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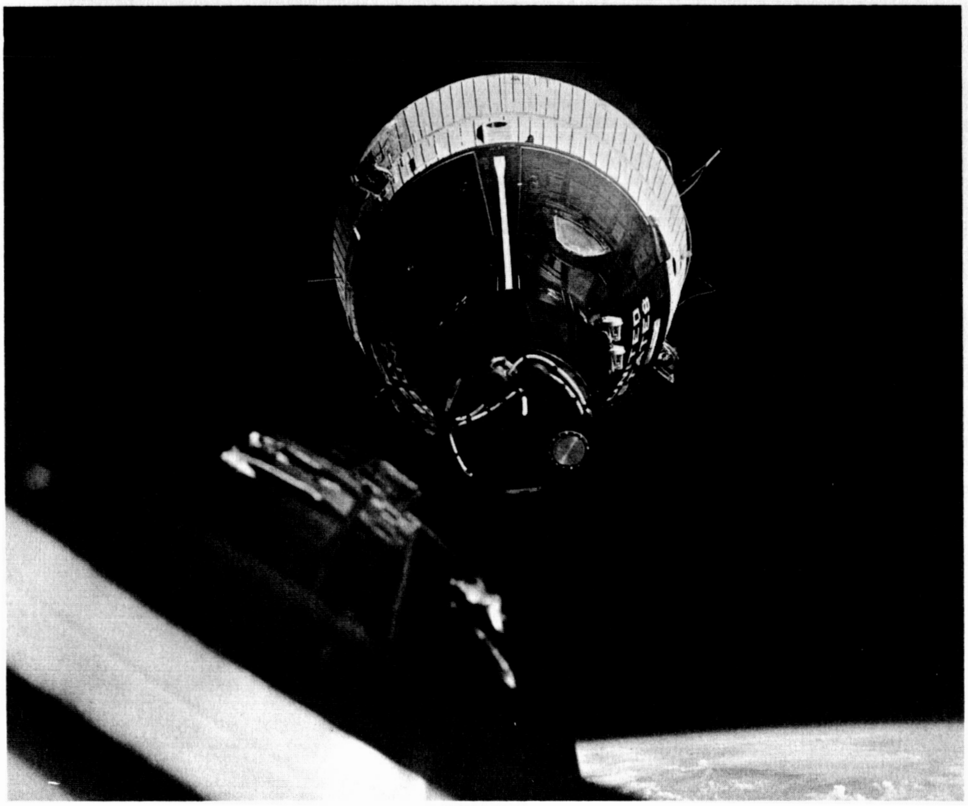
NASA SP-121

GEMINI MIDPROGRAM CONFERENCE

INCLUDING EXPERIMENT RESULTS



NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

FEBRUARY 23-25, 1966 MANNED SPACECRAFT CENTER HOUSTON, TEXAS

GEMINI

MIDPROGRAM CONFERENCE

INCLUDING EXPERIMENT RESULTS

GEMINI SPACECRAFT FLIGHT HISTORY

MISSION	DESCRIPTION	LAUNCH DATE	MAJOR ACCOMPLISHMENTS
Gemini I	Unmanned 64 orbits	Apr. 8, 1964	Demonstrated structural integrity.
Gemini II	Unmanned suborbital	Jan. 19, 1965	Demonstrated heat protection and systems performance.
Gemini III	Manned 3 orbits	Mar. 23, 1965	Demonstrated manned qualifications of the Gemini spacecraft.
Gemini IV	Manned 4 days	June 3, 1965	Demonstrated EVA and systems performance for 4 days in space
Gemini V	Manned 8 days	Aug. 21, 1965	Demonstrated long-duration flight, rendezvous radar capability, and rendezvous maneuvers.
Gemini VI	Manned 2 days rendezvous (canceled after failure of GATV)	Oct. 25, 1965	Demonstrated dual countdown procedures (GATV and GLV-spacecraft), flight performance of TLV and flight readiness of the GATV secondary propulsion system. Mission canceled after GATV failed to achieve orbit.
Gemini VII	Manned 14 days rendezvous	Dec. 4, 1965	Demonstrated 2-week duration flight and station keeping with GLV stage II, evaluated "shirt sleeve" environment, acted as the rendezvous target for spacecraft 6, and demonstrated a controlled reentry to within 7 miles of planned landing point.
Gemini VI-A	Manned 1 day	Dec. 15, 1965	Demonstrated on-time launch procedures, closed-loop rendezvous capability, and station keeping techniques with spacecraft 7.

EVA—Extravehicular activity
GATV—Gemini-Agena target vehicle
GLV—Gemini launch vehicle
TLV—Target launch vehicle

THE COVER—*Gemini VII as seen from Gemini VI-A just prior to rendezvous*

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FOREWORD

The Gemini Midprogram Conference presented a summary of the Gemini Program to date with emphasis on the first seven missions. This report contains the papers presented at that conference. These papers discuss the program development as it grew to meet the mission complexity and the stringent requirements for long-duration and rendezvous flight.

The papers are divided into two major groups: The first concerns spacecraft and launch-vehicle description and development, mission operations, and mission results; and the second reports results of experiments performed.

CONTENTS

PART I

C. Flight Operations

D. Mission Results

Concluding Remarks

PART II

A. Physical Science Experiments

PART I

1. INTRODUCTION

By ROBERT R. GILRUTH, *Director, NASA Manned Spacecraft Center, and* GEORGE M. LOW, *Deputy Director, NASA Manned Spacecraft Center*

In our first manned space-flight program, Project Mercury, man's capability in space was demonstrated. In the Gemini Program our aim has been to gain operational proficiency in manned space flight. At the midpoint in the Gemini flight program this aim has, in a large measure, been achieved.

The Gemini Program has produced numerous technical and management innovations through contributions of a large number of space-oriented organizations. At the peak of the Gemini activities more than 25 000 people in the aerospace industry were involved. This document will highlight the technical results of the program at the midpoint, with the management aspects to be reported more fully at a later opportunity.

The papers presented are representative of the contributions of the Gemini team. Participation by industry in the Gemini Program has been led by McDonnell Aircraft Corp., Martin-Marietta Corp., Lockheed Missiles & Space Co., and all of their associates. This participation has included more than 50 major contractors, more than 150 subcontractors, and, of course, a host of vendors and suppliers. The excellent performance of both the flight systems and the ground systems demonstrates graphically the strong capabilities of American industry in its support of these exploratory flights. Each of the companies involved deserves special recognition and credit for these accomplishments.

Many Government agencies have also been deeply involved in Gemini. In addition to NASA, the program has received support from the Department of Defense; the State Department; the Department of Health, Education, and Welfare; the Department of Commerce; the Atomic Energy Commission; and many others. The contributions of the Air Force Space Systems Division's National Range Division and the Navy Recovery Forces are well known. All of the astronauts who have flown to date in the Gemini Program have been trained as test pilots by either the Air Force or the Navy. In addition, the Air Force has provided the Gemini launch vehicle, which has performed with near perfection. There have been many other contributions by the military services in support of ejection-seat tests, centrifuge tests, and weightless trajectories utilizing the KC–135 aircraft.

Within NASA, every center has participated in direct technical support and, in many instances, in sponsorship of experiments. Of particular note is the contribution of the Goddard Space Flight Center in the implementation and operation of the worldwide network of tracking stations. Many nations of the free world have augmented or otherwise supported these stations, which are so vital to the manned space-flight program. Sponsorship of experiments and consultation services have been provided by universities and other institutions whenever and wherever they were needed. The Gemini Program is truly a national enterprise with international cooperation and support.

The Gemini team has been led by one of this country's outstanding engineers and program managers, Charles W. Mathews. Under his direction, significant advances have been made in this Nation's manned space-flight program. Gemini achievements in 1965 include five manned flights, yielding more than 1300 hours of manned flight in space; long-duration flight in steps of 4, 8, and 14 days; extravehicular activity, including the use of a self-propelled maneuvering gun; precise maneuvers in space, culminating in rendezvous; and controlled landing of a lifting spacecraft.

The results of the Gemini Program contribute directly to the Apollo Program and to other manned space-flight programs, such as the Air Force Manned Orbiting Laboratory. The lessons which have been learned, and the knowledge gained, have been rewarding, and give us confidence as we meet the problems and the programs of the future.

2. GEMINI PROGRAM FEATURES AND RESULTS

By CHARLES W. MATHEWS, *Manager, Gemini Program, NASA Manned Spacecraft Center;* KENNETH S. KLEINKNECHT, *Deputy Manager, Gemini Program, NASA Manned Spacecraft Center; and* RICHARD C. HENRY, *Manager, Office of Program Control, Gemini Program Office, NASA Manned Spacecraft Center*

Summary

This introductory paper has the objective of highlighting some of the intrinsic features of the Gemini Program and relating general results to these features, thereby furnishing a background for the more detailed papers which follow.

Introduction

Less than 5 years ago, men ventured briefly into space and returned safely. These initial manned space flights were indeed tremendous achievements which stirred the imagination of people worldwide. They also served to provide a focus for the direction of future efforts. Gemini is the first U.S. manned space-flight program that has had the opportunity to take this early experience and carry out a development, test, and flight program in an attempt to reflect the lessons learned. In addition, Gemini has endeavored, from its conception, to consider the requirements of future programs in establishing techniques and objectives.

Gemini Program Features

The purpose of the Gemini Program has usually been stated in terms of specific flight objectives; however, somewhat more basic guidelines also exist, and these are described in the following paragraphs.

Reliable System Design

The first guideline, reliable system design, is an objective of all programs, but in the Gemini Program several aspects of the approach are worth noting. One is the concept of independence of systems in which, to the degree practical, systems are designed in modules than can be developed and tested as a single unit. In this manner the inherent reliability of a system is not obscured by complex interacting elements. Advantages of this approach also exist in systems checkout and equipment changeout.

A second factor in Gemini systems design is the use of manual sequencing and systems management to a large extent. This feature affords simplicity by utilizing man's capability to diagnose failures and to take corrective action. It facilitates flexibility in the utilization of necessary redundancy or backup configurations of the systems. For example, in the spacecraft electrical-power system, the redundancy involved would make automatic failure sensing, interlocking, and switching both complex and difficult, if not impossible.

As already implied, the use of redundant or backup systems is an important facet of the Gemini spacecraft design. An attempt has been made to apply these concepts judiciously, and, as a result, a complete range of combinations exists. For systems directly affecting crew safety where failures are of a time-critical nature, on-line parallel redundancy is often employed, such as in the launch-vehicle electrical system. In the pyrotechnics system, the complete parallel redundancy is carried to the extent of running separate wire bundles on opposite sides of the spacecraft. In a few time-critical cases, off-line redundancy with automatic failure sensing is required. The flight-control system of the launch vehicle is an example of this type. In most crew-safety cases which are not time critical, crew-controlled off-line redundancy or backup is utilized. In the spacecraft propulsion system, the backup attitude control is used solely for the reentry operation. This reentry propulsion in turn involves parallel redundancy because of the critical nature of this mission phase. Many systems not required for essential mission phases are basically single systems with internal redundancy features commensurate with the requirements for overall mission success. The spacecraft guidance system is an example of this application. Certain systems have sufficient inherent reliability, once their operation has been demonstrated, that no special redundant features are required. The heat protection system is one of this type.

Future Mission Applicability

In the selection of systems and types of operations to be demonstrated, a strong effort was made to consider the requirements of future programs, particularly the manned lunar landing. It was not anticipated that Gemini systems necessarily would be directly used in other programs; however, their operating principles would be sufficiently close that the concepts for their use would be validated.

Where possible and to minimize development time, systems that already had some development status were selected; the spacecraft guidance and control system (a simplified block diagram is shown in fig. 2–1) typically represents this approach. The system is capable of carrying out navigation, guidance, and the precise space maneuvers needed for such activities as rendezvous, maneuvering, reentry, and launch guidance. At the same time, such major elements of the system as the inertial platform,

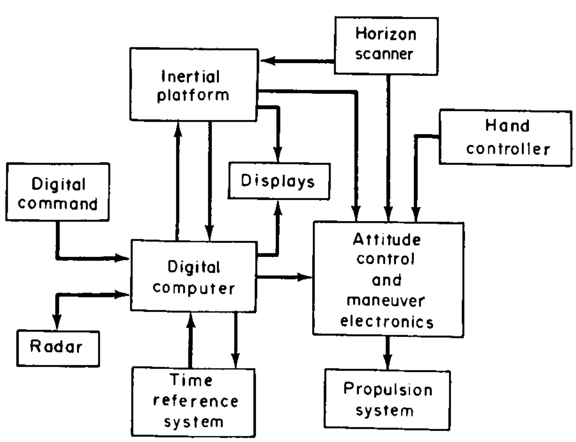

FIGURE 2–1.—Example of Gemini systems applicable to future programs and missions (guidance and control system shown).

the digital computer, the radar, and the flight-director display drew heavily on previous developments. Reliability, system operating life, and the sizing of consumables were also selected to afford durations corresponding to the requirements of oncoming programs.

These ground rules were applicable to many other systems. In the case of the Gemini launch vehicle, great benefit was obtained from the Titan II development program, even to the extent of validating certain Gemini-peculiar modifications in the test program prior to their use in Gemini.

Minimum Flight Qualification Tests

Because flying all-up manned space vehicles is expensive, time consuming, and exceedingly sensitive to failures, the Gemini development was based on the premise that confidence could be achieved through a properly configured program of ground tests and that a very limited number of unmanned flights could serve to validate the approach. With this in mind, a comprehensive ground program was implemented in the areas of development, qualification, and integrated systems tests. In addition, certain other measures were taken to further this approach, such as the utilization of the external geometric configuration and general heat protection approach of the Mercury spacecraft. The Titan II applicability has already been mentioned.

The ground-test program not only involved rigorous component and subsystems qualification and the usual structural testing, but also included many special test articles for integrated testing. These test articles included an airborne systems functional test stand for the launch vehicle and production spacecraft elements for ejection-seat tests, electrical and electronic compatibility tests, landing-system drop tests, at-sea tests, zero-g tests, and also a complete flight spacecraft for thermal-balance tests.

As indicated on figure 2–2, a high level of ground test effort commenced at the outset of the program and was sustained past the first several flights. The ability to fly with some qualification testing incomplete is related to the differences between the early spacecraft configurations and the long-duration and rendezvous spacecraft configurations. It was hoped that

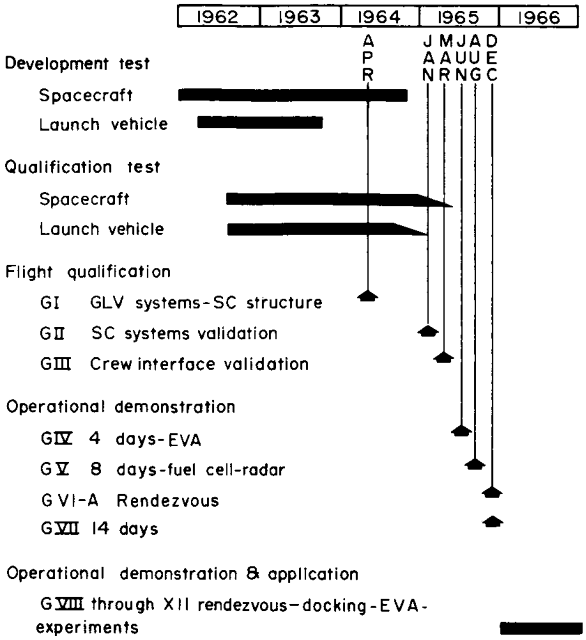

FIGURE 2-2.—Gemini test program.

the ground testing could be completed earlier, but the problems that were isolated and the required corrective action prevented earlier accomplishment. In spite of the great effort involved, it was better to utilize a ground-test program to ferret out problems than to encounter them in flight.

The ability to minimize flight qualification tests is also indicated in figure 2-2. Two unmanned flights were required prior to the first manned flight, and one manned flight test was required before proceeding into the operational program. No problems that significantly impacted following flights were encountered on these early flights.

Streamlined Launch Preparations

Activities aimed at streamlining the launch preparations and the other checkout activities commenced with the design. In the case of the spacecraft, the majority of equipment was placed outside the pressure vessel, with large removable doors providing a high percentage of equipment exposure during tests. Connectors were designed integral with each piece of equipment so that, when aerospace ground equipment was required for tests, the flight wire bundles need not be disconnected. These and similar features allow multiple operations to take place around the spacecraft and minimize damage while testing or replacing equipment.

Although repetitive testing still exists, it has been possible to curtail it because of the preservation of integrity features previously discussed and because of the improvement in test flow, to be discussed later. An outcome of the Gemini Program experience is that system reliability is achieved as a result of the basic development, qualification, and reliability testing; consequently, repetitive testing of the space vehicle need not be used for this purpose.

Another important aspect of the program is the delivery of flight-ready vehicles, including Government-furnished equipment, from the manufacturer's plant. This objective dictates complete integrated testing at the factory and includes crew participation in system tests, simulated flights, stowage reviews, and altitude-chamber runs. Equally important, it means the delivery of vehicles with essentially zero open items. All elements of the Gemini team, both launch vehicle and spacecraft, have worked extremely hard to achieve this end.

At Cape Kennedy the checkout plans have not been inflexible. They are continuously under review and are changed when the knowledge gained shows that a change is warranted. Some of the testing required for the first flights is no longer required or, in some cases, even desirable. Improvements in test sequences have also been achieved, and these avoid excessive cabling-up or cabling-down, or other changes in the test configuration. These alterations in test plans are carefully controlled and are implemented only after detailed review by all parties concerned.

Buildup of Mission Complexity

Although the Gemini flights have built up rapidly in operational capability, the planning endeavors have been orderly in order to make this buildup possible. The progressive buildup in mission duration is obvious from figure 2-2, but this philosophy also applies to most categories of the flight operations and will be discussed in more detail in subsequent papers. It can be stated that, from systems considerations alone, the 14-day flight of Gemini VII might not have been possible without the prior experience of the 8-day flight of Gemini V.

Another aspect of the buildup idea is the control of configuration to avoid flight-to-flight impact. The fuel cells and the cryogenic stowage of their reactants are by far the newest developments of all the Gemini systems. They were first flown "off-line" on Gemini II to obtain information on prelaunch activation and on their integrity in the launch and weightless environment. The next planned use was on Gemini V, where a fuel-cell power system was a mission requirement. To permit concentration on the basic flight objectives, the intermediate flights were planned with batteries as the source of electrical power. Similarly, the Gemini VI–A spacecraft utilized battery power so that possible results of the Gemini V flight would not impact on the first space rendezvous. This arrangement resulted in an excellent integration of these new systems into the flight program. The good performance of the fuel-cell systems now warrants their use on all subsequent flights.

Flight Crew Exposure

Gemini objectives require that complex operational tasks be demonstrated in earth orbit, but it is also desired to provide the maximum number of astronauts with space-flight experience. As a result, no flight to date has been made with crewmembers who have flown a previous Gemini mission. In fact, two significant flights, Gemini IV and VII, were made with crews who had not flown in space before. In the other three flights, the command pilot had made a Mercury flight. The results achieved attest to the character and basic capabilities of these men and also reflect the importance of an adequate training program. Again, a more detailed discussion of the subject will be presented in subsequent papers.

The flight crew require detailed familiarity with and confidence in their own space vehicle. This is achieved through active participation in the flight-vehicle test activities. The flight crews require many hours of simulation time to gain proficiency in their specific mission tasks, as well as in tasks common for all missions. With short intervals between missions, the availability of trained crews can easily become a constraint, and careful planning is necessary to avoid this situation. Much of this planning is of an advanced nature in order to insure the adequate capability and flexibility of simulation facilities.

Complex Mission Operations

The fundamentals of manned-mission operations were demonstrated in the Mercury Program where the flight-control functions of orbital insertion, orbit determination, systems monitoring, retrofire time, orbital landing-point prediction, and recovery were developed. These features also apply to Gemini flight control, but in a greatly expanded sense. There are many reasons for the increased requirements. On a rendezvous mission, the Gemini space vehicle is launched on a variable azimuth that is set-in just prior to launch, and the vehicle yaw-steers into orbit. These features affect both the flight-control function and the recovery operations for launch aborts. Also during rendezvous missions, flight control must be exercised over two vehicles in orbit at the same time, both of which have maneuvering capability. The orbit maneuvering further complicates the recovery operation by requiring mobility of recovery forces. These factors, combined with the relatively higher complexity of the Gemini spacecraft, require the rapid processing and display of data and a more centralized control of the operation. The maneuvering reentry is another aspect of the Gemini Program that complicates the flight control and recovery operations.

The long-duration missions have required shift-type operations on the flight-control teams and their support groups. This mode of operation increases the training task and introduces additional considerations, such as proper phasing from one shift to the other.

The Mission Control Center at Houston was designed to support these more complex functions, and these functions have been carried out with considerable success. It is felt that the implementation and demonstration of this part of the Gemini capability will be one of the largest contributions in support of the Apollo Program.

Flexible Flight Planning

Another facet of the Gemini flights is flexibility in flight planning and control. Requirements for flexibility have existed in both the preflight activities and in the manner in which the actual flight is carried out. The prime example of preflight flexibility is the implementation of the Gemini VII/VI–A mission subsequent to the aborted rendezvous attempt of the original Gemini VI mission. Although strenuous effort was required in all areas, these activities did take place essentially in accordance with the plan.

During actual flights, the need has often arisen to alter the flight plans. These changes have been implemented without affecting the primary objectives of the mission. They have also been initiated in a manner to obtain a high degree of benefit from the mission in terms of all the predetermined flight objectives. In some cases, new tasks have been incorporated in the flight plan during the flight, as was the phantom rendezvous and ground transponder interrogation on Gemini V when difficulties forced abandonment of the rendezvous-evaluation-pod exercise. While detailed premission flight planning is a requirement, the ability to modify rapidly has been of great benefit to the program.

Postflight Analysis and Reporting

In a manned operation, it is necessary to isolate and resolve problems of one flight before proceeding with the next. In the Gemini Program, an attempt has been made to establish an analysis and reporting system which avoids this potential constraint. The general plan is shown in figure 2–3. In targeting for 2-month launch centers, the publication of the mission evaluation report was set at 30 days. In turn, a major part of the data handling, reduction, and analyses activities takes place during a

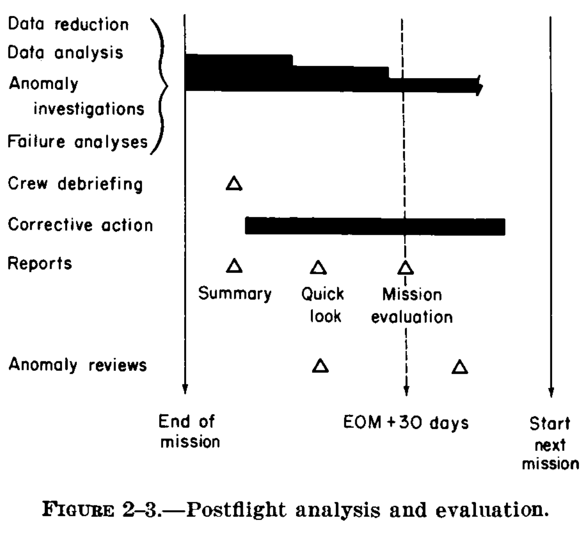

FIGURE 2–3.—Postflight analysis and evaluation.

period of approximately 2 weeks following each mission. All problems are not necessarily solved at the end of the 30-day period, but isolation of problems, evaluation of their impact, and initiation of corrective action have been possible.

In carrying out these activities, a formal task group is set up. Rather than having a permanent evaluation team, personnel are assigned who have been actively working in the specific areas of concern before the flight and during the flight. This approach provides personnel already knowledgeable with the background of the particular flight. Corrective action is initiated as soon as a problem is isolated and defined. At this point in the program, impact of one flight on another has not proved to be a major constraint.

Personnel Motivation

Although good plans and procedures are needed in a major program, well-motivated people must be behind it. Teamwork comes primarily from a common understanding through good communications. In the Gemini Program, an effort has been made to facilitate direct contact at all levels. Good documentation is necessary but should not constrain direct discussions. Individual people, right down to the production line, must fully realize their responsibility. This effort starts with special selection and training, but it is necessary to sustain the effort. With this in mind, a number of features directly related to the individual have been included in the flight-safety programs. The launch-vehicle program is an outstanding example of this effort. People working on Gemini hardware are given a unique badge, pin, and credentials. Special awards are presented for outstanding work. Special programs are held to emphasize the need for zero defects. A frequent extra feature of such programs is attendance and presentations by the astronauts. Much interest has been exhibited in this feature, and it serves to emphasize the manned-flight safety implications of the program.

Before leaving this subject, the effect of incentive contracts should also be pointed out. All major Gemini contracts, although differing in detail, incorporate multiple incentives on performance, cost, and schedule. The experience with these contracts has been very good in providing motivation throughout the contractor organization, and they have been structured to provide this motivation in the desired direction. The incentive features have served to enhance program visibility, both for the Government and for the contractors.

Gemini Flight Results

Gemini Objectives

At the outset of the Gemini Program, a series of flight objectives was set forth. As stated previously, these objectives were directed at the demonstration and investigation of certain operational features required for the conduct of future missions, particularly the Apollo missions. These original objectives include: long-duration flights in excess of the requirements of the lunar-landing mission; rendezvous and docking of two vehicles in earth orbit; the development of operational proficiency of both flight and ground crews; the conduct of experiments in space; and controlled land-landing. Several objectives have been added to the program, including extravehicular operations and onboard orbital navigation. One objective, controlled land-landing, has been deleted from the program because of development-time constraints, but an important aspect of this objective continues to be included—the active control of the reentry flight path to achieve a precise landing point. Initial demonstrations of most of these objectives have been made, but effort in these areas will continue in order to investigate the operational variations and applications which are believed to be important. In addition, the areas yet to be demonstrated, such as docking and onboard orbital navigation, will be investigated on subsequent flights.

Mission Results

The flight performance of the launch vehicle has been almost entirely without anomalies (fig. 2–4). There have been no occasions to utilize backup guidance or any of the abort modes. On two occasions, the Gemini II and VI–A missions, the automatic-shutdown capability was used successfully to prevent lift-off with launch-vehicle hardware discrepancies.

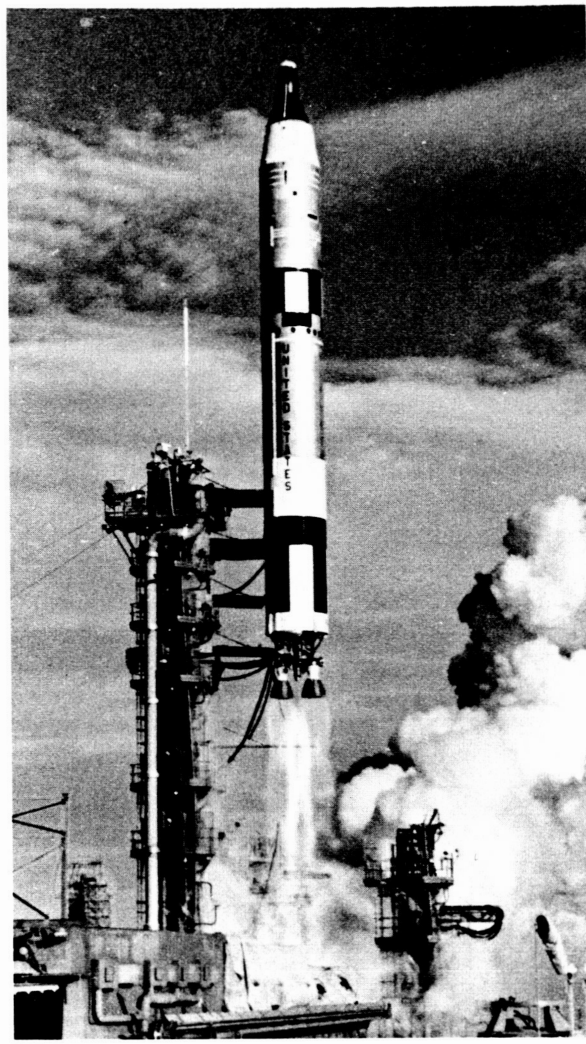

FIGURE 2–4.—Lift-off of Gemini space vehicle.

In orbital operations, all missions have taken place with no significant crew physiological or psychological difficulties (fig. 2–5). The proper stowage, handling, and restowage of equipment has been a major effort. There has been a tendency to overload activities early in the mission. This is undesirable because equipment difficulties are quite likely to become evident early in the mission. It has always been possible to develop alternate plans and to work around these equipment difficulties in carrying out the basic flight plan. The cabin environment has proved satisfactory, but pressure-suit comfort and mobility considerations make doffing and donning capabilities desirable. The performance of the spacecraft maneuvering and attitude control has been outstanding. Special orbital

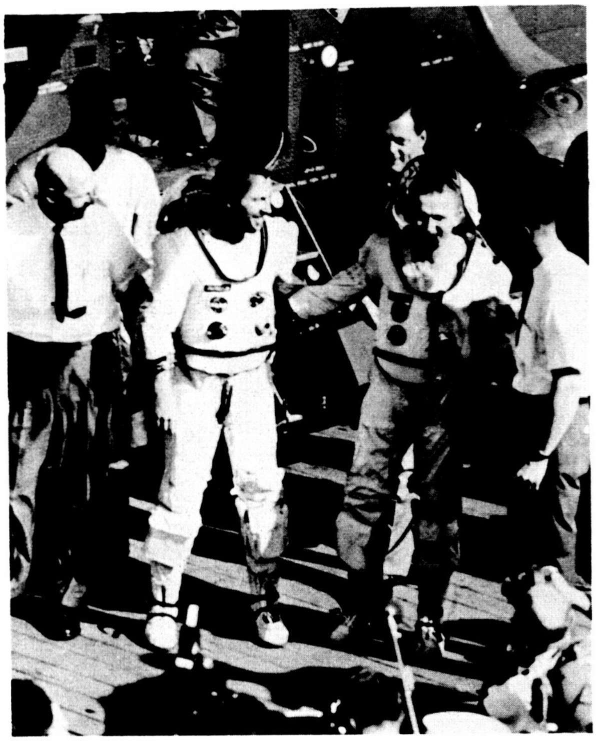

FIGURE 2-5.—Gemini VII flight crew onboard recovery ship.

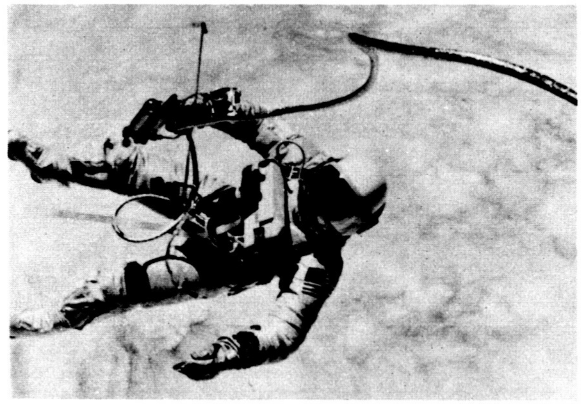

FIGURE 2-6.—Extravehicular activity during Gemini IV mission.

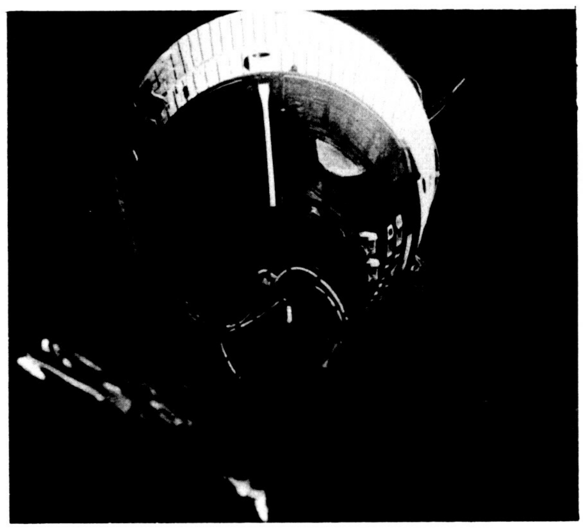

FIGURE 2-7.—Rendezvous during Gemini VI-A and VII missions.

FIGURE 2-8.—Typical experiment activity.

tasks, such as extravehicular activities, rendezvous, and experiments, have been conducted very satisfactorily. During the extravehicular investigation on Gemini IV (fig. 2-6), no disorientation existed, and controlled maneuvering capability was demonstrated. This capability is felt to be a prerequisite to useful extravehicular operations. The straightforward manner with which the rendezvous was accomplished (fig. 2-7) does indeed reflect the extremely heavy effort in planning, analysis, and training that went into it.

The Gemini experiments have been of a nature that required or exploited man's capability to discriminate for the collection of data, and then retrieve the data for postflight evaluation. During the flights, 54 experiments were conducted (fig. 2-8). All of the experiment flight objectives, except for about three, have been accomplished.

All retrofire and reentry operations have been performed satisfactorily, although only the last two missions demonstrated precise controlled maneuvering reentry (fig. 2-9). In the Gemini VI-A and VII landings, an accuracy of about

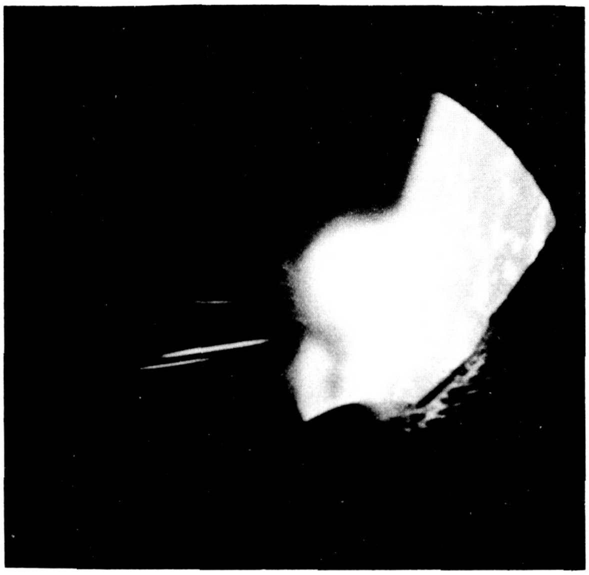

FIGURE 2-9.—View through spacecraft window during reentry.

6 miles was achieved, and this is approaching the capabilities of the system being utilized. Recovery has always been rapid, and the support of recovery by the Department of Defense has been excellent (fig. 2-10).

FIGURE 2-10.—Recovery operations.

Concluding Remarks

The Gemini design concepts and comprehensive ground test program have enabled the flight program to be conducted at a rapid pace and to meet program objectives. Much credit in this regard must be given to James A. Chamberlin, who spearheaded the conceptual effort on the Gemini Program.

Although flight operations have been relatively complex, they have been carried out smoothly and in a manner to circumvent difficulties, thereby achieving significant results from each flight.

The flights, thus far, have served to provide an initial demonstration of most of the Gemini flight objectives. Future flights will explore remaining objectives as well as variations and applications of those already demonstrated.

The Gemini team has worked exceedingly hard to make the program a success, and the special effort in developing teamwork and individual motivations has been of considerable benefit.

A

SPACECRAFT

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3. SPACECRAFT DEVELOPMENT

By DUNCAN R. COLLINS, *Manager, Office of Spacecraft Management, Gemini Program Office, NASA Manned Spacecraft Center;* HOMER W. DOTTS, *Deputy Manager, Office of Spacecraft Management, Gemini Program Office, NASA Manned Spacecraft Center;* WILBURNE F. HOYLER, *Gemini Program Office, NASA Manned Spacecraft Center; and* KENNETH F. HECHT, *Gemini Program Office, NASA Manned Spacecraft Center*

Summary

The flight sequence of the two-man Gemini spacecraft from lift-off through reentry and landing is similar to that of the Mercury spacecraft; however, additional capabilities are incorporated in its design for each phase of flight. The Gemini spacecraft has the capability of adjusting its own insertion velocity after separating from the launch vehicle. It also can maneuver in space, as well as control its trajectory during reentry. The Gemini spacecraft is configured to facilitate assembly, testing, and servicing. Its two-man crew has provided the capability to accomplish complicated mission objectives. Its built-in safety features cover all phases of flight and have greatly increased the confidence in the practicality of manned space vehicles.

Introduction

The Gemini spacecraft with its launch vehicle, shown in figure 3–1, is the second generation of manned space vehicles produced in the United States. The Gemini launch vehicle is a modified version of the Air Force Titan II ballistic missile. The spacecraft incorporates many concepts and designs that were proved during Project Mercury, as well as new designs required by the advanced Gemini mission objectives and more operational approach.

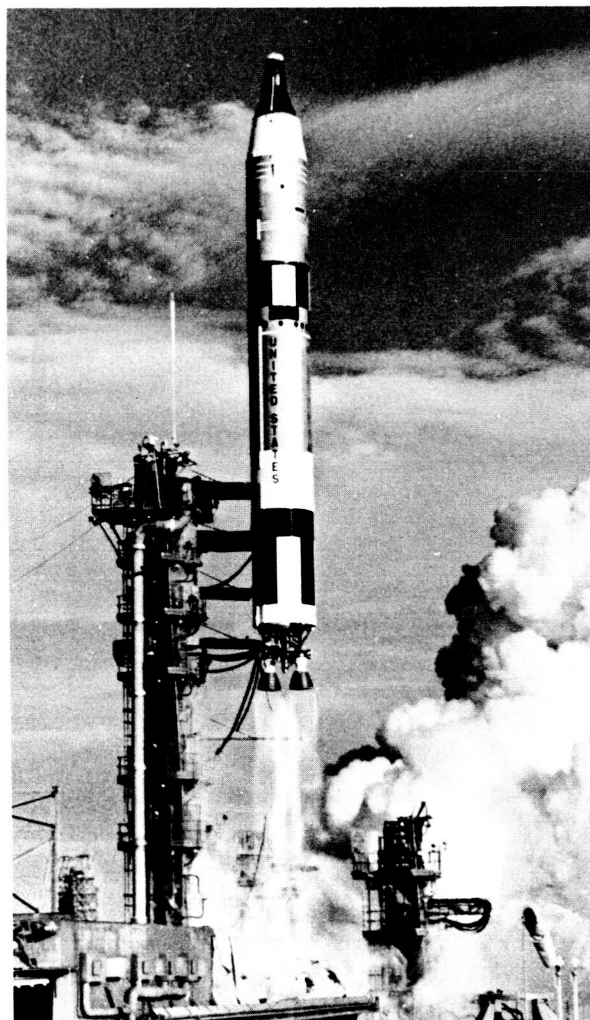

FIGURE 3–1.—Gemini space vehicle at lift-off.

Flight Sequence

Launch

The combined length of the Gemini launch vehicle and spacecraft is approximately 110 feet. The maximum diameter of both vehicles is 10 feet, which is constant from their common interface to the base of the launch vehicle. The diameter of the spacecraft decreases forward of the interface.

The launch vehicle consists of two stages: the first stage separates approximately 155 seconds after lift-off; the second-stage engine is shut down approximately 335 seconds after lift-off. These values vary somewhat depending upon performance, atmospheric conditions, and the insertion velocities required for a particular mission. Separation of the spacecraft from the second stage is initiated by the crew approximately 20 seconds after second-stage engine shutdown. This time delay assures that the thrust of the second-stage engine has decayed sufficiently to prevent recontact between the two vehicles during separation. Two 100-pound thrusters, located at the base of the spacecraft, are used to separate the two vehicles. These thrusters are nominally fired for several seconds; however, this time may be extended, if necessary, for insertion velocity adjustment. On two missions, this time was held to a minimum to permit launch-vehicle station-keeping exercises.

In-Orbit Configuration and Capability

Figure 3–2 shows the in-orbit configuration of the spacecraft. The spacecraft is manufactured in two major assemblies: the reentry vehicle and the adapter. These assemblies are held together by three structural straps spaced approximately 120° apart at the interface. Electrical cables and tubing cross this interface at these three points. The adapter serves not only as the transition structure between the reentry vehicle and the launch vehicle, but also as the service module for the reentry vehicle while in orbit. The adapter is separated into two compartments: the retrorocket-adapter section and the equipment-adapter section. The retrorocket-adapter section contains the four retrorockets, and the equipment-adapter section contains systems or parts of systems which are used only in orbit and are not required for reentry and recovery. The reentry vehicle contains the pressurized cabin, the crew, flight controls, displays, the life-support system, and the crew provisions. It also contains the reentry-control-system section and rendezvous and recovery section. Other systems, some used only for reentry and some used during all flight phases, are installed in the reentry vehicle.

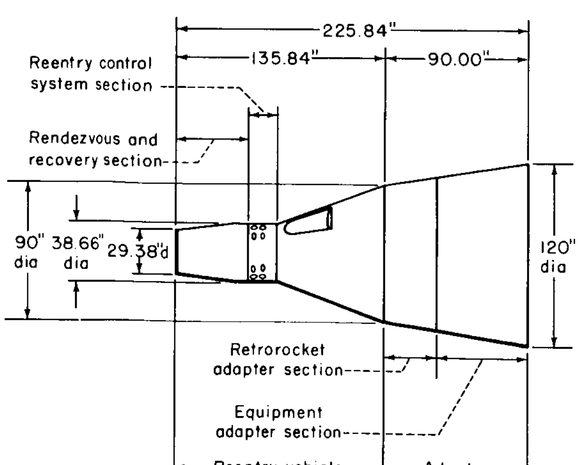

FIGURE 3–2.—Configuration of Gemini spacecraft.

The Gemini spacecraft has the capability to maneuver in space with an orbital attitude and maneuver system, which is located in the adapter section. Spacecraft attitude is controlled with eight 25-pound thrusters, and translation along any axis is accomplished with six 100-pound thrusters and two 85-pound thrusters. This system has been used extensively during all Gemini flights to make in-plane and out-of-plane maneuvers. The successful rendezvous between the Gemini VI–A and VII spacecraft was accomplished with this system and the associated guidance system.

Reentry Sequence

In preparation for the reentry sequence, the spacecraft is placed in retrograde attitude using the orbital attitude and maneuver system (fig. 3–3). The reentry control system, located in the reentry vehicle, is then activated and provides attitude control through the rentry phase. The equipment-adapter section is then separated with a shaped-charge pyrotechnic, followed by the sequential firing of the four retrorockets. After retrograde, the retrorocket-adapter section, containing the spent retrorockets, is separated from the reentry vehicle and is jettisoned by a spring which exerts a force at the center line of the heat shield.

The concept of jettisoning the spacecraft section containing systems not required for reentry was adopted for the following reasons:

(1) It reduced the size and weight of the reentry vehicle. As the reentry vehicle had to be provided with external heat-protection materials for reentry, it follows that its size should be minimized to reduce overall spacecraft weight.

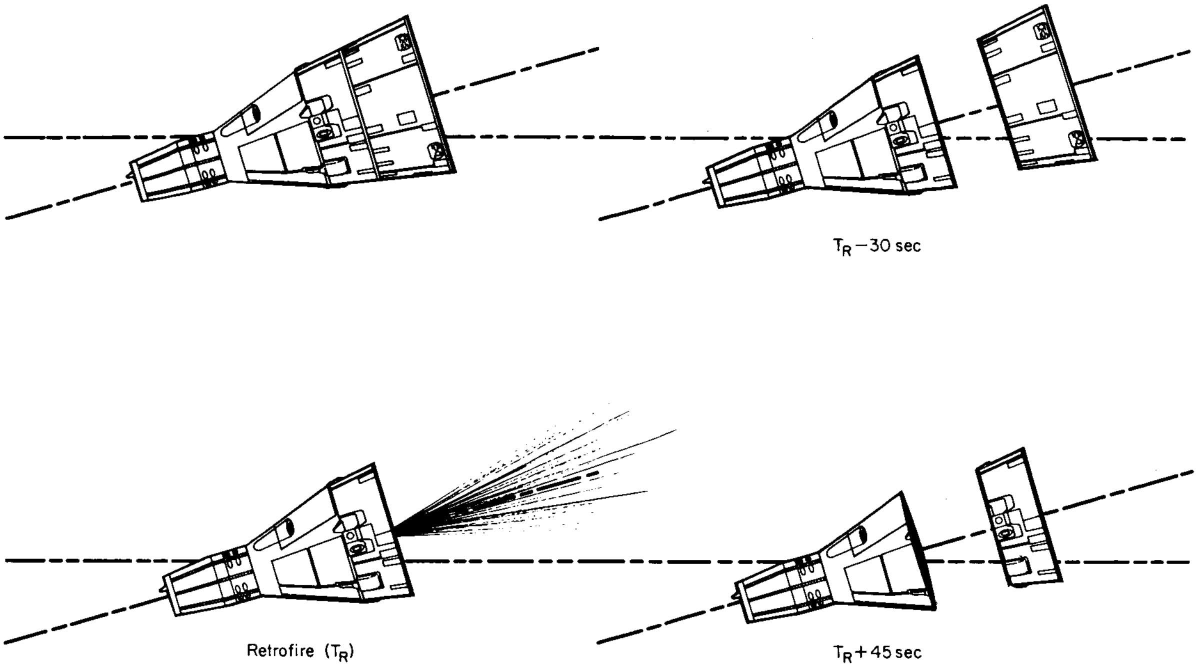

FIGURE 3-3.—Retrograde sequence.

(2) The adapter skin and stringers provided a radiator for the environmental control system in orbit. The configuration of this structure, which was designed for the launch and orbit environment, made it easily adaptable as a radiator.

(3) Space and center-of-gravity constraints do not exist in the adapter sections to the degree they do in the reentry vehicle; therefore, the adapters are less sensitive to equipment location and design changes.

(4) It provided a configuration with much flexibility. The design of systems located in the adapter has varied considerably with each mission. As an example, the Gemini III and VI–A systems were designed to support a 2-day mission using battery power. Gemini IV design supported a 4-day mission using battery power. Gemini V and VII were powered with fuel-cell electrical systems which supported long-duration missions of up to 14 days. Although the configuration of the systems installed in the adapter varied to a great extent, little change was required in the reentry vehicle.

The Gemini reentry vehicle is provided with the capability to control the reentry trajectory and to land at a predetermined touchdown point. An asymmetric center of gravity (fig. 3–4) causes the vehicle to trim aerodynamically at an angle of attack, thus providing a lift vector normal to the flight path. A controlled trajectory to a desired touchdown point (fig. 3–5) is made by varying the bank angles to the right or to the left. A maximum-lift trajectory is obtained by holding a zero bank angle through reentry. A zero-lift ballistic trajectory is obtained by rolling the vehicle continuously at a constant rate, which nullifies the lift vector. When making a controlled reentry, bank angles greater than 90° are avoided (except when flying a zero-lift trajectory) to preclude excessive heating rates and loadings. A controlled reentry may also be executed using a combination of the zero-lift trajectory and bank technique.

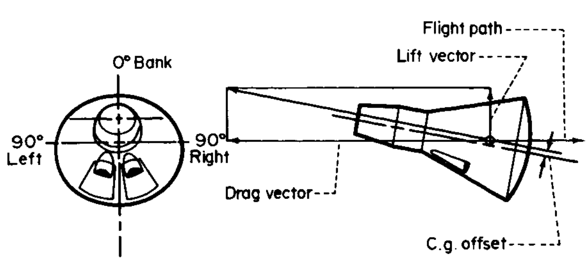

FIGURE 3–4.—Reentry vehicle trim.

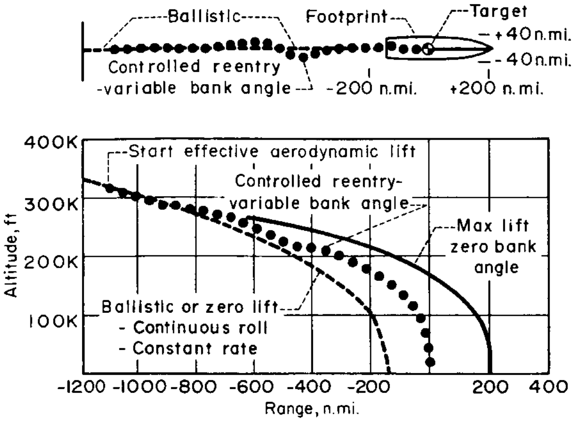

FIGURE 3-5.—Reentry control.

Landing Sequence

A single-parachute landing system is used on Gemini spacecraft, with the ejection seats serving as a backup. In the normal landing sequence (fig. 3-6), an 8-foot-diameter drogue parachute is deployed manually at approximately 50 000 feet altitude. Below 50 000 feet, this drogue provides a backup to the reentry control system for spacecraft stabilization. At 10 600 feet altitude, the crew initiates the main-parachute deployment sequence, which immediately releases the drogue, allowing it to extract the 18-foot-diameter pilot parachute. At 2.5 seconds after sequence initiation, pyrotechnics release the recovery section, to which the pilot parachute is attached and in which the main parachute is stowed. As the reentry vehicle falls away, the main parachute, an 84-foot-diameter ring-sail, deploys. The pilot parachute diameter is sized such that recontact between the recovery section and the main parachute will not occur during descent. After the crew observes that the main parachute has deployed and that the rate of descent is nominal, repositioning of the spacecraft is initiated. The spacecraft is rotated from a vertical position to a 35° noseup position for landing. This landing attitude reduces the acceleration forces at touchdown on the water to values well below the maximum which could be tolerated by the crew or by the spacecraft.

Spacecraft Design

Reentry Vehicle

The reentry vehicle (fig. 3-7) is manufactured in four major subassemblies: the ablative heat shield, the section containing the pressur-

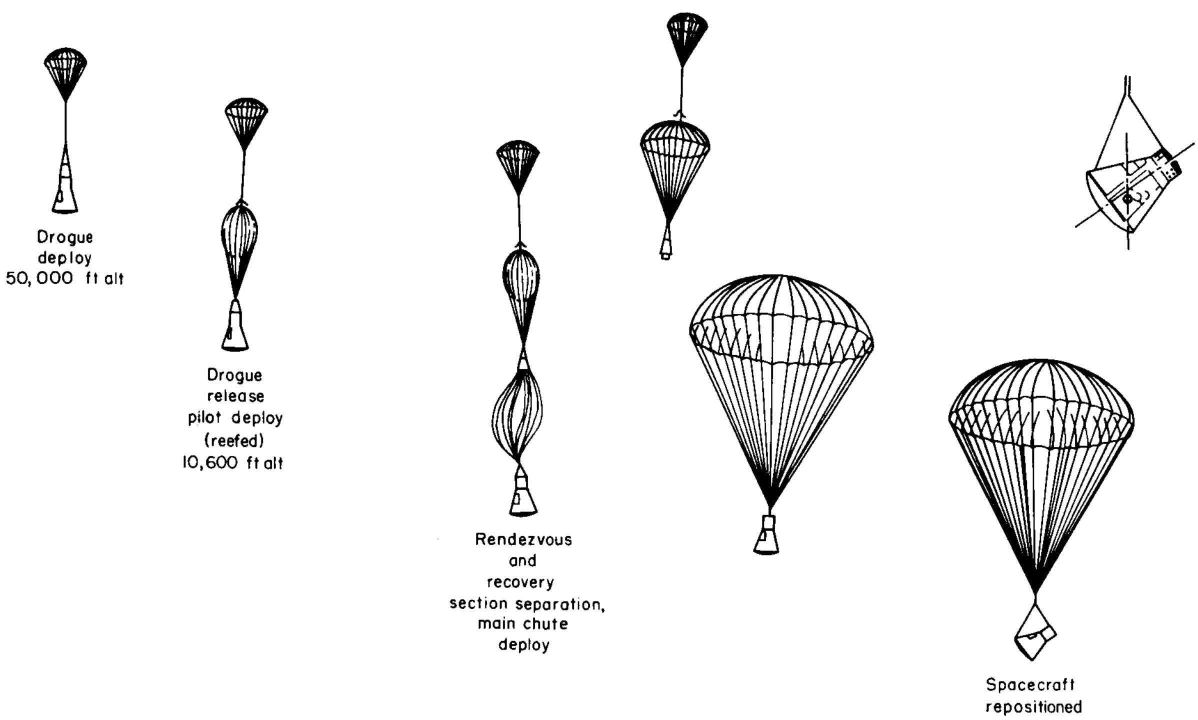

FIGURE 3-6.—Landing sequence.

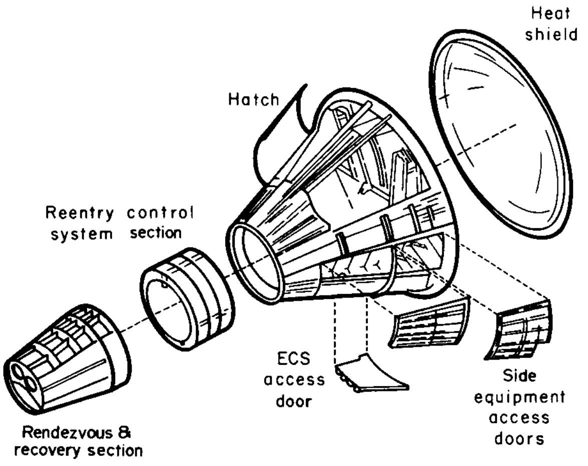

FIGURE 3-7.—Reentry vehicle structure.

ized cabin, and the reentry control system and the rendezvous and recovery sections. The vehicle was sized to house the pressurized cabin with two crewmembers and associated equipment, and other systems required to be located in the reentry vehicle. The use of two crewmembers on Gemini flights, as opposed to the one-man crew in Project Mercury, has resulted in expanded flight accomplishments and flexibility in flight planning and operation. For example, experiment activity would have been sharply curtailed had only one crewmember been aboard. With only one crewmember, extravehicular activity would have been unlikely as an added objective. Teamwork in preparation for each flight is considered to be a major asset in the crew training programs. Furthermore, the number of trained crew personnel is expanded, and this will substantially assist the Apollo Program. Many major program objectives involving inflight control and crew management of spacecraft systems could not have been accomplished had only one crewmember been aboard.

The Mercury blunt-body concept was selected for the Gemini spacecraft and provides a configuration which is compatible with the design requirements necessary to meet mission objectives. From a reliability, cost, and schedule standpoint, the advantages of using this concept are obvious, as much of the experience and technology gained on Project Mercury could be directly applied to the development and design of the Gemini spacecraft.

The structure of the reentry vehicle is predominately titanium, and it is skinned internally to the framing. The vehicle is protected from the heat of reentry by a silicone elastomer ablative heat shield on the large blunt-end forebody of the vehicle, by thin René 41 radiative shingles on the conical section, and by beryllium shingles which provide a heat sink on the small end of the vehicle. MIN-K insulation is used as a conductive barrier between the shingles and the structure, and Thermoflex blankets are used as a radiative barrier. Flat, double-skinned shear panels form a slab-sided pressure vessel, within the conical section, for the crew. Two large, hinged hatches provide access to the cabin. The reentry vehicle structure is designed with an ultimate factor of safety of 1.36.

The highest reentry heating rates are attained if the spacecraft aborts from a launch trajectory several thousand feet per second short of the orbital insertion velocity and reenters along a ballistic trajectory, whereas the highest total heat is sustained during reentry from orbit along a maximum-lift trajectory (fig. 3-8). The Gemini spacecraft was designed for a maximum stagnation-point heating rate of 70 Btu/ft²/sec and a maximum total heat of 13 138 Btu/ft². Maximum total heat is the critical design condition for the ablative heat shield and for the beryllium shingles located on the small end of the vehicle, while maximum heating rate is the critical design condition on the René shingles on the conical section.

The trajectory for the Gemini II mission was tailored to produce high heating rates as a test of the critical design condition on the René

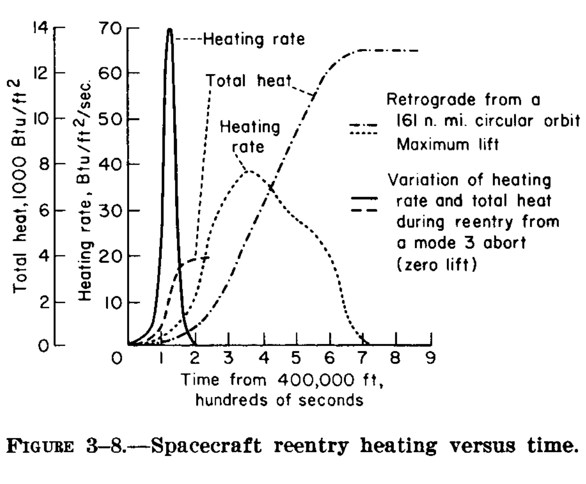

FIGURE 3-8.—Spacecraft reentry heating versus time.

shingles. Based on the Gemini II trajectory, the stagnation heating rate reached a calculated value of 71.8 Btu/ft^2/sec, slightly in excess of that predicted. The René shingle temperatures were generally as expected. However, in one localized area—in the wake of a fairing located on the conical section near the heat shield on the most windward side (fig. 3–9)—several small holes were burned in the shingles. An additional wind-tunnel test was conducted on a 10-percent model, and results indicated that minor changes in the fairing configuration would not decrease the heat intensity. The intensity was, however, a function of Reynolds number and of the angle of attack. As a result of this test, the trim angle on subsequent spacecraft was slightly reduced, and the thickness of two René shingles aft of the fairing was increased from 0.016 to 0.025 inch.

Heat-shield bond-line temperatures and beryllium shingle temperatures were lower than those predicted. The hottest area at the heat-shield bond line measured only 254° F at landing, although it was predicted to be 368° F. The peak temperature of the beryllium was recorded as 1032° F, against a predicted value of 1109° F.

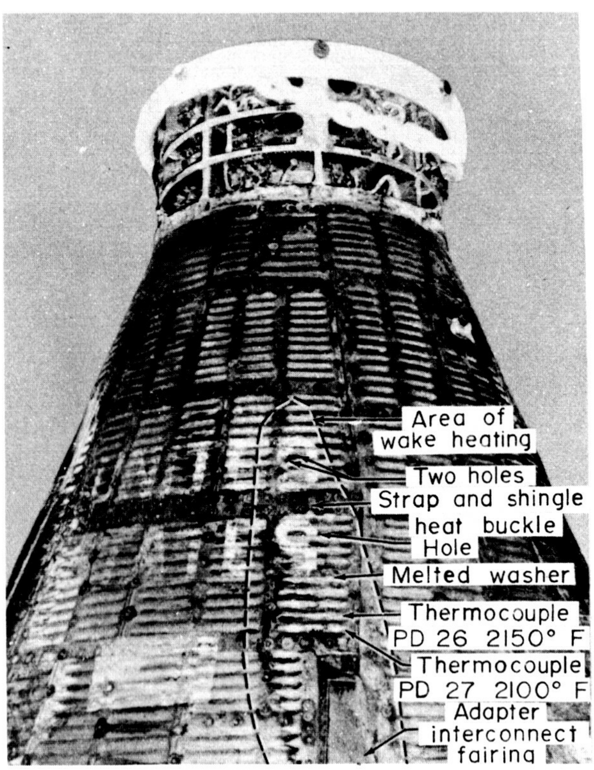

FIGURE 3–9.—Effects of reentry heating on the Gemini II spacecraft.

