goes on to describe the systems design alternatives which were visualized and those implemented.

The choice of sensors and data processors for guidance, navigation, and control used in APOLLO is governed by the nature of the spacecraft and the diverse demands of its mission. First, although full use is made of all earth-based help, the spacecraft systems are designed to complete the mission and return without the use of earth-based tracking data or computation support to the extent that onboard computer programs provide. For safety,. this provides protection against critical lack of earth coverage or communication failure. However, earth-based data are available most of the time with support from the onboard equipment measurements.

To provide the required flexibility, identical inertial and computer subsystems are used in the two independent systems controlling the command and lunar modules.

5.1 INERTIAL MEASUREMENT SUBSYSTEM

The choice of inertial guidance over radio command guidance can be justified most dramatically by the velocity change maneuvers that must occur in back of the moon. The guidance measurements for the lunar orbit insertion and escape maneuvers out of sight of the earth must be made by onboard sensors.

Inertial guidance measurements might be made in either of two major configurations: gyro-stabilized gimbal-mounted platform, or vehicle frame-mounted sensors. Each has advantages.

The gyro-stabilized gimbaled platform has had many years of successful use in guidance of military ballistic missiles. Its gyros and accelerometers are kept inertially non-rotating by the isolation provided by the gimbals and their stabilization servos. Its outputs are in a convenient form: vehicle attitude Euler angles appear directly as the angles of the gimbals; acceleration measurements appear directly as components in the non- rotating coordinate frame of the stable member “platform.”

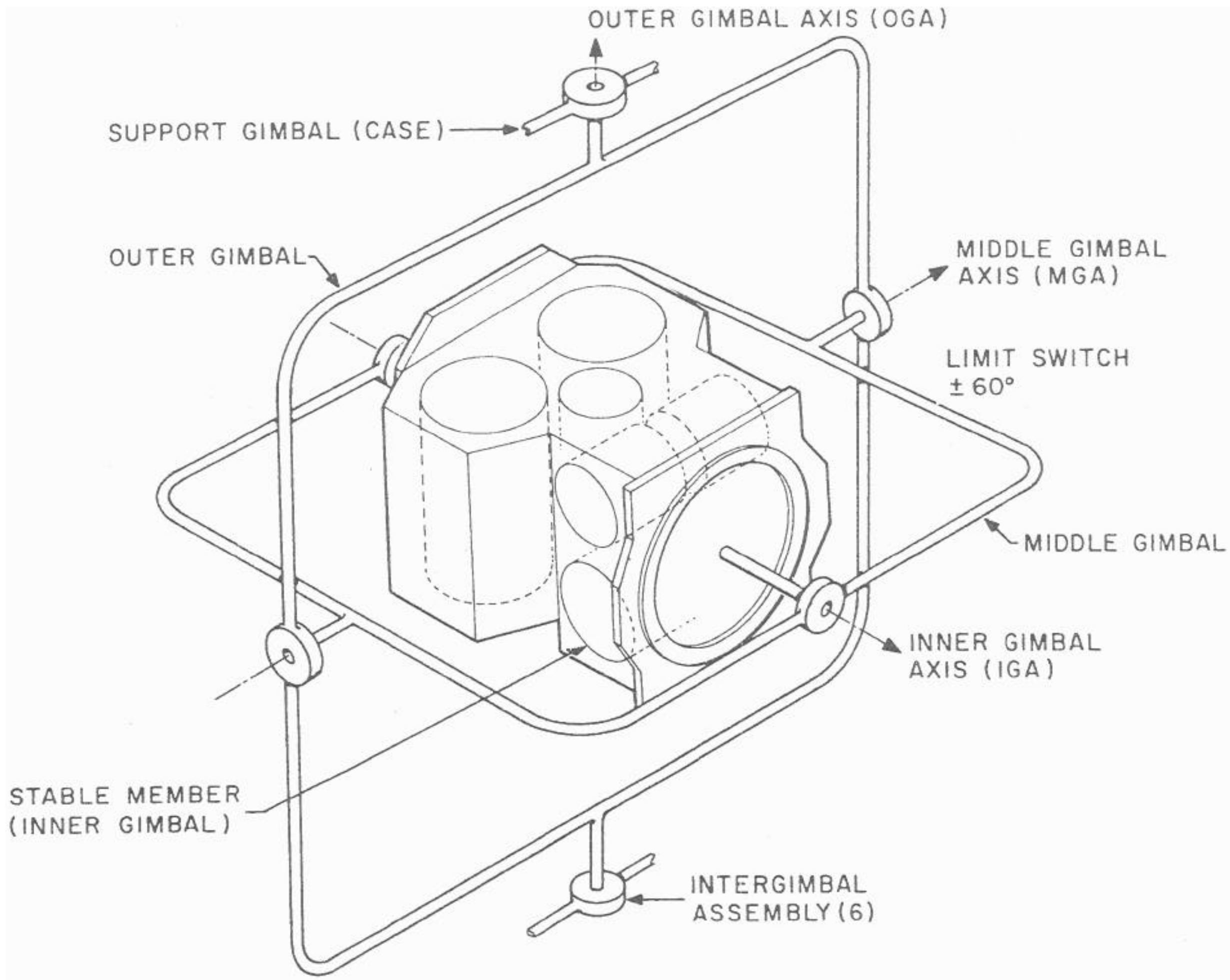


Fig. 5-1 Schematic of the Inertial Measurement Unit

Alternately, the vehicle frame- or body-mounted inertial subsystem offers dramatic savings in size, weight, and mounting convenience. Unlike the gimbaled system gyros which merely must indicate the small deviations from initial attitude for closed-loop gimbal control, the body-mounted gyros must measure precisely the vehicle's whole angular velocity. Good gyro and accelerometer performance is made more difficult, because of the large angular movements the units must tolerate about all axes. Finally, the outputs are not always in a direct useful form. Vehicle angular orientation is indicated only by properly transforming and integrating the body-fixed coordinates of angular motion indicated by the gyros into either an Euler-angle set or a matrix of direction cosines. With either of these, the body-mounted accelerometer signals can be resolved from the rotating spacecraft coordinates into an inertial frame. All these calculations require a computer of considerable speed and accuracy to prevent accumulation of excessive error.

The design choice made for both the APOLLO command and lunar modules was, with due consideration of computer technology available in the early 60's, the gimbal-mounted platform. Its superior performance provides a conservative margin of safety in economic use of rocket fuel for the major mission maneuvers. At the same time, secondary backup or abort guidance systems in each spacecraft capitalize on the size and convenient installation advantages of body-mounted sensors. Here the more modest performance is quite ample for abort maneuvers in case of primary guidance system failure.

The APOLLO inertial measurement unit is shown schematically in Figure 5-1. This unit carries three single-degree-of-freedom gyros that providenecessary error signals to servo drives on each axis to stabilize in space the angular orientation of the inner member. The gimbal system has three rotational axes. The three degrees of freedom can present problems due to a phenomenon called "gimbal lock." This occurs when the outer axis is carried by spacecraft motion parallel to the inner axis. In this position, all three axes of gimbal freedom lie in a plane, and no axis is in a direction to absorb instantaneous rotation about an axis perpendicular to this plane. Thus, at gimbal lock, the inner stable member can be pulled off its space alignment. Even though a three-degree-of-freedom gimbal system allows all geometric orientations, the required outer gimbal angular acceleration needed to maintain stabilization at gimbal lock will exceed servo capability. In the APOLLO design, servo loop stabilization is adequate for a middle gimbal angle up to within 15 degrees of gimbal lock. An automatic "coarse alignment" to a spacecraft-referenced frame is performed in the remote event that gimbal lock is impending,

One direct solution to gimbal lock problems is to add a fourth gimbal and axis of freedom driven to keep the other three axes from getting near a common plane.

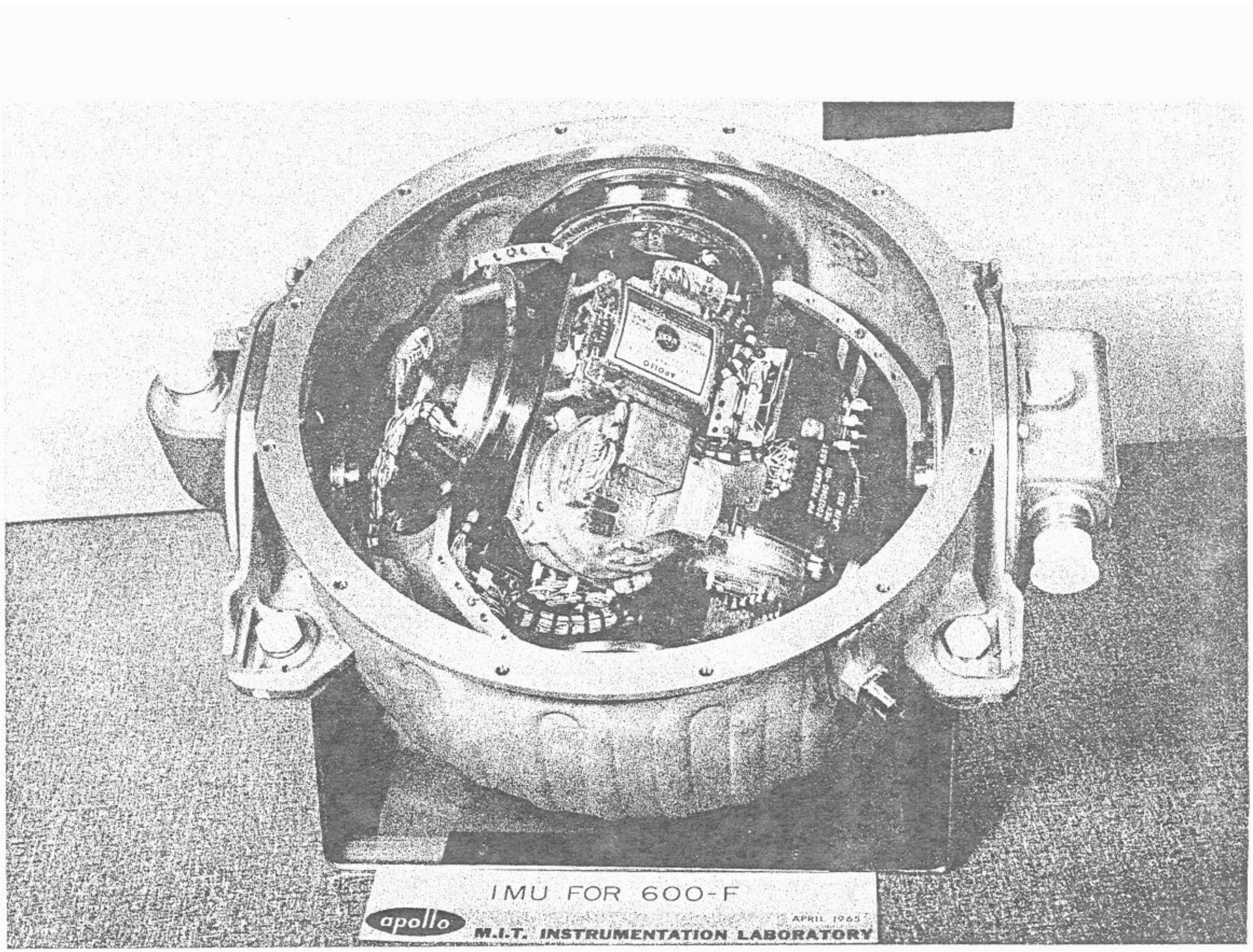


Fig. 5-2 Inertial Measurement Unit for System 600F (LM Functional)

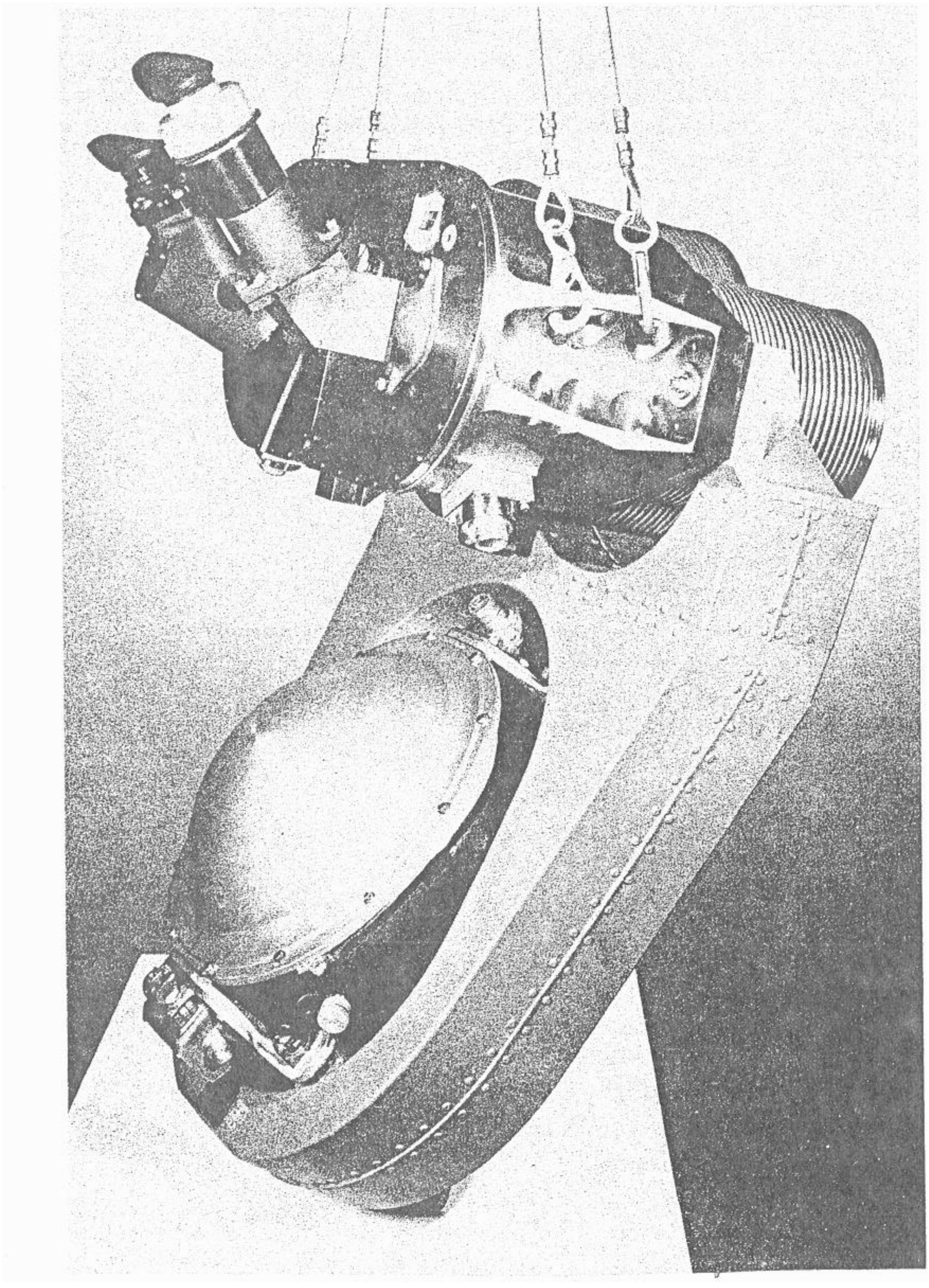
However, the cost in complexity and weight for a fourth gimbal is considerable. Fortunately, in APOLLO the inertial measurement unit operations are such that gimbal lock can be easily avoided; although operational experience has shown the constraints of the simple three-degree-of-freedom gimbal system are sometimes a nuisance.

The guidance, navigation, and control system must provide for inflight inertial subsystem alignment and realignment against star references before the start of each accelerated mission phase. The inner stable member alignment chosen is the most logical one-the "X" accelerometer axis on the stable member is aligned in some direction near parallel to the expected thrust (or entry atmospheric drag). This also minimizes inertial sensor measurement error effects in velocity measurement. Since the X accelerometer is perpendicular to the inner gimbal axis, the direction of this inner axis can be chosen as required. For each mission phase involving rocket burning or atmospheric drag, the trajectory and the thrust or drag lie close to some fixed plane. The inner gimbal axis is then aligned nearly perpendicular to this plane. All required large maneuvers cause primarily inner gimbalmotion, and avoid the gimbal lock associated with large middle gimbal angles. Finally, because large roll maneuvers are desirable (for instance during command module atmospheric entry), the outer gimbal axis is mounted to the spacecraft along or near the roll axis so that no restriction on roll maneuver exists,

An overall view of the inertial measurement unit is shown in Figure 5-2. In this photogaph, the spherical gimbal halves and case cover are removed to show the components mounted on the stable member and on the gimbal axes.

5.1.1 Inertial Subsystem Alignment

Periodic in-flight alignment and realignment is required for measuring both the large guided maneuvers and drift in the inertially derived attitude reference. Alignment determines the inertial attitude after turn-on or after the inertial measurement unit has been in a spacecraft-fixed mode of operation, Realignment negates accrued errors due to gyro drift. The sensed star directionmust be physically related to the inertial subsystem stable member orientation. The problem could be minimized by mounting the star sensor or sensors directly on the stable member. This would impose a most severe limitation of field-of-view of sky available and puts unpermissible constraints on spacecraft attitudes during alignment. Even a measured two-degree-of-rotational-freedom of the star sensor axis on the stable member limits flexibility and compromises design more than can be tolerated,



Fig, 5-3 Optics and IMU on Navigation Base

Mounting the star sensor telescope separately near the spacecraft skin where its line of sight can be articulated to cover a large portion of the sky produces far more freedom in spacecraft attitude during inertial subsystem alignment and realignment sightings, In APOLLO, a rigid structure called the navigation base—which is strain-free mounted to the spacecraft-provides a common mounting structure for the star alignment telescope and the base of the inertial measurement gimbal system. Figure 5- 3 shows this arrangement for the subsystem in the command module. By means of precision angle transducers on each of the telescope axes and on each of the axes of the inertial subsystem gimbals, the indicated angles can be processed in the onboard computer to generate the star direction components in inertial subsystem stable member coordinates. This provides the computer with part of the needed stable member orientation data, although no information is provided for rotation about the star line. The use of a second star, at an angle far enough removed from the line of the first, completes the full three-axis stable member orientation measurement. With this information, the stable member inertial orientation can then be changed under computer command to the optimum orientation for use of the guidance maneuvers.

The above procedure has many error sources. Each axis of rotation of the star telescope and the inertial subsystem gimbals must be accurately orthogonal (or at a known angle) with respect to the adjacent axis on the same structure. This is a problem of precision machining, accurate bearings, and stable structures. Each angle transducer on each axis of the star telescope and on the inertial subsystem gimbals must have minimum error in indicated angle. This includes initial zeroing, transducer angle function errors, and digital quantization errors for the computer inputs. By careful attention to minimizing each of these and other error sources, alignment probable error of the order of 0.1 milliradian is achieved-an accuracy which exceeds requirements by a comfortable margin.

5.2 OPTICAL MEASUREMENT SUBSYSTEM

Besides providing for inertial alignment the optical subsystem also provides the onboard measurement capability for orbital and midcourse navigation of the command module. The single-line-of-sight directionmeasurement used for inertial alignment can be used also in low earth or lunar orbit navigation. However, for onboard navigation during the translunar and transeartn mission phases, accuracy requirements are met only by a two-line-of-sight sextant.

Two separate optical instruments in the command module are mounted in an integral housing on the navigation base supporting the inertial measurement unit. These

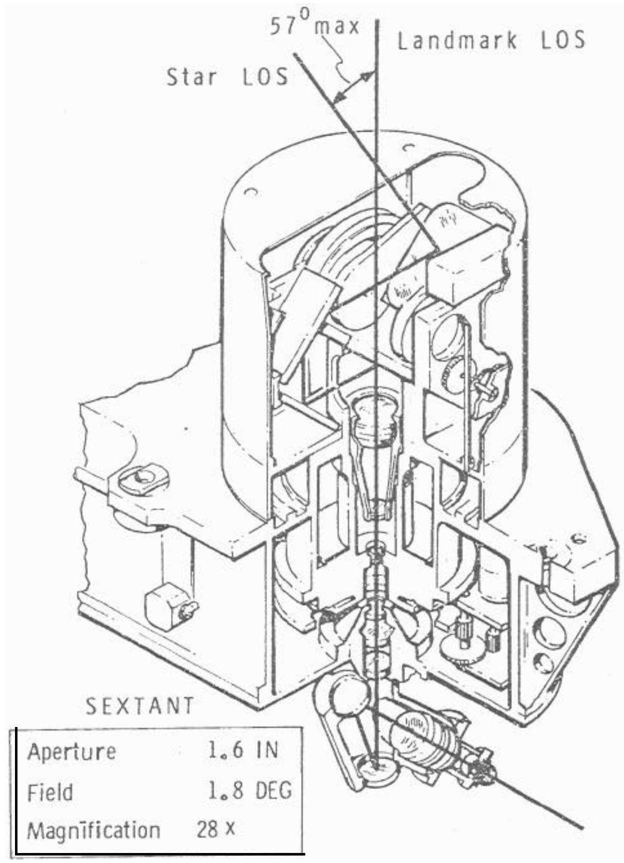


Fig. 5-4 Optical Subsystem (Sextant)

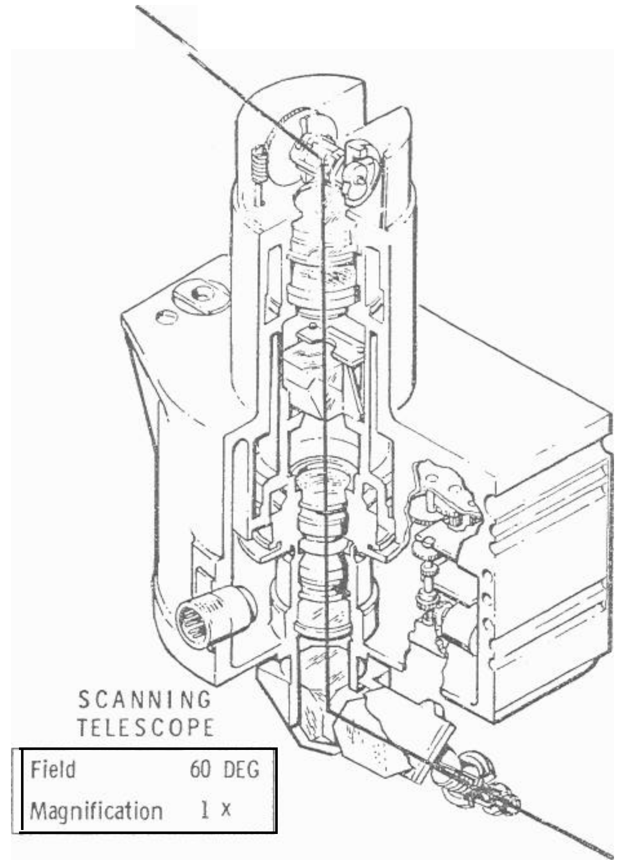


Fig. 5-5 Optical Subsystem (Scanning Telescope)

are the two-li,ne-of-sight sextant and the single-line-of-sight scanning telescope. The sextant and its features are illustrated in Figure 5-4. It is a two-line-of-sight instrument providing magnification for manualvisualuse. The design includes special sensors for automatic use, although they have not been carried aboard mainline APOLLO flights (see Part 2, Chapter V)] One of the lines of sight of the sextant, identified with the landmark or horizon side of the navigation angle, is undeflected by the instrument and is thereby spacecraft-fixed. To aim this line, the spacecraft must be turned in space by orientation commands to the reaction control system rockets. The second line identified with the star side of the navigation angle can be pointed in space through the use of two servomotor drives illustrated schematically. One axis of this motion-called the shaft axis-is parallel to the landmark line and changes the plane in which the navigation angle is measured by rotating the instrument head. The second axis-trunnion axis-sets the navigation angle by tilting the trunnion axis mirror. A 64-speed resolver on this mirror provides a direct measure of the navigation angle for the computer's navigation routine. An angle transducer on the shaft axis provides indicated star direction dataneeded by the computer for inertial alignment.

Because the sextant's 28-power magnification results in approximately a 2-degree diameter field-of-view, the second instrument-the scanning telescope-is necessary to provide a wide-field acquisition capability for the sextant. The use of a separate optical instrument-rather than a combined variable-power instrument using one set of line-of-sight articulation drives-is justified by the simpler mechanical and optical configuration and the sighting redundancy that two units provide.

The scanning telescope illustrated in Figure 5-5 has shaft and trunnion pointing of its single line of sight. The telescope shaft angle always follows the sextant shaft angle via servo action when the optics subsystem power is on. The trunnion can be selected by the astronaut to (1) follow the sextant trunnion and look along the star line, (2) be driven to zero and look along the landmark line, or (3) be driven to a fixed offset angle of 25 degrees, The last was intended for ease in simultaneous acquisition of landmark and star, since the scanning telescope will indicate both the image along the landmark line by a reticle point 25 degrees from the center of the field and possible stars available by trunnion motion in the sextant field of view on a diametrical reticle line. Figure 5- 6 shows the view through the scanning telescope during acquisition. Generally, as the preliminary step in the proposed acquisition process, the navigator can preset the trunnion to the expected navigation angle indicated by the computer. In the missions to date, the star line-of-sight acquisition has been accomplished via an automatic optics pointing routine in the computer, rather than by the manual procedure. However, the manual procedure is available

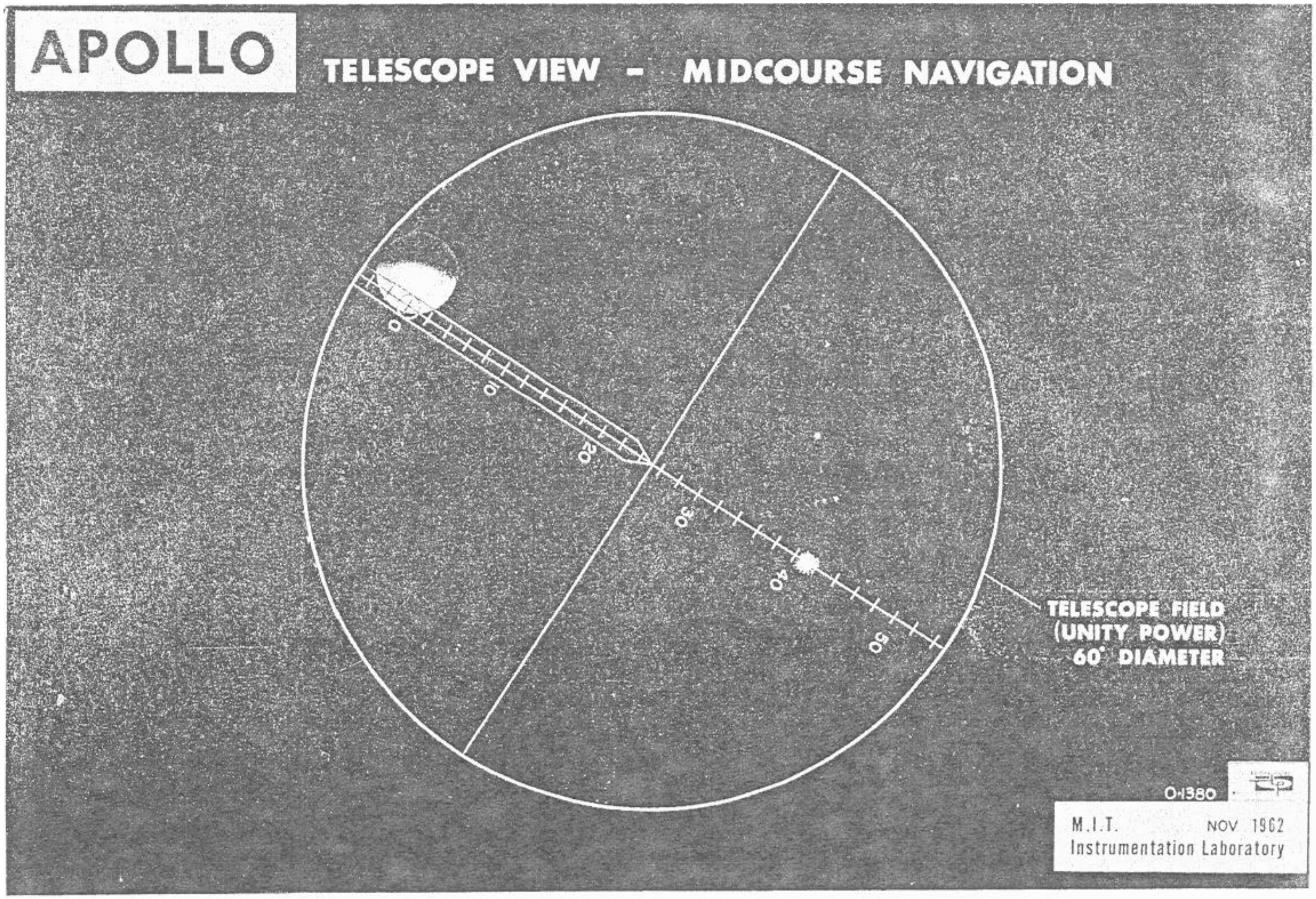


Fig. 5-5 Telescope View — Midcourse Navigation

in the event that inertial orientation knowledge was not available when the inertial measurement unit is off.

Under manual visual control, the shaft and trunnion drives of the telescope are commanded by a left-hand two-axis controller. By this controller, the navigator can point the scanning telescope and the sextant star line. At his right hand, the navigator has the spacecraft attitude controller with which he can rotate the spacecraft to position the landmark line on the target horizon or landmark. His midcourse cislunar sighting strategy is to set up the measurement situation illustrated in Figure 5-7. Computer-controlled attitude maneuvers are available to get the identified horizon or landmark within the sextant field-of-view at a slow spacecraft rotation drift. He need then only provide occasional minimum impulses from the appropriate attitude jets to keep the target within the field; the automatic optics pointing routine can then get the selected navigation star in the sextant field, after which with his left hand he positions the star image to superimposition. When this is achieved (Figure 5-8), he pushes a "mark" button which signals the computer to record the navigation trunnion angle and time, From these data, the computer updates the navigation state vector if the astronaut approves the displayed data. Thus far the navigation measurements in midcourse flight have been of the star-horizon type.

The unity-power, wide-field-of-view scanning telescope is also suitable for navigation direction measurements to landmarks in low earth or moon orbit, The wide field of view makes landmark recognition easy. Landmark direction measurement accuracy of the order of 1 milliradian (as referenced to the prealigned attitude of the inertial subsystem and as limited by the unity magnification) is sufficient for landmark ranges under a few hundred miles. Computer-aided sextant pointing to landmarks is an available alternative which has the advantage of the more accurate instrument and the disadvantage of decreased recognition capability in the small field of view.

In low orbit, the inertial measurement unit must be on and prealigned with two star sightings. The navigator acquires and tracks landmarks as they pass beneath him, pushing the mark button when he judges he is best on target (Figure 5-9). The computer then records optics angles, inertial unit gimbal angles, and time.

In all uses of stars and horizons or landmarks for navigation, the computer must be told by the navigator the identifying code or coordinates of the star and horizon or landmark. These appear on the navigator's maps and charts,

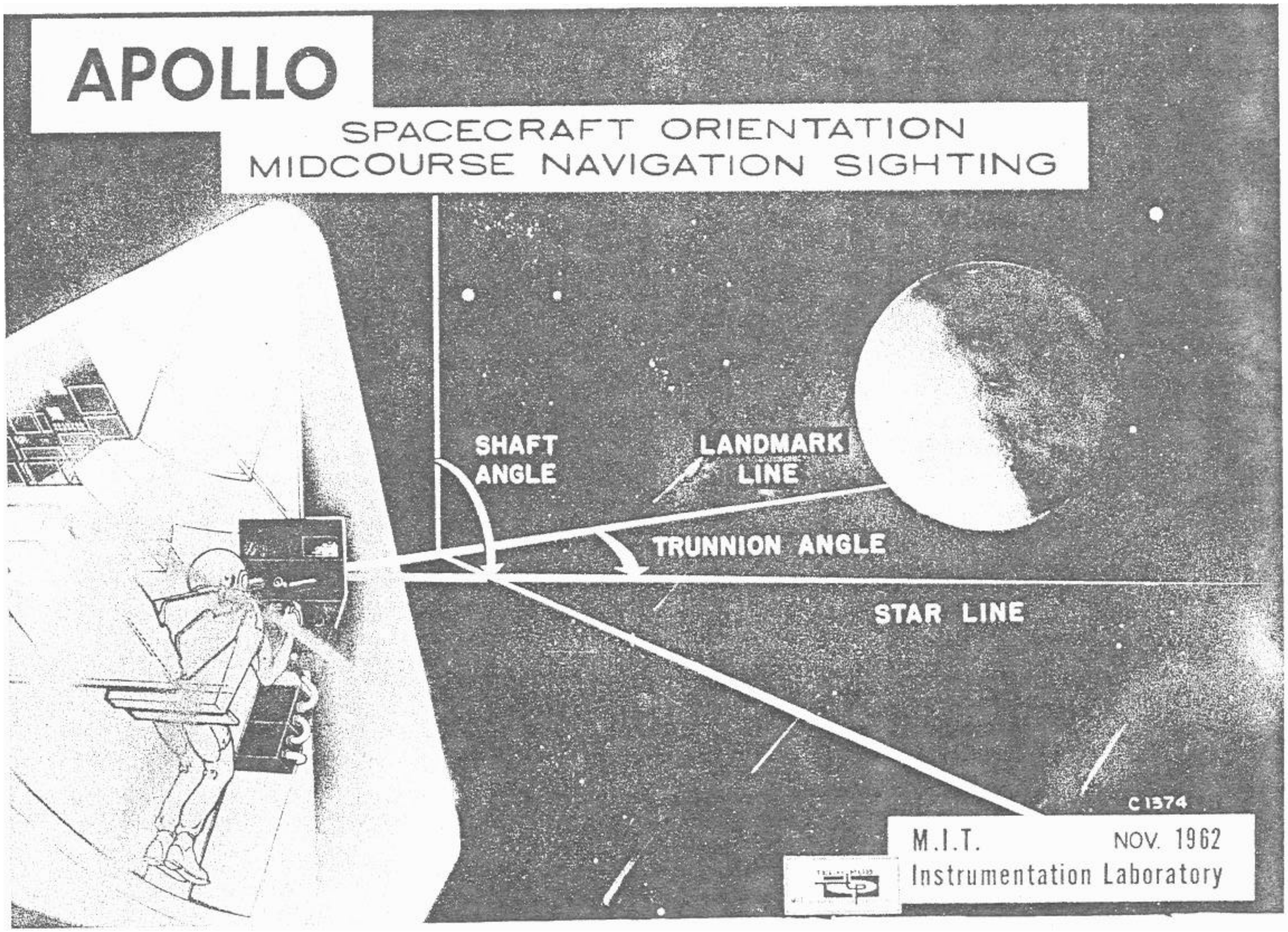


Fig. 5-7 Spacecraft Orientation — Midcourse Navigation Sighting

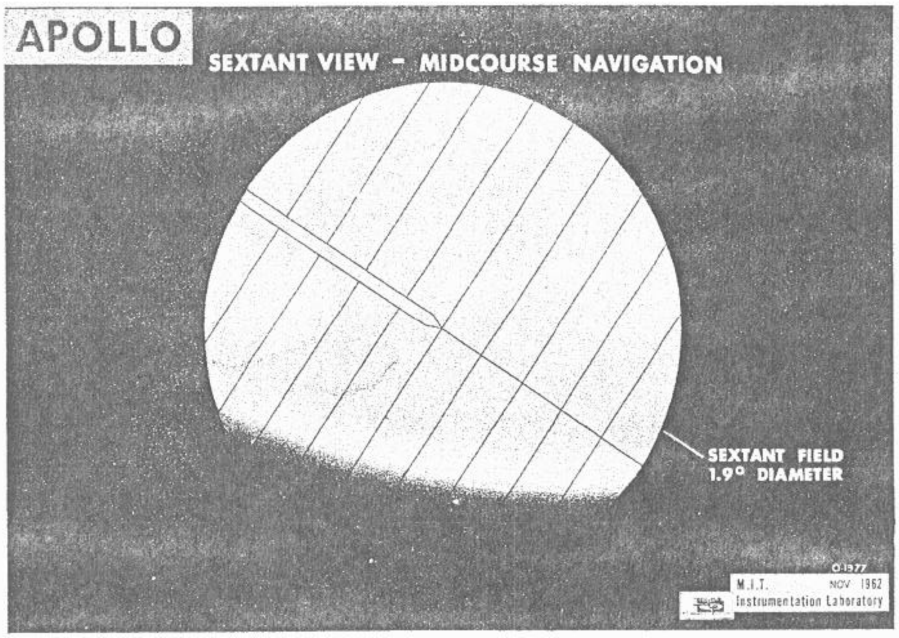


Fig. 5-8 Sextant View -- Midcourse Navigation

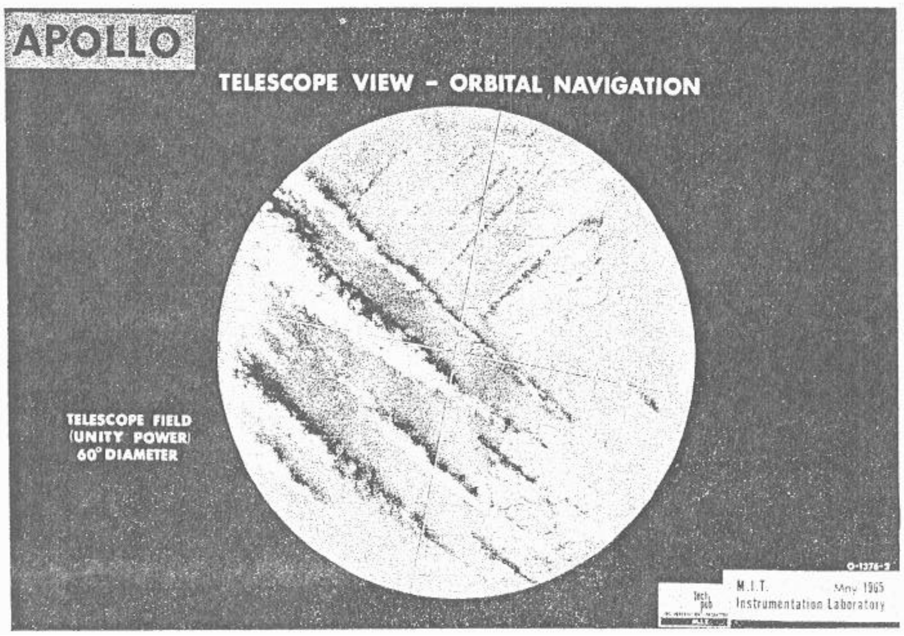


Fig. 5-9 Telescope View -- Orbital Navigation

5.3 ONBOARD COMPUTER SUBSYSTEM

The relatively large amount of onboard data processing required for APOLLO guidance, navigation, and control can be met only by the capabilities of a specially designed digital computer. The special requirements include:

1. Logic, memory, word length, and speed capability to fit the needs of the problems handled
2. Real-time-data processing of several problems simultaneously on a priority basis
3. Efficient and yet easily understood communication with the astronauts for display of operations and data as well as manual input provisions for instructions and data
4. Ground control capabilitythrough radio links and telemetering of onboard operations and data to the ground
5. Multiple signal interfaces of both a discrete and continuously variable nature.

The many input and output signals play a large part in determining system configuration and system tasks and operation. Inputs to the computer of a discrete or two-state nature are handled as contact closures or voltage signals. These offer no difficulty except for the computer activity needed to keep appraised of them. Urgent signals of this nature-such as an abort command or the time-critical “mark” signals-go to special circuits that interrupt computer activity and process them before other activity is resumed or modified. Less critical signals are examined periodically as necessary.

Discrete signal outputs are of two types. Time-critical ones-for example, the signal for engine thrust cutoff-consist of high frequency pulse trains gated on at the time of the programed event and detected remotely when requested. Slower discrete outputs are either gated dc voltages or relay contact closures set by a state matrix driving the appropriate relay coils. Most of these relays are used to set the states of the electroluminescent number display readout of the computer display and keyboard. Others change operating modes of the associated spacecraft systems or are used to light status or warning lights.

Direct earth communications to and from the computer requires circuits to convert between the serial code of the telemetry and the parallel format of the computer.

Other variables fed into the computer are handled by input counters that sum pulses through fixed increments as the indicated variable changes. Velocity increments measured by the inertial subsystem accelerometers are handled this way.

Some variable outputs, such as the command torquing to change alignment of the inertial subsystem gyros, appear as output increment pulses on appropriate lines.

Perhaps the most difficult class of computer interfaces is handled by the use of the coupling dataunits. These units couple with the digital computer the sine and cosine analog signals from the resolver angle transducers on the optics and inertial gimbal system axes. There are Five coupling data units-one each associated with optics shaft, optics trunnion, and the three inertial measurement unit axes.

5.4 DISPLAYS AND CONTROLS

APOLLO displays and controls were designed to provide the crew with visibility into and command over the guidance, navigation, and control tasks. In most of these tasks, the astronaut can select either to be intimately involved in the procedures or to allow automatic operation which he may monitor at his discretion.

In the command module, the navigator has displays and controls as illustrated in Figure 5-10. The eyepieces of the sextant and scanning telescope appear prominently side by side. Just below these eyepieces is a control panel used primarily for optics operation, The left-hand optics hand controller and the right-hand spacecraft attitude minimum-impulse controller appear at the top of this panel. At the bottom of this panel are operating-mode selector switches. To the right is the display and command keyboard associated with the computer. Miscellaneous analog electronics that operate the equipment appear under the center panel. Below this is the digital computer and to the left the coupling data units. A second display keyboard is mounted on the main panel in front of the pilot's couch. This unit is identical to and functions in parallel with the navigator's unit, so that the majority of guidance and navigation functions can be completed or observed from either station. Also visible to the command pilot is a flight director attitude indicator and associated position and rate needles, Figure 5-11. The spacecraft orientation is indicated by the attitude of the ball which is driven in three inertial measurement unit axes. The three components of attitude error generated by the guidance system are also displayed by the position of three pointers that cross the face of the instrument. Vehicle attitude rates measured by three vehicle-mounted rate gyros are displayed by three more pointers around the side of the instrument. Other displays showing guidance, navigation, and control system status with regard to caution and alarm functions are available to the pilot on the main panel along with a complex array of equipment associated with other systems for control mode selection and display.

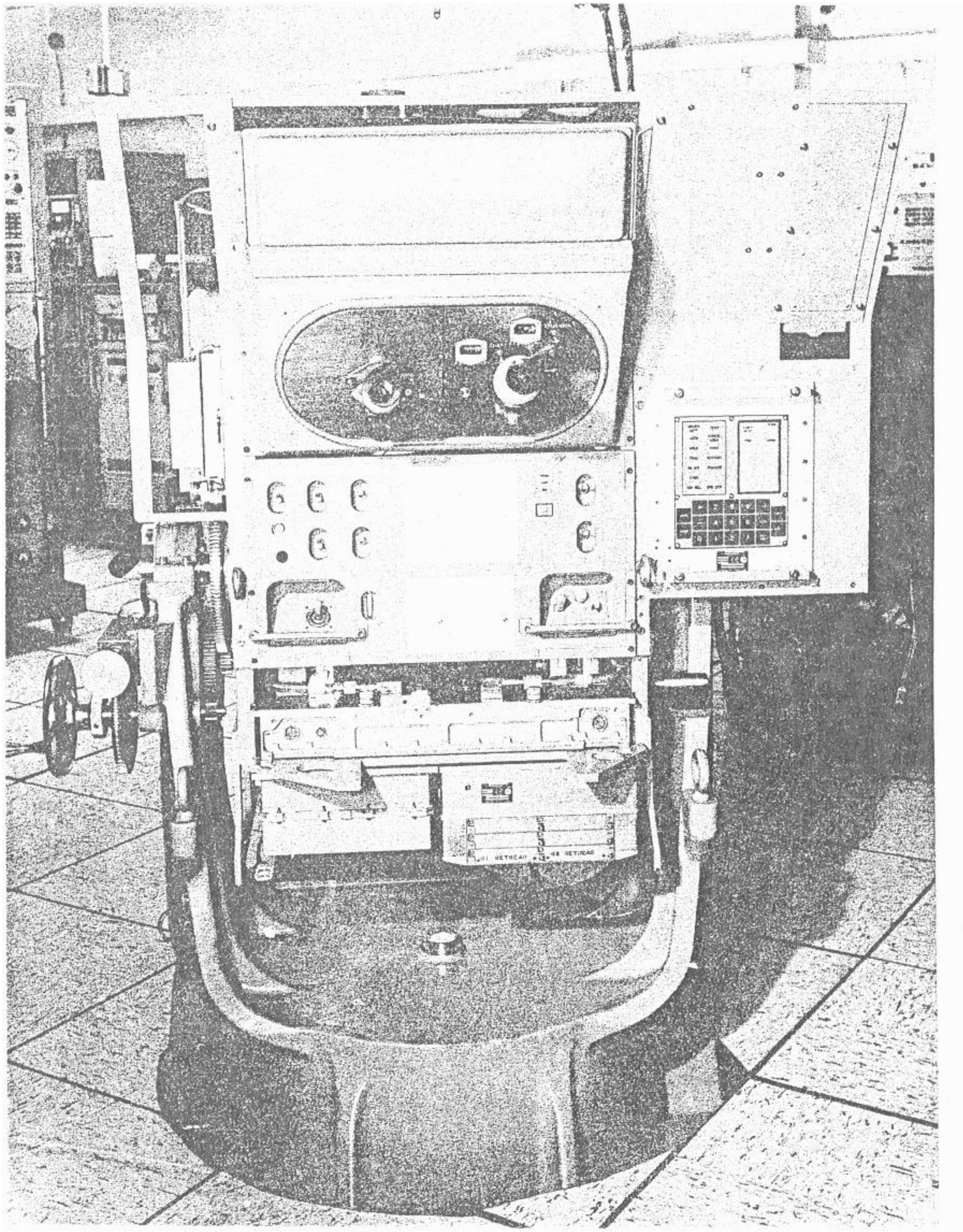


Fig. 5-10 G&N System — Command Module Lower Equipment Eay

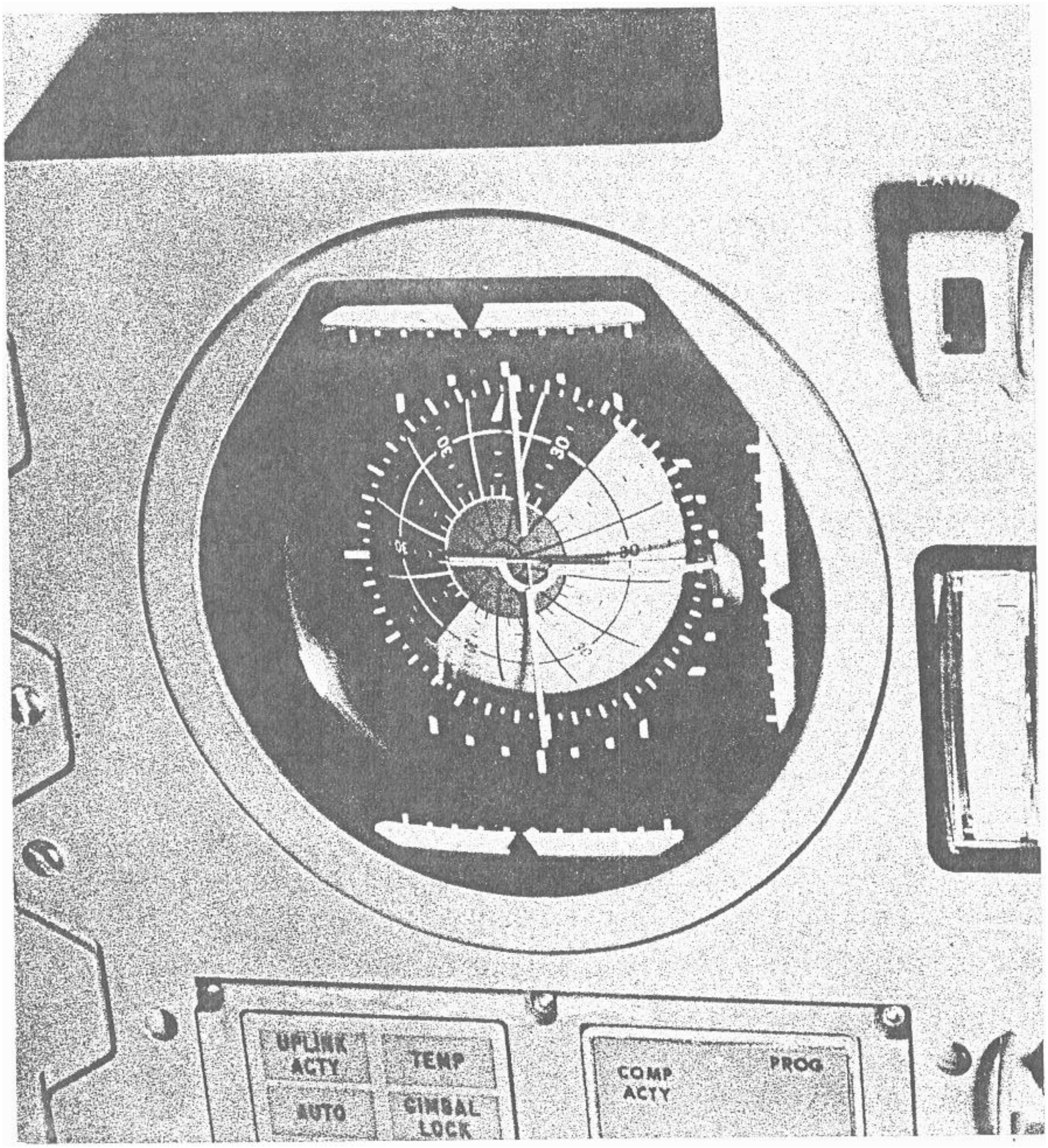


Fig. 5-11 Ball Attitude Indicator

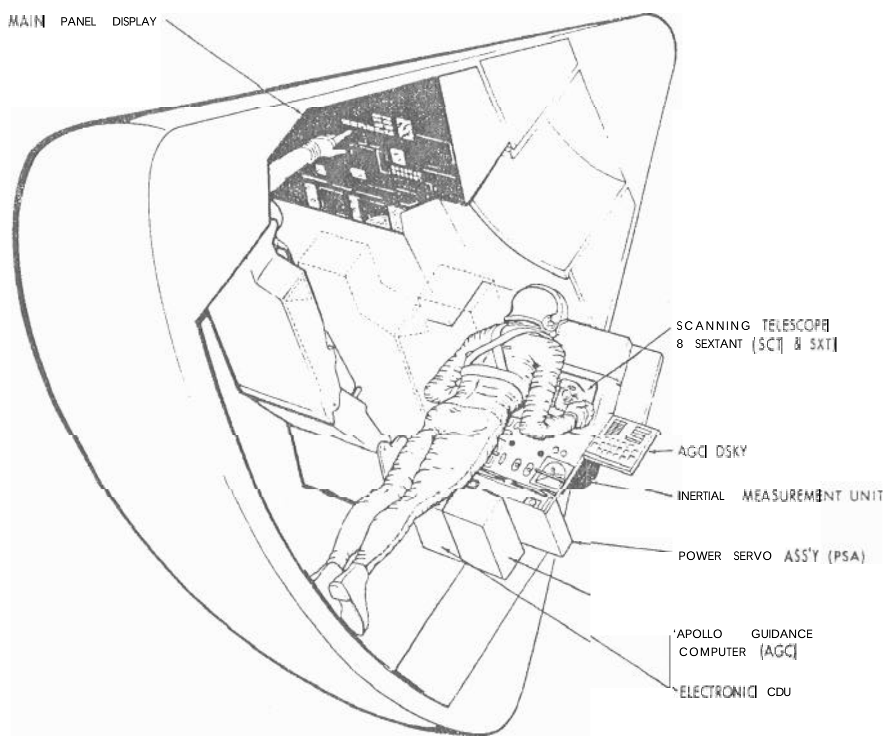


Fig. 5-12 Location of the Guidance and Navigation System in the Command Module

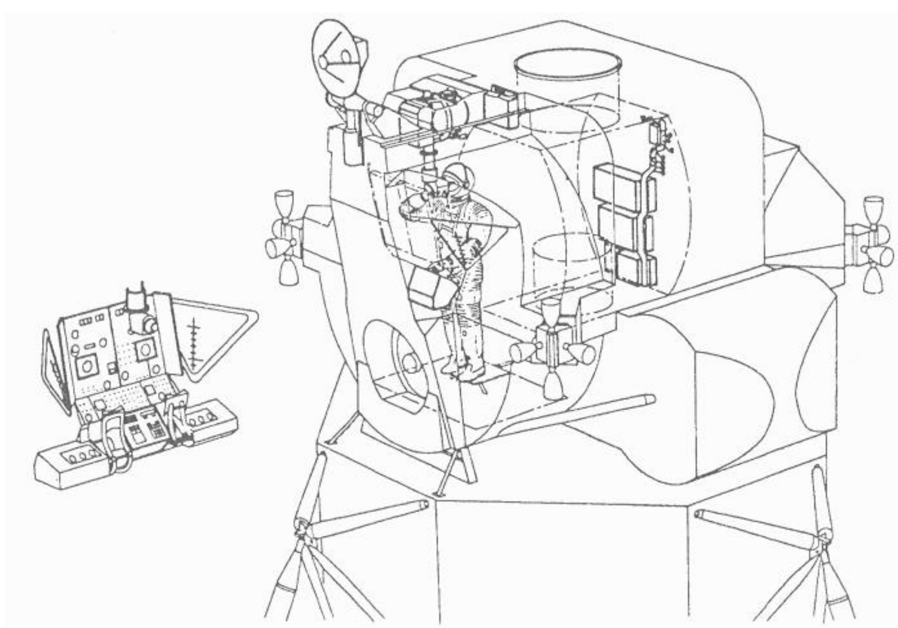


Fig. 5-13 Location of the Guidance and Navigation System in the Lunar Module

5.5 EQUIPMENT INSTALLATION IN SPACECRAFT

The installation of the command module guidance, navigation, and control equipment is illustrated in the cutawayviewof Figure 5-12. This shows the navigator operating the displays and controls at the lower equipment bay where the majority of the guidance and navigation equipment is located. Other control equipment is distributed around the spacecraft. When the acceleration forces are high, during launch boost into earth orbit and during return entry, the navigator must leave his station and lie in the center protective couch, between his companions. Sufficient controls and displays are on the main panel in front of the couches to perform all the guidance, navigation, and control functions except for those requiring visual use of the optics.

The lunar module installation is shown in Figure 5-13. The inertial measurement unit, the guidance computer, the coupling data units, and support electronics of the power servo assembly, are all essentially identical to those used in the command module. Since the lunar module, when separated from the command module, does not require optical navigation sightings, a simpler alignment optical telescope is installed on a navigation base with the inertial unit and is used only for aligning the stable member. Also unique to the lunar module are the two radars. The rendezvous radar is mounted near the inertial measurement unit so that direction data can be related between the two. The landing radar (not shown) is used on the descent stage and is discarded on the lunar surface.

5.6 OVERALL BLOCK DIAGRAMS

The signal interconnections among the guidance, navigation, and control equipment are illustrated in Figures 5-14 and 5-15 for the command and lunar module respectively. The intent here is to show only the general nature of the equipment interfaces, the similarity and differences between the command and lunar module systems, and the central role of the guidance computer in each case.