With the exception of the suit-circuit module in the environmental control system and that equipment which must be accessible to the crew, all other major system components in the reentry vehicle are located in accessible areas outside the cabin (fig. 3–10). This concept was used on the Gemini spacecraft to reduce the size of the pressurized cabin and to provide better access to the equipment during manufacturing assembly and during the entire test phase up to launch. This arrangement also allows manufacturing work tasks and tests to be performed in parallel, thus shortening schedules. It has the added advantage of "uncluttering" the cabin, which is the last area to be checked out prior to launch.

The suit-circuit module in the environmental control system is located in the cabin to circumvent the possibility of oxygen leakage to ambient. The module is installed in an area below the crew and, for servicing or replacement, it is accessible from the outside through a door located in the floor of the cabin. This results in a minimum of interference with other activities.

Adapters

The retrorockets are the only major components located in the retrorocket-adapter section (fig. 3–11). These critical units are isolated in this section from other equipment in the spacecraft by the reentry-vehicle heat shield and by the retrorocket blast shield located on the forward face of the equipment-adapter section.

FIGURE 3–10.—Installation of equipment in the reentry vehicle.

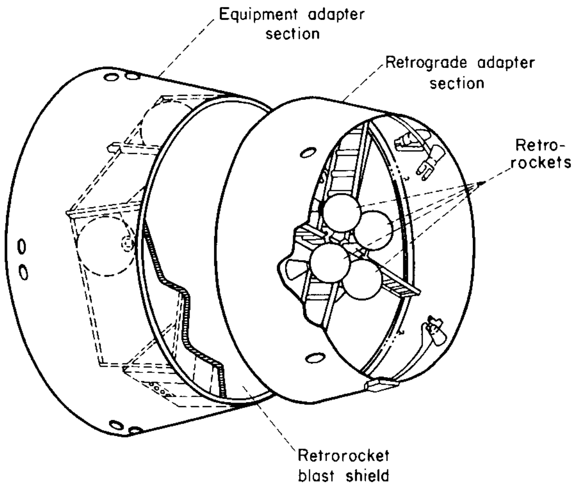

FIGURE 3-11.—Spacecraft adapter assembly.

This isolation protects these units from shrapnel in the event a tank ruptures in the equipment-adapter section. In addition, when the retrorockets are fired in salvo in the event of an abort during launch, the blast shield prevents the retrorocket blast from rupturing the tanks located in the equipment-adapter section and the launch-vehicle second-stage tank. Such an event could possibly damage the retrorocket cases before the firing was complete.

Systems not required for reentry and recovery are located in the equipment-adapter section. Most of this equipment is mounted on the aft side of the retrorocket blast shield. The systems in this area are designed and assembled as modules to reduce assembly and checkout time.

The adapter section is a conventional, externally skinned, stringer-framed structure. The skin stringers are magnesium, and the frames are aluminum alloy. The stringers incorporate passages for the environmental-control-system coolant fluid and are interconnected at the ends. This structure provides the radiator for the environmental control system, and its external surface is striped to provide temperature control within the adapter. The retrorocket blast shield is a fiber-glass sandwhich honeycomb structure. The adapter structure is designed with an ultimate factor of safety of 1.36.

Pyrotechnic Applications

As shown in figure 3-12, pyrotechnics are used extensively in the Gemini spacecraft. They perform a variety of operations including separation of structure, jettisoning of fairings, cutting tubing and electrical cables at separation planes, dead-facing electrical connectors, functioning and sequencing the emergency escape system, and initiating retrograde and reentry systems.

Because of the varied applications of the pyrotechnics, the individual designs likewise vary. However, all pyrotechnics have a common design philosophy: redundancy. All pyrotechnic devices are powered redundantly or are redundant in performing a given function, in which case the redundant pyrotechnics are ignited separately. For example, in a drogue-parachute cable cutter where it is not practicable to use redundant cutters, two cartridges, each ignited by separate circuitry, accomplish the function (see fig. 3-13); whereas, for cutting a wire bundle at a separation plane, two cutters, each containing a cartridge ignited by separate circuitry, accomplish the function redundantly.

Escape Modes

Ejection seats, as shown in figure 3-14, provide a means of emergency escape for the flight crew in the event of a launch vehicle failure on the launch pad, or during the launch phase up to 15 000 feet. Above 15 000 feet, retrorocket salvo firing is used to separate the spacecraft from the launch vehicle, after which the parachute is used to recover the spacecraft. The seats, however, remain a backup to that escape mode up to approximately 50 000 feet, and were designed and qualified for the higher altitudes and for the condition of maximum dynamic pressure. In addition, the seats provide a backup landing system in the event of a main parachute failure, and become the primary landing system if the reentry vehicle is descending over land during landing. The usual function of the seat, however, is to provide a contoured couch for the crewman and adequate restraint for the forces attendant to launch, reentry, and landing.

Extensive tests were conducted on the ejection seat system early in the program before it was qualified for flight. These tests included simulated off-the-pad ejections, sled runs at maximum dynamic pressure, and ejection from an F-106 airplane at an altitude of 40 000 feet.

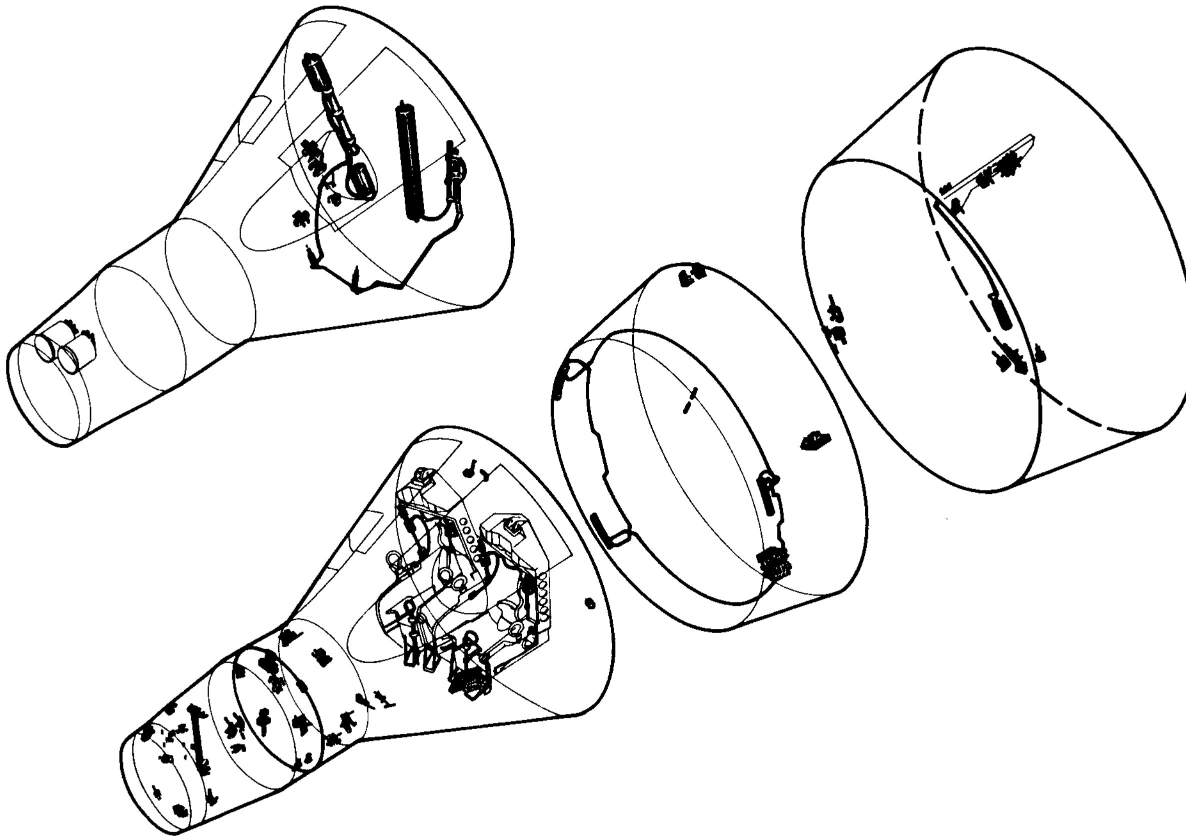

FIGURE 3-12.—Location of pyrotechnic devices in the spacecraft.

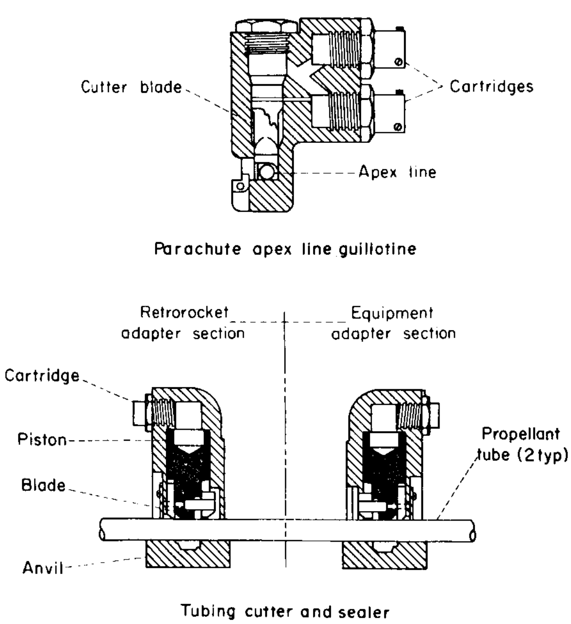

FIGURE 3-13.—Typical pyrotechnic devices used in the spacecraft.

Ejection seats were selected for the Gemini Program in lieu of other escape systems primarily for two reasons:

(1) This escape method was independent of all other systems in the spacecraft. A failure of any other system would not prevent emergency escape from the spacecraft.

(2) Ejection seats provided an escape mode for a land landing system which was planned for Gemini early in the program.

The use of hypergolic propellants in the launch vehicle also influenced the decision to use ejection seats. The reaction time to operate the system was compatible with the usage of hypergolic propellants with regard to size of the fireball and its development rate.

Safety Features

Redundancy is incorporated into all Gemini systems which affect the safety of the crew should a failure occur. Redundancy is also

FIGURE 3-14.—Gemini ejection seat.

incorporated into selected components in nonflight safety systems, with the objective of increasing probability of mission success. Crew safety has been emphasized throughout the program, both in the design and in the operational procedures. Some of the major spacecraft safety features are as follows:

(1) The spacecraft inertial guidance system serves as a backup to the launch-vehicle guidance system during the launch phase.

(2) As described earlier, ejection seats and retrorockets provide escape modes from the launch vehicle during the prelaunch and the launch phases.

(3) Two secondary oxygen bottles are provided, either of which will support the crew for one orbit and reentry in the event a loss of the primary oxygen supply occurrs. All other flight safety components in the environmental control system are redundant.

(4) In the event that a loss of reference of the guidance platform should occur, the crew has the capability of performing reentry control using out-the-window visual aids.

(5) The reentry control system is completely redundant. Two identical but completely independent systems are used, either of which has the capability of controlling the reentry vehicle through reentry. These systems are sealed with zero-leakage valves until activated shortly before retrograde.

(6) A drogue parachute, which is normally deployed at 50 000 feet altitude after reentry, backs up the reentry control system for stability until the main parachute is deployed.

(7) Ejection seats provide an escape mode if the recovery parachute fails to deploy or is damaged such that the rate of descent is excessive.

Conclusions

Although many advanced systems and concepts are used in Gemini, the capability to maneuver in space is considered to be the most important and useful operational feature incorporated in the vehicle. With this proved capability, many important mission objectives have been met, and avenues are now open for more advanced exercises in orbit. This basic technology obtained on the program provides a wealth of data for the planning and design of future space vehicles.

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4. GUIDANCE, CONTROL, AND PROPULSION SYSTEMS

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Summary

In accomplishing the Gemini Program objectives, an onboard digital computer system, an inertial platform reference system, a radar system, and control systems using hypergolic bipropellant propulsion have been developed and successfully demonstrated.

Introduction

The program objectives of long-duration, rendezvous, and controlled-reentry missions have placed special requirements on the spacecraft guidance and control systems. These objectives required maximum reliability and flexibility in the equipment. This was accomplished by utilization of simple design concepts, and by careful selection and multiple application of the subsystems to be developed.

Guidance and Control System Features

In the development of an operational rendezvous capability, the geographical constraints on the mission are minimized by providing the capability for onboard control of the terminal rendezvous phase. To complete the rendezvous objectives, the spacecraft must be capable of maneuvering, with respect to the target, so that the target can be approached and a docking or mating operation can be accomplished.

For failures in the launch vehicle, such as engine hardover and launch vehicle overrates, where effects are too fast for manual reaction, the automatic portion of the launch-vehicle malfunction-detection system switches control from the primary to the secondary system. The secondary system receives command signals from the spacecraft system for launch guidance.

To develop an operational guided reentry, onboard control has been provided. The use of the flight crew for control mode selection and command of attitudes, as well as for detection of malfunctions and selection of redundant systems, simplifies the system design and reduces the need for complicated protective interlocks.

Guidance, Control, and Propulsion Systems Implementation

The features just discussed dictated the configuration of the Gemini guidance, control, and propulsion equipment. Figure 4–1 is a block diagram of the systems.

The guidance system consists of: (1) a digital computer and an inertial measuring unit operating together to provide an inertial guidance system, and (2) a radar system which provides range, range rate, and line-of-sight angles to the computer and to the crew-station displays. The ground stations and the spacecraft are equipped with a digital command system to relay information to the spacecraft digital computer.

The control system consists of: (1) redundant horizon-sensor systems, (2) an attitude controller, (3) two translation-maneuver hand controllers, and (4) the attitude-control and maneuvering electronics which provide commands to the reentry-control and to the orbit-attitude and maneuvering portions of the propulsion system. The retrorocket propulsion engines are normally fired by a signal from the spacecraft time-reference system.

Figure 4–2 shows the arrangement of the guidance, control, and propulsion equipment in the spacecraft. The locations are shown for the thrust chamber assemblies, or engines, for the reentry control system, and for the orbital attitude and maneuver system. The attitude controller is located between the two crewmembers,

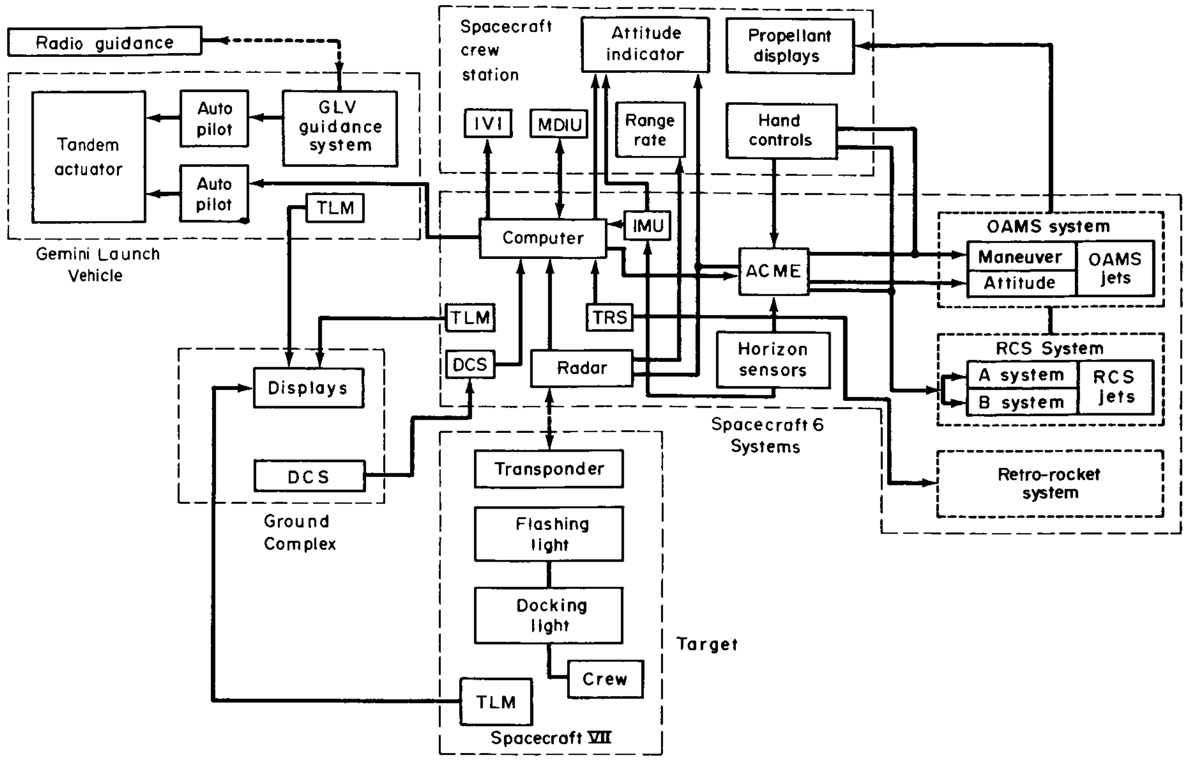


FIGURE 4–1.—Spacecraft guidance and control system.

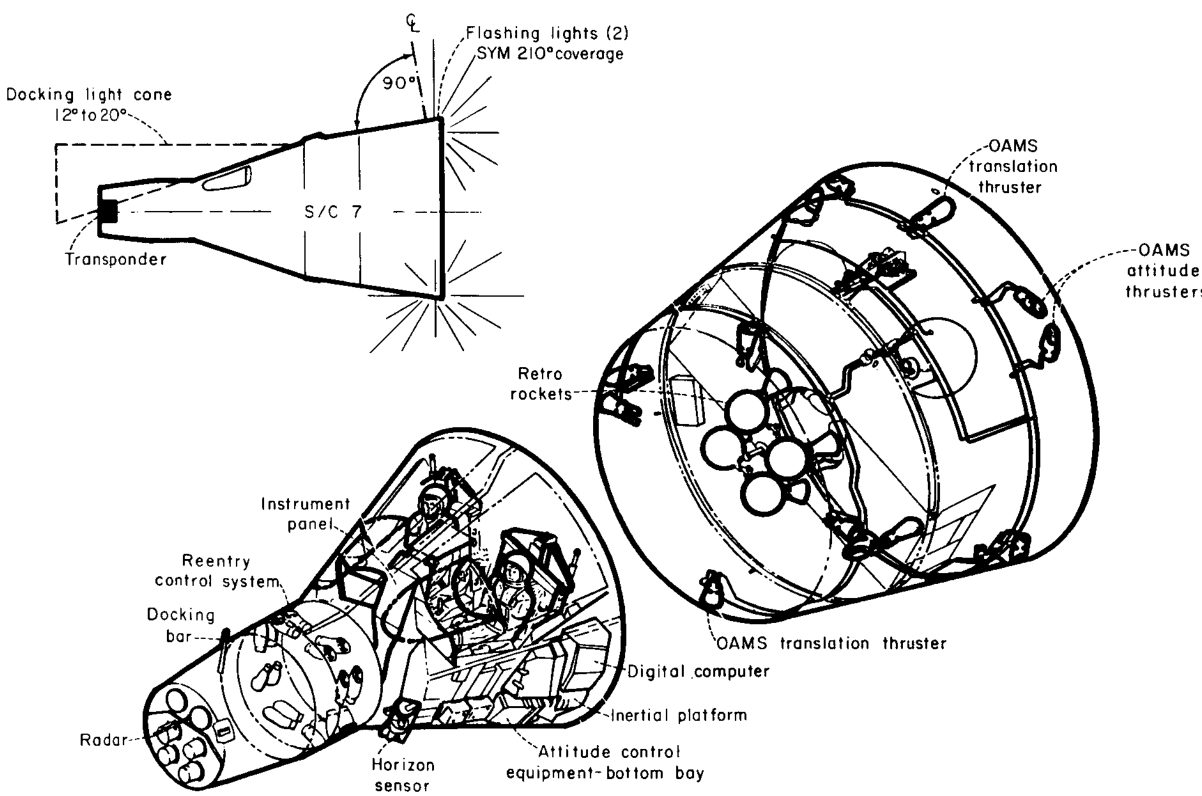


FIGURE 4–2.—Arrangement of guidance and control system components in the spacecraft.

and a translation controller is located on each side of the cabin.

Two attitude display groups, located on the instrument panel, use an eight-ball display for attitude orientation, and are equipped with three linear meter needles called flight director indicators. During launch or reentry, these needles can be used to indicate steering errors or commands and permit the flight crew to monitor the primary system performance. The needles can also be used to display attitude errors and to provide spacecraft attitude-orientation commands. The radar range and range-rate indicator used for the rendezvous missions is located on the left panel.

Gemini Guidance System

The inertial guidance system provides backup guidance to the launch vehicle during ascent. This system also determines the spacecraft orbit insertion conditions which are used in computing the velocity increment required for achieving the targeted orbit apogee and perigee. This computation is performed using the insertion velocity adjust routine.

A low-gain antenna, interferometric, pulsed radar utilizing a transponder on the target vehicle was selected to generate the information used by the computer to calculate the two impulse maneuvers required to achieve a rendezvous with the target.

The need to reference acceleration measurements and radar line-of-sight angles, as well as to provide unrestricted attitude reference to the crew, resulted in the selection of a four-gimbal stabilized platform containing three orthogonally mounted accelerometers. It provides an inertial reference for launch and reentry, and a local vertical earth-oriented reference for orbit attitude, using orbit-rate torquing.

The inertial guidance system also generates commands which, together with a cross-range and down-range steering display, are used to reach a landing point from dispersed initial conditions. Either an automatic mode, using the displays for monitoring, or a man-in-the-loop reentry-guidance technique can be flown.

The digital computer utilizes a random-access core memory with read-write, stored program, and nondestruct features. This memory has a capacity of 4096 39-bit words. The computer system provides the data processing necessary for launch guidance, rendezvous, reentry, and other calculations.

Control System

The control system (fig. 4–3) is basically a redundant rate-command system with the flight crew establishing an attitude reference and closing the loop. Direct electrical commands to the thrusters and a single-pulse-generation capability are also provided. The control system can be referenced to either of the two horizon-sensor systems to provide a redundant, low-power, pilot-relief mode. This mode controls the vehicle to the local vertical in pitch and in roll. Either horizon sensor can also supply the reference for alining the platform in a gyrocompassing-type automatic or manual mode as selected by the crew. To achieve the desired degree of reliability, the spacecraft is equipped with two separate reentry-control systems which include propellants, engines, and electrical-control capability. Either reentry-control system is adequate for controlling spacecraft attitude during the retrofire and reentry phases of the mission.

The control system was designed to operate with on-off rather than proportional commands to the propulsion engine solenoids. This simplified operation reduced the design requirements on the system electronics, solenoids, and valves, and on the dimensions and injector design of the thrust chamber assemblies, and also allowed the use of simple switch actuation for direct manual control. The engine thrust levels selected were those which would provide translation and rotational acceleration capability adequate for the completion of all tasks even with any one engine failed, and which would allow reasonable limit-cycle propellant-consumption rates for a long-period orbit operation.

Propulsion System

The orbital attitude and maneuver system (fig. 4–4) uses a hypergolic propellant combination of monomethylhydrazine and nitrogen tetroxide which is supplied to the engines by a regulated pressurization system that uses helium gas stored at 2800 psi. The choice of these propellants, along with the on-off mode of operation, minimized ignition requirements and permitted simplification of engine design. Controlled heating units prevent freezing of the

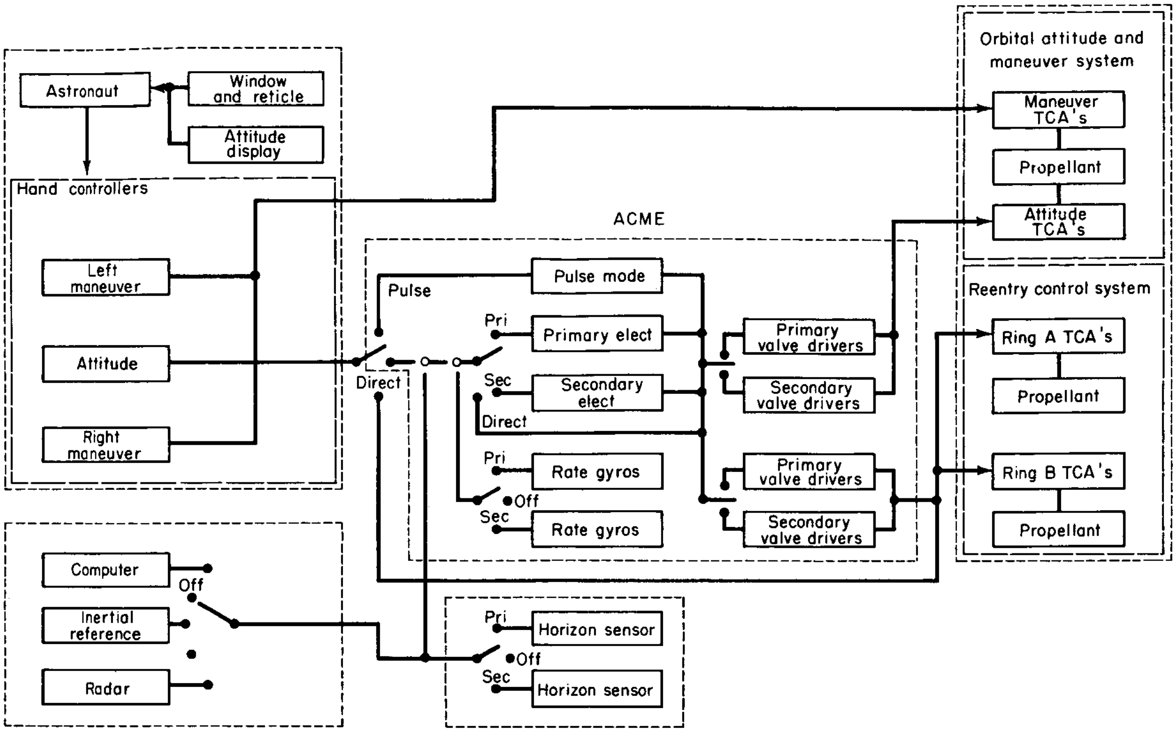
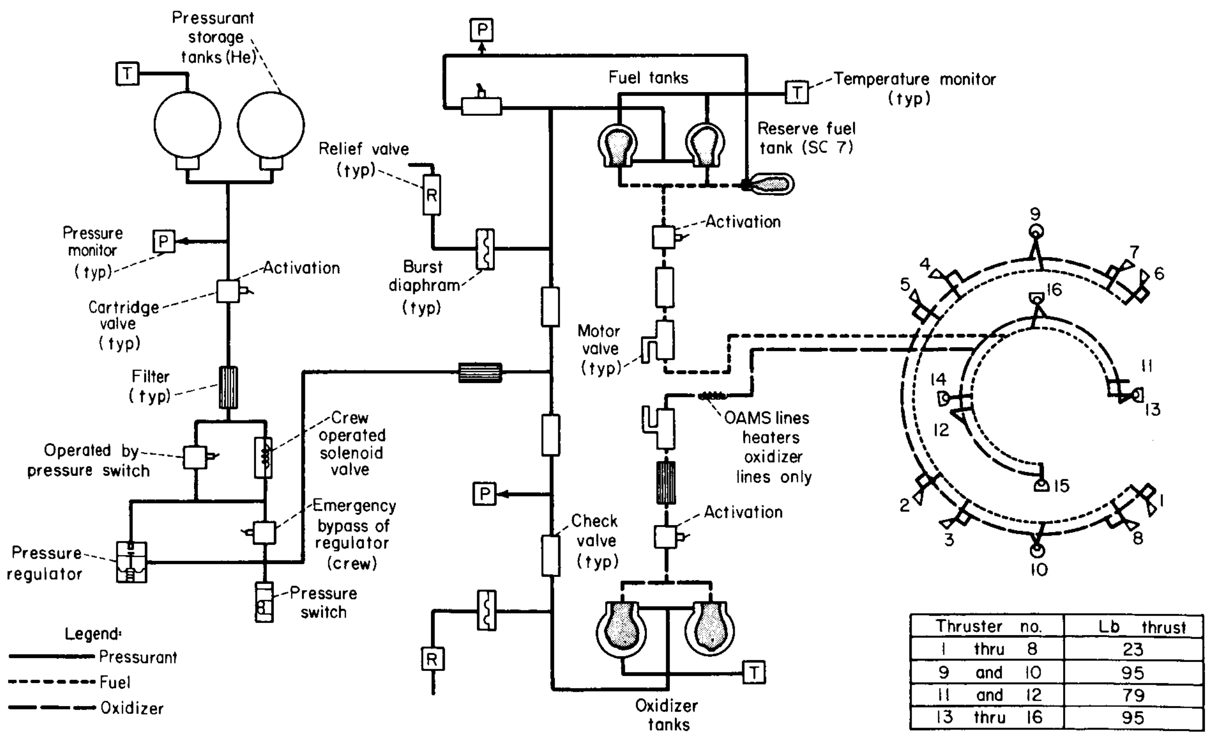


FIGURE 4–3.—Control system.



Thruster no.	Lb thrust
1 thru 8	23
9 and 10	95
11 and 12	79
13 thru 16	95

FIGURE 4–4.—Orbital attitude and maneuver system.

propellants. A brazed, stainless-steel plumbing system is used so that potential leakage points and contamination are eliminated. Positive expulsion bladders are installed in the propellant tanks. Table 4–I shows the system characteristics for steady-state engine operation.

The reentry-control system is of similar design to the orbital attitude and maneuver system. Ablative-type engines to limit reentry heating problems are used on the reentry vehicle. To reduce hardware development requirements and to permit a clean aerodynamic configuration, submerged engines, similar in design concept, are used in the orbital attitude and maneuver system.

The separate retrograde propulsion system consists of four spherical-case, polysulfide-ammonium-perchlorate, solid-propellant motors. The system is designed to assure safe reentry after any three of the four motors have been fired. The design also allows the system to be used for emergency separation of the spacecraft from the launch vehicle after lift-off.

Development Program

During the development phase, each guidance and control component underwent a comprehensive series of ground tests, both individually and after integration with interfacing components. These included engineering tests beyond the qualification level; qualification tests; and overstress, reliability, and complete systems tests at the vendor's plant. The computer and inertial-measurement-unit systems, engineering models as well as flight hardware, were integrated at the computer manufacturer's plant. Flight units were delivered to the prime contractor with the flight computer program loaded, for installation in the spacecraft prior to spacecraft systems tests. During the development of the guidance and control hardware, it was established that temperature and random vibration environments were needed as part of the predelivery acceptance tests on each flight unit to verify system capability and to establish and maintain effective quality control. A two-sigma flight environment was used to uncover conditions not apparent in the normal testing environment. Unsatisfactory conditions were corrected, and the units retested until proper operation was obtained as a means for insuring high reliability of the flight equipment.

For the Gemini guidance and control program, many special tests were developed. As an example, a special inertial component run-in test procedure (fig. 4–5) was used to determine gyro normal-trend data and also to reject unstable gyros before installation in platforms. After a 40-hour run-in period, five runup-to-runup drift measurements are obtained, followed by subsequent sets of run-in and runup-to-runup measurements. The units are rejected as having unstable characteristics if the drift trend is excessive, or if the effect of the run-in and the storage-temperature-soak on the performance of the gyro creates an unusual spread within the sets of measurement bands or the amount of shift of the bands. Tests of this nature assure adequate selection of inertial components and, along with 100 percent inspection of parts and similar techniques, have significantly improved system reliability.

TABLE 4–I.—*Gemini Propulsion System Characteristics*

Propulsion system	Number of engines	Thrust, lb_f [a]	Total impulse, lb_f-sec	Propellant weight, lb_m [b]	Specific impulse, lb_f-sec/lb_m
Orbital attitude and maneuver system	8	23	180 000	710	258
	2	79			273
	6	95			273
Reentry control system	16	23	18 500	72	283
Retrorockets	4	2490	56 800	220	255

[a] lb_f = pounds of force.
[b] lb_m = pounds of mass.

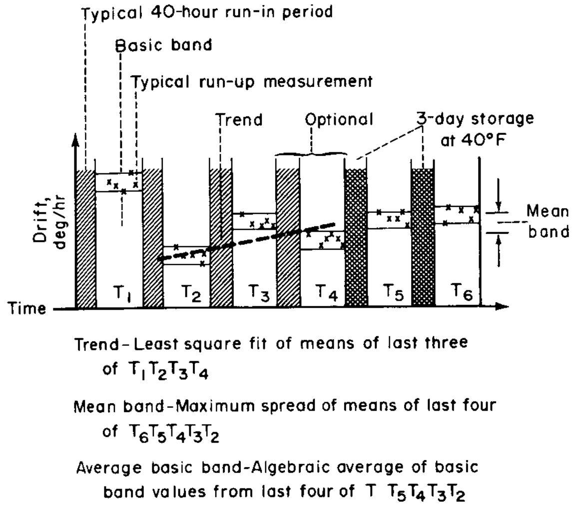


FIGURE 4-5.—Gyro test procedure.

Onboard Computer Program Development

An extensive development program for the computer-stored program was established to assure timely delivery, adequate verification, and good reflection of mission requirements. Figure 4-6 shows the basic organizational arrangement that was established. A critical feature is the monthly issue of the detailed system description authorized and provided to all users to assure common understanding, and integrated and coordinated implementation of supporting requirements. The programs are subjected to rigorous tests, including a mission verification simulation program. These tests provide dynamic simulation of the flight computer, which has been loaded with the operational program; all interfaces are exercised and all computer logic and mode operation thoroughly demonstrated. Figure 4-7 indicates a few of the detailed steps and iterations required in the development of a successful computer program. Figure 4-8 shows the computer-program development schedule, and also indicates the required lead time and development background.

Propulsion System Preflight Background

A similar, extensive ground-test program was conducted on the propulsion systems during research, development, qualification, reliability, and complete systems-test programs. A full-scale retrorocket abort test was conducted in an altitude chamber which determined the required nozzle-assembly design.

An analysis of the reentry control system and

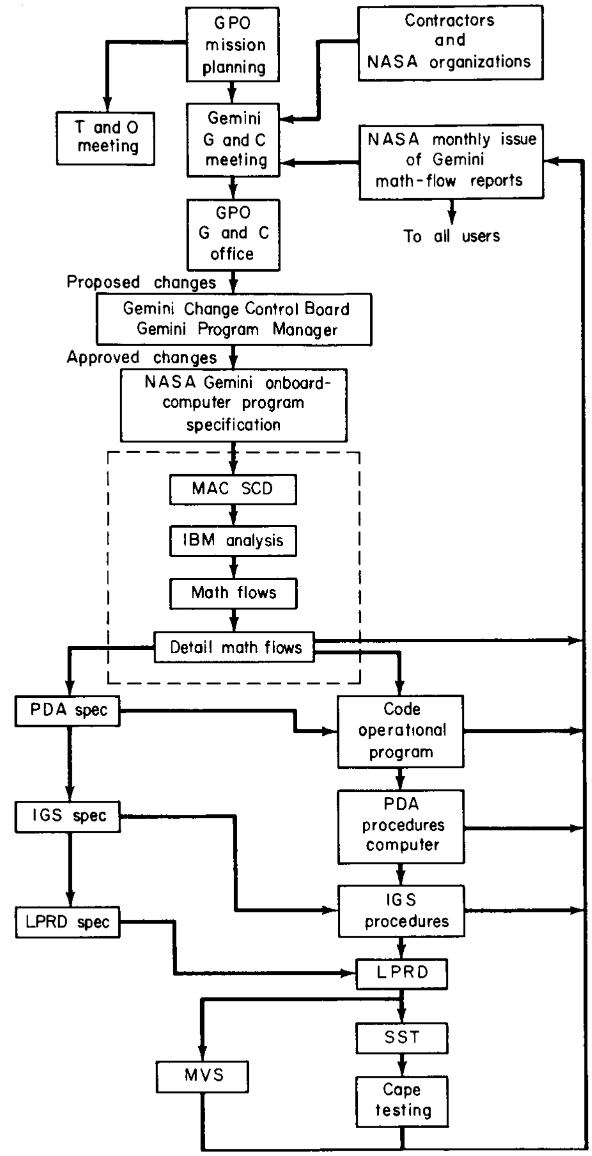


FIGURE 4-6.—Math flow control procedures and required intermediate goals.

the orbital attitude and maneuver system engine operation reveals that engine life is a function of the firing history (fig. 4-9). A long engine life results from low-percent duty cycles which, however, decrease specific impulse. To meet the duty-cycle requirements of the Gemini spacecraft, the mixture ratio of the propellants was decreased so that the combustion gas temperatures would be reduced. Major design changes also were instituted to provide greater engine integrity by permitting fuel-film-cooled walls and reorientation of the thrust-chamber-

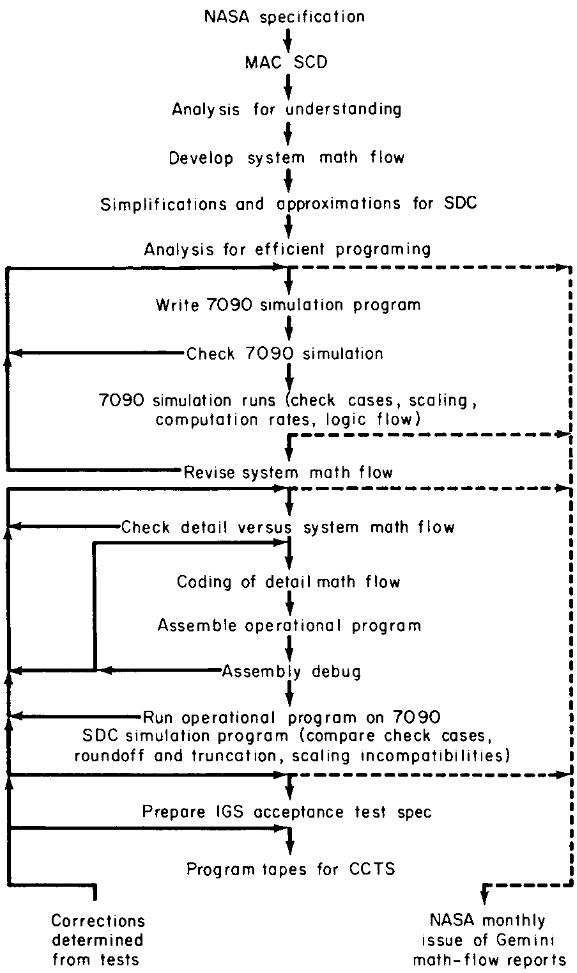


Figure 4–7.—Required intermediate goals in math flow development.

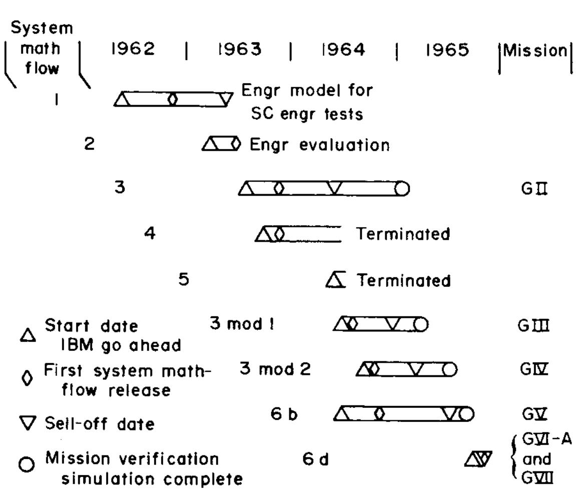


Figure 4–8.—Computer program development status chart.

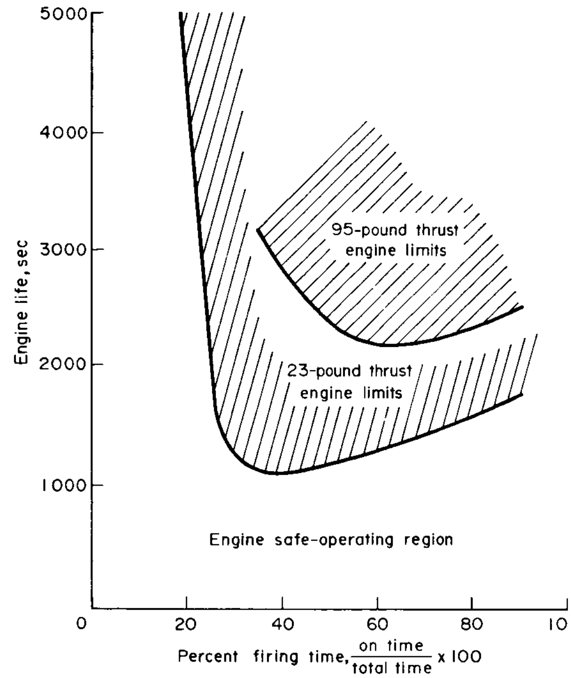


Figure 4–9.—Engine firing capability.

assembly ablative layers. Special hot-fire tests of the injector assemblies provided a basis for rejection of undesirable injectors prior to engine assembly.

Flight Performance

Guidance System Performance

The accumulated hours that the guidance and control system was in operation during the various missions are shown in table 4–II. Of all the missions, Gemini V required the maximum number of operating hours on the following systems and components:

(1) Platform—32 hours

(2) Attitude control and maneuver electronics—142 hours

(3) Primary horizon sensor—38 hours

(4) Secondary horizon sensor—45 hours

The maximum operating time required for the computer was 20 hours during the Gemini VI–A mission.

Beginning with the Gemini IV mission, the systems were subjected to repeated power-up and power-down cycling. After a periodic update of the emergency-reentry quantities for the Gemini IV computer, the flight crew was

TABLE 4–II.—*Gemini Component Operating Hours*

Component	Gemini II	Gemini III	Gemini IV	Gemini V	Gemini VI–A	Gemini VII	Total
Computer	0. 2	4. 7	6. 3	16. 0	20. 0	6	53. 2
Inertial measurement unit (platform)	. 2	4. 7	9. 7	32. 7	20. 0	14	81. 3
Attitude control and maneuver electronics	. 2	4. 7	37. 0	142. 0	25. 7	91. 5	301. 1
Horizon scanner (primary)	. 2	2. 2	33. 0	38. 4	25. 4	16. 0	115. 2
Horizon scanner (secondary)	. 2	2. 5	. 1	45. 0	. 3	0	48. 1

unable to power-down the computer system using normal procedures. Power was removed using an abnormal sequence which altered the computer memory and, therefore, prevented its subsequent use on the mission. Subsequent inflight cycling of the switch reestablished normal power operation. During postflight testing of the computer, 3000 normal cycles were demonstrated, both at the system level and with the system installed in the spacecraft. This testing was followed by a component disassembly program which revealed no anomalies within the computer, auxiliary computer power unit, or the static power supply.

The primary horizon sensor on the Gemini V spacecraft failed at the end of the second day of the mission. The mission was continued using the secondary system. The horizon-sensor head is jettisoned prior to reentry, which makes postflight analysis difficult; however, the remaining electronics which were recovered operated normally in postflight testing.

During ascent, the steering-error monitoring, along with selected navigation parameters which are available as onboard computer readouts, has given adequate information for onboard switchover and insertion go—no-go decisions. Table 4–III contains a comparison of the nominal preflight targeted apogee and perigee altitudes, with the flight values actually achieved. The table also shows, in the IVAR column, the values which would have resulted from the use of the insertion velocity adjust routine (IVAR) after insertion with the primary guidance system, and, in the IGS column, the values which would have been achieved had switchover to inertial-guidance-system (IGS) steering occurred early in stage II flight and assuming that no insertion correction had been made. A range of apogees from 130 to 191 nautical miles was targeted on the flights. Comparison of the actual values with those in the IVAR column shows that, after the Gemini III mission, the insertion velocity adjust routine would have reduced the dispersion of the actual from nominal. The IGS column shows that, had the backup system been selected, it would have given insertion conditions resulting in a safe orbit and a go-decision for all flights. Although the primary guidance was adequate on all flights, the inertial guidance system, subsequent to the Gemini III mission, would have provided guidance values closer to nominal than the primary system. The use of the insertion velocity adjust routine would have further reduced these dispersions.

Table 4–IV compares the nominal, actual, and inertial-guidance-system insertion values of total velocity and flight path angle. The actual value was computed postflight from a trajectory which included weighted consideration of all available data. The comparison indicates that, for missions after the Gemini III mission, the interial-guidance-system performance has been well within expectations.

During the orbital phases of flight, the inertial guidance system was utilized for attitude control and reference, for precise translation control, and for navigation and guidance in closed-loop rendezvous. Performance in all of these functions is dependent upon platform alinement. The alinement technique has proved to be satisfactory, with the residual errors, caused by equipment, in all axes being on the order of 0.5° or less.

TABLE 4-III.—*Comparison of Orbital Parameters at Insertion* [a]

Mission	Absolute value, nautical miles							
	Nominal		Actual		IVAR [b]		IGS [c]	
	Apogee	Perigee	Apogee	Perigee	Apogee	Perigee	Apogee	Perigee
Gemini II [d]	141	90	N/A	N/A	111 (−30)	87 (−3)	N/A	N/A
Gemini III	130. 1	87. 1	121. 0 (−9. 1)	87. 0 (−0. 1)	121 (−9. 1)	90 (2. 9)	128 (−2. 1)	78 (9. 1)
Gemini IV	161. 0	87. 0	152. 2 (−8. 8)	87. 6 (0. 6)	164. 3 (3. 3)	87. 0 (0)	163. 9 (2. 9)	87. 0 (0)
Gemini V	191. 2	87. 0	188. 9 (−2. 3)	87. 4 (0. 4)	189. 9 (−1. 3)	87. 0 (0)	192. 7 (1. 5)	86. 9 (−0. 1)
Gemini VI-A	146. 2	87. 1	140. 0 (−6. 2)	87. 0 (−0. 1)	146. 5 (0. 3)	87. 0 (−0. 1)	140. 5 (−5. 7)	87. 0 (−0. 1)
Gemini VII	183. 1	87. 1	177. 1 (−6. 0)	87. 1 (0)	181. 0 (−2. 1)	87. 0 (−0. 1)	180. 0 (−3. 1)	87. 0 (−0. 1)

[a] Values in parentheses are differences from nominal.
[b] Insertion velocity adjust routine.
[c] Inertial guidance system.
[d] Values shown from Gemini II are those targeted to exercise the IVAR routine.

TABLE 4-IV.—*Comparison of Insertion Conditions*

Mission	Insertion condition	Nominal (targeted)	Actual	Inertial guidance system
Gemini II	Total velocity, fps	25 731	25 736	25 798
	Flight path angle, deg	−2. 28	−2. 23	−2. 20
	Time from lift-off, sec	356. 5	352. 2	351. 8
Gemini III	Total velocity, fps	25 697	25 682	25 697
	Flight path angle, deg	+0. 01	+0. 01	+0. 32
	Time from lift-off, sec	358. 4	353. 8	353. 7
Gemini IV	Total velocity, fps	25 757	25 746	25 738
	Flight path angle, deg	+0. 00	+0. 04	+0. 06
	Time from lift-off, sec	355. 8	353. 8	353. 8
Gemini V	Total velocity, fps	25 812	25 805	25 808
	Flight path angle, deg	+0. 02	0. 00	−0. 01
	Time from lift-off, sec	356. 9	353. 2	353. 2
Gemini VI-A	Total velocity, fps	25 730	25 718	25 720
	Flight path angle, deg	0. 00	+0. 03	+0. 03
	Time from lift-off, sec	356. 7	358. 7	358. 7
Gemini VII	Total velocity, fps	25 806	25 793	25 801
	Flight path angle, deg	0. 00	0. 03	0. 03
	Time from lift-off, sec	358. 6	357. 0	357. 0

Figure 4–10 contains a time history of the radar digital range and computed range rates during the rendezvous approach for the Gemini VI–A mission. Rendezvous-approach criteria limit the permissible range rate as a function of range for the closing maneuver. The figure shows that, prior to the initial braking maneuver, the range was closing linearly at approximately 40 feet per second. If the effect of the braking thrust is ignored, an extrapolation of range and range rate to the nominal time of interception indicates that a miss of less than 300 feet would have occurred. A no-braking miss of this order is well within the requirements for an easy manual approach and docking with the target vehicle. Solid lock-on was achieved at 232 nautical miles and was maintained until the spacecraft had closed with the target and the radar was powered down.

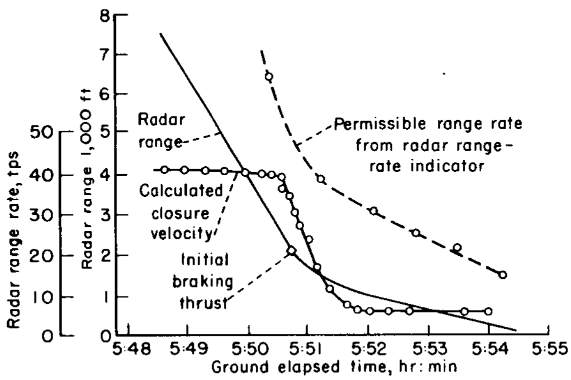


FIGURE 4–10.—Radar trajectory range comparison for Gemini VI–A and VII rendezvous.

The rendezvous performed on the Gemini VI–A/VII missions was nominal throughout. A computer simulation has been completed in which actual radar measurements were used to drive the onboard computer program. A representative value of the computed total velocity to rendezvous is compared with the telemetered values and shown in table 4–V. The close agreement verifies onboard computer operation. A trajectory simulation has verified total system operation. Using the state vectors obtained from the available tracking of the Gemini VI–A and VII spacecraft prior to the terminal phase, and assuming no radar, platform, alinement, or thrusting errors, the values of the total velocity to rendezvous and the two vernier midcourse corrections were computed. The simulated values and the actual values agree within the uncertainties of the spacecraft ground tracking for the conditions stated. The flyby miss distance resulting from this simulation was 96.6 feet.

The Gemini VI–A and VII spacecraft both demonstrated successful onboard-controlled re-

TABLE 4–V.—*Rendezvous Velocity Comparisons*

[Angle to rendezvous equals 130°]

Computer simulation			
Time from lift-off	Radar, nautical miles	Simulated ΔV_t,[a] feet per second	Data acquisition ΔV_t,[a] feet per second
5:15:20	36.20	70	69

Trajectory simulation			
First midcourse correction, incremental velocity indicators		Second midcourse correction, incremental velocity indicator	
Simulated, feet per second	Actual, feet per second	Simulated, feet per second	Actual, feet per second
3 aft 0 right/left 3 down	7 forward 5 left 7 up	2 aft 0 right/left 1 down	4 forward 6 right 2 up

[a] ΔV_t=total velocity to rendezvous.

entries. The cross-range and down-range error indications of the flight director indicator permitted both flight crews to control the spacecraft landing point to well within the expected tolerance of 12 nautical miles.

Table 4–VI is a summary of reentry navigation and guidance performance. The first line on the figure shows the inertial-guidance-system navigation error after the completion of steering at 80 000 feet and is obtained from comparisons with the best estimate trajectory. These values show that the system was navigating accurately. The next line shows the miss distances as a difference between the planned and actual landing points. The Gemini II mission had an unguided reentry from a low-altitude-insertive reentry condition which tended to reduce dispersions. Gemini III was planned and flown so that a fixed-bank angle, based on the postretrofire tracking as commanded from the ground, was held until the cross-range error was brought to zero. During this flight, however, the aerodynamic characteristics and the velocity of the retrograde maneuver performed with the orbital attitude and maneuver system differed from those expected. This difference reduced the spacecraft lifting capability to such an extent that, with the open-loop procedure flown, the targeted landing area could not be reached using the planned technique. The onboard computer predicted this condition and gave the correct commands to permit the flight crew to achieve the correct landing point. The Gemini IV reentry dispersion is that resulting from reentry from a circular orbit and being flown without guidance. The Gemini V reentry miss was caused by an incorrect quantity being sent from the ground. This quantity was used to initialize the inertial guidance system prior to reentry, and the incorrect quantity caused the inertial guidance system to show the incorrect range to the targeted landing area. The flight crew determined that a discrepancy existed in the system and, at that time, started flying a constant bank-angle reentry. The last two lines in table 4–VI indicate some of the factors causing shifts in the landing-area footprints for the Gemini missions. This table indicates generally good system performance.

Control and Propulsion System Performance

The control system has been thoroughly exercised, and all design objectives have been demonstrated. The platform mode has proved well suited for in-plane translations, for platform alinement, and for general pilot relief in busy exercises such as station keeping. The rate-command capability has been most useful for

TABLE 4–VI.—*Gemini Reentry Navigation Summary*

Flight	Gemini II	Gemini III	Gemini IV	Gemini V	Gemini VI–A	Gemini VII
	Trajectory difference, nautical miles					
Inertial guidance system—best estimate trajectory difference at 80 000 feet	1.2	0.8	([a])	[b] 1.1	[c] 2.5	2.3
Planned—best estimate trajectory difference at touchdown	18	64	47	97	[c] <7	6.6
Footprint shift, nautical miles						
Retrofire	14	48	50 [d]	5	22	41
Aerodynamics	([a])	160	([a])	([a])	([a])	40

[a] Not determined.
[b] With corrected value for ground update.
[c] Based on extrapolated radar data.
[d] Preretrofire and retrofire.

translations, such as retrofire and rendezvous maneuvers, and for damping aerodynamic oscillations during reentry in order to ease the reentry guidance task. Pulse mode has provided the fine control necessary for manual platform alinements, for station keeping, and for experiments and maneuvers requiring accurate pointing. Reentry rate command has been used on the Gemini II and IV missions for reentry control. The wide deadbands mechanized in this mode conserve propellants while retaining adequate control.

The horizon mode has been utilized extensively to provide pilot relief through automatic control of pitch and roll attitude based upon horizon-sensor outputs. Performance, in general, has been excellent, although several instances of susceptibility to sun interference have been noted. On the Gemini VI–A mission, this mode operated unattended for approximately 5 hours while the flight crew slept. The final or direct mode has been utilized effectively by the crew when they wished to perform a maneuver manually with the maximum possible control authority.

Typical retrofire maneuver performance is shown in table 4–VII. During the first manned mission, the Gemini III spacecraft retrofire maneuver was performed with the roll channel in direct mode and with the pitch and yaw channels in rate command. This method of operation provided additional yaw authority in anticipation of possible high-disturbance torques. Only nominal torques were experienced, however, and the remaining missions utilized rate-command mode in all axes. Attitude changes during retrofire have resulted in velocity errors well within the lifting capability of the spacecraft and would not have contributed to landing-point dispersions for a closed-loop reentry. A night retrofire was demonstrated during the Gemini VI–A and VII missions. In summary, the performance of the attitude-control and maneuvering electronics has been exceptional during ground tests as well as during all spacecraft flights.

The Gemini III spacecraft demonstrated the capability to provide orbital changes which included a retrograde maneuver that required a 111-second firing of the aft engines in the orbital attitude and maneuver system. The

TABLE 4–VII.—*Typical Gemini Retrofire Maneuver Velocity Comparison*

[Values in parentheses are differences from nominal]

Flight	ΔX, feet per second	ΔY, feet per second	ΔZ, feet per second	Total
Gemini VI–A	−308	0	117	329.5
	(1)	(−1)	(−1)	(.6)
Gemini VII	−296	0	113	316.8
	(2)	(3)	(−1)	(1.6)

propulsion system maneuvering capability was used for the rendezvous maneuvers during the Gemini VI–A mission.

There have been two flights with known anomalies which could definitely be attributed to the propulsion systems. The two yaw-left engines in the orbital attitude and maneuver system of the Gemini V spacecraft became inoperative by the 76th revolution, and neither engine recovered. Rate data also showed that other engines exhibited anomalous behavior but subsequently recovered, and this suggested the cause to be freezing of the oxidizer. During this flight the heater circuits had been cycled to conserve power. During the Gemini VII mission, the two yaw-right engines in the orbital attitude and maneuver system were reported inoperative by the crew approximately 283 hours after lift-off. Postflight analysis of rate data verified this condition. However, because these engines are not recovered, failure analysis is difficult, and inflight testing was insufficient to identify the cause of the failure on Gemini V and VII. Further studies are being conducted in an attempt to isolate the cause.

On the Gemini IV spacecraft, one of the pitch engines in the reentry control system was inoperative; however, postflight examination revealed a faulty electrical connector at the mating of the reentry-control-system section and the cabin section.

The propellant quantity remaining in the spacecraft during the flight is determined by calculating the expanded volume of the pressurizing gas using pressure and temperature measurements. Flight experience has shown that, due to inaccuracies in this quantity-gaging system, a significant quantity of propellants must be reserved for contingencies. A reserve propellant tank has been added to assure that a known quantity of propellant remains even though the main tanks have been depleted, thus insuring the capability of extending the mission to permit recovery in the planned primary landing area.

Conclusions

As a result of developing onboard capability, greater flexibility in mission planning and greater assurance of mission success have been achieved. In addition, information obtained from systems such as the inertial guidance system and the radar system has significantly improved the knowledge of the launch, orbital, and reentry phases of the mission and has made a thorough analysis more practical.

For the guidance, control, and propulsion systems, the design, development, implementation, and operating procedures have been accomplished, and the operational capabilities to meet the mission requirements have been successfully demonstrated.

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5. COMMUNICATIONS AND INSTRUMENTATION

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Summary

The Gemini spacecraft communications and instrumentation system consists of subsystems for voice communications and tracking, a digital command system, recovery aids, a data acquisition system, and a data transmission system. Development and qualification testing were completed rapidly to meet launch schedules, and the engineering problems encountered were solved in an expeditious manner. The first seven missions have proved the overall adequacy of the system design. The problems encountered have not prevented the fulfillment of mission objectives and have not interfered significantly with mission operations. Although some telemetry data have been lost, sufficient data support has been provided for design verification and operational purposes.

Introduction

The Gemini spacecraft communications system consists of subsystems for voice communications and tracking, a digital command system, a telemetry transmission system, and various recovery aids. The instrumentation system consists of the data acquisition system and the data transmission system. Experience with Project Mercury was a valuable aid during system design and gave increased confidence in design margin calculations which have since been borne out by successful flight experience. A communications-system block diagram is shown in figure 5–1, and equipment locations are illustrated in figure 5–2.

Communications System

Voice communications in the Gemini spacecraft employ an integrated system which has as the central component a voice-control-center package which performs the function of an audio-distribution system.

The primary voice communications system for the Gemini spacecraft is the very-high-frequency system. The redundant transmitter-receiver units transmit and receive on a frequency of 296.8 megacycles with an output power of 3 watts. Conventional double-sideband amplitude modulation with speech clipping is employed. The units are mounted in the unpressurized reentry-section equipment bay, and either may be selected.

The very-high-frequency antenna system consists of quarter-wave monopoles mounted in selected locations (fig. 5–2) to provide the satisfactory radiation patterns for each mission phase. Flight experience has shown that circuit-margin calculations were adequate. Two antenna systems are used while in orbit, one predominantly during stabilized flight and one for drifting flight. Special tests conducted during the Gemini V mission verified the proper antenna selection for drifting and oriented modes of flight which had previously been derived from radiation-pattern studies. The very-high-frequency ground-to-air voice quality has been excellent. Even during the launch phase with the very high ambient noise level in the cabin area, the flight crews have reported high intelligibility. Although operationally satisfactory, the intelligibility of the air-to-ground link has not been as good, especially during the time of high launch-vehicle noise following lift-off. There are instances of communication fades encountered during drifting flight when regions of high attenuation are encountered in the antenna radiation patterns and when multipath interference is encountered at low antenna look angles. Interference from atmospheric effects, even storms, has been of

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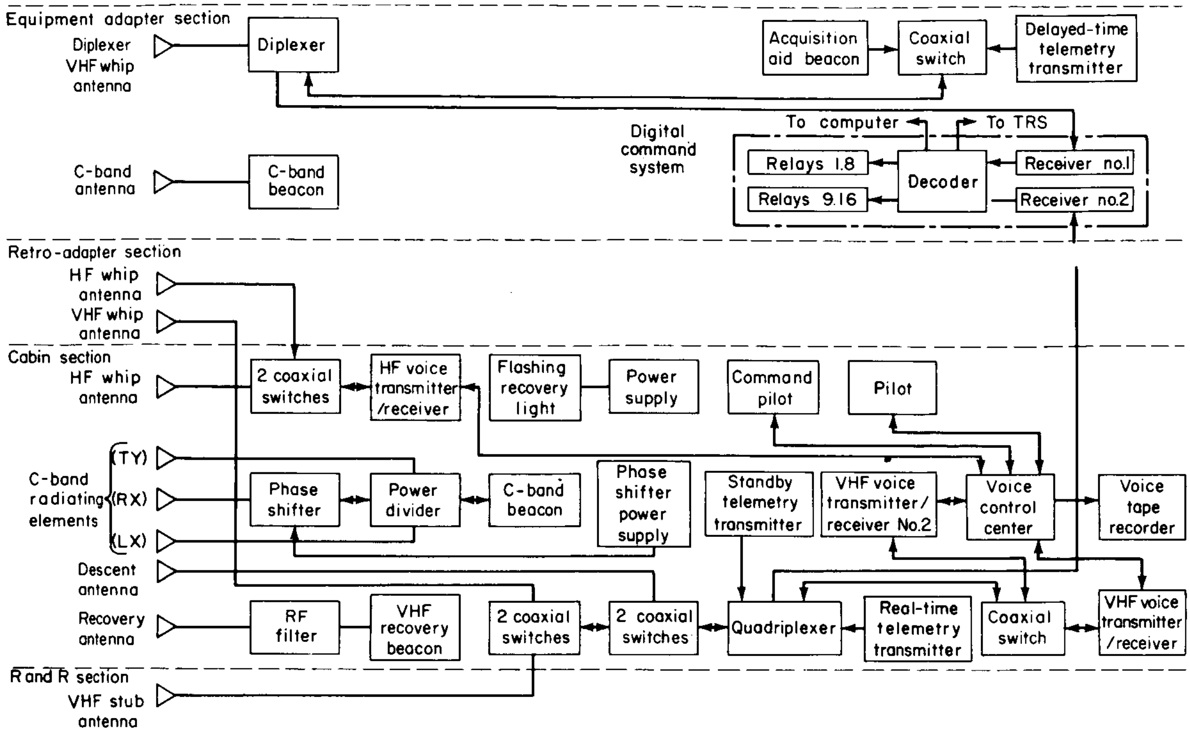

FIGURE 5-1.—Communications system.

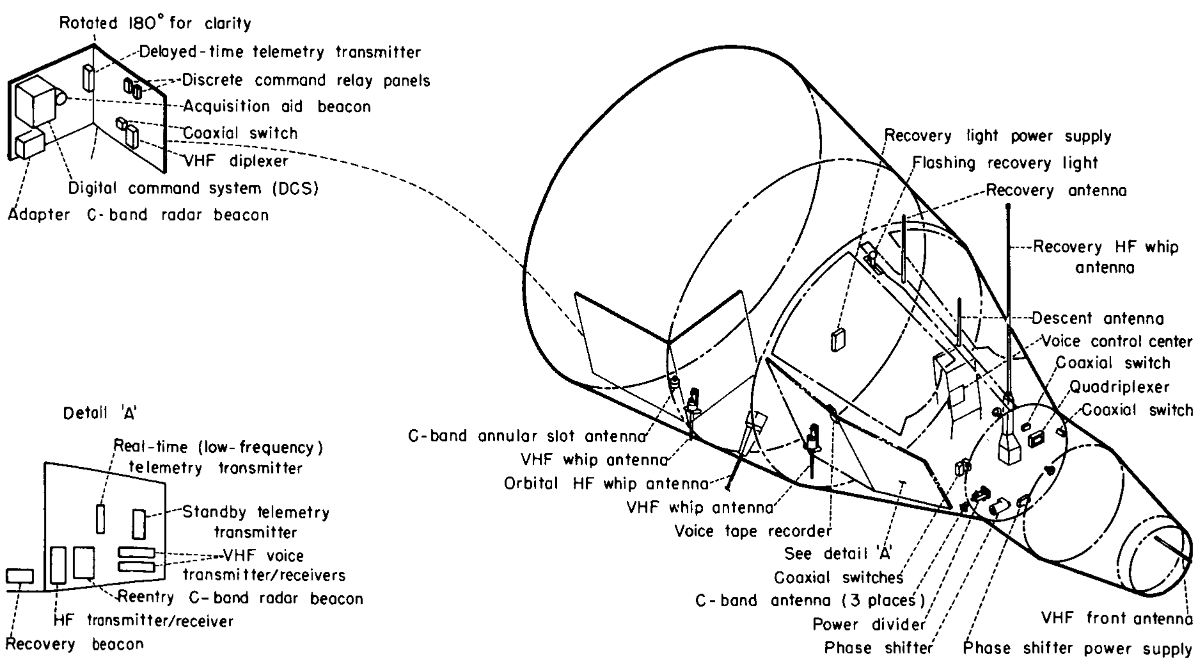

FIGURE 5-2.—Location of Gemini spacecraft communications equipment.

very minor significance. All of these effects combined have not significantly interfered with mission operations.

A high-frequency voice transmitter-receiver is included in the spacecraft communications system to provide an emergency postlanding long-distance voice and direction-finding communications link for use if the landing position of the spacecraft is unknown. It can also be used for beyond-the-horizon transmissions in orbit, and as a backup to the very-high-frequency communications link. The high-frequency link operates on a frequency of 15.016 megacycles with an output power of 5 watts. Manmade electromagnetic interference is of primary concern to communication links utilizing the high-frequency range for long-range transmission. Many occurrences of interference at the Gemini frequency are reported during each mission. The need for the high-frequency communications link would occur with land-position uncertainties of several hundred miles or greater. However, the high-frequency direction-finding equipment is usually tested during the postlanding phase, and postlanding high-frequency voice communications between Gemini VI–A and the Kennedy Space Center were excellent. Transmissions from Gemini VI–A and VII were received with good quality at St. Louis, Mo. Many good direction-finding bearings were obtained on Gemini VI–A and VII. Figure 5–3 is an illustration of bearings made on Gemini VI–A.

The spacecraft tracking system consists of two C-band radar transponders and one acquisition-aid beacon. One radar transponder is mounted in the adapter for orbital use, and the other in the reentry section for use during launch and reentry (fig. 5–2). The adapter transponder peak-power output is 600 watts to the slot antenna mounted on the bottom of the adapter. The reentry transponder peak-power output is 1000 watts to the helix antenna system mounted on the reentry section. The power is divided and fed to three helix antennas mounted at approximately 120° intervals around the conical section of the reentry assembly, forward of the hatches. Flight results have been very satisfactory. The ground-based C-band radar system is capable of beacon-tracking the spacecraft completely through the reentry-plasma blackout region, and has done so on more than one occasion.

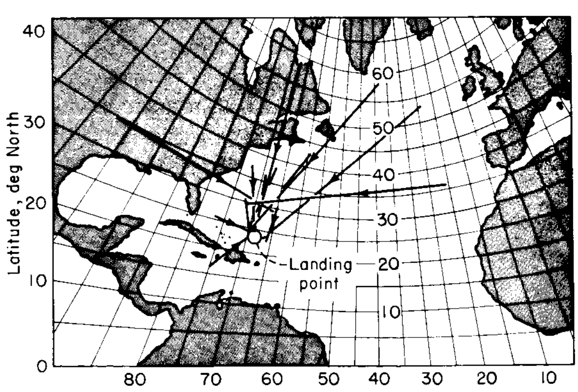

FIGURE 5–3.—HF–DF bearings to Gemini VI–A after landing.

A 250-milliwatt acquisition-aid beacon is mounted in the adapter section. The beacon signal is used by the automatic antenna-vectoring equipment at the ground stations to acquire and track the spacecraft prior to turning on the telemetry transmitters. This system has operated normally on all flights.

The digital command system aboard the spacecraft consists of a dual-receiver single-decoder unit and two relay packages mounted in the equipment section of the adapter. The two receivers are fed from different antennas, thus taking advantage of complementary antenna patterns which result in fewer nulls. The receiver outputs are summed and fed to the decoder, which verifies and decodes each command, identifies it as being a real-time or stored-program command, and either commands a relay operation or transfers the digital data, as indicated by the message address. The decoder sends a message-acceptance pulse, via the telemetry system, to the ground when the message is accepted by the system to which it is addressed. The probability of accepting an invalid message is less than one in a million at any input signal level. The stored-program commands are routed to the guidance computer or to the time reference system for update of the time-to-go-to-retrofire or equipment reset.

The digital command system has performed most satisfactorily in flight. The ground stations are programed to repeat each message until a message-acceptance pulse is received; therefore, the occasional rejection of a command because of noise interference or other reasons has not caused a problem. Completion of the transmission is an indication that all commands have been accepted at the spacecraft.

The telemetry transmission system consists of three transmitters: one for real-time telemetry, one spare transmitter, and one for delayed-time recorder playback. Either the real-time or the delayed-time signal can be switched to the spare transmitter by the digital command system or by manual switching. Recorder playback is also accomplished by command or by manual switching. The transmitters are frequency-modulated with a minimum of 2 watts power output, and solid-state components are used throughout. Transmitter performance has been normal during all flights through Gemini VII. The delayed-time transmitter on Gemini III failed a short time before launch; however, the spare transmitter functioned throughout the short mission. The telemetry signal strengths received at the network stations have been adequate. However, some data have been lost by the ground stations losing acquisition and failing to track the spacecraft. This was usually due to signal fades, which were sometimes caused by localized manmade electromagnetic interference or multipath signal cancellation.

A recovery beacon is energized when the spacecraft goes to two-point suspension on the main parachute and transmits until the recovery is complete. A flashing light mounted on the top of the spacecraft deploys after landing and can be turned on by the crew. Direction finding is sometimes employed using continuous-wave transmission from the very-high-frequency voice transmitter, and, if necessary, a signal is available from the high-frequency voice transmitter for long-range direction finding.

The recovery beacon transmits a pulse plus continuous-wave signal on the international distress frequency. The signal was specifically designed to be compatible with the AN/ARA–25 and the search and rescue and homing (SARAH) direction-finding systems but is also compatible with almost all other direction-finding equipment. The transmission range is limited to horizon distances and, therefore, limited by the altitude of the recovery aircraft. The Gemini recovery-beacon signal is received by all aircraft within line of sight and has been received by aircraft at distances up to 200 nautical miles.

The flashing recovery light is used as a visual location aid during the postlanding phase. It is powered by a separate 12-hour battery pack composed of several mercury cells, and can be turned on and off by the crew. The flashing rate is approximately 15 flashes per minute.

The performance of all communications systems has met or exceeded the design criteria. Ground acquisition of both voice and telemetry signals has always occurred on the approach horizon and has been maintained with excellent circuit margins to the departing horizon. No significant design objectives remain to be achieved.

Instrumentation System

Three instrumentation systems (table 5–I) have been flown. These were the PAM–FM–FM instrumentation and telemetry system used only on spacecraft 1, the standard production system supplemented by a special instrumentation system on spacecraft 2, and the standard production system used on spacecraft 3 and subsequent spacecraft.

TABLE 5–I.—*Instrumentation Systems*

Spacecraft	Equipment type	Measurements
Gemini I	PAM–FM–FM	Structural temperatures, structural vibrations, and cabin acoustic noise
Gemini II	Special and standard pulse code modulation	Structural temperatures, structural vibrations, and crewman simulator functions
	Analog tape recorder	Structural vibrations
	Cameras	Instrument panel and window views
Gemini II to Gemini VII	Standard pulse code modulation	Operational and diagnostic measurements

The PAM–FM–FM system was employed on spacecraft 1 to determine the Gemini spacecraft launch environment. This system measured the noise, vibration, and temperature characteristics of the spacecraft during launch and orbital flight. Excellent data were obtained throughout the mission.

To obtain launch and reentry environment data in addition to flight performance data on spacecraft 2, it was necessary to use special instrumentation as well as the standard production instrumentation system. Data on crewman simulator functions, structural dynamics measurements, many of the temperature measurements, and photographic coverage of the instrument panels and of the view out of the left-hand window were obtained. These contributed materially to evaluation of other onboard systems.

The spacecraft instrumentation and recording system also serves as a significant tool in the checkout of the spacecraft during contractor systems tests and Kennedy Space Center tests. During flight, the standard instrumentation system provides operational data and facilitates diagnostic functions on the ground.

The instrumentation system (shown in fig. 5–4) is composed of a data acquisition system and a data transmission system. Instrumentation packages contain signal-conditioning modules which convert inputs from various spacecraft systems into signals which are compatible with the data transmission system. Redundant dc-to-dc converters provide controlled voltages for those portions of the instrumentation and

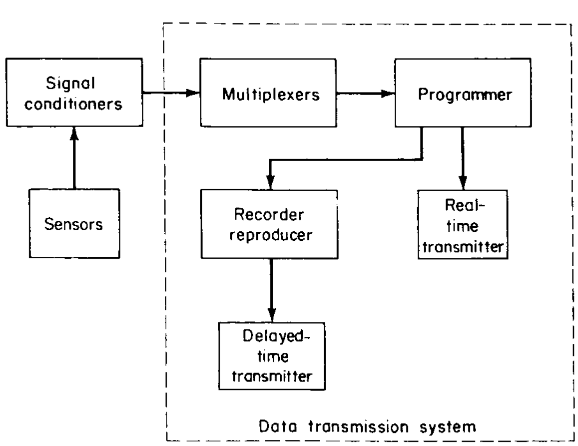

FIGURE 5–4.—Block diagram of the instrumentation system.

recording system which require a constant input for operation. Pressure transducers, temperature sensors, accelerometers, a carbon-dioxide partial-pressure sensing system, and synchro-repeaters are provided to convert physical phenomena into electrical signals for handling by the system.

Biomedical instrumentation sensors were attached to each astronaut's body, and signal conditioners were contained within the astronaut's undergarments. Physiological parameters were supplied by these sensors and signal conditioners to the biomedical tape recorders and to the data transmission system for transmission.

The delayed-transmission recorder/reproducer records data during the time the spacecraft is out of range of the worldwide tracking stations. When the spacecraft is within range of a tracking station, the recorder/reproducer will, upon receiving the proper signal, reverse the tape direction and play back the recorded data at 22 times the real-time data rate.

The data transmission system is composed of the pulse-code-modulation (PCM) multiplexer-encoder, the tape recorder/reproducer, and the telemetry transmitters. The PCM multiplexer-encoder includes the PCM programer, two low-level multiplexers, and two high-level multiplexers. The programer provides the functions of data multiplexing, analog-to-digital conversion, and digital data multiplexing, while also providing the required timing and sampling functions needed to support the high-level and low-level multiplexers. The two high-level multiplexers function as high-level analog commutators and on-off digital data multiplexers, providing for the sampling of 0-to-5-volt dc measurements and bilevel (on-off) events. The two low-level multiplexers function as differential input analog commutators and provide for the sampling of 0-to-20-millivolt signals.

The PCM multiplexer-encoder is made up of plug-in multilayered motherboards. Each motherboard contains numerous solid-state modules which employ the cordwood construction technique, and each module performs specific logic functions. The data transmission system contains approximately 25 000 parts, giving a component density of approximately 37 000 parts per cubic foot, or over 20 parts within each cubic inch.

The PCM system accepts 0-to-20-millivolt signals, 0-to-5-volt dc signals, bilevel event signals, and digital words from the onboard computer and time reference systems, as shown in table 5–II. The total system capacity of 338 measurements has been more than adequate, since the manned missions have not required more than 300 measurements.

To meet program objectives, three significant problems had to be overcome. These are shown in table 5–III.

The PCM tape recorder would not perform properly at the specification vibration levels during the development tests. This problem was one of the most difficult development problems encountered. The final solution required over 10 major modifications, numerous minor modifications, and a special ball-socket vibration-isolation mount. After the Gemini II flight-vibration data were obtained, a vibration specification was established for the operation of the PCM tape recorder and was met.

TABLE 5–II.—*Instrumentation System Capacity* [a]

Number of signals	Type of signal	Sample rate, samples/sec
6	0–20 mV dc	640
6	0–20 mV dc	160
9	0–20 mV dc	80
16	0–20 mV dc	1.25
48	0–20 mV dc	.42
3	0–5 V dc	40
3	0–5 V dc	20
6	0–5 V dc	10
96	0–5 V dc	1.25
120	Bilevel	10
1	Digital	10
24	Digital	.416

[a] Available channels:

Analog	193
Bilevel	120
Digital	25
Total	338

During spacecraft systems tests, switching functions caused inductive transients on the voltage supply buses, introducing spurious resets into the multiplexers which caused a loss of data. A simple modification which inserted diodes in the reset drive lines eliminated most of the problem. Unfortunately, this modification lowered the reset drive voltage to a level which made the multiplexers susceptible to "lockup," or not sending data out to the PCM programer in the proper sequence. The reset drive and counterdrive circuitry in the programer and the remote multiplexers were modified and flown in spacecraft 3 and subsequent spacecraft.

During spacecraft 3 testing, it was found that the combination of the Gemini PCM prime-frame format with the bit jitter of the tape recorder would not allow optimum recovery of the recorded data. By changing the output of the tape recorder from non-return-to-zero-change to non-return-to-zero-space, recovery of the dump data during high bit-jitter periods was enhanced by a factor of 15 to 1. The non-return-to-zero-space code tends to give an output which is optimum for the Gemini data format and also minimizes the sync adjustment sensitivities of the PCM ground stations.

For all Gemini missions to date, the instrumentation system has performed exceptionally well. Out of the 1765 measurements made, only 10 parameters were lost, or 0.57 percent. A summary of the real-time telemetry data actually received for Gemini missions II through VII reveals that the usable data exceed 97.53 percent.

TABLE 5–III.—*Instrumentation Problem Areas*

Equipment	Hardware phase	Difficulty	Corrective action
Pulse-code-modulation multiplexer-encoder	Spacecraft systems test	Spurious resets	Redesign circuitry
Tape recorder	Development	Failed in vibration	Major modifications made
Tape recorder	Spacecraft systems test	"Bit jitter"	Pulse-code-modulation output code changed

Table 5–IV summarizes the delayed-time data quality. During orbital flight, 416 data dumps have been made. Of these, 135 data dumps have been processed and evaluated. The results show that 96.57 percent of the evaluated data was completely acceptable.

TABLE 5–IV.—*Summary of Delayed-Time Pulse-Code-Modulation Data Dumps* [a]

Dumps		Percent of data retrieved from evaluated dumps
Total	Evaluated	
416	135	96. 57

[a] Data for 5 missions.

The failures which occurred during Gemini flights are shown in table 5–V. The majority of the problems are associated with the playback tape recorder, the most significant of which was due to a playback clutch ball-bearing seizure. This bearing seizure resulted from a design deficiency which allowed the bearing shield to cut into an adjacent shoulder, generating metallic chips which entered the bearing itself. Modifications to correct this problem have been made in the remaining flight recorders. The other failures could not be verified because the failure modes could not be reproduced, or because the suspect components were jettisoned prior to reentry.

The Gemini instrumentation system has met the mission requirements on all flights and has been of significant importance in preflight checkout of spacecraft systems. The design criteria which established parameter capacity, sampling rate, circuit margin, et cetera, proved to be completely adequate throughout the missions to date. The instrumentation system accuracy of 3 percent has been more than adequate to satisfy the program requirements. The problems encountered to date have all been resolved, and no major objectives remain to be achieved.

TABLE 5–V.—*Instrumentation Flight Failures*

Flight	Failure	Effect	Corrective action
Gemini IV	Recorder stopped running	Lost data after 2000 feet during descent and landing	Cause undetermined (possible bearing seizure)
Gemini V	Oxide flaked off tape	Poor delayed-time data, revolutions 30 through 45	Improved assembly procedures
Gemini VI–A and Gemini VII	Recorder bearing seized	Lost delayed-time data	Rework bearing clearances
Gemini VI–A	Possible solid-state switch malfunction	Lost 5 parameters, regained after retrofire	Cause undetermined (still under investigation)
Gemini VII	Transducer stuck at 910 psi	After 170 hours lost data on reactant-supply-system oxidizer supply pressure	None (failure analysis impossible)

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6. ELECTRICAL POWER AND SEQUENTIAL SYSTEMS

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Summary

The electrical and sequential systems successfully supported the Gemini spacecraft in meeting the objectives of the first seven missions. The development of a fuel-cell electrical-power system was required to meet the 8-day and 14-day objectives of the Gemini V and VII missions.

Introduction

The development of an electrical system to support the Gemini spacecraft long-duration missions required a significant advance in the state of the art. Conventional battery systems were used in some missions, but, for the more complex rendezvous and long-duration missions, a new power system was required. An ion-exchange-membrane fuel cell was chosen as the new power source, and, to take advantage of the available space in the spacecraft, fuel-cell consumables, oxygen and hydrogen, were stored at cryogenic temperatures in a supercritical state. The new fuel-cell power system has flown on the Gemini V and VII missions, and has met all the spacecraft requirements.

A major step forward was taken in the design of the sequential system of the spacecraft by inserting the man in the loop. The resulting sequential system is straightforward and more reliable. It has performed successfully on all flights.

Electrical System

The electrical power system of the Gemini spacecraft, shown in figure 6–1, is a 22- to 30-Vdc two-wire system with a single-point ground to the spacecraft structure. During the launch and orbital phases of the mission the main bus power has been supplied by either silver-zinc batteries or by a fuel-cell power system. The main bus power sources, which will be discussed later, are placed on the bus by relays powered from a common control bus, and through diodes. The diodes prevent a shorted battery or shorted fuel-cell stack, or a short in the line to bus, from being fed by all remaining power sources. During the reentry and postlanding phases of the mission, the main bus power is supplied by four 45-ampere-hour, silver-zinc batteries. Each battery is first tested, then placed directly on the bus by a switch. Systems that require alternating current or regulated direct current have special inverters or converters tailored to their own requirements. Circuit protection in the spacecraft is provided mainly by magnetic circuit breakers, although fuses are used in branches of heater circuits and in the inertial guidance system. Fusistors are used in the squib-firing circuits.

The isolated bus system contains two completely redundant squib-firing buses connected through diodes to a third common-control bus, and it is powered by special batteries capable of a 100-ampere discharge rate. This bus is separate from the main bus to prevent transient spikes from reflecting into systems on the main bus. Such transients, which might come from thruster solenoids or squib firings, could damage the computer or other sensitive components of the spacecraft. The main and other buses can be linked together by the bus-tie switches, if necessary. This was done on spacecraft 7 to conserve squib battery power.

Power Sources

Batteries were used as the only source of power on three of the five manned orbital Gemini missions completed thus far (table 6–I). The development of the fuel-cell system was completed in time to meet the electrical power requirements of the 8-day mission of Gemini V and the 14-day mission of Gemini VII.

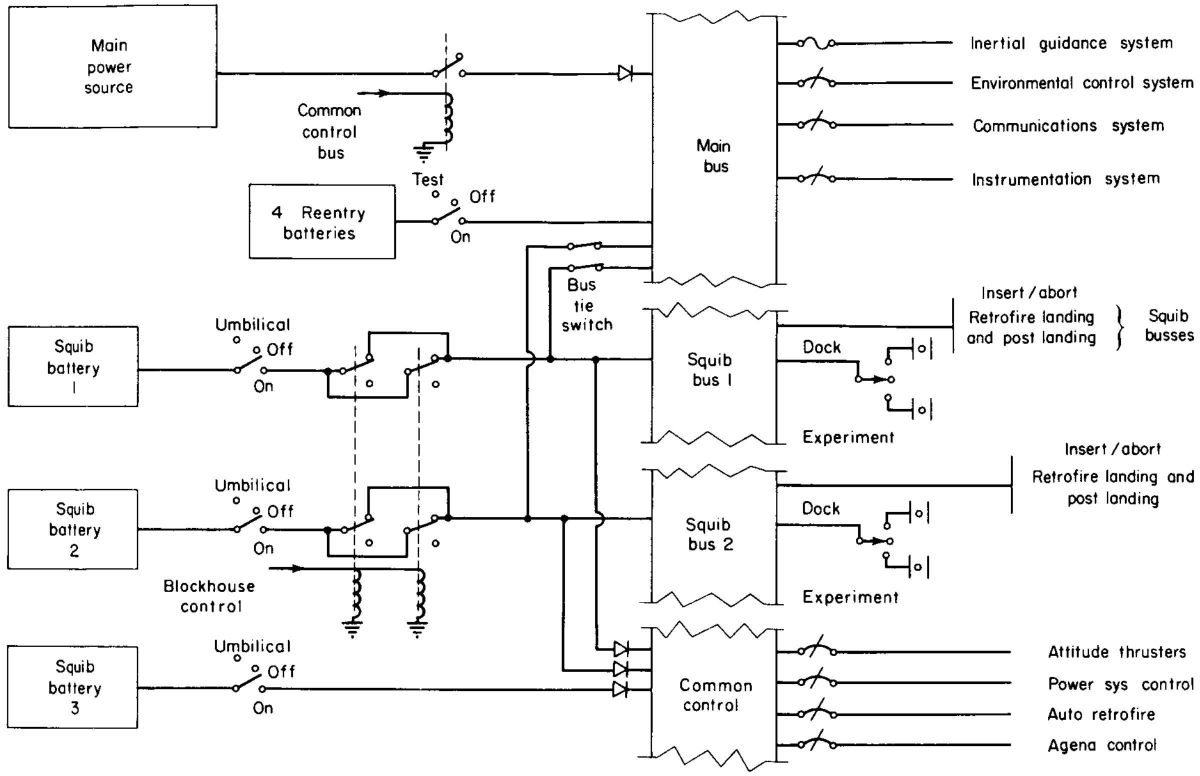


FIGURE 6-1.—Gemini electrical system.

TABLE 6-I.—*Main Power Source for Gemini Spacecraft*

Spacecraft	Power source	Estimated usage, ampere-hours
3	3 silver-zinc batteries [a]	354.3
4	6 silver-zinc batteries	2073.0
5	Fuel-cell power system	4215.8
6	3 silver-zinc batteries	1080.0
7	Fuel-cell power system	5583.6

[a] Each silver-zinc battery had a capacity of 400 ampere-hours.

Table 6-II shows load sharing of the batteries and gives the ampere-hours remaining in each reentry and squib battery after completion of the mission. The highest usage of squib batteries was 59.2 percent on spacecraft 5, whereas the highest usage of reentry batteries was 29 percent on spacecraft 7.

The fuel-cell power system provided Gemini with a long-duration mission capability. For missions requiring more than 800 ampere-hours, the fuel cell has the advantage of low weight and low volume over a silver-zinc battery system.

The fuel-cell power system (fig. 6-2) consists of two sections, plus an associated reactant supply system. Each section is approximately 25 inches long and 12.5 inches in diameter, and weighs approximately 68 pounds including accessories. The section contains 3 stacks of 32 cells and can produce 1 kilowatt at 26.5 to 23.3 volts. The system is flexible in operation. Each stack or section can be removed from the bus at any time. A section can be replaced on the bus after extended periods of open circuit.

Two stacks are required for powered-down flight (17 amperes), and five stacks are needed for maximum loads. To provide electrical power, each cell must interface with the hydrogen and oxygen supply system and with the water system.

The oxygen and hydrogen reactants for the fuel cell are stored in a supercritical cryogenic state in tanks located in the spacecraft adapter section. Each tank contains heaters for main-

TABLE 6-II.—*Reentry and Squib Batteries Postflight Discharge Data* [a]

[All data are in ampere-hours]

Silver-zinc batteries rated capacity	Spacecraft					
	2	3	4	5	6	7
45 (reentry)	35	35. 4	36. 67	41. 0	42. 5	32. 5
45 (reentry)	35	38. 9	41. 67	42. 9	38. 8	32. 5
45 (reentry)	35	38. 9	40. 00	42. 3	36. 7	30. 5
45 (reentry)	35	35. 0	44. 83	40. 65	41. 3	32. 5
15 (squib)	12	10. 27	10	7. 52	12	8. 8
15 (squib)	12	10. 67	11	4. 86	12. 7	9. 4
15 (squib)	12	10. 67	8	6. 0	12. 6	8. 9

[a] Discharge at 5 amperes to 20 volts.

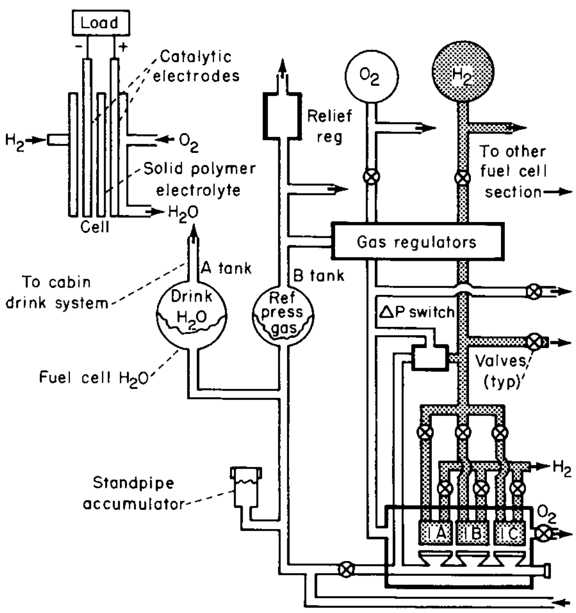


FIGURE 6-2.—Spacecraft 7 fuel-cell/RSS fluid schematic.

taining the oxygen operating pressure between 800 and 910 psia and hydrogen pressure between 210 and 250 psia. Relief valves prevent pressures in excess of 1000 psia for oxygen and 350 psia for hydrogen.

Between the storage tanks and the main control valves, the reactants pass through heat exchangers which increase the temperature of the reactants to near fuel-cell temperatures, thus preventing a thermal shock on the cell. The temperatures in the heat exchangers are controlled by the primary and secondary coolant loops.

The dual pressure regulators supply hydrogen at a nominal 1.7 psi above water pressure and oxygen at 0.5 psi above hydrogen pressure. One regulator is provided for each section, with a crossover network that enables one of the regulators to supply both sections in the event the other regulator should fail. Separate control valves provide gaseous hydrogen to each stack. Each stack is provided with a hydrogen purge valve and an oxygen purge valve for removing accumulated impurity gases. Should it become necessary to shut down a section, a water valve and separate hydrogen and oxygen valves upstream of the regulators are provided.

The smallest active element of the fuel-cell section is the thin, individual fuel cell, which is 8 inches long and 7 inches wide. Each cell consists of an electrolyte-electrode assembly with associated components for gas distribution, electrical current collection, heat removal, and water control. The cell is an ion-exchange type which converts the energy of the chemical reaction of hydrogen and oxygen directly into electricity.

The metallic-catalytic electrode structure of the fuel cell contains an anode and a cathode which are in contact with a thin, solid plastic electrolyte, or ion-exchange membrane, to stimulate the exchange of hydrogen ions between electrodes. In the presence of the metallic catalyst, hydrogen gives up electrons to the electrical load, and releases hydrogen ions which migrate through the electrolyte to the cathode. At the cathode, the ions combine with oxygen and electrons from the load circuit to produce water which is carried off by wicks to a collection point. Ribbed metal current carriers are in contact with both sides of the electrodes to conduct the produced electricity.

The water formed in each cell during the conversion of electricity is absorbed by wicks and transferred to a felt pad located on a porcelain gas-water separator at the bottom of each stack. Removal of the water through the separator is accomplished by the differential pressure between oxygen and water across the separator. If this differential pressure becomes too high or too low, a warning light on the cabin instrument panel provides an indication to the flight crew. The telemetry system also transmits this information to the ground stations. A similar warning system is provided for the oxygen-to-hydrogen gas differential pressure so that the appropriate action may be taken if out-of-specification conditions occur.

The water produced by the fuel-cell system exerts pressure on the Teflon bladders in water tanks A and B. Water tank A also contains drinking water for the flight crew, and the drinking-water pressure results from the differential between the fuel-cell product-water pressure and cabin pressure. Tank B has been precharged with a gas to 19 psia, and the fuel-cell product water interfaces with this gas. However, the 19-psia pressure changes with drinking-water consumption, fuel-cell water production, and temperature. Should the pressure exceed 20 psia, the overpressurization is relieved by two regulators. This gas pressure provides a reference pressure to the two dual regulators that control the flow of the oxygen and hydrogen gases to the fuel-cell sections.

Another system which interfaces with the fuel cell is the coolant system. The spacecraft has two coolant loops: the primary loop goes through one fuel-cell section, and the secondary loop goes through the second section. In each section the coolant is split into two parallel paths. For the coolant system, the stacks are in series, and the cells are in parallel. The coolant-flow inlet temperature is regulated to a nominal 75° F.

Ground Test Program

To achieve the necessary confidence required before a completely new system is certified for flight, considerable ground testing of the fuel-cell power system was necessary (table 6–III). As part of the development program, two fuel-cell sections were operated at electrical load profiles simulating prelaunch and rendezvous, followed by powered-down flight. The first section lasted 1100 hours, and the second section lasted 822 hours. A third section endured 10 repeated rendezvous missions. In qualification, one section was subjected to random vibration, and a month later it was placed in an altitude chamber at −40° F for 4 hours. Still another section successfully experienced acceleration, and a month later it was placed in an altitude chamber with chamber-wall temperatures cycling each 24 hours from 40° to 160° F. This section was supplying power to a simulated 14-day-mission electrical load.

TABLE 6–III.—*Major Tests of Fuel-Cell Power System*

Section no.	Environments	Electrical load profile	Remarks
1516	Ambient	Prelaunch simulation rendezvous powered-down	1100 hours' duration
1519	Ambient	Prelaunch simulation rendezvous powered-down	822 hours' duration
1524	Ambient	Repeated 2-day rendezvous	10 cycles
1514	Vibration (random) (7.0g RMS for 8 minutes per axis)	30 amperes	Satisfactory
	Altitude (1.47×10^{-5} psia)	7.5 amperes, 4 hours	Satisfactory
1527	Acceleration linearly from 1 to 7.25g in 326 seconds	45 amperes	Satisfactory
	Altitude (1.6410^{-6} psia); temperature cycled 40° to 160° F every 90 minutes	14-day mission profile	Monitored with cockpit instrumentation; successfully completed mission

An extensive development, qualification, and reliability test program was conducted on the reactant supply system. A total of 14 different environmental conditions, in addition to 7 simulated 14-day missions, was included in the tests. The environments included humidity, thermal shock, cycle fatigue, high and low temperature and pressure, proof, burst, and also all expected dynamic environments. Subsequent ground testing revealed that the thermal performance of the hydrogen container degrades with time at cryogenic temperatures. It was found that the bosses in the inner shell allowed hydrogen to leak into the annulus, thus degrading the annulus vacuum, even though this leak rate was almost infinitesimal. A pinch-off tube cutter was added to allow venting the annulus overboard should the container degrade excessively during a mission. Also, as added protection for the Gemini VII spacecraft, a regenerative line and insulation were added to the outside of the hydrogen container to limit the heat leak into the container.

The evaluation of the complete fuel-cell power system was successfully completed with a series of tests that checked out the integrated system. Additional tests included a full-system, temperature-altitude test, and finally a vibration test of the entire system module mounted in a spacecraft equipment adapter.

Fuel-Cell Flight Results

Gemini V

The fuel-cell power system was first used in the Gemini V mission. During the launch phase, the fuel cells supplied approximately 86 percent of the overall main-bus load. During the orbit phase, the fuel cells provided 100 percent of the main-bus power. The maximum load supplied by the fuel cells was 47.2 amperes at 25.5 volts.

Section performance.—The performance of the fuel-cell section 1 is shown in figure 6–3. Between the first launch attempt and the actual launch, the fuel-cell power system was operated on a 1-ampere-per-stack dummy load for 60 hours. At a load of 15 amperes, approximately a 0.4-volt decline was observed between the second activation of the section on August 18, 1965, and the performance on August 21, 1965, the first day of flight. Continuing operation showed a gradual increase in performance until the eighth day of flight, when the performance was approximately equal to that experienced at the second activation. The performance of fuel-cell section 2 is shown in figure 6–4. At a load of 15 amperes, section 2 showed a decline of approximately 0.6 volt between the second activation on August 18, 1965, and the performance on August 21, 1965, the first day of flight. Over the 8 days of the mission, the section performance declined an additional 0.66 volt, most of which occurred during the three periods of open circuit. During the flight, section 2 was placed on open circuit, without coolant flow, for three 19-hour periods. Open-circuit operation was desirable to conserve the ampere-hours drawn by the coolant pump. The voltage degradation, compared at 8 amperes for each of

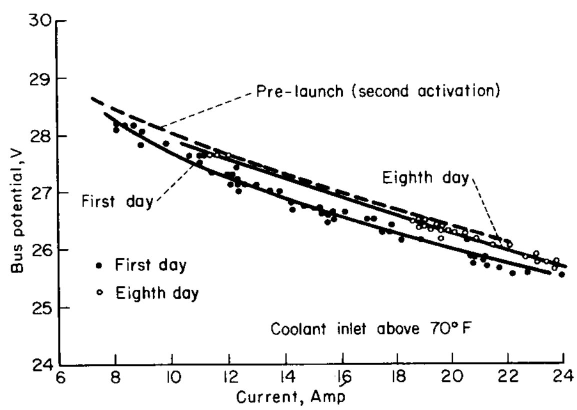


FIGURE 6–3.—Fuel-cell section 1 performance for the Gemini V mission.

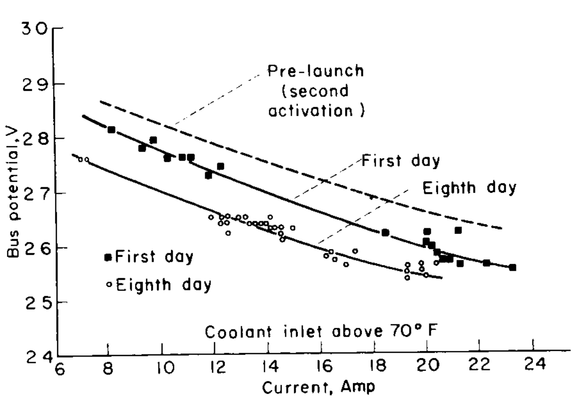


FIGURE 6–4.—Fuel-cell section 2 performance for the Gemini V mission.

these three periods, was 0.27 volt. A comparison of the performance following each open-circuit period shows a net rise of 0.15 volt in section 2 performance.

The purge sensitivity exhibited during the mission was found to be normal. An average recovery of 0.1 volt resulted from the oxygen and hydrogen purge sequences.

Three differential-pressure warning-light indications occurred: during launch, during the first hydrogen purge of section 1, and during an attempt to purge section 1 without opening the crossover valve. These pressure excursions caused no apparent damage to the fuel-cell power system.

Load sharing of the six fuel-cell stacks is shown in table 6–IV. While the inflight performance of section 2 declined, the performance of section 1 improved and resulted in a shift of 7.7 percent in load sharing between the two sections.

Reactant-usage rate and water-production rate.—Since the Gemini V mission was the first mission to use the fuel-cell power system, it was important to future mission planning that the reactant-usage rates be determined and compared with theoretical and ground-test experience (table 6–V). The reactant-usage rate and water-production rate agreed within 2 and 4 percent, respectively, with the theoretical, and within 5 percent in each case with ground-test observations.

TABLE 6–IV.—*Fuel-Cell Load Sharing*

[Bus potential, 25.8 volts]

Fuel-cell stack	1st day of mission		Change in percent of total load between 1st and 8th days	8th day of mission	
	Current, amperes	Percent of total load		Current, amperes	Percent of total load
Stack 1A	7.02	16.70	+3.69	8.25	20.39
Stack 1B	6.45	15.35	+1.82	6.95	17.17
Stack 1C	7.65	18.20	+2.15	8.23	20.35
Section 1	21.12	50.2	+7.7	23.43	57.9
Stack 2A	6.65	15.82	−2.45	5.42	13.37
Stack 2B	6.63	15.77	−1.92	5.62	13.85
Stack 2C	7.65	18.21	−3.34	6.02	14.87
Section 2	20.93	49.8	−7.7	17.06	42.1
Total	42.05	100	------	40.49	100

TABLE 6–V.—*Fuel-Cell Cryogenic Usage Rates and Water-Production Rate*

	Hydrogen usage, lb/amp-hr	Oxygen usage, lb/amp-hr	Water production, lb/amp-hr	
			Method 1	Method 2
Theoretical	0.0027	0.0212	0.0238	
Ground test	.0029	.0252	0.0253	
Flight data [a]	.00275	.0220	0.0247	0.0244

[a] These are averages of 4 calculated rates taken at 15, 24, 30, and 34.5 hours after lift-off.

The cryogenic-oxygen heater circuit failed after about 26 minutes of flight. Therefore, the oxygen-usage rate was calculated from hydrogen data, applying the ratio of 8 to 1 for the chemical combinations of oxygen and hydrogen. The water-generation rate of the fuel cell was determined by two different methods. In method 1, hydrogen and oxygen usage rates were combined, assuming that all of the gases produced water. In method 2, the amount of drinking water consumed by the flight crew was added to the amount required to change the gas pressure in the water storage tank over a given interval of time, and the ratio of this water quantity to the associated ampere-hours resulted in the production rate.

Prior to the Gemini V launch, the hydrogen tank in the reactant supply system was filled with 23.1 pounds of hydrogen to satisfy the predicted venting and the power requirements of the planned mission. Prelaunch testing of the hydrogen tank showed that it had an ambient heat leak greater than 9.65 Btu per hour, and this provided data for an accurate prediction of inflight performance. The tank pressure increased to the vent level of 350 psia at 43 hours after lift-off. Venting continued until 167 hours after lift-off, with a brief period of venting at approximately 177 hours. At the end of the mission, 1.51 pounds of hydrogen remained. The oxygen container in the reactant supply system was serviced with 178.2 pounds of oxygen and pressurized to 815 psia. Operation was normal until 25 minutes 51 seconds after lift-off when the heater circuitry failed. The pressure then declined gradually until stabilization occurred at approximately 70 psia, around 4 hours 22 minutes after lift-off. Although 70 psia was far below the 200 psia specified minimum supply pressure, the gas regulators worked perfectly. Analysis indicates that the fluid state at the 70-psia point was coincident with the saturated liquid line on the primary enthalpy curves for oxygen. Subsequent extraction from the tank resulted in penetration of the two-phase, or liquid and vapor, region for operation during the remainder of the flight. Analysis showed that the majority of fluid extracted from the container was low-energy liquid instead of high-energy vapor. This was a result of the characteristics of a fluid in a zero-gravity environment and the internal arrangement of the container. A more detailed postflight analysis indicated that, at all times during the mission, the extracted fluid, by weight, was more than 60 percent low-energy liquid. The energy balance between extraction and ambient heat leak permitted a gradual pressure increase to 260 psia at the end of the mission. The mission was completed with an estimated 73 pounds of the oxygen remaining in the tank. Postlandings tests of all associated circuits and components in the reentry portion of the spacecraft did not uncover the problem. To prevent a similar occurrence on spacecraft 7, a crossfeed valve was installed between the environmental-control-system primary-oxygen tank and the fuel-cell reactant-supply-system oxygen tank.

Gemini VII

The 14-day Gemini VII flight was the second mission to use a fuel-cell power system. This mission would not have been possible without the approximately 1000-pound weight saving provided by the fuel cell. In addition to the man-bus loads, during orbital flight, fuel-cell power was switched to the squib buses, and the squib batteries were shut down. During this mission the maximum load supplied by the fuel-cell power system was 45.2 amperes at 23.4 volts.

Section performance.—Figure 6–5 shows the performance of the fuel-cell section 1 during its second activitation and on the first and last days of the Gemini VII mission. During these periods the voltage decay averaged 3 and 5

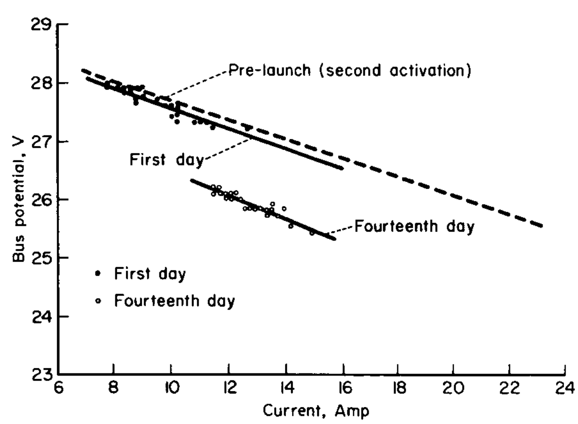


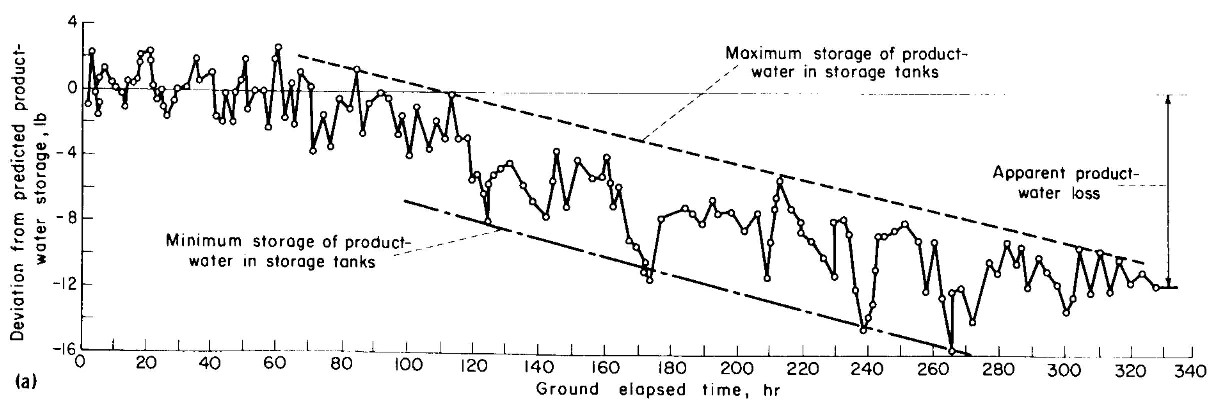
FIGURE 6–5.—Fuel-cell section 1 performance for the Gemini VII mission.

millivolts per hour at 10 and 24 amperes, respectively. These decay rates are within the range experienced in the laboratory section life tests. Through the first 127 hours of the mission, the performance decay rate of the fuel-cell section 2 was also within the range experienced in the laboratory section life tests. At that time, the first of several rapid performance declines was observed, with each decline showing severe drops in stack 2C performance. At 259 hours after lift-off, the last rapid performance decline in section 2 began and resulted in the removal of stacks 2A and 2C from the spacecraft electrical-power bus.

During all but 16 hours of the mission, the oxygen-to-water differential pressure warning light of section 2 indicated an out-of-limit oxygen-to-water pressure across the water separators. With an out-of-tolerance differential pressure, the extraction rate of water from the section would have been severely reduced. Therefore, when the performance of stack 2C, which was carrying 45 to 50 percent of the section load, started dropping, it was concluded that water was accumulating in section 2. Excessive water reduces the active membrane area in each cell by masking; consequently, section 2 was purged more often in order to move water out through the ports. In addition, this section was placed on open circuit to stop the production of water while permitting water removal to continue.

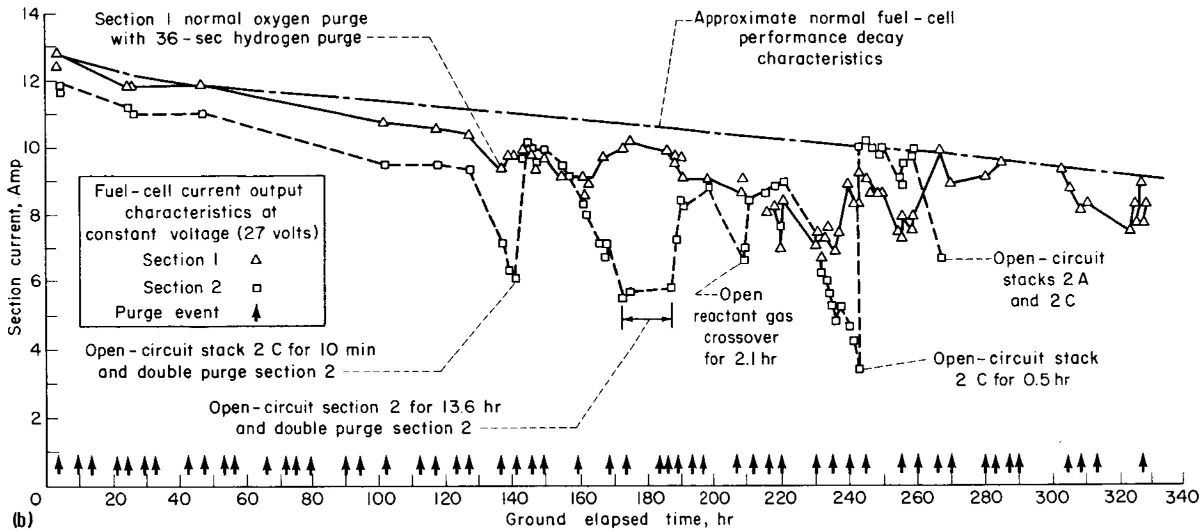
Figures 6–6(a) and 6–6(b) show the deviations in product-water storage with the performance of the fuel-cell sections as a function of time from lift-off. Between 100 and 265 hours after lift-off, a maximum storage fluctuation of 8 pounds occurred around the gradual storage reduction. The gradual storage reduction, totaling 12 pounds at the end of the mission, is attributed to losses of water during purges of oxygen and hydrogen or to a possible loss of nitrogen in the water-reference system. A significant observation is that, when periods of maximum product-water storage occurred, the section current characteristics at a constant voltage show good fuel-cell performance. When periods of minimum or decreasing product-water storage occurred, section 2 and, to a lesser extent, section 1, had very low or degrading performance. The responses to the corrective actions were significant increases in stored water (presumably from sec. 2) and immediate return to normal performance.

Photographs of the Gemini VII spacecraft, taken by the Gemini VI–A flight crew during the rendezvous exercise, revealed an ice formation around the hydrogen-vent port on the equipment adapter (fig. 6–7). The presence of this ice formation raised questions about the ability to purge hydrogen from the fuel-cell sections. Purge effects were not discernible from the data. The Gemini VII flight crew did report water crystals going by the spacecraft window during hydrogen purges late in the mission. At these particular times, the vent port was at least partially open. The hydrogen-to-oxygen differential-pressure light, normally illuminated during hydrogen purging, did not illuminate during this flight or the Gemini V mission. Freezing of the purge moisture at the vent port could cause restriction



(*a*) Fuel-cell product-water storage.

FIGURE 6–6.—Comparison of fuel-cell performance with fuel-cell product-water storage.



(b) Fuel-cell current supplied.
FIGURE 6-6.—Concluded.

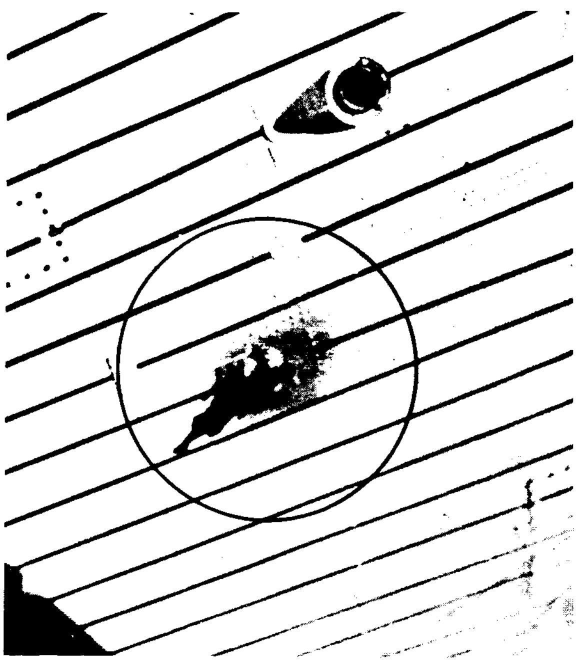


FIGURE 6-7.—Ice formation at hydrogen vent.

of flow and prevent illumination of the differential-pressure light.

Reactant usage rate.—The hydrogen container of the reactant supply system was serviced to 23.58 pounds and pressurized to 188 psia. Container performance was normal throughout the flight. At the end of the flight, 8.55 pounds of hydrogen remained.

The oxygen container of the reactant supply system was serviced to 181.8 pounds and pressurized to 230 psia. Container performance was normal throughout the flight. The oxygen quantity remaining at the end of the flight was 60.95 pounds.

Sequential System

The sequential system consists of indicators, relays, sensors, and timing devices which provide electrical control of the spacecraft. The sequential system performs launch-vehicle–spacecraft separation, fairing jettison, equipment-adapter separation, retrofire, retroadapter jettison, drogue-parachute deploy, main-parachute deploy, landing attitude, and main-parachute jettison. Generally, the flight crew receive their cue of the sequential events from the electronic timer which lights a sequential telelight switch. When the switch is depressed and released, the sequence is initiated.

The major sequential functions are operated through a minimum of two completely independent circuits, components, and power sources. As an example, figure 6-8 shows the redundancy in the launch-vehicle–spacecraft separation system; the flight crew depress and release the SEP SPCFT telelight switch. This action supplies power to the redundant launch-vehicle–spacecraft wire guillotines, to the pyrotechnic switch that open-circuits the interface

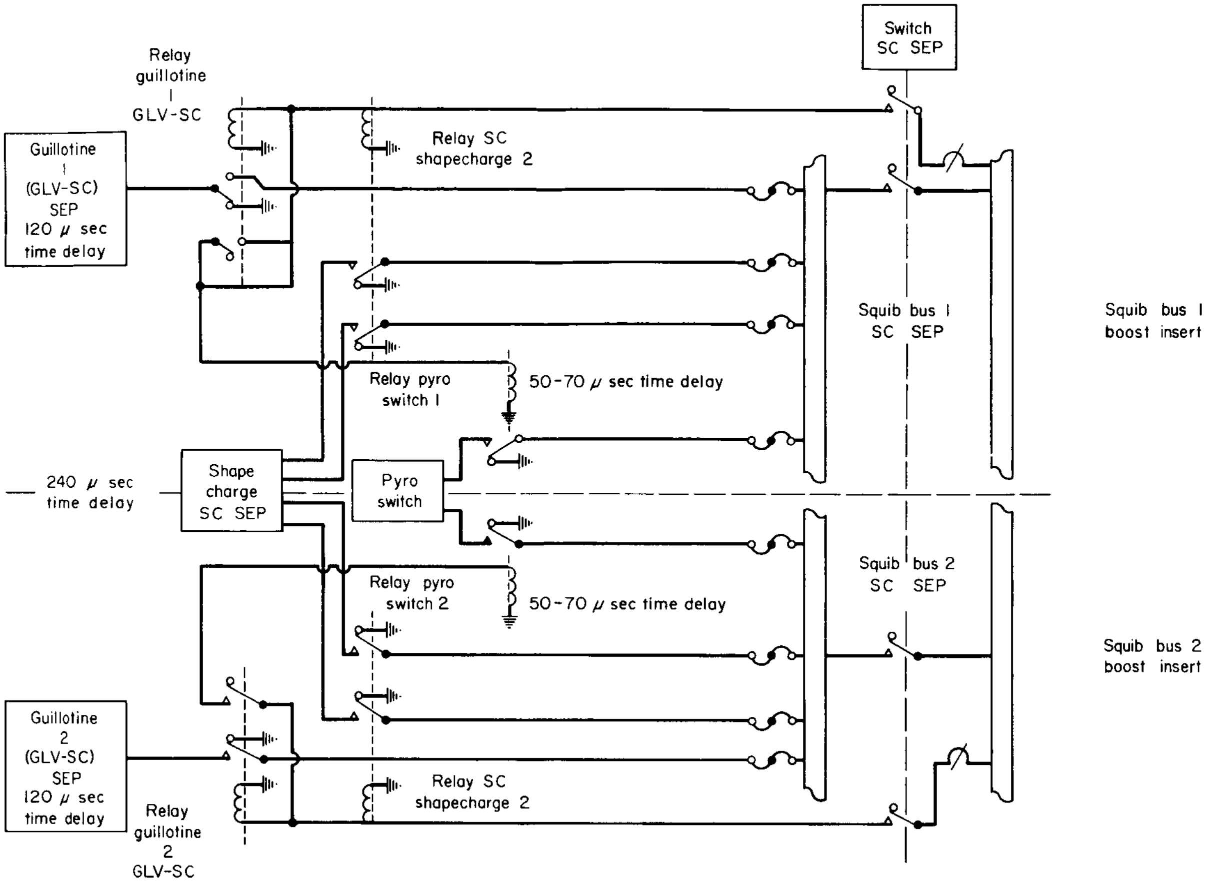


FIGURE 6-8.—Launch-vehicle–spacecraft separation circuitry.

wire bundles prior to severing, and to the shaped charges that break the structural bond between the launch vehicle and the spacecraft.

The sequential system is checked out frequently before the spacecraft leaves the launch pad. Each sequential function is performed first with one circuit, then with the backup, and finally with both. The timeout of all time delays is checked and rechecked. High-energy and low-energy squib simulators were fired to insure that the firing circuits were capable of handling the sure-fire current of the pyrotechnic initiators. Thus far in the program, all sequential timeouts have been nominal.

Concluding Remarks

It can be concluded from Gemini flight experience that fuel cells and their associated cryogenic reactant supply systems are suitable and practical for manned space flight applications. It can also be concluded that the man-in-the-loop concept of manually performing non-time-critical sequential functions is a reliable mode of operation.