5.7 SPACECRAFT SAFETY CONSIDERATIONS

Although the risk is actually small, the APOLLO crew admittedly put their life in jeopardy when they embark in their spacecraft. However, unlike the more traditional pioneers and adventurers, the men flying the APOLLO missions leave in a spacecraft only after their safety is assured. Crew survival is an urgent concern in mission preparation. NASA has set high safety standards. The crew of a checked out vehicle leaving the earth launching pad for the lunar surface should have a 90 percent probability of completing the lunar landing mission and a 99.9 percent probability

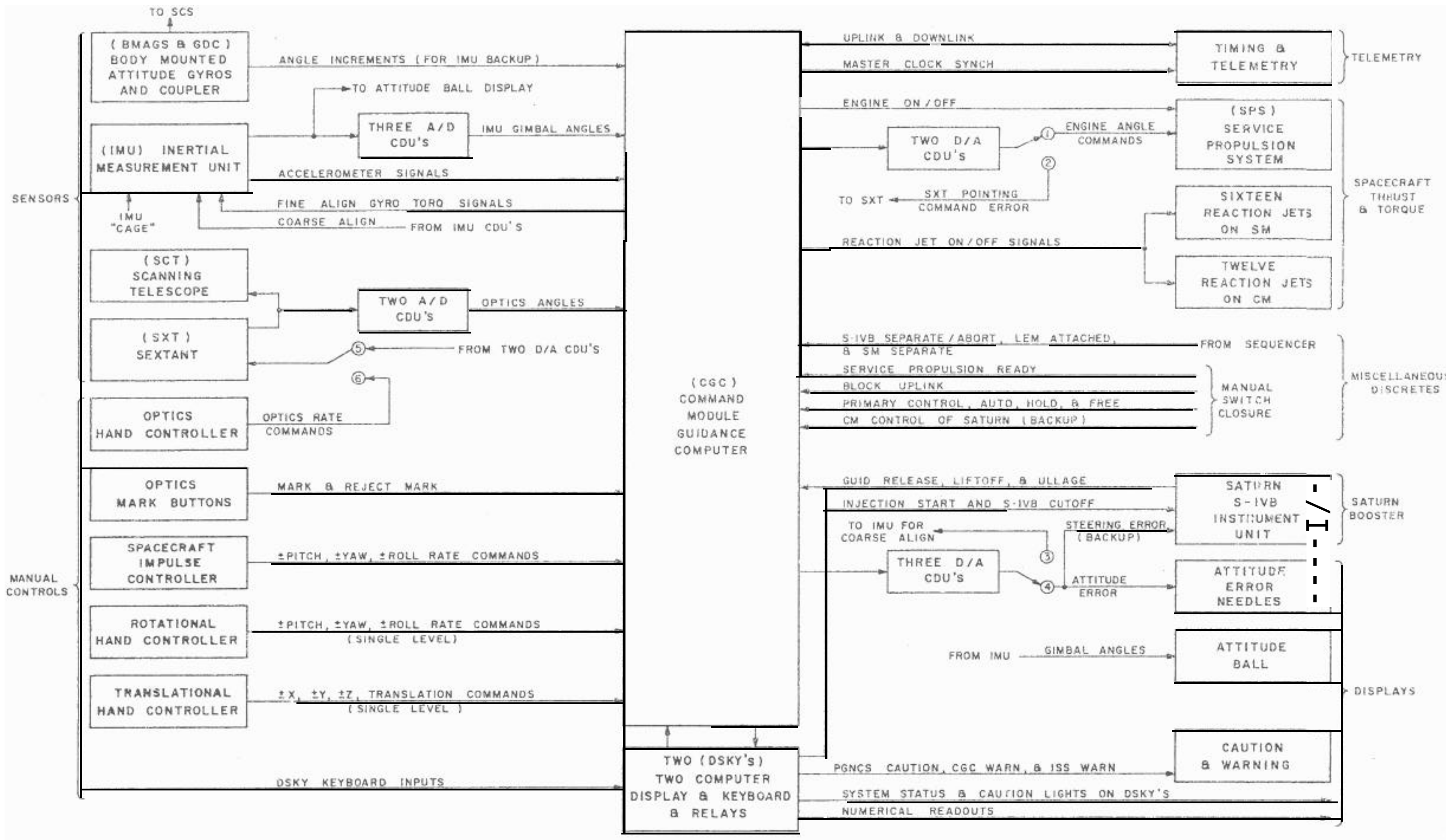


Fig. 5-14 Guidance, Navigation, & Control Interconnections in Command Module

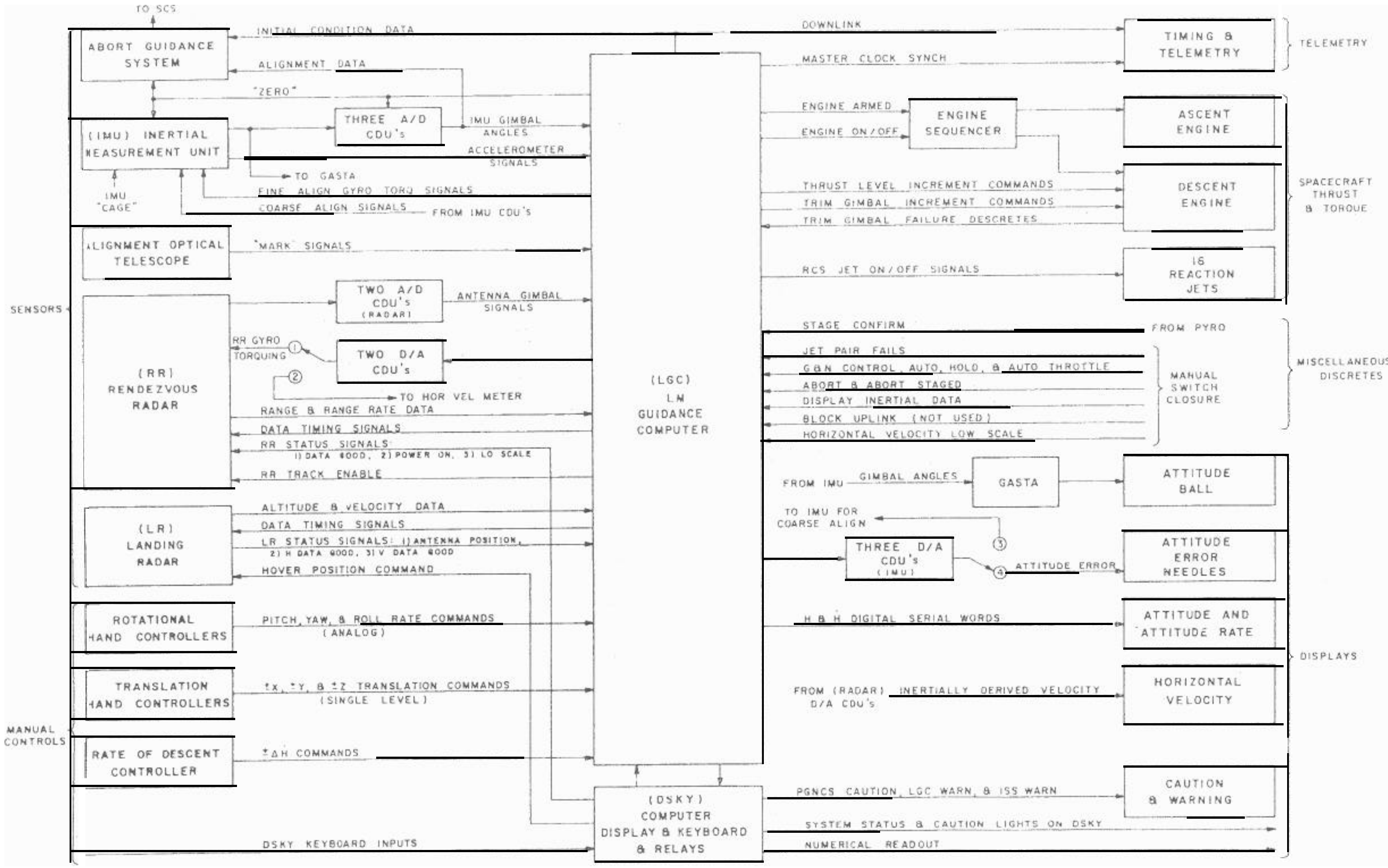


Fig. 5-15 Guidance, Navigation and Control Interconnections in LM

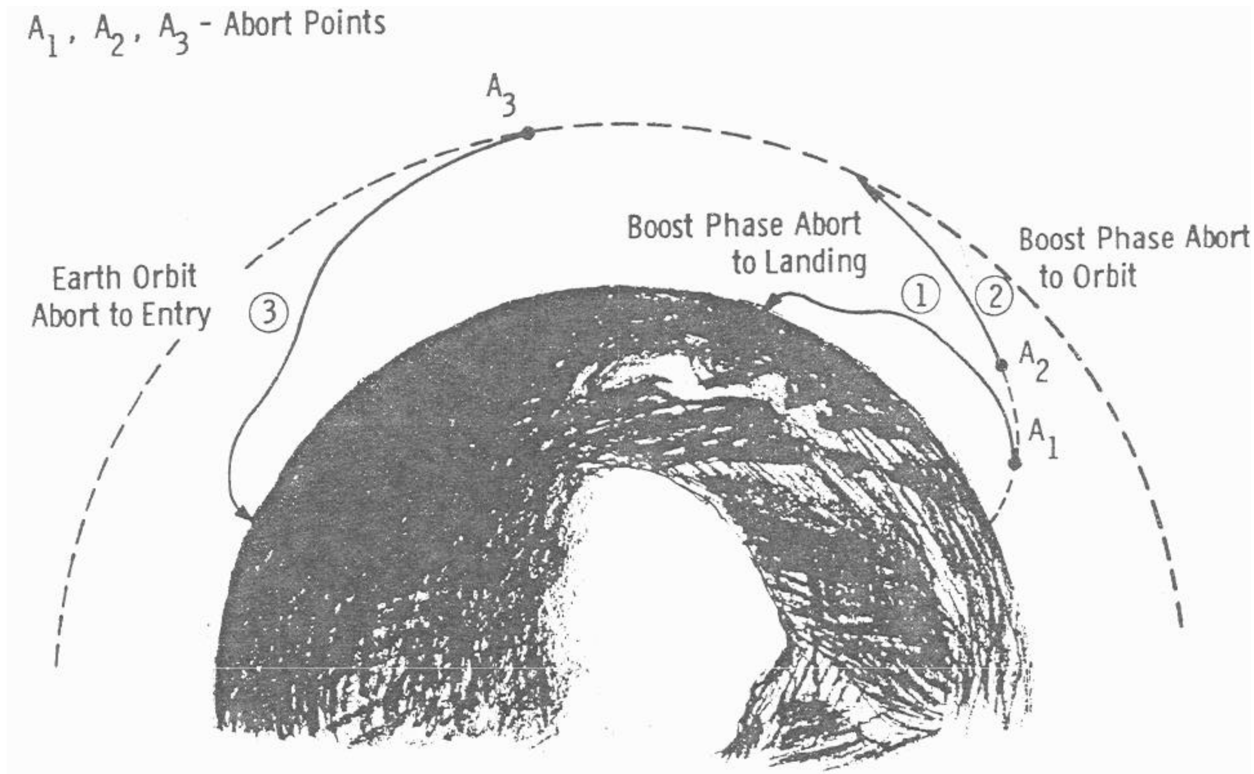


Fig. 5-16 Near Earth

of returning to earth safely. These goals pertain to all parts of the APOLLO program: mission planning, spacecraft design, crew training, testing methods, and so forth; but the focus here is only with safety aspects of the guidance, navigation, and control systems. The nominal mission phases are outlined in Section 2 of this report; the following paragraphs deal with abort capabilities from the various phases.

5.7.1 Abort Trajectories

The failure tolerance in APOLLO systems is based on the deliberate design guideline that any single failure should, if at all possible, leave enough working equipment remaining to bring the crew safely home. Although for practical reasons, this guideline cannot be met everywhere, the number of safety critical flight items that have no backup is quite small.

The guidance, navigation, and control equipment in particular is designed with enough flexibility in both equipment and computer programs to support the measurements and maneuvers necessary for all reasonable mission abort trajectories caused by failures in other parts of the spacecraft. Depending upon the nature of the failure and the mission phase in which failure occurs, the crew can initiate an abort by informing the computer and setting the appropriate propulsion systems. In some situations, the spacecraft commander can inform the computer which of three types of aborts he wishes: (1) time-critical aborts requiring fastest return using all available propulsion, (2) propulsion-critical aborts, requiring optimum use of available fuel in energy-efficient orbital transfers, and (3) normal aborts, using trajectories constrained to achieve a landing on one of the prepared earth recovery areas. The computer can inform the crew of the times of flight and propulsion usage for each of the above aborts so that the abort mode decision can be made.

The abort trajectory depends upon the mission phase in which the abort decision is made. Figure 5-16 illustrates the three abort types pertinent to operations near the earth. Trajectory 1 is direct abort to earth during launch boost ascent. It is flown when the failure requires immediate return to earth or when sufficient propulsion is not available to fly trajectory 2. Abort trajectory 2 continues the flight into earth orbit using an upper stage of the vehicle. It provides a better choice of landing recovery area by selecting the phasing of the return maneuver. It does permit flight continuation, but of more limited mission scope. The descent from orbit, trajectory 3, is a simple maneuver similar to the earth orbit return.

Aborts that can be initiated after APOLLO has been committed to translunar injection are illustrated in Figure 5-17. Trajectories 1 and 2 on this figure are typical of

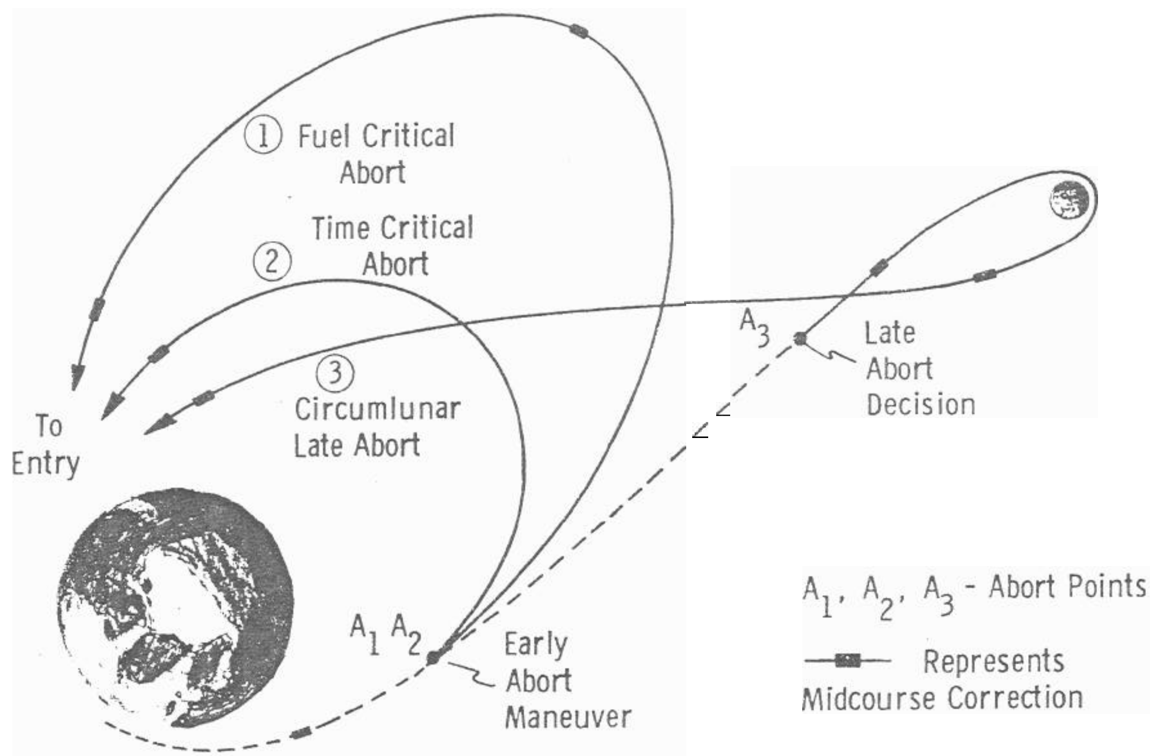


Fig. 5-17 Translunar Coast Abort Trajectories

the paths flown for aborts initiated during the first part of the translunar coast. Trajectory 1 illustrates a fuel-optimum direct return to earth. Trajectory 2 illustrates the full-fuel-usage quick return to earth. At some point in the translunar coast, the time-to-earth return is quicker if the spacecraft coasts around back of the moon and then continues home (trajectory 3). All these cislunar aborts require careful navigation.

After arrival into lunar orbit, aborts may either involve an immediate transearth injection or be preceded by recovery of the two men in the lunar module. Figure 5-18 illustrates the trajectories and operations involved with lunar module aborts. Trajectory 1 illustrates a typical abort initiated during the lunar descent. The abort trajectory injection, begun at point A_1, is guided and controlled to put the spacecraft in a fairly high elliptical trajectory so that the phasing is proper for rendezvous to meet the orbiting command module at point R_1. Midcourse corrections, not shown, are necessarily based upon navigation from the rendezvous radar or earth-tracking data. Unfortunately, much of the trajectory occurs in back of the moon out of sight of the earth tracking facility. Trajectory 2 of Figure 5-18 illustrates a typical lunar module emergency abort from the lunar surface. Here it is supposed that a failure has occurred-such as fuel tank leakage or life support system failure -that requires immediate ascent without waiting until the command module is in the proper position for a normal ascent and rendezvous. The lunar module is guided into a low-altitude clear perilune orbit which it will hold until it catches up with the orbiting command module. At the proper point, the ascent engine is fired again for transfer and rendezvous, using midcourse corrections as required. Alternately, once the lunar module succeeds in getting into a holding orbit, it can assume a passive role and allow the other spacecraft to maneuver for rendezvous and crew pickup.

5.7.2 Control of Propulsion Failure Backups

The APOLLO guidance and control equipment is designed to operate with abnormal propulsion and loading configurations for given mission phases to provide abort capabilities covering failure in any of the primary rocket engines. Figure 5-19 shows examples of these aborts. The heavy ascending line traces the normal mission phases from prelaunch to lunar orbit. The dashed lines trace abort paths using alternate propulsion sources to cover failures of the normal rocket used in each phase, These paths are numbered on the diagram and are explained briefly below:

1. The launch escape system could provide aborts from pre-liftoff until atmospheric exit during the early part of the second stage burn. No

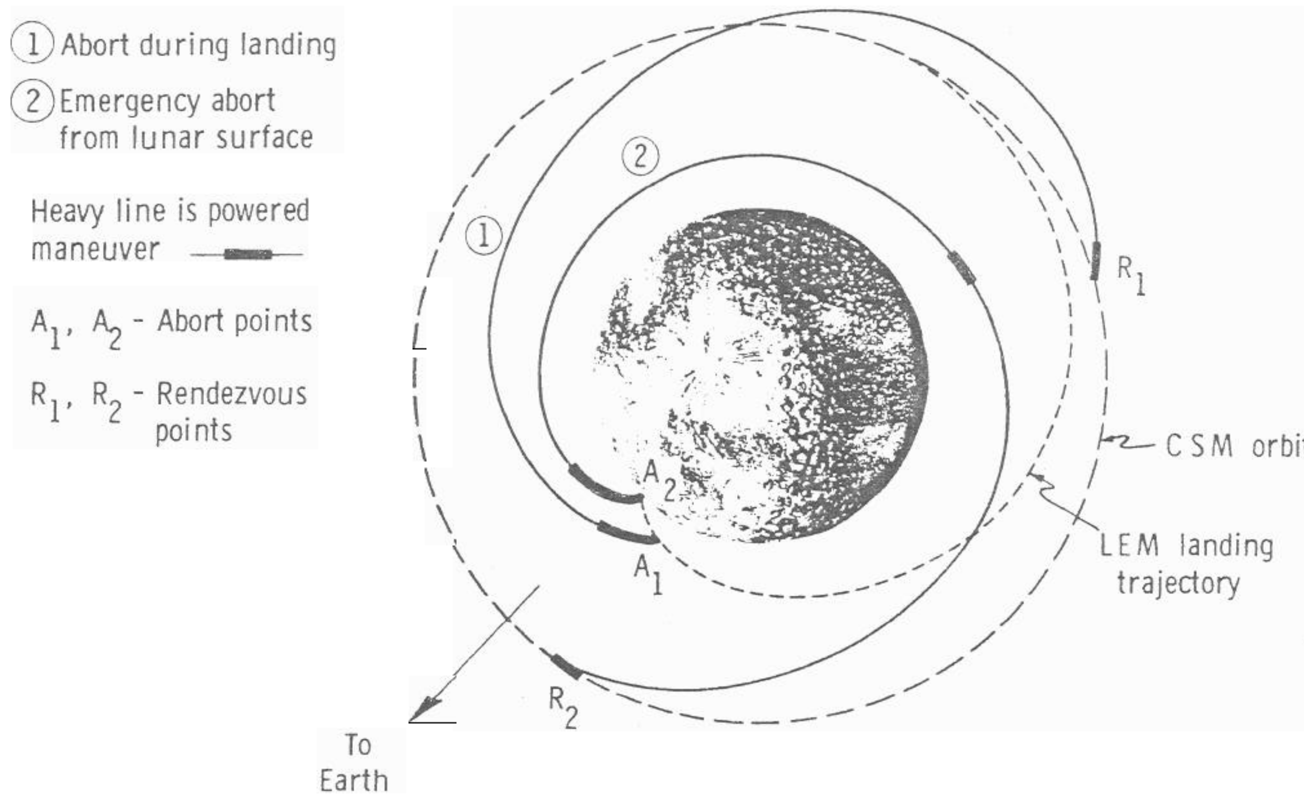


Fig. 5-18 LM Abort Trajectories

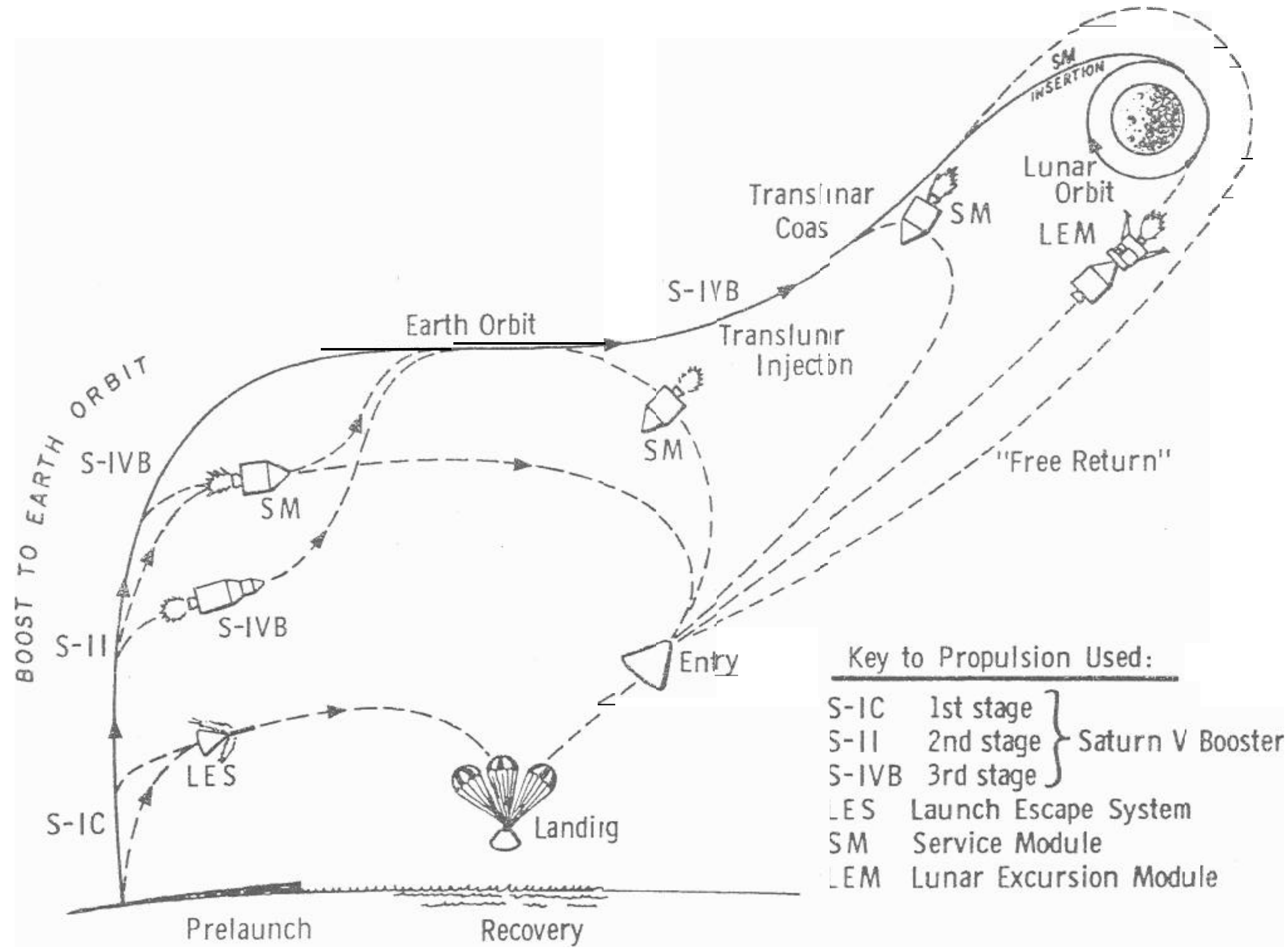


Fig. 5-19 Failure Abort Paths

measurement is necessary by the guidance system for launch escape aborts. The system is designed to pull the command module safely past and far enough away from an exploding booster for a low velocity entry and normal parachute landing.

2. A second stage failure during ascent might allow thrusting in the S-IVB SATURN third stage into earth orbit. This would deplete S-IVB fuel sufficiently to prevent continuation of a lunar mission.
3. Again during second stage boost and during third stage as well, the abort could be made to an immediate entry trajectory and landing using the command service module propulsion and the spacecraft guidance and control systems.
4. Aborts using service module propulsion during third stage boost could also be made into earth orbit. A second burn of the service module would then initiate descent to a selected landing site.
5. If the abort is initiated while in earth orbit, the service module propulsion could be used for descent, assuming it still functions. If it doesn't, the small reaction jets could be used in a limited retrograde translational burn or series of burns in order to capture the atmosphere and achieve suborbital velocity.
6. On the way to the moon, the service module propulsion could be used to inject into the return orbits described previously.
7. If the service module rocket has failed, the flight could continue around the moon on the "free return" path using the reaction jets in translation maneuvers to perform the midcourse maneuvers required for navigation.
8. If service module propulsion fails while in lunar orbit before the separation and descent of the lunar module, its propulsion, guidance and control systems could be used to inject the command module into the necessary transearth trajectory.

The examples of propulsion failure abort paths above illustrate the necessary flexibility and universalness needed of the APOLLO guidance, navigation, and control systems.

5.7.3 Failure Detection and Alarm

A central aspect of mission safety is the early detection of system failure. Part of this detection is systematic onboard testing during the stress-free coasting phases to assure that needed systems are functioning. Of more interest is the automatic failure-detection features that immediately signal appropriate alarms during the accelerated phases of rocket thrust or entry. With the help of these alarms, the crew can initiate appropriate abort action immediately if necessary,

Figure 5-20 is a simplified diagram of the guidance failure detection system for the command module. The box at the left represents the inertial. subsystem. The signals coming out are error detections. Shown is “gyro error”-a signal that exists when any of the gyro gimbal stabilization loop servo errors exceed a preselected detection level. The “accelerometer error” and "CDU error” have similar properties -detection of any of the accelerometer servo loop errors and coupling data unit servo loop errors. The “power supply fail” signals deviations of the inertial subsystem power supply voltages from preselected levels. Each of these detections is sent to the computer and is also separately summed to light a master inertial subsystem error display light. During system turn-on or mode switching, this light is expected to operate briefly, but will extinguish itself in a normal system.

The computer contains its own error detection programs and circuitry that are used to light the master computer error light and/or inertial subsystem error indicators. By pushing a reset button, the astronaut can extinguish an error indication if the error cause no longer exists. The computer program examines computer operation error circuitry and the inertial subsystem error detectors and lights appropriate fail lights. The “accelerometer fail” indicates that the acceleration data in the guidance are faulty and that the primary guidance steering cannot be used. In the latter case, however, the inertial attitude data may still be correct for use in a backup mode. A similar situation occurs with the "CDU fail” light.

The last light is a "master guidance fail” with special features that makes it fail-safe. The computer program examines periodically, at a fixed frequency, all the previous failure detections and, if it findsnone, the program sends out a pulse of a particular duration. If this pulse keeps occurring at the expected frequency, then the detector inhibits lighting the “master guidance fail.” Otherwise this signal lights.

If any of these lights operate, the crew is trained to take appropriate emergency backup action.

5.7.4 Guidance and Control Backup

Failure in the primary guidance equipment requires the use of an alternate backup system to provide safe return of the crew. The redundancy concept used is illustrated in Figure 5-21. The primary system involving the inertial measurement unit, the flight digital computer, and the associated coupling data unit constitutes a complete, flexible, accurate, and fuel-efficient guidance and control system able to perform all the maneuvers required to complete the mission. The backup system is simpler, smaller, and of more modest endowment, but it is able to make the more simple

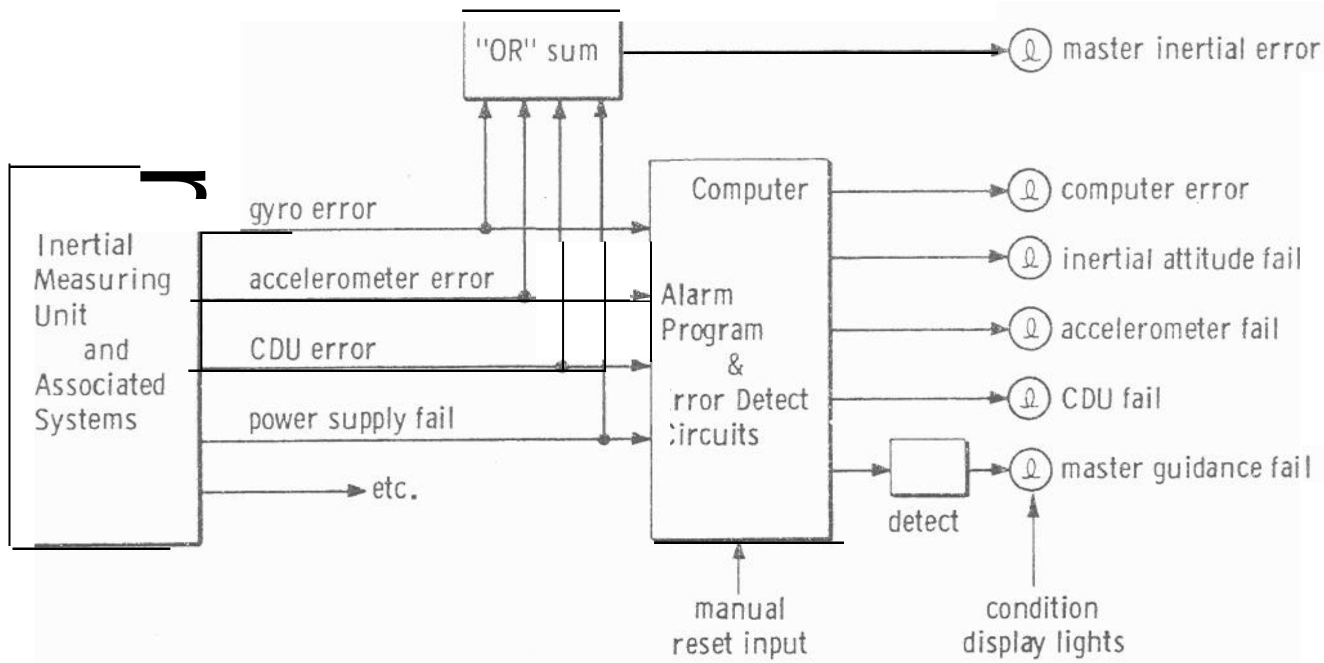


Fig. 5-20 Detection — Command Module Primary Guidance System

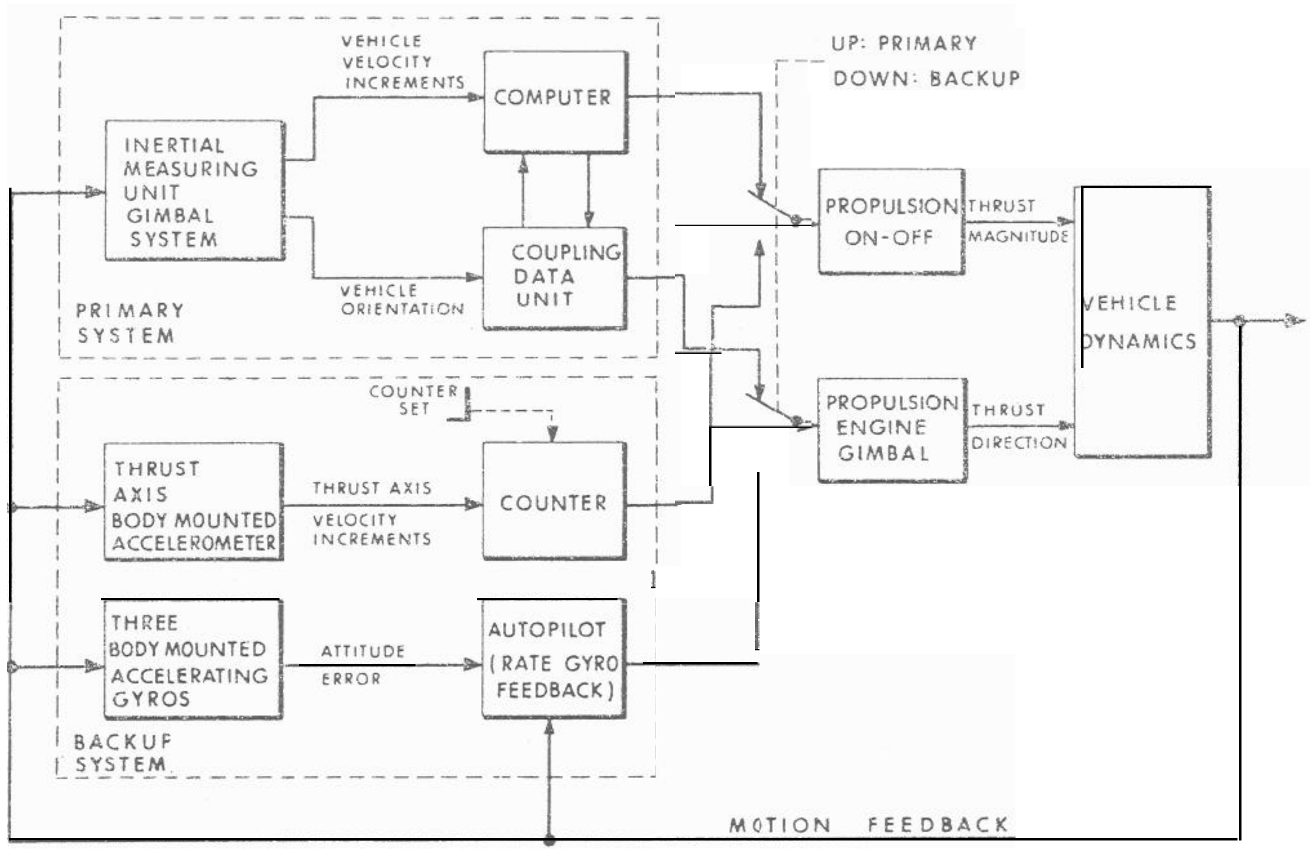


Fig. 5-21 Primary and Backup Guidance Concept

maneuvers to return the crew after the failure of the primary system has aborted the mission.

Figure 5-21 shows a backup guidance system like that of the command module using three body-mounted single-degree-of-freedom integrating gyros to provide attitude-error signals over a limited angular range. These errors are treated as steering commands to a simple autopilot to hold vehicle direction fixed during an abort thrusting maneuver. Engine cutoff is signaled by the integrated output of a single-body-axis accelerometer mounted with its sensitive axis along the nominal-thrust axis. The integrator is a simple preset counter whose output is an engine shutdown signal, when the sum of the accelerometer output velocity increments reaches a level equal to the total velocity change desired of the maneuver. Using data telemetered from the ground, the crew prealigns the thrust axis in the required abort maneuver direction by aiming the vehicle with respect 'LO the stars. The ground instructions recognize the expected offset of the thrust axis from the vehicle roll axis. The maneuvers are accelerations in a fixed direction hut of any magnitude set into the acceleration counter. If large magnitude accelerations are required, then the more limited accuracy of this backup system results in significant errors. These can be corrected by a much smaller maneuver after a short coast based upon the ground tracking of the abort trajectory.

There is a more complex abort guidance system in the lunar module to perform more accurately'the critical and complex abort maneuvers near the moon's surface. This system consists of three body-mounted rate-measuring gyros, three body-mounted accelerometers, and a small computer to perform necessarytransformations of gyro and accelerometer data. This computer also generates steering commands appropriate for abort up to rendezvous with the orbiting command module.

Satisfactoryvehicle control also requires three-axis spacecraft torquing during the free-fall and accelerating phases. Optimum hardware redundancy is provided to drive engine gimbals. The 16 reaction jets on the service module and on the lunar module allow a limited number of jet failures without unacceptable loss of rotational control or translational control. The necessary rotational control of the command module is provided by a redundant assembly of 12 reaction jets for use during earth atmospheric entry. Various levels of automatic, semiautomatic, and manual control can be selected by the crew to keep the subsystems available and working, As the last level of emergency backup, the pilot or a surviving companion can use direct hand-control commands to the reaction jets and engine gimbals with a view of the stars as reference directions.

SECTION 6.0

SPACECRAFT INTERFACES

6.1 COMMAND MODULE INTERFACES

The CM GN&C system, in performing its basic functions of inertial guidance, optical navigation, and spacecraft control, has direct interface with the following six spacecraft systems and the crew:

1. Stabilization and Control System
2. Service Propulsion System
3. Reaction Control System
4. Electrical Power System
5. Environmental Control System
6. Communication and Instrumentation System
7. Crew

A brief descriptionof each of these interfacing systems is provided in the following paragraphs.

The Stabilization and Control System (SCS) is located in the CM and can sense and control spacecraft attitude and velocity changes during any flight phase. Steering and thrust signals can be generated by either the GN&C system or the SCS.

The Service Propulsion System (SPS) provides the thrust capability to change the spacecraft's velocity. The system is normally operated by the GN&C system and is utilized for mission abort, midcourse velocity corrections, and orbital injection.

The two Reaction Control Systems (RCSs) areutilized for spacecraft attitude control and stabilization. One system is required for the CM and the other for the SM. The systems can be operated directly by the GN&C system.

The primary power required to operate the GN&C is supplied by SM fuel cells that are used during all mission phases except entry and recovery.

The Environmental Control System (ECS) is used to sustain life in space. The system provides breathable atmosphere, acceptable temperatures, food and water, waste disposal, and radiation protection. The system also circulates an ethylene-gylcol-water coolant about the temperature-sensitive electronics equipment in the GN&C system and other spacecraft systems to provide thermal stability.

The Communication and Instrumentation System (C&IS) provides voice, television, and telemetry communications with the earth and voice communications between crew members.

6.1.1 Crew Interfaces

The astronaut is an active and controlling part of the GN&C system. Astronaut activities include the monitoring of systems performance, the loading of data for computation, optical sightings, and the selection of operational modes. The astronaut also provides a backup for all systems, thus comprising both an integrating and controlling interface with the GN&C system.

The astronaut plans the problems to which the system will address itself, such as preparing for a velocity correction; he also organizes the major modes (programs) of operation for the problem. Control by the astronaut is accomplished through the data fed to the GN&C system and by response to computer-displayed data. He is thus a major communicator with the system.

6.1.2 Interface Signals

The GN&C system interface can be classified into the following types:

1. Mode and status
2. Engine and jet control
3. Angle data
4. Maneuver command
5. Timing and telemetry
6. Caution and warning
7. Power, reference, and lighting
8. Astronaut interface displays
9. Astronaut interface controls

These interface signal types are defined in the following Interface Control Documents (ICDs):

1. MH01-01380-216, "Command Module Guidance Computer Electrical Interfaces"

2. MH01-01344-216, "Mode Control Signals S/C to ISS Discretes"
3. MH01-01307-216, "CDU to TVC Servo Amps"
4. MH01-01324-216, "Attitude Error Signals"
5. MH01-01325-216, "Total Attitude Signals"
6. MH01-01386-216, "G&N Attitude Error Signals to SATURN Guidance"
7. MH01-01342-216, "G&N system Caution Warning System Interface"
8. MH01-01327-216, "Guidance and Navigation Electrical Input Power"
9. MH01-01388-216, "Interior Lighting Functional Performance Criteria"

6.2 LUNAR MODULE INTERFACES

The LM GN&C system performs the same basic functions as the CM GN&C system and interfaces directly with the following similar systems and the crew:

1. Stabilization and Control System
   a. Control Electronics System (CES)
   b. Abort Guidance System (AGS)
2. Propulsion System
   a. Ascent System
   b. Descent System
3. Reaction Control System
4. Environmental Control System
5. Electrical Power System
6. Instrumentation System
7. Communication System
8. Crew

The CES provides data processing for flight control data during all phases of the LM mission. The CES provides signals to command any combination of the 16 RCS thrusters to stabilize the LM by controlling vehicle attitude and translation during all mission phases.

The AGS provides abort capability from any point in powered descent or powered ascent and increases crew safety by acting as a backup system to the GN&C system.

The LM utilizes separate descent and ascent propulsion systems, each of which is complete and independent of the other. The descent propulsion system utilizes a throttleable, gimballed engine to inject the LM into the descent transfer orbit and to control the path and rate of descent of the LM to the lunar surface, The ascent propulsion system utilizes a fixed, constant-thrust engine to launch the ascent stage from the lunar surface and place it in orbit.

The RCS provides rocket thrust impulses to stabilize the LM during descent, ascent, and coasting flight to control the vehicle attitude and translation about or along all axes.

6.2.1 Interface Signals

The LM GN&C systeminterface signal types are the same as the CM GN&C system (Section 6.1.2), with the addition of the following:

1. Rendezvous Radar
2. Landing Radar
3. Display data
3. Abort guidance system initialization

These interface signal types are defined in the following ICDs:

1. LIS 370-10004, "LGC - LM Electrical Interface"
2. LTS 350-10001, "Total Attitude Signals, Attitude Error Signals, and IMU Cage"
3. LIS 370-10006, "GN&C System to Rendezvous Radar Angle Electrical Interface"
4. LIS 370-10003, "LM-GN&C System Measurement Interface Provision"
5. LIS 390-10002, "GN&C System Prime Power Requirements and Characteristics"
6. LIS 370-10007, "LM-GN&C System 800-Hz Power Electrical Interface"
7. LIS 350-10002, "LM-GN&C System Lateral and Forward Electrical Interface"
8. LIS 300-10002, "Abort Guidance System Electrical Interface with GN&C System."

SECTION 7.0

SPECIFICATIONS, TEST REQUIREMENTS, AND TEST PROCEDURE GENERATION

7.1 TEST PRINCIPLES AND PROCEDURES

The developmental efforts in the MIT/IL System Test Laboratory resulted in a set of system-level acceptance tests for use at the AC Electronics Division facility to assure that completed equipment met specifications. This detailed "test package" was used at each field site as the standard set for GN&C systems prior to spacecraft installation or at any time systems were removed from the spacecraft for laboratory evaluation. Tests were allocated to flight computer fixed memory, tape-loaded erasable memory, and/or manual procedures, depending upon the nature of each test, the characteristics of the GN&C and ground equipment and human operator factors. The initial apportionment was revised from time to time on the basis of operational experience and such factors as the availability of flight computer fixed memory.

Extensive test verification prior to use in critical factory or spacecraft operations minimized delaying schedules. Development of a set of tests for use at APOLLO field sites (North American Rockwell, Grumman, and KSC) in post-installation checkout of the systems in the spacecraft required duplication of the laboratory tests to enable meaningful comparisons among data gathered at various phases and levels. In turn, a comprehensive group of tests was developed to verify every interface and polarity of the GN&C system with other spacecraft systems.

Simulated flight tests were developed to enable reasonablynominalmission sequences to be activated using the onboard mission programs (and, at times, the ground mission control system) to uncover compatibility problems between hardware and software, operational problems, and possible electromagnetic interference problems.

Visits and meetings with NASA and contractor personnel provided for the exchange of information and established mutually acceptable test policies and equipment, software, and facility requirements. Adequate measurement procedures for all testing levels enabled requirements to be met.

Finally, a data processing capability was set up for timeiy analysis and dissemination of test results. Agreements and controls among the various involved organizations prevented misunderstandings and duplication of effort.

7.2 GENERATION OF REQUIREMENTS AND PROCEDURES

Test requirements were prepared to be consistent with system performance specifications, so there could be a sufficient check to establish whether specifications were being met. The underlining criteria for test requirements were the following:

1. Generate requirements critical to the performance of the total mission program.
2. Generate test requirements critical to overall hardware performance at a system level; these were characteristically not directly significant to mission program performance.
3. Test the GN&C system without simulation of any of its components and, hence, without breaking the system harness.
4. Write each Job Description Card (JDC) so that anengineerwith minimal experience in the GN &C system and associated ground support equipment could perform each test.
5. Verify all GN&C system interfaces which are controlled by interface control agreements by signal measurements using simulated loads. The power and servo assembly (PSA), for example, is verified cumulatively throughout the Job Description Cards by observing nominal signal values that verify connections to the pre-calibrated test adapter. The system monitor connector is also verified cumulatively and calibrations compared on the basis of data processing requirements.
6. Each fundamental test is contained in its own independent card series: for example, all accelerometer scale factor and bias requirements are demonstrated in one card series.
7. Maintain continuity of test methods through all phases and levels of checkout.

For preparation of the early Block I system test requirements, complete specifications were not initially available as a package. Information about system, subsystem modules, and component performance became available from design personnel as the design evolved. Thus, the preparation of the initial Block I requirement involved discussion with designers, review of existing design documentation, and the resourceful use of previous experience by test personnel with similar systems. On this basis, initial test requirements were synthesized and routed to designers for correction and comment. Concurrently, more complete system specifications were evolved, providing increasingly sound documentation sources for test requirements.

Preparation of the Block I requirements was difficult; preparing Block II and lunar module system test requirements was more routine. tlssociated measurement and telemetry requirements were prepared concurrently with the test requirements, since the lead times necessary for signal test point buffering and telemetry equipment necessitated early decisions about these requirements.

7.2.1 Major Milestones

The most significant milestone was the completion of Block I test requirements, since Block II and lunar module system test requirements consisted primarily of increments and changes to Block I. (Some new tests requiring substantial work were added; for example, the Spacecraft Controi Test.)

Except for the early Block I requirements contained in ATP101500, system test requirements were corn pleted sufficiently ahead of actual testing to enable an orderly preparation and verification of test procedures. For references, Block I/100 requirements were provided in ATP1026000, Block II/CM requirements in PS2015000, and Block II/LM requirements in PS6015000.

The development of requirements and procedures merged into a continuous effort, with the requirements document being a preliminary form of the procedure.

7.2.2 Problem Areas and Resolutions

The mos-t persistent problem was obtaining sufficiently reliable design information. As the system design became well documented and understood, this problem diminished, but other problems continued.

7.2.2.1 Relation of Mission Requirements to Test Requirements

Although test design personnel were well aware of the overall need to provide tests which would assure that each and every mission requirement could be met, it was never possible to comprehensively relate test requirements to mission requirements. In fact, the definition of specific mission requirements usually lagged actual test operations by many months.

The soundest basis for test specifications has proven to be reliability assurance, as, in the case of inertial parameters, the gyro ADIA term[1] In certain instances, a useful specification was developed on the basis of using three times the RMS combination of estimated hardware performance errors and measurement errors.

7.2.2.2 Relation of Design Configuration to Test Requirements

In order to provide adequate detailed test requirements in sufficient time to allow for timely development and verification of procedures, sufficient information on the system configuration changes was required prior to design maturity. Since detailed changes in system design occurred quite rapidly, approved design changes were frequently not available in time for requirements and procedures to be generated prior to test runs-despite formal configuration control by the Change Control Boards. This problem was solved by frequent contact with designers as the design evclved. Since design changes usually affected the system wiring, for example, intensive collaboration with the wiring harness design personnel gave the most comprehensive early indication of possible change. The subsystem design personnel worked quite closely with the wiring harness interface to provide an early indication of probable changes. In turn, this information alerted test requirements and procedures personnel and enabled them to anticipate probable additional or modified requirements.

7.2.2.3 Verification of Procedures

In the case of Block I Series 0 acceptance test procedures, there was sufficient time-due to the availability of both checkout requirements and test design personnel -to prepare and verify them thoroughly in the System Test Laboratory before they were needed in formal system acceptance at AC Electronics. As the Series 50 and Series 100 Block I systems evolved, design changes occurred so near to the time of hardware production that it became very difficult to validate associated procedure modification in the laboratory. Instead, test procedure personnel worked in direct support of revised tests being run at AC Electronics in those relatively few unavoidable instances when unverified procedures had to be employed. This approach avoided delays due to the formal communication links and mailing times between MIT/IL and AC Electronics personnel. It also provided in-house training for AC Electronics personnel who would subsequently run the system tests; the approach was well received by AC Electronics.

1. See Reports R-568 and E-2141, Part 4, Appendix A, Abstracts.

7.2.2.4 Corrections to Procedures

The formal correction of procedures was essentially a communication problem involving extensive detailed paper work and configuration control. As a result, the final updating of procedures was delayed. Difficulties growing out of such delays were alleviated considerably by MIT/IL resident personnel from AC Electronics, Raytheon, and Kollsman, who helped develop the procedures subsequently used by their fellow employees a-t the system fabrication facilities and field sites.

7.2.2.5 Education of Test Personnel

There was some initial difficulty in -the use of detailed step-by-step procedures for acceptance testing. While the procedures gave all the information necessary for a human operator to run the tests "by the number," they did not explain the system or why a particular test procedure was used rather than another. As a result., improper procedures were occasionally used, often because test engineers departed from the specified procedure for one they -thought better. The outcome can readily he imagined; results were often different from those anticipated and data were out-of-tolerance when the system was actually operating within tolerances. This difficulty arose most frequently when engineers ran initial tests. Those test personnel who tested according to the verified procedures had little difficulty-they "went by the book." Instead of adding explanatory notes to already lengthy procedures, the tests were explained to engineering test personnel who were-urged to use the available documentation. In time, they acquired increased faith in the procedures and this problem largely disappeared.

7.2.2.6 Method of Executing Tests

As early as 1.962, the desirability was recognized of placing certain test sequences or exercises in the flight computer's fixed memory. On the other hand, studies soon revealed that there might be no more than 1000 words of fixed memory available for the test programand that perhaps 200-500 words or erasable memory would be available. At the time, total fixed memory was about 10 000 words. Other alternatives for test execution were to use the flight computer's erasable memory, to use manual sequences, or to use some combination of the first three alternatives. Furthermore, there was doubt whether a means would be provided for routinely loading sequences into erasable memory during post-installation testing. (The planning of the Spacecraft Acceptance Checkout Equipment—later called SPACE-was incomplete and the use of a punched-tape reader was not in favor.)

To resolve these questions temporarily, several ground rules were established:

1. The pre-flight test program in fixed memory would be limited to that required in the computer prototype, AGC4.
2. The feasibility of using erasable memory programs, either with interpretive programing or machine language, would be thoroughly explored and the availability of erasable memory carefully watched.
3. More extensive use of fixed memory test programs would be considered if the fixed memory were increased. (Later it was increased to approximately 38 000 words.)

After screening proposed tests, the initial ones to be executed with the flight computer were chosen: the Gyro Drift Test, the Accelerometer Scale Factor Test, and the IMU Operational & Computer Self-Test. To avoid delaying the test program preparation, it was decided to encourage initiative in writing any important test programs, although they might not be allowed to reside in fixed memory. These early decisions led to the inclusion of much of the important testing in fixed memory and to the fabrication of test ropes.

In addition to the first three tests, other tests programed for flight computer execution were the Gyro Torquing Scale Factor and IMU Gimbal Torquing Test, the Sextant-Navigation Base-IMU, Fine Alignment Test, and the Gyro Compassing Verification Test. The Semi-Automatic Operational Check and the Computer Self-check also became identified as separate tests.

Eventually, as there was continued need for fixed memory to perform the mission programs, major parts of the IMU Performance and Computer Self-Test checks were programed for erasable memory, The IMU Fine Align Test was deleted.

SECTION 8.0
PROTOTYPE SYSTEMS ASSEMBLY AND TEST

The topics of this section are the typical system-level problems that occurred in developing the guidance, navigation, and control equipment prototypes and the solutions thereto that were resolved by the System Test Group. The discussion centers on the hardware development problems and then either the hardware or software solutions implemented as a result of the work in the System Test Laboratory. This focus is different from the hardware-software compatibility testing that took place later in the systems development cycle (1965 onward) and involved checkout of the mission computer programs in the same laboratory (see Section 10.2).

In the following paragraphs the specific problems are identified by the GN&C system on which they first occurred: prototype No. 4, prototype No. 5, Block I No. 104, and Block II No. 200 for the command module; systems No. 600F and 600 for the lunar module. Neither the problem set nor solutions are intended to be exhaustive, but they are indicative of the type and scope encountered. Full descriptions of both problems and solution techniques are available through the office of the Director, System Test Group.

8.1 SYSTEM LEVEL TESTING PROBLEMS

8.1.1 Prototype No. 4

Five changes were documented by the Technical Data Release or Revision board to correct deficiencies in lighting levels and in variation control.

Mechanical tolerance buildup and improper keying arrangements resulted in poor mating when the power servo assembly trays were installed, Symptoms of this problem were intermittent, gyro torquing loop, pulsed integrating pendulous accelerometer loop operation, and so forth. The bolts and the torquing requirements on the holding bolts were changed to insure proper pin mating.

Electromagnetic interference inputs to the computer, or random pulses generated within it, caused discrepancies in program operation. Protective measures were taken in program design to avoid such problems (see Report E-2307). These measures involved the temporary storage of critical data as the computer proceeded through a particular program. If a restart occurred, the computer was forced to begin at an intermediate and known point in the program. Degradation of system performance was thus limited. Unexplained hardware, software, or interference-originated restarts have not occurred in APOLLO flights to date.

8.1.2 Prototvpe No. 5

Multiple inputs to the computer, such as a double ENTER command, were caused by the microswitch contact bounce on operator depression of the keyboard button. This was rectified by instructing the computer programs to ignore double keystroke commands within a minimum specified time.

Accelerometer bias shifted on turn-on as a function of the particular instrument stop which the pendulum had fallen to after last being used. The mechanical stops had been designed at ±5 degrees and were reduced to ±1 degree to reduce this problem. This change minimized the accelerometer loop closure time of which the bias shift is a function.

Themoding of any givenaccelerometer was found to be dependent on the orientation of the others with respect to gravity. This problem was traced to ground currents in the precision-voltage-reference zero lines and was corrected by wiring changes. The changes improved the balance of the O-volt reference of the three accelerometers.

Re-routing of the stable-member wiring harness was required as a result of power distribution inequalities to the inertial components from the 3200-Hz, 2V RMS, 1% supply. This was detected sirnply by out-of-tolerance voltage measurements at the gyro and accelerometer excitation inputs.

The gyro torquing commands had the wrong pulse polarity resulting in inability to achieve the commanded gimbal angles. The power servo assembly and computer interface wiring were changed to correct the pulse polarity.

The accelerometer bias was sensitive to the 120-V power supply level and binary current switch temperatures as detected by GN&C system-level studies of instrument stability. A balance network was included in the binary current switch design and the 120-V power supply was regulated.

The temperature controller, while operating normally, generated an electromagnetic interference output on the O-volt reference and bus power line. A filter capacitor was added to correct this erroneous output.

Redesign of grounding points was necessitated by the electromagnetic interference ouputs from the D/A converters. When the converters were incremented or decremented, there was an increase in the noise level of the output to the subsystem. This problem was solved by r-e-orienting the ground points (i.e., the O-Volt and signal return).

The accumulation of noise hits in the telemetry inlink register required shielding of wiring, rerouting wires, and the arrangement of uplink data to clear the inlink register. The previous shielded wire groupings allowed compu-ter-driver pulses to be coupled into the inlink lines. The wires were regrouped to another shield group.

8.1.3 Block II No. 104

The accelerometer binary current switch was redesigned to eliminate turn-on oscillations which were caused by the unmatched turn-on characteristics of the flip-flop in the previous design. A balance unit was later added to match the characteristics of the flip-flop during turn-on and temperature changes.

An oscillation problem was solved by installing a balance potentiometer to change the enter- range instrumentation group current switching characteristics. This was the same remedy as applied in the preceding binary current switch problem.

In the zero encode mode, a false null in the sextant trunnion loop due to the large saturationvoltage in the fine channel of the 2-speed switch was removed by lowering the saturation voltage from 0.9 to 0.5V.

A phase reversal found on the 800-Hz, 1% supply input of the D/A converter was a design error which was corrected.

In the coupling data unit difficulty occurred in mating a connector and mounting plate because the harness was too short, had to be stretched, and resulted in broken wires. Remedial action was taken.

The optics trunnion in the coupling data unit did not zero within the 15-second specification. No hardware change was made. Based on analysis of the requirements and test method, the test procedures and specification were changed.

The inertial measurement unit did not sit properly in its base because its protruding pad locating pins were hitting base pads. Adjustment was made by relocating the pins upward to avoid possible misalignments and stress upon the case.

Generated noise from the display and keyboard relay switching (400-600 mV peak, 1 MHz) cycled the G&N fail light. No action was taken since it was decided that a legitimate G&N fail would occur before this condition could exist.

In the series 100 workhorse system, checkout and compatibility tests for system No. 121 at North American Rockwell revealed several wiring errors; a dead channel in the trunnion 16x resolver was repaired, and calibration curves were changed to compensate for optics section loading of buffer circuits in the GN&C system.

8.1.4 System No. 600 F

There was a difference in the normalized bias drift term measured with the gyro output axis oriented up and with the axis oriented down relative to gravity. New test positions and a different sequence of positions were used to improve measurement accuracy.

Phase-shifting capacitors in the accelerometer preamplifier failed due to insulation breakdown or pinhoiing during development testing. A new capacitor manufacturer was selected, and more stringent tests were specified to ensure the capacitor would meet the procurement specification.

3.1.5 System No. 600

Erroneous outputs from the rendezvous and landing radars were caused by strobe pulse splitting produced through interface channel circuitry design and interface channel usage by the down-telemetry data-processing technique. A blocking oscillator that would cause all pulses to extend over the possible split pulses was used in the APOLLO 11 landing radar; programing changes were introduced for subsequent missions. The rendezvous radar was not corrected for APOLLO 11, since there were backup modes of operation for avoiding the pulse-splitting problem.

Testing at Grumman determined that reverse power could be applied to the coupling data units and computer as a result of a difference in grounding levels when the command and lunar modules were docked and the lunar module was being operated from the translunar bus. (Ref. AS letter 537-68.) This was corrected by the addition of a new capacitor in the coupling data unit that could tolerate the slight reverse voltage and by replacement of a capacitor in the power servo assembiy filter module with a component that would survive the reverse power. The crew procedures were also changed to ensure that the proper sequence of power up and down would not result in incorrect circuit breakers remaining closed during translunar bus operations.

A series of tests confirmed that the redundant gyro scheme for logical voting control of the rendezvous radar antenna pointing would perform as specified in all modes, including partial failure operation. This test series is fully described in Chapter IV (Radar Subsystem) in Part 2 of the report.

Erroneous accelerometer float motion caused by low-voltage transients was rectified by changing the lunar module battery switchover sequence to minimize the transients. (Ref. STG Memo No. 1080.) These tests also indicated that the computer V-FAIL limit level rather than the inertial measurement unit was the limiting factor on system performance. The V-FAIL limit was lowered in the appropriate lunar module computer by modification to the voltage monitor circuitry.

A series of thermal tests simulated loss of coolant to the inertial measurement unit under flight conditions. Test results were used to define interface control document levels for coolant temperatures (see APM-1717).

As a result of spacecraft-level testing at North -American Rockwell and subsequent testing at MIT/IL, it became apparent that the coupling data unit, while operating normally, had transient errors of a few tenths of a degree (see STG Memo No. 1055 for a complete description). If a transient occurred at the instant that the computer was reading the coupling data unit for a mark, inaccurate data would be stored in the lunar or command module computer. The error probability was about 1/7000. A partial fix was made to the command module flight program by incorporating alarm number 121 to indicate and to reject rapidly changing coupling data unit read counter values. A hardware fix was designed and tested by December 1967 (STG Memo No. 1094) but NASA did not authorize the recommended change.

8.1.6 Block II No. 200

During inertial measurement unit performance preflight testing, the accelerometer output varied nonlinearly due to the gravity input passing through the dead zone at certain orientations with respect to local vertical. The performance test procedure was altered to change stable-member orientation so that accelerometers would always sense a component of gravitational acceleration.

Electromagnetic interference in the form of large spikes on the resolver signal return lines, such as that caused hy switching the ORDEAL, could cause 11.25-degree shifts on the coupling data unit read counter by coupling the interference into the read counter ladder gate. Filters were placed on the spacecraft interface lines to remedy this problem. Also see the paragraph on transients in Section 8.1.4.

The coupling data unit frequently put out extraneous pairs of pulses close together and of either sign. The only important effect of these pulses was in the inertial rate integrating gyro scale factor test. A software solution was developed to look for the uncorrelated extraneous pulses, i.e., to discriminate between the polarity of the pulses at the beginning and end of the scale factor test.

An observable image hop occurred when the optics hand controller was moved initially because of the motor drive amplifier. This was remedied by modifying the amplifier to leave the first stage on and to shut off only the output stage with anti-creep relays. A detailed description of this problem and the implemented solution is included in Chapter III, Part 2 of this report.

The interface between the tracker X- and Y-error signal outputs and the shaft and trunnion motor drive amplifiers was corrected so that the line of sight would be driven to center and track the star image in the field of view.

It was determined that a gyro bias shift would result from leaving -the inertial measurement unit in the coarse align mode of operation for long periods of time (more than 2 hours). This hysteresis effect decayed exponentially after entering the inertial mode of operation. The drift term was a function of the gyro being against its mechanical stop and differed, depending on which stop was encountered. As a solution, the inertial measurement unit was left in the inertial mode for non-use periods, the gimbals were properly oriented, and the unit was monitored to guard against gimbal lock.

Large tolerances in the coupling data unit fail circuitry led to incorrect computer usage of the data relating to synchronization between the inertial measurement and coupling data units with an original computer program usage delay of 3 seconds. This problem was solved by extending the computer non-use delay for the coupling data unit zero, after command from the computer, to 10 seconds.

Multiple test capability for the guidance and navigation indicator control panel alarm lights (i.e., computer-controlled testing and indicator panel-controlled testing) caused all lights to be erroneously turned on when one light was commanded on by the computer relays. This was caused by wiring path interaction and was cured by removing the indicator panel test switch capability with a wiring harness change.