7. CREW STATION AND EXTRAVEHICULAR EQUIPMENT

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Summary

The crew station provides a habitable location for the flight crew and an integrated system of displays and controls for inflight management of the spacecraft and its systems. The results of the first manned Gemini flights have shown that the basic crew-station design, the displays and controls, and the necessary crew equipment are satisfactory for rendezvous and long-duration missions. Space suits have been developed for both intravehicular and extravehicular use. These space suits have been satisfactory for flight use; however, the flight crews favor operation with suits removed for long-duration intravehicular missions. The initial extravehicular equipment and space suits were satisfactory in the first extravehicular operation. This operation proved the feasibility of simple extravehicular activities, including self-propelled maneuvering in the immediate vicinity of the spacecraft. Increased propellant duration is desirable for future evaluations of extravehicular maneuvering units. The Gemini crew station and equipment are satisfactory for continued flight use.

Introduction

The experience gained in Project Mercury proved and demonstrated the capability of the flight crew to participate effectively in the operation of the spacecraft systems. This experience was carried over into the design of the Gemini spacecraft. Manual control by the flight crew is a characteristic design feature of every system in the spacecraft. Automatic control is used only for those functions requiring instantaneous response or monotonous repetition. Ground control of the spacecraft is used only for updating onboard data and for on-off control of ground tracking aids and telemetry transmitters. Manual backup is provided for all automatic and ground-control functions. The flight crew has the key role in the control of all spacecraft systems.

To enable the flight crew to perform the necessary functions, the crew station provides an integrated system of displays and controls. The displays provide sufficient information to determine the overall status of the spacecraft and its systems at any time. The controls enable the crew to carry out normal functions and corrective actions. In addition, the crew station provides a habitable location for the crew, with a large amount of equipment to support the crew's needs and activities.

Basic Design

Cabin Arrangement

The flight crew is housed within the pressurized structural envelope shown in figure 7–1. The total internal pressurized volume is 80 cubic feet. The net volume available for crew mobility after equipment and seat installation is approximately 20 cubic feet per man. This volume was adequate for the Gemini missions up to 14 days; however, it was less than optimum for crew comfort and mobility. The interior arrangement is shown in figure 7–2. The crewmembers are seated side by side, in typical pilot and copilot fashion, facing the small end of the reentry assembly. This seating arrangement provides forward visibility for both pilots and permits either one to control the spacecraft during orbit and reentry with minimum duplication of displays and controls.

Cabin Lighting

The basic lighting provisions in the crew compartment consist of three incandescent floodlight assemblies. Continuously variable dimming controls and alternate selection of red or white light are provided. The cabin lighting has been adequate for the missions to date; however, during darkside operation, the crews have found it difficult to see the instruments without reducing their dark adaptation for external visibility. Floodlighting is not well suited to this requirement.

Stowage Provisions

The equipment stowage provisions consist of fixed metal containers on the side and rear walls of the cabin, and a large stowage frame in the center of the cabin between the ejection seats, as shown in figures 7-3 and 7-4. Food packages and other equipment are stowed in the side and aft containers. All items in the aft containers are normally stowed in pouches, with all the pouches in a container tied together on a lanyard.

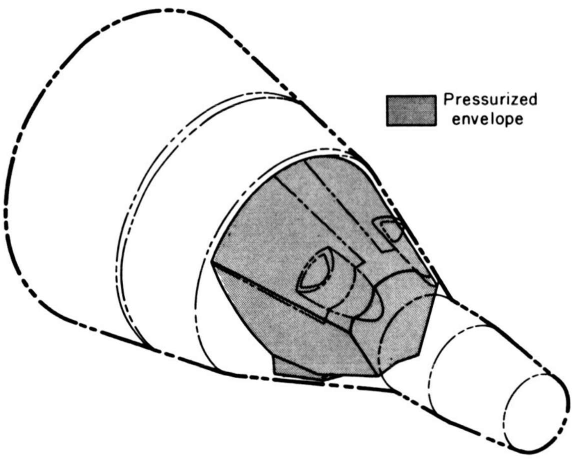

FIGURE 7-1.—Crew-station pressure vessel.

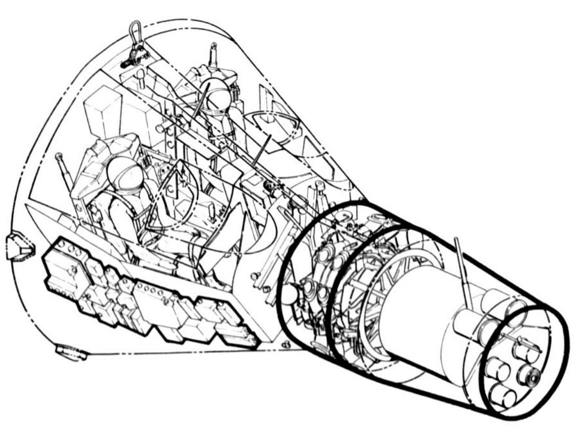

FIGURE 7-2.—Crew-station interior arrangement.

The center stowage frame holds fiber-glass boxes containing fragile equipment. These boxes are standardized, and the interiors are filled with a plastic foam material molded to fit the contours of the stowed items. This foam provides mechanical and thermal protection. Figure 7-5 shows a typical center stowage box with equipment installed. The concept of using standardized containers with different interiors has made it possible to use the same basic stowage arrangements for widely varying mission requirements.

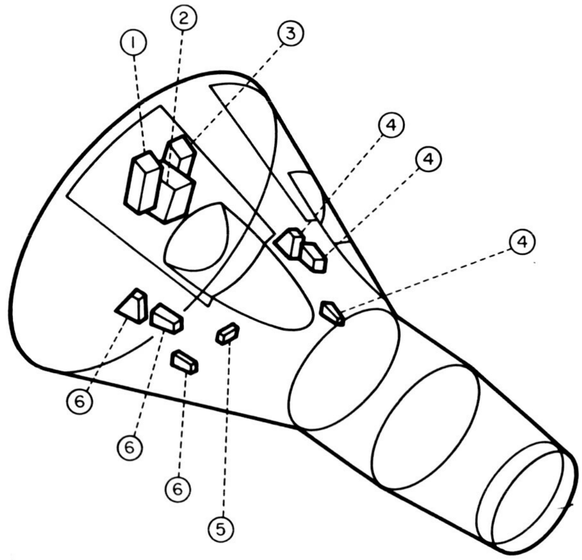

FIGURE 7-3.—Crew-station stowage arrangement: (*1*) right aft stowage container; (*2*) center stowage container; (*3*) left aft stowage container; (*4*) left-side stowage containers; (*5*) orbital utility pouch (under right instrument panel); (*6*) right-side stowage containers.

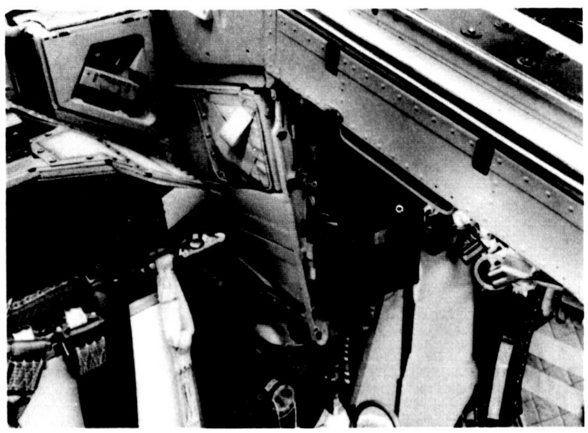

FIGURE 7-4.—Spacecraft center and right-aft stowage containers (viewed from right side looking aft).

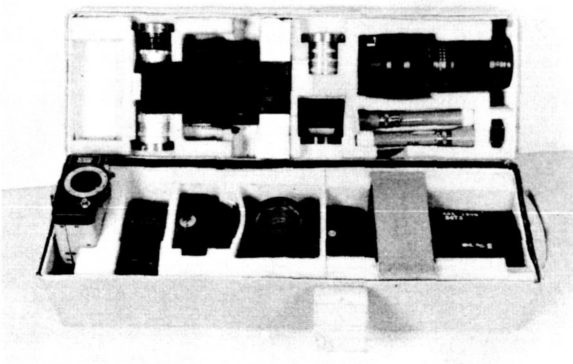

FIGURE 7-5.—Stowage of equipment in center stowage box.

In order to establish practical stowage plans for each mission, formal stowage reviews and informal practice-stowage exercises were conducted with each spacecraft and crew. The tasks of unstowing equipment in orbit and restowing for reentry were practiced in the same sequence as planned for flight. The use of authentic mockups for stowage exercises and actual flight hardware for spacecraft fit checks was essential for successful prelaunch stowage preparations.

The equipment stowage provisions proved satisfactory for long-duration and rendezvous missions. The mission results showed that with adequate stowage preparations and practice, the stowage activities in orbit were accomplished without difficulty.

Displays and Controls

General

The command pilot in the left seat has the overall control of the spacecraft. The pilot in the right seat monitors the spacecraft systems and assists the command pilot in control functions. This philosophy led to the following grouping of displays and controls (fig. 7-6):

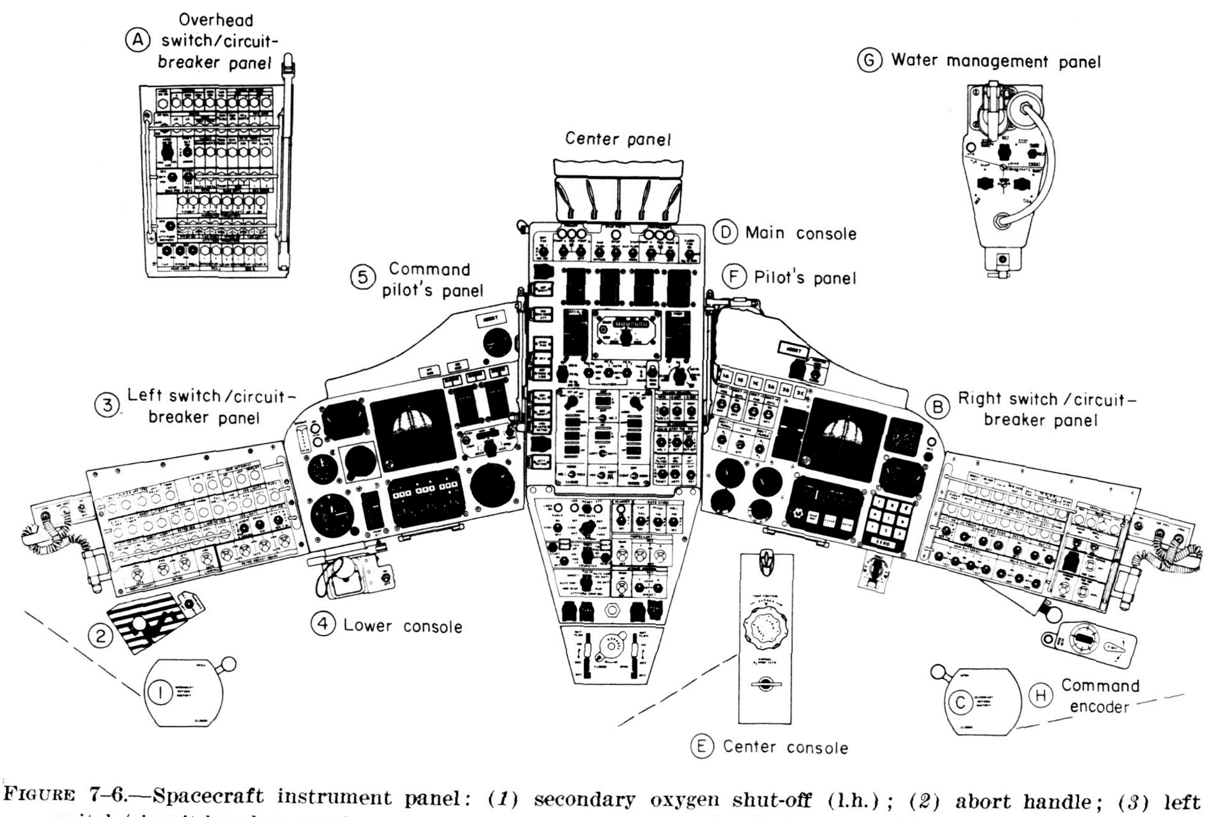

FIGURE 7-6.—Spacecraft instrument panel: (*1*) secondary oxygen shut-off (l.h.); (*2*) abort handle; (*3*) left switch/circuit-breaker panel; (*4*) lower console; (*5*) command pilot's panel; (*A*) overhead switch/circuit-breaker panel; (*B*) right switch/circuit-breaker panel; (*C*) secondary oxygen shut-off (r.h.); (*D*) main console; (*E*) center console; (*F*) pilot's panel; (*G*) water management panel; (*H*) command encoder.

The left instrument panel (fig. 7–7) contains the flight command and situation displays and the launch-vehicle monitoring group. The maneuver control handle is located under the left instrument panel. The left switch panel contains the sequential bus and retrorocket arming switches, as well as circuit breakers for electrical-sequential functions and communications functions. The abort control handle is just below the left switch panel. These displays and controls are normally operated only by the command pilot.

The right instrument panel (fig. 7–8) contains displays and controls for the navigation system, the electrical power system, and experiments. A flight director and attitude indicator is also installed in the right instrument panel. The right switch panel contains switches and circuit breakers for the electrical power system and experiments. Below the right switch panel is the right-hand maneuver control handle. These displays and controls are operated by the pilot.

The center instrument panel (fig. 7–9) contains the communications controls, the environmental displays and controls, and the electrical-sequential system controls. The pedestal panel contains the guidance and navigation system controls, the attitude and maneuvering system controls, the landing and recovery system controls, and the space-suit ventilation flow controls. The attitude control handle and the

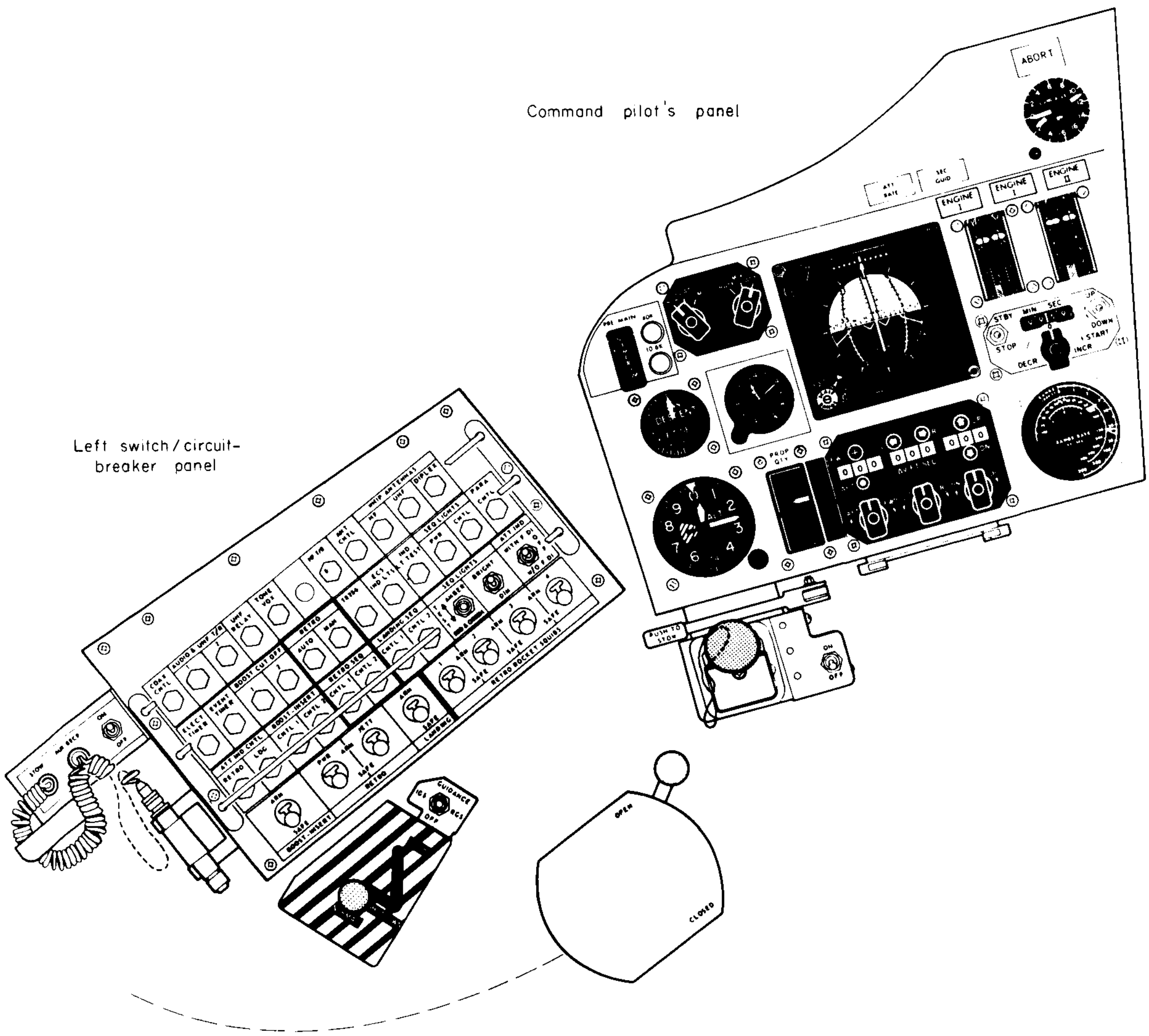

Figure 7–7.—Command pilot's displays and controls.

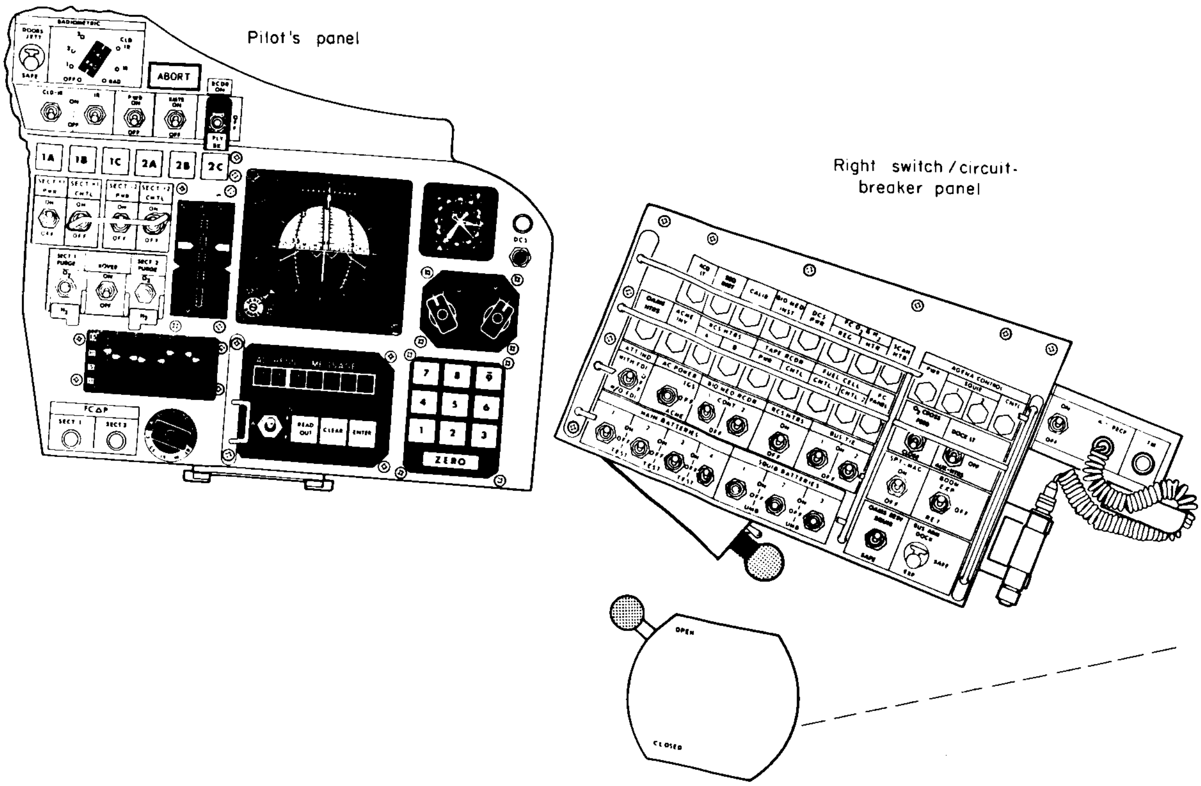

FIGURE 7-8.—Pilot's displays and controls.

cabin and suit temperature controls are located on the center console. The water management controls are located on a panel between the ejection seats. The overhead switch panel contains switches and circuit breakers for the attitude control and maneuvering systems, the environmental control system, and the cabin lighting. These controls and displays are accessible to both pilots and may be operated by either one.

Displays

The primary flight displays consist of the flight director and attitude indicator, the incremental velocity indicator, and the radar indicator. The flight director and attitude indicator is composed of an all-attitude sphere and flight director needles for roll, pitch, and yaw. The incremental velocity indicator provides the command pilot with either the command-maneuver velocities from the guidance computer or the velocities resulting from translation maneuvers. The radar indicator displays the rendezvous-target range and range rate when the radar is locked on.

The launch-vehicle monitoring group, or the malfunction-detection-system display, consists of launch-vehicle tank-pressure gages, thrust-chamber pressure lights, an attitude overrate light, and a secondary guidance light.

The primary-navigation-system display and control unit is the manual data insertion unit located on the right instrument panel. Guidance computer values may be inserted or read out with the manual data insertion unit.

The environmental and propulsion system displays and the electrical-power-system monitor display all utilize vertical scales on which deviations from nominal are readily detected. In the electrical power system, the current values for all six stacks of the fuel cell are displayed simultaneously. The concept of a single ammeter with a stack-selector switch did not prove satisfactory, since frequent monitoring of the stack currents is required. For relatively static parameters such as cryogenic tank pressures and quantities and propellant temperatures, the use of one display and a selector switch for several parameters was adequate.

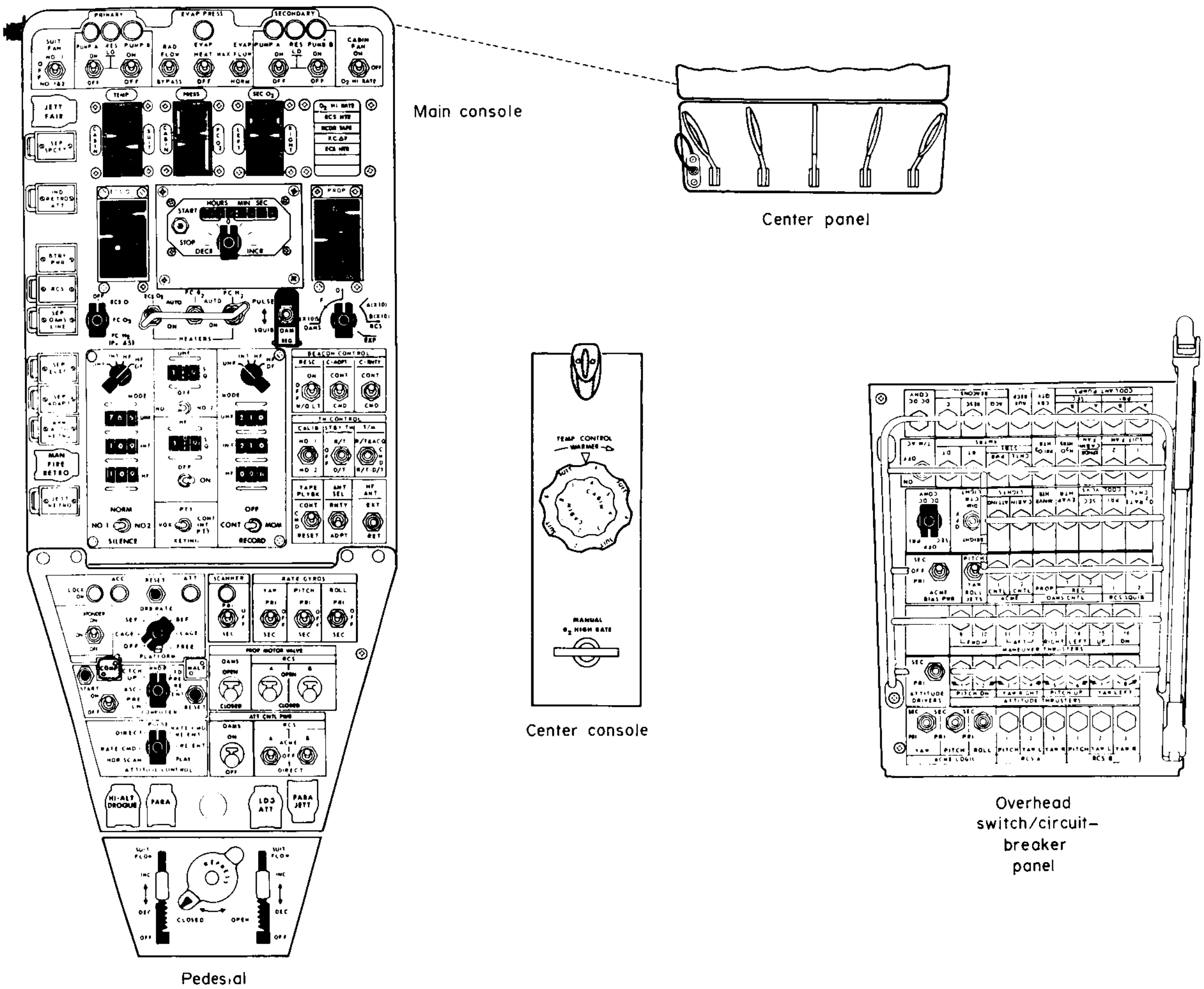

FIGURE 7-9.—Displays and controls used by both sides.

Controls

The three-axis attitude control handle, shown in figure 7-10, enables the flight crew to control the spacecraft attitude in pitch, roll, and yaw. This single control handle is located between the two pilots and can be used by either one. The three axes of motion correspond to the spacecraft axes. The axes of the control handle are located to minimize undesirable control inputs caused by high accelerations in launch and reentry, and to minimize cross-coupling or interaction of individual commands.

The primary translation-maneuver control handle (fig. 7-11) is located beneath the left instrument panel. The motion of this control corresponds to the direction of spacecraft motion.

Special system controls, such as the environmental-control-system levers and valve handles, are oriented and sized for use by the crew in pressurized space suits. Actuation forces are within crew requirements but are sufficient to prevent inadvertent actuation or change of position due to launch and reentry forces. All critical switches are guarded by locks or bar guards.

Flight Results

The best indications of the adequacy of the displays and controls have been the results of the flights to date and the ability of the crew to accomplish assigned or alternate functions as required. In general, the displays and controls have been entirely satisfactory.

During the first launch attempt for the Gemini VI-A mission, the flight crew was able to assess correctly the launch-vehicle hold-kill

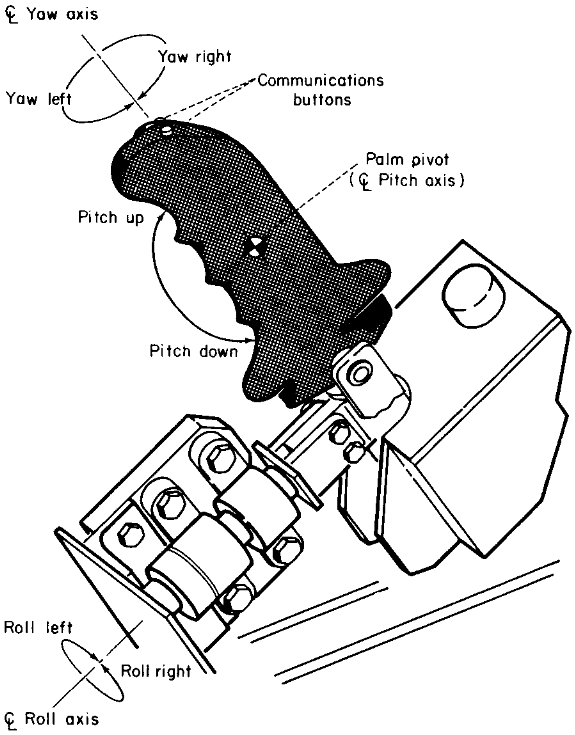

FIGURE 7-10.—Attitude hand control.

situation, initiate the proper action, and avoid an unnecessary off-the-pad ejection. As a result, there was only a minor delay in the launch schedule, rather than the loss of an entire mission.

Flight results have shown that the crews were able to determine the spacecraft attitude and rates and to control the spacecraft more accurately than initially anticipated. Accordingly, the markings on the attitude indicator and flight director needles have been increased to provide greater precision in reading pitch and roll attitudes and pitch and yaw rates.

The only other significant change to the displays and controls was the addition of a mission-elapsed-time clock to spacecraft 6 and subsequent spacecraft. Prior to the use of this clock, there had been occasional confusion between Greenwich mean time and mission elapsed time for timing the onboard functions. The installation of a mission-elapsed-time clock in the spacecraft enabled the crew and the ground control network to use a single, common time base for all onboard functions. The addition of this mission-elapsed-time clock was found to

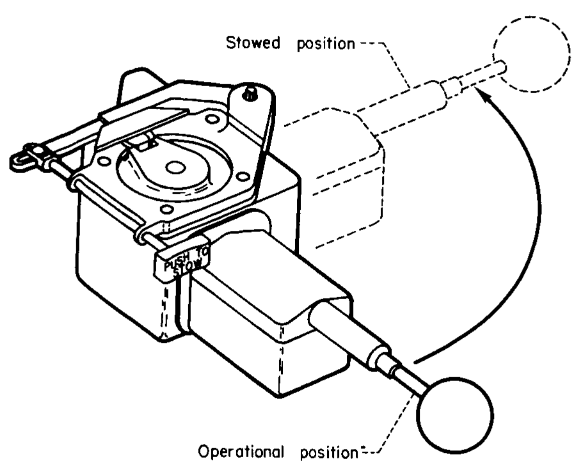

FIGURE 7-11.—Maneuver hand control.

be a significant simplification for all mission-timing activities.

An overlay concept is used to make maximum use of the available display panel space. Since the launch-vehicle display group is not used after reaching orbit, checklists and flight procedure cards are mounted in this area for ready reference during orbital operations.

The use of pressure-sealed switches in the attitude and maneuver controls, as well as other applications in the crew station, led to some difficulty because of the sensitivity of these switches to pressure changes. In one altitude chamber test, several of these sealed switches failed to close because of the pressure trapped inside. Fabrication and test procedures were established to screen out those pressure-sensitive switches. The pushbutton-lighted switches also gave some difficulty in the development phase because of the critical dimensional requirements of small components and frequent mechanical failure. Sturdy toggle switches were used inside all critical, pushbutton-lighted switches to obtain the desired reliability of operation. No difficulties with the sturdier switches were encountered in flight.

As a result of the experience of the early Gemini flights, the crew-station displays and controls are now standardized for the remaining spacecraft. The only future changes planned are those resulting from the differences in experiments assigned to each mission.

Space Suits and Accessories

G3C Space Suit

The G3C space suit used in the first manned Gemini flight is shown in figure 7–12. The outer layer is a high-temperature-resistant nylon material. The next layer is a link-net material, especially designed to provide pressurized mobility and to control ballooning of the suit. The pressure layer is a neoprene-coated nylon. An inner layer of nylon is included to minimize pressure points from various space-suit components. The space-suit vent system (fig. 7–13) provides ventilating flow to the entire body. Sixty percent of the ventilation flow is ducted by a manifold system to the boots and gloves. This gas flows back over the legs, arms, and torso to remove metabolic heat and to maintain thermal comfort. The remaining 40 percent of the inlet gas passes through an integral duct in the helmet neck ring and is directed across the

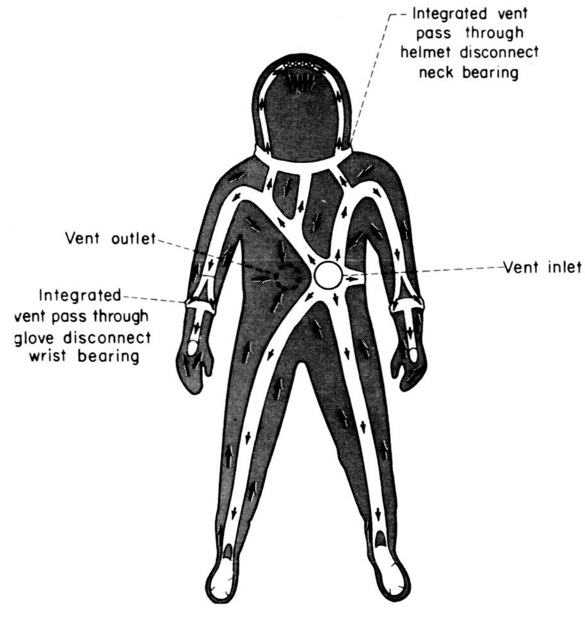

FIGURE 7–13.—Ventilation distribution system for the G3C space suit.

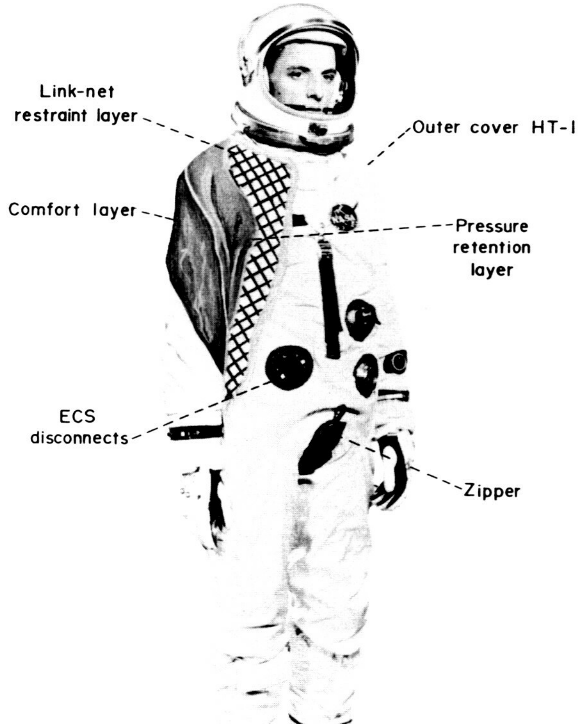

FIGURE 7–12.—Gemini G3C space suit.

visor to prevent fogging and to provide fresh oxygen to the oral-nasal areas. Flight experience with the G3C space suit indicated that it met all the applicable design requirements for short-duration missions. There were no space-suit component failures nor any significant problems encountered in flight.

G4C Space Suit

The G4C space suit, as shown in figure 7–14, is a follow-on version of the G3C suit, with the necessary modifications required to support extravehicular operation. The outer-cover layer of the G4C suit incorporates added layers of material for meteoroid and thermal protection. The inner layers of the space suit are the same as the basic G3C suit. The G4C helmet incorporates a removable extravehicular visor which provides visual protection and protects the inner visor from impact damage. A redundant zipper was added to the pressure-sealing closure of the suit to protect against catastrophic failure and to reduce the stress on the pressure-sealing closure during normal operation.

The G4C suits worn by the flight crews of the Gemini IV, V, and VI-A missions were satisfactory for both intravehicular and extravehicular operation. Some crew discomfort resulted from long-term wear of the suits, and

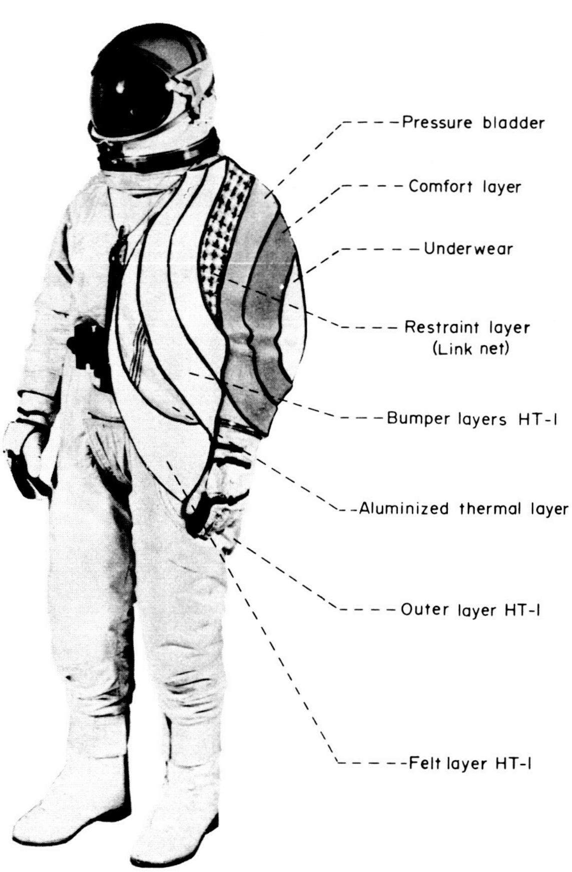

FIGURE 7–14.—Gemini extravehicular space suit.

this discomfort increased significantly with time. After the Gemini IV and V missions, it was concluded that the characteristics of a space suit designed for extravehicular operation were marginal for long-term intravehicular wear.

G5C Space Suit

The G5C space suit was developed for intravehicular use only, and it was used on the Gemini VII mission. It was designed to provide maximum comfort and freedom of movement, with the principal consideration being reduction in bulk. As shown in figure 7–15, the G5C suit is a lightweight suit with a soft fabric hood. The hood, which is a continuation of the torso, incorporates a polycarbonate visor and a pressure-sealing zipper. The zipper installation permits removal of the hood for stowage behind the astronaut's head. The G5C suit provided much less bulk, less resistance to movement, and fewer pressure points than previous space suits. It also was satisfactory for doffing and donning in the crew station. Donning time was about 16 to 17 minutes. In summary, the G5C suit met all its design objectives.

The significant flight results were that the crewmembers felt more comfortable, perspired less, and slept better when they removed the suits entirely. Elimination of the pressure garment resulted in a thermal environment more nearly approximating the conditions of street clothes on earth. With this comfort goal in mind, the Gemini VII crew strongly recommended removal of the space suits during future long-duration manned space-flight missions.

Flight-Crew Equipment

A substantial amount of operational equipment was required in each spacecraft to enable the crew to carry out their mission tasks. This equipment included flight data items, photographic and optical equipment, and a large number of miscellaneous items such as small tools, handheld sensors, medical kits, wristwatches, pencils, and pens. A 16-mm sequence camera and a 70-mm still camera were carried on all the flights. Good results were obtained with these cameras.

An optical sight was used for alining the spacecraft on specific ground objects or landmarks, and it was also effective in aiming at the rendezvous target. The backup rendezvous techniques being developed depend on the aiming and alinement capabilities of the optical sight. The extensive use of this sight for experiments and operational activities made it a necessary item of equipment for all missions.

All of the flight-crew equipment served useful purposes in flight and contributed to the crew's capability to live and work in the spacecraft for short or long missions. The large number of items required considerable attention to detail to insure adequate flight preparation. The most important lesson learned concerning flight-crew equipment was the need for early definition of requirements, and for timely delivery of hardware on a schedule compatible with the spacecraft testing sequence.

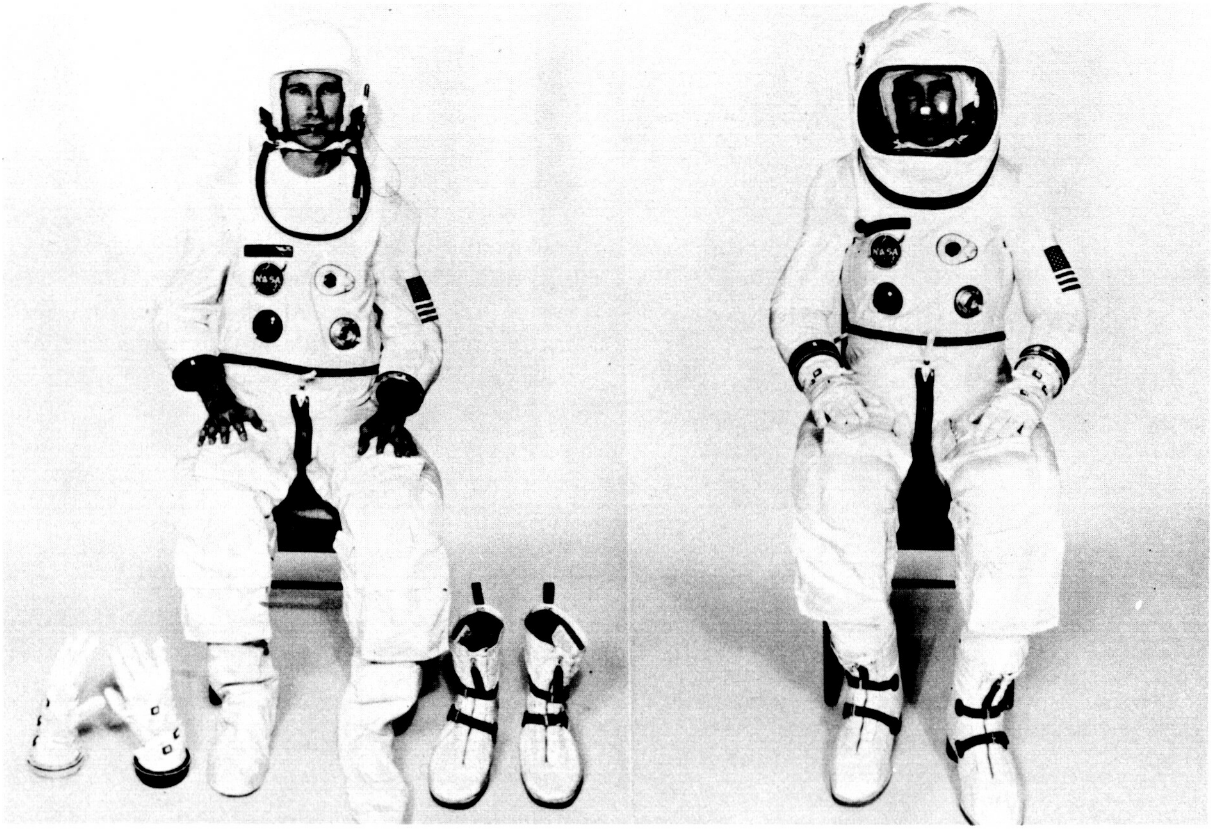

FIGURE 7-15.—Gemini G5C space suit.

Food, Water, Waste, and Personal Hygiene System

Food System

The Gemini food system consists of freeze-dried rehydratable foods and beverages, and bite-sized foods. Each item is vacuum packed in a laminated plastic bag. The items are then combined in units of one or two meals and vacuum packed in a heavy aluminum-foil overwrap. (See fig. 7-16.) The rehydratable food bag incorporates a cylindrical plastic valve which mates with the spacecraft water dispenser for injecting water into the bag. At the other end of the bag is a feeder spout which is unrolled and inserted into the mouth for eating or drinking the contents.

A typical meal consists of two rehydratable foods, two bite-sized items, and a beverage. The average menu provides between 2000 and 2500 calories per man per day. The crews favored menus with typical breakfast, lunch, and dinner selections at appropriate times corresponding to their daily schedule. Occasional leakage of the food bags occurred in use. Because of the hand pressure needed to squeeze the food out of the feeder spout, these leaks were most prevalent in the chunky, rehydratable items. A design change has been made to increase the spout width. The bite-sized foods were satisfactory for snacks but were undesirable for a sustained diet. These items were rich, dry, and, in some cases, slightly abrasive. In addition, some of the bite-sized items tended to crumble. In general, the flight crews preferred the rehydratable foods and beverages.

Drinking-Water Dispenser

The drinking-water dispenser (fig. 7-17) is a pistol configuration with a long tubular barrel which is designed to mate with the drinking port on the space-suit helmet. The water shutoff valve is located at the exit end of the barrel to minimize residual-water spillage. This dispenser was used without difficulty on Gemini III, IV, and V.

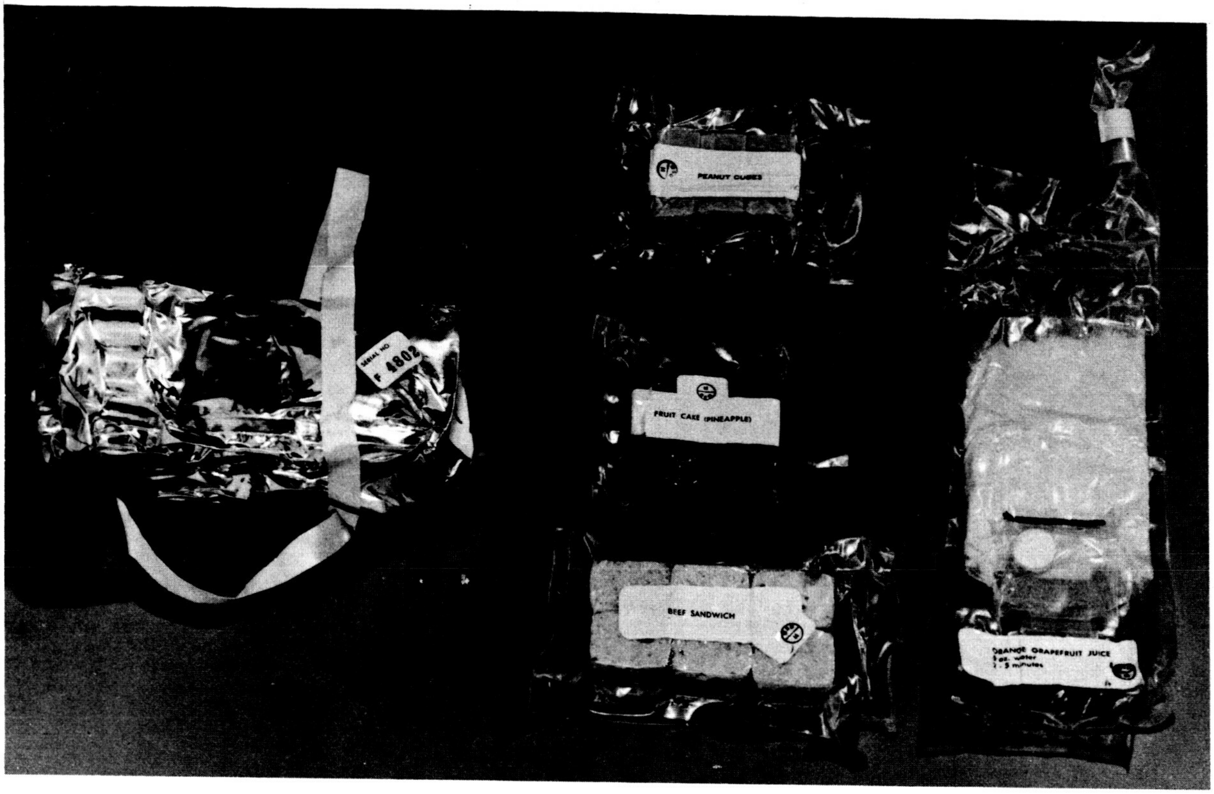

FIGURE 7-16.—Gemini food pack.

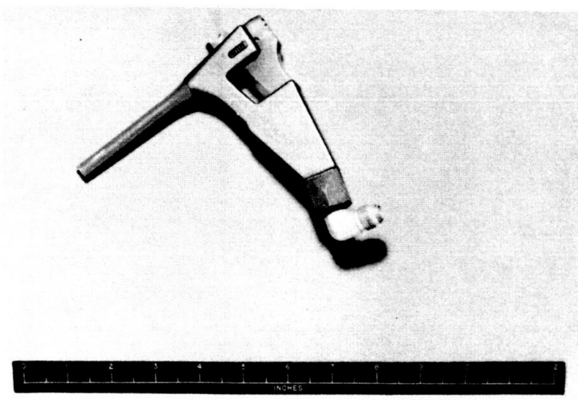

FIGURE 7-17.—Original Gemini water dispenser.

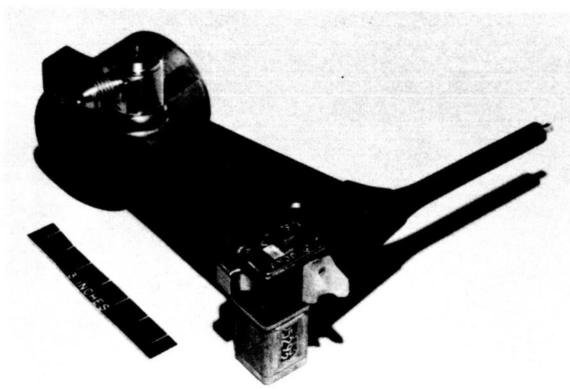

FIGURE 7-18.—Gemini water-metering device.

In order to measure the crew's individual water consumption, a water-metering dispenser (fig. 7-18) was used on Gemini VI-A and VII. Similar to the basic dispenser, this design incorporates a bellows reservoir and a valve arrangement for dispensing water in ½-ounce increments. A digital counter on the handle records each increment dispensed. This dispenser operated satisfactorily on both missions.

Urine Collection System

The Gemini urine system consists of a portable receiver with a Latex roll-on cuff receptacle and a rubberized fabric collection bag. After use, the receiver is attached to the urine-disposal line, and the urine is dumped directly overboard. This system was used without difficulty on the Gemini V and VI-A missions.

On Gemini VII, a chemical urine-volume-measuring system was used to support medical experiments requiring urine sampling. Although this system was similar to the Gemini V system, the increased size and complexity made its use more difficult, and some urine leakage occurred.

Defecation System

The defecation system consisted of individual plastic bags with adhesive-lined circular tops. Hygiene tissues were provided in separate dispensers. Each bag contained a disinfectant packet to eliminate bacteria growth. Use of the bags in flight required considerable care and effort. Adequate training and familiarization enabled the crews to use them without incident.

Personal Hygiene System

Personal hygiene items included hygiene tissues in fabric dispenser packs, fabric towels, wet cleaning pads, toothbrushes, and chewing gum for oral hygiene. These items were satisfactory in flight use.

Extravehicular Operation

Extravehicular Equipment

Early in 1965 the decision was made to conduct self-propelled extravehicular operation on the Gemini IV mission. The extravehicular space suit was the G4C suit described previously. The primary oxygen flow to the extravehicular space suit was supplied through a 25-foot umbilical hose. This oxygen hose was connected to the spacecraft oxygen system in the center cabin area, and the other end was connected to the space-suit inlet fitting. The umbilical provided a normal open-loop oxygen flow of 8.2 pounds per hour. The umbilical also contained communications and bioinstrumentation wiring.

A small chest pack, called the ventilation control module, was developed for control of the space-suit pressurization and ventilation flow (fig. 7-19). Existing Gemini environmental-control-system components were used where possible, since they were already qualified. The

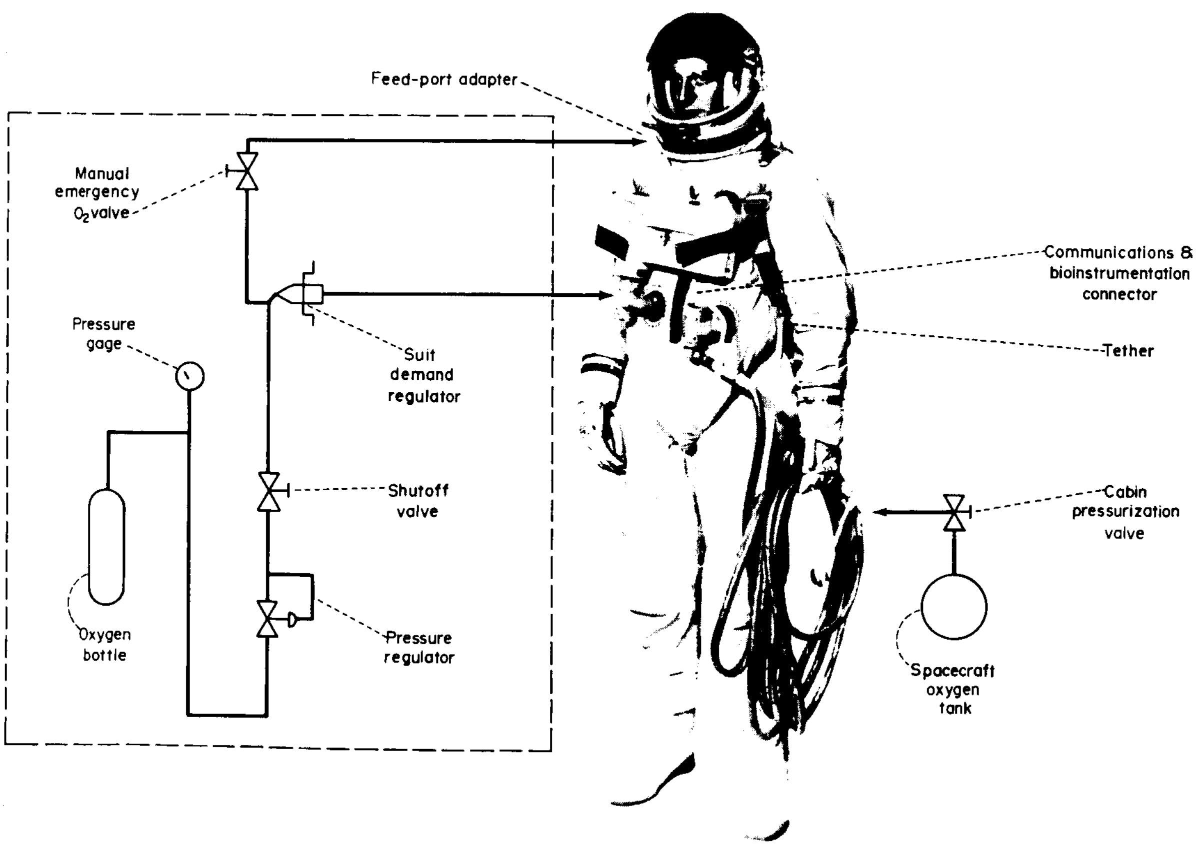

FIGURE 7-19.—Gemini IV extravehicular life-support system.

ventilation control module consisted of a Gemini demand regulator, a 3400-psi oxygen bottle, and suitable valving and plumbing to complete the system. The ventilation control module was attached to the space-suit exhaust fitting and maintained the suit pressure at 4.2 psia. The nominal value was 3.7 psia; however, the pressure in the space suit ran slightly higher because of the pressure drop in the bleed line which established the reference pressure. The reserve-oxygen bottle in the ventilation control module was connected by an orificed line to a port on the helmet. When manually actuated, this reserve bottle supplied oxygen directly to the facial area of the extravehicular pilot.

The handheld maneuvering unit consisted of a system of manually operated cold-gas thrusters, a pair of high-pressure oxygen bottles, a regulator, a shutoff valve, and connecting plumbing (fig. 7–20). The two tractor thrusters were 1 pound each, and the single pusher-thruster was 2 pounds. The flight crew received extensive training in the use of the handheld maneuvering unit on an air-bearing platform, which provided multiple-degree-of-freedom simulation.

The principal spacecraft provisions for extravehicular operation in the Gemini IV spacecraft were the stowage provisions for the ventilation control module and the handheld maneuvering unit, the oxygen supply line in the cabin, and a hatch-closing lanyard. These provisions and all the equipment were evaluated in mockup exercises and zero-gravity aircraft flights. Flight-crew training was also accomplished as a part of these tests and evaluations.

The extravehicular equipment for the Gemini IV mission was subjected to the same rigorous qualification test program as other spacecraft hardware. Prior to the mission, the flight and backup equipment was tested in a series of altitude-chamber tests, following the planned mission profile and culminating in altitude runs with the prime and backup pilots. These altitude-chamber tests, conducted in a boilerplate spacecraft at the Manned Spacecraft Center, provided the final system validation prior to flight.

Flight Results

The flight results of Gemini IV confirmed the initial feasibility of extravehicular operation. Ventilation and pressurization of the space suit were adequate except for peak workloads. During the initial egress activities and during ingress, the cooling capacity of the oxygen flow at 8.2 pounds per hour did not keep the extravehicular pilot cool, and overheating and visor fogging occurred at these times. During the remainder of the extravehicular period, the pilot was comfortably cool.

The mobility of the G4C space suit was adequate for all extravehicular tasks attempted

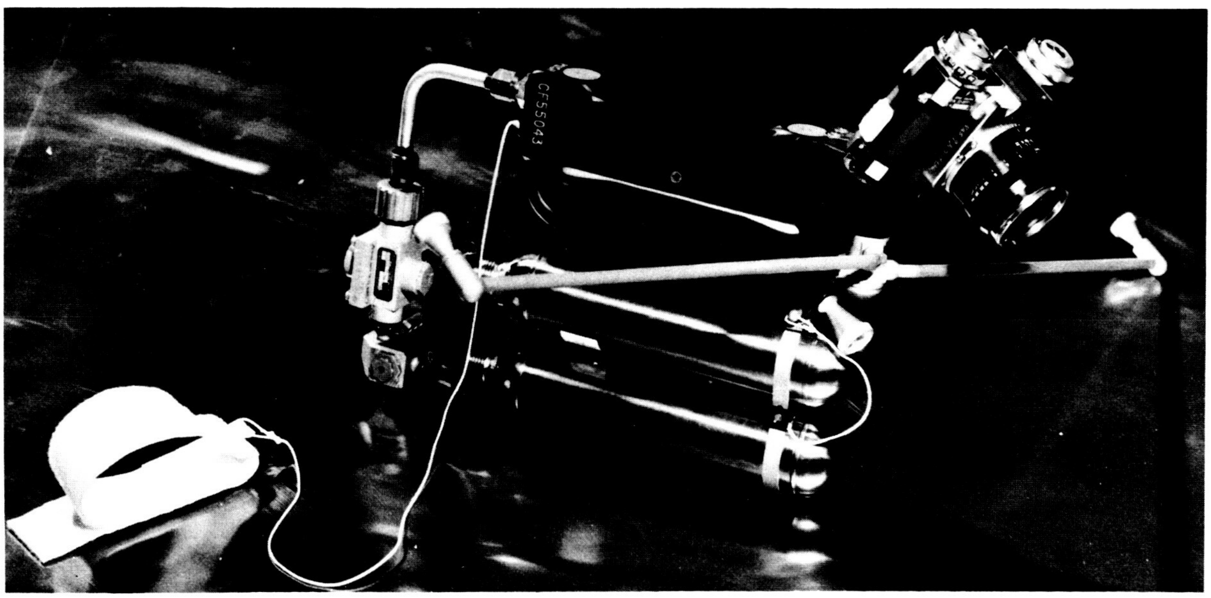

FIGURE 7–20.—Handheld maneuvering unit.

during the Gemini IV mission. The extravehicular visor on the space-suit helmet was found to be essential for looking toward the sun. The extravehicular pilot used the visor throughout the extravehicular period.

The maneuvering capability of the handheld maneuvering unit provided the extravehicular pilot with a velocity increment of approximately 6 feet per second. He executed short translations and small angular maneuvers. Although the limited propellant supply did not permit a detailed stability evaluation, the results indicated that the handheld device was suitable for controlled maneuvers within 25 feet of the spacecraft. The results also indicated the need for longer propellant duration for future extravehicular missions. After the maneuvering propellant was depleted, the extravehicular pilot evaluated techniques of tether handling and self-positioning without propulsive control. His evaluation showed that he was unable to establish a fixed position when he was free of the spacecraft because of the tether reaction and the conservation of momentum. Any time he pushed away from the spacecraft, he reached the end of the tether with a finite velocity, which in turn was reversed and directed back toward the spacecraft. Throughout these maneuvers the extravehicular pilot maintained his orientation satisfactorily, using the spacecraft as his reference coordinate system. At no time did he become disoriented or lose control of his movements.

The ingress operation proceeded normally until the pilot attempted to pull the hatch closed. At this time he experienced minor difficulties in closing the hatch because one of the hatch-locking control levers failed to operate freely. The two pilots operated the hatch-closing lanyard and the hatch-locking mechanism together and closed the hatch satisfactorily. The cabin repressurization was normal.

The results of this first extravehicular operation showed the need for greater cooling capacity and greater propellant duration for future extravehicular missions. The results also showed that extravehicular operation could be conducted on a routine basis with adequate preparation and crew training.

Concluding Remarks

Evaluation of the crew station and the related crew equipment was somewhat subjective, with varying reactions from different crews. In summary, the crew station, as configured for the Gemini VI–A and VII missions, met the crew's needs adequately, and the flight results indicate that this configuration is satisfactory for continued flight use.

8. ENVIRONMENTAL CONTROL SYSTEM

By ROBERT L. FROST, *Gemini Program Office, NASA Manned Spacecraft Center;* JAMES W. THOMPSON, *Gemini Program Office, NASA Manned Spacecraft Center; and* LARRY E. BELL, *Crew Systems Division, NASA Manned Spacecraft Center*

Summary

The environmental control system provides thermal and pressure control, oxygen, drinking water, and waste-water disposal for the crew, and thermal control for spacecraft equipment. An extensive test program was conducted by the spacecraft prime contractor, the subcontractor, and the NASA Manned Spacecraft Center to develop and qualify the system for the Gemini Program. Flight results to date have been good. A minimum number of anomalies have occurred, thus confirming the value of the extensive ground test program.

Introduction

The environmental control system maintains a livable 100-percent-oxygen atmosphere for the crew; controls the temperature of the crew and of spacecraft equipment; and provides a drinking water supply and a means for disposing of waste water. The environmental control system may be subdivided into a suit subsystem, a water management subsystem, and a coolant subsystem. The suit subsystem may be further divided into three systems: the suit, cabin, and oxygen supply systems. The location of these systems in the spacecraft is shown in figure 8–1. All components are grouped into modules where possible to facilitate installation, checkout, and replacement.

The environmental control system design incorporates several redundancies so that no single failure could be catastrophic to the crew. Additional redundancy is included in certain areas to enhance the probability that the system will satisfy requirements for the full duration of the mission. Redundant units are provided for the suit demand regulators, the suit compressor and power supply, the cabin outflow valve, the oxygen supply system, the cooling circuits, and the coolant pumps in each cooling circuit. The cabin pressure regulator and the cabin pressure relief valve are internally redundant.

Suit Subsystem

A schematic of the space-suit, the cabin, and the oxygen-supply systems is shown in figure 8–2. The space-suit module is shown in figure 8–3.

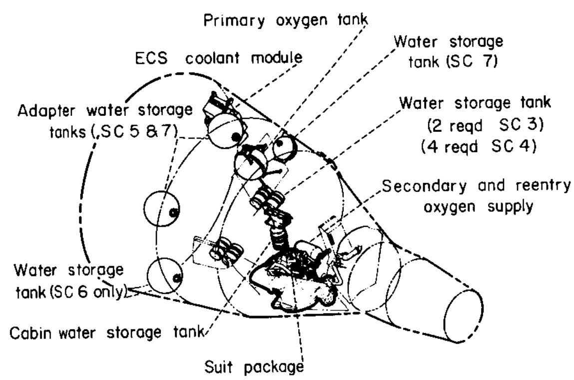

FIGURE 8–1.—Environmental control system.

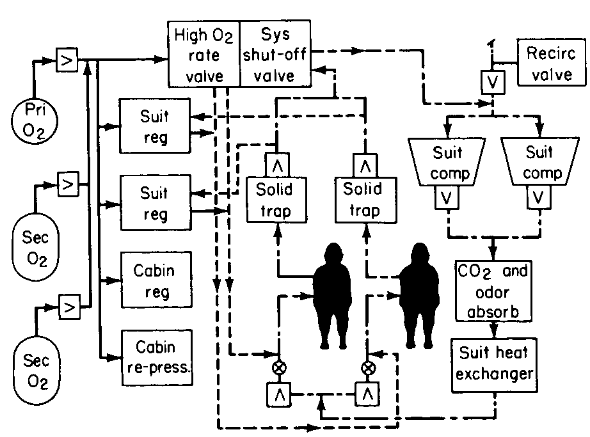

FIGURE 8–2.—Suit subsystem.

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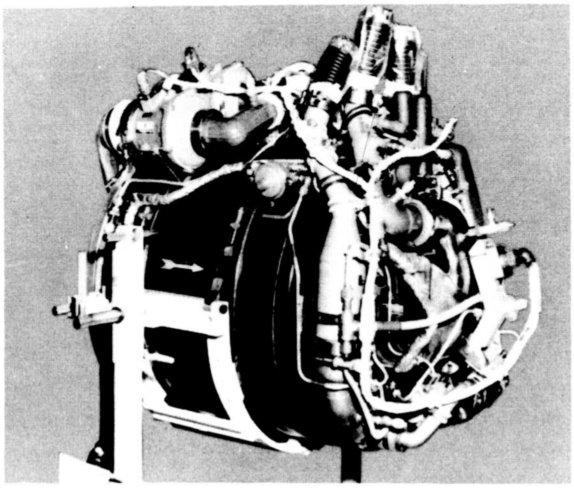


FIGURE 8-3.—Environmental control system suit subsystem module.

Space-Suit System

The space-suit system is a single, closed recirculating system, with the two space suits in parallel. The system provides ventilation, pressure and temperature control, and atmospheric purification. Centrifugal compressors circulate oxygen through the system at approximately 11 cubic feet per minute through each space suit. The two compressors may be operated individually or simultaneously. Carbon dioxide and odors are removed from the oxygen by an absorber bed containing lithium hydroxide and activated charcoal. The amount of lithium hydroxide varies according to the requirements of the mission. The oxygen can be cooled in the suit heat exchanger to as low as 48° F; however, the actual temperature is a function of crew activity, coolant subsystem operating mode, and system adjustments made by the crew. Adjustments can be made both for coolant flow rate through the suit heat exchanger and for oxygen flow rate through the space suit.

Water given off by the crew as perspiration and expiration is condensed in the suit heat exchanger and routed to the launch-cooling heat exchanger.

The two demand regulators function to maintain a suit pressure approximately equal to cabin pressure. The demand regulators also maintain a minimum suit pressure of 3.5 psia any time the cabin pressure drops below that level. Should the suit pressure drop to a level between 3.0 and 3.1 psia, the absolute-pressure switch actuates, closing the dual secondary-flow-rate and system-shutoff valve, thereby changing to an open-loop configuration having a flow of 0.08 to 0.1 pound of oxygen per minute through each space suit. The recirculation valve is normally open so that, when the suit visors are open, cabin gas will be circulated through the suit system for purification.

Cabin System

The cabin system includes a fan and heat exchanger, a pressure regulator, a pressure-relief valve, an inflow snorkel valve, an outflow valve, and a repressurization valve. The cabin fan circulates gas through the heat exchanger to provide cooling for cabin equipment. The cabin pressure regulator controls cabin pressure to a nominal 5.1 psia.

Oxygen-Supply System

The oxygen-supply system uses two sources of oxygen. The primary source, located in the equipment-adapter section, is a tank containing liquid oxygen stored at supercritical pressures. The second supply is gaseous oxygen stored at 5000 psi in two bottles located inside the cabin section. The secondary supply supplements the primary supply in case of failure and becomes the primary supply during reentry. Each secondary bottle contains enough oxygen for one orbit at the normal consumption rate, plus a normal reentry at the oxygen high rate of 0.08 pound of oxygen per minute to each astronaut.

Water Management Subsystem

Drinking Water Systems

The water management subsystem includes a 16-pound-capacity water tank, a water dispenser, and the necessary valves and controls, all located in the cabin, plus a water storage system located in the adapter. The adapter water storage systems for the battery-powered spacecraft consisted of one or more containers, each having a bladder with one side pressurized with gas to force water into the cabin tank.

The water storage systems on fuel-cell-powered spacecraft is similar to the battery configuration. Fuel-cell product water is stored on the gas side of the bladder in the drinking-water storage tanks. Regulators were added to control the fuel-cell product water pressure as required by the fuel cell. The initial design concept called for the flight crew to drink the fuel-cell product water; however, tests revealed that fuel-cell product water is not potable, and the present design was adopted.

Waste-Water Disposal System

Waste-water disposal is accomplished by two different methods. Condensate from the suit heat exchanger is routed to the launch-cooling heat exchanger for boiling, if additional cooling is required, or is dumped overboard. Urine is dumped directly overboard, or it can be routed to the launch-cooling heat exchanger should the primary systems fail or additional cooling be required. To prevent freezing, the outlet of the direct overboard dump is warmed by coolant lines and an electric heater.

Coolant Subsystem

The coolant subsystem provides cooling for the crew and thermal control for spacecraft components. Electronic equipment is mounted on cold plates. The system, shown schematically in figure 8-4, consists of two completely redundant circuits or loops, each having redundant pumps. For clarity, the coolant lines for the secondary loop are omitted from the figure. All heat exchangers and cold plates, except for the regenerative heat exchangers and the fuel cells, have passages for each loop. On spacecraft 7, the secondary or B pump in each coolant loop was equipped with a power supply that reduced the coolant flow rate to approximately half that of the primary or A pump. This change was made in order to reduce total power consumption, to maintain higher adapter temperatures during periods of low power

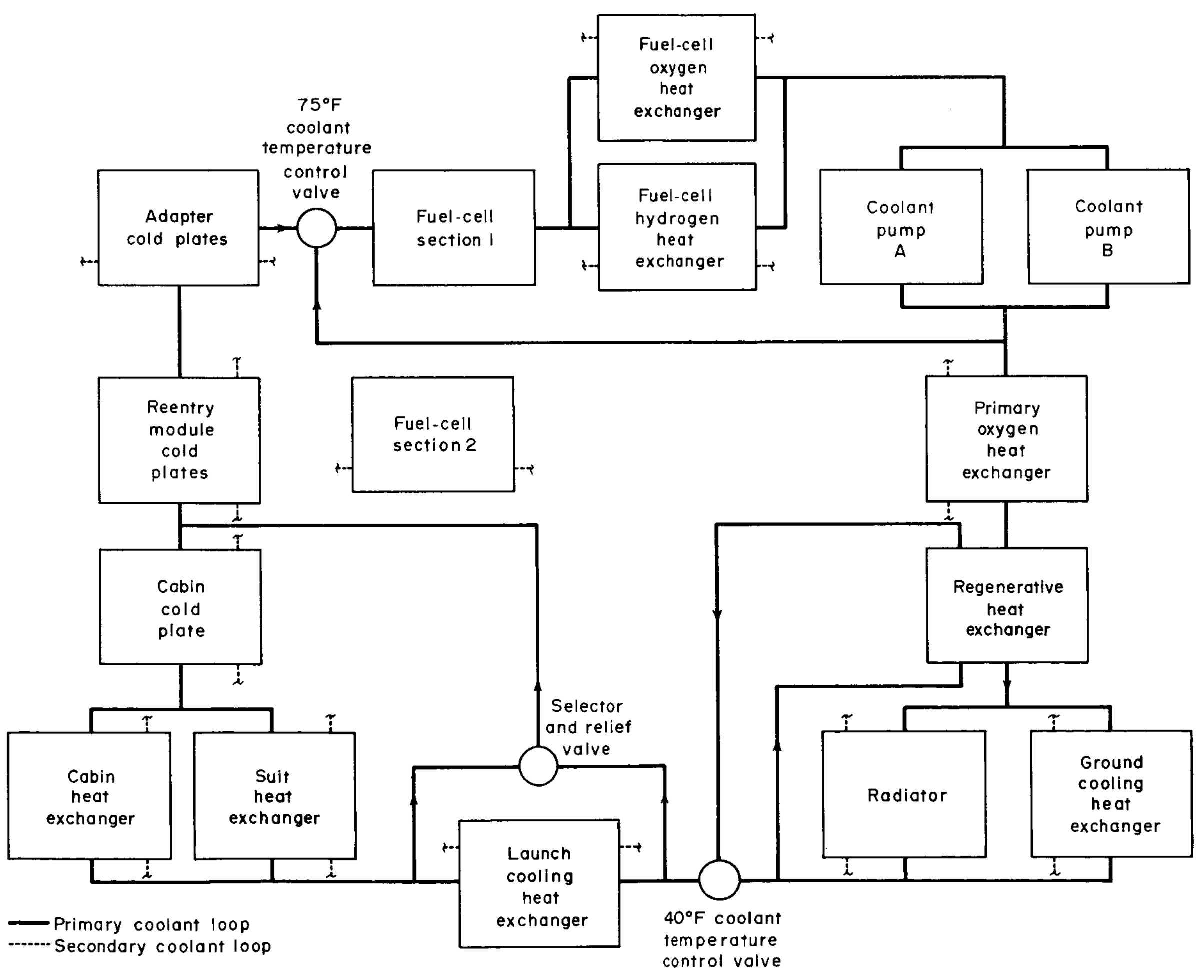


FIGURE 8-4.—Coolant subsystem.

usage, and also to allow greater flexibility in maintaining optimum coolant temperatures for the resultant variations in thermal loads.

Battery-powered spacecraft require the use of only one coolant loop at a time, whereas the fuel-cell-powered spacecraft require both loops, as each fuel-cell section is on a different loop. By using both coolant pumps simultaneously, one loop is capable of handling the maximum cooling requirements should the other loop fail. The coolant loops have two points of automatic temperature control: radiator outlet temperature is controlled to 40° F, and fuel-cell inlet temperature is controlled to 75° F. Prelaunch cooling is provided through the ground-cooling heat exchanger. The launch-cooling heat exchanger provides cooling during powered flight and during the first few minutes of orbital flight until the radiator cools down and becomes effective. The heat exchanger also supplements the radiator, if required, at any time during flight by automatically controlling the heat-exchanger outlet temperature to a nominal 46° F.

The spacecraft radiator (fig. 8–5) is an integral part of the spacecraft adapter. The coolant tubes are integral parts of the adapter stringers, and the adapter skin acts as a fin. Alternate stringers carry coolant tubes from each loop, and all tubes for one loop are in series. Coolant flows first around the retrosection and then around the equipment section of the adapter. Strips of high-absorptivity tape are added to the outer surface of the adapter to optimize the effective radiator area for the cooling requirements of each spacecraft.

Test Programs

The environmental-control-system program consisted of development, qualification, and reliability tests, covering 16 different environments, conducted by the vendor, and of systems tests conducted by the spacecraft contractor and by Manned Spacecraft Center organizations.

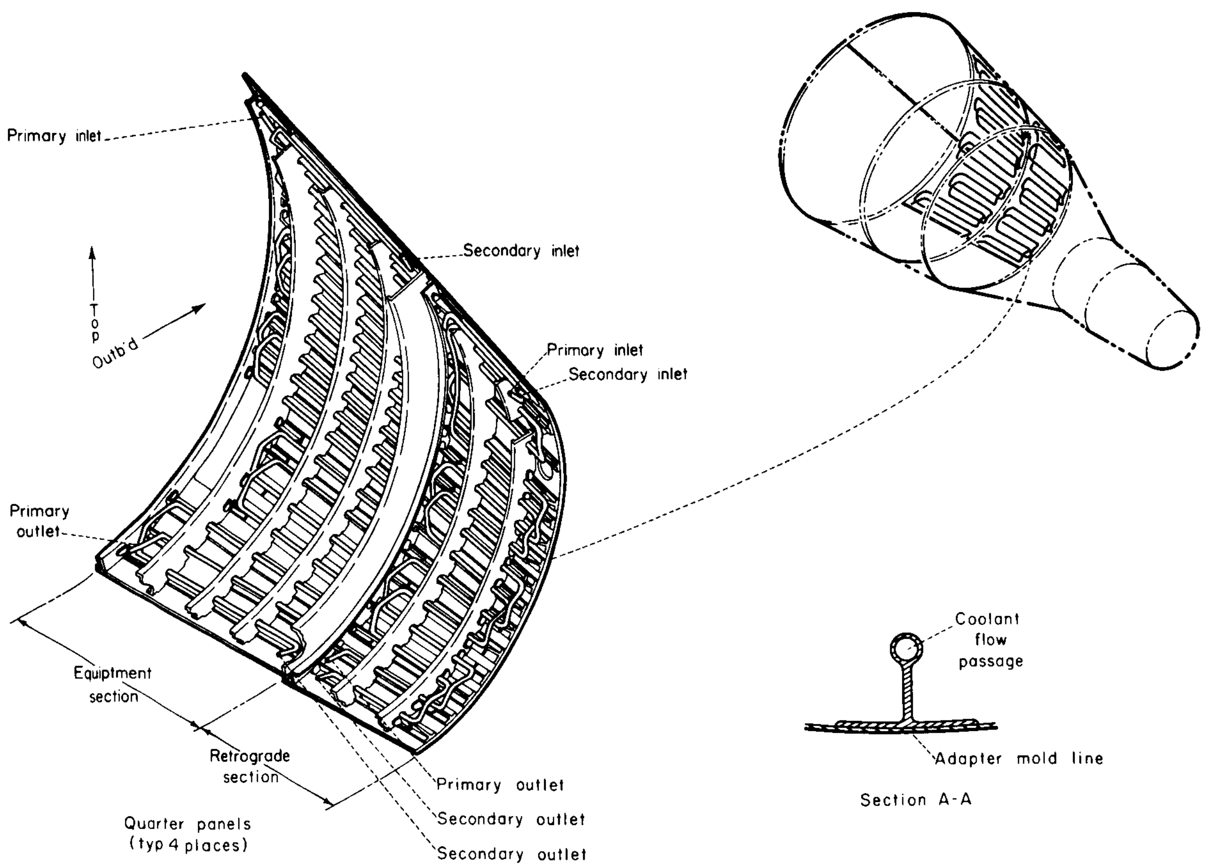


FIGURE 8–5.—Spacecraft radiator.

During the development of the components for the environmental control system, designs were verified with production prototypes rather than with engineering models. For example, if a pressure regulator was to be produced as a casting, the test model was also produced as a casting. As a result, additional production development was eliminated, and confidence with respect to flightworthiness was accumulated from developmental tests as well as from later qualification and system reliability tests. Development tests included manned altitude testing on a boilerplate spacecraft equipped with the suit and cabin portion of the environmental control system.

Where possible, qualification of the environmental control system has been demonstrated at the system level, rather than at the component level, because of the close interrelationships of components, especially with respect to thermal performance. Test environments included humidity, salt-water immersion, salt-solution, thermal shock, high and low temperature and pressure, proof, burst, vibration, acceleration, and shock.

System qualification tests were followed by simulated mission reliability tests consisting of eight 2-day, three 7-day, and eight 14-day tests of a single environmental control system. In these tests, all the environmental-control-system components mounted in the cabin and spacecraft adapter section were exposed to simulated altitude, temperature cycling, and temperature extremes in an altitude chamber. Moisture and carbon-dioxide atmospheric conditions were provided by crewman simulators. After each of these tests, the oxygen containers were serviced, and the lithium hydroxide canisters were replaced; otherwise, the same components were used for all tests.

These tests revealed that heat transfer from the lithium hydroxide canister to ambient was greater than expected. This increased heat transfer caused chilling of the gas stream near the outer periphery of the chemical bed, sufficient to cause condensation of water from the gas stream. The condensation reduced the life of the chemical bed by approximately 45 percent based on a metabolic input rate of 500 Btu per hour per man. The canister was redesigned to include a layer of insulation between the chemical and the outer shell of the canister. Also, the estimate of the metabolic rate was reevaluated and was reduced based on the results of previous flights. Test reruns then used metabolic rate inputs of 370 and 450 Btu per hour per man. The new design successfully met all mission requirements.

Early in the Gemini Program, a boilerplate spacecraft was fabricated to simulate the cabin portion of the reentry assembly, with adequate safety provisions for manned testing under any operating condition. Sixteen manned tests were conducted—four at sea level, six at altitude with a simulated coolant subsystem, and six at altitude with a complete system, except that the radiator was simulated only by pressure drop. System cooling was provided through the ground-cooling heat exchanger. After satisfactory completion of the spacecraft contractor's test program, the boilerplate model was shipped to the Manned Spacecraft Center, where it was used in numerous manned tests.

The boilerplate proved a valuable test article, as it pointed out several potential problems which were corrected on the flight systems. The most significant of these was the crew discomfort caused by inadequate cooling during levels of high activity. The inadequate cooling was determined to be a result of excessive heat gain in the coolant fluid between the temperature control valve and the suit heat exchanger. Insulation was added to the coolant lines and to the heat exchanger. In addition, a flow-limiting orifice was added between the suit and cabin heat exchangers to assure adequate flow of coolant in the suit heat exchanger. Also, the capability to run both suit compressors was added to cover any activity level. With these changes, the environmental control system was demonstrated to have adequate capability.

During the boilerplate tests at the Manned Spacecraft Center, no problems were encountered with the environmental control system. The boilerplate played a valuable role in qualification of the Gemini space suit, the Gemini IV extravehicular equipment, and the extravehicular life-support systems for future missions.

Static article 5 was a production spacecraft reentry assembly and was used in flotation and postlanding tests. The portions of the environmental control system required for use after landing were operated during manned tests in the Gulf of Mexico. This testing demonstrated satisfactory cooling and carbon-dioxide removable for up to 19 hours of sea recovery time.

A series of three thermal qualification tests was conducted on spacecraft 3A, which was a complete flight-configuration spacecraft with the exception of fuel cells. Fuel-cell heat loads were simulated with electric heaters. The entire spacecraft was placed in an altitude chamber equipped with heat lamps for solar simulation and with liquid-nitrogen cold walls to enable simulating an orbital day-and-night cycle.

During the first test, which lasted 12 hours, the adapter temperatures were colder than desired, indicating that the radiator was oversized for the thermal load being imposed by the spacecraft systems. As a result, the drinking and waste-water lines froze, and the oxidizer lines and components in the propulsion system became marginally cold. After the data from the first test were analyzed, resistance heaters were added to the adapter water lines, flow-limiting valves were installed in the fuel-cell temperature-control-valve bypass line, and provisions were made to vary the effective radiator area.

The second test lasted 135 hours, and the spacecraft maintained thermal control. The resistance heaters kept the water lines well above freezing, but the propulsion-system oxidizer lines remained excessively cold, indicating the need for similar heaters on these lines.

The most significant gains were the successful raising of the adapter temperature and the improved environmental-control-system performance with the reduced effective area of the radiator. By adding strips of high-absorptivity tape, the effective area of the radiator can be optimized for each spacecraft, based on its specific mission profile.

Excellent thermal control was maintained for the entire 190 hours of the third test, demonstrating the adequacy of the environmental control system with the corrective action taken after the first and second tests. The only anomaly during the test was condensate forming in the cabin. The spacecraft contractor and NASA both studied the possibility of condensate forming during orbital flight, and two approaches to the problem were examined. The Manned Spacecraft Center initiated the design and fabrication of a humidity-control device that could be installed in the cabin. In the interim, the spacecraft contractor took immediate precautions by applying a moisture-absorbent material on the interior cabin walls of the Gemini IV spacecraft. During the Gemini IV mission, humidity readings were taken, and no moisture was observed. Consequently, development of the humidity-control device was terminated after initial testing, as condensation did not appear to be a problem during orbital operation.

The validity of the thermal qualification test program has been demonstrated on the first five manned flights. The high degree of accuracy in preflight predictions of thermal performance and sizing of the radiator area is due, in large part, to the spacecraft 3A test results.

Flight Results

Performance of the environmental control system has been good throughout all flights, with a minimum number of anomalies. Crewman comfort has been generally good. A review of the data from all flights shows that an indicated suit inlet temperature of 52° to 54° F is best for maintaining crew comfort. Actual suit inlet temperatures are 10° to 20° F higher than indicated because of heat transfer from the cabin to the ducting downstream of the temperature sensor. Suit inlet temperatures were in or near the indicated range on all flights except during the Gemini VI–A mission. During this flight, except for the sleep period, the temperature increased to over 60° F, causing the crew to be warm. Detailed postflight testing of the environmental control system showed no failures. The discomfort is attributed to a high crewman metabolic-heat rate resulting from the heavy workload during the short flight. The design level for the suit heat exchanger is 500 Btu per hour per man. Experience gained since the design requirements were established has shown that the average metabolic rate of the crew is around 500 Btu per hour per man on short flights and between 330 and 395 Btu per hour per man on long-duration flights. (See fig. 8–6.)

The most comfortable conditions proved to be during the suits-off operation of the Gemini VII flight. Preflight analysis had determined

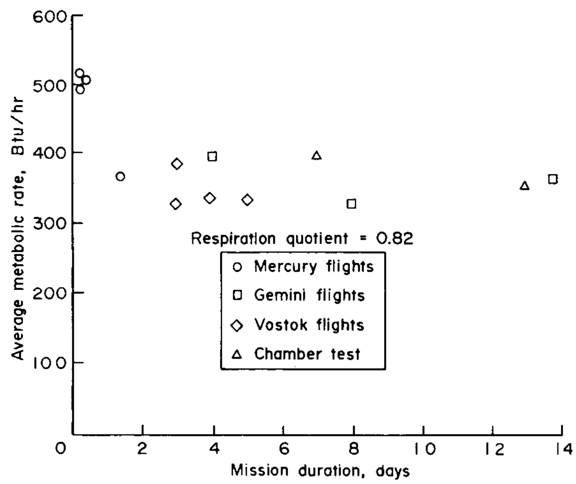


FIGURE 8-6.—Crewman metabolic rate.

that, because of insufficient gas flow over the body, the crew might not be as comfortable as would be desired. However, the crew found that relatively little air flow over the body was necessary. The suits-off operation had very little effect on the cabin environment. Cabin air and wall temperatures were between 75° and 80° F, which was normal after stabilization on all flights. Cabin relative humidity was between 48 and 56 percent during suits-off operation, which was lower than the 50 to 72 percent experienced on other flights. This was as expected because the sensible-to-latent cooling ratio was higher with the suits off than with the suits on.

Condensation has not been a problem during flight, contrary to the indications during the spacecraft 3A testing. Spacecraft 3A testing assumed a fixed spacecraft attitude. This would cause greater temperature gradients in the cabin than the drifting mode normally used during the missions. Significant condensation has occurred only once during the program. During the Gemini VII mission, the crew reported free moisture leaving the suit inlet hoses at approximately 267 hours after lift-off and again at 315 hours. Also, a buildup of condensation was noted on the floor and on the center pedestal at this time. The exact cause has not been determined, but two possibilities are that some ducts experienced local chilling as a result of spacecraft attitude and that a degradation or failure occurred in the condensate removal system. Circumstances both support and reject these possibilities.

Cabin temperature has not increased during reentry as was originally expected. Initial calculations showed an increase of 70° to 120° F during reentry, whereas the actual increase has been less than 10° F. The thermal effectiveness of the insulation and structural-heat flow paths is greater than could be determined analytically.

During the Gemini II mission, the pressure in the cryogenic containers dropped approximately 30 percent just after separation of the spacecraft from the launch vehicle. Extensive postflight testing determined that the pressure drop resulted from thermal stratification within the cryogen. The separation maneuver caused mixing, which reduced the stratification and resulted in a lower stabilized pressure. The prelaunch procedures have been modified to bring the container pressure up to operating levels at a much slower rate, thus minimizing the stratification. A pressure drop has been experienced on only one mission since Gemini II.

Concluding Remarks

The excellent flight results to date, with a minimum number of anomalies, confirm the value of the extensive ground test program conducted on the system. Condensation in the cabin has not been a problem, as was originally indicated. Also, it appears that the metabolic heat load of the crew during periods of high activity may be more than 500 Btu per hour per man.

9. SPACECRAFT MANUFACTURING AND INPLANT CHECKOUT

By Walter F. Burke, *Vice President and General Manager, Spacecraft and Missiles, McDonnell Aircraft Corp.*

Introduction

The technology of space exploration is expanding at an extremely rapid rate. McDonnell Aircraft Corp. of St. Louis, as the prime contractor to NASA for the design and manufacture of the Gemini spacecraft, has been able to meet this challenge with its highly integrated operations, covering all aspects of the technical disciplines required. Figure 9–1 shows the physical layout of their facilities. Of particular interest to this presentation is the location of the Engineering Campus, the Fabrication Building, the Laboratory Complex, and the McDonnell Space Center. The latter includes its self-contained Engineering Office Building, in which the major portion of the Gemini engineering activity is conducted.

Corporate Organization

To support the Gemini Program a combination of functional and project-line organizations has been found necessary to provide a rapid response and to assure the maximum utilization of knowledge, personnel, and equipment for the diverse disciplines required. This dual breakdown has been demonstrated to be a very satis-

Figure 9–1.—McDonnell Aircraft Corp., St. Louis, Mo.

factory arrangement for getting corporate-wide action at a very fast response rate.

The officers in charge of the functional sections are responsible for providing the required number of personnel to accomplish the various disciplines in all the programs, to evaluate the caliber of the individual's effort, and to establish means of crossfeeding information between projects.

Project Organization

Upon receipt of a specific contract, a project organization is set up with its project manager reporting directly to the vice president and general manager for that line of business. The nature of the Gemini Program made it desirable for this to be one and the same person. The project organization, in a sense, is a company within a company. The project manager is responsible for all decisions on that particular project and has full authority over the personnel assigned to the task. It is this line organization which has proven so successful, enabling management to concentrate all necessary attention to problem areas as quickly as they arise, and to carry out the necessary action at a very rapid pace. In the project organization, for example, the manufacturing manager is responsible for all of the following functions:

(1) Establishment of the manufacturing plan.

(2) Tool design.

(3) Establishing process development requirements.

(4) Training of personnel to productionize new manufacturing processes.

(5) Determination of facility requirements.

(6) Arrangement of spacecraft production lines and associated facilities.

(7) Tool manufacture.

(8) Production planning (preparation of individual operation sheets).

(9) Production control.

(10) Mockup construction.

(11) Final assembly.

(12) Test participation.

(13) Preparation for the shipping of completed vehicles.

In addition, the Gemini Program Technical Director, Procurement Manager, Spacecraft Product Support Manager, and Program Systems Manager have similar authority in the project organization.

Gemini Modular Concept

From the very beginning, the Gemini spacecraft was designed to be an operational vehicle with capabilities for late mission changes and rapid countdown on the launch pad. Based on experience with Project Mercury, this definitely dictated the use of a modular form of spacecraft in which complete systems could be added to, subtracted from, or replaced with a minimum impact on schedule. Figure 9–2

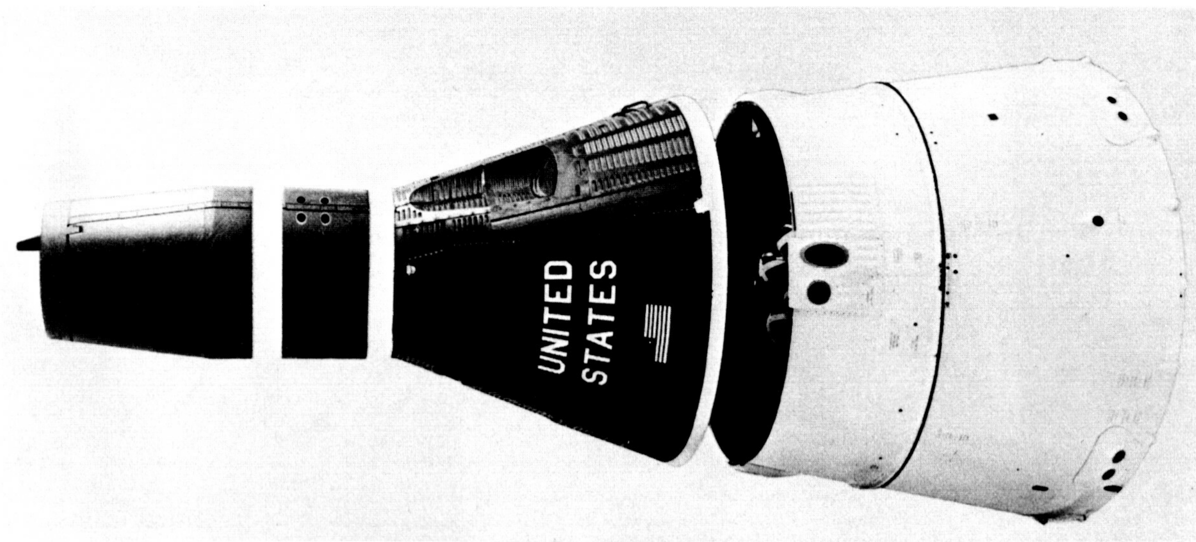


FIGURE 9–2.—Gemini spacecraft modular assembly.

shows how this was accomplished in the Gemini spacecraft, where, reading from left to right, the individual sections are the—

(1) Rendezvous and recovery.
(2) Reentry control system.
(3) Reentry cabin.
(4) Retrograde-adapter and equipment-adapter sections (adapter assembly).

Each of these sections is fabricated and assembled in the manufacturing area of the Space Center, and furnished with its equipment and checked as a separate entity in the Gemini white room before being mated with any of the other sections. With this form of modular construction, it is possible to accomplish the work as a series of parallel tasks, thus permitting a larger number of personnel to be effectively working on the total spacecraft on a noninterference basis, thereby greatly reducing the overall cost of such a vehicle. In addition, during the test program, the effect of a variation in test results will affect only that section, and not slow down the overall test program. In like manner, when a spacecraft has been mated, any module may be removed from a section and replaced by another with little or no impact on the launch schedule, as has been evidenced on several occasions during the Gemini Program to date.

Care was paid in design, particularly in the reentry section, so that no components are installed in a layered or stacked condition. In this way, any component can be removed or installed without disturbing any other. Another requirement was that each wire bundle be so designed that it could be manufactured and electrically tested away from the spacecraft, and that its installation primarily be a lay-in operation. No soldering is planned to be done on the spacecraft during the installation and assembly period. This provided for much greater reliability of terminal attachments and permitted the manufacture of many wire bundles to proceed simultaneously without interference. As a measure of its effectiveness in providing a quality product, spacecraft 5 had zero defects in the 6000 electrical check points monitored. It was also required that each component be attached in such a manner that access to it be possible by the technicians without the use of special tools. For ease of testing, each black-box component was designed with an aerospace-ground equipment test plug, bringing those necessary test parameters right to the surface of the box, and permitting the hooking-up of the test cabling with no disruption of the spacecraft wiring to the box. In this way, particularly during the development phase, it was possible to evaluate the performance of each component while it was connected directly into the spacecraft wiring and to minimize the number of times connections had to be made or broken.

Gemini Manufacturing Work Plan

With the modular concept established and with the engineering progressing, manufacturing planners, under the manufacturing manager, began the layout of the manufacturing work plan, as shown in figure 9–3. The bottom of figure 9–3 shows the work plan for the adapter, with subassemblies of the retrorocket support structure, the panels of the space radiator, the buildup of the basic adapter structural assembly, and the time span allotted to installation. This workload was broken down into three units—A3, A2, and A1—each of which is a station for installation of the equipment spelled out in the attached blocks of the diagram. Upon completion of these installations, an engineering review was held prior to beginning the sectional spacecraft system tests.

In a similar manner, the rendezvous and recovery section and the reentry control system section have been displayed. The longest cycle time and, therefore, the critical path involve the reentry section. Because of the complexity of this section, it is broken down into many more subassemblies, beginning with hatch sills, main frames, left-side and right-side panels, cabin structural weld assemblies, and the cabin intermediate assembly. Upon completion of this portion of the manufacturing, the assembly is submitted to a detailed inspection and cleanup and transported to the white room. In the white room, the components which will be installed in the cabin are first put through a preinstallation acceptance test and then mounted in the cabin as defined by the attached planning sequences shown in figure 9–3. Upon completion of these installations, an engineering review is again performed, and then the reentry section is subjected to a very detailed space-

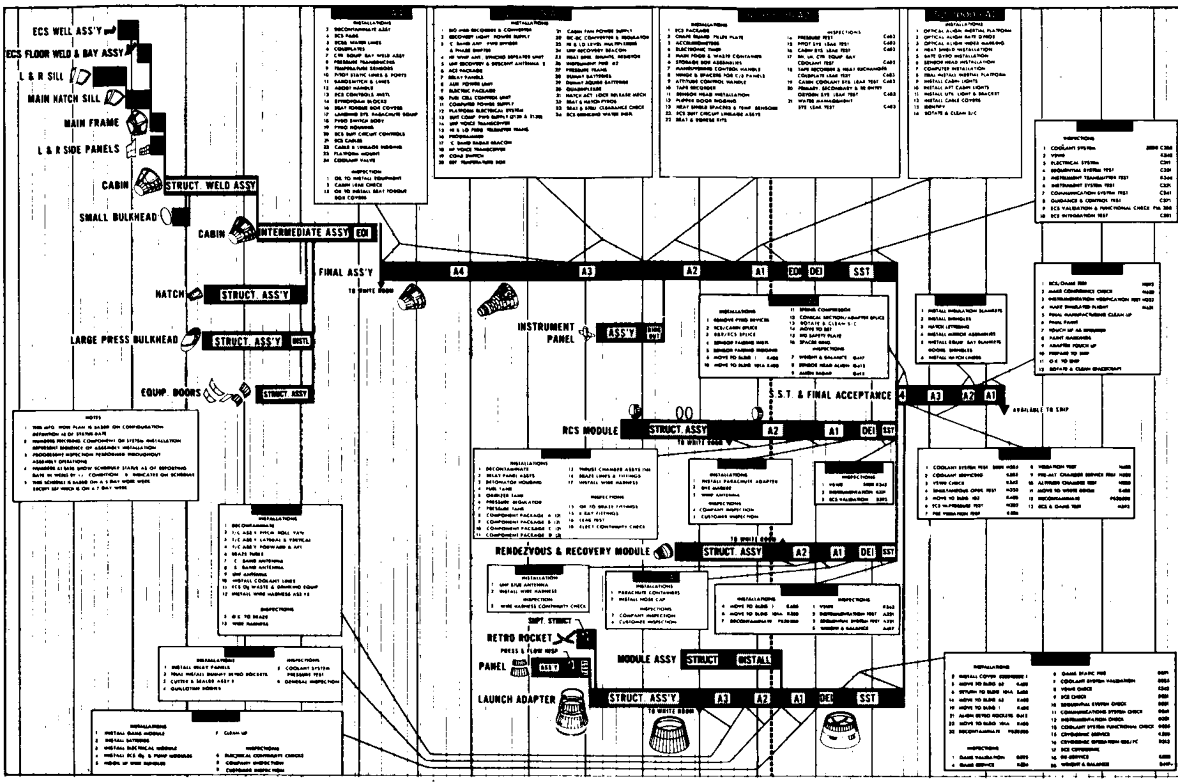


FIGURE 9-3.—Gemini spacecraft 4 manufacturing work plan.

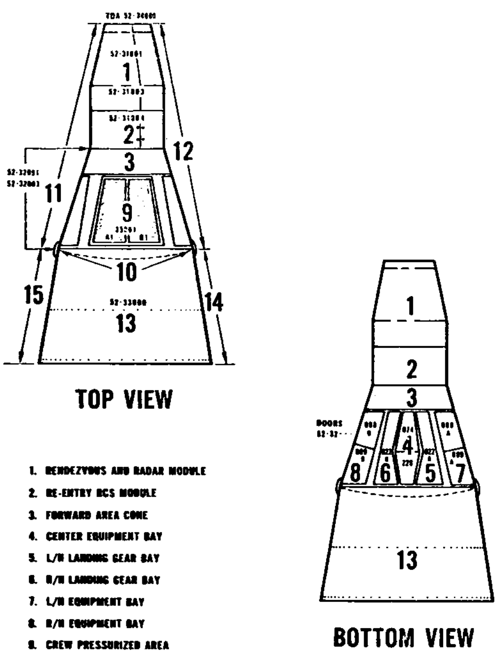
craft systems test at the module level. At this point in the manufacturing cycle, the three sections and the adapter assembly are assembled and the end-to-end spacecraft systems tests performed. From this manufacturing work plan, it can be seen that activities can be conducted on many zones of the spacecraft simultaneously, thus permitting significant reduction in the overall cycle time and minimizing the impact of problems arising in the individual sections.

Control of Work Status

Manufacturing planners have the responsibility for determining the sequence in which individual installations are made. Obviously, this requires an evaluation of the time to make a particular installation and requires the assignment of the tasks to prevent delays due to interference between the production personnel. To accomplish this, the spacecraft was divided into work zones as shown in figure 9-4, which is a typical work sheet. In each one of these numbered areas is work that can be accomplished, either in the structural assembly or installation areas. The key for this breakdown is shown in the lower left corner of figure 9-4 and is self-explanatory.

Manufacturing production control is responsible for bringing the necessary parts to the jig or installation station in time to meet the schedule. As an aid in the performance of this job, the status of the equipment for each zone was maintained in the form shown on the right side of figure 9-4, where zone 9 is typical. Here, it can be seen that production control has determined the number of pieces of equipment required, the number on hand, what additional pieces are still expected to arrive on the required schedule date, and, most significant, what pieces of equipment are at that particular time, to be late for installation. Each piece of late equipment is analyzed as to its point of normal installation and the amount of delay expected, and then a decision is made as to its installation at a point farther down the line. Along with this information, the time required to install the late pieces of equipment is tabulated so that the production supervisor will be constantly

GEMINI ZONE CHART



1. RENDEZVOUS AND RADAR MODULE
2. RE-ENTRY RCS MODULE
3. FORWARD AREA CONE
4. CENTER EQUIPMENT BAY
5. L/H LANDING GEAR BAY
6. R/H LANDING GEAR BAY
7. L/H EQUIPMENT BAY
8. R/H EQUIPMENT BAY
9. CREW PRESSURIZED AREA
10. EXTERIOR AREA OF LARGE PRESS. BULKHEAD
11. SPACECRAFT EXTERIOR L/H
12. SPACECRAFT EXTERIOR R/H
13. INTERNAL ADAPTER AREA
14. R/H EXTERNAL ADAPTER AREA
15. L/H EXTERNAL ADAPTER AREA
16. ZONE FOR ENTIRE SPACECRAFT

GEMINI SPACECRAFT NO. 4
ZONE NO. 9
TOTAL EQUIPMENT REQUIRED 118
EQUIPMENT ON HAND 91
EQUIPMENT TO SCHEDULE 15
EQUIPMENT LATE 12

PART NO.	PART NAME	NORM. INST.	SLACK	DEF. INST.	SLACK	HOURS TO INSTALL
83700-123	MANUAL SHUT OFF VALVE	00005 A4	-8	00005 A2	+1	3
83700-277	E.C.S SECONDARY	00005 A4	-7	00005 A2	+2	2
41703-183-121	BRIDLE DISC ASSY	00005 A4	-6	00005 A2	+1	4
83162-15	BUTTERFLY S/O VALVE	00005 A4	-7	00005 A2	+1	2
83149-17	CABIN PRESS. VALVE	00005 A4	-7	00005 A2	+2	3
72714-7-123	SWITCH BODY (2)	00005 A4	-6	00005 A2	0	3
72714-9-123	SWITCH BODY (5)	00005 A4	-6	00005 A2	0	3

FIGURE 9-4.—Gemini spacecraft 4 zone chart.

aware of any overload of work coming to his station, and therefore, making the necessary provisions, either of added manpower or overtime.

Management Control

While figures 9-3 and 9-4 have shown the formal nature in which the work is planned and controlled, it still takes personal action on the part of all levels of supervision to accomplish the task. At McDonnell Aircraft Corp., this is accomplished through the medium of three particular action centers, as shown on figure 9-5. A project management meeting is held daily, chaired by the Project Manager. In this meeting are discussed the manpower assignments, comparison of the work accomplished versus the man-hours expended, status of the spacecraft to the schedule, and situations resulting in red-flag items; then management directives are issued.

In a similar manner, the technical staff conducts a daily meeting, chaired by the Engineering Manager. Here, the design is coordinated in compliance with customer technical inputs, study assignments are made, and test feedback is discussed as to its effect on engineering specifications.

A configuration control board meets on a bidaily interval, chaired by the Project Control Manager. Here, engineering change proposals are discussed, thus keeping up to date all elements of the project regarding the spacecraft configuration. Analyses of the schedule impact of these changes are made, and a spacecraft effectivity for the change incorporation is established.

As shown by the arrows on figure 9-5, there is a three-way distribution of this information

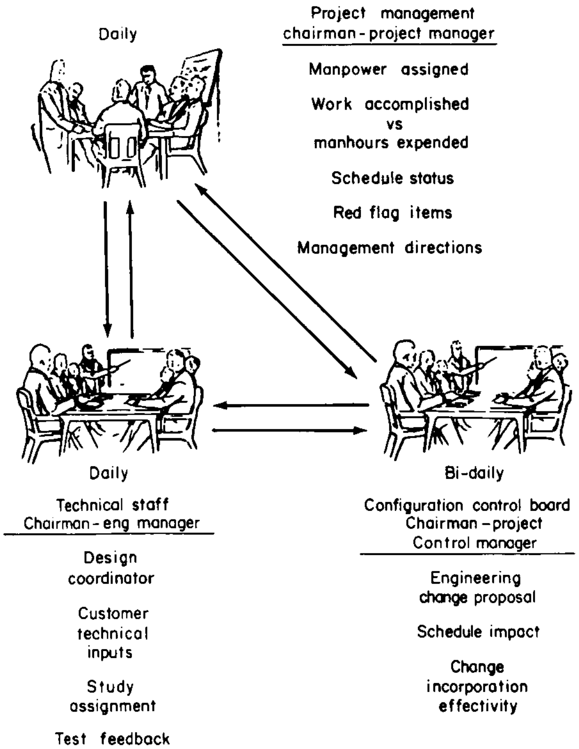


FIGURE 9-5.—Management control.

as decisions in any one of these meetings have their effects on the others. Only with the project-manager concept has it been found possible to keep this form of control in the hands of a sufficiently small group which can be counted on for rapidity of response.

Management Control Communications

Because of the short development time and the short elapsed time between launches, it is essential that almost an hour-by-hour status of the program be available to the Gemini Program Office at the Manned Spacecraft Center. To assist in making this possible, the Project Manager at McDonnell Aircraft Corp. and the Program Manager at the Manned Spacecraft Center are kept in close communication by means of the establishment of two identical control centers. At McDonnell Aircraft Corp. in St. Louis, the project group keeps detailed track of spacecraft manufacturing, assembly, test status, schedule, and cost, primarily based on the action of the three activity centers described in figure 9-5. A Gemini control room in which these results are under constant attention is in communication by a direct hot line to an identical room at the Manned Spacecraft Center. In addition to the phone communications there is a Datafax transmission link because much of the information cannot be readily transmitted verbally. With this form of communications link, the Manned Spacecraft Center has extremely up-to-date information of every facet of the Gemini operation under the contractor's direction, whether it be fiscal, engineering, manufacturing, developmental test, or subcontractor performance.

Spacecraft Assembly

The Gemini spacecraft uses titanium almost exclusively for the basic structure. One of the interesting manufacturing processes involves the spot, seam, and fusion welding of this material. Of particular interest is the weld line where the titanium sheets, ranging from 0.010 to 0.180 inch in thickness, are prepared for spot and seam welding. In preparing sheets of the 0.010-inch-gage titanium for spot welding, it was found necessary to overlap and then cut with a milling-type slitting saw to secure the parallelism required to gain the quality type welding needed. In addition, it was found necessary to supply an argon atmosphere right at the seam to prevent oxidation, and, by the use of these two devices, it was possible to perform this operation with the result that there has been no inflight structural problem throughout either the Mercury or the Gemini Program. Typical of the care taken to obtain this result is the assembly welding machine. Here the components are jig mounted and fed through the electrodes. To prevent spitting during this welding with the consequent burn-throughs, the weld fixtures are mounted on air pads, and air is provided to lift the fixtures a few thousandths of an inch off the ground surface plates over which they travel. This eliminates any possibility of a jerky or intermittent feeding of the work through the electrodes. There are many instances where welding is required in places not accessible with the welding machines. In these instances, fusion welding is employed, and the welds are made in a series of boxes as shown in figure 9-6. These boxes are made of Plexiglas. Argon is fed into the box to provide an inert gas atmosphere. The rubber gloves seen in the fig-

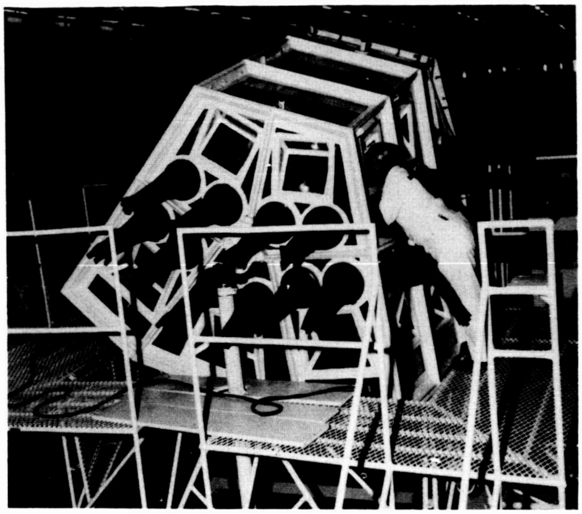


FIGURE 9-6.—Plexiglas welding boxes.

ure provide the access for the operator's arms, and the complete work is done within the transparent box. A variety of sizes and configurations is provided to permit the most efficient use of the device.

Installation and Checkout, White Room

The operational environment of a spacecraft is such that a life-support capability must be carried along in onboard systems. Perfection in functional operation of this equipment must be the goal. To comply with these requirements, extensive use is made of the white room facilities in the manufacture of wire harnesses, preparation of functional systems, manufacture of critical components, and conduct of spacecraft systems tests, including those conducted in the space simulation chamber. There is a twofold benefit in this form of operation: (1) the extreme attention focused on cleanliness in the manufacturing area, and (2) the increased awareness of the personnel engaged in the operation. An area equivalent to 54 000 square feet is utilized in the performance of the various operations on the spacecraft. Figure 9-7 shows a typical white room in the McDonnell Space Center. The white room is the major installation and test room for the Gemini spacecraft.

For individual systems of the spacecraft, engineering specifications have established different degrees of environmental cleanliness, and this has brought about the creation of three different classes of white rooms. This was done to make efficient use of facilities, to properly grade the requirements for air filtering and thermal and humidity control, and to establish personnel clothing and access standards in a practical manner. A few of the specifications established for our maximum cleanliness white room are as follows:

(1) The area shall be completely enclosed.

FIGURE 9-7.—White room at the McDonnell Space Center.

(2) The area shall be supplied with clean filtered air. The filters used in the circulating system shall be capable of removing 99.9 percent of all particles above 1 micron in size and 90 percent of all particles 0.3 to 1 micron in size.

(3) A positive pressure shall be maintained in this area at all times. Pressure in the maximum cleanliness area shall be higher than the pressure in adjacent areas.

(4) The area shall be maintained at a temperature of not over 75° F and a relative humidity of not over 55 percent.

(5) Vinyl floor coverings shall be used.

(6) The walls shall be painted with gloss white or a light pastel color enamel.

(7) Recessed or flush-mounted light fixtures shall be used.

This is typical of the type area provided for work on environmental control systems, and those components such as valves which may have extremely fine orifices.

Spacecraft Systems Tests Flow Plan

The environment of space is one demanding near perfection of operation of the equipment in the spacecraft. The spacecraft systems tests flow plan of figure 9–8 describes in sequence the actual tests performed on each of the spacecraft. The reactant supply system module in the adapter contains the tanks and valves sup-

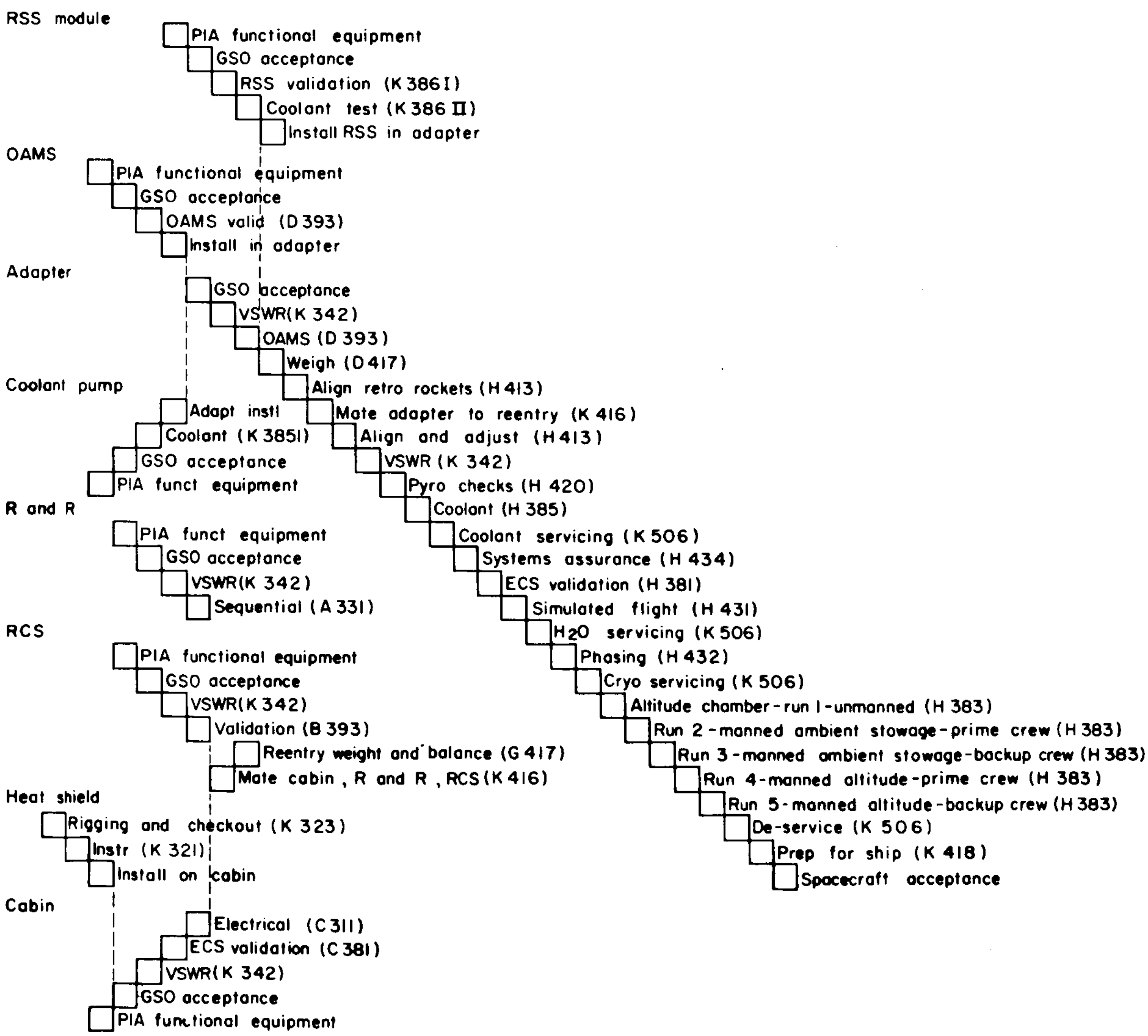


FIGURE 9–8.—Spacecraft systems tests flow plan.

plying the cryogenic oxygen and hydrogen to the fuel cells. The first step is to make a complete functional test of each individual component before assigning it to the spacecraft for installation into the module or section. Following this, the test data are reviewed by the contractor and the customer, and the equipment is then actually installed. When the submodule has completed buildup, it is then subjected to two systems-level tests, each defined by a detailed, documented test plan which has had engineering review and concurrence by the customer. Each section follows this pattern, with the number of tests obviously dependent upon the amount of equipment installed. Upon completion of the section-level tests, the spacecraft is erected into a vertical stand (fig. 9–7) and a complete end-to-end series of tests conducted in the order shown in figure 9–8. Here again each individual test is done in an extremely detailed manner, thoroughly documented and reviewed both by McDonnell Aircraft Corp. and NASA engineering and quality personnel before proceeding to the next step. All test discrepancies are submitted to a review board jointly manned by NASA and McDonnell Aircraft Corp. for evaluation and resolution. A complete log is maintained of all the test results on each spacecraft and forwarded to the launch site for ready reference during launch-site tests. Among the numerous tests shown on figure 9–8 is listed simulated flight. In this test the spacecraft, with the actual selected astronaut crew, is put into a flight condition functionally, and the equipment is operated in the manner planned for its mission from launch through landing. This test includes not only those functions which would occur in a completely successful flight, but also evaluates all emergency or abort capabilities as well. When the spacecraft has successfully passed this test, it is then prepared for a simulated flight test in the space simulation chamber, where altitude conditions are provided, and both the prime crew and the backup crew have an opportunity to go through the complete test.

Space Simulation Chamber

All of the components, modules, and even sections of the Gemini spacecraft were qualified under conditions simulating as closely as possible the space environment in which they must operate. As previously discussed, each complete Gemini spacecraft undergoes the final simulated flights at altitude. This capability has been made possible by the provision at McDonnell Aircraft Corp. of a sizable number and variety of space simulation chambers. These vary in size from 32 inches to 30 feet in diameter. The large altitude chamber (fig. 9–9), in which the complete spacecraft is put through manned simulated flight test, is 30 feet in diameter by 36 feet in length. It has the capability for emergency repressurization from vacuum to 5 psia in 18 seconds. This latter capability permits access through a special lock for conduct of emergency operations should such ever be required. The chamber also has numerous observation hatches.

Spacecraft Delivery

At the conclusion of the manned simulation run in the chamber, the spacecraft is delivered

FIGURE 9–9.—McDonnell altitude chamber.

via aircraft furnished through NASA direct to the Kennedy Space Center. Figure 9–10 shows

FIGURE 9–10.—Spacecraft being loaded into aircraft for shipment to Cape Kennedy.

the early stage of loading into the aircraft, and is typical of the manner in which all spacecraft have been delivered. The goal of delivering vehicles in as near to flight-ready condition as practical has been met for each of the seven production spacecraft shipped to the launch site.