Repeated use of the pushbuttons on the keyboard led to sticking after depression. This was caused by chafing of the mounting shaft and the pushbutton inside diameter. Teflon coating applied to the pushbutton mounting shaft for self-lubrication solved the problem. Other types of dry lubricants were eliminated during life testing of several alternative solutions for this problem.

The sextant trunnion was driven into the stop after hand controller commands had been removed. The anti-creep circuitry was removed so that the amplifier was always powered. The resultant small residual drift of the line of sight was considered acceptable. (See also Chapter III, Part 2.)

Unterminated leads in the command module computer test connector were picking up noise from other active leads in the test connector. The test connector cover was redesigned to provide proper terminations for the sensitive leads whenever the computer was not connected to computer test equipment.

8.1.7 Coupling Data Unit Hunting (APOLLO 11)

During the landing maneuvers on the -APOLLO 11 mission, computer alarms indicated that it was being overloaded with data. This had not occurred previously in mission simulations. The problem stemmed from extraneous inputs from the rendezvous radar coupling data unit when not in the computer control mode during this mission phase and was a direct result of coupling data unit hunting for a match between the read counter and improperly phased resolver angles. The solution was to place the rendezvous radar coupling data units at zero by lunar module computer command until the radar was used in the lunar module computer mode.

SECTION 9.0
GN&C SYSTEM INTEGRATION

9.1 FIELD TEST PHILOSOPHY DEFINITION

The Laboratory's field test philosophy was designed to fit within the ground rules supplied by NASA for Manned Spacecraft Preflight Acceptance Tests. It included the following elements:

1. The missions must provide for astronaut safety and assure the accomplishment of mission objectives.
2. A comprehensive test plan must be worked out before the start of testing.
3. The "building block" approach to testing must be used. 'This meant that no assumption would be made about the operational status of any spacecraft subsystem: the GN&C system was verified functionally before being installed in the spacecraft and again before being operated concurrently with other interfacing systems.
4. End-to-end testing must be done. This provided for the initiating and ending functions to occur sequentially as they would in flight.
5. Interface testing and verification must be done.
6. Mission profile simulation must be designed to reproduce actual flight conditions encountered.
7. Astronaut participation must be an integral part of the system during testing.
8. Diagnostic assistance must be provided for fault isolation and correction prior to flight.
9. Test repetition must be included for operational confidence.
10. Test results must be evaluated against the test plans and specifications for unusual operation.

Within the overall test philosophy, specific objectives and requirements were developed to:

1) operate all system modes,
2) test all parameters,
3) check all interfaces-electrical and manual,
4) permit no undetected malfunctions, and
5) provide prelaunch initialization.

A. G & N SYSTEM TESTS

1 OPERATIONAL CHARACTERISTICS
2 AGC FUNCTIONAL TEST
3 OPTICS FUNCTIONAL TEST
4. FUNCTIONAL MODES TEST
5 INERTIAL COMPONENT PARAMETERS CHECK
6 ALIGNMENT TESTS
   a. SPACE ALIGNMENT
   b. PRE-LAUNCH ALIGNMENT
7 PILOT FUNCTIONS

B. COMBINED SYSTEM TESTS

1. FUNCTIONAL COMPATIBILITY
2. SIMULATED MISSION TESTS
3. INTEGRATED SYSTEMS TESTS
4. POWER TRANSFER TESTS
5. RF COMPATIBILITY
6. AGC CLOCK ALIGNMENT

Fig. 9-1 G&N System Spacecraft Functional Test

The corollary objectives included:

1. Testing without exceeding limit-life times
2. Doing end-to-end tests and polarity checks
3. Maximizing test similarity
4. Maximizing self--test capability
5. Decreasing acceptance checkout equipment requirements
6. Decreasing manual operation
7. Decreasing test time
8. Increasing inflight error detection

The GN&C system tests provided for the checkout of operational modes, interfaces, and performance testing. Among the tests were star-landmark measurement, inertial measurement unit alignment, inertial reference, and acceleration measurement. Figure 9-l shows the scope of GN&C system functional tests. Typical test flow and sequences are shown in Figures 9-2, 9-3, and 9-4.

9.2 DEFINITION OF FIELD SITE REQUIREMENTS

MIT/IL defined the requirements for the field support of GN&C systems based on past experience at Cape Kennedy with the Fleet Ballistic Missile Program. Field operations included all the activities necessary to furnish manpower, materiais, and equipment to support a GN&C system at the site: program management, facilities, logistics, personnel, training, transportation, packing, maintenance, repair, procedures, and field documentation. Field operations also included the reporting of all ground and flight test data. The designated field sites and MIT/IL activities were Kennedy Space Center (supplying preflight testing and launch support), Manned Spacecraft Center (supplying support to special test vehicles and mission simulations), and at the associated contractors-Grumman Aerospace Corporation, Bethpage, N.Y., and North American Rockwell, Downey, California (supplying support for GN &C system checkout prior to installation and spacecraft checkout support after installation was accomplished).

9.2.1 Original Site Management Structure

Individual field sites were originally directed by an MIT/IL site manager who was assisted by an administrator and a chief engineer. Each participating contractor (AC Electronics, Raytheon, Kollsman) had a lead or senior engineer in charge of its personnel who reported directly to the site manager. The contractors determined the type of work and quantity of people required. Each contractor was specifically

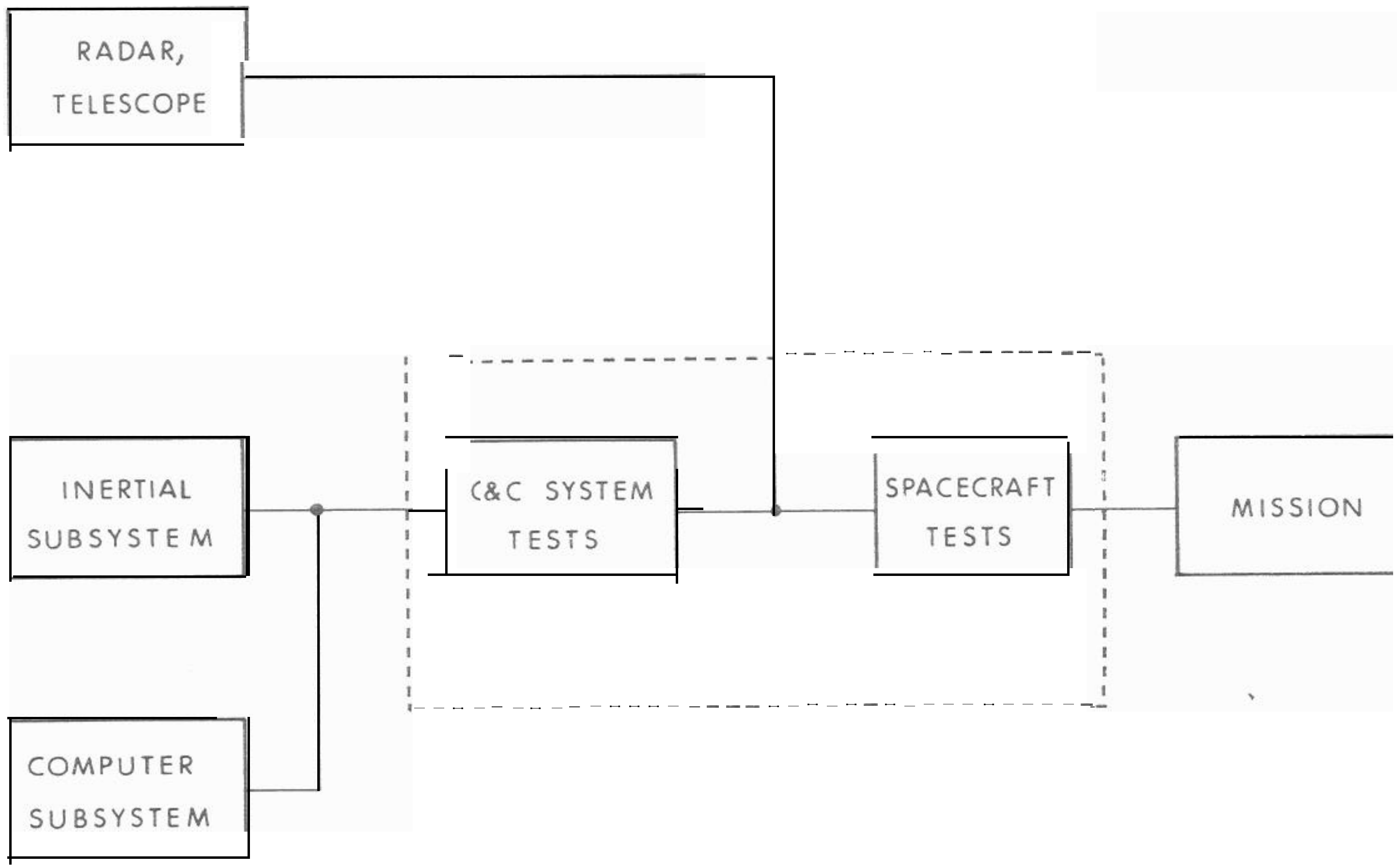


Fig. 9-2 G&C Test Flow (LM)

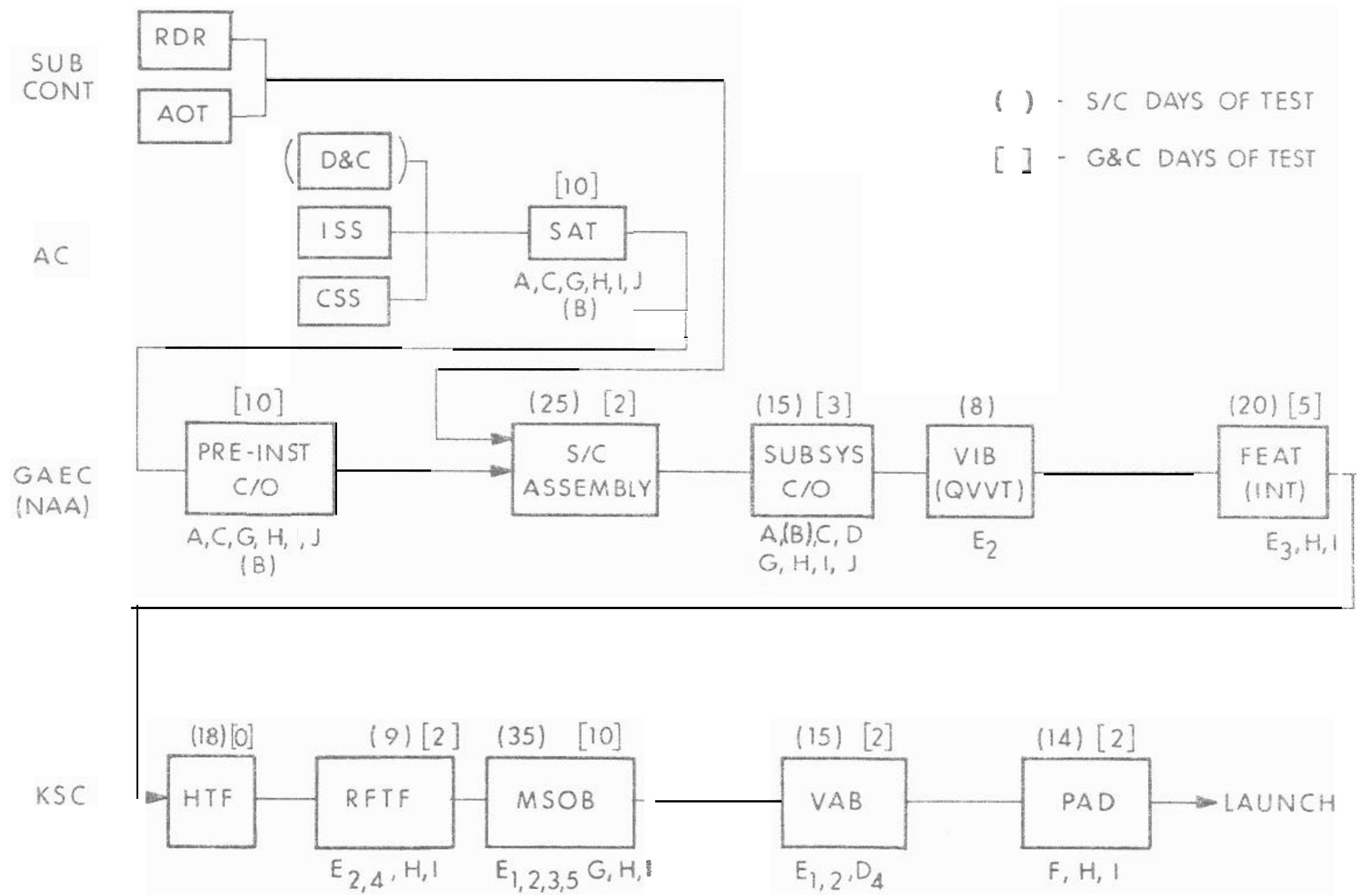


Fig. 9-3 G&C — LM Test Sequence

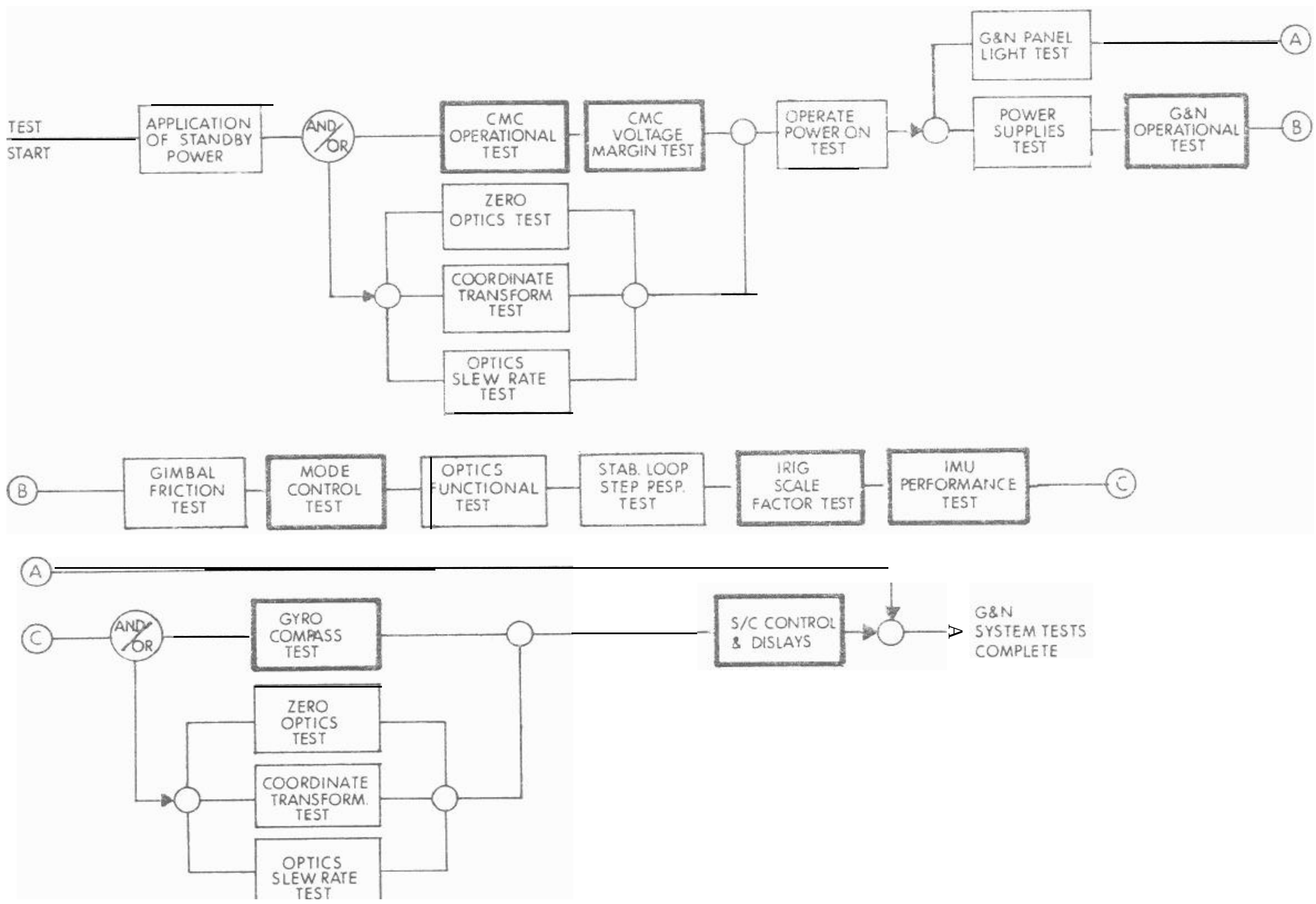


Fig. 9-4 CM G&N Individual Systems Test Sequence

responsible for its end item and its functions: Raytheon for computer functions, Kollsman for optics functions, and AC Electronics for the inertial and system functions. AC **Electronics** had an additional function to provide guidance for the system testing team. Originally MIT/IL supplied approximately 20 percent of the personnel, Raytheon 25, Kollsman 15, and AC Electronics 40.

The logistics function was the responsibility of AC Electronics; its personnel maintained a central file **and** reporting point for all field logistics information such as spares on hand and expended, tests accomplished, failures incurred, shipping information, usage data, test data sheets, and so forth.

AC Electronics was also responsible for chairing each of the provisioning conferences attended by NASA, MIT/IL, Kollsman, and Raytheon and for integrating the requirements and reporting conference results. Preparation and revision of field-use documents-maintenance and repair manuals, checkout manuals, familiarization manuals, launch operations and flight operation manuals-rested with AC Electronics.

The policy from the Manned Spacecraft Center that the management of the field programs should be as close to the design source as possible resulted in **its giving** MIT/IL overall responsibility for field site testing and development. Fulfilling this responsibility required the intricate coordination of three separate contractors. Theresponsibility could not have been readily placed elsewhere in the early formation phases of the Project: design, reliability, safety, and mission success all argued for this decision.

9.2.2 Revised Site Management Structure

In April 1964, as the APOLLO effort moved toward its peak year of effort, 1965, NASA gave new organizational direction to field operations. Field sites were placed directly under AC Electronics contractual control. The site manager would be an AC Electronics employee. MIT/IL would not exercise detailed direction or administration of routine tests; that function would reside with the AC Electronics site manager. The senior MIT/IL representative would be the test operations director (TOD); he would have a small complement of four or five engineers. This staff would work directly with XC Electronics, Raytheon, and Kollsman engineers on the various site activities. The test operations director and his staff would thus obtain for MIT/ IL the required visibility into system performance. In addition, the director would be the single contact point for the **flow** of technical information relevant to the APOLLO GNRC systems at the test site. The 1964 reorganization has remained in practice essentially unchanged.

9.3 LABORATORY FACILITIES

Approximately 5200 square feet of laboratory space were required. To prevent excessive handling of the systems, all working areas were to be contiguous, and all laboratories were to be located as close to the spacecraft as possible. MIT/IL was responsible for selecting all items—benchs, racks, rotary tables, surface plates, and other equipment.

The guidance checkout laboratory, approximately 3000 square feet in area, was designed primarily to test the GN&C system before installation in the vehicle; the same area was also used for diagnostic tests. Two systems could be worked on or checked out simultaneously. The environment specification for these facilities conformed to Federal Standards No. 209—particle size class of 100 000; temperature 73° ±2°F; humidity 45 ±5%; and ambient pressure above the surrounding area.

The system test laboratory's equipment, excluding the precision rotary tables, was placed at the level of a IO-inch raised false floor. The floor panels were removable; each 2-foot square section was capable of supporting at least 250 lb/ sq ft distributed load and 1000 lb/sq ft concentrated load. The ceiling height was to be a minimum of 10 feet above the false floor. This height was required to permit installation of two mechanical monorails of l-ton capacity with a minimum hook height of 90 inches. The monorails (with hoist) were required to move a GN&C system on its ground support equipment holding fixture through the laboratory and onto the rotary tables. Each laboratory required a North survey to an accuracy of 5 arcseconds for placing optical targets and gravity determinations to 10^{-6}g accuracy for inertial testing. Stable-mounting base level requirements were to 2 arcseconds per hour.

The original laboratory plans included a guidance repair laboratory of approximately 700 square feet, equipped for failure analysis and the authorized repair and modification of systems, subsystems, and assemblies. A thermal hood and oven permitted opening the inertial measurement unit or other components with special temperature requirements. The repair laboratory activity was discontinued when NASA directed that repairs were not to be performed at any field location. Instead, field problems were to be resolved by isolating the problem to the black-box level. After identifying a given computer, inertial measurement unit, etc, it was shipped back to AC Electronics. A replacement unit was shipped to, or available at, the field site to permit testing to proceed. AC Electronics would then analyze the failed assembly to determine the exact cause of field failure. In practice, spare units were always kept at the field sites in order that spacecraft-level testing not be held up by the lack of replacements.

The general repair laboratory (500 squat-e feet) was developed for the repair of guidance ground support equipment and general laboratory equipment and for cleaning, welding, potting, and depotting. The guidance shipping and receiving area (600 square feet) was used for the receipt and unpacking of guidance equipment and all support units required for field site operation. All shipping and receiving were done in this special area; sufficient space was also allocated for equipment storage.

The bonded storage area was used for all flight hardware, ground support equipment, and test equipment. Engineering and administrative offices were usually located away from the laboratory and the spacecraft assembly area and were similar to the offices supplied for contractor personnel.

9.3.1 Sorth American Rockwell Field Site

The first. operational field site was at North American Rockwell. AGC 3 was the first system to undergo testing there. The North American Rockwell objectives were to verify test procedures in a field environment and to certify the ground support equipment, Information was collected about personnel requirements, test times, and site and system problems. The first system arrived at North American Rockwell on 19 June 1964 and posed the following types of problems:

1. The first system did not meet all acceptance test procedures (ATP) on performance characteristics such as tolerances on excitation voltages, power supply tolerances, parallelism of optics lines of sight, etc.
2. There were assorted mechanical problems associated with module mounting screws, connectors for coolant, problems with obtaining shipping containers, and so forth,

The AGC 4's total operating time on arrival was approximately 600 hours. Its total failures were 10.

The following types of facility problems were noted during site activation:

1. The optical targets were misaligned, Corrective measures had to be taken also to align the GN&C mounting fixture relative to the height of the targets,
2. The holding fixture required modification due to mechanical interference,
3. Condensation dripping from the inertial. measurement unit and the water-glycol lines required coolant line insulation and inertial unit insulation to prevent water drippage during spacecraft level testing. The problem occurred because the test area air conditioning system did not meet specifications.

9.3.2 Grumman Aircraft Engineering Corporation Field Site

The Grumman field site laboratory became active in January 1966 when PI-3, a production prototype, was utilized to certify the ground support equipment. GN&C System 601 arrived in March 4966 and was checked out in the laboratory and made ready for spacecraft installation in the Grumman, ESI rig, and later spacecraft LTA-1. System turnover to Grumman was made in July 1966.

The high-humidity problem also occurred in the Grumman laboratory; dripping from coolant lines and the inertial measurement unit was the result. The air conditioning system 's capacity was exceeded during the summer months; during the cooler months the problem vanished.

The most difficult facility problem was the cleanliness requirement. The original requirement was the maintenance of a class of 100 000 level throughout the guidance repair laboratory, systems test laboratory, and flight hardware storeroom. Grumman proposed that the air conditioning for the three areas meet a cleanliness level of 200 000 particles per cubic foot. It was agreed that the cleanliness level in the system test laboratory and the guidance repair area was of significant concern and it was recommended that any critical work in the guidance repair area be conducted at one or more l.aminar-flow clean-work benches. These benches would supply air directed at the workpiece with a contamination level of 10 000 to 100 000 particles per cubic foot,

The problem was resolved through the use of laminar-flow benches in the repair laboratory and laminar-flow tunnels that projected clean air over the rotary table area, so that the cleanliness level in the system checkout laboratory could be maintained during systems test.

9.3.3 Kennedy and Manned Space Center Field Sites

The Kennedy Space Center facility certification test runs activating the site were made on GN&C System No. 8 in February 1966, Laboratory humidity was too high, causing condensation; the optical piers were inaccurately aligned requiring correction.

The Manned Spacecraft Center facility was activated with the validation of GN&C System 109 in February 1966. The problems reported at other facilities did not occur at the Manned Space Center, indicating that the learning process in site activation from facility to facility was beneficial.

SECTION 10.0
SYSTEM TESTING

There are an important set of performance parameters to be measured in system-level verification of the Guidance, Navigation, and Control equipment at the manufacturer's facility, the spacecraft contractors' site, and at the launch site. Such parameters include the functioning of the inertial components (e.g., gyro drift, and scale facto:), the status of the optics-for instance transmitting sighting marks to the computer-and the operational integrity of the computer, including interfaces with **other** spacecraft systems. This section of the report outlines the types of computer-contained systems tests that were developed, problems encountered in development, and the solutions implemented. Detailed descriptions of the test techniques may be found in the first section of the guidance system operations plan for the command or lunar module (K-577, 567).

In retrospect, using the flight computers for the vehicle of system testing was workable. Generally, it provided for standardization of test techniques at all three levels of APOLLO testing-the laboratory, the installation, the launch site-and it permitted technical control of test design by MIT/IL. Moreover, the computer-contained checks were highly automated and thereby provided for repeating tests as many times as necessary to evaluate trends in system performance very near the time of launch.

The central problem **with** the system tests was that they used fixed memory in a flight computer already crowded with flight programs and data. Thus, the general lesson learned was properly sizing the computer to include system testing. The interim solution for the APOLLO case was to move the tests to the erasable memory part of the computer where required.

In support of the mission computer program development, the System Test Group performed a series of hardware-software compatibility tests. This series, a task which is prerequisite to computer program release, is also summarized in this portion of the report beginning with Section 10.2.

10.1 COMPUTER-CONTAINED TESTS

Responsibility for the design, coding, testing, and subsequent release of the test routines for development testing both at MIT/IL and at the APOLLO field sites was given to the System Test Group. The requirements for each of these tests were formally documented inmemoranda by the staff in the group. The memoranda were discussed among NASA, MIT/IL, and the APOLLO contractor; and changes were made as required. Using these memoranda as a basic guide, the required test routines were coded and the following computer ropes were manufactured to support the prelaunch checkout:

Name	System	Release Date
SUNRISE 38	BLK I	Not Released
SUNRISE 45	BLK I	28 November 1964
SUNRISE 69	BLK I	15 March 1965
AURORA 85	LM BLK II	15 March 1966
AURORA 88	LM BLK II	15 July 1966
SUNDIAL B	CM BLK II	16 June 1966
SUNDIAL C	CM BLK II	24 June 1966
SUNDIAL D	CM BLK II	1 March 1967
SUNDIAL E	CM BLK II	13 March 1967

Each of the test ropes contained various interface routines common to all. Such routines included INTERPRETER, EXECUTIVE, WAITLIST, PHASE TABLE MAINTENANCE, FRESH START and RESTART, T4RUPT including DSKY servicing, IMU TURN-ON, FAILURE MONITOR, OPTICS servicing, IMU MODE SWITCHING, OPTICS MARKING, EXTENDED VERBS, UPLINK and DOWNLINK, ALARM and ABORT, INFLIGHT ALIGNMENT, PINBALL, AND AGC SELF-CHECK. The Mission Development and Digital Development Groups were responsible for these routines. They were assembled on the MH 1800 computer and checked out using both the digital simulation method and a prototype computer in the Digital Development Laboratory. After successful verification of the basic routines, work began in early 1964 on the SUNRISE test routines.

10.1.1 SUNRISE 38 Testing

SUNRISE 38 contained the aforementioned interface routines; SUNRISE 45 was the released version. SUNRISE 69 used four additional memory banks containing the following required system test functions:

1. GYRO DRIFT-Determines the drift terms including the normal excitation total bias drift (NBD), acceleration sensitive drift along the positive gyro input axis (ADIA), acceleration sensitive drift due to a case acceleration of 1 g along the positive gyro spin reference axis (ADSRA),

2. PIPA SCALE FACTOR-Determines the average scale factor and acceleration bias for each accelerometer.
3. SXT-NB-IMU-FINE ALIGNMENT—Measures the system's ability to align to coordinates defined by optical sightings or to entry of known navigation base orientation.
4. IMU OPERATIONAL-Designed to provide a quick check into system operation and performance; the test is particularly useful as a tool to establish effects of electromagnetic interference on system operation.
5. IRIG SCALE FACTOR-Determines the average torquing scale factor for each gyro.
6. DSKY CHECK-Checks each of the electroluminescent elements, pushbuttons, response, lighting, and the resetting of the lamps on the display and command keyboard.
7. C-RELAY CHECK-Exercises each of the 33 relays controlled by the OUTO register by turning them on and off in sequence.

10.1.2 SUNRISE 69 Testing

In addition to the seven test routines above, SUNRISE 69 contained the PRELAUNCH ALIGNMENT and GYROCOMPASS program. This program uses the gyrocompass loop to maintain the stable member's Y-Z plane horizontal and torques about the X-axis to seek and maintain a specified azimuth.

10.1.3 AURORA Testing

Work began on the AURORA test routines in late 1965. The AURORA ropes were designed for the lunar module and utilized the Block II computer. Five sets of AURORA 85 were manufactured for distribution to the various field sites. A few improvements and corrections made to AURORA 85 were incorporated into AURORA 88. Four sets of these ropes were manufactured. Changes and corrections were made to such items as the fail-detect routines, NO ATT lamp, gimbal lock protection, coupling data unit synchronization time, lunar module computer standby, rendezvous and landing radar routines, alignment optical telescope reticle tilt, and system test routines.

In addition to the GYRO DRIFT, PIPA SCALE FACTOR, AOT-NB-IMU FINE ALIGNMENT, IMU OPERATIONAL, and IRIG SCALE FACTOR tests described in the SUNRISE rope, AURORA contained the following tests:

1. SEMI-AUTOMATIC MODIKG-This test is used to verify IMU CDU moding, rendezvous radar CDU moding, CDU repeating accuracy, CDU

command rate, FDAI commands, GASTA, Velocity Meter, and radar positioning commands.

2. AOT ANGLE CHECK-Provides a simple means of verifying the accuracy of the telescope.
3. LM FLIGHT CONTROL SYSTEM TEST-Checks interfaces by which the computer controls the lunar module including:
   a. IG RCS jets and eight JET FAIL switches
   b. On-Off control for ascent and descent engines
   c. Trim position control of the descent engine gimbal
   d. Throttle control of the descent engine thrust
   e. Monitors changes of signals to input channels 30-33.
4. SEMI-AUTOMATIC INTERFACE-Exercises the digital interfaces with the spacecraft control and display groups.

The EXTENDED VERB section of this rope also contains routines for positioning the rendezvous radar, for driving rendezvous and landing radars to position 2, and for reading data from both radars. These radar routines are very similar to the routines that are actually used in manned missions.

10.1.4 SUNDIAL Testing

The first test program for the Block II command module guidance system was termed SUNDIAL B. Various changes and additions resulted in the release of SUNDIAL C, D, and E. The GYRO DRIFT, PIPA SCALE FACTOR, SXT-NB-IMU FINE ALIGNMENT, IMU OPERATIONAL, and IRIG SCALE FACTOR tests described for the SUNRISE rope are also included in the SUNDIAL ropes. In addition, the following tests were included:

1. SEMI-AUTOMATIC MODING-This test verifies IMU CDU moding, CDU repeating accuracy, CDU command accuracy, CDU command rate, and attitude error positioning commands.
2. SEXTANT ANGLE CHECK-Provides a simple means of verifying the accuracy of the onboard sextant.
3. CSM and SATURN INTEGRATED-Checks interfaces by which the computer controls the command service module and SIVB including:
   a. CM/SM RCS Jets
   b. SPS engine on-off control
   c. Trim position control of SPS engine gimbal
   d. Monitors changes of signals to input channels 30-33
   e. SIVB steering.

4. SPACECRAFT CONTROL, and DISPLAY INTERFACE-This test consists of three parts:

   a. FDAI INTERFACE-Computer generates attitude error signals to produce an attitude error display on the ball and needle indicator.
   b. SPACECRAFT CONTROL of SATURN-Computer generates attitude error signals for the guidance of the SIVB via the inertial subsystem D/A converters.
   c. THRUST VECTOR CONTROL-Computer generates analog signals for positioning the SPS engine gimbals through the optics coupling data unit D/A converters.

5. SEMI- AUTOMATIC INTERFACE--Exercises the digital interfaces with the control and display groups.

10.1.5 Verification Problems

Verification of the system test routines was based on individuals being made responsible for the routines that they had coded. Those test designers who had not coded their own routines also assisted in verification. Weekly reports also provided for allocation of GN&C system testing for the following week.

Some of the problems encountered during SUNRISE verification were also experienced during the AURORA and SUNDIAL programs, although on a somewhat smaller scale. Time was the biggest problem. The test programs required extensive testing, but there was a limited period of time to accomplish it before program release. Effective use of the GN&C system in the System Test Laboratory did ease this problem. With only one system available for the testing of the SUNRISE, AURORA, and SUNDIAL routines, operating time had to be stringently controlled.

Other problems encountered, not necessarily in order of importance, were:

1. Learning to use INTERPRETER correctly.
2. Low reliability of the paper tapes used for the core rope simulator (CRS).
3. Reliability of the simulator hardware. Fortunately, qualified personnel were available for troubleshooting but, nevertheless, valuable time was lost.
4. Time required to get a paper tape punched and to get listings of the latest program assembly.
5. The input deck format for digital simulation was always a problem since the environment programs were constantly changing.

Most of the problems encountered were caused by the continuing development of each component routine. Problems were overcome with a minimum of lost time because (1) paper work for the testing was minimal; (2) groups involved in coding and testing were in the same building enabling direct communication of changes in environment programs, input deck format for digital simulation, program assembly, and bench testing, and (3) the smallness of each group involved in assembly changes enabled optimum awareness of the changes and the effect on their own routines.

Program verification was accomplished using the guidance computer monitor (CRS) in the System Test Laboratory and using the MH 1800 computer for digital simulation.

10.1.6 AGC Monitor/Core Rope Simulator

The guidance computer monitor has two functions; it can be used as a computer monitor to debug programs under development; and it provides core rope simulation -replacing all, or part of, the fixed memory with a read and write memory. Loading the monitor memory is done automatically, using perforated paper tape or magnetic tape, or manually using the keyboard on the display panel. Tape loading allows a rapid loading of the entire memory, while manual loading permits the loading of small programs or the revision of data already loaded.

The monitor may be used with either the Block I or II computer configurations. It has control and monitor capabilities over the computer's operation. The time counter can be stopped under a variety of conditions; data within the computer can then be monitored, and a transfer of control can be initiated from the monitor. In addition, the monitor may be used to load and read the erasable memory using paper or magnetic tape or to dump the contents of the erasable memory onto magnetic tape. All. these functions were used extensively as diagnostic aids during program development.

10.1.7 Digital Simulator

Digital simulations run on the MH 1800 involved transferring the computer program to the disk; the desired routine was executed using machine instructions and timing to simulate an actual computer. In addition to running the program, inserted input cards gave added capability-special request, printing during the run, post-run dumps, instruction-by-instruction printout, and running with a simulated environment. The output of each run could be saved for later editing if investigation into the present output revealed that further data were necessary. Changes could also be inserted into the program if there were no problems in the present revision.

No erasable memory programs were developed for the SUNRISE, AURORA, and SUNDIAL programs, with the exception of the short routines to check out the alarm circuitry. RESTARTS were generated via TC TRAP, RUPT LOCK, PARITY FAIL, and NIGHTWATCHMAN, using short erasable routines. The INTERPRETIVE and BANK switching routines in SUNRISE, AURORA, and SUNDIAL were not compatible with programs executed from erasable memory. During this development phase, it was assumed that fixed memory space would be adequate for the system test routines in flight ropes. If inadequate, a module would be inserted containing the system tests compatible with that particular flight rope. This test module would be replaced by the flight module following completion of the desired tests.

10.1.8 Computer Restart Monitor

A special test connector "cover" was fabricated for the Block II computer in the spring of 1967. The features included in the "cover" were: nine flip-flop registers set by computer alarms (available to interrogation as Channel 77 in the computer) and loading of all test connector inputs with 270 ohms to ground. By interrogating Channel 77 after a RESTART, the type of restart (PARITY FAIL, TC TRAP, etc.) could be determined. This "cover" was never used in field testing because it was not flight-qualified and therefore altered the flight configuration. It would have been useful during field testing, because of the large number of restarts caused by the K-START tapes needed to perform the required spacecraft interface tests.

10.2 HARDWARE-SOFTWARE COMPATIBILITY TESTING IN SUPPORT OF MISSION PROGRAMS

10.2.1 Specific Test Information

There were six categories of compatibility testing: prelaunch alignment, optics operational programs, command and lunar module alarms, lunar module inertial measurement unit alignment, radar, and the miscellaneous tests which included inertial measurement unit checks. The complete objectives, descriptions, initializations, and results of all these tests are found in System Test Group memoranda 1035, 1120, 1178, 1207, 1225, 1268, 1321, 1323, 1340, 1343, and 1348.

In the following paragraphs each of the categories of tests is briefly explained as to type, technique, and purpose. This is followed by an outline of some of the compatibility problems encountered and their resolutions, 10.2.1.1 Command Module Prelaunch Alignment

The computer programs associated with aligning and gyrocompassing the inertial platform (P01, P02, P03) prior to launch were tested with the GN&C system in the laboratory. Optical targets mounted on piers at known azimuth and elevations were used to verify the results. Also, a test was performed by removing the inertial measurement unit from its mounting fixture and performing stable member alignment with the case at the actual launch azimuth. Both normal and stress (off-nominal) tests of the programs were performed in the laboratory and on the all-digital simulator.

10.2.1.2 Command Module Optical Program Tests

All the mission programs and routines which involve use of the command module optics were verified for hardware-software compatibility. This included the programs and routines associated with inertial alignment (P51), realignment (P52), rendezvous navigation sightings (P20), landmark navigation (P22), and midcourse navigation (P23).

Duplicating mission conditions exactly in the laboratory was not praticable nor was it possible to obtain precisely accurate performance measurements due to limiting factors such as the single degree of freedom in the mounting fixture rotation. Rather than performance limits, the testing verified that the proper optics and program functions were compatible in terms of the sequences required by the guidance system operations plans. For example, by using the K-START tape reader to initialize the computer and represent a reasonable flight situation, the conditions could be established for checking the polarity and scaling between hardware and software.

Typical flight situations which were simulated for the optics operational programs included realigning the inertial measurement unit (P52) after having performed the S-band pointing routine (R05) to earth and the optics trunnion calibration (R57). The. initial conditions for this phase of the testing were based on conducting the prelaunch (P01), gyrocompassing (P02), and earth orbital insertion monitor (P11) programs to simulate an actual sequence of events.

Also in the testing were the checks of the program alarms and astronaut options associated with optics operations. Thus, all program branches were checked in the compatibility testing.

Inertial orientation determination of the inertial measurement unit after turning it on is accomplished with Program 51. This program was verified in the compatibility testing as was P23, the midcourse navigation program that would be needed for self-contained spacecraft navigation in the event of loss of communications.

Simulating optical rendezvous program P20 to verify compatibility was a rather complex matter involving the initialization of the command and lunar modules in carefully constructed orbits at known times at known inertial orientations. By using this information, it was possible to predict the correct operation of the program in terms of the required optics shaft and trunnion pointing angles from command to lunar module as a function of time. Additionally, it was possible to predict the changes in lunar module state vectors which would result from the simulated sightings.

10.2.1.3 Command and Lunar Module Alarm Tests

Methods were developed to trigger each program alarm associated with the hardware. The alarms were generated in a manner as near actual flight situation as possible and hardware failures were simulated. Some of the alarms were generated by special tests and some as parts of other general tests.

An example of a special test is the simulated failure of an accelerometer in the inertial measurement unit which is expected to cause display of its associated alarm code No. 205. The register (DELVX) which accumulates X-accelerometer output pulses is loaded to its maximum capacity. Then the routine which periodically observes the accelerometer outputs (SERVICER) is called by a special erasable program. When the SERVICER routine detects that the accelerometer events are at a maximum-that is, beyond the number of pulses which would be accumulated in a normal thrusting maneuver—then a program alarm light informs the system operator. On request, the alarm code No. 205 is displayed as the required test result.

An example of a program alarm which was exercised during general testing was the optics alarm, code No. 404, “Target out of view,” which is used to indicate that the computer-contained automatic optics routine was not able to locate a suitable star in the field of view at a particular spacecraft inertial altitude. This exercise is typically performed during the inertial measurement unit realignment program (P52) testing and is evaluated by selecting stars which are known not to be within the optics field of view at the particular inertial altitude of the test situation.

Using either the special test or the general test approach, all the alarms were checked for both the command and lunar module computer programs (e.g., COLOSSUS, LUMINARY). More than 70 COLOSSUS alarms and almost 90 LUMINARY alarms were tested with these techniques, the exact number depending on the particular flight program and revision under test.

10.2.1.4 Lunar Module Alignment Program Tests

The computer programs were tested for aligning or realigning the inertial measurement unit in the lunar module. These programs were inflight inertial orientation determination (P51), inflight realignment (P52), and lunar surface alignment (P57). As part of this work, the routines associated with alignment and realignment were checked also: fine alignment (R51), automatic maneuvers of the spacecraft to the star sighting attitude (R52, auto-optics positioning routine), the sighting mark routine (R53), the sighting data display (R54), the gyro torquing routine (R55), the lunar surface sighting marks, and the automatic maneuver (R60).

Verification of the programs and routines above included exercising the astronaut checklist codes, options, and alarms. In the alarm cases, those which had not been checked as part of the special tests previously mentioned were evaluated. There was, however, some redundancy in testing the alarms by both techniques.

An interesting aspect of the lunar module alignment programs compatibility checkout in the system test laboratory was that there were no alignment optical telescope or optical targets included; this equipment was entirely simulated by appropriately designing the individual tests. The absence of integrated optics with the computer and inertial platform in the program verification made the simulations more difficult and moved system testing requirements (hardware) to the spacecraft manufacturer's facility where the guidance, navigation, and control equipment was installed in the vehicles (e.g., AOT accuracy test, fine alignment testing). In retrospect, a more complete testing facility comprised of a totally integrated system on the rotary table would have been appropriate.

One example of the compatibility testing on the lunar module programs is the exercise of the inflight realignment. In this case the stable member was aligned relative to the basic reference frame (i.e., via REFSMMAT) with the inertial measurement unit mounting fixture positioned so that the simulated telescope would be pointing in a known direction. Simulated sightings were then made to a known planet and star with appropriately timed mark sequences and a half-unit vector of the selected planet loaded into the computer. The desired spacecraft maneuvers, star data test results (i.e., difference between known and measured included angle between celestial bodies), and gyro torquing angles calculated by the computer were then compared with the precomputed values of the same parameters.

Initialization data for the test above included simulating the lunar module orbit in a moon- centered reference coordinate system and loading the viewing detent calibration for use of the alignment optical telescope. All the computer flagwords which would be normally set in actual flight were also set in the simulation. In effect, effort was made to represent as closely as possible the configuration and operation to be encountered during the mission.

Limitations of the realignment tests were in the effects of earth rate on the inertial measurement unit case, and the initialization with gimbal angles set by the coarse alignment method, thus resulting in slight errors. However, the test results were within 0.25 degree of the predicted value for the Y-gyro and 3 percent for the X- and Z-gyros. Moreover, the testing was comprehensive to the extent that the polarities and scaling of all interfaces between the inertial measurement unit, the telescope, and the realignment program were verified; also successful completion of the test required processing of the sighting mark data and calculation of the required gyro torquing angles exactly as in the APOLLO mission.

Other tests similar to the type above were performed to provide assurance that the stable member could be realigned to various inertially specified reference frames; e.g., a preferred orientation for a thrusting maneuver.

10.2.1.5 Radar Programs

The programs and routines associated with use of the rendezvous and landing radars were tested. The lunar module guidance computer was initialized with state vectors that simulated orbits of both vehicles and then the rendezvous radar response during tracking was observed and recorded (i.e., in P20, the rendezvous program). For most of these rendezvous radar tests the angle and interface simulators were used; however, the tests were performed once with a flight-type radar,

The landing radar interfaces with the primary GN&C system were also checked. In the previously referenced System Test Group memoranda the total test program for the radars is described; it is also discussed in some detail in Chapter IV of this report. The following paragraphs outline a particular rendezvous radar test to illustrate the system-level approach to the compatibility evaluation,

Exercising the rendezvous navigation program, P20 (lunar module), and all options of operation was the test objective. Loss of input discretes were also introduced with the intent of simulating deviations from normal operation. Alarms associated with radar malfunctions or computer problems were also checked for the same purpose, After calling P20, the radar search option was tested and antenna motion was observed, When the antenna stopped at a dwell point of the search, the positions of its shaft and trunnion gimbals were recorded for comparison with required pointing given the state vector of the simulated target command module. During the search pattern, multiple computer restarts were also introduced to represent an off-nominal situation.

By advancing the computer-contained time, the required pointing angles to the target vehicle were changed through the change in its state vector. When the angular separation from the boresight increased beyond 30 degrees, the preferred attitude maneuver was called by the computer as required. Insertion of a "data good" discrete from the radar simulator then resulted in exiting the search routine (R24) and entering the data read routine (R22).

Through reinitialization of the computer with the proper state vectors, tests of the type above could be repeated and sequenced through all the required program branches for compatibility verification (e.g., the manual radar pointing option (R23), operation of the 3-degree sidelobe lock-on check, and so forth). All the computer displays and interactions with the crew could be verified also by repeating the tests and following each of the program branches, in turn, as listed in the guidance system operations plan. This included evaluation of radar tracking marks and the updates in target position and velocity resulting therefrom.

10.2.1.6 Miscellaneous Command and Lunar Module Tests

The moding of the inertia! measurement unit was checked; including turn-on tests, turn-on with failure, gimbal caging, error monitor, and so on. Extended verbs which are available to the crew for various hardware control functions were tested also (V40, V41, V42, V43, V44, and V55). Additionally, the inertial measurement unit compensation routines were tested for correct magnitude and polarity. The program for aligning the abort guidance system in the lunar module was checked. When command module programs having the VHF ranging capability became available, their compatibility with the laboratory hardware was verified and compatibility was tested of the ranging marks with optical tracking marks frorn sightings to the lunar module.

10.2.2 Compatibility Problems and Their Resolution

During the testing, all problems which seemed to indicate a program error, conflict with the operations plans, or compatibility problem were reported on System 'Test

Group anomaly reports. Fifty-four of these were generated on the lunar module system and 26 on the command module system. These were all eventually closed either by reporting them formally to NASA as program anomalies, by correcting the program, or in some cases the test procedure was found to be in error. A few interesting problems (some were not discovered in regular testing and so were not anomaly reports) are discussed below:

1. Shortly before the launch of Mission 202 it was discovered that the accelerometer scale factor compensation had the wrong sign. This was caused by the hardware test groups defining the scale factor as $\frac{cm/sec^2}{pulse}$ while the computer program expected $\frac{pulse}{cm/sec^2}$. The problem was easily corrected by changing the erasable load, but this points out the importance of detailed testing in all hardware/software interface areas.
2. As a result of spacecraft-level testing at North American Rockwell and subsequent testing at MIT/IL, it became apparent that the coupling data unit will, while operating normally, have transient errors of a few tenths of a degree. The errors exist for a short time. However, if a transient occurs at the instant that the computer is reading the coupling data unit for a mark, inaccurate data are stored. The probability of an error is about 1/7000. A partial fix was made to the command module program by incorporating Alarm 121 to indicate rapidly changing coupling data unit read counter values. A hardware fix was designed and tested by December 1967 (STG Memo No. 1094), but NASA would not authorize the change. These transients remain a potential problem.
3. After the inertial measurement unit performance test was programmed, it was discovered that the accelerometers could have an appreciable dead zone and that the data processing in the performance test was not compatible with such a dead zone. This problem was corrected by positioning the stable member at the start of the test so that the horizontal accelerometer would not be in a dead zone.
4. It was discovered that, in the spacecraft environment, noise could put extraneous bits into the uplink buffer at times when no uplink transmission was taking place. Subsequent uplink transmission would cause these bits to overflow and cause erroneous results into the computer. This was corrected by sending a string of all zeros at the beginning of a transmission, thereby setting all bits to zero.
5. The sextant shaft has a physical stop at 270 degrees rotation in either direction from zero. In early programs, auto-optics would always drive the shortest way from Star 1 to Star 2 and so it was possible for auto-optics to drive into the stops. This was fixed by correcting the auto-optics routine.

6. The restart logic would allow the inertial measurement unit to go inertial for one cycle of T4RUPT even if the stable member was in gimbal lock. This could result in considerable stable member motion and conceivable damage to the gyros. This was corrected by having the restart logic keep the stable member non-inertial if it was at the time of the restart.

7. V41N72 is used before Program 52 to place the radar antenna out of the field of view of the alignment optical telescope. However, the required position was out of the mode limits allowed by the rendezvous radar monitor routine and the radar could not be designated as desired. This was corrected by changing the limits for this special case.

8. It was possible to cause restarts by entering inertial measurement unit moding commands such as coupling data unit zero or coarse align before the previous moding was complete. This was corrected when interlocks were provided that prevented two extended verbs from being entered at once. Also some K-START tapes were not compatible with the hardware in that insufficient time was allowed for moding to be completed, This was corrected by requiring larger wait times on the tapes.

10.2.3 Milestones

The following tabulation shows some milestones in the testing programs.

Program	Level 3 Testing Complete	Level 5 Testing Complete
SUNDISK	Sept. 67	Nov. 67
SUNDANCE	April 68	July 68
COLOSSUS I	July 68	Nov. 68
COLOSSUS II	Feb. 69	April 69
LUMINARY 1	Nov. 68	May 69
LUMINARY 1A	April 69	June 69
LUMINARY 1B	August 63	Sept. 69

10.2.4 Conclusions

The System Test Group laboratory testing provided a valuable test of the mission programs. It was possible to design laboratorytests that were comprehensive enough to substantially increase confidence in the programs.

10.3 SYSTEM TEST PROGRAM VERIFICATION FOR MISSION PROGRAMS

The inertial measurement unit performance test (gyro drifts and accelerometer bias and scale factor) and the gyro scale factor test were verified. The programs were tested both with the digital simulator and on the laboratory systems. Particular attention was required when portions of the programs were transferred to erasable memory to save fixed memory locations. For the laboratory test, the programs were evaluated by comparing the results of the test with the program being tested to the results of tests with previous programs on the same instruments. When available, the K-START tapes to be used at the Kennedy Space Center were used in the laboratory to run these tests.

SECTION 11.0
GROUND SUPPORT EQUIPMENT DEVELOPMENT

11.1 BASIC GOALS

The basic goals of ground support equipment (GSE) development were:

1. to demonstrate adequately the ground performance characteristics of the airborne hardware while employing the same equipment and techniques at all test levels to the maximum extent practicable,
2. to ensure uniformity and consistency of test data at all levels of test,
3. to minimize duplication of design activity, and
4. to facilitate configuration control.

The major element of the ground support equipment is a test complex capable of verifying the entire GN&C system or any of its individual subsystems-optics, inertial, or computer. Ancillary equipment is, for the most part, limited to that available from general laboratory and field support sources. Exceptions include the ground support equipment used to test the optical unit assembly, the signal conditioner, the power and servo assembly adapter module (PSAAM), and the alignment optical telescope.

NASA drawing 1900030 lists al! ground support equipments, their configurations, and usages. A detailed description of the equipment is contained in ND 1021039, Auxiliary Ground Support Equipment Manual, and in ND 1021040, APOLLO Bench Maintenance Ground Support Equipment.

MIT/IL generated the overall test concepts, requirements, configurations, and the test and equipment accuracies. MIT/IL also approved the equipment mechanizations, provided a functional review of the detail design, and maintained configuration control. The ground support equipment was fabricated by the same industrial contractors responsible for production of the comparable airborne hardware. MIT/IL did not have detail design responsibility for the equipment but was responsible for technical supervision of the industrial contractors. In addition, MIT/IL served whennecessary as technical consultants to the industrial contractors.

The major problems in development of the ground support equipment stemmed from the difficulty of providing both an adequate simulation of interface and/or environment and an adequate measure of individual component performance.

Other difficulties in achieving the basic goals resulted from inadequate consideration of ground support equipment design and tolerances in the airborne equipment design and in certain cases use of the ground support equipment to test for quality and workmanship.

11.2 EQUIPMENT EVOLUTION

The original design concept for the APOLLO ground support equipment indicated the possibility of four individual checkout units:

1. Inertial Subsystem Checkout Equipment (ISSCE)
2. Optical Subsystem Checkout Equipment (OSSCE)
3. Computer Subsystem Checkout Equipment (CSSCE)
4. System Checkout Equipment (SCE)

Further study showed this preliminary structure produced a great deal of test equipment redundancy. It became evident that, with the addition to the inertial subsystem checkout equipment of a few special controls and some special accessory gear, three of the four test requirements (inertial, optical, and system checkout equipments) could be combined into a general purpose universal console, leaving the computer subsystem checkout equipment as a separate entity. This console evolved into the present optical inertial test set (OITS). The test set was the major element of the general test complex.

11.3 OPTICAL INERTIAL TEST SET DESIGN EVOLUTION

The test set meets the following requirements;

1. Provides input power to the subsystems and their associated electronics.
2. Monitors all output signals from the subsystems.
3. Simulates certain computer-oriented input stimuli (for subsystem testing) to the subsystems.
4. Provides monitoring and control of the thermal environment of the subsystems.
5. Provides manual features to simulate programed computer outputs and/or astronauts' manual selections.

6. Supplies the necessary handling and holding fixtures for mechanically mounting and aligning the subsystems for test.
7. Provides monitoring and measurement facilities for power supplies (magnitude, frequency, noise level, and phase verification).
8. Exercises the systems to determine dynamic response characteristics,
9. Verifies all displays and controls.
10. Provides for malfunction isolation to a module level where possible.

Since the airborne prototy-pe hardware was to be manufactured at MIT/IL, the ground support prototype design was divided into two separate tasks: the design and construction of prototype consoles to support subsystem testing at MIT/IL and the design and development of equipment to eventually evolve into a production Configuration. The latter was performed by the industrial support contractors on the basis of airborne system design requirements and results of the MIT/IL ground support equipment design effort that incorporated an actual marriage to a system.

The first breadboard unit for inertial subsystem testing was constructed by MIT/IL in early 1963. This unit was used to evaluate prototype inertial subsystems AGE 4 and 5 at MIT/IL. This unit's flexibility was shown in its support of Block I Series 0 and 100 systems and, with some modifications, the initial Block II lunar module prototype systems.

A second breadboard unit was fabricated by AC Electronics and delivered to MIT/IL in late 1963. This unit was configured as a "pre-production" unit reflecting the final design of the universal checkout console and was used to support testing of the Block I Series 0 and 100 prototype designs in their system configuration.

The successful marriage of a partially configured GN&C system to a pre-production universal-type checkout station in early 1964 indicated the successful implementation of some of the early design objectives, Design deficiencies discovered during this marriage and checkout did much to pave the way for the construction of a second pre-production unit to support airborne equipment "Learner Model" testing at the factory and to advance the design and fabrication effort of the production units.

In mid-1964, a third portion of the early phase checkout was completed-the installation and functional check of an optics subsystem. The control and monitoring capabilities of the ground support equipment were demon&rated. Few of the problems encountered were ground support equipment oriented.

During mid-1964, the design and fabrication efforts of the groups concerned with ground support equipment underwent major changes. Most important were numerous changes to support the Series 100 GN&C system modifications. These changes were scheduled for new production units with retrofit kits for updating existing stations. The next major effort was equipment design to support the Block II GN&C system. The major constraint for the new design was a requirement that the equipment be capable of supporting testing for all versions of GN&C systems. Universal compatibility was achieved by special interchangeable components for the basic consoles and special cabling and accessory hardware for each desired configuration.

While the Block II design was being implemented, a program was initiated to prove the compatibility of the new ground support equipment with all airborne configurations. This plan, "GSE First Article Test," was initiated to determine the testing capability of the optical inertial test set and accessory hardware. Test set design testing and documentation had to ensure compatibility between ground support equipment, the lunar module GN &C system (and its various subsystems), and the command module GN&C system (and its various subsystems). The ground support equipment had to meet all the requirements of its Procurement Specification. Any modifications, repairs, or replacements not resulting from engineering change proposals to the GN&C or ground support equipment, hut established during testing as necessary to attain system compatibility, were accomplished by marriage kits. Any ground support equipment which was accepted prior to completion of first article testing was considered a partial acceptance and its final acceptance was contingent upon completion of all marriage kits.

The marriage of the new optical inertial test set to a GN&C system in late 1965 resulted in the generation of the first of a series of marriage kits. This kit, P/D; 8102159, corrected certain ground support equipment design deficiencies, fabrication errors, some airborne equipment design, and test procedures (particularly dealing with sextant reticle power measurement, gyro pulse-torque polarity changes, and GN&C system interface revisions). In addition, this kit implemented changes to support the lunar optical rendezvous system (subsequently canceled).

Continued ground support equipment production acceptance testing, airborne equipment testing, revisions, and interface changes indicated the need for additional design modifications. A second marriage kit, P/N 8102157, was issued in early 1966. This kit made changes to satisfy added test requirements and minor product improvement additions. From April through October 1966, special tests were conducted at the factory, using each version of the GN&C system to verify the testing and support capability of the universal test station. These tests were run using the accepted airborne equipment job description cards as a test base. Included in the tests were the following:

Inertial Subsystem Guidance Computer Moding Interface Tests	15073
Accelerometer Fail Detector Test	15070
Coupling Data Unit Fail Detector 'Test	15097
Zero Encoder Test	15107
Coarse Align Test	15108
Coupling Data Unit Manual Test	15199
Fine Align Test	15111
Coupling Data Unit Dynamic Repeating Test	00091
Coupling Data Unit Dynamic Positioning Test	00094
Attitude Control Test	15112
Optics Hand Controller Operations Test	10711
Trunnion Servo Loop Test	10714
Shaft Servo Loop Test	10715
D/A Converter Sensitivity Test	10716
Scanning Telescope Trunnion Accuracy Test	10710

The systems used during these tests were the following:

	Unit No.	Date
Block I GN&C	123	16 April 1966
Lunar Module Inertial Subsystem	603	14 June 1966
Lunar Module GN&C	603	14 - 16 June 1966
Block II Inertial Subsystem	204	25 July 1966
Block 11 Optical Subsystem	204	25 July 1966
Block I Optical Subsystem	ACSP II	25 August 1966
Block I Inertial Subsystem	Inertial Measurement Unit 7	21 October 1966

The successful completion of unit testing-implying a satisfactory First Article Test -concluded the compatibility test program. Program completion was recorded by AC Electronics with concurrence by MIT/IL and NASA representatives. Similar first article test programs were accomplished for the special optical and computer ground support equipment.

11.4 COMPUTER SUBSYSTEM CHECKOUT EQUIPMENT DESIGN REQUIREMENTS

The computer subsystem checkout equipment was designed to support the operation of the APOLLO guidance computer. The primary requirements were to:

1. Provide a means of calibrating the master computer clock.
2. Provide programing control for computer operation.

3. Provide the logic, hardware, and commercial equipment required for loading programs or data into the computer.
4. Provide monitoring and logging equipment for sampling computer activity.
5. Provide means for mounting and establishing environmental control of the computer during subsystem testing.

The ground support equiprnent for the computer subsystem, designed and manufactured by Raytheon Company under the technical supervision of MIT/IL, consists of the following:

1. A calibration console assembly, for the calibration of the computer oscillator frequency, aging characteristics, and temperature. (The console also provides a 1-MHz time base for use with other calibrating systems,)
2. A guidance computer/computer test set operation console, which must
   (a) provide mounting facilities for the guidance computer,
   (b) provide environmental control,
   (c) provide mounting for the main display and keyboard and the navigation display and keyboard, and
   (d) act as an interface coupler between the computer and the computer test set,
3. A computer test set that
   (a) supplies program-loading capabilities,
   (b) monitors key computer program functions,
   (c) records and displays data,
   (d) provides dummy loading, and
   (e) has self-testing capabilities.

Additional special equipment to facilitate computer and program testing consisted of:

1. DSKY recorder, to help determine operator procedural errors.
2. Camera, to record switch settings and to help determine operator error in event of a disturbance in testing.
3. Core rope simulator (monitor), to monitor computer functions (contents of various registers) and to provide erasable memory simulation of the core ropes for program testing purposes (there are also devices with similar purposes called PACS).
4. Trace (Coroner), a digital recorder of the time history of selected computer register contents used to determine previous computer behavioral activity in the event of a disturbance,

Most computer ground support equipment problems were due to timing (usually capacitive delays) and to noise spikes. Some resulted from not taking as many or more precautions in support equipment packaging as with the airborne components.

11.5 OPTICAL TEST EQUIPMENT

The optical test equipment does not come in a single package and must be discussed individually. (In many cases, Block I part numbers are used even when a Block II part also exists.)

The optical equipments provide the following:

1. A means of establishing a known true azimuth line of sight,
2. A means of measuring pointing and readout accuracy,
3. A means of measuring optical resolution,
4. A means of verifying the operation of gear trains, indicators, etc, and
5. A means of simulating known stellar and luminescent bodies.

The Zero Degree Autocollimator Plate Assembly (10173808, Figure 11-1) and the Forty-Five Degree Autocollimator Plate Assembly (1017381, Figure 11-2) were used in the system test laboratories to provide optical targets at the respective optics trunnion angles. The zero-degree assembly contained one 5-inch autocollimator and one 2-1/2-inch autocollimator. The 5-inch autocollimator was used for sextant star line of sight (SLOS) to landmark line of sight (LLOS) parallelism checks. It was also used as the base reference for the optics trunnion accuracy sighting tests, in which the optics viewed the 0- and 45-degree targets simultaneously, and the optics angular readout was compared with known angular separation of the two targets. The 2-1/2-inch autocollimator in the 0-degree plate was used to check the scanning telescope alignment. The 45-degree autocollimators plate assembly contained two 2-1/2-inch autocollimators-one for sextant sightings and one for scanning telescope sightings. The 2-1/2-inch aperture was sufficient for the sextant at 45 degrees since only the star line of sight needed to be viewed (landmark line of sight remained fixed at 0 degree).

The 5-inch aperture of the 5-inch autocollimator was a compromise choice. The aperture's limited size did not permit the sextant landmark line of sight and sextant star line of sight to be viewed at both the 0- and the 90-degree shaft angles without vignetting one or the other star line of sight position.

The only other problem with these units again occurred with the 5-inch autocollimator. The relatively short focal length of the collimator and the tight tolerances on

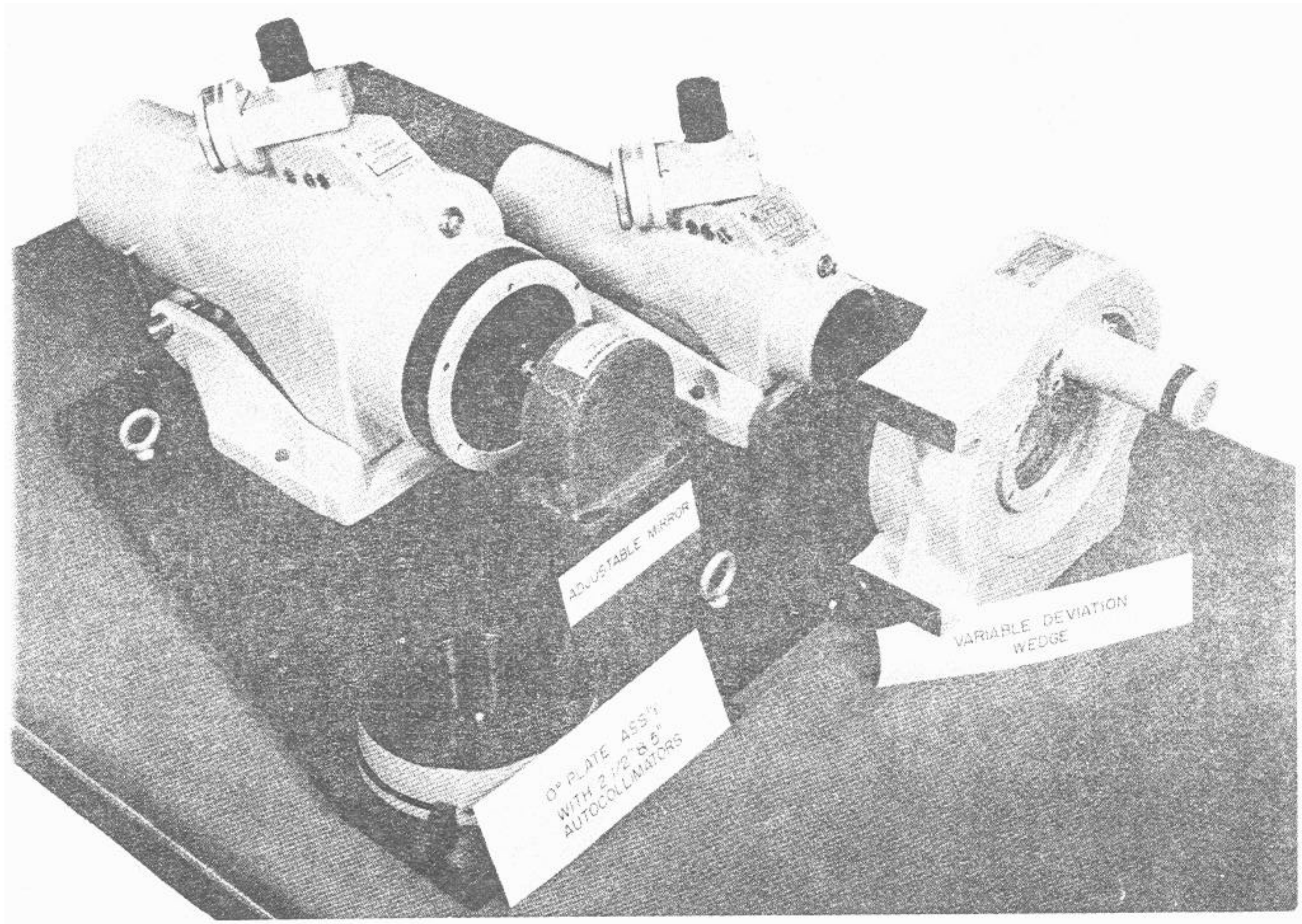


Fig. 11 1 Zero Plate Assembly

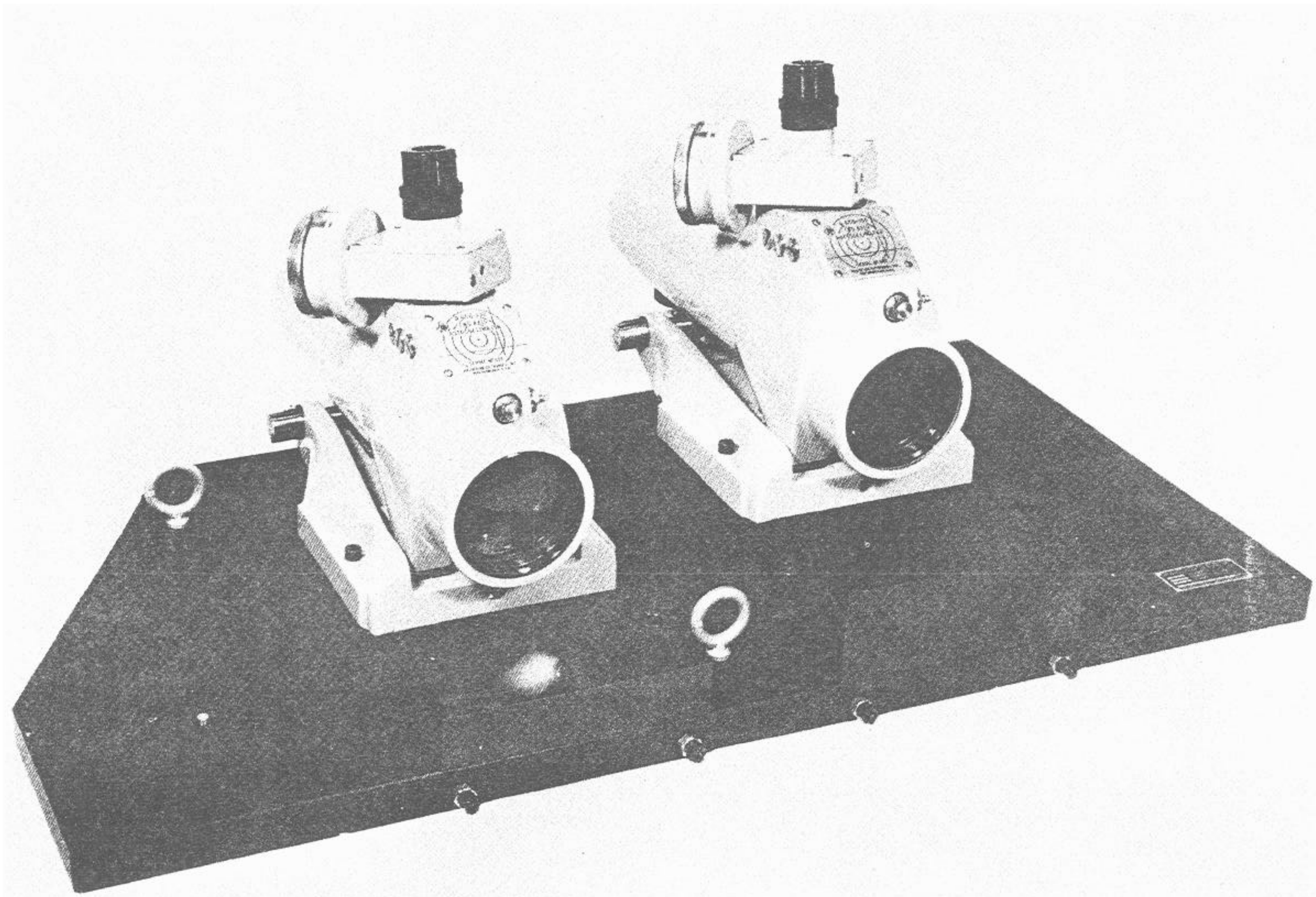


Fig. 11-2 Forty-Five Degree Autocollimator Plate Assembly

Fig. 11-3 Adjustable Mirror and Pedestal Assembly

collimation over the aperture required a very precise focus. There were problems initially in the focusing procedure, but these were corrected and are reflected in the present methods.

Other problems with the collimators involved the use of resolution targets. The initial resolution targets for the scanning telescope were in arcseconds rather than arcminutes. Charts with resolution patterns had to be made up for both system and subsystem levels. Additional charts were required later for post-installation testing.

The Adjustable Mirror and Pedestal Assembly (1019759, Figure 11-3) is used to establish a known azimuth reference for setup and alignment of the 0- and 45-degree autocollimator plate assemblies for optical subsystem and system tests.

The Vertical Leveling Mirror (1017445, Figure 11-4) is used to obtain a true earth-vertical reference in conjunction with the alignment of

1. Alignment Mirror Assembly (1016951, Figure 11-5),
2. Azimuth Reference Fixture (1017382, Figure 11-6), and the
3. GN&C Installation Qualification Fixture (1017383, Figure 11-7).

The mirror consists of two reflecting surfaces supported by a pendulous support system. The pendulous movement positions the mirror surfaces to the earth vertical once the mirror base is leveled. A locking device, incorporated in the mirror support system, is used to prevent damage due to sudden shock or jarring of the mirror.

The theodolite (1017444, Figure 11-8) is a Kern DKM-3X used as a general purpose laboratory tool, together with its own tripod (10016531, to align the adjustable mirror assembly and the 0- and 45-degree autocollimator assemblies. It is also used to certify the GN&C Installation Qualification and Azimuth Reference Fixtures and to align these fixtures in azimuth for post-installation testing. The theodolite is also apart of the alignment optical telescope tester and is used on the tooling bar (1019758, Figure 11-9) at AC Electronics.

The Variable Deviation Wedge (1017376, Figure 11-10) is basically two counter-rotating optical wedges and is used in conjunction with the 2-1/2-inch autocollimator of the 45-degree autocollimator plate assembly to deviate the sextant line of sight in precise angular increments to test azimuth accuracy during GN&C testing. The wedge is basically a piece of laboratory support equipment. The only problem with

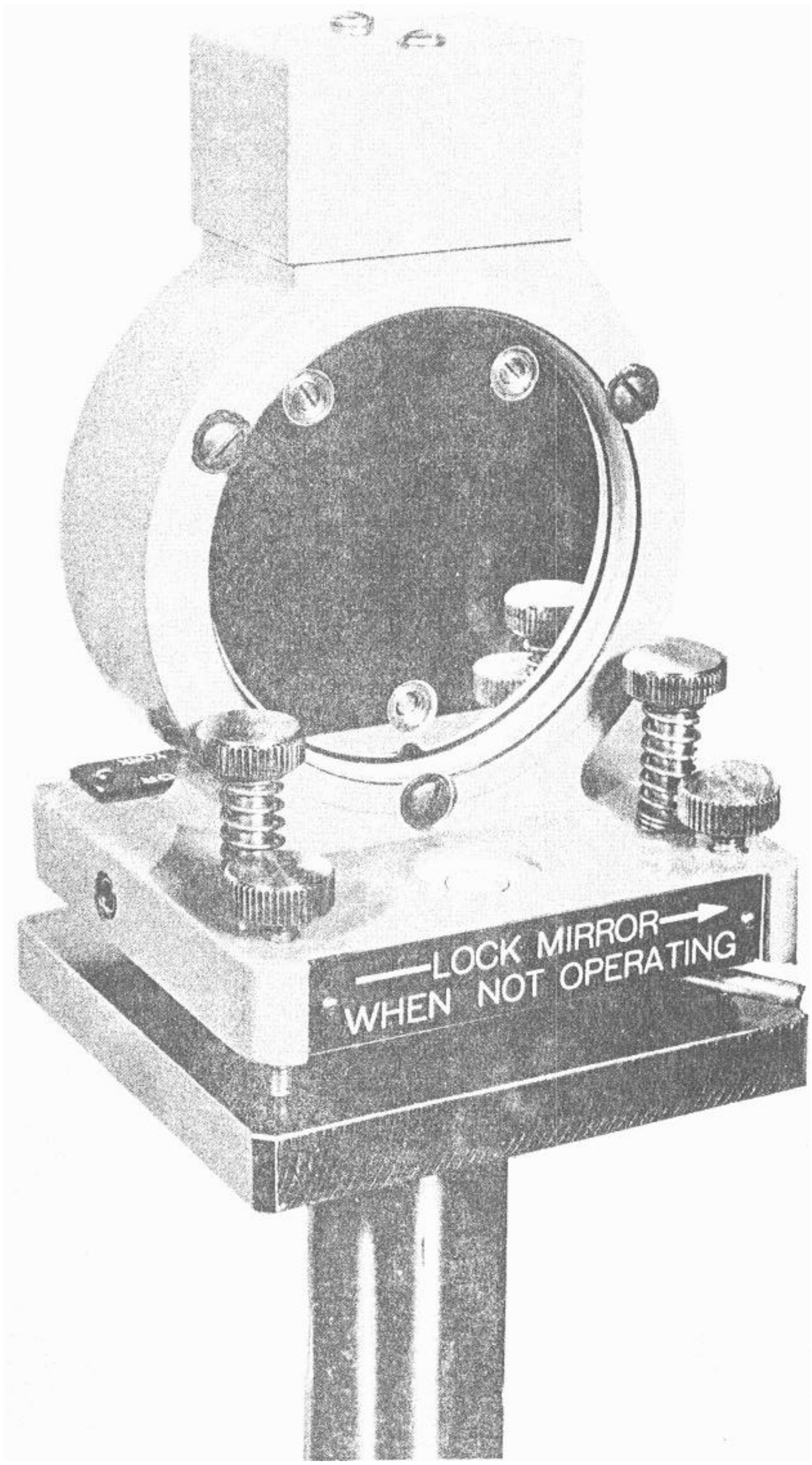


Fig. 11-4 Vertical Leveling Mirror

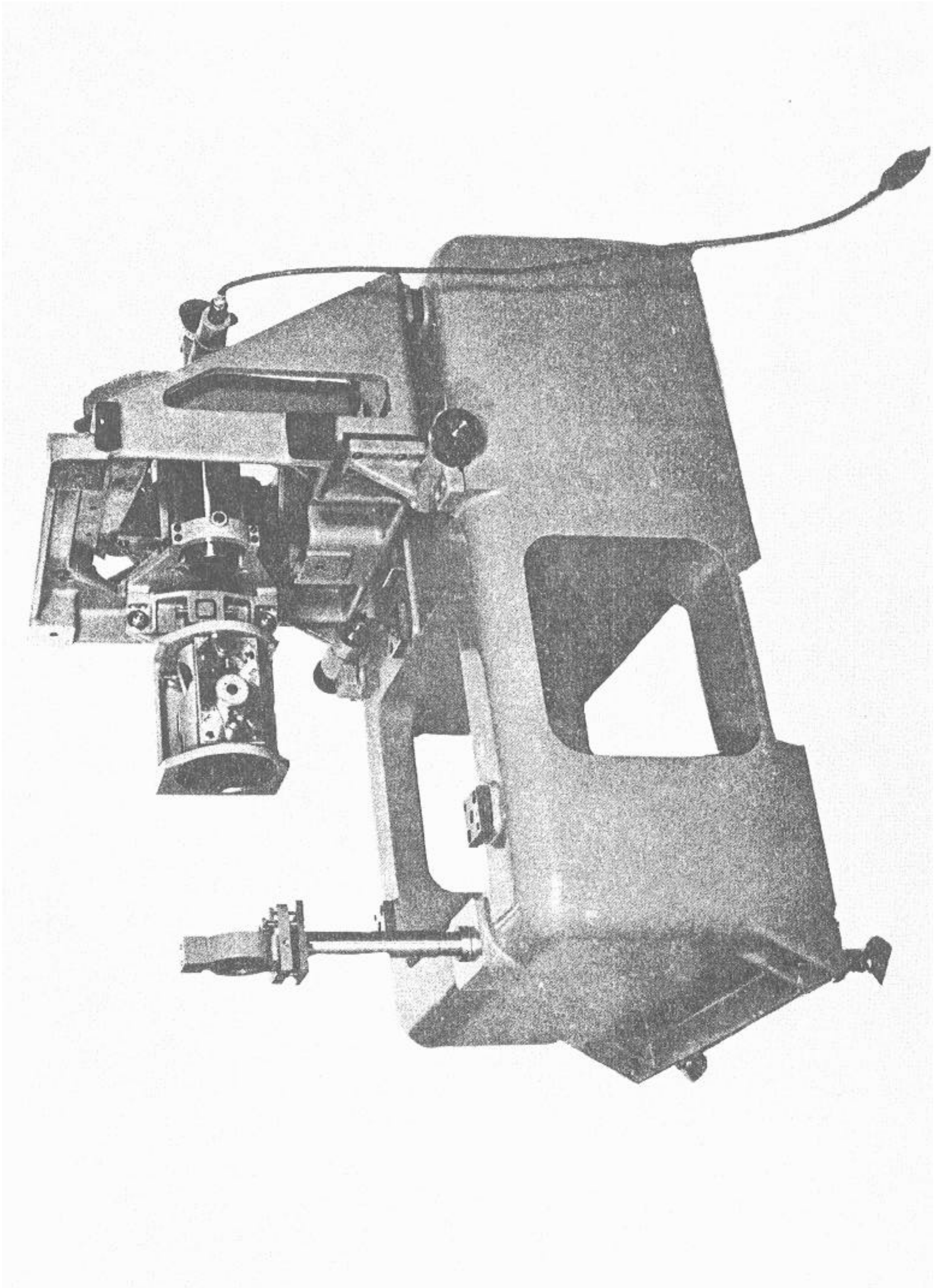


Fig. 11-5 Alignment Mirror Assembly

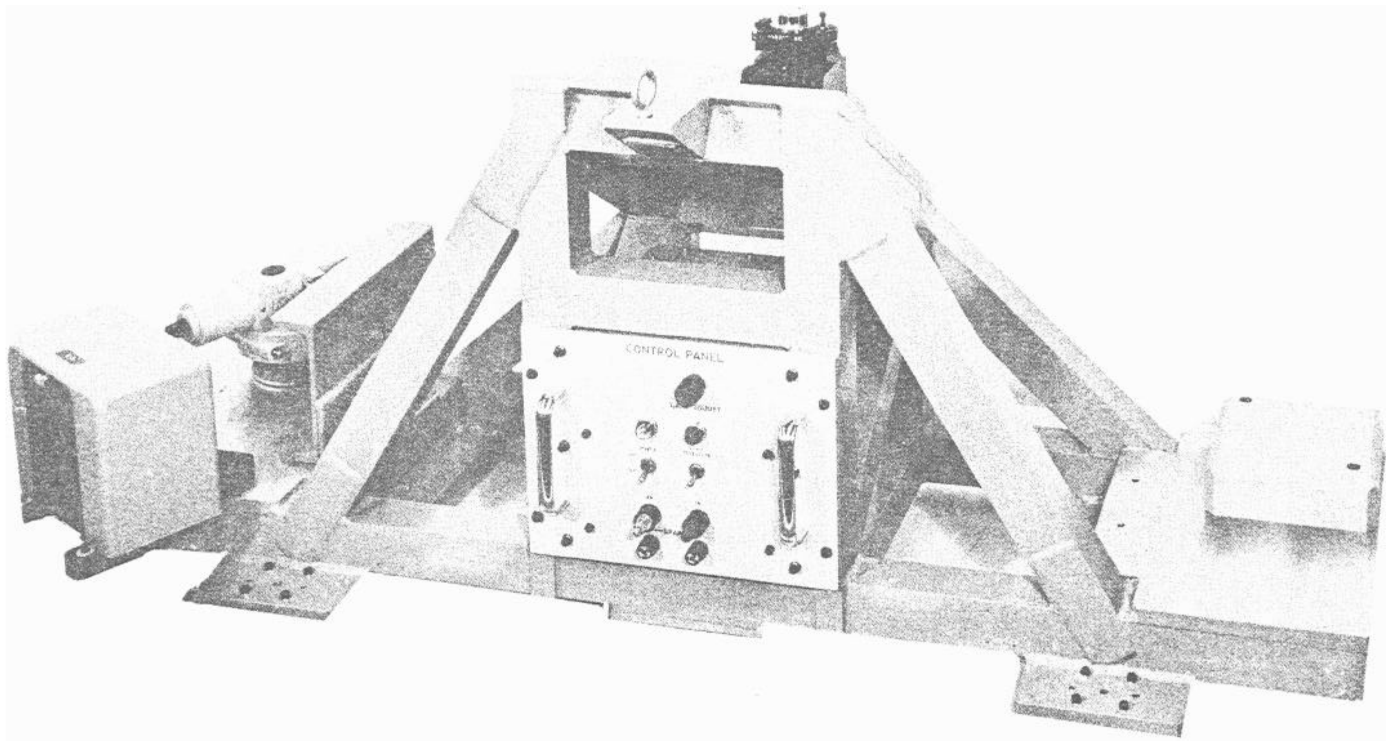


Fig. 11-6 Azimuth Reference Fixture

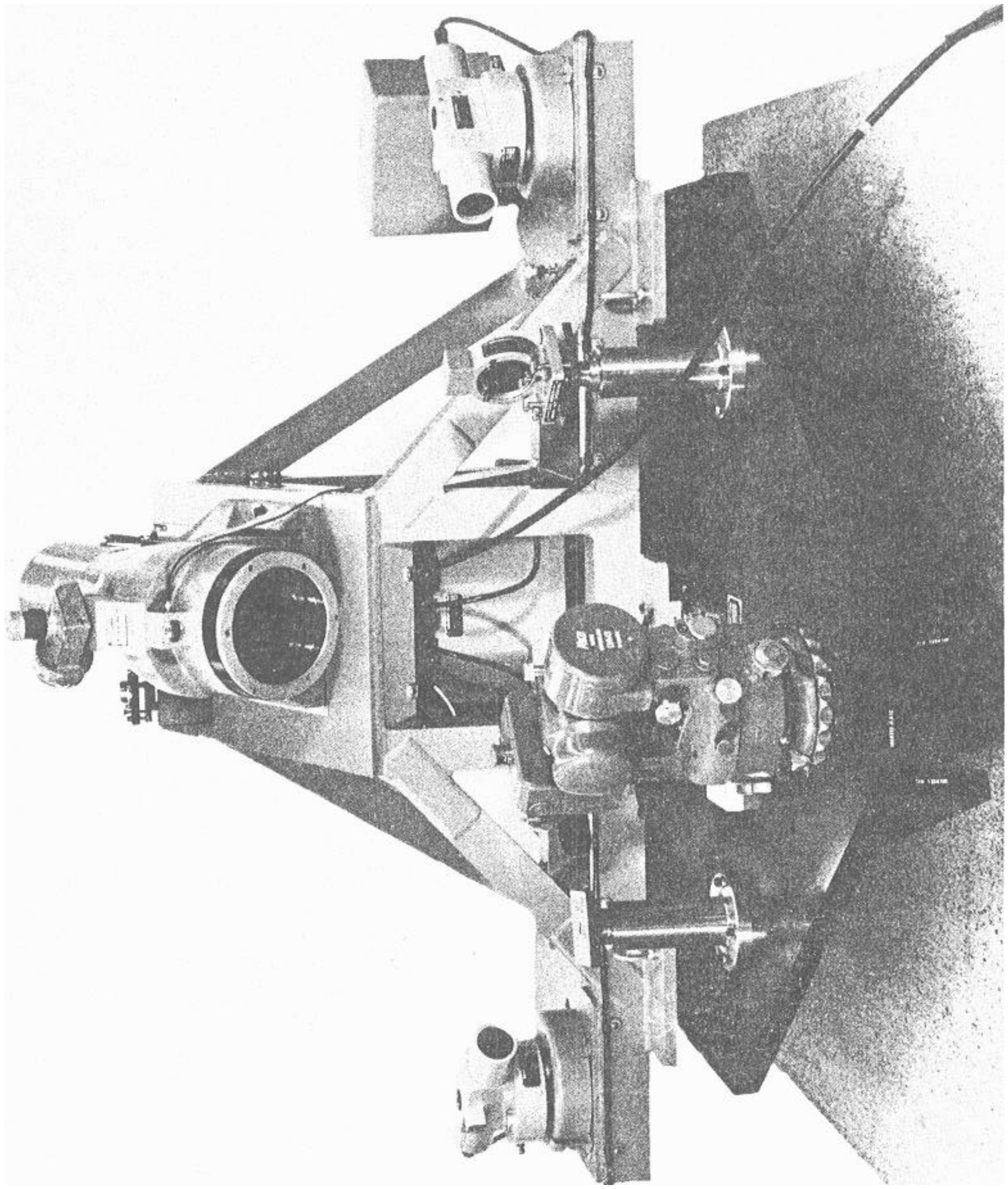


Fig. 11 Z GN&C Installation

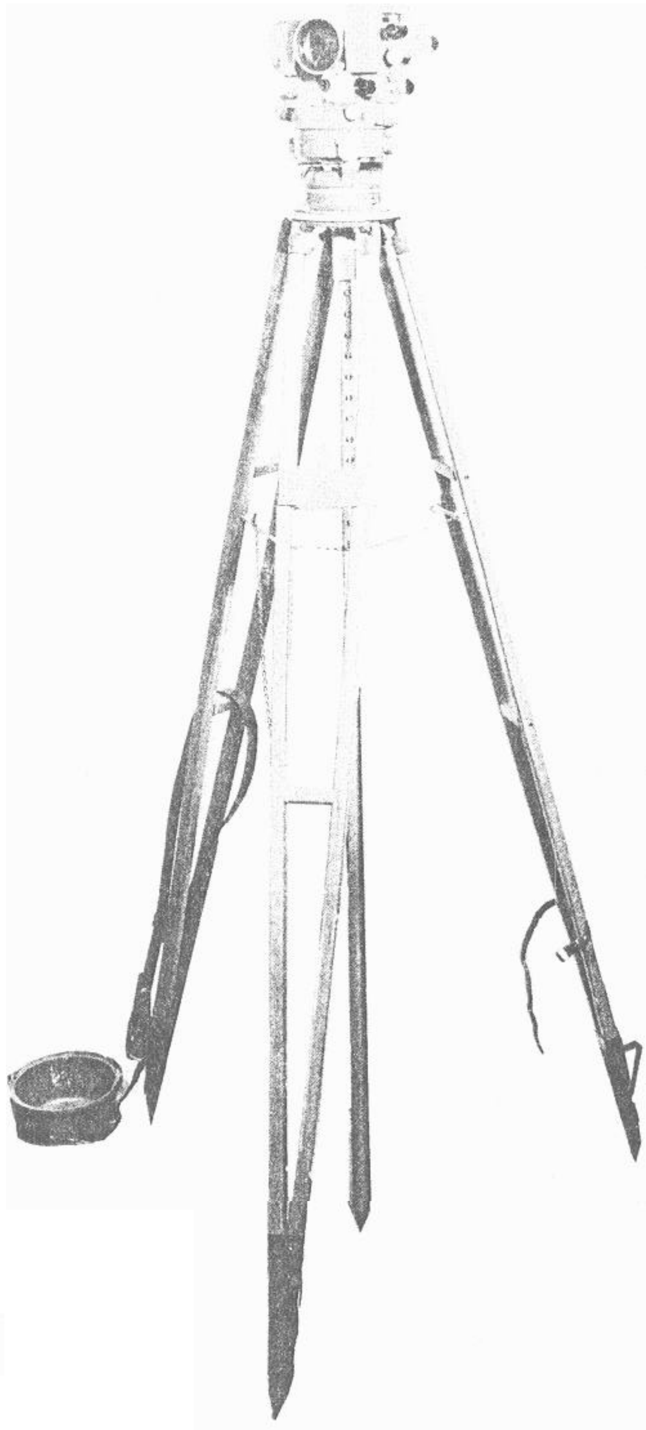


Fig. 11-8 Theodolite

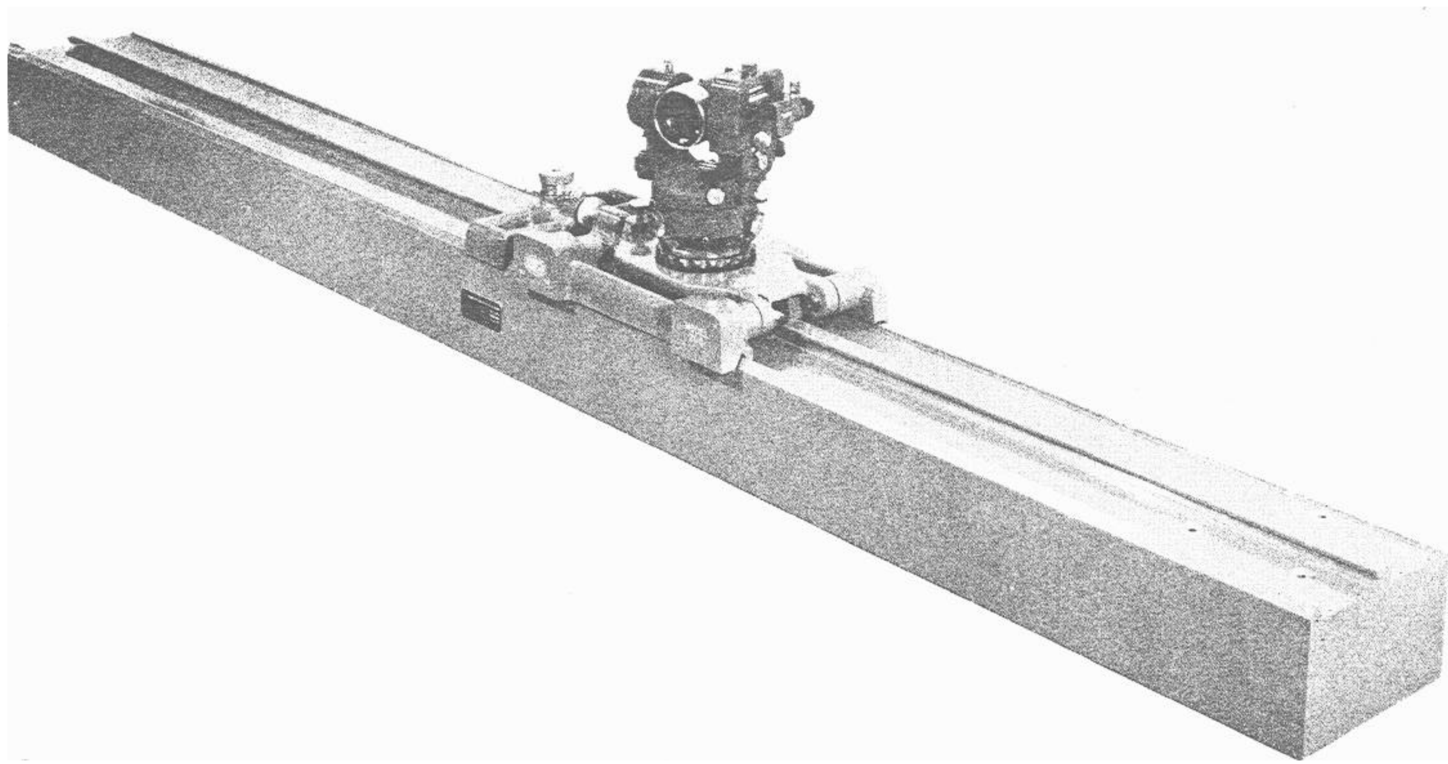


Fig. 11-9 Alignment Optical Telescope Tester

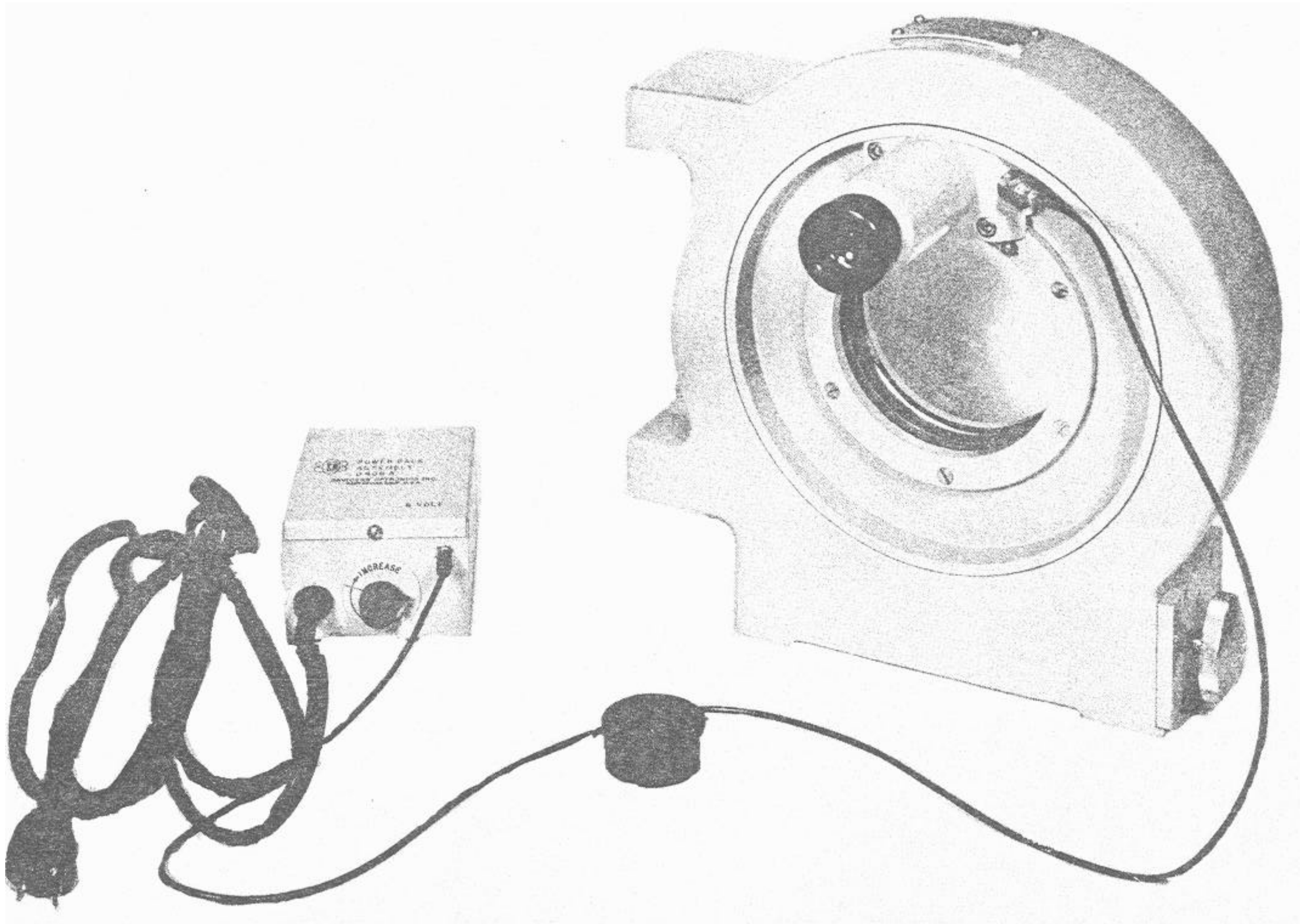


Fig. 11-10 Deviation Wedge

it is that some of the manufactured units were assembled wrong and had deviations counter to the normal deviation; hence, all units required a polarity identification plate,

The Portable Light Assembly (1019837, Figure 11-11) is used to backlight the optical unit sextant reticle for autocollimation purposes, (The sextant prism cover plate must be removed.) The portable light assembly was mainly used when the optics unit assembly was on the precision test fixture, Problems with the lamp included insufficient intensity adjustments, inadequate diffusion, and poor color characteristics.

The Retroreflecting Prism (1019840, Figure 11-12) is used to transfer the scanning telescope line of sight into the sextant star line of sight during system and subsystem testing. The prism is positioned in front of the optical unit in order to compare the telescope reticle with the sextant reticle through the sextant star line of sight. This comparison indicates relative shaft position accuracies of the two units through 360 degrees of rotation. The prism is mounted in an aluminum holding fixture. The holding fixture has hand grips for hand holding and can be supported by a holding stand. The retroreflecting prism is insulated with rigid polyurethane foam to limit degradation of resolution or alignment through handling during testing due to thermal and mechanical stresses. The entrant surface of the prism has nonreflective coating. While there were no problems with the retroreflecting prism, its predecessor, the short periscope, had thermal problems,

The GN&C Installation Qualification Fixture (101’7383, Figure 11-7) is a structural frame on which are mounted two automatic optical levels (autosets) separated by 90 degrees, a porro prism referenced in azimuth to the autosets, and a 5-inch autocollimator. The two automatic optical levels are mounted on a common horizontal plane at an angle in azimuth of approximately SO degrees from each other. Optical Reference No. 1 is located approximately 45 degrees in the horizontal plane from the command module %-axis. Both levels provide horizontal targets viewed through the optical subsystem sextant (star line of sight), The levels are pre-focused to infinity. Level stability about horizontal and azimuth axes are maintained within specified requirements. The Optical Reference No. 1 azimuth angular position is measured with referencsto a theodolite on the ground, Position stability is monitored by autocollimating the reference theodoiite off the fixture-mounted porro prism. Optical Reference No. 3 is a 5-inch autocollimator capable of viewing from the sextant landmark and star line of sight simultaneously, thus determining parallelism between the two lines of sight. The autocollimator is located at a nominal position of 32 degrees 31' 23” from horizontal, so that the two sextant lines of sight may be

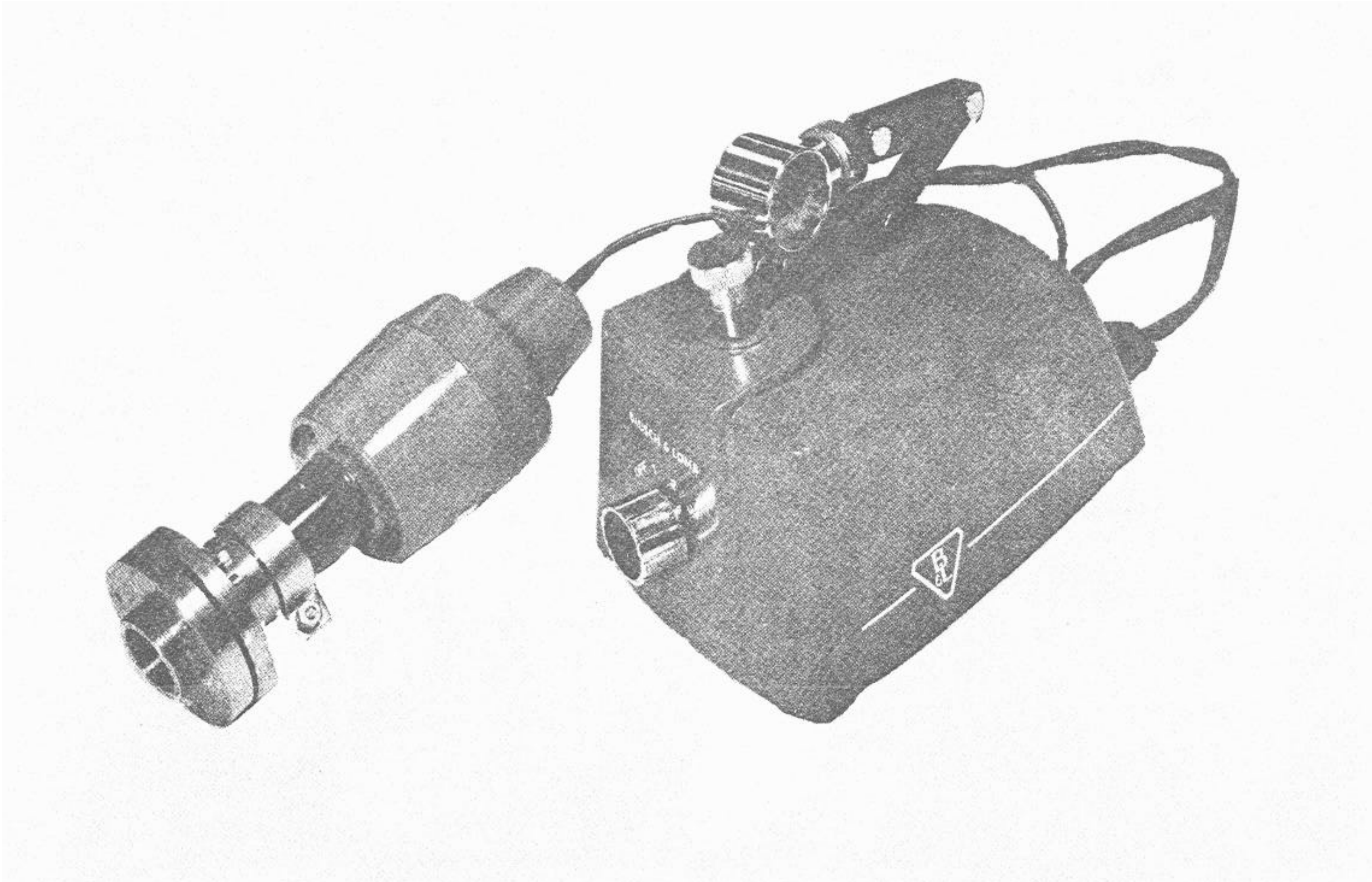


Fig. 11-11 Light Assembly

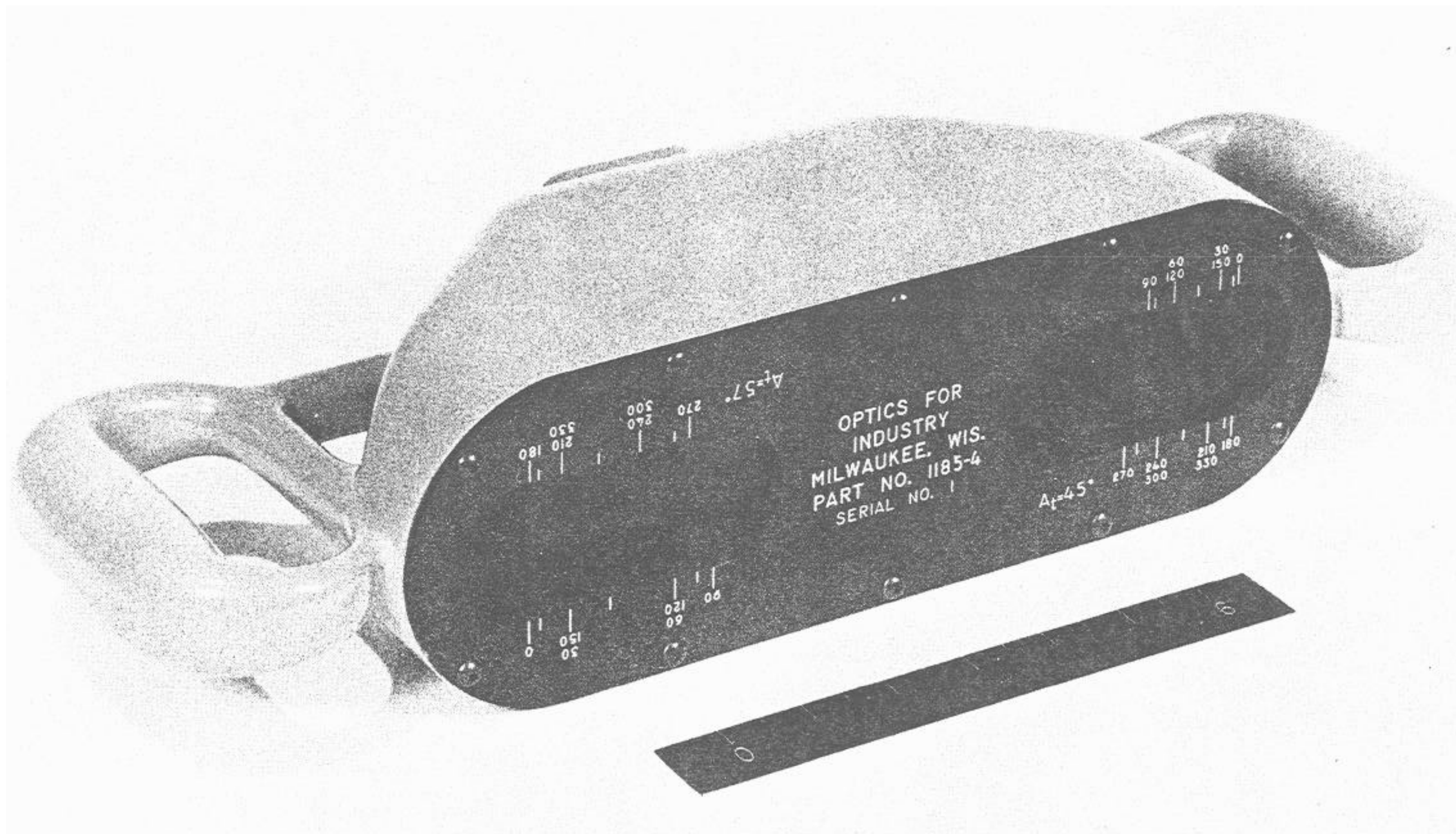


Fig. 11-12 Retroreflecting Prism

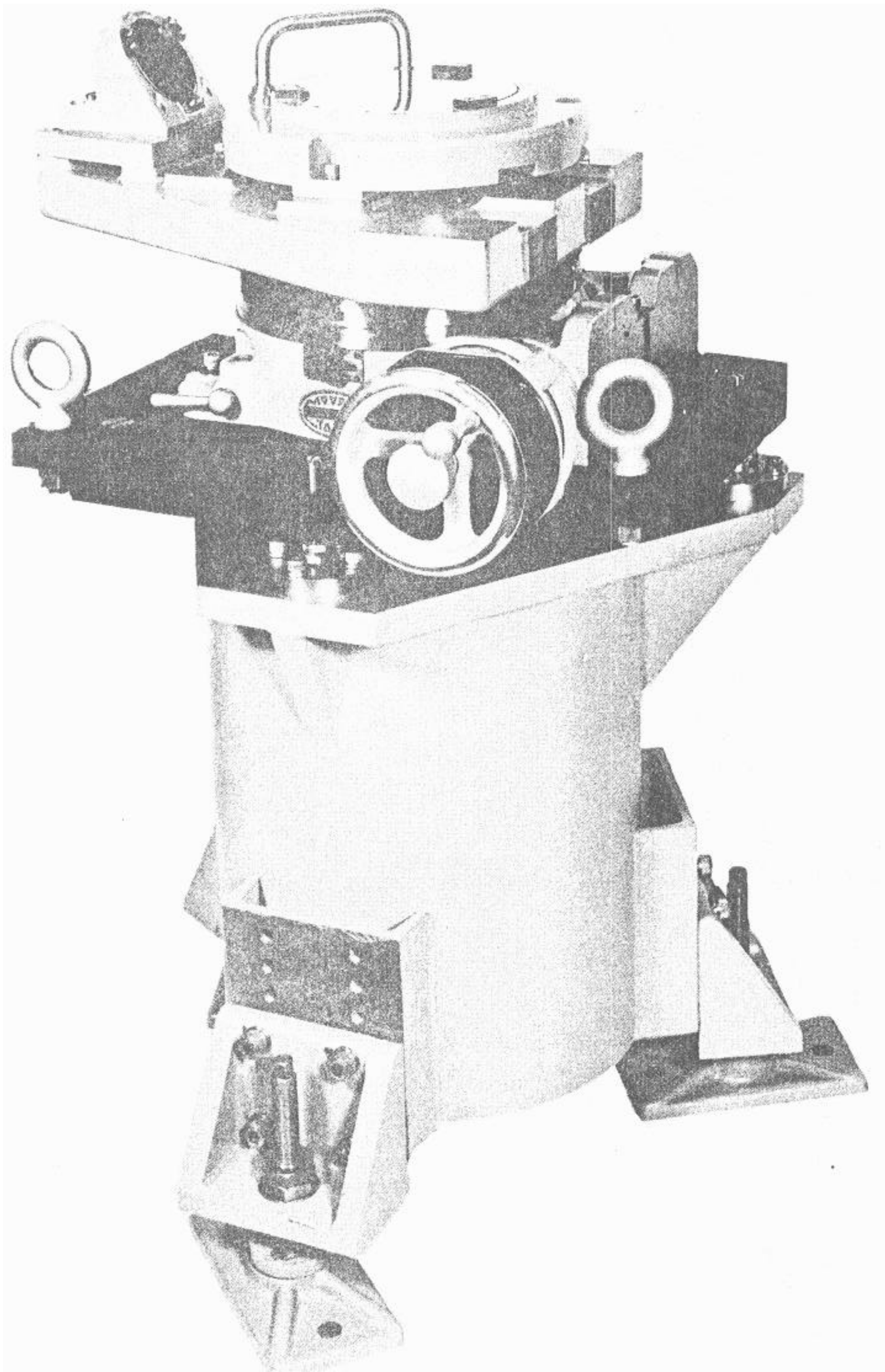


Fig. 11-13 Shaft Accuracy Tester

viewed at the zero optics position. The 5-inch autocollimator is also used for determining sextant optics resolution. The porro prism is positioned to provide constant azimuth monitoring by the ground-located theodolite.

The Shaft Accuracy Tester (1019769, Figure 11-13) is used to certify accuracy of the sextant shaft axis alignment during optical subsystem testing. The tester consists of a rotary table on which are mounted a flat horizontal mirror and a mirror which is off-axis of the rotary table and 57 degrees away from the horizontal. The optical unit assembly landmark line of sight is autocollimated off the leveled flat mirror, then the sextant star line of sight is autocollimated off the 57-degree mirror at various shaft angle settings, and the rotary table angle setting is checked against the shaft angle settings.

The Alignment Mirror Assembly (1017957) is used to determine angular deviation in three axes between the navigation base mounting pads to assure proper orientation between the coordinate systems of the optical unit sextant and the inertial measurement unit pads and pins of the navigation base. The tests were performed with the assembly seated on the inertial unit mounting pads of the navigation base, so that the optical unit, the base, and the alignment mirror assembly in effect became one unit. During optical subsystem tests, this unit was mounted on the Optics Navigation Base Mounting Fixture (1900013) which, in turn, was mounted on the Ultra Precision Rotary Table (1900006).

Since proper orientation of the optical unit and inertial measurement unit mounting surfaces will produce parallel lines of sight in the sextant and in the alignment mirror assembly autocollimator, deviation from the required parallelism can easily be determined. By placing self-leveling reflective surfaces (mercury pools) below the two optical instruments, sighting through the sextant and the autocollimator should produce, in each case, autocollimation off the reflective surfaces, if proper system coordination has been maintained. In case of autocollimation errors, the amount of error can be measured to determine angular deviation present in the mounting system.

Azimuth deviation measurement between the optical unit and inertial unit mounting pads was obtained after placing the alignment mirror assembly pellicle to the 28.5-degree position, and adjusting the sextant shaft axis to 180 degrees and trunnion axis to 57 degrees. While sighting through the alignment mirror assembly autocollimator, the sextant trunnion angle was fine-adjusted so that the reticle image was superimposed on the horizontal centerline of the autocollimator reticle. The azimuth error (deviationof sextant reticle image from superimposure on the autocollimator reticle vertical centerline) represented azimuth deviation between the optical unit and inertial measurement unit mounting pins of the navigation base.

Numerous problems occurred with this unit and with its certification fixture. The mercury pool used as a gravity horizontal level had poor reflective qualities, was subject to various vibration frequencies, and was corrosive. Although several investigations were made into mercury pool shapes and varying thicknesses of mercury and silicone oil layers, none proved really adequate. Other techniques involved level sensors in one form or another, but none could yield the measurement precision required from one position to another (in a reasonable test environment). A question of device stability arose, since several calibrations were done with the device in one position, and thereafter it was used in a position 90 degrees (gravity sensitive) away from the calibration position. The differences in line-of-sight angles due to the gravity vector were greater than the angle measurements required. Cleaning the plastic membrane in the pellicle assembly was difficult, because the membrane punctured easily. A full assessment of the alignment mirror assembly, in the light of these problems, concluded that the mechanical inspection tolerances and the inherent stability of the navigation base material obviated the need for this unit.

The Azimuth Reference Fixture (1017382, Figure 11-6) was essentially the same as the GN&C installation qualification fixture, but it did not contain a 5-inch autocollimator. It was used at post-installation to verify the azimuth drift of the inertial measurement unit relative to a known geodetic reference.

When it was planned that the optical unit assembly (Block I-100) would contain a star tracker and horizon photometer, these units were modified to hold a star and horizon simulator for post-installation testing of the tracker and photometer. When the tracker was canceled, these units were obsoleted.

The Functional Tester (1016949) simulates the electronics and components of the optical subsystem and is used in testing the optical unit's operational accuracy. The functional tester positions the optical unit and measures the sextant and scanning telescope resolver readouts. It is used together with the optical alignment and readout capabilities of the precision test fixture (PTF) to verify the optical-to-electrical alignment and readout accuracy. In addition,. the functional tester is used to verify the functional dynamics of the optical unit assembly portion of the optical subsystem servo loop. The functional tester was modified in Block I Series 100 development to provide additional electronics for testing the star tracker and horizon photometer. This modification became unnecessary when the star tracker and horizon photometer were deleted from the system. At the time of the Block I Series 100 First Article Test, it was determined that the motor drive amplifiers (MDA) in the functional tester were not good enough to adequately position the optical unit servo loops. These amplifiers were simiiar to those in the Block I system, and did not have sufficient quadrature rejection to ensure that the optical-electrical alignment measurements were accurate. (The majority of measurement variations were due to the test electronics rather than the item under test.) The problem was resolved by tuning the amplifiers and placing stricter limits on the amplifier gain. Another problem was posed by the narrow bandpass amplifiers used in conjunction with the star tracker and horizon photometer. These were exact copies of the airborne design (in accordance with simulation principle), but failed to yield an adequate signal-to-noise ratio to adequately test the optical unit tracker alignment. Both cases illustrate the necessity for ground support equipment design to be superior to airborne equipment design in signal amplification variations, frequency response, phase response, and signal-to-noise characteristics, Conversely, precision resolver zeroing, while accurate enough, failed to match the impedance characteristic of individual GN&C systems and required that a resolver trim module be zeroed at a higher assembly level with its peculiar system components. Another minor problem with the functional tester required the removal (isolation) of critical measurement circuits from the system ground during measurements.

The Precision Test Fixture (1021070) is an optical alignment test fixture for the optical unit assembly which has proven to be extremely accurate and stable. Its only difficulties were with the simulated optics certification fixture (which proved unstable) and with the single-axis autocollimators (which were subject to shifts between their fixed and filar reticles and were difficult to calibrate).

The alignment optical telescope tester is the same in function, but not structure, as the precision test fixture. The alignment optical telescope tester comes in two versions: afield test version with an incorporated rotary table and a ground support equipment version mounted on the optical inertial test set rotary table. Two versions were necessary because the field test version would not fit into the Grumman laboratory. It was decided to utilize the existing rotary table for the base, but the first attempt was too heavy for the rotary table and the design was rejected. Even on the second attempt, it was not realized that the coordinate systems of the two versions were different until after production and initial test. This was one of the most serious unanticipated problems in ground support equipment design, since the deficiency was in a fixture and not merely in a circuit. The problem was resolved through the use of transformation equations.

Other ground support equipment include

1. Shipping and storage containers (problems included inability to measure shock conditions well, failure to purge, high altitude pressure on sealed units, exposure to severe atmospheric conditions),

2. Connector covers,
3. Optics covers (including eyepiece covers),
4. Cleaning and purging equipment, and
5. Test equipment to certify the ground support equipment.

11.6 GROUND SUPPORT EQUIPMENT DOCUMENTATION

Throughout the design and production phase of APOLLO ground support equipment, proper configuration control was maintained through an extensive system of documentation, designed to reflect the "as built" structure of the hardware.

11.6.1 Drawing Structure

The first formal drawings of the system are mechanizations depicting the functional characteristics. These are followed by assembly drawings for manufacturing direction. The assembly drawings and any necessary schematics are displayed down to the module level. A third required item is an interface wire listing. These items are assigned a number within the document number structure. The top- assembly drawing number is generally the same as the item's part number.

11.6.2 Number Structure

Each subassembly drawing is referenced to its next higher assembly. Since all top- assembly drawings contain the applicable subassembly part numbers, traceability is achieved by reference to next higher assembly numbers. If the top assembly is classified as an "End Item" (highest individual deliverable item), its part number is included in the 1900030 drawing which is the master ground support equipment file.

11.6.3 Drawing Release

The support contractors forwarded all relevant drawings to MIT/IL for review. Mechanization drawings were generated after design definitions had been established and were presented for Design Review Board review and amendments as required. These drawings became the controlling documents. Assembly drawings were presented after the release of the mechanization drawings but prior to start of production. Procurement specifications were for end-item acceptance testing and were presented prior to release of production units. Technical Data Release or Revision (TDRR) and Engineering Change Proposals (ECP) were used for equipment changes after unit release to the customer.