Concluding Remarks

In this paper, only a selected few high points have been treated. Although it is equally impossible to list all the many contributors to the development of this program for NASA, McDonnell Aircraft Corp., and other Government agencies, the writer wishes to point out that teamwork was the key element in its accomplishment.

10. SPACECRAFT RELIABILITY AND QUALIFICATION

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Summary

The Gemini spacecraft reliability and qualification program was based on conventional concepts. However, these concepts were modified with unique features to obtain the reliability required for manned space flight, and to optimize the reliability and qualification effort.

Emphasis was placed on establishing high inherent reliability and low crew-hazard characteristics early in the design phases of the Gemini Program. Concurrently, an integrated ground-test program was formulated and implemented by the prime contractor and the major suppliers of flight hardware. All data derived from all tests were correlated and used to confirm the reliability attained.

Mission-success and crew-safety design goals were established contractually, and estimates were made for each of the Gemini missions without conducting classical reliability mean-time-to-failure testing.

Design reviews were conducted by reliability engineers skilled in the use of reliability analysis techniques. The reviews were conducted independently of the designers to insure unbiased evaluations of the design for reliability and crew safety, and were completed prior to specification approval and the release of production drawings.

An ambitious system to control quality was rigidly enforced to attain and maintain the reliability inherent in the spacecraft design.

A closed-loop failure-reporting and corrective-action system was adopted which required the analysis, determination of the cause, and corrective action for all failures, malfunctions, or anomalies.

The integrated ground-test program consisted of development, qualification, and reliability tests, and was conducted under rigid quality-control surveillance. This test program, coupled with two unmanned Gemini flights, qualified the spacecraft for manned flights.

Introduction

The level of reliability and crew safety attained in the Gemini spacecraft and demonstrated during the seven Gemini missions is the result of a concerted effort by contractor and customer engineers, technicians, and management personnel working together as one team within a management structure, which permitted an unrestricted exchange of information and promoted a rapid decisionmaking process.

Stringent numerical design goals for Gemini mission success and crew safety were placed on the spacecraft contractor, who incorporated these goals into each specification written for flight hardware. To meet this specification requirement, the suppliers had to give prime consideration to the selection, integration, and packaging of component parts into a reliable end item. Reliability analyses were required from the major equipment suppliers to assess the design for the inherent capability of meeting the established design goal.

The spacecraft contractor was required to integrate the subcontractor-supplied hardware, and to effect the necessary redundancy in the spacecraft to meet the overall reliability goal.

Examples of the spacecraft redundant features are:

(1) Every function in the pyrotechnic system incorporates a redundant feature.

(2) Two completely independent reentry-control propulsion systems are installed in the spacecraft.

(3) Redundant coolant subsystems are incorporated in the environmental control system.

(4) Duplicate horizon sensors are incorporated in the guidance system.

(5) Six fuel-cell stacks are incorporated in the electrical system, although only three are required for any long-duration mission.

Redundant systems or backup procedures were provided where a single failure could be catastrophic to the crew or the spacecraft.

Concurrent with design and development, an integrated ground-test program was established. Data from all tests were collected and analyzed to form a basis for declaring the Gemini spacecraft qualified for the various phases of the flight test program. The integrated ground-test program, shown in figure 10–1, shows the density of the test effort with respect to the production of the flight equipment.

Development tests were initially performed to prove the design concepts. Qualification tests were conducted to prove the flight-configuration design and manufacturing techniques. Tests were then extended beyond the specification requirements to establish reasonable design margins of safety. The unmanned flight tests were conducted to confirm the validity of design assumptions, and to develop confidence in spacecraft systems and launch-vehicle interfaces prior to manned flights.

Specific test-program reviews were held at the prime contractor's plant and at each major subcontractor's facility to preclude duplication of testing, and to insure that every participant in the Gemini Program was following the same basic guidelines.

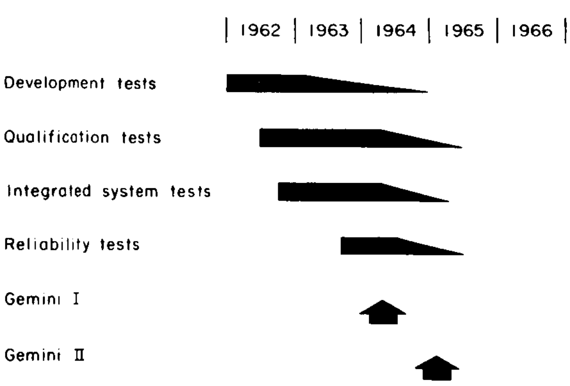

FIGURE 10–1.—Gemini test program.

Mission Success and Crew Safety

A numerical design goal was established to represent the probability of the spacecraft performing satisfactorily for the accomplishment of all primary mission objectives. The arbitrary value of 0.95, which recognizes a risk of failing to meet 1 primary objective out of 20 on each mission, was selected. The 0.95 mission-success design goal was included in the prime contract as a design goal rather than a firm requirement, which would have required demonstration by mean-time-to-failure testing. The prime contractor calculated numerical apportionments for each of the spacecraft systems and incorporated the apportioned values in major system and subsystem contractor requirements. Reliability estimates, derived primarily from component failure-rate data and made during the design phase, indicated that the design would support the established mission-success design goal. The reliability estimates, by major spacecraft system, for the Gemini III spacecraft, are shown in table 10–I.

Crew safety design goals were also established but for a much higher value of 0.995 for all missions. Crew safety is defined as having the flight crew survive all missions or all mission attempts.

Planned mission success, gross mission success, and crew safety estimates were also made prior to each manned mission, using the flight data and data generated by the integrated ground-test program; each program reflected assurance of conducting the mission successfully and safely.

A detailed failure mode and effect analysis was conducted on the complete spacecraft by the prime contractor and on each subsystem by the cognizant subcontractor, to investigate each failure mode and assess its effect on mission success and crew safety. The analysis included an evaluation of—

(1) Mode of failure.

(2) Failure effect on system operation.

(3) Failure effect on the mission.

(4) Indications of failure.

(5) Crew and ground action as a result of the failure.

(6) Probability of occurrence.

Corrective action was taken when it was determined that the failure mode would grossly

TABLE 10–I.—*Spacecraft 3 Reliability Estimates*

	Planned mission success [a]	Gross mission success [b]
Electrical power	0.999	0.999
Guidance and control:		
Propulsion	.952	.991
Orbital attitude and maneuver system	.9602	.9992
Reentry control system	.9919	.9919
Electronics	.967	.9998
Communications	.999	.999
Instrumentation	.999	.999
Environmental control	.989	.989
Landing	.985	.985
Sequentials, rockets, and pyros	.957	.988
Total	.856	.951

[a] Planned mission success is having the spacecraft function as necessary and perform the objectives of the mission as established in the mission directive.

[b] Gross mission success is inserting the spacecraft into orbit, having the capability of completing the prescribed orbital duration, and recovering the flight crew and spacecraft.

affect mission success or jeopardize the safety of the crew.

A single-point failure mode and effect analysis was conducted for all manned missions to isolate single failures which could prevent recovery of the spacecraft or a safe recovery of the crew. The single-point failure modes were evaluated, and action was taken to eliminate the single-point failure or to minimize the probability of occurrence.

Design Reviews

Critical reliability-design reviews were conducted as soon as the interim design was established. The reviews were conducted by reliability personnel independent of the designer and resulted in recommended changes to improve the reliability of the respective systems or subsystems. The reviews included the use of—

(1) Numerical analyses.
(2) Stress analyses.
(3) Analyses of failure modes.
(4) Tradeoff studies to evaluate the need for redundant features.

A typical design change is shown schematically in figure 10–2. This change was incorporated because the 2-day Gemini rendezvous flight requires four of the six fuel-cell stacks, three stacks to a section, to meet mission objectives. The failure of a single supply pressure regulator would have caused the loss of a fuel-cell section. Therefore, it was necessary that each of the two regulators which control the reactant supply be capable of supplying reactants to both fuel-cell sections. The crossover provided this capability. Figure 10–3 shows the electrical power system reliability slightly increased for the 2-week mission. The reliability was increased from 0.988 to 0.993 for an assumed failure rate of 10^{-4} failures per

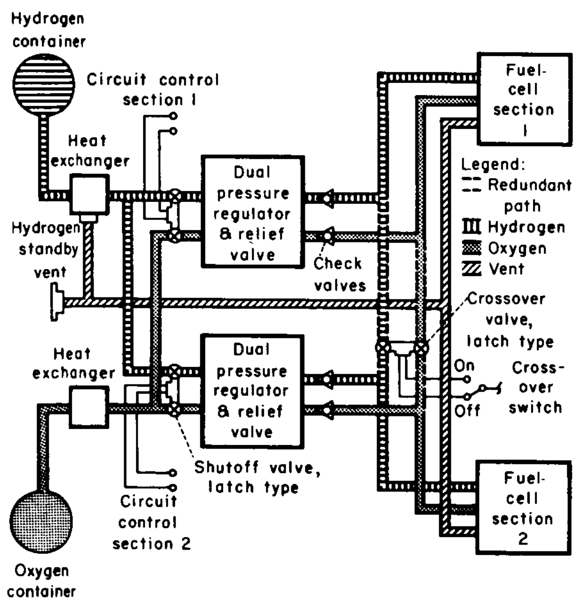

FIGURE 10–2.—Fuel-cell reactant supply system.

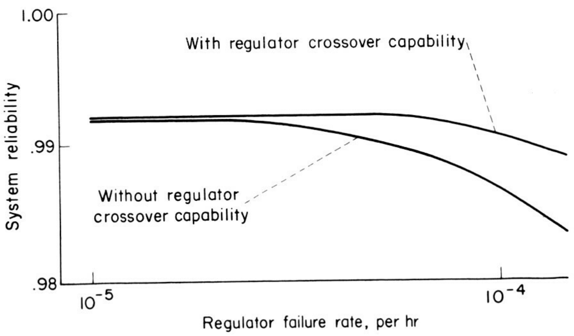

FIGURE 10–3.—Fuel-cell power system reliability for a 2-week mission.

hour. Figure 10–4 shows the reliability greatly increased for the 2-day mission.

It cannot be overemphasized that reliability is an inherent characteristic and must be realized as a result of design and development. Inherent reliability cannot be inspected or tested into an item during production; at best, that which is inherent can only be attained or maintained through a rigid quality control. These reliability design reviews and the numerical analyses were conducted as early as November 1962, prior to the fabrication of the first production prototypes.

Development Tests

Development tests using engineering models were conducted to establish the feasibility of design concepts. These tests explored various designs and demonstrated functional performance and structural integrity prior to committing production hardware to formal qualification tests. In some cases, environmental tests were conducted on these units to obtain information prior to the formal qualification.

Integrated System Tests

Integrated system tests were conducted during progressive stages of the development to demonstrate the compatibility of system interfaces. Such systems as the inertial guidance system, the propulsion system, and the environmental control system were especially subjected to such tests. Early prototype modules were used in static articles or mockups, which represented complete or partial vehicles. They served to acquaint operating personnel with the equipment and to isolate problems involving

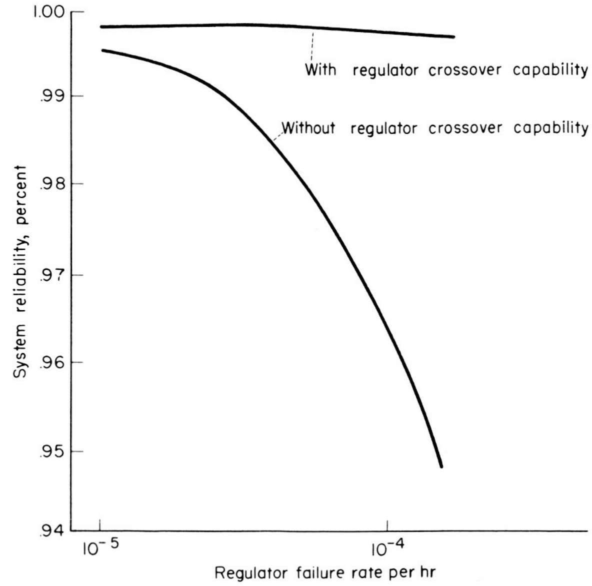

FIGURE 10–4.—Fuel-cell power system reliability for a 2-day mission.

electrical-electronic interface, radiofrequency interference, and system-design compatibility.

When production prototype systems became available, a complete spacecraft compatibility test unit was assembled at the prime contractor's facility (fig. 10–5). During these tests, system integration was accomplished by end-to-end test methods. These tests permitted the resolution of problems involving mechanical interface, electrical-electronic interface, radiofrequency interference, spacecraft compatibility, final-test-procedures compatibility, and compatibility with aerospace ground equipment (AGE), prior to assembly and checkout of the first flight vehicle.

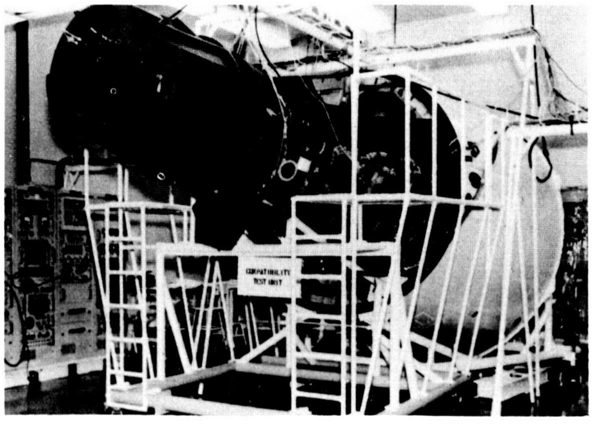

FIGURE 10–5.—Gemini compatibility test unit.

One of the more significant integrated systems tests was the thermal qualification or the spacecraft thermal-balance test. This was conducted on a complete production spacecraft (fig. 10–6). Tests were conducted in a cold-wall altitude chamber that simulated altitude and orbital heating characteristics with the spacecraft powered up.

The test results demonstrated the need for heating devices on the propulsion system oxidizer lines, on thrust-chamber assembly valves, and on water lines to prevent freezing conditions during the long-duration mission.

System Qualification Test

Each item of spacecraft equipment was qualified prior to the mission on which the item was to be flown. The equipment was considered qualified when sufficient tests had been successfully conducted to demonstrate that a production unit, produced by production personnel and with production tooling, complied with the design requirements. These tests included at least one simulation of a long-duration flight or one rendezvous mission, or both, if necessary, with the system operating to its expected duty cycle.

Qualification requirements were established and incorporated in all spacecraft equipment specifications. The specifications imposed varied requirements on equipment, depending on the location of the equipment in the spacecraft, the function to be performed by the equipment, and the packaging of the equipment.

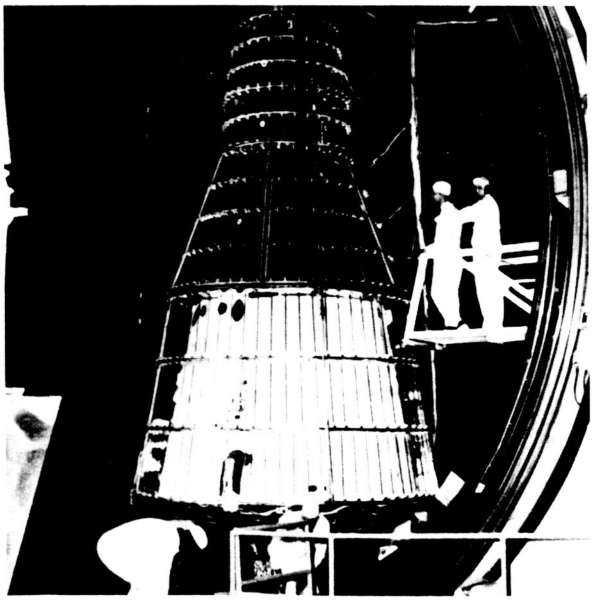

FIGURE 10–6.—Gemini spacecraft 3A preparation for thermal qualification test No. 1.

The environmental levels to which the equipment was subjected were based on anticipated preflight, flight, and postflight conditions. However, the environmental levels were revised whenever actual test or flight experience revealed that the original anticipated levels were unrealistic. This is exemplified by—

(1) The anticipated launch vibration requirement for the spacecraft was based on data accumulated on Mercury-Atlas flights. The upper two-sigma limit of this data required a power spectral density profile of approximately 12g rms random vibration. This level was revised because the Gemini I flight demonstrated that the actual flight levels were less than expected. The new data permitted the power spectral density to be changed, and by using the upper three-sigma limits the requirement was reduced to approximately 7g rms random vibration in the spacecraft adapter and to 8.8g rms random vibration in the reentry assembly.

(2) An aneroid device used in the personnel parachute was expected to experience a relatively severe humidity; therefore, the qualification test plan required the aneroid device to pass a 10-day 95-percent relative humidity test. The original design of the aneroid could not survive this requirement and was in the process of being redesigned when the Gemini IV mission revealed that the actual humidity in the spacecraft cabin was considerably lower than expected. The requirement was reduced to an 85-percent relative humidity, and the new aneroid device successfully completed qualification.

(3) The tank bladders of the propulsion system did not pass the original qualification slosh tests. Analysis of the failures concluded that the slosh tests conducted at one-g were overly severe relative to actual slosh conditions in a zero-g environment. The slosh test was changed to simulate zero-g conditions more accurately, and the slosh rate was reduced to a realistic value. The tests were then successfully repeated under the revised test conditions.

The development and timely execution of a realistic qualification program can be attributed, in part, to a vigorous effort by Government and contractor personnel conducting test-program reviews at the major subcontractor plants during the initial qualification phase of the program. The objective of the reviews was to aline the respective system test program to conform to an integrated test philosophy. The original test reviews were followed with periodic status reviews to assure that the test programs were modified to reflect the latest program requirements and to assure the timely completion of all testing which represented constraints for the various missions.

The qualification test environments required for Gemini equipment are shown on table 10–II. This chart, which was extracted from the spacecraft qualification status report, shows the qualification status of the digital command system and provides a typical example of a supplier's qualification test requirements. All environmental requirements are not applicable, since the digital command system is located in the adapter and will not experience such environments as oxygen atmosphere and salt-water immersion. Those environments which were required are noted with a "C" or "S" in the appropriate column. The "C" designates that the equipment has successfully completed the test, and the "S" designates that the equipment has been qualified by similarity. A component or assembly is considered qualified by similarity when it can be determined by a detailed engineering analysis that design changes have not adversely affected the qualification of the item.

Reliability Testing

For programs such as Gemini, which involve small production quantities, the inherent reliability must be established early in the design phase and realized through a strict quality control system. It was not feasible to conduct classical reliability tests to demonstrate equipment reliability to a significant statistical level of confidence. Consequently, no mean-time-to-failure testing was conducted. Confidence in Gemini hardware was established by analyzing the results of all test data derived from the integrated ground and flight test program, and by conducting additional reliability tests on selected components and systems whose functions were considered critical to successful mission accomplishment.

Equipment was selected for reliability tests after evaluating the more probable failure modes. The tests were designed to confirm the design margins or to reveal marginal design characteristics, and they included exposure to environmental extremes such as—

(1) Temperature and vibration beyond the design envelope.

(2) Applied voltage or pressure beyond the normal mission condition.

(3) Combined environments to produce more severe equipment stress.

(4) Endurance beyond the normal mission duty cycles.

The reliability tests conducted on the digital command system are shown in table 10–III. These tests overstressed the digital command system in acceleration, vibration, voltage, and combinations of altitude, temperature, voltage, and time. These overstress tests confirm an adequate design margin inherent in the digital command system.

Typical reliability tests on other systems and components included such environments as proof pressure cycling, repeated simulated missions, and system operation with induced contamination. The contamination test was conducted on the reentry control system and the orbital attitude and maneuver system because these systems were designed with filters and pressure regulators which contained small orifices susceptible to clogging.

Some reliability tests were eliminated when Gemini flight data revealed that in some instances qualification tests had actually been overstress tests. This was particularly true with respect to vibration qualification, where the overall rms acceleration level of 12.6g (fig. 10–7) exceeded the actual inflight vibra-

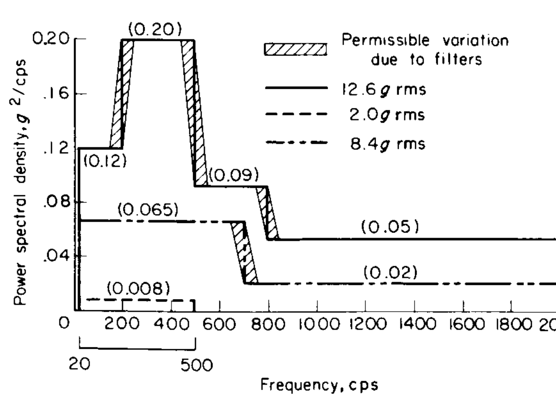

FIGURE 10–7.—Spacecraft random vibration test.

TABLE 10–II.—*Typical Test Sheet for Digital Computer Components*

NASA—MANNED SPACECRAFT CENTER
QUALIFICATION STATUS REPORT
GEMINI

Page no. — of —

System title Communications and Signaling

Subsystem title Digital Command System

Part number (vendor no.)	Part name	Current status (X, F, Q)	Spacecraft effectivity	A Hi temp.	B Lo temp.	C Vib.	D Shock	E Temp. Alt.	F Accel.	G RFI	H Humid (Opr)	I Humid (N Op)	J Acoustic N	K Exp.	L Rain	M O_2 Atmos.	N Low Pres.	O Pres. Drop	P Hi Pres.	Q Fungus	R Salt Spray	S S.W. Imm.	T Struct.	U Exp. Decom.	V Temp. Cont.	W Life	X Proof Pr.	Y Ht. Trans.	Z	Planned start	Actual start	Planned completion	Complete	Similarity to part no.
52-85714-15	Receiver decoder	....	2	C	C	C	C	C	C	C	—	C	C	C	C	—	—	—	—	C	C	—	—	—	—	—	—	—	C	....	....	....	10/63	
52-85714-17	8-unit relay	....	2, 3	C	C	C	C	C	C	C	—	C	C	C	C	—	—	—	—	C	C	—	—	—	—	—	—	—	C	....	....	....	10/63	
52-85714-27	Receiver decoder	....	3 to 13	S	S	S	S	S	S	S	—	S	S	S	S	—	—	—	—	S	S	—	—	—	—	—	—	—	S	....	....	....	....	52-85714-15
52-85714-21	8-unit relay	....	3A, 4 to 13	S	S	S	S	S	S	S	—	S	S	S	S	—	—	—	—	S	S	—	—	—	—	—	—	—	S	....	....	....	....	52-85714-17

Environments required and status: columns A–Z. Test schedule: Planned start, Actual start, Planned completion, Complete.

LEGEND

*=Environment required.
—=Environment not required.
C=Completed.
S=Similarity.
△=Part failure with amplifying note indicated by number.
X=Operates but has not started qualification.
F=Functionally qualified with major environment's complete.
Q=Fully qualified.

TABLE 10-III.—*Digital Command System Reliability Tests*

Environments	Qualification tests	Overstress tests
Acceleration	7.2g in 326 sec	9.0g in 326 sec
Random vibration	Overall rms acceleration level of 12.6g for 15 min per axis	Overall rms acceleration level of 15.6g for 3 min per axis
Combined altitude, high temperature, high voltage	No combined-environment qualification tests required	Pressure, 1.7×10^{-6} psia Temperature, 200° F Voltage, 36 V dc
Combined low temperature, low voltage	No combined-environment qualification tests required	Temperature, −60° F Voltage, 17 V dc
Applied high voltage	30.5 to 33.0 V dc	36 V dc
Applied low voltage	18.0 to 20.0 V dc	17 V dc

tion levels by a significant margin. Consequently, the test level was reduced to an overall rms acceleration level of 7g for the adapter blast shield region and to 8.8g in the reentry assembly region (figs. 10-8 and 10-9), respectively. Equipment which had been subjected to the initial requirement, therefore, did not require additional testing.

All failures which occurred during the reliability tests were analyzed to determine the cause of failure and the required corrective action. Decisions to redesign, retest, or change processes in manufacturing were rendered after careful consideration of the probability of occurrence, mission performance impact, schedule, and cost.

For the most part, the reliability tests were conducted as a continuation of the formal qualification tests on the same test specimens used in the qualification tests after appropriate refurbishment and acceptance testing. When the previous testing expended the test specimen to a state that precluded refurbishment, additional new test units were used.

Quality Control

A rigid quality control system was developed and implemented to attain and maintain the reliability that was inherent in the spacecraft design. This system required flight equipment to be produced as nearly as possible to the qualified configuration.

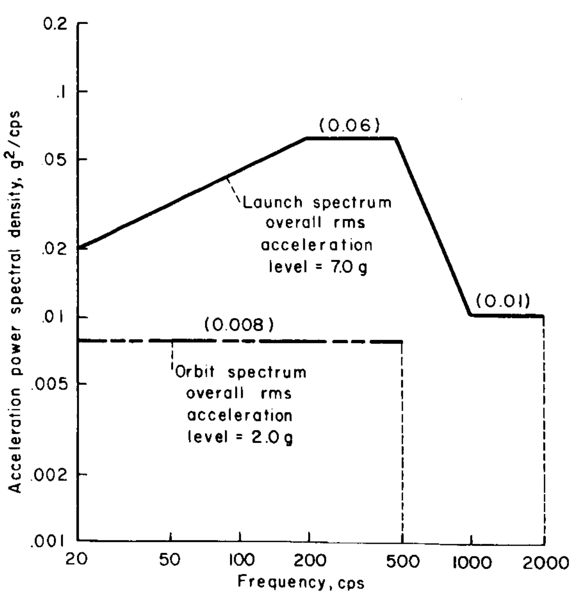

FIGURE 10-8.—Random vibration of test adapter blast-shield region.

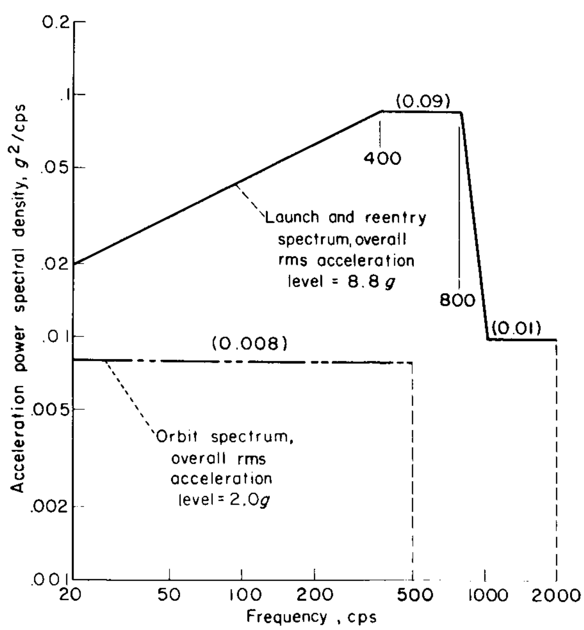

FIGURE 10-9.—Random vibration test of reentry assembly region.

The unique features of the quality control system which contributed to the success of the Gemini flight program are:

(1) Configuration control.
(2) Material control.
(3) Quality workmanship.
(4) Rigid inspection.
(5) Spacecraft acceptance criteria.

Configuration control is necessary to maintain spacecraft quality; therefore, the contractor and customer management developed and implemented a rigid and rapid change-control system which permitted required changes to be documented, approved, implemented, and verified by quality control, with the inspector being fully aware of the change before it is implemented on the spacecraft. When a change is considered necessary, and the program impact has been evaluated for design value, schedule, and cost, the proposed change is formally presented to the management change board for approval and implementation. All changes made to the spacecraft are processed through the change board.

Each article of flight equipment is identified by a unique part number. Components, such as relay panels, tank assemblies, and higher orders of electrical or electronic assemblies, are serialized, and each serialized component is accounted and recorded in the spacecraft inventory at the time it is installed in the spacecraft.

Exotic materials such as titanium, René 41, and explosive materials used in pyrotechnics are accounted for by lots to permit identification of any suspect assembly when it is determined that a part is defective because of material deficiency.

Inspection personnel and fabrication technicians who require a particular skill such as soldering, welding, and brazing are trained and certified for the respective skill and retested for proficiency at regular intervals to retain quality workmanship.

The very strict control of parts and fabricated assemblies is maintained by rigid inspection methods. All deficiencies, discrepancies, or test anomalies are recorded and resolved regardless of the significance that is apparent to the inspector at the time of occurrence. All equipment installations and removals require an inspection "buy-off" prior to making or breaking any system interfaces.

Formal spacecraft acceptance reviews are conducted at strategic stages of the spacecraft assembly and test profile. The reviews are conducted with both the customer and the contractor reviewing all test data and inspection records to isolate any condition which occurred during the preceding manufacturing and test activity and may adversely affect the performance of the equipment.

All failures, malfunctions, or out-of-tolerance conditions that have not been resolved are brought to the attention of the management review board for resolution and corrective measures. The reviews are conducted prior to final spacecraft system tests at the contractor's plant, immediately prior to spacecraft delivery, and approximately 10 days preceding the flight.

Flight Equipment Tests

A series of tests are conducted on all flight articles to provide assurance that the reliability potential of the design has not been degraded in the fabrication and handling of the hardware. The tests conducted on flight equipment include—

(1) Receiving inspection.
(2) In-line production tests.
(3) Predelivery acceptance tests (PDA).
(4) Preinstallation acceptance tests (PIA).
(5) Combined spacecraft systems tests (SST).
(6) Spacecraft-launch vehicle joint combined system tests.
(7) Countdown.

In receiving inspection, critical parts are given a 100-percent inspection which may include X-ray, chemical analysis, spectrographs, and functional tests.

While the equipment is being assembled, additional tests are performed to detect deficiencies early in manufacturing. Mandatory inspection points are established at strategic intervals during the production process. These were established at such points as prior to potting for potted modules and prior to closure for hermetically sealed packages. As an example, certain electronic modules of the onboard computer receive as many as 11 functional tests before they go into the final acceptance test.

A predelivery acceptance test to verify the functional performance of the equipment is performed at the vendor's plant in the presence of vendor and Government quality control representatives. Many of these tests include environmental exposure to vibration and low temperature whenever these environments are considered to be prime contributors to the mechanics of failure. For complex or critical equipment, spacecraft contractor engineering and quality control and Government engineering representatives were also present to witness the test for initial deliveries.

Prior to installation in the spacecraft, the unit is given a preinstallation acceptance test to verify that the functional characteristics or calibration has not changed during shipment. This test is conducted identically to the predelivery acceptance test when feasible, unless a difference in test equipment necessitates a change. When differences in test equipment dictate a difference in the testing procedure, the test media (such as fluids, applied voltages, and pressures) are identical, and test data are recorded in the same units of measure in order to compare test results with previous test data. This permits a rapid detection of the slightest change in the performance of the equipment.

Spacecraft systems tests are performed on the system after installation in the spacecraft, prior to delivery. They include individual systems tests prior to mating the spacecraft sections, integrated systems tests, simulated flight tests, and altitude chamber tests after mating all of the spacecraft sections. These tests use special connectors built into the equipment to prevent equipment disconnection which would invalidate system interfaces.

Similar systems tests are repeated during spacecraft premate verification at the launch-site checkout facility. After the spacecraft has been electrically connected to the launch vehicle, a series of integrated systems functional tests is performed. Upon completion of these tests, simulated flights, which exercise the abort mode sequences, are conducted in combination with the launch vehicle, the Mission Control Center, the Manned Space Flight Network, and the flight crew.

The countdown is the last in a series of systems functional tests to verify that the spacecraft is ready for flight. It should be pointed out again that any abnormality, out-of-tolerance condition, malfunction, or failure resulting from any of these tests is recorded, reported, and evaluated to determine the cause and the effect on mission performance.

Failure Reporting, Failure Analysis, and Corrective Action

Degradation in the inherent reliability of the spacecraft systems is minimized through the rigid quality control system and a closed-loop failure-reporting and corrective-action system. All failures of flight-configured equipment that occur during and after acceptance tests must be reported and analyzed. No failure, malfunction, or anomaly is considered to be a random failure. All possible effort is expended to determine the cause of the anomaly to permit immediate corrective action.

Comprehensive failure-analysis laboratories were established at the Kennedy Space Center and at the spacecraft contractor's plant to provide rapid response concerning failures or malfunctions which occur immediately prior to spacecraft delivery or launch.

However, in cases where the electronic or electromechanical equipment is extremely complex, the failed part is usually returned to the vendor when the failure analysis requires special engineering knowledge, technical skills, and sophisticated test equipment.

A tabulated, narrative summary of all failures which occur on the spacecraft and spacecraft equipment is kept current by the prime contractor. This list is continuously reviewed by the customer and the contractor to assure acceptable and timely failure analyses and resulting corrective action. The contractor has established a priority system to expedite those failure analyses which are most significant to the pending missions.

A simplified flow diagram of the corrective action system is shown in figure 10–10. A material review board determines the disposition of the failed equipment, and an analysis of the failure may be conducted at either the supplier's plant, the prime contractor's plant, or at the Kennedy Space Center, depending on the nature of the condition, the construction of the equipment, and the availability of the facilities

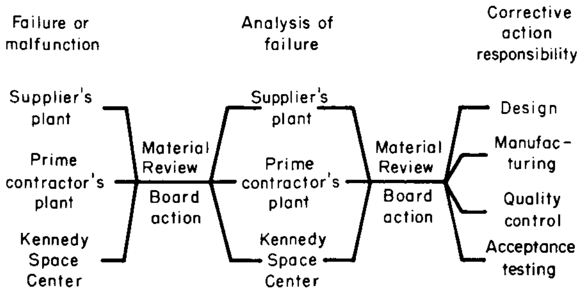

FIGURE 10-10.—Gemini corrective action flow schematic.

at each of the respective locations. If the analysis of a supplier's equipment is conducted at the prime contractor's plant or at the Kennedy Space Center, the respective supplier's representative is expected to participate in the analysis.

When the failure-analysis report is available, the recommended corrective action is evaluated, and a decision is rendered to implement the required corrective action. This may require management change board action to correct a design deficiency, a change in manufacturing processes, establishment of new quality control techniques, and/or changes to the acceptance-testing criteria. Each change must also be evaluated to determine whether qualification status of the equipment has been affected. If the equipment cannot be considered qualified by similarity, additional environmental tests are conducted to confirm the qualification status.

Unmanned Flight Tests

The final tests conducted to support the manned missions were the unmanned flights of Gemini I and II. Gemini I verified the structural intergrity of the spacecraft and demonstrated compatibility with the launch vehicle. Gemini II, a suborbital flight, consisted of a production spacecraft with all appropriate onboard systems operating during prelaunch, launch, reentry, postflight, and recovery. Each system was monitored by special telemetry and cameras that photographed the crew-station instrument panels throughout the flight. The flight demonstrated the capability of the heat-protection devices to withstand the maximum heating rate and temperature of reentry. The successful completion of the Gemini II mission, combined with ground qualification test results, formed the basis for declaring the spacecraft qualified for manned space flight.

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B

LAUNCH VEHICLE

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11. LAUNCH VEHICLE MANAGEMENT

By WILLIS B. MITCHELL, *Manager, Office of Vehicles and Missions, Gemini Program Office, NASA Manned Spacecraft Center; and* JEROME B. HAMMACK, *Deputy Manager, Office of Vehicles and Missions, Gemini Program Office, NASA Manned Spacecraft Center*

Summary

The management of the Gemini launch vehicle program has been characterized by a successful blending of the management philosophies of the NASA Gemini Program Office and the Air Force Space Systems Division. The management activity discussed in this paper represents those measures taken to achieve this degree of cooperation in order to maintain cognizance of the progress of the launch vehicle program, and to provide the necessary integration between the launch vehicle development activity and the rest of the Gemini Program.

Introduction

A modified version of the Air Force Titan II was selected as the launch vehicle for the Gemini flights early in the proposal stage of the Gemini Program, in the fall of 1961. The selection was based on the payload capability of the Titan II and on the fact that it promised to be an inherently reliable vehicle because of the use of hypergolic propellants and the simplified mechanical and electrical systems. Although the selection was made before the completion of the Titan II development program and a number of months before the first flight, this early technical evaluation was accurate.

The selection early in the Titan II development phase also offered the opportunity to flight-test some of the changes which were desirable to rate the vehicle for manned flight. The purpose of the changes was to enhance further the basic reliability of the vehicle through the use of redundant systems. Modifications were made in the flight control and electrical systems. A malfunction detection system was incorporated to give the crew sufficient information to diagnose impending problems and to determine the proper action. Details of the modifications will be covered in subsequent papers.

The Gemini launch vehicle was, therefore, composed of the basic Titan II plus the changes discussed in the preceding paragraph. In January 1962, a purchase request was issued to the Space Systems Division of the Air Force Systems Command for the development and procurement of a sufficient number of these vehicles to satisfy the needs of the Gemini Program.

Management Organization

The basic document underlying the relationship between the Air Force and the NASA in the management of the Gemini Program is the "Operational and Management Plan for the Gemini Program," often referred to as the NASA–DOD agreement. This document was prepared in the fall of 1961 and agreed to by appropriate representatives of the NASA and of the Department of Defense (DOD) in December 1961. The document delineates the responsibilities and the division of effort required for the conduct of the Gemini Program. In general terms, the agreement assigns to the Air Force the responsibility for development and procurement of the launch vehicle and launch complex, and for technical supervision of the launch operations under the overall management and direction of the NASA Gemini Program Manager.

The management of the integration of the launch vehicle development program into the overall Gemini system is a function of the NASA Gemini Program Office organization. Within the Gemini Program Office, the monitoring of the technical development of the launch vehicle is, primarily, the responsibility of the Office of Vehicles and Missions. This office serves as the major point of contact with the

Air Force management office and is responsible for the launch vehicle coordination and integration activities within the Manned Spacecraft Center. The Test Operations Office in the Gemini Program Office has the responsibility for the integration of the launch vehicle into the overall plan for preflight checkout, countdown, and launch of the combined Gemini space vehicle. In order to accomplish these tasks, the Test Operations Office works closely with Kennedy Space Center organizations and with the Gemini Program Office Resident Manager at the Kennedy Space Center.

The magnitude of the management task is illustrated in figure 11–1, which shows the contractor and Government organizations involved in the launch vehicle effort. For completeness, the Manned Spacecraft Center organizations which are directly concerned are also shown. The figure shows that 2 major Government agencies, 5 major industrial contractors, and 43 industrial subcontractors participate in the Gemini launch vehicle development program. The major Government agencies involved in the program are the two NASA centers (the Kennedy Space Center and the Manned Spacecraft Center) and the Air Force Systems Command (AFSC). Within the Air Force, the Gemini launch vehicle program is managed through the Space Systems Division Program Office, which is supported strongly by the Aerospace Corp. The Aerospace Corp. is responsible to the Space Systems Division Program Office for systems integration and technical direction on the overall Gemini launch vehicle program. The Aerospace Corp. also supplies the launch-vehicle guidance equations and predicted payload capabilities, and performs the postflight evaluation.

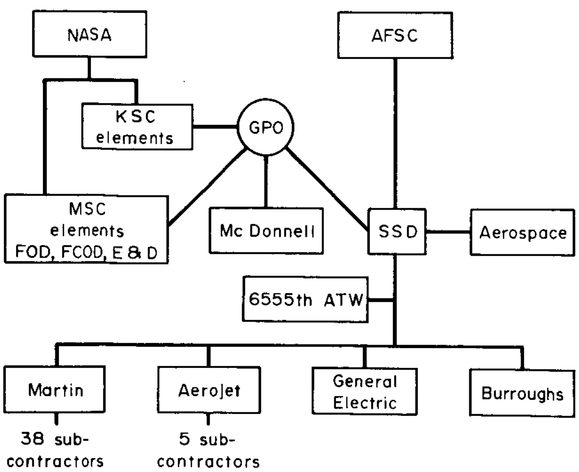


FIGURE 11–1.—Management structure (Gemini launch vehicle).

The airframe contractor is the Martin Co., with 38 major subcontractors. The Aerojet-General Corp. and its five subcontractors supply the engine system. The General Electric Co. produces the airborne guidance system components, and the Burroughs Co. supplies the ground computer and implements the guidance equations. The Air Force 6555th Aerospace Test Wing at Patrick Air Force Base, Fla., has been assigned the responsibility for preflight checkout of the launch vehicle at Cape Kennedy and for the launch operations. In the NASA organization, this responsibility is supported by the Kennedy Space Center and by a Gemini Program Office Resident Manager assigned from the Manned Spacecraft Center.

Within the Manned Spacecraft Center, organizations other than the Gemini Program Office involved in the program are the Flight Operations Directorate, which is responsible for operational mission planning and for the overall direction and management of flight control and recovery activities; the Flight Crew Operations Directorate, which is responsible for the flight crew training and crew inputs to the launch vehicle systems; and the Engineering and Development Directorate, which is responsible for additional technical support as required for the Gemini Program. The spacecraft contractor, the McDonnel Aircraft Corp., is also shown on the figure because interface relationships are maintained with this contractor, especially in the areas of the malfunction detection system and backup guidance.

Management Coordination Group

Obviously, with such a large, diverse, and far-flung group of organizations participating in the program, the two major management problems are (1) adequate and timely communications and (2) proper control and coordination of the activities of the separate participants. These problems occur in identifying and resolving the difficulties which arise in the various elements of the program hardware and in determining the ramifications of these solutions on all interfacing hardware and procedures.

Communication and control are also problems in the identification and transmittal of interface requirements among the groups involved. The interfaces are not only physical but many times are philosophical or ideological in nature.

When these management problems were further considered in the light of the relatively short time allowed for development and procurement of the launch vehicle, both the NASA and the Air Force recognized early in the Gemini Program that a system of cooperative program direction and problem reporting would be beneficial. Time simply was not available for the conventional chain-of-command operation. Consequently, a launch vehicle coordinating organization was formed, headed by a Chairman from the NASA Gemini Program Office and an Associate Chairman from the Space Systems Division Program Office. The group is composed of representatives of all the Government and industrial organizations which participate directly in the launch vehicle program, plus representatives of all Government or industrial groups which have an interface with the launch vehicle program. The organization of this group went through a number of changes and eventually arrived at the form shown in figure 11-2. This panel-type organization has the advantage of grouping people of like specialties, and it results in smaller discussion groups which allow more detailed treatment of problems. A normal coordination meeting lasts 2 days, the first of which is devoted to panel meetings. On the second day, reports from the panel chairmen are presented to the assembled committee, and recommendations for courses of action are proposed. This is followed by a Government session devoted to discussions of action items and financial matters. Meetings were originally held at intervals of 2 weeks, later increased to 3 weeks, and then monthly. Presently, one meeting is held before each mission. The present frequency of meetings indicates the maturity of the program. The key results of the meetings are translated into action items which are put into a telegram format. After coordination with responsible groups within the NASA Gemini Program Office, the action items are approved by the NASA Gemini Program Manager and are implemented. Other study items and records of discussions are put into abstract form and mailed to responsible agencies and participants.

In operation, the coordination group provides the status monitoring required to properly assess the progress of the launch vehicle program. It also makes possible the rapid identification of problem areas in hardware development, and, more importantly, it allows the talents of a large group of knowledgeable people to be brought to bear on these problems. The effects of proposed solutions on other facets of the total program are evaluated quickly, and knowledge of changes is disseminated rapidly. While a detailed discussion of the function of each of the

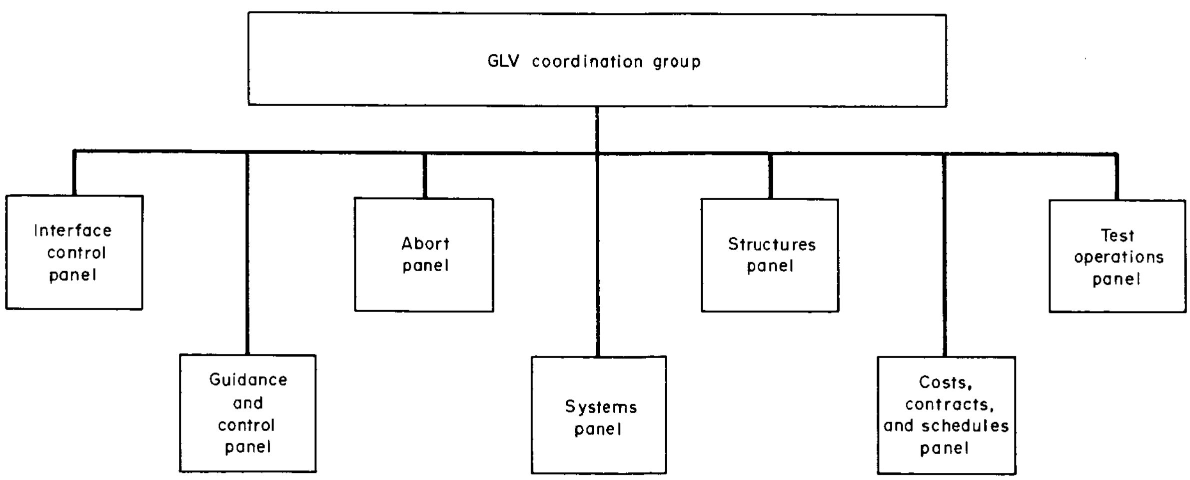


FIGURE 11-2.—Gemini launch-vehicle coordination group and reporting panels.

panels is not appropriate, the implications of the work of three of the groups is important because of their interrelation with the other elements of the Gemini Program:

(1) The interface control panel brings together the appropriate members of the industrial contractors representing the Gemini launch vehicle and the spacecraft for the interchange of information and requirements. The actions of this panel led to the preparation of the interface specification and the interface drawings. These drawings were the joint product of the two engineering departments and are indicative of the cooperation which was achieved.

(2) The abort panel outlines the required studies of the flight-abort environment, makes hazard analyses, and recommends abort procedures. Test programs to define the magnitude and extent of a launch-vehicle fireball were conducted under the surveillance of the abort panel. These activities were the basis of the crew-escape procedures.

(3) The guidance and control panel is concerned with the airborne and ground-based guidance equipment, as well as the interfacing requirements of the launch vehicle flight-control equipment with the redundant spacecraft inertial-guidance-system equipment. This panel is concerned with both hardware and software requirements.

A coordination activity at the Air Force Eastern Test Range has also proved to be a useful tool. This group, the Gemini Launch Operations Committee, brings together all elements that participate in the Gemini Program at the Air Force Eastern Test Range. The main purpose of this group is to resolve all launch-complex-oriented problems and, where necessary, to submit action requests back through the NASA Gemini Program Office.

Configuration Management

The NASA–DOD agreement provides to the NASA the authority to establish a configuration management system for the launch-vehicle program. This includes the establishment of a reference configuration, a configuration control board, and a change-status accounting system. Although an overall Gemini Program Configuration Control Board exists, the NASA Gemini Program Manager chose to delegate the detail authority for launch vehicle change control to the Air Force Configuration Change Board, which is operated by the Space Systems Division in accordance with Air Force Manual AFSCM–375–1. This manual specifies the configuration management system for Department of Defense programs during the definition and development phases. To provide the necessary integration of launch vehicle changes into the general program development plan, a member of the NASA Gemini Program Office has been appointed to sit with the Air Force Configuration Change Board as an associate member. It is his function to provide the liaison between the two boards. Generally, all Gemini launch vehicle changes are well coordinated with the NASA through the coordination group; consequently, the primary action of the NASA Change Board, concerning Gemini launch vehicle changes, is to review the key actions of the Air Force Change Board and to act on those changes referred to the NASA Change Board. This latter group of changes are those specifically requested by the NASA, those which affect the interface with the spacecraft or affect pilot safety, and those which materially affect launch schedules or funding.

Concluding Remarks

It is axiomatic that no organization will function well, no matter how carefully devised are the organization charts nor how well documented are the authorities and responsibilities, unless it is manned with well-motivated and dedicated people who work cooperatively toward the objective. On the Gemini launch vehicle program, a spirit of cooperation has been developed between the two Government agencies involved that has extended throughout the contractor structure and has generally surmounted any differences that arose. This cooperation and excellent communication, together with the competence of the Air Force Space Systems Division and its associated contractors, is the key to the successful Gemini launch vehicle program.

12. GEMINI LAUNCH VEHICLE DEVELOPMENT

By WALTER D. SMITH, *Program Director, Gemini Program, Martin-Marietta Corp.*

Summary

This paper presents a brief description of the basic modifications made to the Titan II to adapt it to a Gemini launch vehicle (GLV), the ground rules under which they were made, how the principal systems were initially baselined, how they evolved, and how they have performed to date.

Introduction

An original concept of the GLV program was to make use of flight-proven hardware; specifically, the modified Titan II would be used to insure a high level of crew safety and reliability. This decision was based on the fact that more than 30 Titan II vehicles were scheduled to be flown prior to the flight of the first GLV, and, as a result of these flights, a high level of confidence would be established in the hardware unchanged for the GLV.

Modifications Required To Adapt the Titan II to a Gemini Launch Vehicle

The fundamental modifications made to the Titan II (fig. 12-1) to adapt it for use as the GLV were—

(1) The Titan II inertial guidance system was replaced with a radio guidance system.

(2) Provision was made for a redundant flight-control and guidance system which can be automatically or manually commanded to take over and safely complete the entire launch phase in the event of a primary system failure. This system addition was required because of the extremely short time available for the crew to command abort and escape, in the event of critical flight-control failures during the high-dynamic-pressure region of stage I flight. This redundant system was added primarily to insure crew safety in case of a critical malfunction; however, it also significantly increases the probability of overall mission success.

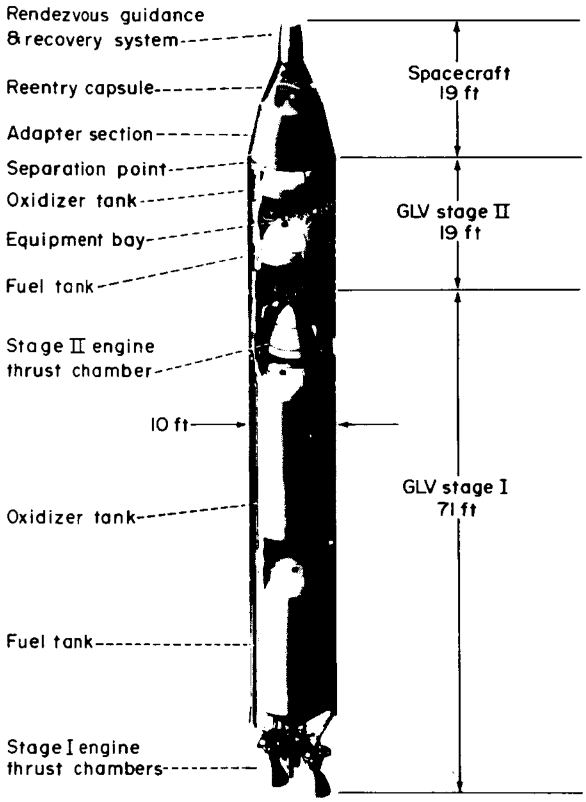


FIGURE 12-1.—Gemini launch vehicle.

(3) A malfunction detection system (fig. 12-2), designed to sense critical failure conditions in the launch vehicle, was included. The action initiated by the malfunction detection system, in the case of flight-control or guidance failures, is a command to switch over to the secondary flight-control and guidance system. For other failures, appropriate displays are presented to the crew.

(4) Redundancy was added in the electrical system to the point of having two completely independent power buses provided to critical components, and redundancy for all inflight sequencing.

(5) The Titan II retrorockets and vernier rockets were eliminated because no requirement

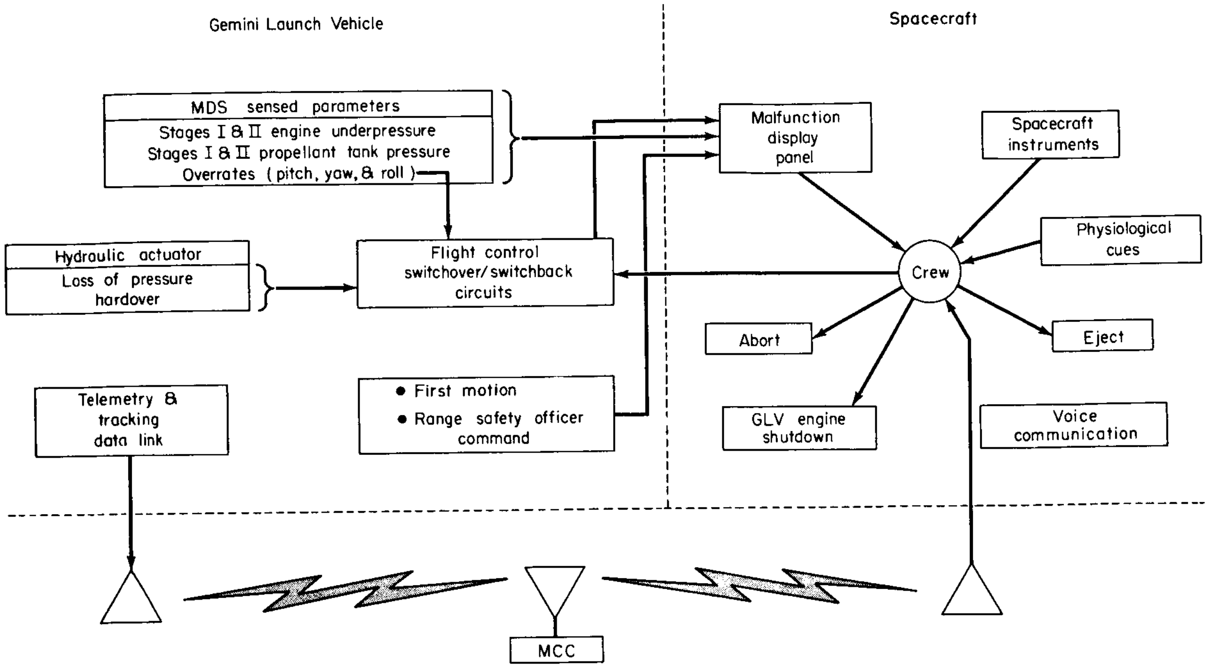


FIGURE 12-2.—Malfunction detection system.

existed for them on the GLV. These deletions resulted in a valuable weight savings and an increase in mission reliability.

(6) A new stage II oxidizer-tank forward skirt assembly was designed to mate the launch vehicle to the spacecraft.

(7) The Titan II equipment-support truss was modified to accommodate GLV equipment requirements.

(8) Devices were added to the GLV stage I propellant lines to attenuate the launch vehicle longitudinal oscillations, or POGO effect.

(9) The Titan II range-safety and ordnance systems were modified, by the addition of certain logic circuitry and by changes to the destruct initiators, to increase crew safety.

A modification not found in this listing but, nevertheless fundamental to the GLV, was the application of special techniques which significantly increased vehicle reliability. Several of these techniques will be mentioned later, but no attempt will be made to detail all the facets as they apply to the GLV. However, disciplines such as the critical-component program, the personnel training-certification and motivation program, the component limited-life program, the corrective-action and failure-analysis program, the procurement-control program, the data-trend-monitoring program, and others have been beneficial.

Pilot Safety

The pilot-safety problem was defined early in the Gemini Program by predicting the failure modes of all critical launch-vehicle systems. For the boost phase, the problem was managed by developing an emergency operational concept which employed concerted efforts by the flight crew and ground monitors, and which employed automatic airborne circuits only where necessary. Detailed failure-mode analyses defined functional requirements for sensing, display, communications, operator training, and emergency controls (fig. 12-3).

During two periods of stage I flight, escape from violent flight-control malfunctions induced by failure of the guidance, control, electric, or hydraulic power systems is not feasible; therefore, the GLV was designed to correct these failures automatically by switching over to the backup guidance and flight-control systems which include the guidance, control, electric, and hydraulic power systems. Sensing parameters for the malfunction detection system and switchover mechanisms were established. Component failure modes were introduced into a breadboard control system, tied in with a

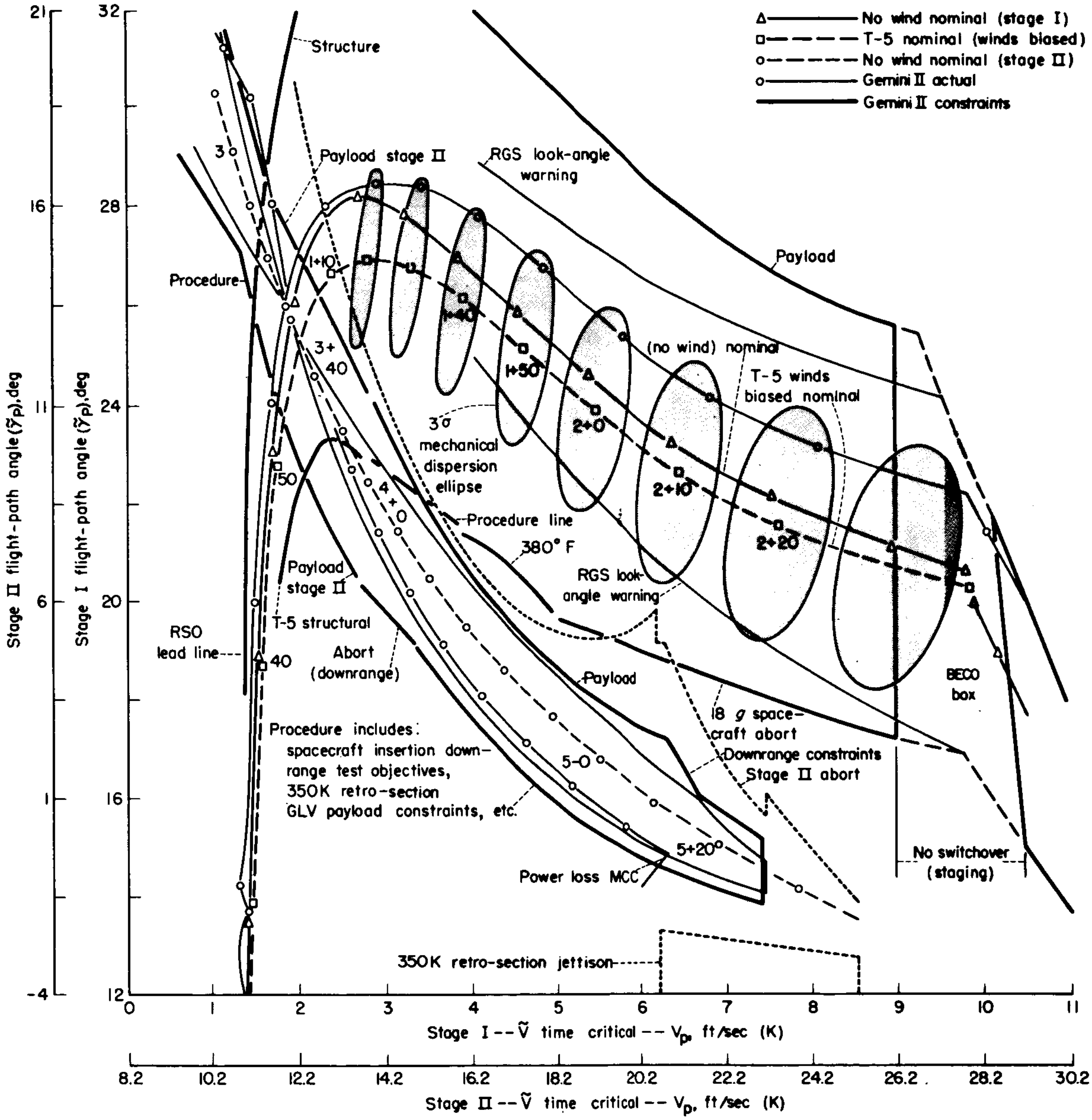


FIGURE 12-3.—Detailed failure-mode analysis.

complete airborne-system functional test stand and an analog simulation of vehicle behavior, to verify the failure mode analysis of system and vehicle effects and to optimize adjustments of the malfunction-detection-system sensors.