The Master Retrofit Kit list contained all changes that affected ground support equipment configuration and interchangeability (even changes which occurred after release). This list was established in late 1965. Included were engineering change proposal-authorized changes from Block I Series 0 up to the present time. There were three separate master listings:

1. Drawing 1021201 - Required Retrofits for AC Electronics Ground Support Equipment Hardware
2. Drawing 8104002 - Required Retrofits for Raytheon Ground Support Equipment Hardware
3. Drawing 8106001 - Required Retrofits for Kollsman Instrument Ground Support Equipment Harclware

11.7 GROUND SUPPORT EQUIPMENT REQUIRED FOR SPACECRAFT GN&C TESTS

Inertial measurement unit thermal support was supplied by the battery power pack and portable temperature controller (PTC) in various configurations and with various cable accessories depending on the application (installation, lighting, transportation etc). The Block II controller provided more power than the Block I and had several interlock features to prevent simultaneous powering of the inertial measurement unit by both the ground support and the airborne equipment. The Block I controllers were limited to general laboratory support, while those of the Block II were used for air transportation as well as in the iaboratory.

The power and servo assembly adapter module was designed to interface most of the GN&C system signal with the automatic checkout equipment. The adapter module converted signals to standard automatic checkout equipment signal levels, buffered the signals to provide the proper interface between the GN&C system and the automatic checkout equipment, and converted signals which were subject to noise or quadrature to more resistant forms using phase-sensitive demodulators, etc. Its main difficulty was calibration of the output/input scaling due to improper calculation of output impedances.

MIT/IL felt that the GN&C system signals could not be adequately handled by the automatic checkout equipment and requested the addition of the accelerometer hardline and display oscilloscopes. The oscilloscopes displayed the in-phase component of the accelerometer signal generator (PIPA SG) outputversus the ducosyn reference. These signals were routed over hardline from the spacecraft to the control rooms. Although the oscilloscopes were used several times, at no time was the accelerometer signal generator the sole determinant of a problem, and the display probably was unnecessary.

The K-Start module was the most valuable variation from the standard automatic checkout equipment concept. It contained a keyboard switching arrangement-identical to the system DSKY -for manual keying of data into the onboard computer. In addition, the K-Start module could automatically read perforated paper tape data into the computer providing for the performance of rapid and accurate testing. The K-Start module was used to load the computer's erasable memory with programs for pre-launch system testing and with special launch constants. No special training or conversion factors were required, and automatic checkout equipment entry into the computer was similar to conventional laboratory or astronaut keying of the DSKY.

The CM Optical Alignment Support Fixture (A14-135) was developed to support optical targets for end-to-end testing of the GN&C system. The command module optical alignment support fixture was attached to the top of the spacecraft and accommodated adjustments rotationally and along all spacecraft axes; it provided a stable mounting position for conventional GN&C system optical test equipment—5-inch autocollimator, autoset levels, and porro prism. Tests performed included sextant "Zero Optics" line-of-sight parallelism checks, trunnion accuracy tests, fine-align tests, and end-to-end gyrocompassing tests.

The lunar module alignment optical telescope was initially tested using two targets mounted externally (on the platform) to the spacecraft. This proved troublesome due to shifts between the spacecraft and the platform (especially at Kollsman), and indirectly demonstrated the necessity of the optical alignment support fixture for the more critical command module measurements. Another problem was the time required for target alignment and for measurement performance with the two targets. A one-target procedure was developed, with the error in line of sight identified by that measured target viewed through two alignment optical telescope detent positions. This procedure greatly reduced the setup and test time, while greatly increasing the accuracy of the line-of-sight measurement.

11.8 DEFINITION OF GN&C MEASUREMENTS

Measurement requirements for the APOLLO GN&C system were needed in order to assure:

1. The availability of GN&C test points,
2. Proper design of GN&C system signal conditioning equipment for use in ground testing and in flight,
3. Compatibility with test requirements and maintenance philosophy,
4. Design of external signal conditioning, cabling, and telemetry (ground and flight),

5. Development of spacecraft automatic checkout equipment program and display requirements,
6. Development of data processing programs,
7. Development of operational and contingency test procedures, and
8. Meeting overall design requirements such as weight, volume, and accessibility to equipment during ground testing and flight.

Since the GN&C equipment shares spacecraft telemetry for both ground testing and flight, the need was urgent early in the program to coordinate GN&C system requirements with APOLLO contractors and with NASA. Also, advanced planning of the vast APOLLO facilities (such as the spacecraft automatic checkout equipment stations at each site and the Mission Spacecraft Real Time Control Center) depended on early agreements on measurement requirements. Measurement thus became a gating item for a great deal of APOLLO planning. Design personnel were pressed for answers about specific measurements even before all the control loops were defined. The completion of NASA-approved measurement requirements involved extensive coordination during the Project's entire duration.

APPENDIX A: ABSTRACTS OF SELECTED ENGINEERING AND RESEARCH REPORTS

INTRODUCTION

Some of the most important work performed at the Instrumentation Laboratory in the past eight years has been documented in engineering and research reports and in graduate theses, These reports have been presented at both national and international technical symposia. They are representntive of the technological base on which the APOLLO Project operated at the Laboratory; and even more important, they served as a disseminating medium to the widespread APOLLO mdustrial community for the design and deveiopment of concepts, analyses, and test criteria generated at the Laboratory and critical to the successful implementation of the APOLLO Primary Guidance, Navigation, and Control Systems by that community.

Appendix A presents selected and categorized reports written between August 1961 and April 1969. Included are engineering reports (E series), research reports (R series), and theses (T series). The intent is to supplement the report proper so that areas of special interest can be pursued further. The abstracts have been categorized into ten broad areas of interest and are listed in reverse chronological order. Many reports are cited in two categories: as an example, E-2274 (Inertial Components Reliability and Population Statistics) is listed under “Inertial Subsystem” and under “Reliability.”

The categories are outlined below:

1. ANALYSIS: mathematical or scientific investigations, derivation of physical equations, experimental studies
2. COMPUTER: hardware or software associated with AGC / LGC
3. DAP: digital autopilot (RCS and TVC)
4. GENERAL,: general information, miscellaneous studies
5. INERTIAL SUBSYSTEM: IMU, CDU, PTA, including components
6. OPTICS: optical subsystem, investigations on visibility, landmarks, horizon signatures
7. PLANNING: design plans, test plans, nonperiodic progress and status reports
8. RADAR: self-evident
9. RELIABILITY: environmental testing
10. SIMULATIONS: all-digital, hybrid

Section 1: ANALYSIS

E-2414: MISSION F (AS 505/CM 106/LM 49
G&N ERROR ANALYSIS
F.D. Grant, J. Werner, E.S. Muller / April 1969

This document comprises three principal sections. The first is concerned with the effects of CM/ LM IMU uncertainties (both Block II and measured data) on navigation and orbital uncertainties. This performance is compared with mission requirements. The effects of MSFN update uncertainties are also included. The second section covers the effects of radar and IMU uncertainties on rendezvous maneuver uncertainties. The third section gives data on G&N system performance and operation.

E-2406: GENERAL QUESTIONS ON KALMAN FILTERING IN NAVIGATION SYSTEMS
George T. Schmidt, Larry D. Brock (Hamilton Standard) / April 1969

The initial steps necessary for the application of Kalman filtering to navigation systems are discussed in general. The application of filtering to terrestrial navigation is illustrated by simple examples. Two methods are suggested for simplifying the problem in order that it can be more easily handled on a practical computer. Details are then given of the alignment and calibration of the inertial system in a spacecraft on top of a swaying launch vehicle. Practical implementation problems and actual hardware difficulties are discussed in detail.

E-2401: APPLICATION OF KALMAN FILTERING TECHNIQUES TO THE APOLLO PROGRAM
Richard H. Battin, Gerald M. Levine / April 1969

After a brief description of the various navigation instruments used aboard the APOLLO CM and LM, the general mathematical method of processing navigation data using the Kalman filter technique is given. The specific procedures used in the cislunar-midcourse, rendezvous, and orbit navigation programs in the two onboard computers are then presented. Finally, there is a discussion of the actual midcourse navigation flight data from the APOLLO 8 lunar mission.

E-2389: STAR-HORIZON MEASUREMENTS FOR ONBOARD SPACECRAFT NAVIGATION
James A. Hand / February 1969

An experiment for assessing the earth's horizon signature as a spacecraft navigational aid is being developed by MIT/IL for a Manned APOLLO Applications Program Flight. Optical measurement of the included angle between a known star and the horizon signature provides data similar to the star-landmark technique employed for implicit navigation. With both a specially designed star tracker and a horizon photometer to execute the automatic sightings, the onboard computer can use this similarity to derive a refined estimate of spacecraft state vector, The principal advantage-that the horizon signature is not subject to cloud cover obscuration as are landmark targets-is thereby exploited. The entire experiment—including sighting, data processing, and evaluation-is being designed for inflight performance. Testing progresses on the star-horizon sensors and the required computer programs.

E-2387: MISSION D (AS 504/CM 104,'LM 3) G&N ERROR ANALYSIS
F.D. Grant, J.E. Miller, E.S. Muller / February 1969

This document presents a performance summary of G&N System 209 in CM 104 and compares this performance with the requirements of Mission D. Test-data-calculated G&N uncertainties are based upon system performance as of 21 January 1969. One-sigma uncertainties for the CM are computed from above data for accelerometer and gyroscope performance parameters. These uncertainties are used to compute position, velocity, and perigee uncertainties.

As a comparison, the Block II Performance Specifications in the Master End Item for IMU uncertainties are also used to compute mission uncertainties. For earth orbit insertion, indicated perigee uncertainties are well above the required minimum perigee. For CM or LM burns, three-sigma perigee uncertainties are always less than 1 n. mi. Data are also given for the de-orbit burn and reentry uncertainties. Alignment uncertainty data are alsopresented. Effects of radar and IMU uncertainties on rendezvous maneuver uncertainties are also presented.

E-2301: G&N ERROR ANALYSIS FOR MISSION D
Frederic D. Grant / July 1968

The results are given of a G&N error study for Mission D covering both CM operations (COLOSSUS) and LM operations (SUNDANCE). The effect of CM and of LM IMU uncertainties on navigational uncertainties are considered. The effects of LM onboard update navigational uncertainties for the LM-active rendezvous maneuver and of uncertainties in ground tracking update (MSFN) are not presented in this report, since neither set of data was available in time. They will be given in a subsequent revision.

E--2287: MISSION C (AS 205/CM 101) G&N ERROR ANALYSIS
Frederic D. Grant, John E. Miller / September 1968

This document presents a performance summary of the G&N System in CM 101 and compares this performance with the requirements of Mission C (AS 205). The calculated G&N errors are based upon system performance as of 20 June 1968. Shown are the updated performance uncertainties derived using data up to this time period. More recent data through 5 September 1968 are included. The system performance of all the major components are shown. The chapter contains an alignment summary for the G&N system. The one-sigma standard deviation for the accelerometer and gyroscope performance parameters is calculated. These uncertainties are used to calculate the error in position and velocity.

As a comparison, the Block II Performance Specifications in the Master End Item for IMU uncertainties are also used to compute the mission position, velocity, and perigee uncertainties. These perigee uncertainties remain well above the required minimum perigee for the whole mission.

E-2262: STUDIES OF ONBOARD LUNAR ORBITAL NAVIGATION WITH UNKNOWN AND KNOWN LANDMARKS AND SOME OBSERVATIONS ON NONLINEAR EFFECTS
David S. Baker / August 1968

This report contains three separate onboard lunar orbital navigation studies with optical measurements:

1. Unknown and known landmark navigation comparisons
2. Measurements only on the landing site with an onboard MSFN matrix
3. Some nonlinearities in lunar orbital navigation

The first study evaluates the two methods of navigation (nine-dimensional state vector) and shows that known-landmark navigation is generally superior to unknown-landmark navigation. The second study indicates that small inertial and relative uncertainties result with the MSFN covariance matrix used. This covariance matrix is overly optimistic in light of recent information. The study of nonlinear effects indicates that Monte Carlo runs must be used to account for possible nonlinearities.

E-2220: MIT AERONOMY EXPERIMENT
J. Lawson, 0. Anderson, R. Newell, J. Geisler / February 1968

This report describes a tangential-viewing satellite experiment using a limb-scanning spectrophotometer specifically designed to measure the vertical distribution of density, temperature, and certain trace constituents on a global scale.

Unsolved meteorological problems related to the atmospheric region from 25-80 km are discussed. A method is outlined for deriving temperature and pressure distributions from measurements of atmospheric density. The physical significance and distributionof the trace substances (oxygen and ozone) are discussed, The utility of accurate atmospheric models for horizon-referenced navigation systems and several phenomena and techniques for establishing a horizon navigation reference are treated.

The design of an optimum mechanized experiment is discussed. A horizon profile inversion technique, designed to extract the maximum amount of data from atmospheric scans by means of Kalman filtering, is presented.

E-2215: ANALYTICAL AND EXPERIMENTAL INVESTIGATION OF THE TKERMAL RESISTANCE OF ANGULAR CONTACT INSTRUMENT BEARINGS
M. Michael Yovanovich / December 1967

This report covers the analytical and experimental investigations on the thermal resistance of a typical angular-contact instrument bearing done at MIT/IL, Thermal Laboratory, Bedford Flight Facility. The analytical investigationconsists of two distinct sections: the elastic contact analysis between a ball and race under typical loading conditions; and the thermal resistance analysis.

The elastic contact study shows that the contact area is always elliptical in shape and has linear dimensions very small relative to the linear dimensions of the ball or race. A detailed description enables the calculation of the contact area knowing only races, load, and the geometry and physical properties of the hall.

The thermal study shows that the thermal resistance at the ball-race interface is a function of the thermal conductivities of the ball and race, the semi-major axis of the elliptical contact area, and the ratio of the semi-major axis to the semi-minor axis. For symmetrical loading and heating conditions, the bearing resistance is inversely proportional to the number of balls. The heat transfer data (load, heat input, and temperature measurements) agree very well with the thermal resistance theory.

This report also analyzes the influence of thermai strain upon the thermal resistance when bearing loads are very light.

E-2204: ALGEBRAIC ANALYSIS AND MODELING OF SEQUENTIAL CIRCUITS
Ramon L. Alonso / November 1967

An algebraic approach to sequential logic circuits is presented which permits accurate simulation of circuit behavior, including gate delays, and convenient modeling of newer, more complicated devices.

E-2172 CLOUD OBSCURATION OF APOLLO LANDMARKS DERIVED FROM METEOROLOGICAL SATELLITE OBSERVATIONS
J. Barnes, D. Beren, A. Glaser / September 1967
(Allied Research Associates, Inc.)

Data from the Nimbus II and ESSA 3 satellites are used to determine the mean cloud cover over each of 100 landmarks to be used in the APOLLO onboard navigation system and the probability of sighting at least a specified number or' landmarks within the first two and one-half revolutions of each of three simuiated APOLLO missions. The sample periods were from 15 May through 31 August 1966 and from 9 October 1966 through 28 February 1967.

In the mission simulation program, a sighting probability based on cloud amount is derived for each observation. A Monte Carlo random numbering method is then employed to determine the number of landmarks sighted on each mission.

Cloud patterns derived from the results of the cloud statistics program are in close agreement with climatology. Satellite observed cloud amounts, however, are generally less than ground observed. The difference is believed to be due to the existence of small cumulus cells not resolved by the satellite and the overestimation of sky cover by ground observers.

The simulation program shows that the probability of sighting a specified number of landmarks on an APOLLO mission depends both on the cloud climatologies of the landmarks and on the number of possible sighting attempts. The number of sightings on a mission also depends on the derivation of the sighting probability for each observation and the time of mission launch. The human confusion factor may limit the sighting of terrestrial landmarks even under ideal cloud conditions.

E-2055: DATA ACQUISITION AND REDUCTION MIT X-15 HORIZON DEFINITION EXPERIMENT: PHASE II

John R. Lawson / December 1966

Theobjectives and data uses of the MIT X-15 Horizon Definition Experiment are discussed. The equipment, facilities, data formats, data reformatting procedures, subsystem calibration procedures, backup data, operational data, and the computations and analysis performed with the reformatted data and calibration data are discussed. A typical experiment flight sequence is described. Detailed associateddocuments are referenced throughout the report.

E-2031: ENTRY FLIGHT DATA FOR MISSION AS-202
Raymond Morth / October 1966

The flight recorder data for the entry portion of Mission AS-202 were excellent. They can be viewed as aerodynamic test results on a full-size model over a wide Mach number range. The data appearing in the original form on the fourth line of the printout come from three sources. Acceleration data (PIPAs), in the form of velocity increments over a 2-second interval, are derived from integrating accelerometers mounted in the IMU. The position and velocity vectors (RN and VN) are generated by the navigation routine in the AGC. Only single-precision numbers are printed for position and velocity, since the lower-order word of the double-precision representation of these numbers is masked by navigation errors. The least significant digit is worth 2.56 ft/sec for velocity and 3359 ft for position. Finally, the gimbal angles (AOG, AMG, and AIG) are derived from resolvers mounted on the gimbal shafts of the IMU. The gimbal angles come from a separate flight recorder that samples at 10 times per second but uses only the even-integer second samples.

The derived data are described in the glossary, Figure 1. The trajectory data appear on the first line of the printout. The aerodynamic data are on the second line. Dollar signs ($$$) for L/D and ROLL appear when the sensed acceleration is zero. In this case, these variables would have no meaning. The data start shortly after separation from the service module and continue until the end of the flight recorder data. At this time the main chutes are deployed. The computer program by which these data were reduced is also attached.

E- 1983: LUNAR ORBIT NAVIGATION PERFORMANCE WITH VARIOUS RANDOM AND SYSTEMATIC ERRORS
D.S. Baker, N.E. Sears, R.L. White / July 1966

A series of lunar-orbit navigation studies conducted for the CSM G&N system are summarized in this report. The navigation measurements used in these studies are chiefly optical sightings to mapped or unmapped lunar landmarks. The primary objective of these studies is to evaluate the orbital navigation performance of six- and nine-dimensional estimation techniques for variations in the following mission parameters:

1. Navigation sighting schedules
2. Lunar iandm ark mapping accuracy
3. Systematic errors in the knowledge of the lunar coordinate system and terrain
4. IMU alignment times during the orbital navigation phase

E- 1981: LEM PGNCS GUIDANCE EQUATIONS FOR A NOMINAL LUNAR LANDING MISSION
Norman E. Sears, P. G. Hoffman (GAEC) / May 1966

This report documents the LEM PGNCS guidance equations required for a nominal lunar- landing mission in a format suitable for GAEC simulations. The nominal lunar-landing mission considered is patterned after the Design Reference Mission I, starting with LGC initialization prior to CSM-LEM separation in lunar orbit, and includes the various LEM maneuvers required for landing, lunar launch, and rendezvous. LEM abort maneuvers are not considered. The guidance equations presented in this report are restricted to LGC mission programs and do not include general service programs involving moding, self testing, and interface monitoring, The mission programs represent current LGC requirements for the lunar-landing mission. This report is issued by GAEC as LMO-500-344.

E- 1973: STATISTICAL ESTIMATION IN INERTIAL NAVIGATION SYSTEMS
Larry D. Brock (Holloman CIGTF), George T. Schmidt / June 1966

The navigation system is developed here as an information filtering problem rather than a simulation of a deterministic physical system. The Kalman optimum linear filter is found to apply to the navigation problem if some techniqueis used to account for possible non-Gaussian maneuver accelerations. A major problem in the applicationof statistical techniques is the tremendous amount of computation required. Two methods are suggested that greatly reduce the amount of computation with a minimum degradationin system performance. In one method, physicalconsiderations are used to divide the total filter into smaller, simpler filters. In the other method, the optimum gains are precomputed and approximated by simple functions in the flight computer. The details are given using statistical filtering to align and calibrate the APOLLO inertial platform.

E- 1906: FOURIER ANALYSIS BY THE METHOD OF SELECTED ORDINATES
James N. Hallock / January 1966

The report presents the selected-ordinate method for approximating Fourier coefficients. Typically, a long and tedious point-by-point numerical integration is required to Fourier-analyze a curve, regardless of the accuracy desired. By the method of selected ordinates, lengthy computation time is replaced by a simple addition of numbers determined by the tables and graphs included in the report,

E-1901: VISIBILITY OF THE LEM WITH VARIOUS BACKGROUND LUMINANCES
Kenyon L. Zapf / January 1966

LEM visibility has been investigated under a variety of circumstances. In a large dark room, a 1/20-scale model of the LEM (finished with a specific aluminum paint) was illuminated by light from a single source to simulate the geometry of visible solar radiation in space. Photometry was used to determine the luminous intensity of the model in various aspects, and photography was used to determine the projected area of the model in each of these aspects. Dividing the intensity by the area and multiplying by the appropriate photometric scale factor results in the average luminance of the real LEM for a large number of aspect angles and solar phase angles.

In Part I of the report, the well-known Tiffany data are employed to calculate liminal range in nautical miles as a function of aspect angle for a single phase angle and uniform backgrounds of 10, 100, or 1000 foot-lamberts.

In Part II of the report, values of the maximum average luminance and the minimum average luminance of the LEM per rotation about its Y (pitch) axis are given as functions of the solar phase angle.

The roughly cubic shape of the LEM and the highly specular characteristics of the specific paint used in the tests produce large fluctuations of liminal range during a rotation of the LEM about its Y axis.

E-1836: STUDIES OF SPACECRAFT ATTITUDE CHANGE MODES AND RESULTING ATTITUDE FUEL CONSUMPTION

Kenneth Nordtvedt / August 1965

This study examines several types of vehicle attitude change modes and presents the logic used to calculate these maneuvers.

The attitude control fuel consumption for these maneuvers is studied with its minimization in mind.

E-1832: ATTITUDE MANEUVER OPTIMIZATION TO CONSERVE REACTION CONTROL PROPELLANTS

Robert Crisp, Donald W. Keene / August 1965

A comparison is made among three methods of spacecraft maneuvering on the basis of RCS propellant use. The first method uses a single direct rotation; the second and third methods use a sequence of rotations about principal axes of the spacecraft. As a starting point for this comparison, it was assumed that all maneuvers are equally likely. In geometrical terms, this implies that the single-equivalent rotation characterizing each maneuver has an axis-of-rotation vector direction uniformly distributed in space and a magnitude of rotation uniformly distributed in angle. In addition, the total time of maneuver for each method is equated to that for a single rotation of constant rate.

The optimum method depends on the specific maneuver performed. For simplicity, however, only one system can be instrumented. The method that uses the least fuel for most cases is the single direct rotation maneuvering at a constant rate. Under the conditions considered here, this is the preferred method to be instrumented in the spacecraft autopilot.

E-1819: A LUNAR PHOTOMETRIC FUNCTION: CURVES FOR CALCULATING THE DIFFERENTIAL LUMINANCE OF THE LUNAR SURFACE

John Gallagher, James M. Hallock / July 1965

A lunar photometric function and curves for calculating the differential luminance of the visible portion of the lunar surface are discussed. Detailed examples and comparisons with the results of several observers are presented.

E-1810: ANALYSIS OF LANDING RADAR GEOMETRY DURING REDESIGNATION OF LUNAR LANDING SITE

Janusz Sciegienny / July 1965

The LR consists of a velocity sensor and an altitude sensor,, The velocity sensor measures velocity components along three beams, and the altitude sensor measures a slant range along one beam. The geometry of radar beams during lunar landing is investigated for the present beam configuration and for a modified beam orientation. Three landing trajectories are simulated on the computer: a "standard" trajectory, a trajectory with forward redesignation of the initially selected landing site, and a trajectory with redesignation to the right of the initial landing site. The trajectory with redesignation to the right may result in unsatisfactory beam geometry in the present beam configuration, but the geometry of modified beam orientation remains within satisfactory bounds. In allcases, the geometry of modified beam orientation results in more favorable radar performance than the geometry of the present beam configuration,

E- 1809: COUPLED TWO-DEGREE-OF-FREEDOM RESPONSE OF RESILIENTLY-SUPPORTED RIGID BODY TO RANDOM EXCITATION

G. Dudley Shepard / December 1965

Vibration-isolation systems are often designed such that the rotational and the translational modes of vibration of the isolated body will be decoupled. Estimates of the system response are then based on a single-degree-of-freedom model. In practice, mode decoupling is only achieved within certain limits. In this report, the effects of mode coupling are studied bycomputing the response of a two-degree-of-freedom model excited by a random translatory ground motion. Graphical response functions are presented from which the angular response, the translational displacement, and accelerationof any point on the body can be calculated.

E-i760: PERFORMANCE MONITORING OF THE PGNS FOR UNPOWERED LUNAR ORBIT FLIGHT PHASES

Bernard A. Kriegsman, Donald S. Millard / September 1965

(Raytheon Resident Staff)

The problem of detecting noncatastrophic G&N equipment failures is considered under conditions where two similar independent G&N systems are provided on the vehicle. The primary interest is the unpowered phases of lunar orbital-rendezvous and landing missions. Failure detection is accomplished during the flight by performing two different statistical tests on the computed vehicle velocity corrections. The first test establishes the existence of the failure; the second test identifies the particular system with the failure. The key quantities used in these tests are the equiprobability-ellipsoid parameters for the velocity-correction estimates from each system and their difference. The basic analytical relations required for the computation of the ellipsoid parameters are given, including the necessary relations for propagating and updating the velocity-correction and velocity-correction-difference covariance matrixes. Results are presented from a digital computer simulation of the unpowered mission phases used to evaluate the proposed failure detection methods.

E-3687: AN OPTICAL EARTH HORIZON PROFILE BASED UPON TABULATED SOLUTIONS OF CHANDRASEKHAR'S EQUATIONS
Milo Wolff / October 1964

Calculations are made of the luminance of the earth's atmosphere as a function of the altitude of a line of sight from outer space, Results are obtained as a function of ground albedo, sun angle, and wavelength. Application to space navigation is discussed. Graphical solutions are given for manyvaried parameters. MIT/IL Report E-1634 is preliminary to this report.

. E-1663: A MANUAL ABORT GUIDANCE PROCEDURE BASED ON RANGE AND ALTITUDE MEASUREMENTS
Allan R. Klumpp / October 1964

A guidance procedure is presented that can be used to guide the LEM from the injection point of a roughly circular orbit to a rendezvous with the CM. The system depends only on measurement of altitude and of range to the CM. The system is based on the principles of the orbit perturbation theory and therefore is not applicable to guidance during thrusting.

E-1634: THE PROFILE OF AN EXPONENTIAL ATMOSPHERE VIEWED FROM OUTER SPACE AND CONSEQUENCES FOR SPACE NAVIGATION
Milo Wolff / September 1964

A review and preliminary analysis of the problem of using the earth's limb as a space-navigational reference presents calculations of the luminance of the earth's limb as a function of the altitude of the line of sight viewed from outer space. An exponential atmosphere model is used assuming single scattering of light with semi-empirical corrections. MIT/IL Report E-1687 continues the work using a more exact analysis.

E- 1560: A MANUAL LEM BACKUP GUIDANCE SYSTEM
Malcolm W. Johnston / April 1964

This report outlines a manual backup guidance system for LEM abort to rendezvous from any point in the powered descent or ascent phase, from subsequent transfer and rendezvous phases, or from the lunar surface. Powered ascent maneuvers are implemented with reference to onboard steering displays. Subsequent transfer and rendezvous maneuvers require steering data obtained through use of the tracking radar and primary G&N system on the CSM. A clear pericynthion is not obtained until after the transfer maneuver. Relinquishing the requirement for an initially clear pericynthion allows' implementation of a more efficient ascent profile, thus permitting low accuracy systems to live within the LEM characteristic velocity budget.

E-1540: A PRELIMINARY STUDY OF A BACKUP MANUAL NAVIGATION SCHEME
Kenneth Nordtvedt / August 1964

A method of manual space navigation for a spacecraft approaching the earth is developed. A transformation of the usual equation for a conic orbit is made which allows the use of a series of star-earth and earth-subtended angle measurements with simple manual arithmetical operations to predict the spacecraft orbit. Monte Carlo computer runs of the method show that one-minute-of-arc measurement errors predict spacecraft perigees well enough to determine "safe" reentry conditions.

Equations are developed to determine the midcourse impulses necessary to correct perigee at earth. The simple strategy of firing the midcourse impulse in the tangential direction of the plane of the spacecraft is within 95% of optimum use of fuel for a given perigee change.

E-1473: ANALYSIS OF LEM MISSION INERTIAL UNCERTAINTIES
John M. Dahlen, Malcolm W. Johnston / December 1963

The major sources contributing to LEM position and velocity uncertainties at perilune, hover, and burnout are individually investigated. The format used to trace eachcomponent uncertainty through these phases also serves as a common basis upon which the following alternate inertial schemes are compared:

1. Gimballed versus gimballess inertial measurement unit
2. Crude versus precision gyroscopes
3. Inertial realignment between injection and perilune versus no realignment

E- 1467: SAMPLED-DATA VELOCITY VECTOR CONTROL OF A SPACECRAFT
Erich K. Bender / March 1964

An investigation is made of two-dimensional steering of a spacecraft, incorporating a time-shared digital computer, during an orbital transfer m aneuver. Sampling-and-holding and pulse-width modulation effects on the attitude correction maneuver performed in response to a prealignment error are studied. The amount of additional fuel consumed for the entire orbital transfer maneuver due to this initial error is dependent on four factors. First, the control system characteristics are most influential. A system designed for fast speed of response causes the consumption of 10 times as much excess fuel as a slower system optimized with respect to fuel consumption. Second, excess fuel consumption is proportional to the square of the initial attitude error for a linear system. Third, the mode of transmitting a digital computer correction signal to the autopilot influences performance. Fourth, the rate at which the digital computer samples the controlled variable affects additional fuel consumption.

E-1429: LUNAR ORBIT DETERMINATION BY STAR OCCULTATIONS AND MSFN TRACKING

D.S. Baker, N.E. Sears, J.B. Suomala, R.L. White /September 1963

The accuracy of lunar-orbit determination by star occultation measurements and MSFN tracking is analyzed. Various performance accuracies are assumed both for star occultationmeasurements and MSFN single station and two station tracking, Combinations of star occultation and MSFN tracking are analyzed to determine the comparative effects of each type of measurement and the resultant performance.

E-1388: SUMMARY OF ERROR PROPAGATION IN AN INERTIAL SYSTEM

Janusz Sciegienny / August 1963

This summary of analytical studies of error propagation in an inertial system during orbital and suborbital flights shows that the error is caused by the initial condition errors and accelerometer bias. The analysis is based on the linearization of the error propagation equations; the analytical results are confirmed by computer simulation.

The tabulated results may be used to determine the propagation of position and velocity errors in the inertial system with no external corrections during the orbits around a planet, with no external corrections during the suborbital flight above a planet, or with an altimeter during the suborbital flight above a planet.

E- 1374: PROPAGATION OF ALTITUDE AND ALTITUDE RATE ERRORS DURING SUBORBITAL FLIGHT

Janusz Sciegienny / June 1963

This report contains a simplified analysis of altitude and altitude-rate error propagation in an inertial system with no external corrections during a suborbital flight. The errors are caused by the initial condition errors in altitude and altitude rate and by acceleration bias in the local vertical direction. During a short flight, the error propagation is determined primarily by the total flight duration. The vehicle position, velocity, and acceleration have only second-order effects on the error propagation, The results of this analysis maybe used to determine the error propagation during lunar descent, lunar hover, lunar ascent, lunar abort, earth ascent, and earth reentry. For a flight time shorter than 15 minutes, the results of analysis and the computer simulations agree within 10%.

E-1350: STATISTICAL DECISION THEORY FOR LOGISTICS PLANNING
Warren G. Briggs / May 1963

This paper presents some preliminary results of an operational interpretation and pilot implementation of some statistical-decision theory concepts. Some techniques are reviewed for Bayesian analysis with the gamma distribution family and the interpretation of life test data. A procedure is described for approximating subjective probability distributions and linear loss function and for optimizing decisions with this data. Some recent experience with these techniques for aerospace systems logistics planning is reported.

E- 1333: CLOUD COVER DATA FOR LANDMARK STUDIES
Chung L. Pu / March 1983

In studying the usefulness of landmarks in making space navigation measurements, data on cloud cover over certain selected landmarks were compiled from information provided'to MIT/IL by the U.S. Air Force. The sample landmarks, distributed around the world within latitudes of $\pm 35^{0}$, were selected for their unique terrain characteristics or other defining features that would make them suitable for recognition and tracking from a space vehicle. A cloud coverage of three-tenths or less was used as a criterion for landmark visibility; the mean number of days for each month of the year that a landmark meets this criterion is listed.

For the landmarks selected, data are presented showing trends in the best months for visibility conditions, the number of available landmarks by months for each visibility category, and annual averages, These data indicate that cloud cover is a problem that may limit the number of useful landmarks for a mission, unless special efforts are made in the flight plan to take advantage of optimum conditions.

E-1285: SOME LUMINANCE VALUES FOR THE SUN, EARTH, AND MOON
Arthur C. Hardy (Consultant) / January 1963

Using the solar constant, the luminous efficiency of solar radiation, and the visual albedo values of the earth and the moon, average luminance values are calculated for those cases with simple geometry.

E-1278: MIT/IL SPACE IMPLEMENTATION: INTERIM REPORT
Gordon R. Gilbert / January 1963

This interim report presents the results of an MIT/IL study of SPACE integration problems. Following a summary of MIT/IL recommendations for SPACE implementation, the report reviews the major study results, including detailed considerations of test procedure definition, G&N data monitoring problems, estimated SPACE programing efforts, and MIT/IL SPACE console requirements. The final section includes considerations of future MIT efforts necessary for SPACE implementation and some further integration probiems to be studied.

E-1261: APPLICATION OF MIDCOURSE GUIDANCE TECHNIQUE TO ORBIT DETERMINATION
Gerald M. Levine / December 1962

A navigation procedure is presented consisting of a sequence of celestial observations together with a procedure for processing each observation as it is made. The technique is applied to the determinationof the position and velocity of a spacecraft in a near orbit of a planet.

Landmark sightings are considered to be the only type of observations used, The angles between the landmark directions and a fixed reference axis system are assumed to be the data obtained.

This method of orbital navigation is considered in detail for the case of landmarks on the earth's surface. Results of adigital-computer simulation are given.

E-1257: ATMOSPHERIC REFRACTION AS A MEANS OF HORIZON DETERMINATION
L.J. Lareau / Demember 1962

This report analyzes the feasibility of employing the phenomenon of atmospheric refraction as a means of horizon determination. Basic equations are developed from two points of view-one purely analytical, based on a differential form of Snell's Law; the other on purely empirical data, namely that available in the 1959 ARDC standard atmosphere. Possible error sources are thenconsidered and analyzed, and conclusions are given about the possible accuracy of this horizon determination method.

E-1256: APOLLO MIDCOURSE GUIDANCE
John W. Hursh / November 1962

Orbital and midcourse G&N requirements for the lunar mission are presented, and techniques for meeting these requirements using an onboard guidance system are discussed. The investigation of phenomena capable of providing navigation measurements is summarized. Navigation sighting instrumentation under design for operation during the orbital and midcourse mission phases are outlined.

E-1196: ANALYSIS OF TWO LUNAR LANDING TECHNIQUES PROVIDING DIRECT LANDING SITE VISIBILITY PRIOR TO TOUCHDOWN
Malcolm W. Johnston / July 1962

Present APOLLO vehicle configurations do not allow direct rearward visibility. If such visibility is necessary during lunar landing, the vehicle must be reoriented with respect to the line of sight to the landing area. Two landing modes (Mode I and Mode II) which provide direct visibility of the landing area for some time interval before touchdown are compared with a “reference” trajectory.

Mode I provides visibility for 60 seconds before a final vertical descent from a finite altitude and vertical velocity. Characteristic velocity penalties, relative to the “reference” trajectory, vary from 300 to 450 ft/sec for this maneuver. Much of this penalty may be eliminated if the final vertical descent phase can be considered as fulfilling part of the hover requirement.

Mode II provides visibility more simply. During this phase, the vehicle "flies" a straight-line path to touchdown. The characteristic velocity penalty for 60 seconds of visibility is 250 ft/sec. None of this can be eliminated as partially fulfilling the hover requirement, because this mode has no final vertical descent phase.

The altitudes, surface ranges, and velocities covered during the Mode I visibility phase are an order of magnitude greater than in Mode II.

E-1195: EARTH ORBITAL RENDEZVOUS
Norman E. Sears / May 1962

This report presents the mission model used for the current analysis and evaluation of the guidance system requirements for earth-orbital rendezvous, Mission phases from initial launch to earth-moon trajectory injection for circumlunar and lunar landing missions are included. Rendezvous in earth orbit is assumed for both missions. Summarized are:

1. Vehicle configurations and propulsion systems assumed for lunar-landing and circumlunar missions
2. Initial launch requirements for the unmanned earth-moon injection vehicle
3. Rendezvous vehicle launch timing requirements and the resulting waiting time requirements in an intermediate parking orbit
4. Orbit-to-orbit transfer phase between the two parking orbits
5. Rendezvous and docking phase
6. Delayed earth-moon injection due to late rendezvous

E-1131: SUMMARY OF DATA FOR A VARIETY OF CIRCUMLUNAR TRAJECTORIES
Richard H. Battin, James S. Miller / February 1962

Presented herein in tabular form is selected information from each of 256 approximate circumlunar trajectories. These trajectories have been calculated using a method of matching pieces of conic trajectories with occupied focus at the center of either the earth or the moon, depending on whether the vehicle is outside or inside the moon's sphere of influence.

E-1124: PRELIMINARY STUDY OF ABORTS FROM CIRCUMLUNAR TRAJECTORIES

James S. Miller, John J. Deyst, Jr. / March 1962

This study presents a technique for determining regions on the earth that can be reached by a space vehicle 'after an abort from a circumlunar flight. The part of the abort path outside the atmosphere is assumed to be a conic section; the effects of the perturbing forces of the sun, moon, and of earth oblateness have been ignored. All paths are computed with the constraints that a specified flight-path (reentry) angle must occur at the altitude of reentry and that not more than a specified velocity increment magnitude be required. Atmospheric flights of constant characteristics are assumed. An extension of this technique is also described enabling the computation of the abort path and required velocity increment for landing at a specified site.

Results are presented showing possible landing locations for aborts from two circumlunar trajectories. The areas are restricted by the upper limit placed on the applied velocity increment and by a landing requirement not more than 24 hours later than the earliest landing possible for the specified velocity limit. A summary is also given of particular abort flights to a site near San Antonio, Texas.

E-1118: PRELIMINARY INVESTIGATION OF MIDCOURSE MANEUVER FUEL REQUIREMENTS FOR APOLLO SPACECRAFT

Roger A. Scholten, Peter J. Philliou / March 1962

Battin's midcourse-guidance analysis and a practical sextant are the tools used in a method presented here for determining the propellant required to orient the APOLLO spacecraft in order to take the best celestial measurements for earth-to-moon and moon-to-earth midcourse trajectories. A study is made using a fuel criterion per measurement and a fuel limit per reading for roll and pitch in order to determine their effect on target error and total fuel expended. Fuel requirements for velocity corrections are considered, but further studies are in progress for refining the results. In the appendixes, the results of the aforementioned studies are applied to a specific trajectory of a specific spacecraft,

E-1106: ANALYSIS OF GUIDANCE TECHNIQUES FOR ACHIEVING ORBITAL RENDEZVOUS
Philip G. Felleman | January 1962

Phase-plane analysis with numerical examples is used to examine several types of satellite-rendezvous terminal guidance techniques. The rendezvous methods differ essentially in the type of propulsion used-for example, on-off thrusting or continuous thrusting. The analysis demonstrates the relationship among propulsive characteristics, initial range and velocity, and onboard-tracking-equipment requirements. A mathematical derivation of a guidance philosophy commensurate with both propulsion capability and initial condition variations is presented, and the relationship between throttling capability and launch timing accuracy is examined. A comparison between propellant requirements for the different types of rendezvous guidance systems is made, and several numerical examples of each are presented.

E-1054: ANALYSIS OF THE PNP-NPN LATCH CIRCUIT
Joseph J. Rocchio | September 1961

The two-transistor latch is a bistable circuit useful as a digital building block. The latch exhibits several interesting properties. These include very low dissipation' in the OFF state, double-polarity outputs, several drive alternatives, and regeneration in the ON state. These notes investigate the latch circuit both qualitatively and quantitatively. Static and transient characteristics are treated, and numerical results are presented based on nominal circuit and transistor parameter values.

R-618: A NEW SOLUTION FOR LAMBERT'S PROBLEM
Richard H. Battin | August 1968

A new universal solution of Lambert's problem, encompassing elliptic, parabolic, and hyperbolic orbits is presented in which the independent variable has an immediate physical interpretation. In terms of this new variable, the time-of-flight equation is the sum of two hypergeometric functions, while the equations for the terminal velocity vectors are characterized by elegant simplicity. Although the hypergeometric functions are expressible in terms of elementary functions, the resulting forms are not computationally useful when the orbit is nearly parabolic. On the other hand, by means of power series and continued fraction expansions, extremely useful algorithms (continuous through the parabolic case) are obtained for calculation of the time of flight. Another distinct advantage of this formulation is the lack of computational difficulty that usually accompanies the case for which the transfer angle is 180°.

R-597: FLUCTUATION ERRORS OF DOPPLER SENSORS AT LOW VELOCITIES
Walter E. Tanner / December 1967

An exact theory is developed for the fluctuation error of an idealized Doppler velocity sensor. Particular attention is given to sensor performance in the low-velocity regime, where signal bandwidth and smoothing bandwidth are of the same order of magnitude. The theory for the idealized sensor is then compared with flight-test data from an actual sensor, and a performance figure (tracker noise figure) is established and discussed.

R-532: REENTRY GUIDANCE FOR APOLLO
Raymond Morth / January 1966

The design of the reentry steering for the APOLLO spacecraft is discussed in detail. Error performance is the critical design factor, A great deal of effort has been spent to find a system that would steer properly in spite of navigation errors. The most significant error is in the initial indicated rate-of-climb. To achieve this error performance, a computed reference trajectory is used during the critical supercircular phase with control gains that have been chosen on a statistical basis to minimize the miss distance.

Other factors affecting the design are presented. Some of these are requirements made by the heat shield and monitor system on the trajectory and limitations by the roll control system fuel on the attitude maneuvers.

The performance of the automatic self-contained system using a digital computer is presented, Guidance to all possible points throughout the entry corridor is demonstrated. The detailed computer logic for the AGC is also presented.

R-531: WHGLE NUMBER STRAPDOWN COMPUTATIONS
J. C. Pennypacker / February 1966

An inertial navigation system employing a gimballess inertial measurement unit requires an analytical transformation of the vehicle coordinate system into the inertial coordinate system. An algorithm is developed for maintaining an up-to-date transformation matrix in a general purpose whole-number computer. A method of implementing the algorithm in the AGC is described. The performance of the algorithm, the effects of flight profile parameters upon the accuracy of the algorithm, and the effects of certain equipment constraints are detailed in the results of computer simulations. Extensive computer simulations were conducted to verify the validity of the algorithm; although conclusions about navigation computer design are drawn from the simulation results, raw simulation data are included for individual interpretation. For comparison, the results from simulation of a digital differential analyzer are included. For at least certain missions, general-purpose computers can be built to perform the strapdown computation with sufficient accuracy and speed while not significantly detracting from the other computer tasks.

R- 505: TEST RESULTS ON A PULSE-TORQUED PENDULUM WITH A PERMANENT-MAGNET-TYPE TORQUE GENERATOR
Standley H. Goodwin / November 1965

An analysis is made of the theory of operation and the high level of performance of an accurate pulse-torqued pendulous accelerometer containing a permanent-magnet torque generator. Experimental results verify the analysis. The key to accelerometer accuracy is the torque generator; therefore, a sirnple analysis of the torque generator is presented, and the results are applied to the instrument design. The operation mode and its effects on the electronics design are discussed. In addition, the effect of the permanent-magnet torque generator on electronics design is considered. Where possible, data are presented to illustrate accelerometer performance. Scale-factor stability, bias stability, and linearity are shown.

R-495: APOLLO SPACECRAFT GUIDANCE SYSTEM
Milton B. Trageser, David G. Hoag / June 1965

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

R-491: ON OPTIMUM STEERING TO ACHIEVE "REQUIRED VELOCITY"

Bairaj G. Sokkapa / April 1965

A well-known method of onboard guidance of space vehicles is based on the concept of a "required velocity." The dynamics of the powered-flight phase of the vehicle can be written in terms of a velocity-to-be-gained:

$$\dot{\underline{v}}_g = -[C^*]\,\underline{v}_g - \underline{a},$$

where

$$\underline{v}_g = \underline{v}_r - \underline{v},$$

$$[C] = \left[\frac{\partial \underline{v}_r}{\partial_r}\right],$$

an $\underline{a}$ is the thrust acceleration, $\underline{v}_r$ is the required velocity.

In general $[C^*]$ is a function of position $\underline{r}$ and, hence, time-varying. With reasonable approximations, this equation can be considered equivalent to the familiar "state equation" of a dynamic system:

$$\dot{\underline{x}} = [A]\,\underline{x} + \underline{u}$$

In this paper, the necessary condition that must be satisfied by a fuel-optimum guidance law is developed for a system where A is linear and time-invariant and -U- is a known function of time. From this condition, with first order approximations, an explicit guidance law is derived. Some conclusions that have been previously obtained by other methods are extracted from the solution.

Numerical examples are included to indicate the performance of this law in comparison to other familiar steering laws. The near-optimum law is shown to yield excellent results in practical problems in which the assumptions of time-invariance and linearity are not quite true. The results are compared with optimum solutions obtained with the calculus of variations. Computational aspects of the law's implementation are discussed. The mathematical form of this law is shown to result in some computational simplifications.

R-482: A METHOD OF ORBITAL NAVIGATION USING OPTICAL SIGHTINGS TO UNKNOWN LANDMARKS
Gerald M. Levine | March 1965

Recursive space navigation and its application to navigation in a near orbit of a planet by means of measuring the directions to known landmarks is discussed. A less restrictive method of recursive orbital navigation is presented in which it is not necessary to identify the landmarks.

Navigational data are obtained from two optical sightings to the same unknown landmark. The landmark position and the two points from which the sightings are made determine a plane. At one position between the two sighting points-the normal point-the velocity vector of the spacecraft has no component perpendicular to the plane. The location of the normal point is obtained as a function of the two sighting points only. It is independent of both the path between the two points and the landmark location. The unknown-landmark orbital navigation procedure is then constructed from these results. Computer-simulation results are presented using this method for both earth and lunar-orbital navigation.

R-479: A UNIFIED METHOD OF GENERATING CONIC SECTIONS
William Marscher | February 1965

This report presents a unified method of generating conic sections under various constraints that are of practical interest. Both the universal and nonuniversal approach are considered. The particular problems selected to demonstrate the utility of the method are Kepler's, Lambert's, the Reentry, the Time-Theta., and the Time-Radius problems.

R-466: VIBRATION EFFECTS ON APOLLO GUIDANCE
Frederic D. Grant / October 1964

This report describes the significant vibration effects on inertial component guidance performance. The transmission of linear and angular vibrations from spacecraft frame through the navigation base to the Stable Member is considered. Most significant vibrationeffects produce rectified errors that result in equivalent bias drift or bias error in a constant vibration field. Vibration effects on trajectory cutoff errors are presented for the different APOLLO trajectories. An appendix contains brief descriptions of the significant vibration effects on inertial components.

R-456: A GENERAL, EXPLICIT, OPTIMIZING GUIDANCE LAW FOR ROCKET-PROPELLED SPACEFLIGHT
George W. Cherry / August 1964

An explicit optimizing method for guiding rocket-propelled vehicles is derived and applied. The method provides a universal solution to the many kinds of boundary-value problems encountered in powered-flight guidance. The derivation of the steering laws is extremely simple and avoids the use of difficult or specialized mathematics. The method is called E Guidance.

An essential feature is the E Matrix, that maps the separation between the current boundary conditions and the desired boundary conditions into thrust-allocation guidance coefficients. These coefficients determine the required allocation of thrust acceleration along controlled coordinate axes. The guidance laws can control final coordinates of position as well as final components of velocity for throttleable as well as fixed-thrust rockets.

Because of the generality of E Guidance, the method is particularly applicable to many-faceted complex space missions. Auniversal powered-flight guidance program for such a mission is described. The program provides for each type of powered-flight guidance problem by linking the appropriate set of stored-program subroutines. The universal powered-flight guidance program is tailored to the peculiar powers of a digital computer by exploiting the machine's switching, branching, and decision-making capabilities.

R-447: NAVIGATION FOR THE APOLLO PROGRAM
J.M. Dahlen, J.L. Nevins | May 1964

This paper explains the basic navigation concepts and techniques used in designing the APOLLO G&N system. This system has the capability to control the spacecraft path throughout its mission, which, for the basic lunar-landingmi.ssion, contains 15 distinct guidance and navigation phases. Also required is the capability to guide aborts from all phases before transearth injection. To perform these functions, three distinct tasks must be accomplished:

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

R-417: Rev. 1 A CLASS OF UNIFIED EXPLICIT METHODS FOR STEERING THROTTLEABLE AND FIXED-THRUST ROCKETS
George W. Cherry | January 1964

This paper deals with the generation of a class of explicit guidance laws for computing rocket- steering and throttling commands. The steering laws provide control of final components of the velocity vector as well as, when it is appropriate, control of final position coordinates. The viewpoint taken in the paper is that the commanded thrust vector can be computed inflight as the explicit solution to a two-point boundary-value problem. Thus, the commanded thrust vector is found by a direct solution of the appropriate equations of motion, subject to the initial boundary condition of the vehicle's instantaneous measured state and final boundary condition of thevehicle's desired state. Three goals motivate the synthesis of the guidance equations: simplicity of the algorithms which must be programed on the vehicle-borne computer; fuel economy in traveling from the initial boundary condition to the final boundary condition; independence of the steering laws from standard conditions and nominal trajectories. To illustrate the guidance method, the paper discusses three principal thrusting phases of a lunar-reconnaissance and landing mission. Programing and simulation of the guidance laws for the lunar-landing mission show the achievement of the three design goals.

R-417: A UNIFIED EXPLICIT TECHNIQUE FOR PERFORMING ORBITAL INSERTION, SOFT LANDING, AND RENDEZVOUS WITH A THROTTLEABLE ROCKET-PROPELLED SPACE VEHICLE
George W. Cherry / August 1963

This paper presents a family of guidance laws for providing rocket-steering and throttling commands during the principal engine-on phases of a planetary reconnaissance and landing. The guidance laws presented are derived as explicit solutions to the two-point boundary value problems of guided powered trajectories. It is emphasized that the subsequent flexibility and independence of these laws from particular mission profiles is an inherent part of this guidance method.

R-38:: ORBITAL ELEMENT VARIATIONS FOR A RODY IN ORBIT AROUND THE MOON
Stephen J. Madden / July 1963

A disturbing function is obtained that represents the effects of the earth's attraction and the nonspherical nature of the moon's gravitational field on a body orbiting the moon. This disturbing function is then used in the variation of orbital-elements equations which are integrated numerically.

R-385: INERTIAL, ORIENTATION OF THE MOON
Richard C. Hutchinson / October 1962

This report presents a method of describing the moon's inertial orientation. A set of inertial axes and a set of moon-fixed axes are defined, and the matrix of direction cosines between the two axis frames is derived. This matrix is the product of four matrixes. One is constant; the others involve the sine and cosine of angles that are linear functions of time.

R-382: UNIVERSAL FORMULAE FOR CONIC TRWECTORY CALCULATIONS
Richard H. Battin / September 1962

A generalization of Kepler's equation, which relates time and position in a conic orbit, has been made by Professor Samuel Herrick. By defining a pair of new transcendental functions, Herrick was able to develop a general form for Kepler's equation. This form is equally applicable for the ellipse, parabola, and hyperbola. This paper extends this idea to a somewhat different trajectory problem. New transcendental functions are used that are necessarily different from those suggested by Herrick, but, as will be shown, are also full capable of providing a generalization of Kepler's equation.

The formulae to be developed for the solution of Kepler's Problem and Lambert's Problem have distinct advantages over the more conventional ones. First, they are truly universal in that they apply to all types of conic trajectories, i.e., ellipse, parabola, or hyperbola. It is unnecessary to know the nature of the conic for their application. Second, transition from one conic to another is continuous and with no ambiguities or confusing indeterminate forms.

R-376: TWO-IMPULSE ABORT TRAJECTORIES FROM TRANSLUNAR FLIGHT
Renwick E. Curry / October 1962

Before embarking on manned translunar flight, a spacecraft must have the capability of immediately returning to earth if it should become desirable to terminate the mission prematurely. It is the responsibility of the guidance system to calculate the one or more trajectories, from widely varying initial conditions, that will bring the vehicle to a safe entry point from which it can maneuver to a preselected landing site. A two-impulse abort may hold a constant landing azimuth (ground track) to facilitate preentry and entry tracking and also to use all the available fuel to return as early as possible, Results of general two-impulse aborts are presented under theassumption of impulsive thrusting in an inverse-square force field. An alternate method is suggested that retains the desirable characteristics of the general two-impulse solution and has the safety factor of entering a proper corridor if the second restart fails. The ease of computation, comparable velocity requirements, and return times make the alternate method an attractive solution for the onboard computation of a two-impulse abort,

R-372: VELOCITY STEERING STUDIES FOR THE APOLLO MISSION
James H. Flanders / August 1962

This report presents a preliminary analysis of a velocity steering loop in the MIT/IL APOLLO guidance equipment for use when vector velocity changes are being commanded during the APOLLO Mission. The study provides preliminary verification of proposed signal quantization levels and computer sampling intervals. Based only on ideal spacecraft attitude-loop dynamics and planar maneuvers, the results indicate that transient response times of transverse velocity errors will be about 7 seconds. The steady-state limit cycle involves velocity excursions that do not exceed one velocity quantum (5.22 cm/sec).

R-353: CIRCUMLUNAR TRAJECTORY CALCULATIONS
Richard H. Battin, James S. Miller / April 1962

This report describes a technique using a realistic model of the earth-moon system to generate circumlunar trajectories. First, an approximate trajectory is determined using pieced conics: an ellipse from earth to the sphere of influence of the moon; a hyperbola around the moon; and an ellipse from the sphere of influence back to earth. Second, an exact trajectory is calculated based on the initial conditions obtained from the simplified model.

In the second phase of the calculations, certain quantities are invariant: the position vectors at injection and at return vacuum perigee; the total time of flight. Thus, the pieced-conic solutions, rapidly generated on a computer, can be used for trajectory planning and analysis.

R-341: A STATISTICAL-OPTIMIZING NAVIGATION PROCEDURE FOR SPACECRAFT FLIGHT
Richard H. Battin / September 1961

This paper provides a basis for determining the best celestial measurements and the proper times to implement velocity corrections. Fundamental to the navigation system is a procedure for processing celestial measurement data that permits incorporation of each individual measurement, as it is made, in order to provide an improved estimate of position and velocity. To "optimize" the navigation, a statistical evaluation of a number of alternative courses of action is made. The following considerations form the basis of a decision:

1. Which star and planet combination provide the "best" available **observation?**
2. Does the best observation give a sufficient reduction **in** the predicted target error to warrant making the measurement?
3. Is the uncertainty in the indicated velocity correction a small enough percentage of the correction itself to justify an engine restart and propellant expenditure?

Numerical results are presented that illustrate the effectiveness of this approach to the space navigation problem.

T-474: A NEW TECHNIQUE FOR THE OPTIMAL SMOOTHING OF DATA
Donald C. Fraser | January 1967

Optimum smoothing is a data reduction technique that uses all available information in the best way possible. Optimum smoothing differs from optimum filtering in that the optimum smoother can use information that is in the future with respect to some particular time of Interest. The subject of this thesis is the solution of the optimum smoothing problem when the system and measurement functions are linear. The application of this solution to linear and nonlinear dynamic **systems** is described.

The optimal smoother is formulated as a combination of two optimum filters, one of which works forward over the data and the other of which works backward. A technique developed to process the filter working backward over the data is extended to treat the general problem of optimal filtering when there is no prior estimate. The resulting technique constitutes a completely recursive method of starting a Kalman filter when no a prior information is available. A smoothability condition is derived. This condition enables the user to determine whether smoothing will yield results that cannot be obtained in a simpler manner from an optimum filter estimate. Several numerical problems are identified and solved.

T-462: A METHOD OF DETERMINING A VEHICLE'S POSITION AND VELOCITY BY TRACKING UNKNOWN LANDMARKS

Howard C. Mathews | June 1966

This paper attempts to find a method of determining the position and velocity vectors of a vehicle at some instant of time such that the vehicle's differential equations of motion can be integrated. The determination of these state elements is carried out by sighting and tracking unknown landmarks. The method differs from more common methods of tracking unknown landmarks in that the state vector does not include landmark coordinates. The state vector is determined solely by observing changes in the tracking angle as the vehicle passes over the landmark.

An expression is derived from the cotangent of the angle measured on the actual orbit projected onto the nominal plane minus the cotangent of the angle measured on a nominal circular orbit at the same time. This difference is a function of the initial in-plane errors away from the nominal. By tracking four landmarks, it should be possible to determine these injection errors. As the problem is formulated in Chapter 4, it appears impossible to determine the injection errors with any degree of certainty. The basic idea does appear to be sound, and it may be possible to derive another method of solution that will reduce this uncertainty.

T-461: SURFACE IRREGULARITY TRACKING SYSTEM

Steven R. Croopnick | June 1966

This paper explores the feasibility of deriving information about the relative motion of a vehicle to a random terrain by sensing the image motion of the terrain. The ensuing analysis differs from conventional techniques in that the information is derived from a comparison of the outputs of sensors in an array and not on a measurement derived from a single sensor. It is shown, through the use of a statistical information theory approach, that the accuracy of an ideal system is dependent on the total distance of image travel, provided there are sufficient terrain irregularities. A simple sensor was constructed and tested to demonstrate the principles involved. Typical examples of systems using these sensor arrays are presented and their accuracies are predicted.

T-446: ON THE DESIGN OF NEARLY OPTIMAL, LINEAR TIME-VARYING SAMPLED-DATA STOCHASTIC CONTROLLERS

William S. Widnall / September 1966

The design of nearly optimal time-varying controllers for stochastic processes is considered. Attention is limited to linear plants and linear-sampled datacontrollers. It is assumed that the performance can be evaluated adequately by some quadratic cost functional. If the capacity of a control computer is limited, a simplified control computer program operating with a short sample interval may outperform the optimal controller forced to operate with a longer sample interval.

A canonical form is developed for time-varying sampled-data controllers. This form minimizes the number of multiplications and additions required to realize the desired input to output relationship. Several examples are given that illustrate the usefulness of optimal control theory in computer-limited applications: the possible performance of practical designs can be computed; lower-order time-varying controllers using optimal solutions to simplified problems can be designed; efficient constant-coefficient stochastic controllers can be designed. A method is developed for optimizing simplified controller designs using a simplified time-varying sampled-data controller with a few parameters left undetermined. The computation of the mean operating cost of the design defines the cost as a function of the free parameters. Efficient numerical procedures for finding the minimum cost optimized design are discussed.

T-414: APPLICATION OF STATISTICAL ESTIMATION TO NAVIGATION SYSTEMS

Larry D. Brock / June 1965

In this thesis, the navigation system is developed from the beginning as a statistical problem, It is recognized that the vehicle path as well as the errors involved can be represented by random signals and that the most that can be known in advance is their statistical characteristics. It is impossible to determine exactly the position and velocity of a vehicle. Only statistical distributions, i.e., the probabilities that the position and velocity have various values, can be determined. Obtaining all of the statistical information available from navigation measurements would be very difficult in general, but if a linearized variation about a known nominal is sufficient to describe the navigation process and if the amplitude distributions of these variations can be assumed to be Gaussian, then the entire estimation procedure can be specified by relatively straightforward mathematical equations. It is found that these assumptions hold for the navigation process except for vehicle maneuvers. The maneuvers are handled by a modification of the basic linear system. Simplifications are also made to adapt the required computations to a practical navigation computer.

T-413: CLOSED-LOOP NEAR-OPTIMUM STEERING FOR A CLASS OF SPACE MISSIONS

Frederic H. Martin | May 1965

A closed-loop control law for space-vehicle guidance is proposed in an attempt to achieve fuel-efficient performance for a class of space missions. Essentially encompassing all objectives that might be accomplished by the application of a single acceleration impulse (if infinite thrust were available), each mission is described by a unique "required velocity" and an accompanying instantaneous "velocity difference." Assuming a non-throttleable "moderate to high" thrust engine with prescribed acceleration time-history, the control task is defined as reducing the velocity-difference magnitude to zero in minimum time.

The formulation of an efficient steering algorithm is approached in two separate steps: variational techniques are applied in order to derive significant information concerning the optimum behavior of the system; this information is then used through the introduction of additional degrees of freedom into a conveniently selected closed-loop control scheme, thereby approximating an optimum controller. The analytic development is facilitated by a linearizing assumption in which certain partial derivatives are considered to be invariant in the neighborhood of an implicit reference trajectory (a trajectory not to be stored as part of the final control law).

T-386: GUIDANCE SYSTEM MONTTORING FOR A LUNAR LANDING

Roger J. Phaneuf | May 1964

The general problem of guidance-system monitoring is discussed briefly, and the particular characteristics of a manned lunar-landing problem are explained. A digital-computer simulation sets guidance-system performance error limits, that are used in turn to define the region of monitor responsibility. Techniques are then described for developing and evaluating a monitoring system for the lunar landing,

T-329: AN INVESTIGATION OF AN EMERGENCY BACKUP GUIDANCE AND NAVIGATION PROCEDURE FOR THE TRANSEARTH PHASE OF THE APOLLO MISSION

A. William Breck, Charles L. Wilson / June 1963

A gross failure situation is postulated for the transearth phase of the APOLLO mission. In the postulated situation, the absolute minimum of electrical power, manually controlled guidance and navigation equipment, vehicle attitude control, and thrusting devices is operable. An emergency guidance and navigation procedure is described that relies upon angular measurements made with the scanning telescope of the APOLLO vehicle. The procedure includes sequential earth disk and earth-to-star angular measurements in optically-established inertial coordinates and subsequent calculation of two-body orbital parameters, vacuum perigee radius, and velocity corrections. An error-analysis procedure permitting quantitative assessment of the guidance and navigation procedure is devised and applied. The probability of reentry in the desired 60-mile-deep corridor is no greater than 10% for the procedure described. One-sigma values of the angular measurement error of about 5 seconds of arc are required for 50% probability of entering the corridor.

E-2307: RECOVERY FROM TRANSIENT FAILURES OF THE APOLLO GUIDANCE COMPUTER
Edward M. Copps, Jr. / August 1968

In the AGC, nearly 100,000 word transfers occur each second. A random error rate of one in 10^{12} actions would be considered good in today's technology, but at that rate an error might occur within several hundred hours. Added to the random error rate are externally induced errors (including power and signal Interface transients), program overloads, and operator errors. The incidence of induced errors has so far been very much greater than random errors (if indeed there have been any at all) in AGC experience.

The AGC is a control computer and has been designed to detect and to recover from random or induced transient failures. The techniques used are the subject of this paper. The software associated with a restart is described, with **a typical** program flow derived from an APOLLO Mission Program. The amount of memory assigned to restart protection is stated. Several interesting sidelights are briefly discussed, such as rnanual break-in to prevent restart looping, the adoption of the restart technique for scheduling the termination of active programs, and the use of the failure recovery technique to remove a temporary computer overload.

E-2280: SOLID STATE DSKY STUDY
L. David Hanley / June 1968

The component problems and the events leading to the DSKY redesign contract are described. In a separate section, the problems involved in modifying the DSKY are discussed, and a review is given of the available options and the reasons for selecting the final design. The appendixes cover the test data measured on some representative modules.

E-2254: AUXILIARY MEMORY SYSTEM: FINAL REPORT ON PHASE I
Donald J. Bowler / April 1968

An auxiliary memory has been proposed as a means of augmenting the storage capability of the AGC. This study analyzes the impact of this additional memory on the present programing structure. Augmenting the function capability of the AGC is desirable and several possibilities are proposed.

E-2204: ALGEBRAIC ANALYSIS AND MODELING OF SEQUENTIAL CIRCUITS
Ramon L. Alonso / November 196'7

An algebraic approach to sequential logic circuits is presented which permits accurate simulation of circuit behavior, including gate delays, and convenient modeling of newer, more complicated devices.

E-2159: COMPUTER-CONTROLLED STEERING OF THE APOLLO SPACECRAFT
Frederic H. Martin, Richard H. Battin / August 1967

The digital guidance computer is the central control element in the APOLLO control, guidance, and navigation system. Efficient operation of the guidance computer during any mission phase requires the performance of many different functions occurring at approximately the same time. The computer must process input data (in the form of velocity increments, gimbal angles, system status signals, astronaut keyboard commands, and ground commands) and produce output (such as steering commands, control of mode and caution lamps, digital display updating, and digital telemetry transmission). To illustrate the diversity of requirements with which this computer must cope, a specific phase of the APOLLO mission is described in detail: the control of the spacecraft to accomplish a powered maneuver.

E-2129: KEYBOARD AND DISPLAY PROGRAM AND OPERATION
Alan I. Green, Robert J. Filene / June 1967

The Keyboard and Display program described in this report is included in the Flight 278 Programs (originally called SUNDISK and SUNDANCE).

The lists of nouns and extended verbs are incomplete. They differ somewhat in CSM and LM and are subject to frequent change. The most complete and up to date information for these is found in the latest revision of MIT/IL "Flight 278 Memo 17" or from the "Assembly and Operation Information" log section at the beginning of the program listing itself.

E-2097: A MULTIPROCESSING STRUCTURE
R.L. Alonso, A. L. Hopkins, Jr., H.A. Thaler / March 1967

Extrapolation of APOLLO experience to spacecraft computers of the next generation indicates a need for digital systems of greater computing and interface activity, and of greater reliability than previously realized.

An idealized collaborative multiprocessor structure in which a number of processing elements are tied together by means of a single multiplexed data bus is explored. At least one job assignment procedure is possible for which no one processor has to act as "master" and which can survive processor malfunctions or the deletion or addition of processors to the bus, thus accomplishing "graceful degradation" and "reconfiguration" of sorts. The single bus structure has implications for compilers and certain bandwidth relationships between processors, bus, and common memory. Rough estimates based on short extrapolations of circuit technology show that the structure is probably realistic.

E-2095: EVALUATION REPORT ON MULTILAYER CIRCUIT BOARD CONNECTORS USING THE SOFT METAL INDIUM FOR ELECTRICAL CONTACTS
Thomas A. Zulon / March 1967

Using weldable, multilayer circuit boards to connect the logic elements in G&N digital computers has resulted in problems associated with the interconnection of these circuit boards. A special connector structure has been designed to study these interconnection problems. The structure has a strongback that compresses an elastomer, that in turn distributes pressure to soft-metal-plated connector fingers and furnishes the system's elastic reserve. The soft-metal plating on the connector fingers is used to improve the electrical characteristics Gf contact interfaces. The soft metals have low crushing strengths and generate larger junction areas. They also anneal out work-hardening at or below room temperature, thereby giving these junction areas tolerance to substantial stress deformation without fracture. In addition, some combinations of junctions between the softer and harder metals exhibit diffusion-alloying effects at room temperature. The Iridium-Nickel reacting system was chosen for this experimental study, because faster acting systems might be detrimental to the repairability characteristic of these special connectors.

E-2092: SUMMARY REPORT BRAID MEMORY
W.H. Aldrich, R.L. Alonso, A.L. Hopkins / February 1967

This is a final report of work on the Braid Memory under Contract NAS9-4065, Part III, DSR Project 55-351. Prior work was reported in MIT/IL Report R-496, and work is continuing under DSR Project 55-29440. The Braid Memory is a high speed, high density, low power, random access computer memory of the read-only class. Its information is permanent and unalterable. It is distantly related to the AGC Rope Memory, but has a word-per-line organizationrather than the Rope's word-per-core organization. The Braid Memory is constructed with the aid of a Jacquard Loom, that "writes" information in the form of woven wire. A prototype package of 2^{19} bits capacity has been designed and fabricated.

E-2065: BLOCK II AGC SELF-CHECK AND SHOW-BANKSUM
Edwin D. Smally / December 1966

This report is in two main sections, The first section contains the operating procedures to be utilized by persons using the SELF-CHECK or SHOW-BANKSUM routines. It also has block diagram flow charts to explain the use of SELF-CHECK for diagnostic purposes. The procedures for SELF-CHECK are slightly different in BLOCK I and BLOCK II, but the procedures for SHOW-BANKSUM are the same.

The second section of this report explains SELF-CHECK and SHOW-BANKSUM. The SELF-CHECK explanation focuses on the computer internal selfcheck and the check of the DSKY electroluminescents. There is a separate description of each subroutine in SELF-CHECK and SHOW-BANKSUM. There is also a separate flow chart, located in the appendix, for each subroutine.

E-2052; AGC4 BASIC TRAINING MANUAL

A.L. Drake, B.I. Savage (both of Computer Consultants, Incorporated) / January 1967

This manual contains a concise description of the APOLLO G&N programing system for computer programers. The following questions are discussed:

1. What are the pertinent machine characteristics ?
2. What programing languages and conventions exist?
3. What systems subroutines may be used?
4. How does the programer communicate with the system subroutines?

This manual does not concern itself with the Mission programing system or with knowledge required by an engineer or mathematician to adequately program a mission phase.

This manual is divided into four sections. Section I discusses the AGC 4 and how to program it in Assembly Language. Section II describes the Interpreter and how to program in Interpretive Language. Section III describes the System Software subroutines and how to interact with them. Section IV contains an outline and suggestions for teaching Sections I-III. Each section has a table of contents.

E-2026; USER'S GUIDE TO THE AGC MONITOR (CORE ROPE SIMULATOR)

James D. Wood / September 1966

This report contains a description of the operation of the AGC Monitor, also known as the Core Rope Simulator, which was designed and built by MIT/IL.

The purpose of the AGC Monitor is: to provide a computer monitor to aid in computer troubleshooting and program debugging; and to provide core rope simulation (replacement of the AGC fixed memory with an erasable memory that can be loaded automatically by using a tape or manually by using a keyboard).

E-1989: KEYBOARD AND DISPLAY PROGRAM AND OPERATION

Alan I. Green, Robert J. Filene / July 1966

The Keyboard and Display Program described in this report is included in the Flight 204 Program (originallycalled SUNSPOT). The nouns listed are correct only for the 204B release.

E-1970: CASE HISTORY OF THE APOLLO GUIDANCE COMPUTER

Eldon C. Hall / June 1966

The characteristics of two complex digital computers designed for space application are presented. These computers make extensive use of microcircuits. The first computer has been in production for approximately eighteen months and has completed all qualification testing required to determine flight readiness. It will be used in the early APOLLO spacecraft missions. The second computer, which has increased computational and control capabilities, is designed to meet the increased requirements of later APOLLO missions. This computer has been in production for about six months. The construction techniques used in these computers are described.

A large volume of data has been collected on the computers during the design, production, and qualification program in preparation for the flight usage. These data confirm the reliability claims for microelectronic systems: with 221 x 10^6 part hoursinsystem operation, the microcircuit components are demonstrating a failure rate of $0.030/10^6$ part hours. Extensive engineering tests demonstrate that microelectronic computers can survive the environments of space missions.

E-1905: KEYBOARD AND DISPLAY PROGRAM AND OPERATION

Alan I. Green / January 1966

The Keyboard and Display Program described in this report is included in the Flight 202 Program (originally called CORONA). This report brings E-1574 up to date.

E-1880: A CASE HISTORY OF THE AGC INTEGRATED LOGIC CIRCUIT

Eldon C. Hall / December 1965

The integrated circuit used for the logic in the APOLLO Guidance Circuit is discussed. Achieving the required goals of low weight, volume, power, and extremely high reliability, necessitates the use of a single, simple integrated circuit for all logic funcilons. A brief description of the evolution of the computer design is given along with a general discussion of some of the engineering and design problems which arise with the use of a standardized semiconductor monolithic integrated circuit,

The flight qualification procedure is described. After the qualified suppliers list has been formed, each lot shipped from any qualified supplier is exposed to a screen and burn-in procedure followed by failure analysis of generated failures. The lot is accepted or rejected on the basis of the number of failures generated and the types of failure modes generated. The reliability history of the NOR Gate is given showing differences among vendors, showing differences among lots shipped from a single vendor, and updated field failure rates.

E-1877: SHOW-BANKSUM AND FINAL BLOCK I AGC SELF-CHECK

Edwin D. Smally / November 1965

This document is in two main sections. The first section contains the operating procedures to be utilized by persons using the SELF-CHECK or SHOW-BANKSUM routines. It also has three block diagram flow charts to explain the use of SELF-CHECK operating procedures for diagnostic purposes.

The second section explains the four main parts of SELF-CHECK and SHOW-BANKSUM. There is a separate description of each subroutine. There is also a separate flow chart for each subroutine located in the appendix.

This is the first time that a sum check of rope memory has been programed in the rope memory. Previously it was possible to sum the contents of each ropememory bank by utilizing the Field Verification Procedure and the Computer Test Set.

E-1808: ROPE MEMORY MODULE ASSEMBLY PROCESSING PROCEDURE
Arthur LaPointe / June 1965

This report establishes the procedures for the generation and process control of AGC programs. The report includes Program Control Boards established by APOLLO Project Memo No. 619, description of forms utilized for process control and release of computer programs, and logistic requirements for fixed memory core ropes.

E-1800: A SEQUENCE OF COMPUTER PROGRAMS USEFUL IN THE ANALYSIS OF FEEDBACK CONTROL SYSTEMS
Donald C. Fraser / July 1965

This report is a description of a sequence of digital computer programs which has been developed to mechanize the tools of servomechanisms analysis. These programs are quite general in nature and are very useful for the analysis and synthesis of high order control systems. Instructions are included describing the use of each program.

E-1786: VERIFICATION PLAN FOR AGC/LGC PROGRAMS
Thomas J. Lawton, Charles A. Muntz / May 1965

The techniques employed in verifying correct programing of the AGC / LGC are described. A system is discussed for automatic verification techniques in initial checkout and in checkout after modifications. The various tools for verification are described in some detail.

E- 1758: ORGANIZATION OF COMPUTATION AND CONTROL IN THE APOLLO GUIDANCE COMPUTER
Thomas J. Lawton, Charles A. Muntz / April 1965

The digital guidance computer is the central control element in the APOLLO G&N system. The difficulties in preparing computer programs for the complex APOLLO missions are aggravated by the need to keep the computer hardware to a minimum. An effective solution to these problems is achieved with a combination of appropriate programing techniques and computer design.

E-1750: COMPUTER PROGRAMS FOR OPTICAL SYSTEM ANALYSIS AND DESIGN

Donald C. Dilworth / February 1965

The two computer programs described are concerned wi-th APOLLO optical system design and analysis. The first, SPOTPLOT, calculates the intersection points at the image plane of a pencil of rays which have passed through an optical system and plots these in the form of a spot diagram. The other, SYNOPSYS (SYNthesis of OPtical SYStems), produces optimum lens designs with specified characteristics.

E-1699: A PROPOSED CIRCUIT STRUCTURE FOR COMPUTER LOGIC BASED ON SEMICONDUCTOR FLAT PACKS INTER-CONNECTED BY MULTILAYER CIRCUIT BOARDS

Theodore C. Taylor / December 1964

The current interest in using weldable, multilayer circuit boards as a means for interconnecting the logic section of an aerospace digital computer results in a study of some utility problems of circuit board structures. In dddition to the necessary characteristics of reliability, thermal and structural integrity, manufacturability, and great interconnectioncapacity; agoodcircuit structure shouldalso possess such features as small size per component, convenient shape, and convenient means of detailed electrical inspection and repair. A structure is proposed that appears to offer a favorable combination of these features.

E-1574: KEYBOARD AND DISPLAY SYSTEM PROGRAM FOR AGC: PROGRAM SUNRISE

Alan I. Green, Joseph J. Rocchio / August 1964

The DSKY System Program described in this report is included in Program SUNRISE, the first mission-oriented program for the AGC. This DSKY Program evolved from early versions for AGC 3 and from AGC 4 Programs, ECLIPSE and MOONGLOW.

E-1539: THERMAL PROPERTIES OF SOME STRUCTURAL MEMBERS FOR SPACE-BORNE COMPUTER ASSEMBLIES

Theodore C. Taylor, Thomas A. Zulon / March 1964

The design of missile and space-borne computer assemblies currently involves a chassis-and-module type of construction, using a considerable amount of structural metal. The amount of metal required in chassis or tray parts is a function of a number of considerations, one of which is the heat-conducting properties of these parts. This report is intended to acquaint computer assembly designers with the thermal conduction properties of some major chassis configurations which have been used or are applicable for use in computer assemblies. The report concludes with an idealized analyticalmodel to show several different heat conduction processes and the general effects of tray design on the thermal weight requirements in an assembly involving a typical configuration.

E-1215: THERMAL MODELS FOR HIGH DENSITY COMPUTER CIRCUIT STRUCTURES

Theodore C. Taylor / September 1962

The application of recent developments in microcircuit technology to the construction of computers for missiles and spacecraft is expected to present increased problems in the removal of component-generated heat. The general nature of the thermal problem due to circuit miniaturization is displayed by means of an idealized model of a circuit structure. The model assumes that heat transport withinthe structure is one-dimensional, and by thermal conduction only, with external cooling provided by a cold plate. Thermal design relationships are derived for heat generation that is uniformly distributed throughout a volume with a uniform thermal conductivity. A typical relationship expresses the required cold plate area in terms of the thermal resistance tolerable to the components, the apparent thermal conductivity of the structure, the external cooling effect, and the maximum length of the heat transfer path.

E-1179: ROPE CORE TESTER

Robert G. Scott / July 1962

The Rope Core Tester was designed to check the wiring and performance of 4096-core rope sense-lines. The device can be used to display complete words from a preselected core or to display the output of a selected sense line for the complete rope or for any portion of the complete rope.

The Rope Core Tester, consisting of a current pulse generator; a four-, a six-, and a twelve-stage counter; two logic sections; a current divider section; and various switching and display arrangements; was developed from standard breadboard components of the APOLLO computer. The Rope Core Tester is sufficiently versatile for its present needs.

E-1158: ERASABLE STORE MOD 3C
David Shansky / July 1962

An erasable ferrite memory consisting of 512 16-bit words has been assembled and is being tested in the Mod 3C computer. The memory has been designed for operation from $0^{\circ}-100^{\circ}C$. Characteristics of the cores selected and the circuits designed to fulfill this objective are discussed.

E-1126: AGC MOD 3C COMPUTER CIRCUITS-GENERAL
Albert L. Hopkins, Jr. / February 1962

The structure of the fundamental logic circuits for the AGC permits power consumption only during periods of activity, i.e., when input variables are in the ONE state. This circuit structure and the logical organization of the computer are generally shown and described. Voltage and power margins and component diversity are briefly indicated.

E-1125: PRELIMINARY RESOLVER ANGLE MEASUREMENT
Ramon L. Alonso / February 1962

A method is presented for measuring the IMU gimbal angles and entering the results into AGC. The interface between the IMU and the AGC is defined, and an AGC program is given that will perform the desired measurement. The measurement method is based on the time elapsed between successive positive-going zero crossings of the signals from resolvers tied to the gimbals.

E-1077: PRELIMINARY MUD 3C PROGRAMERS MANUAL

R. Alonso, J.H. Laning, Jr., H. Blair-Smith / November 1961

The material presented here provides prospective 3C programers with information for coding representative programs and making estimates of storage and speed.

E- 1074: ERASABLE FERRITE MEMORY FOR MOD 3C COMPUTER

David Shansky / October 1961

The operation of a ferrite core memory system, with a capacity of 512 16-bit words, is described with reference to the Mod 3C computer.

R-596: AS-205 VERIFICATION RESULTS: PROGRAM SUNDISK

Joseph F. Vittek / December 1967

This volume describes the testing done on the SUNDISK (AS-205) program and includes the detailed documentation of these tests. This fulfills the requirements set out in the statement of work for 1 December 1966-31 December 1967, Sections D-5.2.2 and D.5.2.3.

R-498: THE "BRAID" TRANSFORMER MEMORY

Ramon L. Alonso, W.H. Aldrich / September 1965

This report describes a read-only digital computer memorywhich utilizes a loom to "braid" information into a wire harness. The memory is called a braid memory and is a variation on the type of transformer memory first described by T.L. Dimond. Such memories are useful when information permanence is desired. Memory braids described in this report were made at the rate of 2^{15} bits/hour, including connections. The use of a loom makes braid manufacture both a fast and an economical process.

A number of trade-offs among speed, capacity, and component count exist; hence, braid memories with many different characteristics can be made. The memory described in this report has a capacity of 16,384 words of 16 bits each. Its semiconductor count is 194 transistors, 244 integrated logic gates, and 1024 diodes. It also employs 256 ferrite transformer cores. The memory has a cycle time of 2 microseconds and consumes less than 3.5 watts of power,

R-489: USERS GUIDE TO THE BLOCK II AGC/LGC INTERPRETER
Charles A. Muntz / April 1965

A description is given of the AGC/LGC algebraic interpreter, a language in which APOLLO Mission computer programs may be conveniently prepared.

R-467: THE COMPLEAT SUNRISE:: BEING A DESCRIPTION OF PROGRAM SUNRISE (SUNRISE 33-NASA DWG #1021102)
R.H. Battin, et al. / September 1964

SUNRISE (SUNRISE 33-NASA DWG #1021102) is a computer program prepared for the AGC and intended for operation with, at least, APOLLO G&N systems 4 through 7. The objectives of SUNRISE arc:

1. Provide the groundwork in terms of input/output, control, **and** utility programs fcr ail subsequent XGC programs.
2. Provide the necessary tools for proving guidance systems 4 through 7.
3. Demonstrate the efficacy of an all-digital simulation as a checkout tool,
4. Demonstrate rapid turnaround time for manufacture and checkout of core ropes.

R-416: THE APOLLO GUIDANCE COMPUTER
Ramon L. Alonso, Albert L. Hopkins / August 1963

The general logical structure of the onboard AGC is presented, and the development of fixed and erasable memory is described. Particular attention is given to the input and output methods.

R-410: GENERAL DESIGN CHARACTERISTICS OF THE APOLLO GUIDANCE COMPUTER
Eldon C. Hall / May 1963

This report describes the AGC's general designcharacteristics, flexibility, reliability, and inflight repair capabilities. Since the CM computer and LEM computer differ basically only in form factor (and therefore weight and volume), the characteristics detailed here apply to both computers.

R-393: LOGICAL DESCRIPTION IN THE APOLLO GUIDANCE COMPUTER (AGC 4)
A.L. Hopkins, R.L. Alonso, H. Blair-Smith / March 1963

This report describes the logical structure of the APOLLO G&N computer. A previous computer, AGC 3, designed for the APOLLO mission, was predominately composed of core-transistor logic. The computer design described here employs miniature integrated NOR logic, whose use will result in the next APOLLO computer (AGC 4) being just over half the size of AGC 3. Fine detail and internal consistency have been underemphasized for the sake of promptness, so that this report could be written within a few weeks of design inception.

R-358: A DIGITAL CONTROL COMPUTER: DEVELOPMENTAL MODEL 1B
R.L. Alonso, A.I. Green, H.E. Maurer, R.E. Oleksiak / April 1962

This report describes in detail a pilot model of a parallel, programable digital control computer. Magnetic cores are used as storage devices and extensively in the logic in order to thus reduce power consumption. Transistors are the active elements, The computer has capabilities for automatic incrementing of counters and automatic program interruption (mode changing). These capabilities are used to advantage in the control of a stepping-motor.

Section 3: DAP

E-2377: LM DIGITAL, AUTOPILOT SIMULATION RESULTS USING PROGRAM SUNDANCE
William S. Widnall / January 1969

The DAP delivered in Program SUNDXNCE has been subjected to exhaustive testing by means of digital and hybrid simulation. The SUNDANCE DAP differs in many ways from the DAP previously delivered in Program SUNBURST. The most noticeable differences are:

1. The SUNBURST DAP was designed only for unmanned flight, but the SUNDANCE DAP contains the basic manual mode capabilities.
2. A capability of using the LM to control the CSM-docked configuration has been added in SUNDANCE.
3. In spite of added capabilities,, the SUNDANCE AUTOPILOT design is simplified, and the coding is significantly shorter than the SUNBURST coding.

E-2353: DIGITAL AUTOPILOT VERIFICATION RESULTS: PROGRAM COLOSSUS I (REV 237)
MIT/IL / November 1968

Verification testing was performed on the digital autopilots for the APOLLO CSM in order to demonstrate suitable operation for an earth-orbital and lunar mission (Program COLOSSUS 237). The COLOSSUS autopilots—essentially the same as earlier-tested SUNDISK autopilots (R-596, Vol. IV: "Digital Autopilot Test Results," AS-205 Verification Results, Program Sundisk, February 1968)-were verified in the following way:

1. All design characteristics either new or modified since SUNDISK were fully tested, using all- digital and real-time hybrid simulations, to verify the coding and to measure performance.
2. Additional tests were performed in areas of performance not tested during the SUNDISK testing.
3. COLOSSUS Level-III testing was examined to verify the DAP software.

Subject to further study regarding bending limitations in the CSM-LM configuration (paragraphs 1.3E and 2.2E), the results reported show that the DAP can perform as described in Report R-577, Guidance System Operations Plan for Manned CM Earth Orbital and Lunar Missions Using Program COLOSSUS, Section 3, "Digital Autopilots," Rev. 1 (June 1968).

E-1964: APOLLO COMMAND AND SERVICE MODULE REACTION CONTROL BY THE DIGITAL AUTOPILOT
Robert Crisp, Donald W. Keene / April 1966

An AGC program developed to control the reaction jets of the APOLLO CSM is described, Design philosophy is discussed. The main design restraints are the existing hardware design and maneuver requirements evolved at the implementation meetings for APOLLO Block II CSM G&N systems, In general, the translation and rotation manual controls are implemented in the same way as the Block I SCS except that simultaneous translation and rotation accelerations are possible. Automatic maneuver and attitude hold are instrumented to conserve reaction control propellants.

The maneuver instrumentation was designed and evaluated using a flexible vehicle model with no propellant motion. Current slosh models look on propellant motion as a source of disturbing torques., However, analog simulations have been made with new slosh models in which the propellant motion is coupled with vehicle response.

A theoretical study of propellant utilization in generalized automatic maneuvers is compared with figures from three-degree-of-freedom digital simulations . The theoretical figures give a good estimate of the simulated propellant utilization. Conclusions are made regarding the current design, future work, and simulation plans.

E-1836: STUDIES OF SPACECRAFT ATTITUDE CHANGE MODES AND RESULTING ATTITUDE FUEL CONSUMPTION
Kenneth Nordtvedt / August 1965

This study examines several different types of vehicle attitude change modes and presents the logic used to calculate these maneuvers, The attitude control fuel consumption for these maneuvers is studied with its miniinizdtion in mind.

E-1832: ATTITUDE MANEUVER OPTIMIZATION TO CONSERVE REACTION CONTROL PROPELLANTS

Robert Crisp, Donald W. Keene / August 1965

A comparison is made among three methods of spacecraft maneuvering on the basis of RCS propellant use. The first method employs a single direct rotation; the second and third methods employ a sequence of rotations about principal axes of the spacecraft. As a starting point for this comparison, it is assumed that all maneuvers are equally **likely.** In geometrical terms, this implies that the single-equivalent rotation characterizing each maneuver has an axis-of-rotation vector direction uniformly distributed in space and a magnitude of rotation uniformly distributed in angle. in addition, the total time of maneuver for each method is equated to that for a single rotation of constant rate.

The optimum method depends on the specific maneuver performed. For simplicity, however, only one system can be instrumented. The method that uses the least fuel for the majority of cases is the single direct rotationmaneuvering at aconstant rate. Under the conditions considered here, this is the preferred method to be instrumented in the spacecraft autopilot.

R-533: A BLOCK II TVC DIGITAL AUTOPILOT COMPENSATION FOR CSM SPACECRAFT

Tan C. Lu (NAA/S&ID) / January 1966

The design of a Block II TVC DAP compensation resulted in a configuration of: a single pole, a single zero, and a unity gain factor. The compensation satisfies both the TVC performance requirements and the design constraints. Both the rootlocus-analysis and the frequency-analysis techniques are described: they yield the same design independently. The compensated system closed-loop dominant poles, the open-loop gain and phase margins, and the crossover frequency are given. AGC implementation of the compensation requires two fixed and two erasable memories, two multiplications, and two additions.

R-503: A BLOCK II DIGITAL, LEAD-LAG COMPENSATION FOR THE PITCH-YAW AUTOPILOT OF THE COMMAND AND SERVICE MODULE
Gilbert S. Stubbs / October 1965

A Block II digital, lead-lag compensation is described for the Pitch-Yaw Autopilot of the CSM. This compensation is designed to minimize the effects of bending-mode resonances on the system response. The worst-case problem of controlling the CSM with the LEM attached is taken as a basis for compensation design and analysis. Both transient and frequency response characteristics of the lead-lag compensated autopilot are analyzed.

R-499: DESIGN PRINCIPLES OF THE LUNAR EXCURSION MODULE DIGITAL AUTOPILOT
George W. Cherry / June 1965

The design principles of the LEM DAP are presented. The application of these principles minimizes the number of RCS jet firings and thereby maximizes RCS reliability while minimizing RCS propellant expenditure, The IMU gimbal angle measurements are processed by an optimum-recursive linear filter to yield vehicle angular orientation, vehicle angular rates, and the vehicle-disturbing angular acceleration impressed by the main propulsion system. The filter output locates the vehicle on the angle-angle-rate phase-plane and delineates the parabolic phase trajectory traversed by the vehicle.

An efficient limit cycle about the commanded angle and angle rate is achieved and maintained by firing RCS jets. If the vehicle is in a coast region, zero RCS torque is commanded. If the vehicle is not in a coast region, the appropriate jets are turned on to move the vehicle from the current phase-plane point to the desired phase-plane point or contour. The jet burning time required to solve this two-point boundary-value problem is computed explicitly,

E-2405: ENTRY FLIGHT DATA FROM APOLLO 8
Raymond Morth / April 1969

In APOLLO 8, Borman, Lovell, and Anders made the first manned entry at speeds greater than orbital velocity, Their entry was accompanied by an excellent set of flight data. This report presents that data showing trajectory, aerodynamic, control, and display parameters. A reconstructed trajectory is also presented.

E-2404: ENTRY FLIGHT DATA FROM APOLLO 7
Raymond Morth / April 1969

APOLLO 7, the flight of Schirra, Eisele, and Cunnmgham, provided an excellent set of entry flight data. This report presents that data showing trajectory, aerodynamic, control, and display par&meters. A reconstructed trajectory is also presented.

E-2214: A STUDY FOR A CALIBRATION DETECTOR FOR ULTRAVIOLET MEASUREMENT
Harold H. Sewdrd, Ian G. McWilliams / January 1968

A quantum detector exhibiting a nearly constant quantum efficiency was developed for themeasurement of ultraviolet radiation from <900 to 4500 Angstroms. The detector exhibits excellent stability of quantum sensitivity through significant variations in:

	Measured*	Projected Estimate
Wavelength	<2000 to 4500A°	<900 to 4500A°
Intensity	500: 1	10^{11}:1
Angle of Incidence on cell	90° to <5°	
Position of Incidence on cell	<2% over sensitive area	-
Temperature of cell	-40 to +140°F	<-40 to +140°F
Age of cell	4 hours to 2 weeks	indefinitely long life

*5% Confidence level in absolute measurements.

E-2119: CRITIQUE OF THE APOLLO GUIDANCE SYSTEM DESIGN
MIT/IL / May 1967

The following discussions represent a Critique of the G&N systems designed by MIT/IL for use in the APOLLO CM and LM. The Critique was prepared in response to Paragraph 2.10 of the Statement of Work for Contract NAS 9-6823, Development and Laboratory Test of Advanced Manned Mission Guidance and Control Techniques. The Critique is presented in sections as follows:

Section 1.0 Introduction
Section 2.0 System (including Crew Interface and Reliability)
Section 3.0 Inertial Subsystem
Section 4.0 Computer Subsystem
Section 5.0 Optics Subsystem

E-2027: PLASTICS: WHAT TO USE
Samuel C. Smith / September 1966

This report presents information concerning dependable potting resins: rigid and flexible, coating materials, adhesives, cleaning agents, solvents, mold releases, primers, plastic stocks, transfer marking, antirust, and etching solution designed for specific jobs. A thorough investigation of each product listed in this report has been made by a study of the relevant literature, some contacts with the manufacturers, and laboratory tests. The products listed were selected after investigating and comparing three or four products of similar quality. In a number of cases, these products have been in continuous use in our program with a high degree of reliability. Except in cases where it was necessary to deviate from the manufacturer's recommendations, "how to use" directions have been omitted; instead, the manufacturer's literature has been referenced,

E-1956: AN AUTOMATED DOCUMENTATION TECHNIQUE FOR INTEGRATING APOLLO CREW' PROCEDURES AND COMPUTER LOGIC
J.C. Dunbar, R.A. Larson (AC Electronics Resident Staff), P.T. Augart / May 1966

This document relates onboard computer activity, ground computer activity, and sequencing of airborne systems, to human operator activity during the APOLLO Mission G&N operations. The document serves as a tool for computer programers, provides a testing device for evaluating mission operations in the various simulators, and also serves as a training device. The crew checklist used to define G&N and related airborne system operating procedures is in a form that can be used directly for flight operations. The documentation technique evolved allows rapid information retrieval and updating. Various forms of the data can be extracted for use in mission planning, computer programing, operations analysis, or crew training.

E-1855: PULSED LIGHT SOURCE SIMULATION OF TRANSIENT NUCLEAR RADIATION EFFECTS IN INTEGRATED CIRCUITS
Harold E. Maurer / October 1965

The feasibility of flashtube light sources to duplicate the transient effects in integrated circuits produced by high energy electrons is demonstrated experimentally. Two flashtube pulsed-light sources were designed, constructed, and used to simulate the transient effect on exposed integrated circuits of sbombardment by 20 MeV electrons. The light sources differed in the spectrum and shape cf the light pulse produced, with the narrower of the pulses being less than one microsecond in width, The integrated circuits contained Nor gates and sense amplifiers: the Nor gates employed reverse-biased p-n junction isolation of components, while the sense amplifiers employed both silicon dioxide and reverse-biased p-n Junction isolation of components. The simulators produced good but not identical results, especially near the thresholds of transient damage, but increasing the electron flux rate caused an eventual deterioration cf the simulators' performance.

E-1828: G&N SYSTEM DATA FOR MISSION AS-202 DEVELOPMENT ENGINEERING INSPECTION
MIT/IL / August 1965

The data in this report have been compiled to support the Spacecraft 011 (G&N 017) AS-202 Development Engineering Inspection. It is intended to describe G&h system functions and performance. G&N functions are defined in the nominal timeline. Performance is defined by the results of an all-digital nominal-mission simulation and a statistical error analysis. Also included are a reliability assessment and an abbreviated failure effects analysis. Additional data may be found in R-477 Rev. 1 (July 19653.

E-1584: DIGITAL TO ANALOG CONVERSION OF BASE MINUS TWO NUMBERS AND APPLICATION TO THRESHOLD LOGIC
Robert J. Filene / June 1964

This paper describes a network for performing digital to analog conversion of numbers expressed with the base minus two (-2). Negative base number systems express both positive and negative numbers without a distinct sign bit; the sign is given by the position of the most significant non-zero bit. Exploiting this property, the network is shown to be useful for threshold logic synthesis; if both positive and negative "weighting" of logical inputs, followed by summing, are required. An example is given.

E-1557: AVERAGE POW-ER PROFILES FOR APOLLO GUIDANCE AND NAVIGATION IN THE COMMAND MODULE AND LUNAR EXCURSION MODULE
William Nadler / April 1964

The accompanying tables summarize the average power and energy as a function of time and APOLLO mission phase for the Block II CM and LEM G&N equipment as defined in the G&N system Status Report. The values reported are center value (average) estimates. They are the latest expected values and should not be taken as "not to exceed" extremes.

E-1548: BRAZING OF BERYLLIUM
Ranulf W. Gras / March 1964

This progress report summarizes the work conducted by the MIT/IL Welding Laboratory to develop techniques for the brazing of beryllium during the period from 1 August 1963 to 1 February 1964. The initial stages of the program were primarily concerned with wetting tests using numerous brazing materials. Data obtained from these tests indicated the most promising processing techniques and brazematerialcombinations to be used for joints. Although the original program was directed towards the brazing of beryllium to itself, subsequent emphasis was later placed on the brazing of berylluim to stainless steel (type 410) and to plain carbon steel. During this period, electron beam equipment was adapted as a useful brazing tool.

E-1524: EVALUATION OF URETHANE FOAM FOR POTTING
Samuel C. Smith / February 1964

Effort has been expended to eliminate some deficiencies in urethane foam potting and to improve its reliability. Causes of such problems as blistering, crazing, coarse cell structures, and improper expansion have been found and corrected.

E-1425: A COMPARISON OF THE READOUT RESOLUTIONS OF THE PROPORTIONAL ELASTANCE/CONSTANT ELASTANCE TORQUE-TO BALANCE SYSTEMS
Charles C. Perez / September 1963

This report establishes read-out voltage signal-to-noise ratios for both the constant elastance torque-to-balance system and the proportional elastance torque-to-balance system. Resolution of low level torques is limited by servo electronics noise in the constant elastance system, but it is limited by gyroscope "noise torques" in the proportional elastance system. These "noise torques" require investigation to further improve gyroscope performance. The advantages of the proportional elastance torque-to-balance system are established.

E-1398: COMMENTS ON THE LUNAR LANDING MISSION DESIGN PLAN OF 15 APRIL 1963
John M. Dahlen, John B. Suomala / August I963

This report comments and recommends changes to the Project APOLLO Lunar Landing Mission Design Plan, MSC, 15 April 1963.

E-1313: THERMAL GROUNDING ANALYSIS FOR CIRCUIT STRUCTURES

Theodore C. Taylor / April 1963

The general use of thermal grounding is considered for circuit structures whose adequate cooling requires specific provision for heat removal. Because both thermal grounding circuits and electrical grounding circuits may have similar continuity in many circuit structures, the dual function capability of thermal grounding circuits is analyzed. For semiconductor logic circuits, the dual function conductor must be sized from thermal requirements rather than electrical requirements, if low frequency electrical grounding requirements are considered.

A formula is derived for the amount of thermal grounding conductor needed for an optimum thermal grounding system (defined as the system of least conductor material). Idealized models are then developed for a geometrically regular arrangement of heat generators for optimum grounding, series grounding with uniform conductors, and parallel grounding with uniform conductors.

E-1.287: BACKUP THRUST VECTOR CONTROL

John M. Dahlen, James H. Long / February 1963

This report contains a description of a backup method of attitude control for use during SM thrusting. An analog computer simulation used to demonstrateits feasibility is also described. Results are given of human factor experiments using the analog computer.

E-1233; TECHNIQUES OF CIRCUIT FABRICATION WITH STATE-OF-THE-ART, WELDABLE MULTILAYER BOARDS

Theodore C. Taylor / October 1962

Multilayer boards (with state-of-the-art welding processes) decrease circuit-structure volume within solid circuit component bodies. The use of weldable board techniques appears to offer significant advantages over present methods of circuit construction, but these techniques will require additional laboratory study to perfect their use for practical application.

E-1203: CALIBRATION TECHNIQUES FOR PRECISION ROTARY COMPONENTS
George A. Davidson, Harold H. Seward / August 1962

Automatic measurement of gear system errors depends on an auto-collimating device that is held at null, keeping the output shafts of two similar gear systems nulled to a predictable degree of accuracy, By phase shifting the two gear systems, harmonic components in each gear system can be measured to a high degree of accuracy. A principal feature of this measurement technique is that a precision standard is not required.

E-1172: THE VISIBILITY OF STARS
(Calculated from the Tiffany Data)
Arthur C. Hardy (Consultant) / June 1962

From the data accumulated on star visibility by the National Defense Research Committee during World War II at the Louis Comfort Tiffany Foundation, aspects relevant to the APOLLO project have been selected and presented in this report. The experiments were performed in the laboratory and in actual field tests frorn a destroyer escort.

E-1167: MIT APOLLO DRAWING STANDARD
Rev.1 MIT/IL / June 1964

The purpose of the MIT/IL APOLLO DRAWING STANDARD is to establish the requirements, format, and procedures governing the preparation, authentication, and approval of drawings and associated documentation required for the production of items, equipment, and assemblies, for use in Project APOLLO.

The material contained in this publication has been prepared from pertinent sections of MIL-D-70327, MIL-STD-2, and NAVORD OSTD 599, 2nd Revision.

E-1134: PHOTOMETRIC UNITS IN THE MKS SYSTEM
Arthur C. Hardy (Consultant) / March 1962

This document lists and discusses briefly photometric units and symbols adopted by the Optical Society of America, The OSA Committee report was published as The Science of Color, Thomas Y. Crowell Company, 1953. (Both radiometric and photometric terms and symbols are treated in Chapter 7 of this volume.)

R-575: FINAL REPORT ON ADDITIONAL SERVICES
MSC / April 1967

This report reviews a number of analytical, research, and development studies completed as part of MIT/IL's overall effort in the development of the APOLLO Primary Guidance, Navigation, and Control System. Of special interest are studies of gyroscope testing, development, and refinement; CDU and DSKY development; computer design and program development; optical system design; and navigation theory. Extensive charts and diagrams are included.

R-539: A MANUALLY RETARGETED AUTOMATIC LANDING SYSTEM
Rev. 1 FOR LM
Allan R. Klumpp / August 1967

During the final few rninutes and several miles of the APOLLO lunar landing, the LM Commander can manually retarget the automatic landing system . At the beginning of this approach phase, the automatic landing system is targeted to a pre-selected landing site. The computer identifies the landing site to which it is currently targeted, by orienting the LM about the thrust axis so that the site appears to coincide with a window-fixed reticle, and by displaying a number indicating the point on this reticle through which to look. The Commander identifies the current site and also selects the desired landing site as a previously landed spacecraft, a natural landmark, or an appropriate unmarked site. Estimating the angular error between the current and desired landing sites, the Commander steers to null this error by repetitively manipulating a hand controller. These signals go only to the automatic system which counts them, moves the current site through a proportional angular increment (as seen by the Commander), and computes the new target parameters and the number to display indicating the new current site. By steering until the point on the reticle currently indicated by the computer coincides with the desired site, the LM Commander can converge on any landing site he may select.

R-500: SPACE NAVIGATION GUIDANCE AND CONTROL: AGARD PAPERS
C. Stark Draper, et al. / June 1965

The material in this book was assembled to support a series of lectures to be given by the authors under the sponsorship of the Advisory Group for Aerospace Research and Development (AGARD), a NATO agency,

Space Vehicle Control Systems are the subject of discussion, with particular application to the present Manned Lunar Landing Program. The mission requirements and man-machine interaction are first described. The specific hardware performance requirements and underlying reasons for choice are next explained. Discussed last are theoretical background, thesystem analysis, and the derivation of the control functions to integrate the hardware into a precision guidance, navigation, and control system. The book is organized into seven sections following the pattern of the lectures.

Section I provides historical background to the fundamental problems of guidance and navigation. The basic physical phenomenon and associated instrument techniques are discussed.

Section II continues with background information going more specifically into the problems and approach of the guidance, navigation, and control of the APOLLO manned lunar landing mission.

Section III concerns in detail the analytic foundation for performing onboard calculations for navigation and guidance.

Section IV covers in detail the mechanization of the inertial sensor equipment of the APOLLO guidance and control system.

Section V discusses the optical navigation sensors and measurement techniques.

Section VI provides background and specific techniques in the mechanization of onboard digital computers.

Section VII discusses the specific problems and solutions of vehicle attitude control under conditions of both rocket-powered flight and free-fall coast conditions.

R-462: DISSIMILAR METALS: COMPILATION OF CONTACTING INTERFACES BETWEEN G&N PARTS IN AIRBORNE EQUIPMENT (BLOCK I)
E.T. Ogrodnick / September 1964

This document describes existingmaterials interfaces within G&N airborne Block I equipment obtained from a detailed review of MIT/IL assembly drawing releases. Except for mounting hardware, the contacting interfaces within vendor supplied parts (such as motors, transformers, etc.) are excluded from this report. This document provides sufficient information and reference on the compatibility of parts interfaces and finishes usage to qualitatively assess adequacy of equiprnent protection against physical degradation that could result in the deterioration of electrical, thermal, mechanical, structural, or optical properties.

Information on fastening/joining of assemblies has been included.

R -447: NAVIGATION FOR THE APOLLO PROGRAM
John M. Dahlen, James L. Nevins / May 1964

This paper explains the basic navigation concepts and techniques used in designing the APOLLO G&N system. This system has the capability to control the spacecraft path throughout the basic lunar landing mission—fifteen distinct guidance and navigation phases. Also required is the capability to guide aborts from all phases prior to trans-earth injection. In order to perform these functions three distinct tasks must be accomplished:

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

R-446: PRIMARY G&C SYSTEM LUNAR ORBIT OPERATIONS:
VOLUMES I & II
Normdn E. Sears / April 1964

This report summarizes the primary G&N system operation and performance during the lunar orbit phases of the APOLLO lunar landing mission. The lunar orbit phases include orbit navigation, descent, landing, surface operations, launch and ascent, rendezvous, and LEM aborts. These phases are primarily concerned with the LEM primary G&N operation; but CSM operations of orbit navigation, LEM back-up guidance capability, and LEM retrieval are included. Each lunar orbit phase is described with respect to:

1) primary G&N system objectives and operating modes,
2) current guidance equations,
3) typical trajectories,
4) and primary G&N performance and error analysis.

A general description and performance specification is included for the basic units of the primary G&N system.

R-445 : TECHNICAL DEVELOPMENT STATUS OF APOLLO GUIDANCE AND NAVIGATION
Norman E. Sears / April 1964

The primary guidance and navigation system of the APOLLO CSM and LEM are described and reviewed. The guidance and navigation system operating procedure is then described for the lunar orbit phases of the APOLLO mission. These mission phases are primarily concerned with the LEM orbital descent, landing, ascent, and rendezvous. The lunar orbit navigation phase in the CM is included, since it establishes the initial inputs for the LEM guidance system. In the description of the lunar orbit phases of operation, the general guidance concepts are briefly described, and the astronauts' use of the system to achieve the required objectives of the individual phases is outlined.

R-415: APOLLO REENTRY GUIDANCE
D.J. Lickly, R. Morth, B.S. Crawford / July 1963

The problem of designing an automatic:, self-contained inertial guidance system for the reentry phase of the APOLLO mission is discussed in detail. The objectives, design criteria, and relationship between overall mission requirements and reentry range requirements are discussed. A system that achieves the desired performance is described in detail.

R-411: APOLLO GUIDANCE AND NAVIGATION: A PROBLEM IN MAN AND MACHINE INTEGRATION
David G. Hoag / April 1963

This report shows the design of the APOLLO guidance and navigation equipment and the displays, controls, and operations used by the astronauts in performing a difficult and necessarily accurate task. The compromise between a completely automatic system and one configured for extreme dependence on man is met with one solution having good features of both approaches. The navigator has complete choice and control of the system operation, using his senses and judgement where they are superior, and depending upon mechanisms where man is unable or too stressed to be used. The sensors, the computer, the displays, and the controls are described in enough detail to illustrate astronaut operation of the APOLLO G&N system.

R-409: PRINCIPLES OF CLEANLINESS IN THE ASSEMBLY OF INERTIAL INSTRUMENTS
Robert J. Schiesser / May 1963

In the development of high-precision gyroscopes and accelerometers, the Instrumentation Laboratory has had to provide Clean Rooms and develop assembly techniques to assure the performance level of these instruments. Attention is directed more to obtaining a consistent level of instrument cleanliness than to the Clean Room itself. Techniques insure getting “Class IV” or better work in a “Class II” or poorer Clean Room.

R-391: THE EFFECT OF RETRO-ROCKET EXHAUST ON VISIBILITY DURING LUNAR TOUCHDOWN
G. Dudley Shepard / December i962

In order to accomplish a successful soft landing on the moon, it is necessary to evaluate visually the small scale lunar terrain as a landing site. This report studies (a) the visual obscuration of the lunar landscape by dust generated by the erosive effect of the landing craft exhaust gases and (b) the washing out of the inherent visual contrast of the lunar landscape by exhaust gas luminescence.

R-373: GUIDANCE AND NAVIGATION SYSTEM FOR LUNAR EXCURSION MODULE
John M. Dahlen, et al. / July 1962

This report describes the LEM guidance equipment and concepts. They are subject to change as the LEM configuration and mission are more clearly defined,

R-367: MULTIRANGE TORQUE-MEASURING DEVICES
P.H. Gilinson, Jr., C.R. Dauwalter, J.A. Scoppettuolo / July 1962

A precision torque-measuring system for torques ranging from 0.010 to 1000 dyne-centimeters is described. The unknown torque is measured by a feedback system that automatically develops a balancing torque by means of anelectromagnetic torque generator. The measure of the unknown torque is the inphase product of the primary excitation and the secondary feedback current into the torque generator necessary to develop this balancing torque. A dynamometer-type wattmeter or Hall Effect current-product device is used to measure the inphase current product. The wide range of torques can be measured to a precision of about 0.1 percent of full scale, by the proper use of a decade attenuator in the torque-generator feedback circuit and by the adaptive control of the system elastance.

R-235: A RECOVERABLE INTERPLANETARY SPACE PROBE
MIT/IL / July 1959

A recoverable interplanetary space probe is investigated and its feasibility established. A preliminary design is presented of a vehicle whosemission is to return safely to earth with a useful payload-for example, a high resolution photograph of a neighboring planet. Navigational techniques and interplanetary orbits are studied in detail to determine basic performance and accuracy attainable for a variety of interplanetary missions. These numerical results, assuming reasonable navigational instrument errors, indicate that only moderate propulsion requirements for orbital velocity corrections are necessary. The vehicle weight is well within the capability of soon- available rocket vehicles for initial boost into a planetary transfer orbit. The vehicle contains an automatic navigation and attitude control system using a digital computer and electromechanical-optical accessories, a micro-rocket system for making navigational corrections, a long-range communication system for transmitting useful intelligence, and a reentry system with associated radio and light beacon aids for the physical recovery of the payload. The overall vehicle weight is approximately 340 pounds.

T-385: HUMAN PERFORMANCE DURING A SIMULATED APOLLO MID-COURSE NAVIGATION SIGHTING
Charles M. Duke, Jr., Michael S. Jones / June 1964

The effects of certain variables on the performance of man doing a precise superposition task are examined. This simulates the task that the Project APOLLO navigator will be required to perform during the mid-course (translunar and transearth) phases of the proposed lunar excursion. For this investigation, the APOLLO Sextant Simulator located at MIT/IL was used. The variables were rate of spacecraft motion, magnification of sextant telescope, orientation of landmark, and star-landmark contrast ratio. In order to determine the effect of each variable individually, only one was varied at a time.

Three subjects were used. Each performed the superposition task by using a set of hand controllers until the star was on top of the landmark, as seen through the sextant telescope. At -this point the subject pressed a “MARK” button, which recorded the error made in seconds of arc. For each given set of conditions, the subject performed the task 25 to 30 times. F’or each such series, the mean error was computed (absolute mean distance from perfect superposition). Statistical tests were then applied to these means to check for significant changes in error due to changing one of the variables.

E-2364: INERTIAL COMPONENTS RELIABILITY AND POPULATION STATISTICS: REPORT IV
Julius Feldman, Stefan Helfant / January 1969

The APOLLO gyroscope failure and operational statistics are presented versus wheel hours from assembly and calendar time from acceptance. The inventory of inertial components in active systems at field sites, including gyroscope removals, is also presented. Gyroscope wear-out characteristics and recommended test procedure changes are discussed.

E-2333: INERTIAL COMPONENTS HE LIABILITY AND POPULATION STATISTICS: REPORT III
Martin Landey/ December 1968

A detailed analysis of scale factor and bias data from system-level tests of the 16 PIPA MOD D is made. For this effort, a relatively unperturbed environment is constructed by editing out data reflecting shifts clearly traceable to system operation anomalies. Analysis is performed using severai computer programs written for this purpose. It is concluded that the system level characteristics are similar to component-level operational characteristics of the instrument.

Particular observations include an estimate of scale factor drift with PIPA aging, normality of test-to-test scale factor shifts, and non-normality of test-to-test bias shifts.

E-2308: A NON-ORTHOGONAL MULTI-SENSOR STRAPDOWN INERTIAL REFERENCE UNIT
Jerold P. Gilmore / August 1968

The multi-sensor non-orthogonal strapdown inertial reference unit described provides redundant capabilities and optimal performance. It utilizes six single-degree-of-freedom gyroscopes and six linear accelerometers, with their input axes arrayed in a unique symmetrical pattern that corresponds to the array of normals to the faces of a regular dodecahedron. Complete adaptive data-processing and statistical weighting, with self-contained failure isolation and detection, is defined to yield "best" estimates of vehicle rate and acceleration for any instrument failure combination. The system has been formulated in a strapdown configuration: instruments are operated in a pulse-torque-to-balance-control mode and configured as prealigned modular units. Representative package configurations are illustrated. The analysis shows a significant reliability improvement over orthogonal redundant configurations that employ nine gyros dnd nine accelerometers.

E-2279: PREALIGNMENT OF THE 16 PIPA MOD D
George J. Bukow / February 1968

Techniques are presented to align the input reference axis of the 16 PIPA MOD D to the axes that define the PIPA mounting surface. An error analysis of these techniques is also presented.

E-2274: INERTIAL COMPONENTS RELIABILITY AND POPULATION STATISTICS: REPORT II
Julius Feldman, Roscoe Cooper / June 1968

APOLLO PIPA and APOLLO gyroscope reliability statistics are presented involving the distribution of failed and non-failed units and their failure rateversus time from acceptance. These distribution and bearing failure rates versus wheel hours and the percentage of gyroscope failures grouped according to manufacturing data are presented for both the APOLLO I and APOLLO II gyroscopes.

E-2252: INERTIAL COMPONENTS RELIABILITY AND POPULATION STATISTICS: REPORT I
Julius Feldman, Roscoe Cooper / March 1968

APOLLO PIPA and APOLLO gyroscope reliability statistics are presented. The APOLLO gyroscope drift terms' standard deviation distribution at component acceptance testing, system acceptance testing, and gimbal systems testing, is presented as bar graphs for both APOLLO I and APOLLO II gyroscopes. The distribution of failed and non-failed gyroscopes and their failure rates versus wheel hours are presented for both APOLLO I and APOLLO II gyroscopes. PIPA failures are shown grouped according to failure mode; and all failed units are tabulated to show cause, symptoms, and unit locations.

E-2141: APOLLO GUIDANCE, NAVIGATION, AND CONTROL SYSTEM: GYROSCOPE RELIABILITY
John E. Miller, Julius Feldman / June 1967

The reliability of the APOLLO PGNCS depends to a large measure upon . the reliability of the inertial reference gyroscope. A proven design, the Polaris Mod II Inertial Reference Integrating Gyroscope (IRIG), is used as the starting point. Although the signal and torque microsyns were redesigned to meet the frequency and computer interface requirements, the critical reliability components-the float and wheel package-are unchanged from a design already in production.

This paper presents the method used to increase the operational reliability of the APOLLO gyroscope by a factor of twenty , despite the fact that the wheel assemblyreliability had exploited. the state of the art. The results of the reliability program lead to a method of forecasting the occurrence of the gyroscope's primary failure mode -the failure of the wheel bearing assembly, This failure predictionmethod, the Delta 25 test, is heuristically derived . Application of this test to a sample of gyroscopes confirms reasonable accuracy in failure prediction, leaving sufficient good- performance operating time to complete a lunar mission.

A second, more efficient method of failure prediction is derived. This method, called the F criterion, is the root mean squared value of successive differences and is applied to the last N data points. A comparison of the F method with the Delta 25 test is made for a sample of the APOLLO gyroscopes.

The overallreiiability of the APOLLO gyroscope is calculated. The history of system performance is also shown and some of the more interesting predicted failure examples are shown in detail.

E-2028: MULTIPLEXED ELECTRONIC COUPLING DATA UNIT

John H. Barker / November 1966

A time-shared shaft-to-digital electronic conversion system was developed to accurately encode the angular positions of the five multispeed resolvers. This time-multiplexed technique evolved from the electronics designed to digitize shaft angles for the inertial guidance system on board the APOLLO spacecraft. Shaft angles are converted to a sixteen-bit binary number with twenty-arcsecond shaft resolution. Instrumentation accuracy is approximately 10 arcseconds, and shaft rotations can be followed up to a 56-1/4 degree per second rate.

E-2014: TEMPERATURE CONTROL OF THE APOLLO BLOCK II INERTIAL MEASUREMENT UNIT

A.H. Arpiarian / August 1966

Temperature control of six inertial components within an IMU—three IRIGs and three PIPAs—is accomplished by mounting one mercury thermostat on the Stable Member as the sensing element in a bistable temperature control system.

Of the two groups of inertial components, the PIPAs were found to be the most temperature sensitive and therefore dictated the maximum allowable temperature changes. Considering external environmental changes, ambient temperature from $0^{\circ}F$ to $130^{\circ}F$, and coolant temperature from $32^{\circ}F$ to $55^{\circ}F$, the average PIPA control temperature is within a $0.25^{\circ}F$ band.

Temperature change due to different gimbal angles of the individual PIPAs is as high as $\pm 0.3^{\circ}F$, but the combined average change of three PIPAs is small. This temperature control system is comparable to a proportional system that senses average PIPA temperature. Only individual PIPA control can eliminate this position sensitivity.

Two additional mercury thermostats are used to provide high- and low-temperature alarm indications, and one other mercury thermostat is used to control two blowers which act as heat flux valves. All four mercury thermostats sense Stable Member temperature.

E-1942: COMMAND ANGLE TORQUING OF GYROSCOPES IN THE APOLLO GUIDANCE AND NAVIGATION SYSTEM

Maurice Lanman, Julius Feldman / April 1966

The APOLLO G&N system stable platform is positioned by digitally torquing the gyroscopes floats, Command angle torquing to an accuracy of 0.1% can be obtained with this technique. System implementation is discussed in general; the mechanics of torquing the gyroscope float and the precision torquing techniques are detailed. Results of investigation into various phenomena that affect accuracy and duplication are reported.