Isolation and analyses of the other time-critical failure modes established engine chamber pressures, tank pressures, and vehicle overrate as malfunction-detection-system sensing parameters for direct spacecraft display and for manual abort warning.

Throughout the entire abort operation, crew safety required certain configuration changes to curb excessive escape environments. The GLV strength envelope was adjusted to loads induced by malfunctions, so that structural failures during attitude divergence would be isolated to the section between stages.

Pilot safety has been actively pursued during the operational phase of the program in the form of astronaut training, development of a real-time ground-monitoring capability, and preflight integrity checks.

A catalog of normal, high-tolerance, and typical malfunction events, describing the time variations of all booster parameters sensible to the flight crew, was supplied to NASA and maintained for astronaut moving-base simulation runs and abort training. In addition to valid malfunction cues, these data emphasized the highest acceptable levels of noise, vibrations, attitude divergence, and off-nominal sequences. The flight crews have demonstrated the effectiveness of this training during the five manned flights to date. In particular, the flight crew correctly diagnosed the fact that no abort was required during the out-of-sequence shutdown event which occurred during the Gemini VI–A launch attempt.

Because a major structural failure in flight would not afford enough warning for a safe escape, a 25-percent margin of safety was provided for the specification wind environment. To insure that the actual flight environment would not exceed the specification environment, wind soundings were taken before each launch and were fed into computer simulation programs which immediately predicted flight behavior, loads, and trajectory dispersions. These results were used to verify structural margins (preflight go—no-go); to adjust the switchover constraints, abort constraints, and real-time trajectory-dispersion displays; and to brief the flight crew on predicted attitude perturbations. Thus, a technique for rapid feedback of the impact of measured weather data in time for prelaunch decisions and prediction of flight behavior had been developed and demonstrated.

Slowly developing malfunctions of the launch vehicle are monitored by ground displays (fig. 12–3) of selected telemetry and radar tracking parameters. Through these displays, the guidance monitor at the Mission Control Center in Houston is able to recommend to the crew either to switch over to the secondary systems or to switch back to the primary systems. In the event the secondary system is no-go for switchover, the monitor can advise the crew and the ground monitors of this situation. The switchover or switchback decisions are based upon potential violation of such launch-vehicle and spacecraft constraints as—

(1) Performance
(2) Structural loads
(3) Structural temperature
(4) Controllability
(5) Hatch opening
(6) Staging
(7) Spacecraft abort boundary

These constraints are developed for each launch vehicle and spacecraft prior to launch and are integrated with the prelaunch winds program to form the displays for the ground monitoring operations. The results of failure mode and constraint analysis for each flight have served to update or change mission rules, and to provide new data for both crew and ground-monitoring training. The constraints and flight results for each mission are updated prior to each launch. Gemini flight results have confirmed the usefulness of the slow-malfunction effort as part of the Mission Control Center ground-monitoring operation, and have demonstrated the feasibility of real-time monitoring, diagnosis, and communication of decisions concerning guidance and control system performance.

System Description

Structures

The basic structure of the GLV is, like Titan II, a semimonocoque shell with integral fuel and oxidizer tanks. Modifications include the addition of a 120-inch-diameter forward oxidizer skirt to accept the spacecraft adapter, and the adaptation of lightweight equipment trusses.

Early in the GLV program, complete structural loads, aerodynamic heating, and stress analyses were required because of the spacecraft configuration and boost trajectories. These analyses confirmed the adequacy of the structural design of the launch vehicle. Additional confirmation of the structure was gained by Titan II overall structural tests, and by tests of the peculiar structure of the GLV. A stage II forward oxidizer skirt and spacecraft adapter assembly was tested to a combination of design loads and heating without failure. The lightweight equipment trusses were vibration and structurally tested without failure.

An extensive structural breakup analysis and some structural testing to failure were performed in support of the pilot-safety studies. A result of these analytical studies was the incorporation of higher-strength bolts in the stage

I manufacturing splice. Strengthening of this splice minimizes the possibility of a between-tanks breakup, with subsequent fireball, in the event of certain malfunctions.

Titan II operational storage in silos is both temperature and humidity controlled. Weather protection of the GLV is provided only by the vehicle erector on launch complex 19. To prevent structural corrosion, the vehicle is selectively painted and is subjected to periodic corrosion control inspections. Stringent corrosion control procedures were established after corroded weld lands and skins were experienced on GLV–1 during its exposure to the Cape Kennedy environment.

Propulsion

Development.—The basic features of the propulsion system remain unchanged from Titan II; however, component changes, deletions, and additions have occurred where dictated by crew safety requirements.

Launch vehicle longitudinal oscillations.—POGO is a limit-cycle oscillation in the longitudinal direction of the launch vehicle, and involves structure, engines, propellants, and feedlines in a closed-loop system response.

The occurrence of longitudinal oscillations, or the POGO effect, on the first Titan II flight, in 1962, caused concern for the Gemini Program. The oscillations were about ±2.5g, and, although this was not detrimental to an intercontinental ballistic missile, it could degrade the capability of an astronaut to perform inflight functions. The POGO problem was studied and finally duplicated by an analytical model, which led to a hardware solution. The hardware consists of a standpipe inserted into the oxidizer feedline which uses a surge chamber to damp the pressure oscillations. In the fuel feedline, a spring-loaded accumulator accomplishes the same damping function.

These hardware devices were successfully tested on three Titan II flights. Considerable improvements in performance, checkout, and preparation for launch have been achieved through the first seven Gemini launches. Major redesigns of the fuel accumulators have helped to reduce POGO to well within the ±0.25g criterion established for the Gemini Program. The one exception, GLV–5, where levels of ±0.38g were recorded, was due to improper preflight charging of the oxidizer standpipe. Charging methods and recycle procedures were subsequently modified, and, on GLV–6 and GLV–7, POGO levels were within the ±0.25g requirements. The new oxidizer standpipe remote-charge system has eliminated a difficult manual operation late in the countdown, and has provided increased reliability and a blockhouse monitoring capability.

Figure 12–4 shows the history of success in eliminating POGO. With one exception, all Gemini results are below ±0.25g, and an order of magnitude less than the first Titan II vehicles.

Electrical

The GLV electrical system was modified to add complete system redundancy, and to supply 400-cycle power and 25-V dc power which the Titan II does not require.

The electrical system consists of two major subsystems: power distribution and sequencing. A block diagram of the electrical power subsystem, illustrating how it is integrated with the launch vehicle systems, is shown in figure 12–5. The power subsystem is fully redundant, with wiring routed along opposite sides of the vehicle. Special fire protection is given to the stage I engine-area wiring by wrapping the wire bundles with an insulating material and also with aluminum-glass tape. Spacecraft interface functions are provided through two electrical connectors, with a com-

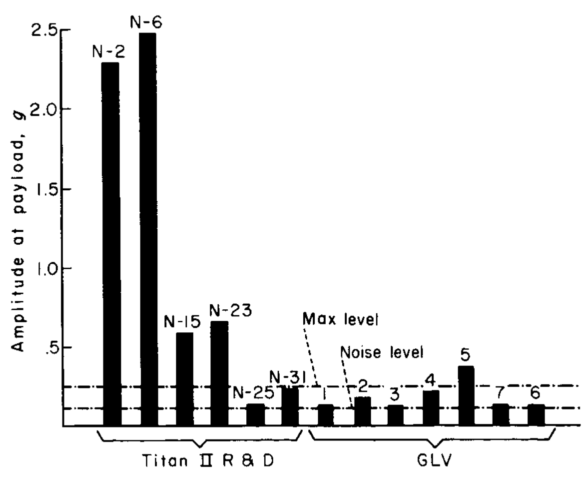


FIGURE 12–4.—History of POGO reduction.

plete set of functions wired through each connector.

The redundant electrical sequencing subsystem consists of relay and motor-driven switch logic to provide discrete signals to the vehicle systems. A block diagram of the sequencing subsystem is shown in figure 12–6. To insure that the critical stage II shutdown function will be implemented when commanded, a backup power supply is provided.

The electrical system has performed as designed on all GLV flights. The 400-cps power,

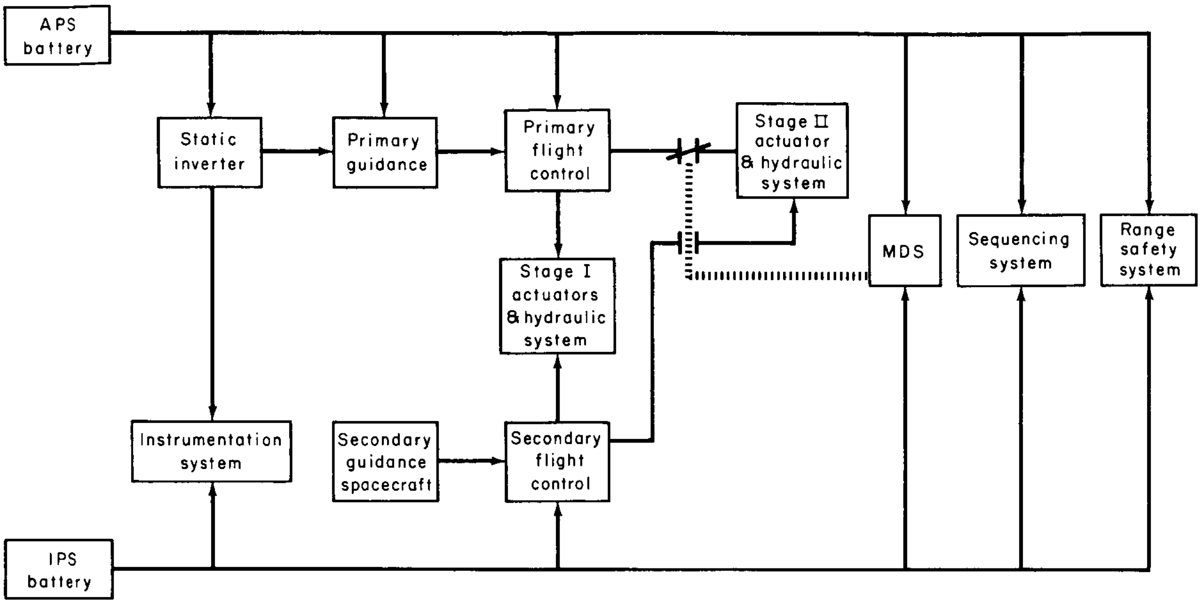


FIGURE 12–5.—Electrical power subsystem.

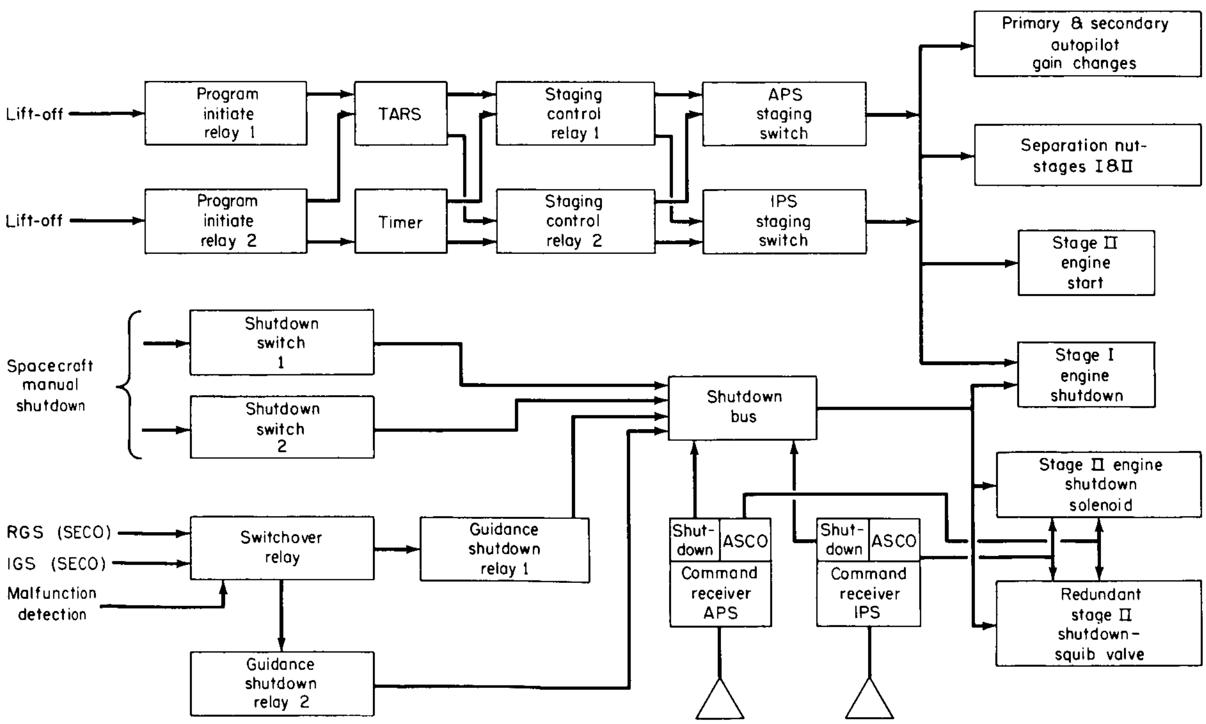


FIGURE 12–6.—Sequencing subsystem.

which is required by the primary guidance flight-control system for timing reference, has not deviated by more than ±0.5 percent, although the specified frequency tolerance is ±1 percent. The discrete timing functions of the sequencing subsystem have been well within the specified ±3 seconds. Power system voltages, with auxiliary and instrumentation power supply, have been within the specified 27- to 31-V dc range. Thus, if switchover to the secondary guidance and control system had occurred, the instrumentation power supply would have performed satisfactorily for backup operations.

Guidance and Control

The GLV redundant guidance and control system (fig. 12-7) was designed to minimize the probability of a rapidly developing catastrophic malfunction, such as a sustained engine hardover during stage I flight, and to permit the use of a manual malfunction detection system. A second objective of the added redundancy was to increase overall system reliability and, consequently, to increase the probability of mission success. Some of the more important system characteristics are:

(1) A mission can be completed after any single malfunction during stage I flight, and there is partial redundancy during stage II flight.

(2) Switchover can be implemented automatically or manually during either stage of powered flight.

(3) Flight-proven hardware from Titan I and Titan II is used wherever possible.

(4) There is complete electrical and physical isolation between the primary and secondary systems.

(5) The relatively simple switchover circuitry is designed for the minimum possibility of a switchover-disabling-type failure or an inadvertent switchover failure.

Even though the GLV guidance and control system is based upon Titan hardware, the system is quite different. The major system changes are the addition of the radio guidance system and the three-axis reference system in the primary system to replace the Titan II inertial guidance system, and the incorporation of new configuration tandem actuators in stage I. The selection of the radio guidance system and three-axis reference system required that an adapter package be added to make the three-axis reference system outputs compatible with the Titan II autopilot control package.

Stage I hydraulic redundancy is achieved by using two complete Titan II power systems.

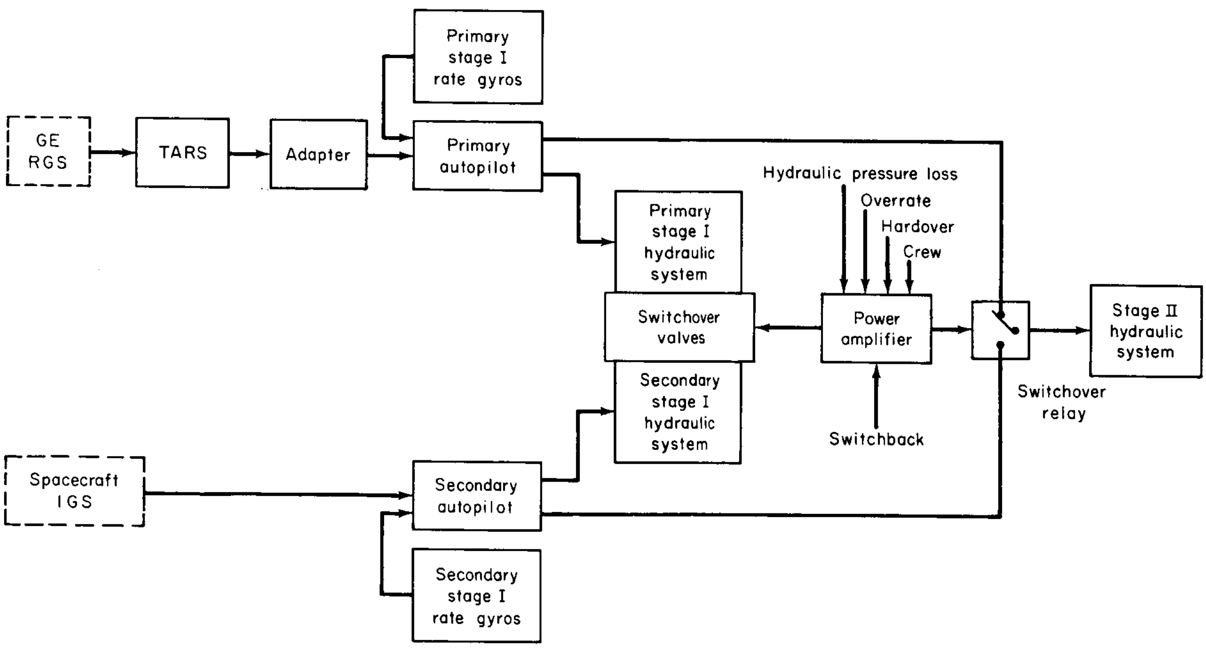


FIGURE 12-7.—Guidance and control subsystems.

The actuators are tandem units with a primary and secondary system section. Each section is a complete electrohydraulic servo, capable of driving the common piston rod. The major components comprising each servoactuator are the same as those used in Titan II actuators. The tandem actuator (fig. 12-8) contains a switchover valve, between the two servovalves and their respective cylinders, which deactivates the secondary system while the primary system is operating, and vice versa, following switchover to the secondary system.

Switchover.—There are four methods for initiating a switchover to the secondary system, and all modes depend on the malfunction detection system.

(1) The tandem actuator switchover valve automatically effects a switchover to the stage I secondary hydraulic system when primary system pressure is lost, and initiates a signal to the malfunction detection system which completes switchover to the secondary guidance and control system.

(2) The malfunction detection system rate-switch package automatically initiates switchover when the vehicle rates exceed preset limits.

(3) The tandem actuator preset limit switches detect and initiate a switchover in the event of a stage I engine hardover.

(4) The crew may initiate a switchover signal to the malfunction detection system upon determining, from spacecraft displays or from information sent by ground-monitoring personnel, that a primary system malfunction has occurred.

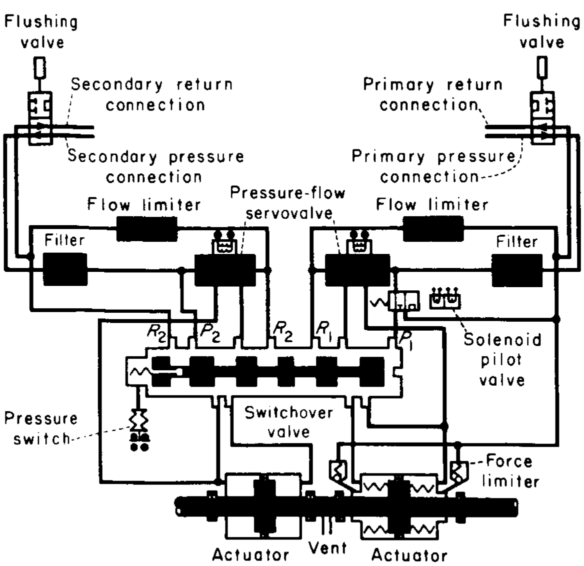


FIGURE 12-8.—Tandem actuator.

Upon receipt of a switchover signal, the inertial guidance system performs a fading operation which reduces the output to zero, and then restores the signal to the system according to an exponential law. This minimizes vehicle loads during the switchover maneuver.

Flight performance.—All GLV flights have been made on the primary system, and performance has been satisfactory, with no anomalies occurring. All flight transients and oscillations have been within preflight analytical predictions.

Although there has not been a switchover to the secondary flight-control system, its performance has been satisfactory on all flights. Postflight analysis indications are that this system could have properly controlled the launch vehicle if it had been necessary.

During the program, the capability of variable-azimuth launch, using the three-axis reference system variable-roll-program set-in capability, has been demonstrated, as has the closed-loop guidance steering during stage II flight.

Malfunction Detection System

The malfunction detection system, a totally new system, encompasses the major inflight launch-vehicle malfunction sensing and warning provisions available for crew safety. The performance parameters displayed to the flight crew are:

(1) Launch-vehicle pitch, yaw, and roll overrates.

(2) Stage I engine thrust-chamber underpressure (subassemblies 1 and 2, separately).

(3) Stage II engine fuel-injector underpressure.

(4) Stage I and II propellant-tank pressures.

(5) Secondary guidance and control system switchover.

The crew has three manual switching functions associated with the malfunction detection system: switchover to the secondary guidance and control system, switchback to the primary guidance and control system, and launch-vehicle shutdown.

The implementation of the malfunction detection system considers redundancy of sensors and circuits and isolated installation of redundant elements to minimize the possibility of a single or local failure disabling the system. Also, probable failure modes were considered in component design and selection and in circuit connection in order to provide the malfunction detection system with a greater reliability than that of the systems being monitored.

The total malfunction sensing and warning provisions, including the malfunction detection system, and the interrelation of these are shown in figure 12–2.

Monitoring techniques.—The malfunction detection system is a composite of signal circuits originating in monitoring sensors, routed through the launch vehicle and the interface, and terminating in the spacecraft warning-abort system (fig. 12–9).

Stages I and II malfunction detection system engine-underpressure sensors are provided in redundant pairs for each engine subassembly. The warning signal circuits for these are connected to separate engine warning lights in the spacecraft. Upon decrease or loss of the thrust-chamber pressure, the redundant sensor switches close and initiate a warning signal.

Except for the pressure operating range, all malfunction detection system propellant-tank pressure sensors and signal circuits are identical. A redundant pair of sensors is provided for each propellant tank. Each sensor supplies an analog output signal, proportional to the sensed pressure, to the individual indicators on the tank pressure meters in the spacecraft.

Launch-vehicle turning rates, about all three axes, are monitored by the malfunction detection system overrate sensor. In the event of excessive vehicle turning, a red warning light in

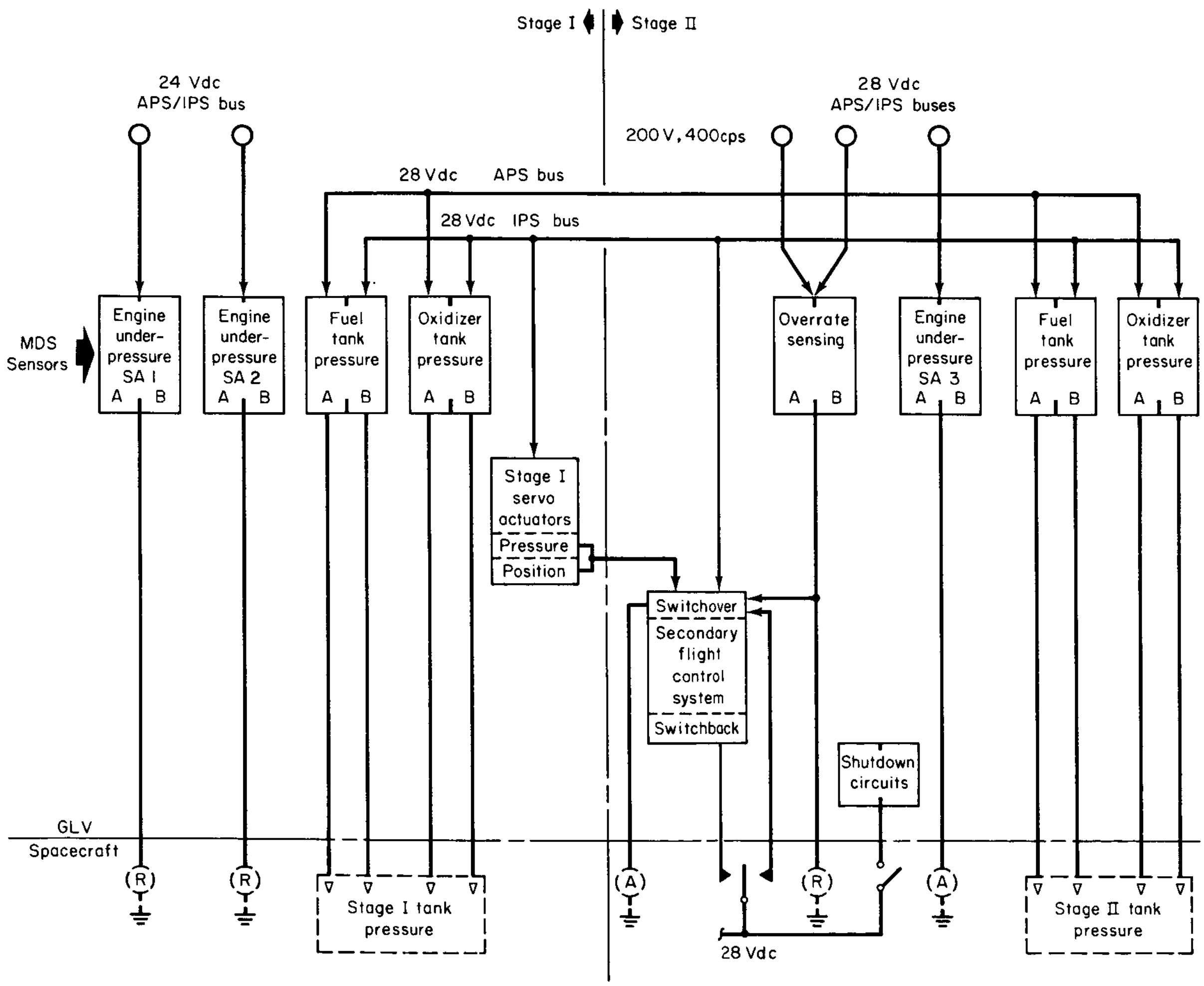


FIGURE 12–9.—Spacecraft monitoring of Gemini launch vehicle malfunction detection.

the spacecraft is energized. Simultaneously and automatically, a signal is provided to initiate switchover to the secondary flight-control system. The overrate sensor is the malfunction detection system rate-switch package, consisting of six gyros as redundant pairs for each of the vehicle body axes (pitch, yaw, and roll). In the malfunction detection system circuits, the redundant rate switches are series connected, and simultaneous closure of both switches in the redundant pair is required to illuminate the warning light in the spacecraft and to initiate switchover.

The dual switchover power-amplifiers are self-latching solid-state switching modules used to initiate a switchover from the primary to the secondary guidance and control system. On the input side, signals are supplied either from the malfunction detection system overrate circuits; from the stage I hydraulic actuators, low pressure or hardover; or from the flight crew in the case of a malfunction. An unlatching capability is provided for the switchover power amplifiers to permit switchback from the secondary to the primary guidance and control system during the stage II flight.

Launch-vehicle engine shutdown can be manually initiated by the flight crew in the case of a mission abort or escape requirement.

There have been several significant changes made to the malfunction detection system since the beginning of the program. These entailed addition of the switchback capability, a change to the stage I flight switch settings of the rate-switch package, and deletion of the staging and stage-separation monitoring signals. Figure 12–10 shows the location of the malfunction detection system components.

Flight performance.—All malfunction detection system components have undergone a similar design verification test program which included testing at both the component and system levels. At the component level, evaluation, qualification, and reliability tests were conducted. System verification and integration with other launch-vehicle systems were performed in the airborne systems functional test set. In addition, flight performance verification was accomplished by means of the Titan II piggyback program. Table 12–I presents the flight performance of the malfunction detection system components. With the exception of two problems which were corrected (a minor oscillation problem occurring on two tank sensors prior to the first manned flight, and a slightly out-of-tolerance indication on one rate-switch operation during the second Piggyback flight),

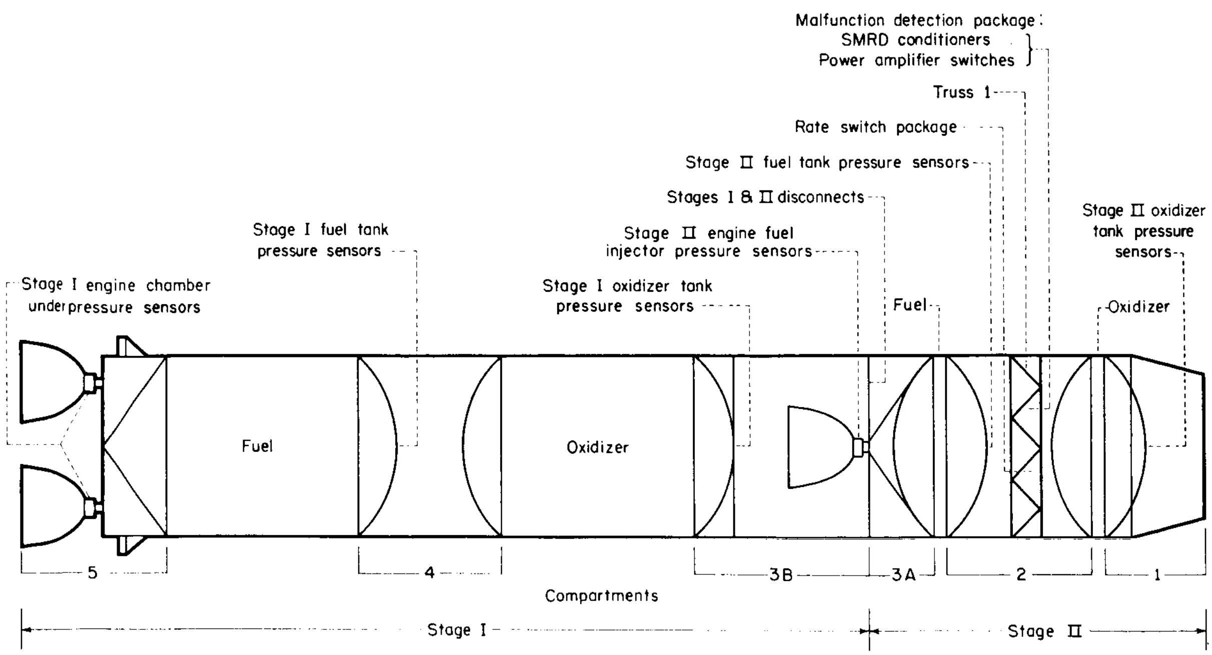


Figure 12–10.—Malfunction detection system components location.

TABLE 12-I.—*Flight Performance of Malfunction Detection System Components* [a]

Malfunction detection system components	Number flown	Results
Tank sensors	96	All units operated satisfactorily; slight output oscillation on 2 units
Rate-switch package	12 (72 gyros)	Of a total of 142 rate-switch operations, 141 were in agreement with rate-gyro data
Malfunction detection package.	12 (24 switchover circuits) (72 rate-switch package gyro spin-motor-rotation-detector monitors)	16 satisfactory operations of switchover circuits; normal operation of 72 spin-motor-rotation-detector monitors
Engine sensors	72	144 satisfactory switch actuations associated with normal inflight engine start and cutoff operations

[a] Data based on 5 Titan II piggyback flights and 7 Gemini flights.

the malfunction detection system has performed as intended.

Test Operations

Airborne Systems Functional Test Stand

The airborne systems functional test stand is an operational mockup of essentially all of the electrical-electronic-hydraulic elements of the launch vehicle, complete with engine thrust chambers and other associated engine hardware. In some systems, such as flight control and the malfunction detection system, the aerospace ground equipment is integrated into the test stand, while in other systems, the aerospace ground equipment is simulated.

The initial purpose of the airborne systems functional test stand was to verify the GLV system design; specifically, systems operation, interface compatibility, effects of parametric variations, adequacy of operational procedures, etc. This was accomplished early in the program so that the problems and incompatibilities could be factored into the production hardware before testing GLV-1 in the vertical test fixture in Baltimore. Even though the formal test-stand test program has been completed, the facility has been used continuously to investigate problems resulting from vertical test fixture and Cape Kennedy testing, and also to verify all design changes prior to their incorporation into the production hardware.

The test stand has proved to be an extremely valuable tool, particularly in proving the major system changes such as guidance and control redundancy and the malfunction detection system. It has also served as a valuable training ground for personnel who later assumed operational positions at the test fixture and at Cape Kennedy. Many of the procedures considered to be important to the program, such as malfunction disposition meetings, handling of time-critical components, and data analysis techniques, were initiated and developed in the test stand.

System verification testing with other launch-vehicle systems was performed in the test stand using flight hardware. This testing was performed on two levels: functional performance and compatibility with other systems, and performance in controlling the launch vehicle in simulated flight.

Vertical Testing at Baltimore

Vehicle checkout and acceptance testing in the Martin-Baltimore vertical test fixture was initiated on June 9, 1963. The baseline test program started with a post-erection inspection followed by power-on and subsystem testing. After an initial demonstration of the combined systems test capability, GLV-1 underwent a comprehensive electrical-electronic interference measurement program during a series of combined systems test runs. Based on recorded and telemetered system data, several modifications were engineered to reduce electrical-electronic interference effects. As part of this program, both in-sequence and out-of-sequence umbilical drops were recorded wih no configuration changes required. Following electrical-electronic interference corrective action, GLV-1 was run successfully through a combined systems acceptance test. Test acceptance was based primarily on several thousand parameter values from aerospace ground equipment and telemetry recordings.

Electrical-electronic interference testing was reduced on GLV–2 because GLV–1 data showed noise levels well within the established criteria. Test results on GLV–2 confirmed the GLV–1 modifications, and the electrical-electronic interference effort on subsequent vehicles was limited to monitoring power sources.

A summary of vertical test fixture milestones is presented in table 12–II.

The vertical test fixture operational experience confirms the importance of program disciplines such as configuration control, rigid work control, and formal investigation of malfunctions as factors establishing test-article acceptability. The detailed review of acceptance test data, including the resolution of every single data anomaly, also facilitated the acceptance process.

Testing at Cape Kennedy

GLV–1 was erected on launch complex 19 at Cape Kennedy on October 30, 1963, and an extensive ground test program in both side-by-side and tandem configurations was initiated. The program included a sequence compatibility firing, in which all objectives were achieved.

Testing in the tandem configuration included fit-checks of the erector platforms, umbilicals, and white room. A series of electrical-electronic interference tests, using a spacecraft simulator with in-sequence and out-of-sequence umbilical drops, and an all-systems test were conducted as part of the program for complex acceptance.

The GLV–2 operations introduced a number of joint launch-vehicle–spacecraft test events. These included verification of wiring across the interface; functional compatibility of the spacecraft inertial guidance system and the launch-vehicle secondary flight-control system; an integrated combined-systems test after mating the spacecraft to the launch vehicle; a similar test conducted by both the spacecraft and launch vehicle, including umbilical disconnect; and final joint-systems test to establish final flight readiness. (See table 12–III.)

The electrical-electronic interference measurements and umbilical drops were recorded during system tests of GLV–2 and spacecraft 2. The only hardware change was a spacecraft correction for a launch-vehicle electronic interference transient during switchover. As a result, further testing on subsequent vehicles was not considered necessary.

A streamlining of all system tests resulted in a test time of 6 to 7 weeks. This program replanning increased the proposed firing rate and allowed overall program objectives to be attained in 1965.

Gemini operations with GLV–5 included the first simultaneous countdown with the Atlas-Agena as part of a wet mock simulated launch. The changes arising from this operation were verified with GLV–6 and resulted in a no-holds, joint-launch countdown.

When the first attempt to launch GLV–6 was scrubbed because of target vehicle difficulties, an earlier Martin Co. proposal for rapid fire of two launch vehicles in succession from launch complex 19 was revived. The decision to implement this plan resulted in GLV–6 being placed in horizontal storage from October 28 to December 5, 1965. In the interim, GLV–7, whose schedule had been shortened by the deletion of the flight configuration mode test and wet mock simulation launch (a tanking test was substituted for the latter), was launched on December 4. GLV–6 was reerected on December 5 and launched successfully on December 15 after an initial launch attempt on December 12. The technical confidence which justified such a shortened retest program was based upon the previous successful GLV–6 operation, the maintenance of integrity in storage, and the reliance on data trend analysis to evaluate the vehicle readiness for flight. During retests, only one item, an igniter conduit assembly, was found to be defective.

Major test events for GLV–1 through GLV–7 are presented in table 12–III.

Test Performance

The vertical test fixture performance is exemplified by indicators such as the number of procedure changes, the equipment operating hours, the number of component replacements, and the number of waivers required at the time of acceptance. These factors, presented in figure 12–11, show a significant reduction fol-

TABLE 12-II.—*Vertical Test Fixture Milestone Summary*

	GLV-1	GLV-2	GLV-3	GLV-4	GLV-5	GLV-6	GLV-7	GLV-8	GLV-9
Date of erection	June 9, 1963	Feb. 7, 1964	June 22, 1964	Oct. 26, 1964	Feb. 5, 1965	Apr. 14, 1965	June 25, 1965	Sept. 28, 1965	Dec. 10, 1965
Post-erection inspection	X	X	X	X	X	X	X	X	X
Modification		X					X		
Subsystem functionals	X	X	X	X	X	X	X	X	X
Data acquisition		X	X	X					
Electrical-electronic interference	X	X							
Umbilical drop	X								
Instrumentation marriage and ambient					X	X	X	X	X
Date of combined systems acceptance test	Sept. 6, 1963	Apr. 22, 1964	Aug. 4, 1964	Nov. 25, 1964			Sept. 20, 1965	Nov. 8, 1965	Feb. 4, 1966
Modification	X		X		X	X			
Date of combined systems acceptance test	Oct. 4, 1963		Sept. 30, 1964		Apr. 21, 1965	June 25, 1965			

TABLE 12-III.—*Launch-Vehicle Test Event Summary—Cape Kennedy*

Test event	Gemini launch vehicle								
	1	2	3	4	5	6	7	6-A	8 and up [a]
Sequenced compatibility firing, erect	X	------	------	------	------	------	------	------	------
Subsystem functional verification tests	X	------	------	------	------	------	------	------	------
Combined systems test	X	------	------	------	------	------	------	------	------
Wet mock simulated flight test	X	------	------	------	------	------	------	------	------
Sequenced compatibility firing	X	------	------	------	------	------	------	------	------
Tandem erect	X	X	X	X	X	X	X	X	X
Subsystem functional verification tests	X	X	X	X	------	------	------	------	------
Subsystem reverification tests	------	------	------	------	X	X	X	X [b]	X
Premate combined systems test	X	X	X	X	X	X	X [c]	------	X
Electrical-electronic interference	X	X	X	------	------	------	------	------	------
Electrical interface integrated validation and joint guidance and controls	------	X	X	X	X	X	X	------	X
Electrical-electronic interference	------	X	------	------	------	------	------	------	------
Joint combined systems test	------	X	X	X	X	X	X	------	X
Flight configuration mode test umbilical drop	------	X	X	X	X	X	------	------	------
Umbilical drop	X	X	------	------	------	X	X	------	------
Tanking	X	X	X	X	X	------	------	------	X
Wet mock simulated launch	------	------	------	------	X	X	------	------	X
Wet mock simulated launch, simultaneous launch demonstration	X	X	X	X	X	X	X	X	X
Simulated flight test	X	X	X	X	X	------	X	X	------
Double launch	------	------	------	------	------	X	------	------	X

[a] Current plan.
[b] Modified.
[c] Umbilical drop added.

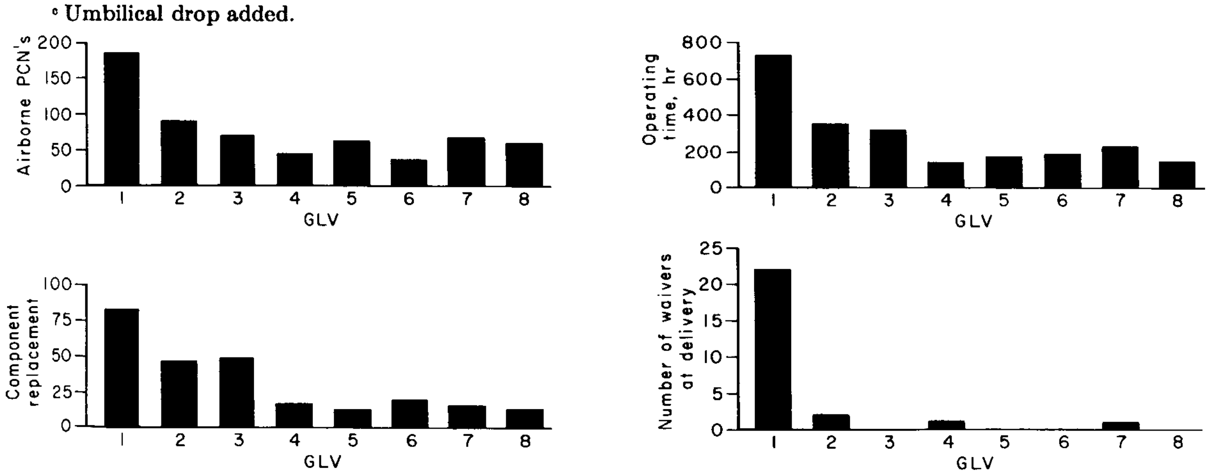


FIGURE 12-11.—Vertical test fixture performance.

lowing the first test fixture operation. This performance improvement is due largely to the vigorous corrective actions initiated to correct the early problems. As such, this action helped produce increasingly reliable hardware and thereby reduced testing time and operating hours. The decrease in procedure changes reflects the rapid stabilization of the testing configuration.

Schedule performance at Cape Kennedy is subject to environment, special testing, and program decisions, and does not indicate improvement in the testing process as effectively as equipment power-on time and component changeout, other than for modification (fig. 12–12). The operating time reductions indicated in figure 12–12 stem primarily from the elimination of one-time or special tests, a decrease in redundant testing, and improvements in hardware reliability. The reduced number of discrepancies when the launch vehicle is received from the vertical test fixture, as well as minimal field modifications, also contributed to improved test efficiency.

As shown in figure 12–12, the decrease in test complexity and the refinement of the testing process are indicated by the decreasing number of procedure change notices generated per vehicle.

An overall measure of test and hardware performance per vehicle is presented in figure 12–13, which shows that the number of new problems opened for each launch vehicle had diminished from 500 to 5 through the launch of Gemini VII.

Data-Trend Monitoring

A data-trend monitoring effort is maintained as part of the launch-vehicle test program. The purpose of the program is to closely examine the performance of components and systems at specified intervals. This is done by having design engineers analyze all critical system parameters in detail during seven prelaunch test operations, which cover a period of 4 to 5 months, and then entering these values into special data-trend books. Because these data have already been analyzed and shown to be within the allowed specification limits, this second screening is to disclose any trend of the data which would be indicative of impending out-of-tolerance performance or failure, or even performance which is simply different from the previous data.

On a number of occasions, equipment has been removed from the vehicle, and at other times special tests were conducted which removed any shadow cast by the trend. In such cases, the history of the unit or parameter, as told by all previous testing on earlier vehicles, was researched and considered prior to package replacement. A typical data-trend chart for the electrical system is shown in table 12–IV.

The launch-vehicle data-trend monitoring program has been of particular significance on two occasions: when GLV–2 was exposed to a lightning storm, and when deerection and re-erection were necessary after a hurricane at Cape Kennedy. A number of electrical and electronic components in both the aerospace ground equipment and airborne areas, some of which were known to be damaged and others which were thought to have been degraded due to overvoltage stress, were replaced. During the subsequent retesting, an even more comprehensive data-trend monitoring program was implemented to insure that the integrity of the launch vehicle had not been impaired due to the prior events. All test data were reviewed by

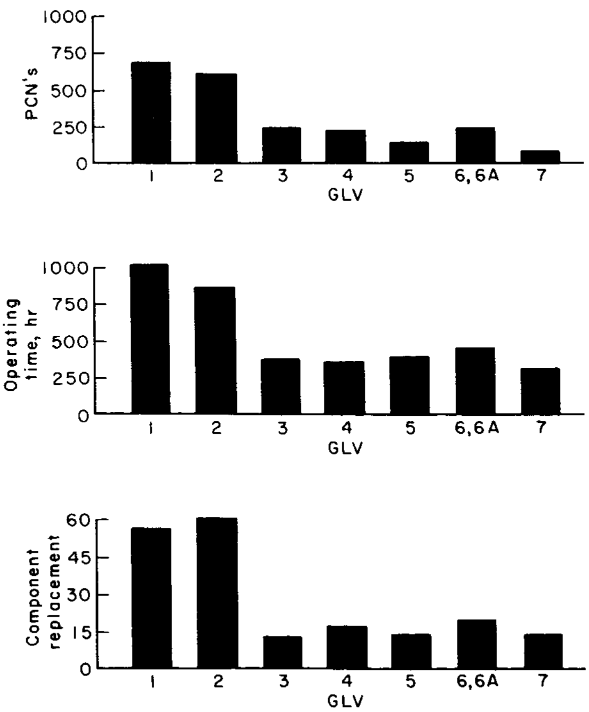


FIGURE 12–12.—Cape Kennedy testing performance.

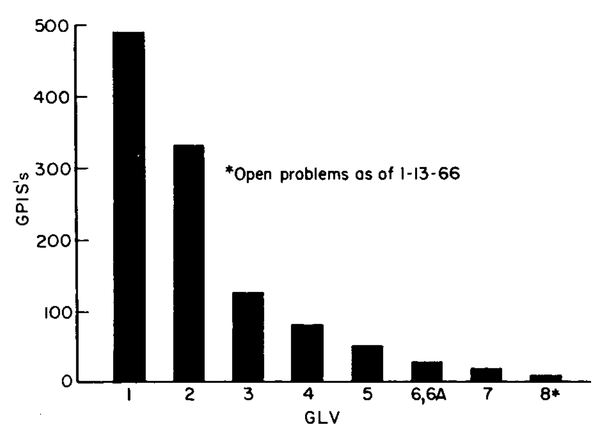


FIGURE 12–13.—Overall measure of test performance.

TABLE 12-IV.—*Gemini Launch Vehicle No. 6 Data-Trend Monitoring (Typical Chart)*

Line no.	Measurement no.	Parameter	Special or nominal value and tolerance	VTF tests		ETR tests							
				CSAT Date 6-25-65 Test no. 011/012		Pre-SC mate Date 9-16-65 Test no. 5547		ELLV (ETR) Date 9-20-65 Test no. 5750	JCST Date 9-23-65 Test no. 5751	FCMT Date 10-1-65 Test no. 5901	WMSL Date 10-7-65 Test no. 6000	SFT Date 10-20-65 Test no. 6260	
				1	2	1	2	1	1	1	1	1	2
1		PS940300011		−001	−001	NiCd	NiCd	NiCd	NiCd			−001	−001
	0800	IPS battery volts	27 to 31 V dc	29.1	29.8	28.5	28.3	29.0	28.2	29.0	29.0	29.0	29.0
	0804	IPS battery amps		29.9	26.9	25.9	25.9	29.9	27.9	29.9	25.4	28.9	29.9
2		PS940300011		−001	−001	NiCd	NiCd	NiCd	NiCd			−001	−001
	0801	APS battery volts	27 to 31 V dc	29.7	30.1	28.7	28.5	29.8	29.0	29.7	29.9	29.9	29.7
	0805	APS battery amps		34.3	28.0	26.9	26.9	25.2	27.3	27.3	24.2	24.2	28.3
3		PS946000001		−007	−007	−007	−007	−007	−007	−007	−007	−007	−007
	0802	Static inv volts	113 to 117 V ac	114.3	114.4	113.7	113.7	113.9	113.5	113.5	113.8	113.9	113.9
	0803	Static inv freq	396 to 404 cps	399.4	399.4	No data [a]	No data [a]	399.8	399.4	[b] 397.7	398.8	400.7	400.5
		Serial number		R31	R31	R31	R31	R31	R31	R31	R31	R31	R31
4		CC19401A11		−1	−1	−1	−1	−1	−1	−1	−1	−1	−1
	0726	Pwr supply 25 Vdc	24.1 to 25.9 V dc	25.1	25.2	25.3	25.1	25.1	25.1	25.1	25.1	25.2	25.2
		Serial number		170	170	170	170	170	170	170	170	170	170

NOTES:
[a] Vehicle access doors not installed.
[b] 392.1-399.4 variation—substitute access doors installed.

design engineers, and any peculiar or abnormal indication or any data point falling in the last 20 percent of the tolerance band was cause for a comprehensive review, with hardware troubleshooting as required.

After the launch-vehicle storage period at Cape Kennedy and prior to the launch, all testing data were reviewed in a similar manner. Additionally, a digital computer program was used to print-out the simulated flight-test data points which differed between the prestorage and poststorage simulated flight tests by more than three telemetry data bits, or approximately 1 percent. All such differences were reviewed and signed-off by design engineers when the investigations were completed.

The data-trend monitoring program has added materially to launch confidence by adding an extra dimension to test data analysis.

Personnel Training, Certification, and Motivation

From the inception of the Gemini Program, it was recognized that the high-quality standards needed could not be achieved by tighter-than-ever inspection criteria alone. Personnel working on the program had to know what was required for the program, and had to personally desire to achieve those requirements. In view of these factors, it was realized that the only thing that was going to make this program better than any other program was properly trained and motivated people.

To meet these challenges, personnel training and certification (fig. 12–14) was used to maximum advantage, with five specific areas of concentration:

(1) Orientation of all program and staff support personnel toward the program goals and objectives.

(2) General familiarization of top management to aid in making decisions.

(3) Detailed technical training for all program personnel to a level commensurate with job position, with training continuously available.

(4) Certification of the launch-vehicle production team.

(5) Certification of the test and the checkout and launch crews.

Within 3 months from the program go-ahead, orientation lectures were being presented in Baltimore, Denver, and Cape Kennedy. Attendance was not confined solely to launch-vehicle personnel; personnel from staff support groups also attended. It was necessary that the manufacturing planning, purchasing, shipping and receiving, and production control personnel understand firsthand that to attain perfection would involve stringent controls and procedures.

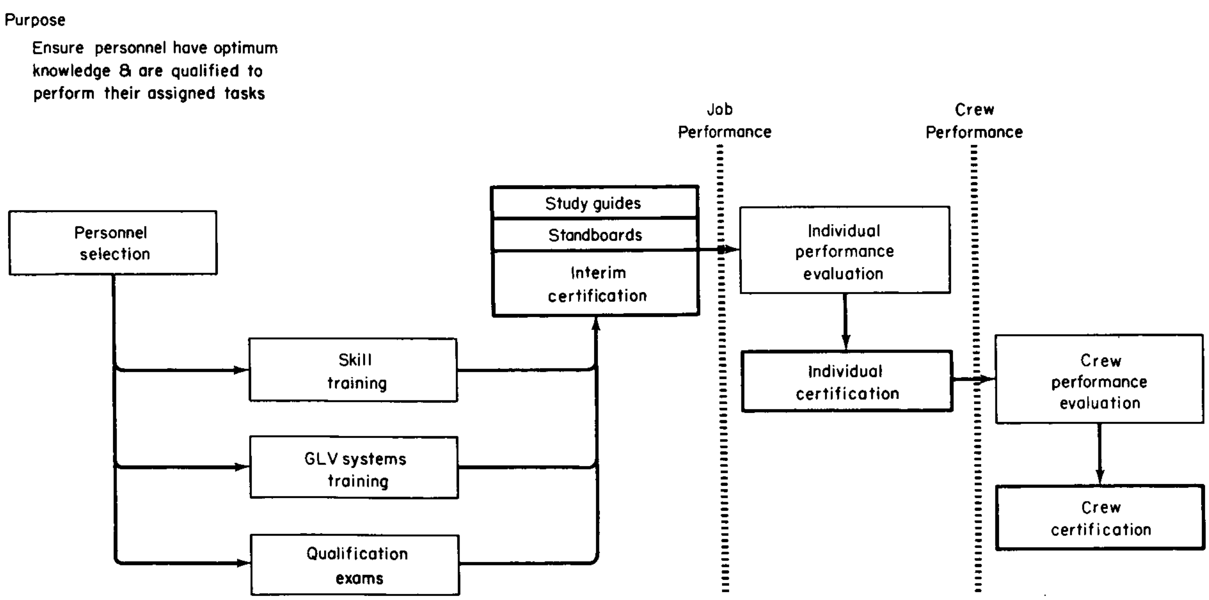


FIGURE 12–14.—Personnel training and certification.

Some of the promotional methods employed were: motivational posters; an awards program which recognized significant meritorious achievements; letters written by the program director to the wives of employees explaining the significance of the program; vendor awards; special use of the Martin-originated zero defects program; visits to the plant by astronauts; broadcasting accounts of launch countdowns to the work areas; and programed instruction texts for use by personnel on field assignments. In these ways, the personnel were continuously kept aware of the importance of the program and of the vital role that each individual played achieving the required success.

In obtaining people for the program, careful screening of potential personnel was conducted in an effort to select people with Titan experience. After selection, the people were trained; for example, some 650 classroom presentations have resulted in more than 7000 course completions. The majority of these have been familiarization courses, the others being detailed courses specifically designed for the test and launch personnel.

After completing written examinations, test personnel are issued interim certifications, permitting them to perform initial test operations. Following this, a performance evaluation is made by a review team which results in formal certification of the technical competence of the individual to perform his job functions.

Through the processes of the motivational programs, training, and certification, the launch-vehicle team has achieved the desired results. However, so long as humans are performing tasks, mistakes will be made. It is these mistakes that command continued emphasis so that the success of the remaining launch vehicles will be insured.

13. PROPULSION SYSTEM

By E. Douglas Ward, *Gemini Program Manager, Aerojet-General Corp.*

Summary

Adapting liquid rocket engines developed for the Air Force Titan II intercontinental ballistic missile to meet the rigid requirements for manned space missions of the Gemini Program was the assignment accomplished by the Liquid Rocket Operations of Aerojet-General Corp., Sacramento, Calif.

Introduction

During the conceptual stages of the Titan II engine, it was recognized that increased reliability could be obtained through simplicity of design. In achieving this goal, the number of moving parts in the stage I and II engines was reduced to a bare minimum. As an example, the Titan I engines had a total of 245 moving parts versus a total of 111 for the Titan II engines. Further, the number of power control operations on Titan I was 107 versus 21 for the Titan II.

Storable propellants were chosen for use because of the requirement for long-term storage in an instant-ready condition that was imposed on the weapons system.

Stage I Engine

The stage I engine (figs. 13–1 and 13–2) includes two independent assemblies that operate

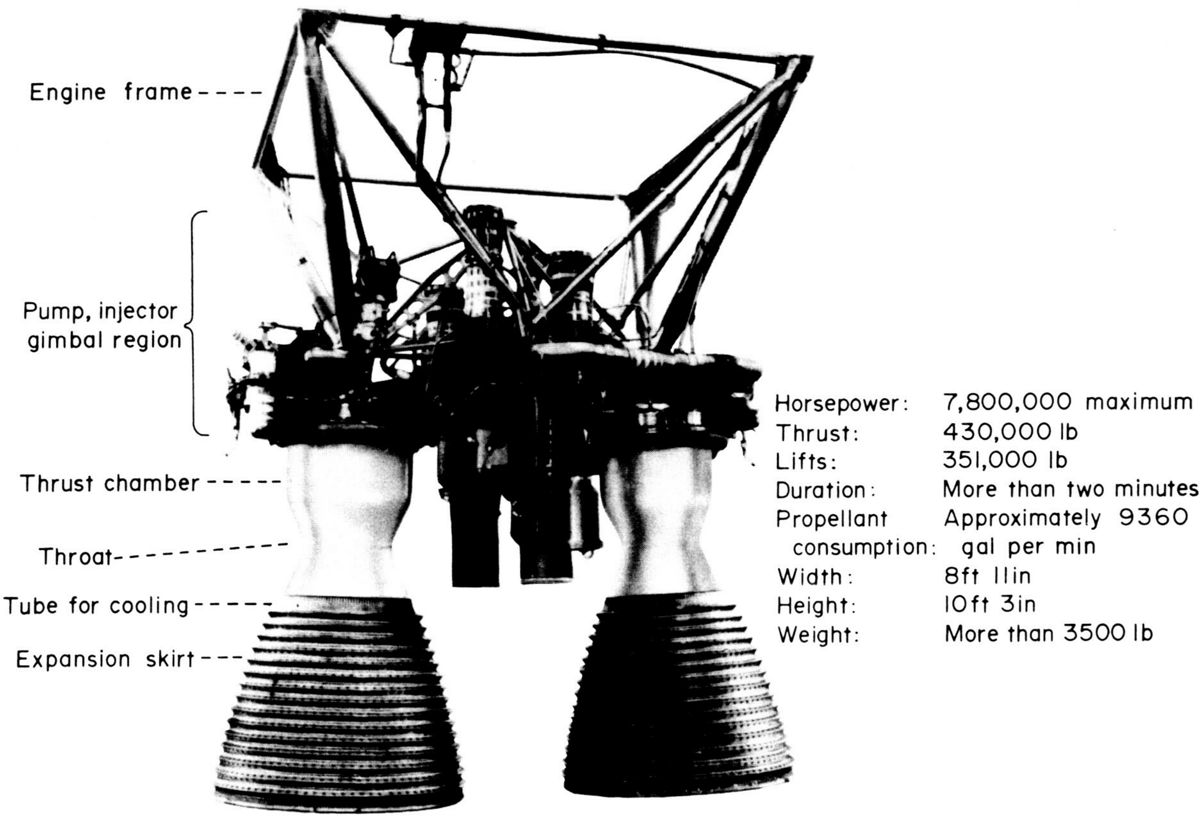


Figure 13–1.—U.S. Air Force first-stage engine for Gemini Program.