E-1792: 16 PIPAs MOD D CENTRIFUGE TESTS

G.J. Reichenbacher / August 1965

This report describes the centrifuge testing of two 16 PIPAs, MOD D, Seriai Numbers 69 and 70. The test object was the determination of the linearity and cross-coupling characteristics of the PIPA over an operating range of 0 to 18 g's.

A brief discussion of the PIPA closed loop system theory of operation, including electronics, is followed by a description of the centrifuge setup. An explanation, comments, and conclusions, are presented about the data, its processing, and the resulting curves.

This test program was performed on the centrifuge of the MIT/IL Special Test Facility between 15 February and 12 March 1965.

E-1759: GYROSCOPE TEMPERATURE DEPENDENT DRIFT

Erich K. Bender / August 1965

Thermally induced drift in a rate integrating gyroscope is studied analytically and experimentally. In order to determine the torque on a spherical float, lateral temperature distributions are assumed to be linear. Convective velocity distributions within the flotation fluid are then found by simplifying and solving the Navier-Stokes equations. Experimental data are gathered by placing a gyroscope between two aluminum blocks maintained at different temperatures. The drift of a stable platform incorporating such a gyroscope would be 3.7 MERU/ $^{\circ}$F of lateral temperature difference and 0.3 MERU/^{o}F of mean temperature deviation from reference. Finally, a preliminary design is presented for a passive device to counteract the effects of thermal gradients. Such a device may be built into the float.

E-1551: FUNCTIONS AND MECHANIZATION OF THE APOLLO GUIDANCE AND NAVIGATION SYSTEM (VOLUME ONE: IN-ERTIAL SUBSYSTEM)
Russel A. Larson, Allan R. Klumpp / August 1964

The purpose of this document is to provide definition of the Block I G&N system for engineering evaluation down to the individual subsystem and also to provide information for Systems Assembly and Test personnel.

E-1541: TEST RESULTS ON APOLLO INERTIAL SUBSYSTEM (ISS) #4
James H. Flanders, Richard A. McKern / April 1964

This report contains the test results for ISS #4—the first ISS to be assembled, tested, and calibrated. Reported are the purpose and description of ISS #4, problems discovered during testing, and recommendations for problem solution.

E-1482: INFLIGHT ALIGNMENT ERRORS OF THE IMU STABLE MEMBER
Frederic D. Grant / December 3963

In the inflight alignment mode of the APOLLO G&N system, the fine alignment of the IMU Stable Member (SM) using optics star sightings will be affected by numerous error sources. These error sources and revised rms estimates are listed in the appendix. SM alignment errors are also a function of sextant-IMU gimbal angle configuration, Computer studies yielded data on rms SM alignment errors for space-erection angle configurations. Graphs are presented showing the effect of increasing SM misalignments on overall target errors for different trajectories.

E-1344: CONSIDERATION OF APOLLO IMU GIMBAL LOCK
David G. Hoag / April 1963

The APOLLO IMU provides specific force measurements within the guidance system as well as orientation signals to the control system and the pilot's attitude display. The proper operation of the IMU requires that the gyroscopes mounted on the stable member-the "platform"- generate signals to the gimbal drive servos so that the stable member is kept nonrotating independent of any vehicle rotations. This report discusses the IMU's limitations in maintaining this stabilizing function and the resulting operational and emergency constraints imposed on both mission success and crew safety.

E- 1343: TERNARY PULSE TORQUING
George Oberbeck / May 1963

This report discusses ternary pulse torquing and describes the modified type "E" Torquer.

E-1288: ALIGNMENT ERRORS OF THE IMU STABLE MEMBER
Frederic D. Grant / February 1963

In the space erection mode, the IMU Stable Member fine alignment is affected byvarious error sources, These include errors in gimbal angle indication, axis non-orthogonalities, and navigation base fixture errors. SM alignment errors are also a function of sextant-IMU gimbal angle configuration. Computer studies yielded data on the variation of SM alignment errors with angle configuration that show therms misalignment about SM axes is approximately isotropic.

E-li64: THE RELATIONSHIP BETWEEN IMU DRIFT, MISALIGNMENTS, AND TARGET ERRORS
Frederic D. Grant / May 1962

Graphs are presented for various trajectories to show the effect of elapsed time between IMU SM alignment and the trajectory start on overall system errors at trajectory end. Alignment errors will develop during this time because of gyroscope drift. These graphical studies indicate how the astronaut might make his last alignments before trajectory start.

E-1135: THE MYF TORQUE GENERATOR FOR PULSE TORQUED INSTRUMENTS
George Oberbeck / March 1962

The requirement of high torque inputs to guidance instruments demands more effective use of the volume and power presently used by existing low time-constant-torque generators. A torque generator to meet these needs is proposed in a general discussion of torque-producing devices.

E-1125: PRELIMINARY RESOLVER ANGLE MEASIJREMENT
Ramon L. Alonso / February 1962

A method is presented for measuring the IMU gimbal angles and entering the results into AGC. The interface between the IMU and the AGC is defined, and an AGC program is given that will perform the desired measurement. The measurement method is based on the time elapsed between successive positive-going zero crossings of the signals from resolvers tied to the gimbals.

R-574: HIGH PRECISION CURRENT SOURCE FOR AN INERTIAL SYSTEM: BASIC DC MEASUREMENT TECHNIQUE
Kee Soon Chun / March 1967

One of the errors in the specific force measured by an inertial system accelerometer is directly proportional to the error in the current applied to the torque generator. This results in a velocity error that can be minimized by reducing the uncertainty of the applied current.

This report describes a method of obtaining highly accurate and precise determinations of both the current and its stability.

R-568: GYROSCOPE RELIABILITY ACHIEVED THROUGH PROPER DESIGN AND EFFECTIVE PERFORMANCE MONITORING
Edward J. Hall / March 1967

Reliability in the APOLLO gyroscope is achieved by careful instrument design and effective monitoring during assembly and evaluation. The design aspect includes selection of materials with desirable characteristics (stability, precision elastic limit, density, thermal expansion coefficient) and control of geometry in the critical areas of the ball-bearing spin-axis support and the electromagnetic signal, torquing, and suspension devices. Also critical to design reliability is the use of flotation fluids with reduced sensitivity to thermogravitational separation. Effective use of magnetic shielding against earth's magnetic field and other fields is also important. Assembly procedures that critically affect reliability include solvent flushing techniques for the dry instrument to minimize particulate contamination and a filling procedure to eliminate the gas or temperature sensitivity problem. Included in the early evaluation testing of each gyroscope is a float-freedom measurement to indicate the presence of particulate contamination in a critical area and a temperature sensitivity test to detect any problem with the fill operation. These tests, gyroscope spin motor-power monitoring techniques, and the orderly collection of other bearing criteria provide the necessary history on each individual gyroscope that makes it possible to accept gyroscopes for assignment to guidance systems.

R-505: TEST RESULTS ON A PULSE-TOHQUED PENDULUM WITH A PERMANENT-MAGNET TORQUE GENERATOR

Standley H. Goodwin / November 1965

An analysis is made of the operation theory and the high performance level of an accurate pulse-torqued pendulous accelerometer containing a permanent-magnet torque generator. Experimental results verify the analysis. The key to accelerometer accuracy is the torque generator; therefore, a simple analysis of the torque generator is presented and the results are applied to the instrument design. The operation mode and its effects on the electronics design are discussed. In addition, the effect of the permanent-magnet torque generator on electronics design is considered. Where possible, data are presented to illustrate accelerometer performance. Scale factor stability, bias stability, and linearity are shown.

R-466: VIBRATION EFFECTS ON APOLLO GUIDANCE

Frederic D. Grant / October 1964

This report describes the significant vibration effects on inertial component guidance performance. The transmissionof linear and angular vibrations from spacecraft frame through the navigation base to the Stable Member is considered. Most significant vibration effects produce rectified errors that result in equivalent bias drift or bias error in a constant vibration field. Vibration effects on trajectory cutoff errors are presented for the different APOLLO trajectories. An appendix contains brief descriptions of the significant vibration effects on inertial components.

R-342: DEVELOPMENT CRITERIA FOR SPACE NAVIGATION GYROSCOPES
C.S. Draper, W.G. Denhard, M.B. Trageser / October 1961

Development criteria for space gyroscopes should include low power components and inertial measurement units as well as the proper handling of dissipated heat. This will lead to low power operation, high performance, improved reliability and operating life, and stability of component calibration.

T-421: SYSTEM TEST DETERMINATION OF APOLLO IRIG DRIFT COEFFICIENTS
George T. Schmidt / June 1965

A method of determining the drift coefficients of the gyroscopes in the APOLLO G&N system is presented. The technique requires the guidance computer to perform optimum statistical filtering on the east accelerometer output for a freely drifting stable platform. By using computer simulations of the APOLLO system coupled with the test procedure, it is shown that the drift of the south gyroscope can be determined under both laboratory and launch pad environments with a significant improvement over existing test procedures. The drift coefficients of each gyroscope are found from a series of platform positions.

T-415: GYROSCOPE PERFORMANCE PREDICTION BY MULTIPLE DISCRIMINANT ANALYSIS
Ralph F. Gerenz / June 1965

The assignment of a gyroscope to a drift rate group is attempted on the basis of a short past history of performance. Multiple discriminant analysis is applied to several variates of past performance to find that line in multidimension space that best separates the groups. Grouping is done by drift at the end of various time intervals and drift ranges during an interval.

Section 6 OPTICS

E-2389: STAR-HORIZON MEASUREMENTS FOR ONBOARD SPACECRAFT NAVIGATION
James A. Hand / February 1969

An experiment for assessing the earth's horizon signature as a spacecraft navigational aid is being developed by MIT/IL for a Manned APOLLO Applications Program Flight. Optical measurement of the included angle between a known star and the horizon signature provides data similar to the star-landmark technique employed for implicit navigation. With both a specially-designed star tracker and a horizon photometer to execute the automatic sightings, the onboard computer can use this similarity to derive a refined estimate of spacecraft state vector. The principal advantage-that the horizon signature is not subject to cloud-cover obscuration as are landmark targets-is thereby exploited. The entire experiment-including sightings, data processing, and evaluation-is being designed for inflight performance. Testing progresses on the star-horizon sensors and the required computer programs.

E-2270: LM ALIGNMENT OPTICAL TELESCOPE: THERMAL-HUMIDITY-INTEGRITY EVALUATION REPORT
Arthur Grossman / June 1968

The LM AOT thermal-humidity-integrity evaluation is reported. Using as a guideline the temperature-humidity test procedure (KIC-AR-DTP-3000-Z) for the AOT, the results indicate that condensation can occur at critical areas within the optical path on the cabin side and the metal shield can replace the foam insulation causing only a small difference in performance. The test performed by KIC indicates that the redesigned sun shield would not greatly influence the AOT thermal characteristics.

E-2262: STUDIES OF ONBOARD LUNAR ORBITAL NAVIGATION WITH UNKNOWN AND KNOWN LANDMARKS AND SOME OBSERVATIONS ON NONLINEAR EFFECTS
David S. Baker / August 1968

This report contains three separate onboard lunar orbital navigation studies with optical measurements:

1. Unknown and known landmark navigation comparisons
2. Measurements only on the landing site with an onboard MSFN matrix
3. Some nonlinearities in lunar orbital navigation

The first study evaluates the twomethods of navigation (nine-dimensional state vector) and shows that known-landmark navigation is generally superior to unknown-landmark navigation. The second study indicates that small inertial and relative uncertainties result with the MSFN covariance matrix used. This covariance matrix is overly optimistic in light of recent information. The study of nonlinear effects indicates that Monte Carlo runs must be used to account for possible nonlinearities.

E-2246: COMPUTER-AIDED INERTIAL PLATFORM REALIGNMENT IN MANNED SPACE FLIGHT
James A. Hand / February 1969

Through establishment of logical communications between a star tracker, a computer, and a previously aligned inertial platform, it is shown practical to bypass the manual sighting function and to execute optics pointing, search moding, target acquisition, and data processing so that a completely automatic realignment is accomplished. The technology represented by the APOLLO G&N system is the foundation for developing this automatic inertial platform realignment capability. The new capability is being considered for an APOLLO Applications Program experiment to be conducted from low earth orbit.

E-2172: CLOUD OBSCURATION OF APOLLO LANDMARKS DERIVED FROM METEOROLOGICAL SATELLITE OBSERVATIONS
J. Barnes, D. Berem, A. Glaser
(Allied Research Associates, Inc.)/September 1967

Data from the NIMBUS II and ESSA 3 satellites are used to determine the mean cloud cover over each of 100 landmarks for the APOLLO onboard navigation system and the probability of sighting at least a specified number of landmarks within the first two and one-half revolutions of each of three simulated APOLLO missions, The sample periods were from 15 May through 31 August 1966 and from 9 October 1966 through 28 February 1967,

In the mission simulation program, a sighting probability based on cloud amount is derived for each observation. A Monte Carlo random numbering method is then employed to determine the number of landmarks sighted on each mission.

Cloud patterns derived from the results of the cloud statistics program are in close agreementwithclimatology. Satellite observed cloud amounts, however, are generally less than ground observed. The difference is believed to be due to the existence of small cumulus cells not resolved by the satellite and the overestimation of sky cover by ground observers.

The simulation program shows that the probability of sighting a specified number of landmarks on an APOLLO mission depends both on the cloud climatologies of the landmarks and the number of possible sighting attempts. The nurnber of sightings on a mission also depends on the derivation of the sighting probability for each observation and the time of mission launch. The human confusion factor may limit the sighting of terrestrial landmarks even under ideal cloud conditions.

E-2055: DATA ACQUISITION AND REDUCTION, MIT X-15 HORIZON DEFINITION EXPERIMENT: PHASE II
John R. Lawson / December 1966

Theobjectives and data uses of the MIT X-15 Horizon Definition Experiment are discussed. The equipment, facilities, data formats, backup data and operational data, datareformatting procedures, subsystem calibration procedures, and the computations and analysis performed with the reformatted data and calibration data are discussed. A typical experiment flight sequence is described. Detailed associated documents are referenced throughout the report.

E-2051: DESIGN EVALUATION OF THERMAL-VACUUM TESTS: FINAL REPORT

William A. Vachon / November 1966

Block II Sextant Head

Thermal-vacuum tests were conducted on the APOLLO Block II sextant head in a vacuum of 10^{-5} Torr or lower. The sextant head was surrounded by a liquid nitrogen shroud for simulation of deep-space conditions. During certain test phases, light from a carbon-arc lamp of one-sun intensity illuminated the head. No measurable sextant errors were found as a result of vacuum alone. Under all environmental conditions, it was found that differences between the electrical and optical readout of trunnion mirror position were directly correlated to the temperature difference between the sextant trunnion and the sextant housing. A method of sextant calibration is suggested whereby the operator can subtract out optical errors at the time that an optical sighting is being made. Tables of expected errors under different environmentalconditions are presented as a function of time.

LM Alignment-Optical Telescope

Thermal-vacuum tests on the AOT showed the total optical error to be less than 30 arcseconds in the presence of thermal-vacuum environment (vacuum, sun, cryogenic shroud). Most of the error is due to shifts of the inner optics in the presence of axial temperature gradients along the telescope tube, A negligible amount of error is found to arise due to shifts of the objective prism in the presence of large temperature changes. The transient temperature response of selected members of the AOT is presented for various environmental conditions.

E-2034: LM ALIGNMENT OPTICAL, TELESCOPE, MECHANICAL-INTEGRITY DESIGN EVALUATION: FINAL REPORT

Eugene K. Gardner / September 1966

One shock and seven vibration tests made in the LM AOT optics mechanical-integrity design evaluation (see interim report, E-1978) indicate the reduction of optical baseline measurement shifts to the practical precision of the baseline measurements by fixing of critical mechanical interfaces with taper pins, the maintenance of high bearing preloads, and the establishment of minimalclearance between bearing outside-diameter and bearing seat.

E- 1978: BLOCK II LEM OPTICS DESIGN EVALUATION: INTERIM REPORT
MIT/IL / June 1966

Mechanical integrity and thermal vacuum tests on the Block II OUA, the Sextant Head, and the LEM AOT are evaluated. Baseline and operational test instrumentation, the equipment used, and actual measurements are discussed. Vibration levels are shown for 11 discrete mechanical-integrity tests and are discussed in terms of severity. Test results are summarized in vibration shift tables showing incremental changes in functional performance determined by baseline measurements before and after vibration. A single thermal vacuum test for the AOT and a series of 20 tests for the sextant head are discussed in terms of operational shifts under varying conditions of temperature and pressure. Functional errors are allocated to fixed and movable portions of the sextant head, and a source of error in the trunnion-resolver is analyzed in terms of servo-motor temperature gradients. Design evaluation is summarized, observed shifts related to design of the optical instruments, and additional tests indicated for further evaluation.

E-1975; SIMULATED LEM INFLIGHT IMU ALIGNMENT SIGHTINGS
James Wolf / June 1966

Determining the accuracy with which human observers can mark the instant a simulated star crosses an optics reticle line is central to the human role in IMU alignment.

The data taken in this experiment show mean absolute mark errors of 170 msec and 1.6 arcmin for the star rates of 0.25°/sec intersecting the reticle line at 45°. Star rates of 5°/sec resulted in mean absolute mark errors of 35 msec and 7.5 arcmin. The star moved at a constant rate in each trial.

Ranges of most probable error were 2-3 arcmin for star rates less than 3°/sec and 3-4 arcmin for star rates $3-4^{\circ}$/sec. Above 4°/sec, absolute mark errors in time leveled off at 35-40 msec.

The experimentation was carried out on the MIT/IL Sextant Simulator. Marking star-reticle superposition consisted of depressing a "MARK" button. The data extracted were the angular distances between the reticle and the star at mark. Three well-trained observers, experienced with other navigation sighting techniques, were used. A simulated 2.5 magnitude star was observed against a uniform 2 ft-L background.

E-1901: VISIBILITY OF THE LEM WITH VARIOUS BACKGROUND LUMINANCES
Kenyon L. Zapf / January 1966

LEM visibility has been investigated under a variety of circumstances. In a large dark room, a l/20-scale model of the LEM (finished with a specific aluminum paint) was illuminated by light from a single source to simulate the geometry of visible solar radiation in space. Photometry was used to determine the luminous intensity of the model in various aspects, and photography was used to determine the projected area of the model in each of these aspects. Dividing the intensity by the area and multiplying by the appropriate photometric scale factor results in the average luminance of the real. LEM for a large number of aspect angles and solar phase angles.

In Part I of the report, the well-known Tiffany data are employed to calculate liminal range in nautical miles as a function of aspect angle for a single phase angle and uniform backgrounds of 10, 100, or 1000 foot-lamberts.

In Part II of the report, values of the maximum average luminance and the minimum average luminance of the LEM. per rotation about its Y (pitch) axis are given as functions of the solar phase angle.

The roughly cubic shape of the LEM and the highly specular characteristics of the specific paint used in the test produce large fluctuations of liminal range during a rotation of the LEM about its Y axis.

E-1819: A LUNAR PHOTOMETRIC FUNCTION: CURVES FOR CALCULATING THE DIFFERENTIAL LUMINANCE OF TH:E LUNAR SURFACE
John Gallagher, James M. Hallock / July 3.965

A lunar photometric function and curves for calculating the differential luminance of the visible portion of the lunar surface are discussed.

Detailed examples and comparisons with the results of several observers are presented.

E-1750: COMPUTER PROGRAMS FOR OPTICAL SYSTEM ANALYSIS AND DESIGN
Donald C. Dilworth / February 1965

The two computer programs described are concerned with APOLLO optical system design and analysis. The first, SPOTPLOT, calculates the intersection points at the image plane of a pencil of rays that have passed through an optical system and plots these in the form of a spot diagram. The other, SYNOPSYS (SYNthesis of OPtical SYStems), produces optimum lens designs with specified characteristics.

E- 1687: AN OPTICAL EARTH HORIZON PROFILE BASED UPON TABULATED SOLUTIONS OF CHANDRASEKHAR'S EQUATIONS
Milo Wolff / October 1964

Calculations are made of the luminance of the earth's dtmosphere as a function of the altitude of a line of sight from outer space. Results are obtained as a function of ground albedo, sun angle, and wavelength. Application to space navigation is discussed. Graphical solutions are given for many varied parameters. MIT/IL Report E-634 is preliminary to this report.

E-1634: THE PROFILE OF AN EXPONENTIAL, ATMOSPHERE VIEWED FROM OUTER SPACE AND CONSEQUENCES FOR SPACE NAVIGATION
Milo Wolff / September 1964

A review and preliminary analysis of the problem of using the earth's limb as a space-navigational reference presents calculations of the luminance of the earth's limb as a function of the altitude of the line of sight viewed from outer space. An exponential atmosphere model is used assuming single scattering of light with semi-empirical corrections. MIT/IL Report E-1687 continues the work using a more exact analysis.

E-1547: OPTICAL EQUIPMENT HANDLING GUIDELINES
P.H. Bowditch, D.A. Koso, D.A. Mudgett / March 1964

This paper establishes guidelines for the handling of optical equipment developed for the APOLLO G&N system. These guidelines, based on a comprehensive design review and functional performance requirements under a wide range of environmental conditions, apply specifically to the OUA.

E-1385: VISIBILITY DATA AND THE USE OF OPTICAL AIDS
Arthur C. Hardy (Consultant) / July 1963

During World War II, an extensive visibility program was carried out at the Tiffany Foundation under a contract with the Office of Scientific Research and Development. This report makes the Tiffany data more readily available for possible use in the evaluation of the performance characteristics of optical aids for space navigation.

E- 1353: THE HORIZON PHOTOMETER AND OTHER EARTH PARKING ORBIT GUIDANCE & NAVIGATION MEASUREMENTS
E.M. Copps, et al, / May 1963

The maneuvers of a SATURN IVB vehicle in an earth parking orbit of altitude 100 miles are presented. Vehicle characteristics and a procedure outline for six possible navigation maneuvers are discussed.

The horizon photometer and derivations of its required alignment accuracies are discussed, and expected views of the earth at various distances are presented. 'The time available to make the star-horizon measurements is derived as a function of the star's eievation from the vehicle orbital plane.

E- 1333: CLOUD COVER DATA FOR LANDMARK STUDIES
Chung L. Pu / March 1963

In studying the usefulness of landmarks in making space-navigation measurements, data on cloud cover over certain selected landmarks were compiled from information provided to MIT/IL by the U.S. Air Force. The sample landmarks, distributed around the world within latitudes of $\pm 35^{\circ}$, were selected for their unique terrain characteristics or other defining features that would make them suitable for recognition and tracking from 2 space vehicle. A cloud coverage of three-tenths or less was used as a criterion for landmark visibility, and the mean number of days for each month of the year that 2 landmark meets this criterion is listed.

For the landmarks selected, data are presented showing trends in the best months for visibility conditions, the number of available landmarks by months for each visibility category, and annual averages. These data indicate that cloud cover is a problem that may limit thenumber of useful landmarks for a mission, unless special efforts are made in the flight plan to take advantage of optimum conditions.

E-1206: THE BLUE-WHITE BOUNDARY HORIZON SENSOR
Harold H. Seward / September 1962

The blue-white boundary sensor is designed to minimize or eliminate measurement errors found in other horizon sensors and due to variations in horizon illumination caused by differences in cloud cover, earth surfaces (such as snow, water, and land), or sun aspect.

E-1172: THE VISIBILITY OF STARS
(Calculated from the Tiffany Data)
Arthur C. Hardy (Consultant) / June 1962

From the data accumulated on star visibility by the National Defense Research Committee during World War II at the Louis Comfort Tiffany Foundation, aspects relevant to the APOLLO project have been selected and presented in this report. The experiments were performed in the laboratory and in actual field tests from 2 destroyer escort.

R-482: A METHOD OF ORBITAL NAVIGATION USING OPTICAL SIGHTINGS TO UNKNOWN LANDMARKS
Gerald M. Levine / March 1965

Recursive space navigation and its application to navigation in a near orbit of a planet by means of measuring the directions to known landmarks is discussed. A less restrictive method of recursive orbital navigation is presented in which it is not necessary to identify the landmarks.

Navigational data are obtained from two optical sightings to the same unknown iandm ark. The landmark position and the two points from which the sightings are made determine a plane. At one position between the two sighting points-the normal point-the velocityvectorof the spacecraft has no component perpendicular to the plane. The location of the normal point is obtained as a function of the two sighting points only. It is independent of both the path between the two points and the landmark location. The unknown-landmark orbital navigation procedure is then constructed from these results. Computer simulationresults are presented using this method for both earth and lunar orbital navigation.

T-436: HUMAN OPTICAL TRACKING OF STROBOSCOPIC BEACON FOR ORBITAL RENDEZVOUS

Ivan S.C. Johnson, Jr. / September 1965

This experiment investigates the flash patterns and intensities of a strobe beacon mounted on the LEM to enable a navigator in the orbiting CM to detect and superimpose the beacon image on optics reticle cross-hairs in order to monitor rendezvous progress. Four well-trained observers were used with the MIT/IL, APOLLO Sextant Simulator.

The experiment is divided into two primary parts:

1. In tests for acquisition time, variables are flash pattern (single flashes with periods 1, 2, and 3 seconds; double pulses every other second with pulse intervals of 0.2 and 0.4 secs), beacon intensity, background (star & earth-lit lunar), and command module motion.
2. In tests for accuracy of superposition, variables are flash pattern (I /4, 1/2 sec periods besides the above), spacecraft motion; tests determine the effect of rapidly-moving lunar background, and a dimmer, although still clearly visible beacon, on superposition accuracy.

T-385: HUMAN PERFORMANCE DURING A SIMULATED APOLLO MIDCOURSE NAVIGATION SIGHTING

Charles M. Duke, Jr., Michael S. Jones / June 1964

The effects of certain variables on the performance of man doing a precise superposition task are examined. This simulates the task that the Project APOLLO navigator will be required to perform during the mid-course (translunar and transearth) phases of the proposed lunar excursion. For this investigation, the APOLLO Sextant Simulator located at MIT/IL was used. The variables were rate of spacecraft motion, magnification of sextant telescope, orientation of landmark, and star-landmark contrast ratio. In order to determine the effect of each variable individually, only one was varied at a time.

Three subjects were used. Each performed the superposition task by using a set of hand controllers until the star was on top of the landmark, as seen through the sextant telescope, At this point the subject pressed a “MARK” button, which recorded the error in seconds of arc. For each given set of conditions, the subject performed the task 25 to 30 times. For each such series, the mean error was computed (absolute mean distance from perfect superposition). Statistical tests were then applied to these means to check for significant changes in error due to changing one of the variables.

E-2411: APOLLO GUIDANCE, NAVIGATION, AND CONTROL SYSTEMS: A PROGRESS REPORT
David G. Hoag / April 1969

The status of the APOLLO PGNCS is examined on the basis of experience with the first eight development flights. This paper discusses the inertial, optical, and computer hardware operation. The application of these hardware subsystems to the DAP, rendezvous navigation, midcourse navigation, and entry is examined. The systems are judged to be fully ready to help a crew of astronauts land on the moon.

E-2345: SOFTWARE CONFIGURATION MANAGEMENT PLAN
Robert C. Millard / October 1968

This document describes the management and testing procedures of the MIT/IL Software Configuration Management Plan being employed in the development of APOLLO mission flight programs.

E-1815: PROJECT APOLLO GUIDANCE AND NAVIGATION PROGRAM: QUARTERLY TECHNICAL PROGRESS REPORT
MIT/IL / June 1965

This Quarterly Progress Report reflects the activities of the MIT/IL APOLLO Program from April 1965 through June 1965. This document covers Block I, Block II, and LEM discussed at the system, subsystem, and component levels.

E-1801: APOLLO GN&C EQUIPMENT CONTINGENCY PLAN: (APOLLO GROUND CHECKOUT)
Joseph D. Fleming, Jr., Randall T. Boyd, Jr. / February 1966

This report presents a plan of action for the minimum capability necessary to handle contingencies in GN&C ground checkout consistent with NASA criteria. It also identifies and lists classes of contingencies presently anticipated and applicable to APOLLO GN&C ground checkout and operations. Contingencies considered include mission or configuration changes, equipment failures, delays, and extreme environmental conditions. Such emergency situations as hurricanes, civil disorders, and war are not covered in the present plan.

E-1679: PROGRESS REPORT ON ATTAINABLE RELIABILITY OF INTEGRATED CIRCUITS FOR SYSTEMS APPLICATION

J. Partridge, L.D. Hanley, E.C. Hall / November 1964

Standardization in systems design allows the predominant use of the same simple integrated circuit and the development and maintenance of high reliability. These failure modes indicate present reliability problems; the need for detection, monitoring, and problem elimination; and the necessity of reliability evaluation among suppliers. Success will be demonstrated by extended use of failure rate data made available through the APOLLO program.

E-1621: A NEW PROCEDURE FOR INITIAL PROVISIONING OF SPARES

Warren G. Briggs / July 1964

The operational procedure necessary to implement a concept developed in the author's doctoral thesis, Report T-374, Statistical Theory for Logistics Planning, incorporates subjective probability distributions of the estimated logistics requirements with a relative evaluation of the consequences of excesses or deficiencies in the quantity provisioned. While the procedure is fully described herein, reference is made to the thesis for theoretical justification and illustrative examples.

E-1588: APOLLO G&N SITE ACTIVATION PLAN

Randall T. Boyd, Jr., Gerald C. Hinds (ACSP) / June 1964

The plan by which APOLLO G&N contractor management policies are established and adapted to the activation of the test sites insures the compatibility of site activation schedules with APOLLO test and flight schedules.

E- 1398: COMMENTS ON THE LUNAR LANDING MISSION DESIGN PLAN OF 15 APRIL 1963

John M. Dahlen, John B. Suomala / August 1963

This report comments and recommends changes to the Project APOLLO Lunar Landing Mission Design Plan, MSC, 15 April 1963.

E-1363: STATUS OF APOLLO FLIGHT SAFETY SYSTEM DESIGN AND DEVELOPMENT

Thomas Heinsheimer / June 1963

This report outlines progress in the design, definition, test planning, and hardware development of the Flight Safety System, All data presented refer oniy to the C-l configuration, as a detailed study of the Advanced SATURN has not begun.

E- 1359: FLIGHT TEST PLAN, APOLLO GUIDANCE AND NAVIGATION SYSTEM (AGE-5)

J. M. Dahlen, T. Heinsheimer, J. B. Suomala / May 1963

This report outlines the flight test objectives to be accomplished by the flight of the MIT AGE-5 APOLLO G&N system.

E-1186: Rev. 5 TECHNICAL, DATA RELEASE PROCEDURES

MIT/IL / August 1966

This report establishes the NASA approval requirements and the procedure for the release and revision of the technical data necessary to fulfill MIT/IL'S design responsibility for the APOLLO G&N system. It also establishes the method by which MIT/IL will control the design configuration represented by the released technical data. This procedure is an integral part of MIT/IL Report E-1087, Documentation Handbook and Plan.

E- 1087: APOLLO GUIDANCE AND NAVIGATION, DOCUMENTATION
Rev. 5 HANDBOOK AND PLAN
MIT/IL / October 1966

This publication is a single source reference for the configuration management plan and implementation procedures, including specific instructions and formats for document preparation. It is divided into twelve sections,

H-496: DESIGN REVIEW PROCEDURES
George W. Mayo, Jr. / May 1965

This document defines the design review activity to be undertaken by MIT/IL in connection with the design and development of inertial guidance and other systems. This review is intended to comply with sponsor requirements and the provisions of applicable Military DOD and NASA Specifications,

R-494: APOLLO GUIDANCE AND NAVIGATION TEST DATA PROGRAM PLAN
Nelson Hanover, Joseph D. Fleming, Jr. / July 1965

The Test Data Program Plan establishes the ground rules, concepts, and procedures required to present data in a meaningful form for evaluation. Data, their sources, their handling, and their control are identified and described for each testing phase. Detailed examples of data uses are included as appendixes.

R-445: TECHNICAL DEVELOPMENT STATUS OF APOLLO GUIDANCE AND NAVIGATION
Norman E. Sears / April 1964

The primary guidance and navigation system of the APOLLO CM and LEM is described. Operating procedure for the lunar orbit phases of the APOLLO mission—LEM orbital descent, landing, ascent, rendezvous, and the lunar orbit navigation phase in the CM (since it establishes the initial inputs for the LEM guidance system)—are described. The general guidance concepts and the manner in which the particular guidance and navigation system units are used by the astronauts to achieve the required objectives of the individual phases are then outlined.

R-434: PRELIMINARY GUIDANCE NAVIGATION TEST PLAN
MIT/IL / December 1963

This APOLLO G&N Test Plan defines and governs the ground and flight test programs. By the processof periodic revision, it will be maintained compatible with engineering changes and higher order documents such as the APOLLO Spacecraft Development Test Plan. System allocations and schedules are purposely not included in order to avoid too frequent revision. Subordinate documents which describe and govern the implementation of individual tests or test programs called for by this plan are cited where appropriate. The flight and ground test programs provide essential data and verify that the G&N system is capable of performing all its required functions.

R-396A: APOLLO GUIDANCE AND NAVIGATION QUALITY ASSURANCE PLAN
Edward T. Driscoll / June 1964

This report describes the APOLLO Quality Assurance Program organized to comply with the provisions of NASA NPC ZOO- 2 in the design, fabrication, and use of APOLLO G&N systems. The quality and reliability controls discussed in detail here have been established by MIT/IL for coordination between its efforts and those of the program contractors-ACED, Kollsman, and Raytheon.

R-396: APOLLO GUIDANCE AND NAVIGATION QUALITY ASSURANCE PLAN
George W. Mayo, Jr., Edward T. Driscoll / April 1963

This report describes the APOLLO Quality Assurance Program organized to comply with the provisions of NASANPC 200-2 in the design, fabrication, and use of APOLLO G&N systems, The quality and reliability controls discussed indetail here have been established by MIT/IL for coordination between its efforts and those of the program contractors-ACED, Kollsman, and Raytheon.

R-388: A PROGRESS REPORT ON THE APOLLO GUIDANCE SYSTEM
David G. Hoag, Milton B. Trageser / December 1962

This report discusses technical approaches, expected performance, operational capability, and system configuration as defined in early November 1962. The presentation includes the guidance approaches used on both the CM and the LEM. There is a brief discussion of the physical phenomena used for guidance in various mission phases. System configuration and CM installation are outlined. Possible configuration and LEM installation are previewed. There are physical descriptions of the several major guidance subsystems. The expected performance of these subsystems is discussed. The method of system operation for various phases of the lunar landing mission is described. The astronaut-guidance system relationship is defined.

R-383: DESIGN REVIEW PROCEDURES
George W. Mayo / September 1962

This report defines the design review activity undertaken by MIT/IL in the design and development of the APOLLO G&N system. It complies with NASA requirements and the provisions of Report R-349 (Rev. A), Guidance and Navigation System Reliability Program, September 1962.

Section 8. RADAR

E-2185: LANDING RADAR-LGC INTERFACE EXPERIMENTS: PART II, DATA PACKAGE
Walter E. Tanner / September 1967

Part II of this report consists of Landing Radar test data from flights 13, 14, 17, 18, 19, 22, and 25. The test flights were conducted in an SH-3A helicopter by MSC personnel and NASA contractors at White Sands, New Mexico, from January through May 1967. Raw Doppler and range data were recorded in analog form on magnetic tape, processed at MIT/IL in the Landing Radar Experimental Assembly 4L and read by an LGC for transmission through a simulated LM downlink.

Data processing consists of comparing MIT/IL data with tine-theodolite tracking data from White Sands. The resulting errors of the radar's threevelocitycomponents and its slant range are plotted versus universal time. Velocity data were taken with three different readout routines that result in tracker smoothing times of 220, 300, and 460 msec. Conclusions of these experiments will be reported in Part I of this report, scheduled for a later release date.

E- 1982: LEM PGNCS AND LANDING RADAR OPERATIONS DURING THE POWERED LUNAR LANDING MANEUVER
B.A. Kriegsman (Raytheon, Resident Staff), N.E. Sears /
June 1966

This report summarizes a study of the LEM PGNCS during automatic-controlled powered landing maneuvers prior to moon touchdown. The navigation system employs as the primary sensor an IMU that is updated at discrete intervals of time by LR altitude and velocity measurements. The document describes the G&N systems (including key mathematical relations) and the development and selection of weighting functions used in updating the navigation system by LR measurements. Extensive data are presented from a digital-simulation study of the G&N system performance in the presence of initial-condition errors, propulsion-system uncertainties, random and bias-sensor errors, and lunar terrain altitude variations.

E-1976: RADAR INTERFACE EVALUTION, PHASE I: FINAL TEST REPORT
Walter E. Tanner, William Saltzberg / May 1966

This final report of &he Phase I Radar-PGNCS Interface Test Program, concluded at MIT/IL in March 1966, evaluates the interface circuit design of the engineering prototypes of the LEM RR and LR. It also evaluates the effects of long transmission-cables on the wave shape of pulse interface signals and on the operational characteristics of the interface circuits. Included are measurements of all digital interface signals of pulse and dc discrete circuits, taken at the interface between the LGC and the two radars connected through simulated spacecraft cabling.

E- 1937: VELOCITY ERROR IN LUNAR LANDING RADAR CAUSED BY DIGITAL PROCESSING
Janusz Sciegienny / March 1966

The vehicle velocity during lunar landing is determined from radar measurements of Doppler shifts along three antenna beams. The velocity components along the antenna axes are computed from the Doppler shifts by counting the number of Doppler pulses, superimposed on a train of bias pulses, during five successive counting gates. During the lunar hover, the velocity round-off error caused by multiple counting gates becomes comparable to the velocity error from all the remaining sources. Synchronization of the counting gates with the bias pulses decreases both the maximum value and the mean value of the velocity round-off error, but increases the standard deviation of the error. A reduction of the velocity round-off error, resulting from counting in multiple gates; can be accomplished by subtracting the mean error and by synchronizing the gates with the bias pulses.

E- 1904: PGNCS LANDING RADAR: FUNCTIONAL AND PERFORMANCE SPECIFICATIONS
Norman E. Sears, Leonard B. Johnson / February 1965

This report specifies LR characteristics necessary for support of the PGNCS during lunar descent, The specifications complement the general radar requirements originally identified in Report R-404: Radar Requirements for Primary Guidance and Navigation Operation, April 1963, and are based on mission plans described both in Report R-404 and in Report R-446: Primary G&N System Lunar Orbit Operations, April 1964. The report was prepared in response to Action Items L64A-000-87 and L64A-000-88.

E- 1903: PGNCS RENDEZVOUS RADAR: FUNCTIONAL AND PERFORMANCE SPECIFICATIONS
Norman E. Sears, Leonard B. Johnson / March 1965

This report specifies radar characteristics necessary for support of the PGNCS during lunar descent, surface operations, and ascent. The specifications complement the general radar requirements originally identified in Report R-404: Radar Requirements for Primary Guidance and Navigation Operation, April 1963, and are based on mission plans described both in Report R-404 and in Report R-446: Primary G&N System Lunar Orbit Operations, April 1964. The report was prepared in response to Action Items L64A-000-87 and L64A-000-88.

E-1810: ANALYSIS OF LANDING RADAR GEOMETRY DURING REDESIGNATION OF LUNAR LANDING SITE
Janusz Sciegienny / July 1965

The LR consists of a velocity sensor and an dltitude sensor. The velocity sensor measures velocity components along three beams, and the altitude sensor measures a slant range along one beam. The geometry of radar beams during lunar landing is investigated for the present beam configuration and for a modified beam orientation. Three landing trajectories are simulated on the computer: a “standard” trajectory, a trajectory with forward redesignation of the initially selected landing site, and a trajectory with redesignation to the right of the initial landing site. The trajectory with redesignation to the right may result in unsatisfactory beam geometry in the present beam configuration, but the geometry of modified beam orientation remains within satisfactorybounds. In allcases, the geometry of modified beam orientation results in more favorable radar performance than the geometry of the present beam configuration,,

E-1803: SYSTEM ANALYSIS OF LUNAR LANDING RADAR
Janusz Sciegienny / June 1965

This report analyzes the current configuration of the LR. The mathematical derivations given in the appendixes may be applied to any configuration of landing radar. The analysis based on the geometry of radar beam configuration and landing trajectories presents the inherent accuracy limitationsof velocity measurement by the LR. The results indicate that the least accurate velocity measurement is in the direction perpendicular to the trajectory plane.

E-1769: TYPICAL MISSILE AND SATELLITE RADAR TRACKING SYSTEMS
Walter E. Tanner / April I965

The different radio and radar tracking systems currently in use or under development for missile and satellite tracking are introduced. Performance characteristics of typical radar systems, information on data readout and data processing, and special requirements for radar sensors are discussed.

R-597: FLUCTUATION ERRORS OF DOPPLER SENSORS AT LOW VELOCITIES
Walter E. Tanner / December 1967

An exact theory is developed for the fluctuation error of an idealized Doppler-velocity sensor. Particular attention is given to sensor performance in the low velocity regime, where signal bandwidth and smoothing bandwidth are of the same order of magnitude. The theory for the idealized sensor is then compared with flight test data from an actual sensor, and a performance figure (tracker noise figure) is established and discussed.

R-541: THE POWER DENSITY SPECTRUM OF THE ECHO FROM AN AIRBORNE DOPPLER RADAR
Martin Schetzen / March 1966

An analytical study of the echo-power density spectrum for an airborne Doppler radar traveling at constant altitude and velocity shows the effect upon the spectrum of the antenna pattern and the terrain's reflecting properties. A detailed analysis is made for the usual case of "rough" terrain; and graphs of the relationship between the echospectrum and the antenna beam width are given for a Gaussian antenna pattern.

R-404: RADAR REQUIREMENTS FOR PRIMARY GUIDANCE AND NAVIGATION OPERATION
MIT/IL / April 1963

This report specifies the radar requirements for the PGNS for primary and secondary mission objectives. The LEM LR and RR and the CSM RR are specified for required operating ranges and performance. Chapter 4 presents detailed description of the LEM and CSM requirements for interface between the radar units and other primary guidance units. Possible RR antenna locations on the LEM and CSM are discussed with associated gimbal-limit requirements. Backup radar guidance and display requirements are not considered in this specification.

E-2364: INERTIAL COMPONENTS RELIABILITY AND POPULATION STATISTICS: REPORT IV

Julius Feldman, Stefan Helfant / January 1969

The APOLLO gyroscope failure and operational statistics are presented versus wheel hours from assembly and calendar time from acceptance. The inventory of inertial components in active systems at field sites, including gyroscope removals, is also presented. Gyroscope wear-out characteristics and recommended test procedure changes are discussed.

E-2333: INERTIAL COMPONENTS RELIABILITY AND POPULATION STATISTICS: REPORT III

Martin Landey / December 1968

A detailed analysis of scale factor and bias data from system- level tests of the 16 PIPA MOD D is made. For this effort, a relatively unperturbed environment is constructed by editing out data reflecting shifts clearly traceable to system operation anomalies. Analysis is performed using several computer programs written for this purpose. It is concluded that the system-level characteristics are similar to component-level operational characteristics of the instrument.

Particular observations include an estimate of scale factor drift with PIPA aging, normality of test-to-test scale factor shifts, and non-normality of test-to-test bias shifts.

E-2299: RELIABILITY AND QUALITY ASSURANCE PROGRAM PLAN FOR PROCUREMENT OF APOLLO IRIG

Reliability Group / December 1968

This report presents an ordered, disciplined plan for controlling the quality and reliability of the APOLLO II IRIGs and for maintaining these features throughout manufacturing and test-life phases.

E-2274: INERTIAL COMPONENTS RELIABILITY AND POPULATION STATISTICS: REPORT II
Julius Feldman, Roscoe Cooper / June 1968

APOLLO PIPA and APOLLO gyroscope reliability statistics are presented involving the distribution of failed and non-failed units and their failure rate versus time from acceptance.

These distribution and bearing failure rates versus wheel hours and the percentage of gyroscope failures grouped according to manufacturing data are presented for both the APOLLO I and APOLLO II gyroscopes.

E-2252: INERTIAL COMPONENTS RELIABILITY AND POPULATION STATISTICS: REPORT I
Julius Feldman, Roscoe Cooper / March 1968

APOLLO PIPA and APOLLO gyroscope reliability statistics are presented. The APOLLO gyroscope drift terms' standard deviation distribution at component acceptance testing, system acceptance testing, and gimbal systems testing is presented as bar graphs for both APOLLO I and APOLLO II gyroscopes. The distribution of failed and non-failed gyroscopes and their failure rates versus wheel hours are presented for both APOLLO I and APOLLO II gyroscopes. PIPA failures are shown grouped according to failure mode; and all failed units are tabulated to show cause, symptoms, and unit locations.

E-2218: LEAD FAILURE STUDY FOR THE MOTOROLA 1-MIL WEDGE-BONDED 1006323 TRANSISTOR
William J. Day, Jayne Partridge / December 1967

This report presents the results of a thermal analysis of the failure modes for the 1-mil diameter aluminum lead wires of the SCD 1006323 wedge-bonded transistor.

E-2141: APOLLO GUIDANCE, NAVIGATION, AND CONTROL SYSTEM GYROSCOPE RELIABILITY
John E. Miller, Julius Feldman / June 1967

The reliability of the APOLLO PGNCS depends to a large measure upon the reliability of the inertial reference gyroscope. A proven design, the Polaris Mod II Inertial Reference Integrating Gyroscope (IRIG), is used as the starting point. Although the signal and torque microsyns were redesigned to meet the frequency and computer interface requirements, the critical reliability components--the float and wheel package--are unchanged from a design already in production.

This paper presents the method used to increase the operational reliability of the APOLLO gyroscope by a factor of twenty, despite the fact that the wheel assemblyreliability had exploited the state of the art. The results of the reliability program lead to a method of forecasting the occurrence of the gyroscope's primary failure mode--the failureof the wheel bearing assembly. This failure predictionmethod, the Delta 25 test, is heuristically derived. Application of this test to a sample of gyroscopes confirms reasonable accuracy in failure prediction leaving sufficient good-performance operating time to complete a lunar mission. A second, more efficient method of failure prediction is derived. This method, called the F criterion, is the root mean squared value of successive differences and is applied to the last N data points. A comparison of the F method with the Delta 25 test is made for a sample of the APOLLO gyroscopes.

The overall gyroscope reliability of the APOLLO gyroscope is calculated, The history of system performance is also shown and some of the more interesting predicted failure examples are shown in detail.

E-2128: A PROPOSED MULTILAYER PLATED-WIRE DEVICE WITH POSSIBLE IMPROVEMENTS IN THE NON-DESTRUCTIVE PROPERTIES OF A PLATED-WIRE MEMORY SYSTEM

Donald Kadish / May 1967

The use of amultilayer plated-wire in a non-destructive read-out memory system is explored. A steering technique is described which offers possible advantages over present operation methods. Plating parameter control and substrate treatment are also discussed.

E-2099: ERASABLE MEMORY INVESTIGATIONS

David Shansky / March 1967

Work proceeded in four distinct areas in this project: wire plating, development of inexpensive solenoidalword-line coils, circuit design aimed at producing anoperable plated-wire memory based on the Toko memory stack, and electronic instrumentation and controls for the wire plating line. The plating line in use during the early part of this year was completely redesigned and rebuilt to allow for more precise control of wire movement through the various baths. Controls were provided for temperature, flow rate, plating current, and pH; and a spectrophotometer was acquired to permit more precise control of solution composition. The plating line is currently in operation and is producing wire of controlled composition suitable for use as the storage element in a non-destructive read-out memory, An exerciser for the Toko memory was designed and constructed. Two selection and drive systems for the Toko memory were designed and breadboarded.

E-2095: EVALUATION REPORT ON MULTILAYER CIRCUIT BOARD CONNECTORS USING THE SOFT METAL INDIUM FOR ELECTRICAL CONTACTS

Thomas A. Zulon / March 1967

Using weldable, multilayer circuit boards to connect the logic elements in G&N digital computers has resulted in problems associated with the interconnection of these circuit boards. A special connector structure has been designed to study these interconnection problems. The structure has a strongback that compresses an elastomer, that in turn distributes pressure to soft-metal-plated connector fingers and furnishes the system's elastic reserve. The soft-metal plating on the connector fingers is used to improve the electrical characteristics of contact interfaces, The soft metals have low crushing strengths and generate larger junction areas. They also annealout work-hardening at or below room temperature, thereby giving these junction areas tolerance to substantial stress deformation without fracture. In addition, some combinations of junctions between the softer and harder metals exhibit diffusion-alloying effects at room temperature. The Lndium-Nickel reacting system was chosen for this experimental study, because faster acting systems might be detrimental to the repairability characteristic of these special connectors.

E-205 1: DESIGN EVALUATION OF THERMAL-VACUUM TESTS: FINAL REPORT

William A, Vachon / November 1966

Block II Sextant Head

Thermal-vacuum tests were conducted on the APOLLO Block II sextant head in a vacuum of 10^{-5} Torr or lower. The sextant head was surrounded by a liquid nitrogen shroud for simulation of deep-space conditions. During certain test phases, light from a carbon-arc lamp of one-sun intensity illuminated the head. No measureable sextant errors were found as a result of vacuum alone. Under all environmental conditions, it was found that differences between the electrical and optical readout of trunnion mirror position were directly correlated to the temperature difference between the sextant trunnion and the sextant housing. A method of sextant calibration is suggested whereby the operator can subtract out optical errors at the time that an optical sighting is being made, Tables of expected errors under different environmentalconditions are presented as a function of time.

LM Alignment-Optical Telescope

Thermal-vacuum tests on the AOT showed the total optical error in the AOT to be less than 30 arcseconds in the presence of thermal-vacuum environment (vacuum, sun, cryogenic shroud). Most of the error is due to shifts of the inner optics in the presence of axial temperature gradients along the telescope tube. A negiigible amount of error is found to arise due to shifts of the objective prism in the presence of large temperature changes. The transient temperature response of selected members of the AOT is presented for various environmental conditions.

E-2034: LM ALIGNMENT OPTICAL TELESCOPE, MECHANICAL-INTEGRITY DESIGN EVALUATION: FINAL REPORT
Eugene K. Gardner / September 1966

One shock and sevenvibrationtests made on the LM AOT optics mechanical-integrity design evaiuation (see interim report E-1978) indicate the reduction of optical baseline measurement shifts to the practical precision of the baseline measurements by the fixing of critical mechanical interfaces with taper pins, the maintenance of high bearing preloads, and the establishment ofminimalclearance between bearing outside-diameter and bearing seat.

E-1971: RELIABILITY ASSESSMENT OF MICROCIRCUITS AND MICROELECTRONIC SYSTEMS
Eldon C. Hail / June 1966

This investigation of microcircuit components and microelectronic systems reliability includes the short-term testing done. Where data are available, a comparison is made between short-term tests and system operating history. For some microcircuit components, a correlation can be made that demonstrates the value of short-term testing for reliability assessment. Some of the problems that affect system reliability of microcircuits in digital systems are discussed. These include worst-case electrical design, noise immunity, etc. Some reliability problems of mechanical packaging and thermal design are also discussed briefly,

In order to reproduce a microelectronic system with high reliability requirements, a set of production process controls is suggested for the microcircuit components. These should include tests and procedures designed to weed out the most prevalent failure modes, and then, during system assembly, repeated inspections and tests to detect the workmanship problems and failure modes most likely in the assembly procedures.

E- 1944: THE IMPACT OF THE FLIGHT SPECIFICATIONS ON SEMI-CONDUCTOR FAILURE RATES
Jayne Partridge, L. David Hanley / June 1967

The procurement, screen and burn-in, and field history of AGC semiconductor parts are given. Both field failures and performance variability through screen and burn-in are directly related to changes occurring in the parts manufacturers' facilities. Developing and sustaining high reliability are discussed.

E-1926: A DESIGN PROCEDURE FOR RADIATION HARDENING CERTAIN NOR GATE LOGIC CIRCUITS
Harold E. Maurer / March 1966

A design procedure is developed which increases the radiation tolerance of switching-and-computing circuits constructed of an RTL Nor gate.

Logic diagrams are hardened by the use of a modified Nor gate constructed of these RTL gates. A unique concept called "minimum fanout" is introduced and restrictions imposed by design procedure on minimum fanout are discussed. The constraints on the hardening procedure imposed by conventional fanin and fanout restrictions result in a set of equations that can be viewed as an integer linear programing problem. The limitations on a solution to this programing problem are also the limitations on this type of permanent-damage, radiation- hardened logic diagram.

E-1859: DESIGN PRINCIPLES FOR A MULTILAYER CIRCUIT BOARD CONNECTOR USING SOFT METAL CONTACTS
Theodore C. Taylor / October 1.965

Some proposed design principles are developed for a connector structure for use in joining multilayer circuit boards in the configuration of a module board on a single masterboard, or back panel. The structure uses a beam (or strongback) to compress an elastomer pad, that in turn, distributes pressure to a number of electrical contacts and furnishes the system's elastic reserve. Formulas are developed to analyze the stress-deflection mechanics of a strongback and are used together with typical stress-stram data for an elastomer pad in an example design.

E-1838: THE APPLICATION OF FAILURE ANALYSIS IN PROCURING AND SCREENING OF INTEGRATED CIRCUITS
L.D. Hanley, J. Partridge, E.C. Hall / October 1965

The procedure for the testing, screening, and lot rejection of integrated circuits for the APOLLO G&N computer is described. The procedure—based on a knowledge of failure modes, failure mechanisms, and contributing causes to failures in device manufacturing-attempts to increase integrated circuit reliability by screening and analyzing weak devices and using the generated data to quantitatively assess the lot for acceptance, rework, or rejection. The technique, which is primarily aimed toward high-usage, high-volume devices, was developed after extensive testing of tens-of-thousands integrated circuits. The process documents included in the appendix contain stress test procedures, classification of failure modes, numerical rejection limits per class of failure modes, internalvisual rejectioncriteria, and leak test procedures, To emphasize the need for the described technique, **data** are presented showing variations among vendors and variation among procurement lots shipped from a single vendor . The contributing factors to the variations are discussed.

E-1823: THE DESIGN OF SERIES-LOADED THERMAL CONDUCTORS FOR MINIMUM WEIGHT
Theodore C. Taylor / July 1965

Aerospace electrical-and-electronic equipment design frequently makes use of thermal conduction as the primary means of heat removal in metal structural members, This use can result in a thermal conductor subject to series-loading by a number of concentrated and distributed heat loads. This report presents techniques for designing one-dimensional, series-loaded thermal conductors of minimum weight, The use of the techniques is illustrated by two design examples.

E- 1802: CHARACTERISTICS OF SOFT METALS FOR ELECTRICAL CONNECTIONS
Theodore C. Taylor / June 1965

The reliability requirements of aerospace electrical and electronic equipment place severe demands on the technology of miniature electrical connection devices. The reliability-promoting features of larger connection devices-large mating forces, substantial deformation of mating surfaces, rigid structures, and diffusion-alloying at metal interfaces-are difficult to achieve in small devices with ordinary contact materials. The. mechanics of contact formation between metals indicate that conventional metals give very small true-junction areas in miniature, low pressure connections. Because of small size and work-hardening effects, these junctions are subject to destruction by both corrosion and displacement fracturecaused bymechanicalstress. Connectioninterfaces could use certain soft metals to improve interface resistance to destruction. The soft metals have low crushing strengths, and also anneal out work-hardening at or below room temperature. They generate larger junction areas tolerant of substantial stress deformation without fracture. In addition, some combinations of junctions between soft and harder metals exhibit diffusion-alloying effects at room temperature. A properly managed application of the rheological and metallurgical properties of the soft metals should produce useful advances in the art of miniature electrical connections.

E- 1792: 16 PIPAS MOD D CENTRIFUGE TESTS
G.J. Reichenbacher /August 1965

This report describes the centrifuge testing of two 16 PIPAs, MOD D, Serial Numbers 69 and 70. The test object was the determination of the linearity and cross-coupling characteristicsof the PIPA over anoperating range of 0 to 18 g's.

A brief discussion of the PIPA closed loop system theory of operation, including electronics, is followed by a description of the centrifuge setup. An explanation, comments, and conclusions, are presented about the data, its processing, and the resulting curves.

This test program was performed on thecentrifuge of the MIT/IL Special Test Facility between 15 February and 12 March 1965.

E-1713: COMBUSTION OF POLYOLEFIN: INSULATION AND TOXICITY OF TEFLON UNDER ELECTRICAL OVERLOADS
A.O. Mirarchi / December 1964

Tests were conducted on polyolefin insulated #24 gauge wires to determine conditions of insulation breakdown, ignition, and explosion. Some wires were treated with water-repellent coatings.

E-1699: A PROPOSED CIRCUIT STRUCTURE FOR COMPUTER LOGIC BASED ON SEMICONDUCTOR FLAT PACKS INTERCONNECTED BY MULTILAYER CIRCUIT BOARDS
Theodore C. Taylor / December 1964

The current interest in using weldable, multilayer circuit boards as a means for interconnecting the logic section of an aerospace digital computer results in a study of some utility problems of circuit board structures. In addition to the necessary characteristics of reliability, thermal and structural integrity, manufacturability, and great interconnection capacity; a good circuit structure should also possess such features as small size per component, convenient shape, and convenient means of detailed electrical inspection and repair, A structure is proposed that appears to offer a favorable combination of these features.

E-1688: SAMPLING RELIABILITY TEST PLANS WITH RELATIVELY SMALL LOT SIZES
Harry H. Cho / December 1964

Some tradeoff problems and solutions for the planning of the reliability test and evaluation are presented in tabular form. The tables provide the minimum sample size to be tested (given a relatively small lot size), extremely large target of mean-time-to-failure, confidence desired, acceptance number of failures, and the number of hours to be tested per sample component in terms of MTTF. The tables are applicable to all major components of the APOLLO G&N system as well as of the other APOLLO systems. The tables may also be useful to all air/space-borne components characterized by a small lot size and high MTTF target,

E-1679: PROGRESS REPORT ON ATTAINABLE RELIABILITY OF INTEGRATED CIRCUITS FOR SYSTEMS APPLICATION
J. Partridge, L.D. Hanley, E.C. Hall / November 1964

Standardization allows systems to use predominantly the same simple integrated circuit and allows the development and maintenanceof the high reliability of integrated circuits, The approach will be justified by the presentation of integrated circuit failure modes detected, The integrated circuit failure modes will indicate present reliability problems, the need for detection, monitoring, problem elimination, and the necessity for reliability evaluation among suppliers. Success will be demonstrated by extended-use failure rate data made available through the APOLLO program.

E-1539: THERMAL PROPERTIES OF SOME STRUCTURAL MEMBERS FOR SPACE-BORNE COMPUTER ASSEMBLIES
Theodore C. Taylor, Thomas A. Zulon / March 1964

The design of missile and space-borne computer assemblies currently involves a chassis-and-module type of construction, using a considerable amount of structural metal. The amount of metal required in chassis or tray parts is a function of a number of considerations, one of which is the heat-conducting properties of these parts, This report is intended to acquaint computer assembly designers with the thermal conduction properties of some major chassis configurations which have been used or are applicable to use in computer assemblies. The report concludes with an idealized analyticalmodel to show several different heat conduction processes and the general effects of tray design on the thermal weight requirements in an assembly involving a typical configuration.

E-1420: ON THE EXTRAPOLATION OF ACCELERATED STRESS CONDITIONS TO NORMAL STRESS CONDITIONS OF GERMANUIM TRANSISTOR
Jayne Partridge / November 1963

Accelerated stressing has become a popular method of evaluating high reliability devices in short periods of testing time. The convenience of accelerated testing should not be the prime factor for its use. The failure modes generated by normal (within rating) and accelerated (exceeding rating) stressing should be compared before failure rates created by accelerated stressing are accepted as valid.

E-1413: SOME REVERSIBLE AND MORE PERMANENT EFFECTS OF MOISTURE ON THE ELECTRICAL CHARACTERISTICS OF GERMANIUM TRANSISTORS FOLLOWING STRESS
Jayne Partridge, Arnold J. Borofsky
(Sylvania Electric Products) / September 1963

Many changes in surface-dependent transistor parameters occur after temperature and voltage stress, because of a change in the amount of junction moisture. Voltage and temperature generally produce opposing effects when parameter changes are due solely to moisture migration. The time constants associated with return to equilibrium after stress vary from minutes to months. This variation depends on the amount of moisture present in the surrounding ambient, the magnitude and time of stress application, and the amount of junction moisture. The differences and similarities among germanium npn and pnp alloy and mesa transistors are considered. The dependence of these effects on the surface potential produced by processing is discussed, since some moisture effects are not present when equal amounts of moisture are present on the junction. The relation of ambient moisture tc stability after operating and storage life testing is shown.

E-1386: REPORT ON CLEAR RESINS
Samuel C. Smith / July 1.963

After Epocast 1786 and Catalyst 9115 were recommended for potting gimbal-mounted electronic modules, certain irregularities were encountered. Two major problems in Epocast 1786 are that it pulls away from the corners of its potted case and softens at temperatures above 120°F. The former problem is corrected by preheating the case to 100°F; the latter, by using Catalyst 955. One major problem of Chomerics 202 is that the resin does not bond to its case. This problem is solved by using Resin 219, a solvent-based primer. However, the susceptibility of the resin to cracking at extreme temperature changes bears watching. In a third resin, Stycast 1264, softening at elevated temperatures is corrected by the addition of Catalyst 9, making the resin a three-part component. Three clear resins are recommended for use in potting the gimbal-mounted electronic modules.

E-1322: APOLLO GUIDANCE AND NAVIGATION FAILURE REPORTING AND CORRECTIVE ACTION PLAN
George W. Mayo, Jr. / April 1963

The Failure Reporting and Corrective Action Plan for APOLLO G&N equipment outlines the procedures, the flow of information, data collection, and reporting applicable to all elements of the airborne and ground test equipment,

Document preparation was a joint effort of MIT/IL and the participating contractors—AC Spark Plug, Kollsman Instrument Corporation, and Raytheon Company.

E-1270: GLOSSARY OF IMPORTANT RELIABILITY TERMS
Edward T. Driscoll, William J. Beaton / December 1962

Since reliability is a major design consideration in space systems contracts, engineers involved with space systems design and development have an increasing need for understanding certain reliability terms and key words in common usage. This manual provides such information by listing those definitions used most frequently.

R-569. APOLLO G&N FLIGHT SYSTEM RELIABILITY
R.R. Ragan, G.W. Mayo, J. Feldman /March 1967

To meet the high-probability success requirement in the APOLLO G&N, early studies showed the importance of redundancy and inflight maintenance. However, these presented a heavy cost in CM weight and power. The same studies showed an order-of-magnitude improvement of only two components would produce an order-of-magnitude improvement in G&N reliability.

Proper specification, controls, testing, and qualification have improved the reliability of integrated circuits by slightly less than an order of magnitude over what it was at the beginning of the program. The most important failure mode of the gyroscope is a performance degradation. Simple monitoring of the drift prior to launch gives ample warning of impending failures. This technique allows removal of enough of the impending failures to leave the remaining sample with a demonstrated MTBF of 173,000 hours.

R-568: GYROSCOPE RELIABILITY ACHIEVED THROUGH PROPER DESIGN AND EFFECTIVE PERFORMANCE MONITORING
Edward J. Hall / March 1967

Reliability in the APOLLO gyroscope is achieved by careful instrument design and effective monitoring during assembly and evaluation. The design aspect includes selection of materials with desirable characteristics (stability, precision elastic limit, density, thermal expansion coefficient) and control of geometry in the critical areas of the ball-bearing spin-axis support and the electromagnetic signal, torquing, and suspension devices. Also critical to design reliability is the use of flotation fluids with reduced sensitivity to thermogravitational separation. Effective use of magnetic shielding against earth's magnetic field and other fields is also important. Assembly procedures that critically affect reliability include solvent-flushing techniques for the dry instrument to minimize particulate contamination and a filling procedure to eliminate the gas or temperature sensitivity problem. Included in the early evaluation testing of each gyroscope is a float freedom measurement to indicate the presence of particulate contamination in a critical area and a temperature sensitivity test to detect any problem with the fill operation. These tests, gyroscope spin motor-power monitoring techniques, and the orderly collection of other bearing criteria provide the necessary history on each individual gyroscope that makes it possible to accept gyroscopes for assignment to guidance systems.

R-439: FAILURE, TROUBLE, AND CORRECTIVE ACTIONS SUMMARY
DESIGN PHASE: REPORT I
G. Kruszewski / February 1964

This document provides a summation of all failures and troubles reported to date on APOLLO G&N design and development type hardware and a detailed account of the corrective action taken with regard to each instance. True catastrophic failures, out-of-tolerance conditions, and quality defects from breadboard and prototype equipments have been included.

This report deals only with data accumulated incidental to design activities. The equipment involved was not necessarily configured or tested as the final deliverable end items, Nor was it manufactured or assembled to Class A documentation, in accordance with required production methods. AGE- 5 is the first production system; it is subject to the integrated and electronically processed G&N failure reporting and corrective action system described in detail in Report E-1322,

H-395: APOLLO GUIDANCE AND NAVIGATION SYSTEM RELIABILITY
APPORTIONMENTS AND INITIAL ANALYSIS
George W. Mayo, Jr., George E. Kruszewski / February 1963

The reliability requirements for mission success are apportioned to elements of the G&N system and an initial analysis of this reliability is presented.

R-389: REQUIREMENTS OF AND INDEX TO DESIGN EVALUATION,
Rev. C QUALIFICATION, AND RELIABILITY TEST PROGRAM FOR APOLLO GUIDANCE AND NAVIGATION SYSTEM
James E. Kent / June 1965

The requirements for test effort on the APOLLO G&N system are indexed; responsibilities are assigned for completing and reporting regularly.

R-389: REQUIREMENTS OF AND INDEX TO DESIGN EVALUATION,
Rev. B QUALIFICATION, AND RELIABILITY TEST PROGRAM FOR APOLLO GUIDANCE AND NAVIGATION SYSTEM
James E. Kent / March 1964

The requirements for test effort on APOLLO G&N system are indexed; responsibilities are assigned for completing and reporting regularly.

R-389: REQUIREMENTS OF AND INDEX TO DESIGN EVALUATION, QUALIFICATION, AND RELIABILITY TEST PROGRAM FOR APOLLO GUIDANCE AND NAVIGATION SYSTEM
George W. Mayo, Jr., Alan R. Schweitzer / March 1963

The requirements for test effort on APOLLO G&N system are indexed; responsibilities are assigned for completing and reporting regularly.

R-368: RELIABILITY HANDBOOK FOR ELECTRONIC ENGINEERS
George W. Mayo, Edward T. Driscoll / August 1962

The Reliability Group has prepared this handbook to enable the engineer to make preliminary reliability calculations during the early planning when changes in parts, application, etc. can most easily and economically be made. Only curves and data are used that permit a quick reliability estimate and help in evaluating trade-offs that would improve reliability. A study of the curves shows the most advantageous combination of parts, stress ratios, and operating temperature for a particular situation.

R-349: GUIDANCE AND NAVIGATION SYSTEM: RELIABILITY
Rev. D AND QUALITY PROGRAM
George W. Mayo / July 1965

The APOLLO G&N system Reliability and Quality Program generates the necessary detailed procedures, instructions, and controls to establish an appropriate organization and to assign responsibilities for insuring that:

1. The guidance and navigation system will meet or exceed required reliability specifications.
2. The attainment of reliability requirements will be within the time specified for the system and its subassemblies.
3. Appropriate effort is expended to achieve reliability requirements in the design phase, and proper action is taken to insure its retention in manufacturing and use phases.

E-2377: LM DIGITAL, AUTOPILOTS SIMULATION RESULTS USING PROGRAM SUNDANCE
William S. Widnall / January 1969

The DAP delivered in Program SUNDANCE has been subjected to exhaustive testing by means of digital and hybrid simulation. The SUNDANCE DAP differs in many ways from the DAP previously delivered in program SUNBURST. The most noticeable differences are:

1. The SUNBURST DAP was designed only for unmanned flight, but the SUNDANCE DXP contains the basic manual-mode capabilities.
2. A capability of using the LM to control the CSM-docked configuration has been added in SUNDANCE.
3. In spite of added capabilities, the SUNDANCE autopilot design is simplified, and the coding is signficantly shorter than the SUNBURST coding.

E- 2353: DIGITAL AUTOPILOT VERIFICATION RESULTS PROGRAM COLOSSUS I (REV. 237)
COLOSSUS I (REV. 237)
MIT/IL / November 1968

Verification testing was performed on the digital autopilots for the APOLLO CSM in order to demonstrate suitable operation for an earth-orbital and lunar mission (Program COLOSSUS 237). The COLOSSUS autopilots—essentiallythe same as earlier-tested SUNDISK autopilots (R-596, Vol. IV: “Digital Autopilot Test Results,” AS-205 Verification Results, Program SUNDISK, February 1968)—were verified in the following way:

1. All design characteristics either new or modified since SUNDISK were fully tested, using all-digitai and real-time hybrid simulations, to verify the coding and to measure performance.
2. Additional tests were performed in areas of performance not tested during the SUNDISK testing.
3. COLOSSUS Level III testing was examined to verify the DAP software.

Subject to further study regarding bending limitations in the CSM/LM configuration (paragraphs 1.3E and 2.2E), the results reported show that the DAP can perform as described in Report R-577, Guidance System Operations Plan for Manned CM Earth Orbital and Lunar Missions Using Program COLOSSUS, Section 3, "Digital Autopilots," Rev 1 (June 1968).

E-2194: HYBRID SIMULATION OF CSM DYNAMICS FOR AS-205
Joseph T. O'Connor / October 1967

The hybrid simulation of the APOLLO spacecraft for mission AS-205 is described. Dynamic equations are presented and data sources are identified. All analog simulation diagrams are included. The associated digital computer program for spacecraft dynamics appears in flow chart form.

E-2176: CSM EARTH ORBITAL, RENDEZVOUS SIMULATIONS
E. Muller, P. Kachmar, N. Sears / August 1967

Presented are simulations evaluating the G&N concepts for CSM earth orbital rendezvous phases of the APOLLO 8 (258) and APOLLO 7 (205) missions. Most of the results were presented at CSM rendezvous meetings at M.I.T. on 6 June 1967 and at MSC on 21 July 1967.

E-2160: VERIFICATION RESULTS SUMMARY FOR LGC STJNBURST (LM-1 MISSION)
Jay A, Sampson (AC Electronics) / August 1967

A description and short summary is presented of the all-digital verification runs performed to qualify the LM- 1 Guidance Computer Program, BURST 116. Individual runs were made of each mission phase: under nominal conditions, with vehicle perturbations, excessive LGC restarts, ignition failures, premature engine shutdowns, and termination commands from the ground. In addition, three end-to-end runs were made from prelaunch through mission phase 13 to verify the protection of the erasable memory load and the integration of the various mission phases under nominal conditions and with prelaunch stable member misalignments. Finally, some miscellaneous runs were made with both critical and non-critical RCS jet failures, a trim gimbal failure, an inflight fresh start recovery, SATURN IVB excess tumbling, and sequences of both legal and illegal uplink commands and updates,

Since not all runs used the final version of the fiight program (SUNBURST Revision 11.6 or BURST 116 Revision 0, which are identical), a revision number is included for each run.

E-2146: THE DIGITAL SIMULATION FOR THE VERIFICATION OF PROGRAM SUNBURST (UNMANNED LM, AS-206)
W.S. Widnall, et al. /July 1967

The digital simulation used for development and verification of LGC mission program (AS-206) is described, including the organization of the simulated environment and descriptions of the simulator component subroutines.

E-2106: SINGLE AXIS BLOCK II CSM/TVC: PRE-SIMULATION REPORT
Joseph T. O'Connor / April 1966

A real-time hybrid simulation of a single axis of the thrust-vector control portion of the CSM G&C system is described. The attitude-control problem using the RCS is not considered. The mathematical models of the system dynamics and of the generated mass functions are discussed. Analog diagrams and the digital program flow chart are presented. An associated problem-the reai-time simulation of the PIPA—is discussed.

E-2081: MAN-MACHINE SIMULATIONS FOR THE APOLLO NAVIGATION, GUIDANCE, AND CONTROL SYSTEM
J.L. Nevins, E.A. Woodin, R.W. Metzinger / January 196'7

This paper describes the APOLLO G&N system, defines the man-machine interfaces and design philosophy, and discussesman-machine simulations and design highlights.

E-2079: LM DYNAMICS SIMULATION: PRE-SIMULATION REPORT
J.D. Goetzinger / September 1966

Simulation of the LM to be programed at MIT/IL hybrid computation facilities is described.

This paper analyzes the simulation and the computer programs. Program techniques and a complete development of motion equations are included. The analog portion of the program is presented in full; the digital portion is presented at the level of logic flow diagrams.

E-2066: HYBRID SIMULATION OF THE APOLLO GUIDANCE NAVIGATION AND CONTROL SYSTEM
Philip G. Felleman / December 1966

The purpose of this hybrid simulation laboratory is to verify APOLLO G&N programs in a real-time simulation using flight hardware. With these simulators, crew procedures are examined and programs can be modified to allow a better interface between the flight crew and the G&N system. The simulator also becomes a task trainer with the dddition of a cockpit mockup.

The facility uses an analog-digital, general-purpose computer and inflight hardware such as the onboard computer, coupling data units, and special purpose simulators for the IMU dnd accelerometers, The facility is designed to allow rapid changeover from Block I to Block II simulations.

E-1923: APOLLO MISSION 202: DYNAMIC SIMULATION BLOCK I
Madeline M. Sullivan, James F. Cutter / January 1966

APOLLO Mission 202 is simulated using actual G&N hardware and an analog computer. The nominal mission from 609.96 secs after liftoff to CM-SRI separation is presented in its entirety. SATURN IVB tumble abort cases begin at 609.96 secs after liftoff, but are terminated after tumble-arrest is either secured or assured.

E-1922: AGC SOFTWARE VERIFICATION (MISSION AS-202): SUMMARY OF RESULTS OF DIGITAL SIMULATIONS
Albrecht L. Kosmala / February 1966

A set of all-digital simulations performed as verification of the AS-202 AGC program are presented. The run description and sequence, and the summary format, are specified as the first priority runs in Reference 1, with three extra runs not detailed. The AGC program used in the simulation is the first flight-released program of Reference 2, approved for manufacture by the MSC Software Control Panel, and released on 11 January 1966. Trajectory and spacecraft data used to generate the environment for the AGC program in the simulation are those defined in References 3 and 4.

E-1864: DYNAMIC SIMULATION BLOCK I (APOLLO MISSION 202): PRESIMULATION REPORT
Madeline M. Sullivan / September 1965

The dynamic simulation of the APOLLO Block I G&N system is principally comprised of the mission timeline, the dynamic equations used to simulate the APOLLO vehicle, and the G&N hardware. The integration of the entire system is shown.

R-599: DIGITAL SIMULATION MANUAL
Pieter R. Mimno / January 1968

The manual is intended both as an introduction to the digital simulator and as a reference document for simulator users. Included are an introduction to the simulator, a description of the structure and operational modes of the simulator, a summary of the models used to simulate the spacecraft and its environment, and acomplete user's guide to simulation input cards, rollbacks, and edits.

R-525: HYBRID SIMULATION OF THE APOLLO GUIDANCE AND NAVIGATION SYSTEM
Madeline M. Sullivan / December 1965

A hybrid simulation of the APOLLO G&N system at MIT/IL combines analog and digital computers with the hardware in order to subject it to dynamic conditions approaching those found in a space environment. APOLLO G&N instrumentation system adaptation to the hybrid simulation is described. Software models are discussed to indicate some of the varied problems considered. Digital procedures designed to support the operator and enhance the reliability of the simulation are described. It becomes apparent that the judicious use of computers, together with G&N hardware and software, makes it possible to simulate in real time any part of the APOLLO mission from liftoff to splashdown.

APPENDIX B

BIBLIOGRAPHY

This bibliography consists of engineering and progress reports (E series), research reports (R series), and theses (T series). These four types are indexed by number in a general chronological order in each category. Some of these documents are abstracted in Appendix A.

TECHNICAL PROGRESS REPORTS

No.	Type	
E-1067	Monthly	11 August through 13 September 1961
E-1068	Monthly	13 September through 4 October 1961
E-1099	Monthly	4 October through 9 November 1961
E-1116	Quarterly	Period ended 11 December 1961
E-1117	Monthly	11 December 1961 through 16 January 1962
E-1139	Monthly	1.6 January through February 1962
E-1140	Quarterly	Period ended 11 March 1962
E-1157	Monthly	11 March through 11 April 1962
E-1177	Monthly	11 April through 1 May 1962
E-1199	Quarterly	Period ended 11 June 1962
E-1236	Monthly	11 June through 17 July 1962
E-1237	Monthly	17 July through 21 August 1962
E-1238	Quarterly	Period ended 11 September 1962
E-1302	Monthly	11 September through 11 October 1962
E-1303	Monthly	11 October through 13 November 1962
E-1304	Quarterly	Period ended 11 December 1962
E-1305	Monthly	11 December 1962 through 11 January 1963
E-1306	Monthly	11 January through 11 February 1963
E-1307	Quarterly	Period ended March 1963
E-1378	Monthly	May 1963
E-1389	Quarterly	Period ended June 1963
E-1410	Monthly	July 1963
E-1445	Monthly	August 1963
E-1474	Quarterly	Feriod ended September 1963
E-1494	Monthly	October 1963
E-1495	Monthly	November 1963
E-1496	Quarterly	Period ended December 1963
E-1534	Monthly	January 1964
E-1544	Monthly	February 1964
E-1573	Quarterly	Period ended March 1964
E-1590	Monthly	April 1964
E-1609	Monthly	May 1965

K-1670	Quarterly	Period ended June 1964
E-1718	Monthly	July 1964
E-1730	Monthly	August 1964
E-1749	Quarterly	Period ended September 1964
E-1761	Quarterly	Period ended December 1964
E-1789	Quarterly	Period ended March 1965
E-1815	Quarterly	Period ended June 1965
E-1872	Quarterly	Period ended September 1965
E-1921	Quarterly	Period ended December 1965
E-1958	Quarterly	Period ended March 1966

ENGINEERING REPORTS

E-1054	J. Rocchio, Analysis of the PNP and NPN Latch Circuit, September 1961
E-1061	AGANI, October 1961
E-1074	D. Shansky, Erasable Ferrite Memory, MOD 3C Computer, October 1961
E-1077	R. Alonso, J.H. Laning, Jr., and H. Blair-Smith, Preliminary MOD 3C Computer Program, November 1961
E-1078	Development Plan, November 18, 1961
E-1087	Documentation Handbook, May 1963 (Rev. 5, October 1966)
E-1091	E.J. Hall, Assembly Procedures and Specifications for Apollo 25 IRIG, June 1962 (Rev. Aug. 1963)
E-1092	J.H. Flanders, IMU Preliminary Operational Duty Cycle, December 1961
E-1097	Work Statement for Industrial Support, January 1962, Addendum, February 1962
E-1100	M. Sappupo, Work Statement for PIP MOD D, 15 December 1961
E-1101	E.J. Hall, Work Statement for Apollo 25 IRIG, 15 December 1961
E-1103	E.J. Hall, Assembly Procedures and Specifications for 25 IRIG MARK 45 MOD 2, 15 December 1961
E-1104	M. Sappupo, Preliminary Assembly and Test Manual 16 PIP MOD D, January 1962
E-1105	Eldon Hall, Computer Displays, January 1962
E-1106	P.G. Felleman, Analysis of Guidance Techniques for Achieving Orbital Rendezvous, January 1962
E-1109	D.G. Hoag, Slide Reprints Used in Support of Apollo Presentation at North American, January 1962
E-1112	M.B. Trageser, Slide Reprints from Apollo Monthly Meeting, January 1962
E-1113	System Identification Data List,, (SIDL), issued 20 August 1962 (to be revised and supplemented monthly)
E-1114	J.E. Levy, Glossary of Terms and Symbols, April 1962 (Revised July 1962)
E-1118	R. A. Scholten and P. J. Philliou, Investigation of Midcourse Maneuver Fuel Requirements, March 1962
E-1123	D.G. Hoag, Systerns and Component Slide Reprints, February 1962
E-1124	J.S. Miller and J.J. Deyst, Jr., Preliminary Study of Aborts from Circumlunar Trajectories, March 1962
E-1125	R. L. Alonso, Preliminary Resolver Angle Measurements, February 1962
E-1126	A. L. Hopkins, Jr., AGC MOD 3C Computer Circuits-General, February i962
E-1130	J.E. Levy, Maintenance Plan, March 1962

E-1131 J.S. Miller and R.H. Battin, Preliminary Summary of Data for a Variety of Circumlunar Trajectories, February 1962

E-1134 A.C. Hardy, Photometric Units in the MKS System, March 1962

E-1135 G. Oberbeck, The MYF Torque Generator for Pulse Torqued Instruments, March 1962

E-1142 J.E. Levy, System Status Report, 15 March 1962, Revised July 1962 (to be revised monthly)

E-1144 J.E. Levy, Apollo Guidance and Navigation Interfaces (to be published)

E-1158 D. Shansky, Erasable Store MOD 3C, July 1962

E-1161 I. Halzel, Technical Directive Procedures for Apollo G&N System Participating Contractors, May 1962

E-1164 F.D. Grant, The Relationship between IMU Drift Misalignments and Target Errors, May 1962

E-1167 Drawing Standards, 15 June 1962 (Rev. 1, June 1964)

E-1168 M.B. Trageser, Slide Reprints - Presentation to Participating Contractors, Houston, June 1962

E-1172 A.C. Hardy, The Visibility of Stars, June 1962

E-1173 R.M. Jansson, The Heat Transfer Properties of Structual Elements for Space Environment, June 1962

E-1174 P.K. Bryant, Slide Reprints - Quarterly Management Review Meeting, Houston, June 1962

E-1179 R.G. Scott, Rope Core Tester, July 1962

E-1186 Technical Data Release Procedures, August 1962 (Rev. 5, August 1966)

E-1195 N.E. Sears, Earth Orbital Rendezvous, May 1962

E-1196 M.W. Johnston, Analysis of Two Lunar Landing Techniques Providing Direct Landing Site Visibility Prior to Touchdown, July 1962

E-1203 G.A. Davidson and H.H. Seward, Calibration Techniques for Precision Rotary Components, August 1962

E-1206 H.H. Seward, The Blue-White Boundary Horizon Sensor, September 1962

E-1212 F.D. Grant, Error Data Summaries for Various Trajectories, (Preliminary), September 1962

E-1215 T.C. Taylor, Thermal Models for High Density Computer Circuit Structures, September 1962

E-1230 J.H. Flanders, IMU Alignment Test, (Preliminary) February 1963

E-1233 T.C. Taylor, Technique of Circuit Fabrication with State-of-the-Art Weldable Multilayer Boards, October 1962

E-1250 Astronauts' G&N Course Notes

Section I, D.G. Hoag, Functional View of the Apollo G&N System, December 1962

Section II, J. Hovorka, Gyro Principles, December 1962

Section III, J. Hovorka, Stabilization, January 1963

Section IV, J. Sciegienny, Electromagnetic Navigation, December 1962

Section V, R.H. Battin, Midcourse Guidance and Navigation, February 1963

Section VI, D.J. Lickly, Re-entry Guidance, March 1963

Section VII, R.J. Magee, Optics, February 1963

Appendixes, M.B. Trageser, Slide Reprints, December 1962

E-1253 C. Stark Draper, Slide Reprint of MSC Guidance System Review at Houston, November 1962

E-1256 J.W. Hursh, Apollo Midcourse Guidance, November 1962

E-1257 L.J. Lareau, Atmospheric Refraction as a Means of Horizon Determination, December 1962

E-1258 M.B. Trageser, Slide Reprints (Shown to Astronauts at MIT in Cambridge) December 1962

E-1261 G.M. Levine, Application of Midcourse Guidance Techniques to Orbit Determination, December 1962

E-1270 E. Driscoll, W.J. Beaton, Glossary of Important Reliability Terms, December 1962

E-1278 G.R. Gilbert, MIT Space Implementation, Interim Report, January 1963

E-1285 A.C. Hardy (Consultant), Some Luminance Values for Sun, Earth, and Moon, January 1963

E-1287 J.M. Dahlen and J.H. Long, Backup Thrust Vector Control, February 1963

E-1288 F.D. Grant, Alignment Errors of the IMU Stable Member, February 1963

E-1293 R. Boyd, AGE Field Operations Program Plan, April 1965 (Rev. 1) Field Operations Program Requirements, March 1965

E-1297 J.H. Flanders, Inertial Subsystem (ISS) Calibration Tests, (Preliminary), June 1963 (Rev. February 1964)

E-1313 T.C. Taylor, Thermal Grounding Analysis for Circuit Structures, April 1963

E-1322 G.W. Mayo, Jr., Apollo G&N Failure Reporting and Corrective Action Plan, April 1963 (Rev. I, March 1965)

E-1329 Apollo 16 PIPA Test Console Reference Manual, December 1963

E-1330 Apollo 16 PIPA Test Console, Job Description Cards for Operation and Maintenance 1963, August 1963

E-1333 C.L. Pu, Cloud Cover Data for Landmark Studies, March 1963

E-1343 G. Oberbeck, Ternary Pulse Torquing, May 1963

E-1344 D.G. Hoag, Consideration of Apollo IMU Gimbal Lock, April 1963

E-1346 C. Stark Draper, M.B. Trageser, D.G. Hoag, R.H. Battin, Slide Reprints of Presentation at NASA Headquarters, Washington, D.C., April 1963

E-1350 W.G. Briggs, Statistical Decision Theory for Logistics Planning, May 1963

E-1353 E.M. Copps, et al., The Horizon Photometer and other Earth Parking Orbit G&N Measurements, May 1963

E-1359 J.M. Dahlen, T. Heinsheimer, J.B. Suomala, Flight Test Plan - Apollo G&N System AGE-5, May 1963

E-1363 T. Heinsheimer, Status of Apollo Flight Safety System Design and Development, June 1963

E-1374	J. Sciegienny, Propagation of Altitude and Altitude Rate Errors During Suborbital Flight, June 1963
E-1381	R. Boyd, Data Analysis and Dissemination Instruction, (to be published)
E-1385	A.C. Hardy, Visibility Data and the Use of Optical Aids, July 1963
E-1386	S.C. Smith, Report on Clear Resins, July 1963
E-1388	J. Sciegienny, Summary of Error Propagation in an Inertial System, August 1963
E-1395	J. McNeil, Date Console Job Description Cards for Operation and Maintenance, (to be published)
E-1398	J.M. Dahlen, J.B. Suomala, Comments on the Lunar Landing Mission Design Plan of 15 April 1963, August 1963
E-1413	J. Partridge and A.J. Borofsky, Some Reversible and More Permanent Effects of Moisture on the Electrical Characteristics of Germanium Transistors following Stress, September 1963
E-1420	J. Partridge, On the Extrapolation of Accelerated Stress Conditions to Normal Stress Conditions of Germanium Transistors, November 1963
E-1421-1	MIT/IL, Slide Reprint Working Copy, Reference Reprints, March-May 1963, February 1964 (Rev. 2, June-September 1963)
E-1425	C.C. Perez, A Comparison of the Readout Resolution of the Proportional Elastance vs Constant Elastance Torque-to-Balance System, September 1963
E-1429	D.S. Baker, N.E. Sears, J.B. Suomala, R.L. White, Lunar Orbit Determination by Star Occultations and MSFN Tracking, September 1963
E-1432	GSE Familiarization Manual, January 1964 (Rev. 2, November 1964)
E-1467	E.K. Bender, Sampled-Data Velocity Vector Control of a Spacecraft, March 1964
E-1473	J.M. Dahlen, M.W. Johnston, Analysis of LEM Mission Inertial Uncertainties, December 1963
E-1482	F.D. Grant, Inflight Alignment Errors of the IMU Stable Member, December 1963
E-1492	G.A. Davidson, A.C. Hardy, On the Perception of Flashes of Light at the Limit of Their Perceptibility, January 1964
E-1522	R. Booth, Assembly Procedures and Specification for Apollo 11 IRIG, March 1964
E-1524	S.C. Smith, Evaluation of Eurethane Foam for Potting, February 1964
E-1539	T.C. Taylor and T.A. Zulon, Thermal Properties of Some Structural Members for Spaceborne Computer Assemblies, March 1964
E-1540	K. Nordtvedt, A Preliminary Study of a Backup Manual Navigation Scheme, August 1964
E-1541	J.H. Flanders and R.A. McKern, Test Results on Apollo Inertial Subsystem, No. 4, April 1964
E-1547	P.H. Bowditch, D.A. Koso, D.A. Mudgett, Optical Equipment Handling Guidelines, March 1964
E-1548	R.W. Gras, Brazing of Beryllium, March 1964
E-1551	A.R. Klumpp and R.A. Larson, Function and Mechanization of Apollo G&N System (Vol. 1 Inertial Subsystem), August 1964

E-1557	W. Nadler, Average Power Profiles for Apollo G&N in the Command Module and Lunar Excursion Module, April 1964
E-1560	M.W. Johnston, A Manual LEM Backup Guidance System, April 1964
E-1574	A.I. Green and J.J. Rocchio, Keyboard and Display System Program for AGC Program SUNRISE, August i964
E-1584	R.J. Filene, Digital to Analog Conversion of Base Minus Two Numbers and Application to Threshold Logic, June 1964
E-1588	R.T. Boyd, Jr., Gerald C. Hines, Apollo G&N Site Activation Plan, June 1964
E-1621	Warren G. Briggs, A New Procedure for Initial Provisioning of Spares, July 1964
E-1634	Milo Wolff, The Profile of an Exponential -Atmosphere Viewed from Outer Space and Consequences for Space Navigation, September 1964
E-1660	J. McNeil, AIDS (Abort Inertial Digital System), November 1964
E-1663	A.R. Klumpp, A Manual Abort Guidance Procedure Based on Range and Altitude Measurements, October 1964
E-1679	J. Partridge, L.D. Hanley, E.C. Hall, Progress Report on Attainable Reliability of Integrated Circuits for Systems Application, November 1964
E-1687	Milo Wolff, An Optical Earth Horizon Profile Based Upon Tabulated Solutions of Chandrasenkhar's Equations, October 1964
E-1688	H.H. Cho, Sampling Reliability Test Plans with Relatively Small Lot Sizes, December 1964
E-1699	T.C. Taylor, A Propose-d Circuit Structure for Computer Logic Based on Semiconductor Flat Packs Interconnected by Multilayer Circuit Boards, December 1964
E-1713	A.O. Mirarchi, Combustion of Polyolefin Insulation and Toxicity of Teflon under Electrical Overloads, December 1964
E-1734	C.M. Adams and S. Weiss, Brazing of Beryllium Final Report on DSR Project 9909, February 1967
E-1750	D.C. Dilworth, Computer Programs for Optical System Analysis and Design, February 1965
E-1758	T.J. Lawton and C.A. Muntz, Organization of Computation and Control in the Apollo Guidance Computer, April 1965
E-1759	E.K. Bender, Gyro Temperature Dependent Drift, August 1965
E-1760	B.A. Kriegsman and D.S. Millard, Performance Monitoring of the PGNS for Unpowered Lunar Orbit Flight Phases, September 1965
E--1769	W.E. Tanner, Typical Missile and Satellite Radar Tracking Systems, April 1965
E-1783	M.B. Trageser, Slide Reprints of MIT/IL Presentation to Presidential Scientific Advisory Committee, May 1965 (Rev. 1) May 1965
E-1786	T.J. Lawton and C. A. Muntz, Verification Plan for AGC/ LGC Programs, May 1965
E-1792	G.J. Reichenbacher, 16 PIP MOD D Centrifuge Test, August 1965
E-1800	D.C. Fraser, A Sequenceof Computer Programs Useful in the Analyzing of Feedback Control Systems, July 1965

E-1801 J.D. Fleming, Jr., and R.T. Boyd, Apollo GN&C Equipment Contingency Plan (Apollo Ground Checkout), February 1966

E-1802 T.C. Taylor, Characteristics of Soft Metals for Electrical Connection, June 1965

E-1803 J. Sciegienny, System Analysis of Lunar Landing Radar, June 1965

E-1808 A. LaPointe, Rope Memory Module Assembly Processing Procedures, June 1965

E-1809 G.D. Shepard, Coupled Two-Degree-of-Freedom Response of Resiliently Supported Rigid Body to Random Excitation, December 1965

E-1810 J. Sciegienny, Analysis of Landing Radar Geometry during Redesignation of Lunar Landing Site, July 1965

E-1815 MIT/IL, Project APOLLO Guidance and Navigation Program: Quarterly Technical Progress Report, June 1965

E-1819 J. Gallagher and J.M. Hallock, A Lunar Photometric Function: Curves for Calculating the Differential Luminance of the Lunar Surface, July 1965

E-1823 T.C. Taylor, The Design of Series-Loaded Thermal Conductors for Minimum Weight, July 1965

E-1828 MIT/IL, G&N System Data for Mission A.S-202 Development Engineering Inspection, August 19 65

E-1832 R. Crisp and D.W. Keene, Attitude Maneuver Optimization to Conserve Reaction Control Propellants, August 1965

E-1836 K. Nordtvedt, Studies of Spacecraft Attitude Change Modes and Resulting Attitude Fuel Consumption, August 1965

E-1838 L.D. Hanley, J. Partridge, and E.C. Hall, The Application of Failure Analysis in Procuring and Screening of Integrated Circuits, October 1965

E-1855 H.E. Maurer, Pulsed Light Source Simulation of Transient Nuclear Radiation Effects In Integrated Circuits, October 1.965

E-1859 T.C. Taylor, Design Principles for a Multilayer Circuit Board Connector Using Soft Metal Contacts, October 1965

E-1864 M.M. Sullivan, Presimulation Report - Apollo Mission 202 Dynamic Simulation Block I, September 1965

E-1877 E.D. Smally, Show-Banksum and Final Block I AGC Selfcheck, November 1965

E-1880 Eldon C. Hall, A Case History of the AGC Integrated Logic Circuits, December 1965

E-1899 J. L. Nevins, Apollo Operations Handbook - G&N System - Command and Service Module Mission AS-204B, Spacecraft 012, June 1966

E-1901 K. L. Sapf, Visibility of the LM with Various Background Luminances, January 1966

E-1903 N.E. Sears and L.B. Johnson, PGNS Rendezvous Radar Functional and Performance Specifications, March 1965

E-1904 N.E. Sears and L.B. Johnson, PGNS Landing Radar Functional and Performance Specifications, February 1965

E-1905 A.I. Green, Keyboard and Display Program and Operations, January 1966

E-1906 J.N. Hallock, Fourier Anaiysis by the Method of Selected Ordinates, January 1966

E-1922 A.L. Kosmala, Mission AS-202, AGC Software Verification, Summary of Resuits of Digitai Simulation, February 1966

E-1923 M.M. Sullivan and James F. Cutter, Apollo Mission 202 Dynamic Simulation Block I, 3 anuary 1966

E-1926 H.E. Maurer, A Design Procedure for Radiation Hardening Certain Nor Gate Logic Circuits, March 1966

E-1937 J. Sciegienny, Velocity Errors in Lunar Landing Radar Caused by Digital Processing, March 1966

E-1942 M. Lanman and J. Feldman, Command Angle Torquing of Gyros in the Apollo Guidance and Navigation System, April 1966

E-1944 J. Partridge and L. D. Hanley, The Impact of the Flight Specification on Semiconductor Failure Rates, June 1967

E-1956 J.C. Dunbar, R.A. Larson, and P.T. Augart, An Automated Documentation Technique for Integrated Apollo Crew Procedures and Computer Logic, May 1966

E-1964 R. Crisp and D.W. Keene, Apollo Command and Service Module Reaction Control by the Digital Autopilot, April 1966

E-1970 E.C. Hall, Case History of the Apollo Guidance Computer, June 1966

E-1971 E.C. Hall, Reliability Assessment of Microcircuits and Microelectronic Systems, June 1966

E-1973 L.D. Brock and G.T. Schmidt, Statistical Estimation in Inertial Navigation System, June 1966

E-1974 D. McCabe, Apollo 16 PIP Test Console Verification Test, June 1966

E-1975 J. Wolf, Simulated LM Inflight IMU Alignment Sightings, June 1966

E-1976 W.E. Tanner and W. Saltzberg, Radar Interface Evaluation Phase I Final Test Report, May 1966

E-1978 MIT/IL, Block II/LM Optics Design Evaluation Interim Report, June 1966

E-1981 N.E. Sears and P.G. Hoffman, LM-PGNCS Guidance Equations for a Nominal Lunar Landing Mission, May 1966

E-1982 B.A. Kriegsman and N.E. Sears, LM PGNCS and Landing Radar Operations During the Powered Lunar Landing Maneuver, June 1966

E-1983 D.S. Baker, N.E. Sears, and R.L. White, Lunar Orbit Navigation Performance With Various Random and Systematic Errors, July 1966

E-1989 A.I. Green and R.J. Filene, Keyboard and Display Program and Operation, July 1966

E-2000 J. L. Nevins, Apollo Operations Handbook G&N System Command and Service Module Mission AS-205A, Spacecraft 014, August 1966

E-2014 A.H. Arpiarian, Temperature Control of the Apollo Block II Inertial Measurement Unit, August 1966

E-2026 J.D. Wood, Users's Guide to the AGC Monitor Core Rope Simulator, September 1966

E-2027 S.C. Smith, Plastics (What to Use), September 1966

E-2028 J.H. Barker, Multiplexed Electronic Coupling Data Unit, November 1966

E-2031 R. Morth, Entry Flight Data for Mission AS-202, October 1966

E-2034 E.K. Gardner, LM Alignment Optical Telescope Mechanical Integrity Design Evaluation Final Report, September 1966

E-2051 W.A. Vachon, Design Evaluation Thermal-Vat Tests Final Report, November 1966

E-2052 B.I. Savage and A.L. Drake, AGC-4 Basic Training Manual, January 1967

E-2055 J.R. Lawson, Data Acquisition and Reduction MIT X-15 Horizon Definition Experiment Phase II, December 1966

E-2065 E.D. Smally, Block II AGC Self-Check and Show-Banksum, December 1966

E-2066 P.G. Felleman, Hybrid Simulation of the Apollo Guidance Navigation and Control System, December 1966

E-2079 J.D. Goetzinger, Pre-Simulation Report: LM Dynamics Simulation, September 1966

E-2081 J. L. Nevins, E. A. Woodin and R.W. Metzinger, Man-Machine Simulation for the Apollo Navigation, Guidance, and Control System, January 1967

E-2092 W.H. Aldrich, R.L. Alonso, A.L. Hopkins, Summary Report Braid Memory, February 1967

E-2095 T.A. Zulon, Evaluation Report on Multilayer Circuit Board Connectors Using the Soft Metal Indium for Electrical Contacts, March 1967

E-2097 R.L. Alonso, A.L. Hopkins, Jr., H. A. Thaler, A Multiprocessing Structure, March 1967

E-2099 D. Shansky, Erasable Memory Investigations, March 1967

E-2106 J.T. O'Connor, Presimulation Report Single Axis Block II CSM/TVC, April 1966

E-2119 MIT/IL, Critique of the Apollo Guidance System Design, May 1967

E-2128 D. Kadish, A Proposed Multilayer Plated Wire Device with Possible Improvements in the Non-Destructive Properties of a Plated-Wire Memory, May 1967

E-21 29 A.I. Green and R. J. Filene, Keyboard and Display Program and Operation, June 1967

E-2141 J.E. Miller and J. Feldman, Apollo Guidance, Navigation and Control System Gyro Reliability, June 1967

E-2146 W.S. Widnall, et al., The Digital Simulation for the Verification of Program Sunburst (Unmanned LM, AS-206), July 1967

E-2150 Guidance, Navigation, and Control Block II Command and Service Module Functional Description and Operation Using Flight Program SUNDISK, April 1968

E-2159 F.H. Martin and R.H. Battin, Computer-Controlled Steering of the Apollo Spacecraft, August 1967

E-2160 J.A. Sampson (AC Electronics), Verification Results Summary for LGC SUNBURST (LM-1 Mission), August 1967

E-2164 Guidance, Navigation, and Control Lunar Module - Using Flight Program SUNDANCE, Vols. I & II, July 1968

E-2172 J. Barnes, D. Beren, and 4. Glaser (Allied Research Associate, Inc.) Cloud Obscuration of Apollo Landmarks Derived from Meterological Satellite Observations, September 1967

E-2176 E. Muller, P. Kachmar, and N. Sears, CSM Earth Orbital Rendezvous Simulation, August 1967

E-2185 W.E. Tanner, Landing Radar/ LGC Interface Experiments, Part II Flight Test Data Package, September 1967

E-2194 J.T. O'Connor, Wyhrid Simulation of CSM Dynamics for AS-205, October 1967

E-2204 R. L. Alonso, Algebraic -Analysis and Modeling of Sequential Circuits, November 1967

E-2210 Quarterly Progress Report for period ending 31 October 1967 - Apollo Applications Program Inertial Measurement Unit, November 1967

E-2214 H.H. Seward and I.G. McWilliams, A Study for a Calibration Detector for Ultraviolet Measurement, January 1968

E-2215 M.M. Yovanovich, Analytical and Experimental Investigation of the Thermal Resistance of Angular Contact Instrument Bearings, December 1967

E-2218 W.J. Day and J. Partridge, Lead Failure Study for the Motorola-l MIL Wedge-Bonded 1006323 Transistor, December 1967

E-2220 J. Lawson, O. Anderson, R. Newell, and J. Geishler, MIT Aeronomy Experiment, February 1968

E-2245 Quarterly Progress Report for Period Ended 28 February 1968 - Apollo Applications Program Inertial Measurement Unit, February 1968

E-2246 J. A. Hand, Computer-Aided Inertial Platform Realignment in Manned Space Flight, February 1969

E-2252 J. Feldman and R. Cooper, Inertial Component Reliability and Population Statistics, March 1968

E-2254 D. J. Bowler, Auxiliary Memory System Final Report on Phase 1, April 1968

E-2260 Guidance, Navigation, and Control Lunar Module - Using Flight Program LUMINARY, Vol. II, May 1968

E-2262 D.S. Baker, Studies of On-Board Lunar Orbital Navigation with Unknown and Known Landmarks and Some Observations on Non-Linear Effects, August 19 68

E-2270 A. Grossman, LM- Alignment Optical Telescope Thermal-Humidity Integrity Evaluation Report, June 1968

E-2271 J. Harper, Quarterly Progress Report for Period Ended 31 May 1968 - Guidance and Navigation System Improvement Program, June 1968

E-2274 J. Feldman and R. Cooper, Inertial Components Reliability and Population Statistics: Report II, June 1968

E-2279 G.J. Bukow, Prealignment of the 16 PIP MOD D, February 1968

E-2280 L.D. Hanley, Solid State DSKY Study, June 1968

E-2287 F.D. Grant and J.E. Miller, Mission C (AS-205/CM101) G&N Error Analysis, September 1968

E-2299 Reliability and Quality Assurance Program Plan for Procurement of Apollo IRIG, December 1968

E-2301 F.D. Grant, G&N Error Analysis for Mission D, July 1968

E-2305 J. L. Nevins, I. Johnson, and R. Sheridan, Reliability and Quality Assurance Program Plan for Procurement of **Apollo** IRIG, June 1968

E- 2307 E.M. Copps, Jr., Recovery from Transient Failures of the APOLLO Guidance Computer, August 19 68

E-2308 J.P. Gilmore, A Non-Orthogonal Multi-Sensor Strapdown Inertial Reference Unit, August 1968

E-2333 M. Landey, Inertial Components Reliability and Population Statistics Report III, December 1968

E-2334 G. Bukow, The PIPA (Pulsed Integrating Pendulous Accelerometer) September 1968

E-2345 R.C. Millard, Software Configuration Management Plan, October 1968

E-2353 MIT/IL, Digital Autopilot Verification Results: Program COLOSSUS I (Rev 237), November 1968

E-2364 J. Feldman and S. Helfant, Inertial Components Reliability and Population Statistics: Report IV, January 1969

E-2377 W.S. Widnall, LM Digital Autopilot Simulation Results Using Program SUNDANCE, January 1969

E-2387 F.D. Grant, J.E. Miller, E.S. Muller, Mission D (AS-504/CM104/LM3) G&N Error Analysis, February 1969

E-2389 J. A. Hand, Star-Horizon Measurements for Onboard Spacecraft Navigation, February 1969

E-2401 R.H. Battin and G.M. Levine, Application of Kalman Filtering Techniques to the APOLLO Program, April 1969

E-2404 R. Morth, Entry Flight Data from APOLLO 7, April 1969

E-2405 R. Morth, Entry Flight Data from APOLLO 8, April 1969

E-2406 G.T. Schmidt and L.D. Brock, General Questions on Kalman Filtering in Navigation Systems, April 1969

E-2411 D.G. Hoag, APOLLO Guidance, Navigation, and Control Systems: A Progress Report, April 1969

E-2414 F.D. Grant, J. Werner, E.S. Muller, Mission F (AS-505/CM106/LM4) G&N Error Analysis, April 1969

TECHNICAL REPORTS

R-235 MIT/IL, et al., A Recoverable Interplanetary Space Probe, July 1959

R-339 M.B. Trageser, Guidance and Navigation System Information for Apollo Spacecraft Bidders, September 1961

R-341 R.H. Battin, AStatistical-Optimizing Navigation Procedure for Spacecraft Flight, September 1961, (Revised May 1962)

R-342 C. Stark Draper, W.G. Denhard, and M.B. Trageser, Development Criteria for Space Navigation Gyroscopes, October 1961

R-348 E.J. Hall, J. Miller, J. Aronson, and A. Lattanzi, Specification for Procurement of Apollo IRIG, December 1961 (Revised August 19631

R-349 G.W. Mayo, Guidance and Navigation System Reliability Program, December 1961 (Rev. D, July 1965)

R-353 R.H. Battin and J.S. Miller, Circumlunar Trajectory Calculations, April 1962

R-358 R. L. Alonso, A.I. Green, H.E. Maurer, and R.E. Oleksiak, A Digital Control Computer Developmental Model 1B, April 1962

R-367 P.H. Gilinson, Jr., C.R. Dauwalter, and J.A. Scoppettuolo, Multirange Precision Torque Measuring Devices, July 1962

R-368 G.W. Mayo and E.T. Driscoll, Reliability Handbook for Electronic Engineers, August 1962 (Revised October 1962)

R-372 J.H. Flanders, Velocity Steering Studies for the Apollo Mission, August 1962

R-373 J.M. Dahlen, et al., Guidance and Navigation System for Lunar Excursion Module, July 1962 (Revised August 1962)

R-376 R.E. Curry, (NASA), Two Impulse Abort Trajectories from Translunar Flight, October 1962

R-380 Development of High Capacity Heat Storage Materials Phase 1 - Study of Materials, July 1962 by Cryo-Therm, Inc.

R-382 R.H. Battin, Universal Formulae for Conic Trajectory Calculations, September 1962

R-383 G.W. Mayo, Design Review Procedures, September 1962

R-385 R.C. Hutchinson, Inertial Orientation of the Moon, October 1962

R-386 Max Petersen, Earth Limb Definition Photography, MA- 7 Mercury Flight, Cmdr. Carpenter, November 1962

R-387 S.J. Madden, Orbital Element Variation for a Body in Orbit Around the Moon, July 1963

R-388 D.G. Hoag, M.B. Trageser, A Progress Report on the Apollo Guidance System, December 1962

R-389 G. W. Mayo, Requirements of an Index to Design Evaluation Qualification and Reliability Test Program, March 1963 (Rev. C, July 1965)

R-391 G.D. Shepard, The Effect of Retrorocket Exhaust on Visibility during Lunar Touchdown, December 1962

R-393 R.L. Alonso, A.L. Hopkins, and H. Blair-Smith, Logical Description for Apollo Guidance Computer AGC 4, March 1963

R-395 G.W. Mayo, Jr., and G.E. Kruszewski, Guidance and Navigation System Reliability Apportionment and Initial Analysis, February 1963

R-396 E.T. Driscoll, Quality Assurance Plan, April 1963 (Rev. A, June 1964)

R-399 W.S. Crocker and I.G. McWilliams, Design and Checking of Indexing, April 1963

R-404 MIT/IL Radar Requirements for Primary Guidance and Navigation Operation, April 1963

R-405 E.T. Driscoll, Quarterly Reliability and Quality Assurance Status Report, April 1963

R-408 Eldon C. Hall, Design Concept of the Apollo Guidance Computer, June 1963

R-409 R. J. Schiesser, Principles of Cleanliness in the Assembly of Inertial Instruments, May 1963

R-410 Eldon C. Hall, General Design Characteristics of the Apollo Guidance Computer, May 1963

R-411 D.G. Hoag, APOLLO Guidance and Navigation: A Problem in Man and Machine Integration, April 1963

R-415 D. J. Lickly, R. Morth, B.S. Crawford, Appollo Reentry Guidance, July 1963

R-416 R.L. Alonso, A.L. Hopkins, Apollo Guidance Computer, August 1963

R-41 7 G.W. Cherry, A Class of Unified Explicit Methods for Steering Throttleable and Fixed-Thrust Rockets, January 1964 (Rev. 1, January 1964)

R-429 E.T. Driscoll, Reliability and Quality Assurance Progress Report, December 1963

R-434 MIT/IL, Preliminary Guidance Navigation Test Plan, December 1963

R-437 R. Kerrigan, Specification for Procurement of Apollo II Inertial Reference Integrating Gyro, February 1964

R-438 T. L. Schuck, High Performance, High Reliability via Magnetic Amplifiers in the Apollo Guidance Temperature Control System (to be published)

R-439 G. Kruszewski, Failure, Trouble, and Corrective Action Summary Design Phase Report #1, February 1964

R-440 E.T. Driscoll, Quarterly Reliability and Quarterly Assurance Progress Report, Apollo G&N Program, Period ended 31 December 1963, December 1963

R-443 D.G. Hoag, C. Stark Draper, R.H. Battin, M.B. Trageser, P. Bryant, Reprints of Presentation to Northeast Executive's Visit at Instrumentation Laboratory, M.I.T., March 1964

R-445 N.E. Sears, Technical Development Status of Apollo Guidance and Navigation, April 1964

R -446 N.E. Sears, Primary G&N System for Lunar Orbit Operations, Vol. I & II, April 1964

R-447 J.M. Dahlen, J.L. Nevins, Navigation for the Apollo Program, May 1964

R-451 E.T. Driscoll, Quarterly Reliability and Quality Assurance Progress Report Apollo Guidance and Navigation Program, July 1964

R-456 G.W. Cherry, Guidance - A General Explicit Optimizing Guidance Law for Rocket Propelled Spacecraft, August 1964

R-457 E.T. Driscoli, Quarterly Reliability and Quality Assurance Progress Report, October 1964

R-462 E.T. Ogrodnick, Dissimilar Metals: Compilation of Contacting Interfaces Between G&N Parts in Airborne Equipment Rlock I, September 1964

R-464 Reliability Staff, Reliability and Quality Assurance Progress Report, July 1964

R-466 F.D. Grant, Vibration Effects on Apollo Guidance, October 1964

R-467 R.H. Battin, The Compleat Sunrise: Being A Description of Program Sunrise, September i 964

R-475 Reliability Staff, Reliability and Quality Assurance Progress Report, August 1964

R-476 Reliability Staff, Reliability and Quality Assurance Progress Report, September 1964

R-477 J. Dahlen, A. Kosmala, ID. Lickly, J. Shillingford, and B. Sokkapa, Guidance and Navigation System Operations Plan Apollo Mission 202, January i965

R-479 W. Marscher, A Unified Method of Generating Conic Sections, February 1965

R-482 G.M. Levine, A Method of Orbital Navigation Using Optical Sightings to Unknown Landmarks, March 1965

R-489 C. A. Muntz, Users Guide to the Rlock II AGC/LGC Interpreter, April 1965

R-491 B.G. Sokkapa, On Optimal Steering to Achieve “Required Velocity”, April 1965

R-493 G.W. Mayo, Jr., Quarterly Reliability and Quality Assurance Progress Report, June 1965

R-494 N. Hanover and J.D. Fleming, Jr., Apollo G&N Test Data Program Plan, July 1965

R-495 M.B. Trageser and D.G. Hoag, Apollo Spacecraft Guidance Systems, June 1965

R-496 G.W. Mayo, Jr., Design Review Procedures, May 1965

R-498 R.L. Alonso and W.H. Aldrich, The ‘Braid’ Transformer Memory, September 1965

R-499 G.W. Cherry, Design Principles of the Lunar Excursion Module Digital Autopilot, June 19 65

R-500 C. Stark Draper, et al., Space Navigation Guidance and Control: AGARD Papers, June 1965, Vols I & II

R-503 G.S. Stubbs, A Block II Digital Lead-Lag Compensation for the Pitch-Yaw Autopilot of the Command and Service Module, October 1965

R-505 S.H. Goodwin, Test Results on a Pulse-Torqued Pendulum with a Permanent-Magnet type Torque Generator, November 1965

R-507 MIT/IL, G&N System Operation Plan Mission AS-204A, January 1966, Vol. I, Rev. 1, October 1966, Vol. II, Rev. 1, October 1966

R- 520 MIT/IL, Quarterly Reliability and Quality Assurance Progress Report, September 1965

R-521 E. Copps, Powered Flight Guidance of the Apollo Command and Service Module, November 1965

R-524 MIT/IL, Quarterly Reliability and Quality Assurance Progress Report, December 1965

R-525 M.M. Sullivan, Hybrid Simulation of the Apollo Guidance and Navigation System, December 1965

R-527 J. Dahlen, PGNC System Operations Plan, Mission AS-206, June 1966

R-529 M. Johnston, Apollo Mission Simulations Utilizing Flight Software and Hardware, July 1968

R-531 J.C. Pennypacker, Whole Number Strapdown Computation, February 1966

R-532 R. Morth, Reentry Guidance for Apollo, Vol. I, Vol. 11, January 1966

R-533 T.C. Lu, A Block II TVC Digital Autopilot Compensation for CSM Spacecraft, January 1966

R-537 GSOP, Mission 501, July 1966; Rev. 1, December 1966; Vol. II, Rev. 1, December 1966, Control Data and Error Analysis

R-539 A.R. Klumpp, A Manually Retargeted Automatic Descent and Landing System for LM, March 1966, Rev. 1, August 1967

R-541 M. Schetzen, The Power Density Spectrum of the Echo from an Airborne Doppler Radar, March 1966

R-547 J. Dahlen, GSOP AS-278, Vol. 1, CM GNCS Operations, October 1966; Vol. II, LM PGNCS, October 1966; Vol. III, Control Data, October 1966, Vol. IV, Error Analysis, October 1966

R-557 GSOP, Manned LM Earth Orbital using Program SUNDANCE, Sections 1 through 6 (updated regularly)

R-567 GSOP, Manned LM Earth Orbital and Lunar Mission using Program LUMINARY, Sections 1 through 6 (updated regularly)

R-568 E.J. Hall, Gyroscope Reliability Achieved through Proper Design and Effective Performance Monitoring, March 1967

R-569 R.R. Ragan, G. W. Mayo and J. Feldman, Apollo Guidance and Navigation Flight System Reliability, March 1967

R-574 Kee Soon Chun, High Precision Current Source for an Inertial System: Basic DC Measurement Technique, March 1967

R-575 Apollo Staff, Final Report on Additional Services, April 1967

R-577 GSOP, Manned CM Earth Orbital and Lunar Mission using Program COLOSSUS, Sections 1 through 6 (updated regularly)

R-596 J.F. Vittek, AS-205 Verification Results, December 1967

R-597 W.E. Tanner, Fluctuation Errors of Doppler Sensors at Low Velocities, December 1967

R-599 P.R. Mimno, Digital Simulation Manual, January 1968

R-600 J.H. Flanders, D. Fraser, F. Grant, J. Lawson, P. Bowditch, R. Crisp, J. Corrigan, J. Gilmore, A. Hopkins, W. Jenkins, and J.S. Miller, Control, Guidance and Navigation for Advanced Manned Missions: Vol. I, Systems; Vol. II, Multiprocessor Computer Subsystem; Vol. III, Radiation Sensor Systems, January 1968; Vol. IV, Inertial Subsystems, November 1968

R-618 R.H. Battin, A New Solution for Lambert's Problem, August 1968

R-620 J. Flanders, Dr. D. Fraser, and J. Lawson, Technology for Guidance and Navigation; Unmanned Deep Space Missions in the 1970's, August 1968

THESES

T-297	R. Kennan, J. Regenhardt, Star Occultation Measurement as an Aid to Navigation in Cis-Lunar Space, May 1962
T-298	D. Scott, C. Januska, R. Willis, Optimum Statistical Operations with Celestial Fix Data for Interplanetary Navigation, May 1962
T-299	M. Mauldin, R. Millard, Optimization of Interplanetary Mid-Course Velocity Correction, June 1962
T-313	D. Yankovich, A Digital Servo Network, June 1962
T-325	H. Wing, Analysis of a Core Rope Memory, August 1962
T-326	R. Curry, Two Impulse Abort Trajectories from Transluna: Flight, October 1962
T-327	G. Cushman, An Improved Technique for Measuring the True RMS Stability of a Sinusoidal Waveform, October 1962
T-328	C. Zicko, Increasing Class B Circuit Efficiency with Supply Voltage Level Switching, September 1962
T-329	A. W. Breck, C.L. Wilson, An Investigation of an Emergency Backup Guidance and Navigation Procedure for the Transearth Phase of the Apollo Mission, June 1963
T-339	P. Marcus, Microsecond, Efficient Current Switching in a Small D.C. Torque Motor, June 1963
T-342	E. DeNezza, M. Dittrich, Orbit and Landmark Determination Driving Lunar Orbit, June 1963
T-344	W. Vachon, The Effects of Lubrication Oil Deposition of Optical System in Vacua, May 1963
T-351	M. A. Lanman, Analysis of a Position Control Servo Incorporating Quantized Feedback, May 1963
T-352	R.L. Fortenbaugh, Description and Analysis of the Apollo Space Sextant Simulator, May 1963
T-353	Gary Stone, The Use of Rate Feedback in Compensation of a Temperature Control System, June 1963
T-355	Erich Bender, Sampled-Data Velocity Vector Control of a Spacecraft, June 1963
T-367	R. Lee, Analysis of a Nonlinear Mixed Feedback Control System, January 1964
T-369	John Deyst, Optimum Continuous Eslimation of Non-stationary Random Variables, January 1964
T-375	E. Shaw, Non-rigid Single-Axis Space Integrator Dynamics, May 1964
T-381	T. Blumenthal, Coordinate Transformation Matrix Errors on a Space Stablilized Inertial Non-System, June 1964
T-384	R. Fitzgerald, A Gradient Method for Optimizing Stochastic Systems, June 1964

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