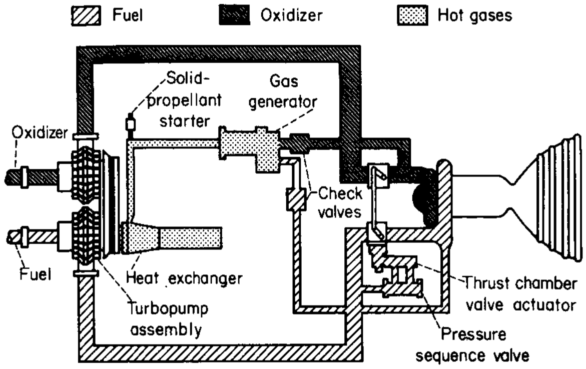


FIGURE 13-2.—Gemini stage 1 engine schematic.

simultaneously. Each subassembly contains a thrust chamber, turbopump, and gas generator assembly, as well as a starter cartridge, propellant plumbing system, and electrical controls harness. In addition, subassembly 2 provides the energy source for the stage I oxidizer and fuel tank pressurization, commonly referred to as the autogenous system (fig. 13-3). Each thrust chamber is gimbaled to provide vehicle pitch and yaw steering and vehicle roll control.

Stage II Engine

The stage II engine system (figs. 13-4 and 13-5) is a scaled-down version of a stage I engine subassembly. The stage II engine does include a thrust-chamber nozzle extension for additional efficiency at high altitudes and a vehicle roll-control nozzle. The stage II engine fuel-tank pressurization system is shown in figure 13-6.

Gemini Unique Engine Components

With the inception of the Gemini Program, rigorous engineering studies were initiated in an effort to identify hardware requiring design and development as a result of the stringent goals imposed on the engines. The requirements for the utmost in manned flight safety and reliability dictated several changes to the Titan II engine design and operation. The design changes evolved from two primary items: (1) crew safety requirements for warning the flight crew in case of incipient failures, and (2) increased reliability of component operation. The reliability of the engine operation is such that crew safety design improvements have not been utilized in any of the five manned launches to date; however, their availability provides added flight-crew safety in case problems do occur.

Hardware Changes

Malfunction Detection System

A malfunction detection system was incorporated to provide a warning to the astronauts in case of an engine performance degradation. The malfunction detection system provides an electrical signal to a spacecraft light as a visual warning to the astronaut. This is accomplished by pressure switches installed in the engine cir-

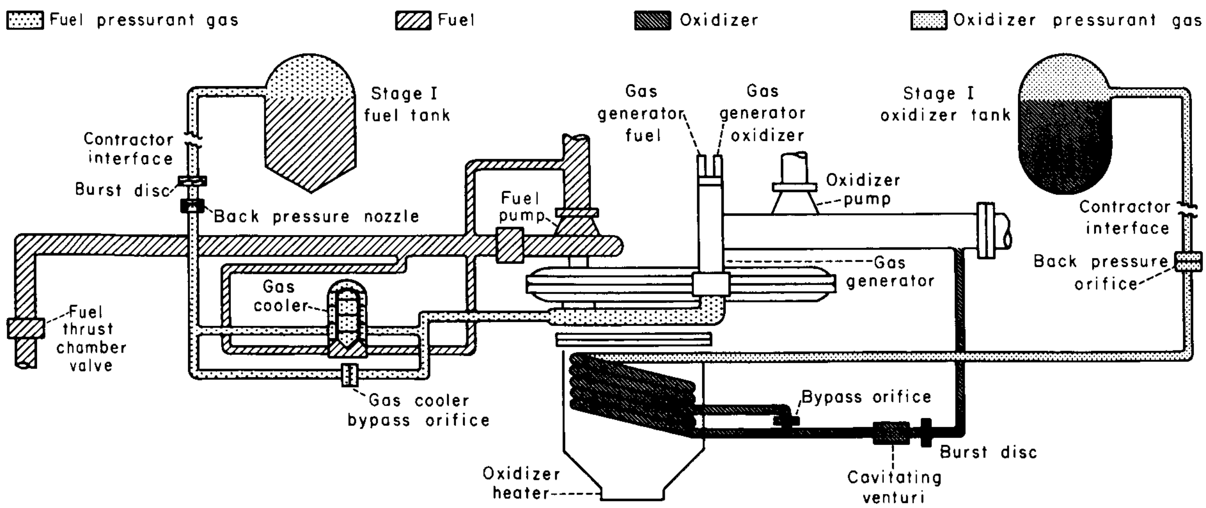
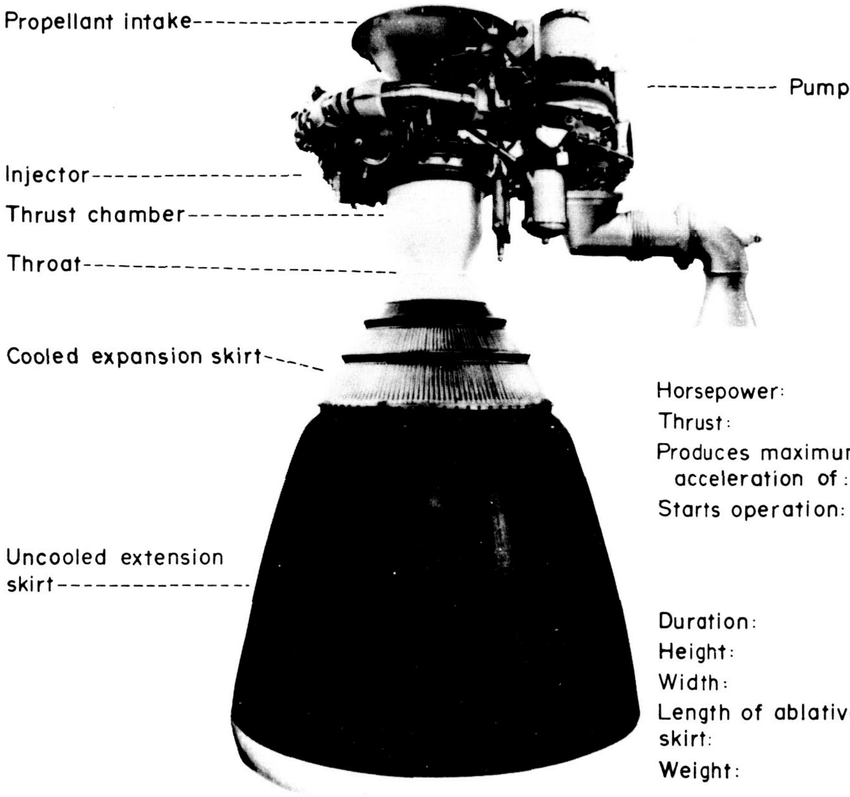


FIGURE 13-3.—Stage 1 autogenous pressurization system.



Horsepower:	4,000,000 maximum
Thrust:	100,000 lb
Produces maximum acceleration of:	7g's at 18,000 mph
Starts operation:	Some 45 miles up and 50 miles downrange while traveling more than 6,000 mph
Duration:	More than 3 minutes
Height:	9 ft 2 in
Width:	5 ft 8.5 in
Length of ablative skirt:	4 ft 7 in
Weight:	More than 1,000 lb

FIGURE 13-4.—U.S. Air Force second-stage space-start engine for Gemini Program.

cuit. These switches monitor the engine system pressures, which are a direct function of engine performance level. In the event of an engine performance decay or termination, the engine system pressure level would also decay and cause the switches to complete the electrical circuitry to the spacecraft light. Reliability of operation is increased through the use of redundant malfunction detection system switches on each thrust chamber. Both malfunction detection system switches on a given thrust chamber must close to complete the electrical circuitry.

Prelaunch Malfunction Detection System

The stage I engine supplies the pressurizing gas for the oxidizer and fuel propellant tanks, and a prelaunch malfunction detection system was developed to monitor the proper operation

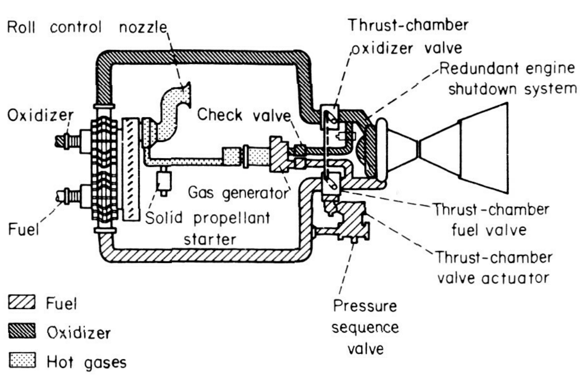


FIGURE 13-5.—Gemini stage II engine schematic.

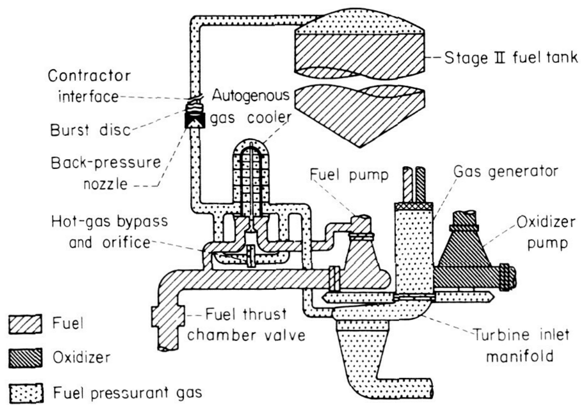


FIGURE 13-6.—Stage II autogenous pressurization system.

of these systems prior to lift-off. The prelaunch malfunction detection system consists of pressure switches installed in the oxidizer and fuel tank pressurization lines. The actuation of these switches during the engine start transient verifies that the stage I oxidizer and fuel tank pressurization gas flow is satisfactory. These switches are monitored prior to lift-off and must actuate before lift-off can occur.

Gemini Stability-Improvement-Program Injector

As a result of a NASA/Department of Defense requirement to develop a stage II injector for the Gemini Program that would have an even higher reliability than the Titan II injector configuration, the Gemini stability improvement program evolved. This program brought forth significant advances in the knowledge of liquid rocket engine combustion stability and has resulted in the development of an injector which fulfills the requirements of dynamic stability, while maintaining the performance of the Titan II and Gemini model specifications. The injector is considered to be dynamically stable, as a result of having met all of the predetermined program objectives defining dynamic stability. The injector design, using cooled-tip ejecting baffles, was developed through extensive thrust-chamber assembly and engine testing, and has been incorporated in the stage II engines on Gemini launch vehicles 8 through 12.

Redundant Engine Shutdown System

A redundant engine shutdown system was developed for the stage II engine in order to assure engine cutoff in the event of a malfunction of the primary shutdown system. To assure engine cutoff, the system terminates the oxidizer flow to the gas generator, concurrent with the normal signal that closes the thrust-chamber valves.

Other Changes

The instrumentation system was changed from a 40-millivolt system to a 5-volt system to provide better data and performance resolution. The stage I engine frame was redesigned to accommodate tandem hydraulic actuators. Selected components of the stage I engine system that are susceptible to fire damage have fire protection insulation which gives protection, during flight, from external temperatures up to 3600° F.

Qualification and Demonstration Test Program

Each of the redesigned systems successfully met their component qualification and flight certification requirements. In addition, a Gemini propulsion system test program and a Titan II piggyback flight test program were conducted. The propulsion system test program was devised to evaluate and demonstrate satisfactory operation of the Gemini unique components and requirements for the stage I and II propulsion systems. The test program was conducted on special test stands in Sacramento, whose "battleship" tankage simulates the flight vehicle. The program was successfully concluded during the early part of 1964.

The Titan II piggyback flight test program was a Titan II flight test demonstration of the malfunction detection system and prelaunch malfunction detection system. This program demonstrated the satisfactory operation of these components under a flight environment prior to a Gemini launch.

In addition to these hardware changes, further action was taken in the areas of reliability and quality in an effort to achieve the 100-percent success goal. Among the most noteworthy of these actions was the implementation of a pilot safety program.

Pilot Safety Program

The Gemini pilot safety program was established as a management tool by the Air Force Space Systems Division and placed the responsibility for implementation and control at the Program Manager level. The objectives, controls, criteria for quality and reliability, and procedures for acceptance of Gemini launch vehicle components and engines were published in an Air Force contract exhibit in January 1963, which specified the responsibilities of the Pilot Safety Team and was the basis for establishment of the goals required for a successful Gemini Program.

The evolution of the Pilot Safety Program at the Aerojet-General Corp. in Sacramento and associated field activities was one of training personnel on the importance of the objectives, of stringent controls in the application of pilot-safety principles, and of the active participation by management in each organization of the team.

The Pilot Safety Program (fig. 13–7) is a program that strives for the quality and reliability necessary to assure the success of manned spacecraft launch systems. The Gemini Program established specific controls, responsibilities, procedures, and criteria for acceptance of the critical components and engine systems to meet and fulfill the requirements of pilot safety. The acceptance of a Gemini engine system and spare components has been accomplished by a team composed of personnel from the Aerojet-General Corp., the Air Force Space Systems Division, and the Aerospace Corp. The acceptance is based on a careful consideration of the following criteria.

The discrepancies noted during all phases of the acceptance of components and engine systems are documented, evaluated, and resolved, and corrective action is taken prior to closeout of each item. In addition, discrepancies which occur on other Titan-family engine systems and which have an impact on Gemini system reliability are evaluated and resolved as to the corrective action required for the Gemini engine system.

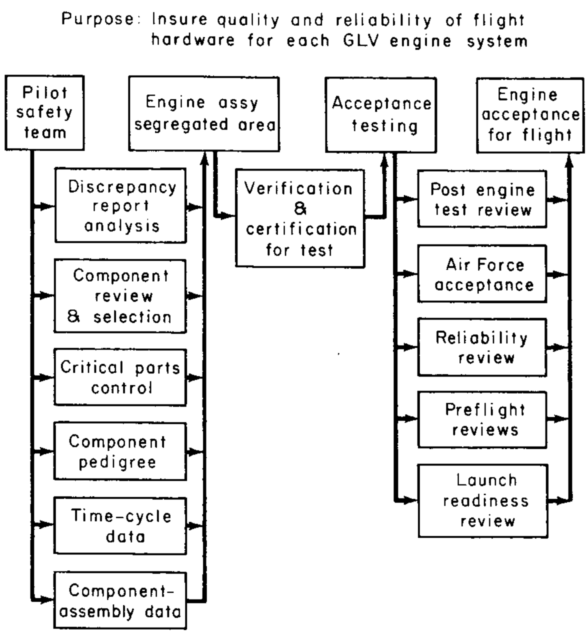


FIGURE 13–7.—Pilot Safety Program.

Each component built into a Gemini assembly and engine is reviewed, selected, and certified by the Aerojet-General Corp. pilot-safety team. All documentation applicable to the components acceptability was reviewed for assurance of proper configuration, design disclosures, and acceptability for manned flight.

A documentation packet is maintained for each critical component and assembly installed on a Gemini engine. In includes all documentation applicable to the acceptance and certification of the component to include discrepancy reports, test data, certification of material conformance, and manufacturing planning with inspection acceptance. The documentation includes certification by the Aerojet-General Corp. pilot-safety review team. The documentation packet includes a history of all rework operations at Sacramento and field sites.

A critical-components program is directed toward additional controls on 97 components of the Gemini engine which, if defective or marginal, could jeopardize the reliability or safety of a manned flight. This program includes the Aerojet-General Corp. suppliers on vendor items as well as the facilities and personnel at Sacramento and field sites. Additional components are included in the program as necessary, based on reliability studies. Containers in which spare critical components are shipped are clearly labeled "critical component." Certain critical components are sensitive to life span—primarily, accumulated hot-firing time during engine and assembly testing; therefore, a complete history of all accumulated firing time is kept on each affected component. These components receive special consideration prior to the release of an engine for flight.

Gemini critical components and engine systems were assembled in segregated controlled areas within the precision assembly and final assembly complex. Personnel assigned to the assembly and inspection operations were designated and certified for Gemini. Documents applicable to the fabrication of components were stamped "Gemini critical component" to emphasize the importance and care necessary in the processing. Approval to proceed with engine acceptance testing is withheld until the acceptance of the critical components and engine assembly are reviewed and verified by the Engine Acceptance Team. Following the acceptance test firings, all test parameters are subjected to a comprehensive review and analysis. Special emphasis is directed in the balancing of an engine to assure optimum performance and mixture ratio for successful flight operation. Hardware integrity is recertified through records review and/or physical inspection.

The engines are then presented to the Air Force, and acceptance is accomplished subsequent to a comprehensive review of the documentation. The engines are then delivered to the launch vehicle contractor's facility, where they become an integral part of the Gemini launch vehicle. After the launch vehicle is delivered to Cape Kennedy, and prior to committing the engines to launch, further reviews are conducted to evaluate the results of the launch preparation checkouts. These reviews are detailed and comprehensive and include participation by Aerojet-General Corp. top management. The engines are released for flight only after all the open items or questions are resolved.

The concept and principles of a pilot safety program can be incorporated into any space systems vehicle, if the management of the organizations involved agree to the procedures, controls, and criteria of acceptance. Specific contractual guidance, negotiation of agreements, and design requirements should be established in the development phase of a program to assure the attainment of the objectives prior to the production and delivery of a system to the Air Force. The responsibility for adherence to the requirements and procedures has to be established by top management and directed to all personnel and functions that support the program. In addition, management participation in the procedural application assures the success of the objectives and purpose of the program.

Reliability of the Gemini propulsion system has been demonstrated by seven successful launches. The reliability of the Gemini engine system is largely attributed to the pilot safety program and personnel motivation in implementing the requirements of the program throughout the entire Gemini team.

Personnel Training, Certification, and Motivation

The potential variability of the human component in system design, manufacturing, quality assurance, test, and field product support requires constant attention to achieve inherent reliability in a total system. The Gemini Program requires the highest degree of personal technical competence and complete awareness of individual responsibility for zero defects. This necessitates a training, certification, and motivation program designed and administered with substantially more attention than is usual in industry. This required—

(1) The complete and enthusiastic support and personal involvement of top management personnel.

(2) The selection, training, and certification of the company's most competent personnel to work on the program.

(3) The development of a Gemini team, each member of which is thoroughly aware of his responsibility to the total effort.

(4) Continuous attention to the maintenance and upgrading of technical competence and the motivation of each Gemini team member to devote his best to the program.

At the inception of the program, all Gemini Program personnel in the Aerojet plant at Sacramento met with an astronaut, key Air Force personnel, and company top management. Program orientation, mission, and importance were duly emphasized. Followup problem-solving meetings were held with line supervision to identify areas for special attention and to emphasize the supervisors' responsibilities with their men.

A coordinated series of technical courses was developed which permitted 218 hours of classroom and laboratory training, administered by instructors qualified by extensive experience with the engine. To qualify for a Gemini assignment, all personnel had to be certified. Certification was accomplished by extensive training and testing, using actual engine and support hardware.

Team membership and awareness of individual responsibility were continuously emphasized. The Program and Assistant Program Managers talked to all Gemini team members in small personal groups. All team members participated in program status briefings after each launch.

As the program has progressed, training has been extensively used as a means of discussing human-type problems and in reacting quickly to their solution through skill development and knowledge acquisition.

More than 1200 Gemini team members have successfully completed over 3600 courses. The courses have ranged from 1.5-hour program orientations to 40 hours for certification.

The high level of personal proficiency and pride in work attained in the Gemini training, certification, and motivation program are attested to by supervision. Since people are, in any man-machine system, the component in greatest need of constant attention, the continued high level of concern evidenced for the human factor in this program is probably the most significant single effort required for the success of the Gemini Program.

Flight Results

The successful operation of the engines on the launches of the Gemini I through VII missions is evidenced by the accuracy of the burn duration obtained versus the duration predicted, since duration is dependent upon proper operation and performance. The fraction of a percentage error in comparing the flight predictions of the engine operation with the actual operation obtained is an indicator of the high degree of repeatability of the engines.

Of interest is the unparalleled record of no engine instrumentation losses on any of the Gemini flights. There have not been any losses of telemetered engine parameters out of 206 measurements to date on the Gemini Program. This is an average of almost 30 engine parameters per vehicle.

The success of the engines on the Gemini I through VII missions is not only due to their design and simplicity of operation, but is also a result of the Air Force/contractor team effort in assuring that everything humanly possible that will enhance the chances of a perfect flight is accomplished prior to launch. The pilot-safety operation, previous flight data review, hardware certification, failure analysis program, and the primary ground rule of not flying a particular vehicle if any open problem exists to which there has not been a satisfactory explanation are all a part of the plan employed to check and doublecheck each and every item prior to flight.

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14. GEMINI LAUNCH VEHICLE GUIDANCE AND PERFORMANCE

By LEON R. BUSH, *Director, Systems and Guidance Analysis, Gemini Launch Systems Directorate, Aerospace Corp.*

Summary

This paper will review flight-test results in terms of success in meeting the overall system performance objectives of the Gemini launch vehicle program. Areas which will be discussed include guidance system development, targeting flexibility, guidance accuracy, trajectory prediction techniques, and achieved payload capability.

Introduction

The guidance system and guidance equations used for the Gemini Program are very similar to those which were used in Project Mercury. The basic guidance scheme is shown in block-diagram form in figure 14–1. The General Electric Mod III system generates rate and position data which are fed to the Burroughs computer. Pitch-and-yaw steering commands are computed in accordance with preprogramed guidance equations and transmitted to the Gemini launch vehicle in order to achieve the proper altitude and flight path angle when the required insertion velocity is reached. A discrete command is generated to initiate sustainer engine cutoff at this time.

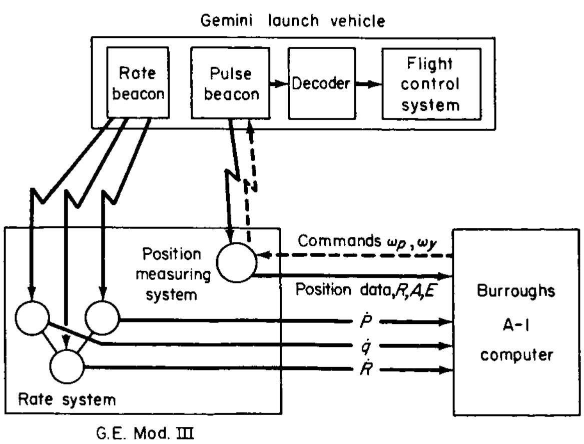

FIGURE 14–1.—Gemini launch vehicle guidance system.

Guidance System Development

Guidance system changes which are unique to Gemini have been mainly in the areas of the Burroughs computing system and auxiliary guidance equations developed by the Aerospace Corp. for targeting. The computing system was modified by the addition of a data exchange unit to provide a buffering capability for the computing system to communicate in real time with the launch facility, the spacecraft inertial guidance system, and the NASA Mission Control Center at the Manned Spacecraft Center.

A block diagram showing computer interfaces and information flow is shown in figure 14–2. Some of the unique functions which are provided include the following:

(1) Automatically receive and verify target ephemeris data from the Mission Control Center.

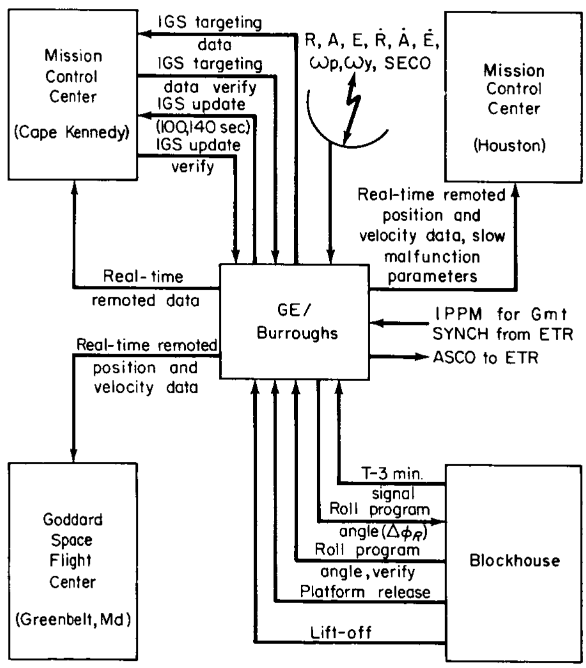

FIGURE 14–2.—RGS computer interfaces.

(2) Perform targeting computations and transfer them to the inertial guidance system for use in ascent guidance (backup mode only).

(3) Compute the required launch azimuth and transmit the corresponding roll program setting to both the block house (for the launch vehicle) and to the inertial guidance system.

(4) Transmit guidance parameters to the Mission Control Center for use in slow-malfunction monitoring.

In addition to these functions, update commands are computed and sent to the inertial guidance system during stage I flight to compensate for azimuth alinement errors in the guidance platform.

Targeting Requirements

In order to achieve rendezvous, considerable flexibility has been built into the targeting equations and procedures. A number of guidance modes have been provided such that the launch azimuth can be chosen prior to flight to allow the Gemini space vehicle to maneuver directly into the inertial plane of the target vehicle, or into a parallel plane which can be chosen to minimize maneuvering and performance loss of the launch vehicle. Logic circuitry is also provided in the computer program to insure that range safety limits and launch vehicle performance and trajectory constraints are not violated.

Flight-Test Results

From a guidance viewpoint, all launch-vehicle flights to date have been gratifyingly successful. All pretargeting and targeting computations and transmissions were performed properly. There have been no guidance hardware failures or malfunctions, and both the flight-test data analysis and comments from the flight crews indicate that guidance on all flights has been smooth and accurate, with minimal transients at guidance initiation. Except for the Gemini I mission, insertion accuracies were well below 3-sigma estimates. On Gemini I, analysis of insertion data showed sizable errors in velocity, altitude, pitch flight-path angle, and yaw velocity. Further analysis resulted in a reoptimization of guidance equation noise filters and gains, and elimination of rate-bias errors in the Mod III radar data. Analysis of the out-of-plane velocity error indicated that the spacecraft center of gravity was considerably offset from the longitudinal axis of the launch vehicle, and this induced attitude drift rates late in flight which were not sensed by the guidance system in time to make proper corrections. As a result, equations were modified to include a center-of-gravity compensator, and a V_y bias constant was added to trim out residual errors. Subsequent flight-test results indicate that these changes were quite effective in removing yaw velocity errors at insertion.

Insertion errors for all flights are shown in table 14–I. It should be noted that these errors are generally well below the 3-sigma predictions obtained by simulation. Some biases in velocity, altitude, and flight-path angle are still apparent. These have been identified with refraction errors in the Mod III rate measurement system and slight errors in prediction of stage II engine tail-off impulse. Modifications have been made to the guidance equation constants to trim these biases out for Gemini launch vehicle 8 (GLV–8) and subsequent vehicles.

Trajectory Performance

Simulation Techniques

Determination of GLV payload capability and evaluation of trajectory constraints are two critical areas in the Gemini Program. Considerable effort has, therefore, been expended by both the Martin Co. and the Aerospace Corp. to develop elaborate simulation techniques. These techniques have involved dynamic six-degree-of-freedom, multistage digital-computer programs combined with the known input parameters to develop trajectories for each specific mission. Since the Titan vehicle does not employ a propellant utilization system, outages at propellant depletion, and therefore payload capability, will be a direct function of how well the engine mixture ratios and propellant loadings are predicted. Engine models are used which take the engine acceptance test data and modify these to account for the effects of nonnominal tank pressures, propellant temperatures, and other inflight conditions. The aerodynamics used in the simulations have been derived from Titan II flight tests modified to reflect the GLV-spacecraft configuration. Dry

TABLE 14–I.—*Gemini Launch-Vehicle Insertion Accuracy*

Gemini mission	Insertion errors [a]			
	Change in total velocity, ft/sec	Change in yaw velocity, ft/sec	Change in altitude, ft	Change in pitch angle, deg
Theoretical 3-sigma dispersion	±29	±25	±2100	±0.13
I	18.5	−79.5	−2424	−0.125
II	7.5	−4.5	−1104	−.010
III	−16.9	−4.5	376	.041
IV	−13.0	0	1252	.066
V	−2.1	3.4	−583	−.008
VI–A	−11.6	−6.7	476	.050
VII	−11.0	−12.9	758	.050

[a] Downrange and crossrange position are not controlled by guidance.

weights are derived from weighings of each launch vehicle made at the factory just prior to shipment to Cape Kennedy. On recent flights, predictions have included measured pitch programer variations based on ground tests, rather than using a nominal value for all vehicles.

Once the nominal trajectory has been generated for a given mission, dispersions are then introduced to evaluate possible violation of trajectory constraints. Constraints which are carefully checked for each mission include pitch-and-yaw radar-look angles, heating and loads during first-stage flight, range safety limits, abort constraints, maximum allowable engine burning time, and acceleration and dynamic pressure at staging. Trajectory simulation results are also used to establish guidance constraints, and to determine payload capability throughout the launch window as a function of propellant temperatures and launch azimuth.

Flight-Tests Results

Analysis of the first three Gemini flights indicated that the trajectories during first-stage flight were considerably higher than the predicted nominals. This resulted in radar-look angles in pitch which were also considerably dispersed from nominal. Further investigation indicated that the basic cause of these dispersions was an apparent bias in vehicle thrust and specific impulse prediction.

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Analysis of vehicle performance at the Aerospace Corp. was accomplished using the best estimate of engine parameters, as shown in the block diagram of figure 14–3. This technique uses engine acceptance data combined with measured pressures and temperatures from inflight telemetry data to compute postflight predictions of thrust and specific impulse versus time. Actual thrust and specific impulse are obtained by combining radar tracking data, meteorological data, and vehicle weights. Figure 14–4 shows the stage I thrust and specific impulse dispersions for all of the Gemini flights to date. The data have been reduced to standard inlet conditions to eliminate effects of variables such as tank pressures and propellant temperatures. Although the first three flights showed a definite positive bias in both thrust

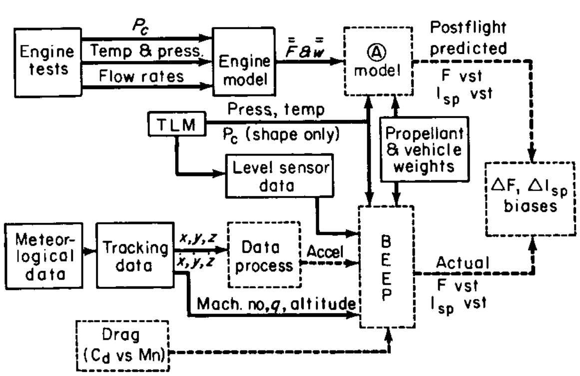

FIGURE 14–3.—Vehicle performance evaluation block diagram.

and specific impulse, the sample size was considered too small for use in determination of engine model prediction corrections. Data were therefore obtained from TRW Systems on their analyses of seven Titan II flights and carefully normalized to account for differences in prediction models. Based on this increased sample size, it was determined that the prediction models should use an increased thrust of 1.92 percent and an increased specific impulse of 1.7 seconds to provide an empirical agreement with flight-test results. This was done on Gemini launch vehicle 4 and subsequent vehicles, and it can be seen from figure 14–4 that the bias errors have been considerably reduced.

A similar technique was also used to analyze stage II engine performance. The results can be seen in figure 14–5. In this case, no bias was observed in specific impulse, but a correction of +0.9 percent in thrust was indicated.

The effect of these changes to the stage I engine model on trajectory dispersions at first-stage engine cutoff can be seen in table 14–II. Note that the altitude dispersions have been considerably reduced for GLV–4 and subsequent, and that dispersions in all parameters are considerably less than the predicted maximums.

The use of the revised engine models also led to a hardware change, in that the pitch programer rates for GLV–4 and subsequent were increased to compensate for the lofting caused by the higher stage I thrust levels.

Payload Performance

Factors Influencing Payload Capability

Many factors affect the launch vehicle payload capability. Some of these are mission oriented, such as requirements on insertion velocity and altitude, launch azimuth, and amount of yaw steering required to achieve insertion in the required target plane. Other factors are characteristics of the launch vehicle subsystems, including engine thrust and specific impulse, vehicle dry weight, loadable propellant volumes, and pitch programer rates. Finally, there are those factors due to external causes such as winds, air density, and propellant temperatures.

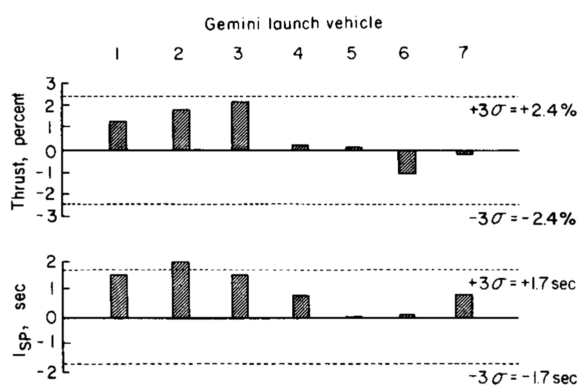

FIGURE 14–4.—Gemini launch vehicle stage I engine dispersions (normalized to standard inlet conditions).

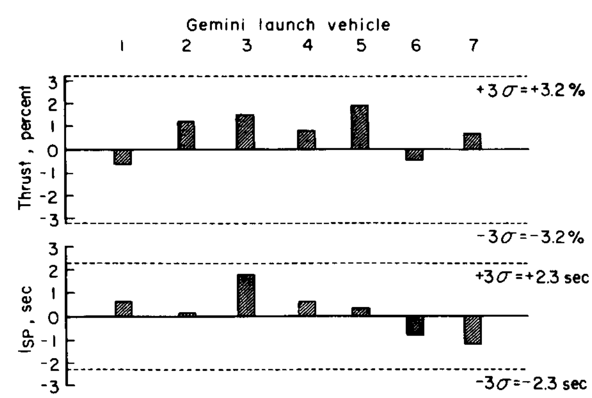

FIGURE 14–5.—Gemini launch vehicle stage II engine dispersions (normalized to standard inlet conditions).

TABLE 14–II.—*Trajectory Dispersions at Booster Engine Cutoff*

Parameter	3-sigma predicted dispersion	Dispersion (actual−predicted), for Gemini missions—						
		I	II	III	IV	V	VI–A [a]	VII [a]
Altitude, ft	±13 226	−580	12 742	14 637	6413	4765	453	3383
Velocity, ft/sec	±192	−58	154	95	−78	−153	−30	125
Flight path angle, deg	±2. 51	−0. 40	0. 69	1. 73	1. 11	0. 90	−0. 64	−0. 42
Burning time, sec	±4. 6	0. 7	−1. 8	−1. 7	−1. 0	−1. 3	0. 83	0. 16

[a] Preliminary.

Dispersions in all these factors will cause corresponding dispersions in payload capability. Sensitivities to these dispersions are shown in table 14–III. As can be seen in the table, outages and engine specific impulse have the greatest influences on payload capability.

TABLE 14–III.—*Gemini Launch-Vehicle Payload Dispersion Sensitivities*

Parameter:	*3-sigma payload dispersion, lb*
Stage II outage	457
Stage II specific impulse	197
Stage I outage	187
Stage I specific impulse	121
Pitch gyro drift	109
Winds	103
Pitch programer error	96
Stage I thrust misalinement	89
Stage I thrust	71
Other	54

Performance Improvement Program

Since the inception of the Gemini Program, a vigorous program of payload capability improvement to meet the ever increasing requirements has been pursued. To date, this effort has resulted in a payload capability increase of over 1000 pounds, over half of which was effective prior to the GLV–1 launch. A summary of the significant improvement items is shown in table 14–IV. A special engine-start test program, and analysis of structural loads and abort considerations permitted loading of additional propellants to reduced ullages, thereby increasing payload capability by 330 pounds. Redesign of telemetry and other equipment and removal of parts formerly used on Titan II and not needed for Gemini resulted in payload gains of 130 pounds. Propellant temperature-conditioning equipment was installed at Cape Kennedy to allow chilling of propellants prior to loading. This allowed a greater mass to be loaded for a given volume and resulted in a payload capability increase of 190 pounds. Analysis of Titan II flights indicated that it was safe to go to propellant depletion rather than have shutdown initiated by a low-level tank sensor. Removal of this function gave a payload capability increase of 180 pounds. Aerojet-General Corp. targeting of the nominal stage I engine mixture ratio at acceptance test to a value more compatible with launch vehicle tank size ratios resulted in a 50-pound increase in payload capability. Finally, the pitch program change and revised engine parameters discussed previously resulted in a combined payload capability increase of 175 pounds.

TABLE 14–IV.—*Summary of Gemini Launch-Vehicle Performance Improvements*

Parameter	Gemini launch vehicle effectivity	Payload capability increase, lb
Reduced ullages	1	330
Weight reduction	5	130
Propellant temperature conditioning	1	190
Low-level sensor removal	2	180
Engine mixture ratio optimization	5	50
Pitch program change	4	65
Revised engine model	4	110
Total increase		1055

Real-Time Performance Monitoring

Although the use of chilled propellants has greatly increased launch-vehicle payload capability, unequal heating of fuel and oxidizer tanks could result in nonnominal mixture ratios and thus have a significant effect upon outages and payload capability. Therefore, a technique was developed for predicting payload capability through the launch window by monitoring the actual temperatures during the countdown. The information flow is shown in block diagram form in figure 14–6. Prior to loading, weather

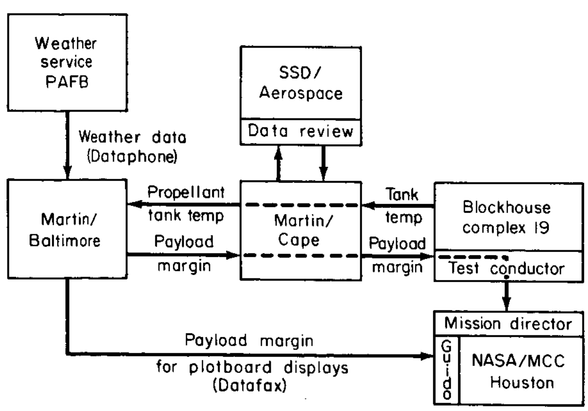

FIGURE 14–6.—Real-time performance monitoring.

predictions of ambient temperature, dew point, and winds are sent from Patrick Air Force Base to the Martin Co. in Baltimore where they are used in a computer program to predict propellant-temperature time histories from start of loading until the end of the launch window. Payload capability is also predicted as a function of time in the launch window. Once loading has been accomplished, the predictions are updated using actual measured temperatures and weather data. The final performance predictions are reviewed by the Air Force Space Systems Division and the Aerospace Corp. prior to transmission to the Mission Control Center. The Martin Co.'s program also includes the effects on payload margins of launch azimuth and yaw-steering variations through the launch window.

Typical variations of fuel and oxidizer bulk temperatures are shown in figure 14–7. As long as the temperatures remain close to the optimum mixture ratio line, the payload variations are small. If deviations in excess of 2° F occur, the payload degradation can be appreciable. Procedures at Cape Kennedy allow for some adjustment in these temperatures early in the countdown by the use of polyethylene wrap on the stage II tanks and by opening and closing of curtains at the various levels of the erector.

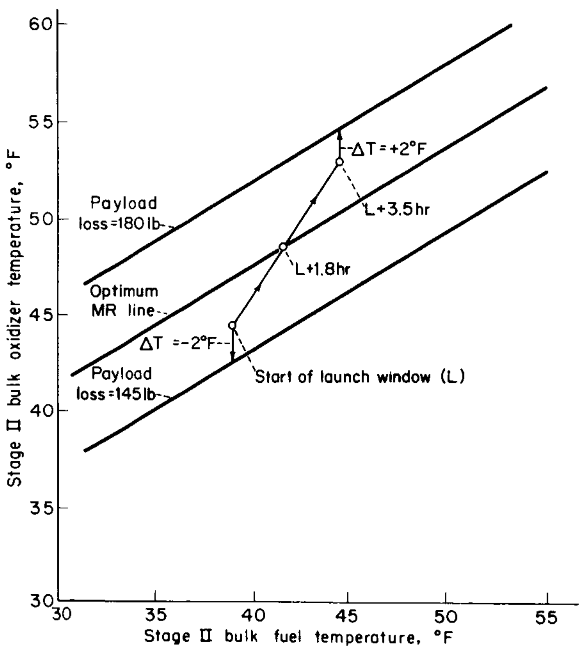

FIGURE 14–7.—Effect of differential propellant temperatures on GLV minimum payload capability.

Flight-Test Results

A summary of achieved payload capability compared to the predicted mean payload capability and 3-sigma dispersions is shown in figure 14–8. The predicted values for the Gemini I, II, and III missions have been adjusted to reflect the increased specific impulse and thrust determined from flight-test analysis. It can be seen that in all cases the actual payload capability falls very close to the mean prediction and well above the actual spacecraft weights. Table 14–V is a summary of the differences between the actual capability and predicted mean for each flight. These figures have been normalized to reflect the current prediction model. Note that the mean error is only 18 pounds higher than the predictions, and the dispersions are relatively small, indicating an extremely accurate prediction technique. Even without normalizing, the mean would be +55 pounds, with a sample standard deviation of 138 pounds. Since the dispersions about the mean are somewhat lower than the maximums predicted by theoretical analysis, current efforts are being directed toward understanding the causes of the reduced dispersions prior to their incorporation in future payload capability predictions.

TABLE 14–V.—*Gemini Launch-Vehicle Performance Dispersions From Flight-Test Analysis*

GLV:	Dispersion, pounds (achieved—predicted)
1	+41
2	−76
3	+118
4	+229
5	−152
6	−112
7	+75
Mean, lb	+18
Sample standard deviation	137
Probability=0.9987 (with 75 percent confidence)	568
Theoretical 3 sigma (probability=0.9987)	648

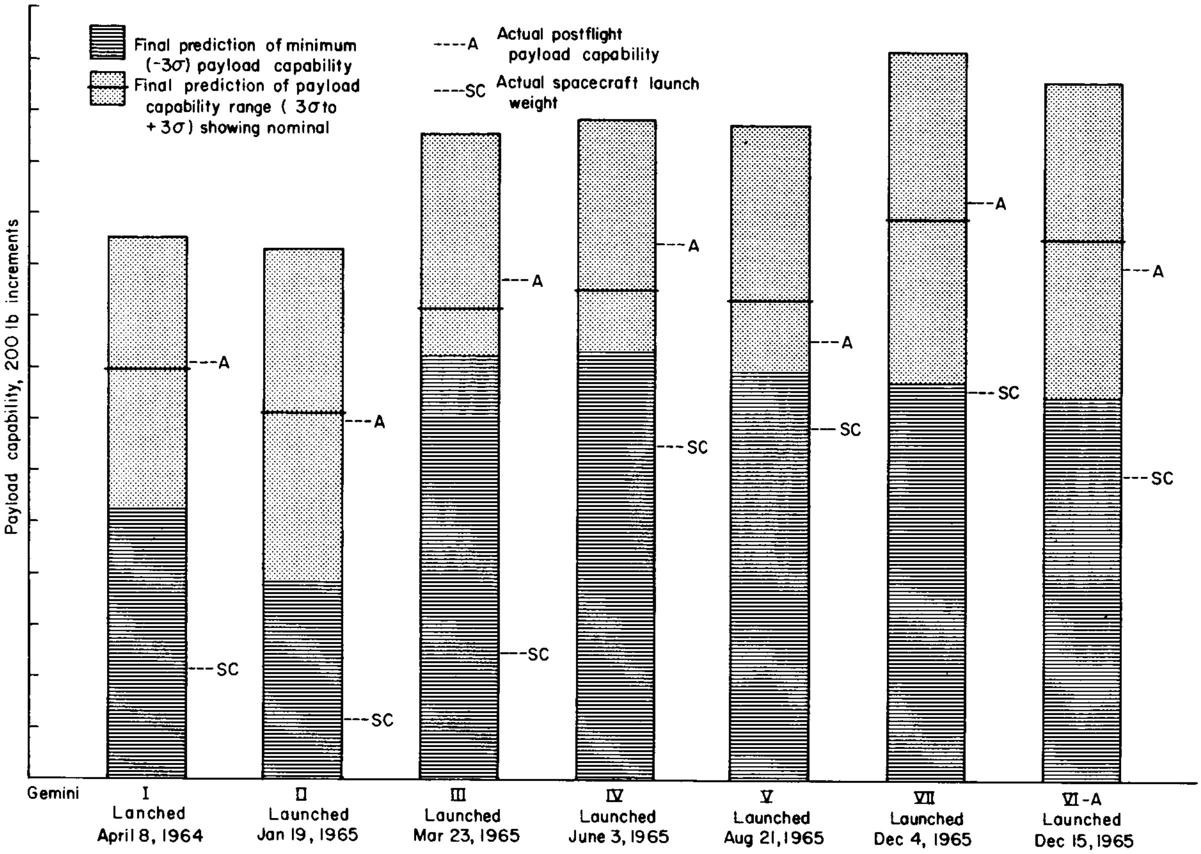

FIGURE 14-8.—Gemini launch vehicle performance history.

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15. PRODUCT ASSURANCE

By ROBERT J. GOEBEL, *Chief, Configuration Management Division, Gemini Launch Vehicle System Program Office, Space Systems Division, Air Force Systems Command*

Summary

In the Gemini-launch-vehicle program, product assurance has been achieved by (1) maximum use of failure data, (2) maximum component maturity, (3) limitation of repair and test, (4) no unexplained transient malfunctions permitted, (5) detailed review by customer, and (6) a strict configuration management policy.

Introduction

In a manned space-flight program such as Gemini, there is no questioning the need for maximum reliability, that is, maximum probability of mission success and, in the event of failure, maximum opportunity for survival of the flight crew. Actions taken in the design area to raise the inherent reliability have already been discussed. A reliability mathematical model was formulated, and from it a reliability allocation and, subsequently, reliability estimates were made. Countdown and flight-hazard analyses were used as inputs for abort studies and provided the basis for design changes aimed at reducing the probability of certain types of flight failure.

The other avenue for raising the achieved reliability of the basic Titan II was a systematic attempt to reduce the unreliability contributed by the nonconformance of people and hardware during the manufacture, test, and preparation for launch.

The word "systematic" implies judgment of what actions were consistent with the limitations and resources available to the program but which, nevertheless, promised every hope of achieving all the requirements for a manned system.

The many elements which comprise the present program stem from a set of principles and ground rules which were established at the outset. The more significant of these principles are listed below, and their purpose, application, and results are discussed.

Maximum Use of All Failure Data

Typical aircraft systems undergo thousands of hours of actual operational testing prior to being placed into service. Affording the system such a broad opportunity to fail with subsequent corrective action probably accounts for the measure of success achieved in commercial aircraft development. A system whose flight experience is recorded in minutes is at a distinct disadvantage. To broaden the data base, it is necessary to use every scrap of information from the piece part to the system level. On the Gemini Program several schemes were used to increase the amount of data available. The data bank of Titan was transferred to Gemini on microfilm and reviewed. Vendors were required to submit in-house test and failure data along with their hardware. Industrywide material deficiency alerts were and are investigated for the Gemini launch vehicle. In the design area, test equipment and aerospace ground equipment were configured to produce variable rather than attribute data, thus permitting trend analysis and data comparison.

The integrated failure-reporting and corrective-action system in use in the Gemini Program requires that every major problem be resolved prior to flight. All problems are identified by subsystem and are made the responsibility of a subsystem quality-reliability engineer for pursuit and ultimate resolution. A failed-part analysis is conducted in every case, and the postmortem is continued until the mode and cause of failure is identified. Over 1500 formal analyses have been made in the past 2½ years. Corrective action, which may involve procedural changes, test specification changes, or physical design changes, is determined and promulgated at the appropriate level. When corrective action is considered to be complete, the package is submitted to the customer for review and approval. This review includes an evaluation of the action taken to assure that the occurrence no longer represents a hazard to the Gemini launch vehicle. Only when this conclusion is reached mutually by the contractor and customer is the problem officially removed from the books. Frequently, problems occur during the last stages of test at the launch site and time may not permit the stepwise processing which is normally accomplished. In this case, the return of the failed part is expedited to a laboratory either at Baltimore, Sacramento, or the vendor's plant which has the capability to do a failed-part analysis. The engineering failure analysis is completed, establishing the mode and cause of failure, and then the flight hazard is evaluated with respect to this known condition. Frequently, it is possible to take short-term corrective action on a vehicle installed on the launch pad. This may be a one-time inspection of that vehicle, an abbreviated test of some one particular condition, or it may be that the probability of occurrence is so low that the risk is acceptable. The point is that, while final actions may not be accomplished, the problem is brought to the attention of that level of management where launch decisions can be made. This system has been extremely useful in permitting an orderly working of problems and it does present a status at any time of exactly what problems are outstanding, who is working them, and the estimated dates of resolution.

Maximum Component Maturity

The basic airworthiness of components has been established by qualification test and flight on Titan missiles. Gemini components whose environmental use was identical to Titan usage were considered qualified by similarity. All others were qualification tested. Qualification test reports were subject to review and approval by the customer. In addition, a reliability test program was established for 10 critical components which were unique to Gemini and hence had no flight history. This special testing consisted of failure mode and environmental life testing. In the first case, the test specimens are made to undergo increasingly severe levels of environment until failure occurs. In the second case, the test specimens are stressed at qualification test levels with time as the variable until failure occurs. Through an understanding of the physics of failure under these conditions, the state of maturity of these components was essentially raised to that of the other critical components. Production monitor tests are performed on 54 items. This test is part of the component acceptance requirements and consists of a vibration test at slightly less than half-qualification test levels. This has proven to be severe enough to uncover latent defects without inducing damage to the units as a result of the test. The malfunction detection system was the only subsystem which was completely new on Gemini. The piggyback program provided for flying a complete malfunction detection system, as well as several other Gemini-peculiar components, on five Titan flights. The successful completion of this program signaled the acceptability of the malfunction detection system as a subsystem for flight.

Limitation of Repair and Test

It is generally recognized that components which have undergone repeated repairs are less desirable than those which have a relatively trouble-free history. The intent was not to fly a component which had been repaired to the extent that potting compound had been removed, and connections had been soldered and resoldered a large number of times. On the other hand, it is not reasonable to scrap a very expensive piece of equipment which could be restored to service by resoldering an easily accessible broken wire. The precise definition of this idea proved to be all but impossible. The solution was to cover the subject in the quality plans as a goal rather than a requirement. The statement, "Insofar as possible, excessively repaired components will not be used on Gemini," may not be enforceable from a contractual standpoint, but it did represent mutual agreement between the contractor and the customer as a basis for internal controls.

Both operating time and vibration were recognized as influencing the probability of survival of the component during flight. Those components subject to wearout were identified together with a maximum useful operating life of each. A system of time recording was established which would pinpoint any component whose operating time would exceed its maximum allowable operating time prior to lift-off and would therefore have to be changed. The production monitor tests are essentially a vibration test at levels deliberately chosen to prevent damage. However, the integrated effect of vibration from multiple production monitor tests was considered to be deleterious and a limit of five production monitor tests was set. This control principally affected repair and modification, since a good, unmodified unit would normally be production monitor tested only once.

In some cases tests were used to determine the condition as well as the function ability of equipment. As an example, there were instances of rate-gyro spin motors failing to spin up immediately on application of power. An improved motor bearing preload manufacturing process was implemented for all new gyros. Data indicated that a correlation existed between the condition of the bearings and the time required to come up to and drop down from synchronous speed. An on-vehicle test was instituted to monitor rate gyro motor startup and rundown times, and thus provide assurance the gyro would spin up when power was applied for the next test operation or countdown.

No Unexplained Transient Malfunctions Permitted

A frequent course of action, in the face of a transient malfunction, is to retest several times and, finding normal responses each time, to charge the trouble to operator error or otherwise disregard it. A ground rule on the Gemini Program has been that a transient malfunction represented a nonconformance which would probably recur during countdown or flight at the worst possible time. Experience has shown that failure analysis of a transient in almost every case did uncover a latent defect. In those cases where the symptom cannot be repeated or the fault found, the module or subassembly within which the trouble must certainly exist is changed.

Customer Review

In order to be assured that the fabrication, test, and preparation for launch were progressing satisfactorily, Air Force Space Systems Division and Aerospace Corp. chose several key points during this cycle at which review would be conducted. These are:

(1) Engine acceptance.
(2) Tank rollout.
(3) Vehicle acceptance.
(4) Prelaunch flight-safety review.

The engine acceptance activity consists of the following sequence of events:

(1) A detailed subsystem-component review is conducted by Aerojet-General Corp. and by the Space Systems Division/Aerospace Team prior to start of engine buildup. All critical components must be approved by the review team prior to initiation of engine buildup.

(2) A detailed system review is conducted prior to acceptance firing of the assembled engine. The review team reviews the final engine buildup records and confirms the acceptability of the engine for acceptance firing.

(3) A preacceptance test meeting is conducted.

(4) Following completion of acceptance firing, a performance and posttest hardware review is conducted.

(5) A formal acceptance meeting is conducted.

The tank rollout review is aimed at determining the structural integrity and freedom from weld defects which could later result in leaks. A set of criteria which defined major repairs was first established. Stress analyses on all major repairs and also on use-as-is minor discrepancies were reviewed, and the X-rays were reread. Only after assuring that the tanks could do the job required for Gemini were they shipped to Baltimore for further buildup as a Gemini launch vehicle.

The next key point at which a customer review is conducted is at the time of acceptance of the vehicle by the Air Force. After the vehicle has undergone a series of tests (primarily several mock countdowns and flights) in the vertical facility in Baltimore, the Space Systems Division and Aerospace vehicle acceptance team meets at Baltimore for the purpose of totally reviewing the vehicle status. Principal sources of information which are used by the vehicle acceptance team are the following:

(1) Launch vehicle history.
(2) Assembly certification logs.

(3) Vertical test certification logs.
(4) Gemini problem investigation status.
(5) Subsystem verification test data.
(6) Combined system acceptance test data.
(7) Configuration tab runs.
(8) Critical component data packages.
(9) Engine logs and recap.
(10) Equipment time recording tab run.
(11) Logistic support status.
(12) Vehicle physical inspection.

The review of these data in sufficient depth to be meaningful represents a considerable task. For the first several vehicles, the team consisted of approximately 40 people and lasted 5 to 6 days. As procedures were streamlined and personnel became more familiar with the operation, the time was reduced to 4 days.

During the review of test data, every response of every system is gone over in great detail. Anomalies must be annotated with a satisfactory explanation, or the components involved must be replaced and the test rerun. After the systems tests are over and while the data are being reviewed, the vehicle is held in a bonded condition. There can be no access to the vehicle either by customer or by contractor personnel without signed permission by the resident Air Force representative at the contractor's plant. The purpose is to assure that if a retest is necessary, the vehicle is in the identical configuration as when the test data were generated. If it is not, and someone has replaced a component or adjusted a system, it may be impossible to determine the exact source and cause of an anomaly.

The customer review of Gemini problems was mentioned earlier in connection with the failure analysis and corrective action system. Those few problems which remain open at time of acceptance and do not represent a constraint to shipping the vehicle are tabulated for final action by personnel at both Baltimore and Cape Kennedy after the vehicle is shipped. It should be understood that, even though a problem may be open against a vehicle, every test required for that vehicle has been passed satisfactorily. The problems referred to may be on related systems or may represent a general weakness in a class of components, but, insofar as the individual vehicle is concerned, there is nothing detectably wrong with it. Prior to each launch, the Flight Safety Review Board takes the final look at the launch vehicle from a performance capability and a reliability standpoint. The factory history of the vehicle is reviewed again, as is its response to tests on launch complex 19 at Cape Kennedy. The contractors' representatives are asked to state the readiness of their equipment to support the mission, and at this time the vehicle is committed to launch.

Configuration Management of the Gemini Launch Vehicle

Configuration control is the systematic evaluation, coordination, approval and/or disapproval of all changes from the baseline configuration. In addition to Air Force System Command Manual (AFSCM) 375–1, Gemini Configuration Control Board Instructions, including Interface Documentation Control between associate contractors, were implemented. To insure configuration control of the launch vehicle subsequent to the first article configuration inspection of Gemini launch vehicle 1 (GLV–1), a Gemini launch-vehicle acceptance specification was implemented, requiring a formal audit of the as-built configuration of the launch vehicle against its technical description. In the area of configuration control, this formal audit consists of airborne and aerospace ground equipment compatibility status, ground equipment complete status, ship comparison status, airborne engineering change proposal/specification change-notice proposal status, ground equipment open-item status, airborne open-item status, specification compliance inspection log, Gemini configuration index, drawing change notice buy-off cards associated with new engineering change proposals, and a sample of manufacturing processes. Worthy of note is the fact that contractors' configuration accounting systems are capable of routinely supplying this body of data at each acceptance meeting.

A first-article configuration inspection was conducted on all end items of aerospace ground equipment, and equipment and facilities comprising launch complex 19. The baseline hardware consisted of 60 Aerojet-General end items, 24 General Electric end items, and 94 Martin Co. end items.

During September 1963, the Air Force Gemini Program Office conducted the first-article configuration inspection on Gemini launch vehicle 1 at the Martin Co. plant in Baltimore, Md. This is a milestone in that it represented the first instance that the first launch vehicle on a given program had been baselined prior to delivery of the item.

Subsequent to the hardware baseline, all engineering change proposals are placed before the Gemini configuration control board which is chaired by the program director. Also represented at the board meeting are engineering, operations, contracts, budget, and representatives of the Aerospace Corp. so that all facets of a change can be completely evaluated. Although all board members are afforded the opportunity to contribute to the evaluation of the proposed change, the final decision for approval or disapproval rests with the chairman. Approved changes are made directive on the contractor by contractual action. The contractor then assures that all affected drawings are changed, that the modified hardware is available and is incorporated at the proper effectivity, and that the change is verified.

Subsequent to the delivery of GLV–1, a substantial number of modifications were accomplished on the vehicle and associated aerospace ground equipment after fabrication. While this is not unusual, it is undesirable because the incorporation of modifications at Cape Kennedy was interfering with the test operations, and, in nearly every case, the work had to be done by test technicians, usually in very cramped or inaccessible places. To eliminate this problem, a vehicle standardization meeting was held by the Air Force Space Systems Division. Contractors were asked to present all known changes which were in the state of preparation or which were being considered. As a result of this forward look, it was possible to essentially freeze the configuration of the vehicle. There have been exceptions to this rule, but the number of changes dropped significantly on Gemini launch vehicle 3 and subsequent. Where necessary, time was provided in the schedule for factory modification periods. A second vertical test cell was activated and provided the capability of retesting the vehicle if modifications were incorporated after combined system acceptance test and before shipment. By comparison, 45 retrofit modifications were accomplished on GLV–1 at Cape Kennedy, and on GLV–7 there were none.

The value of configuration management to the Gemini Program is its accuracy, scope, and, above all, the speed with which it is capable of providing essential basic and detailed information for management decision, both in the normal operations of the program to assure positive, uniform control, and in emergencies when a change of plans must be evaluated quickly. Armed with a sure knowledge of status, management personnel can act with confidence in routine matters and with flexibility in urgent matters. These capabilities of modern configuration control may be illustrated specifically by events prior to the first launch attempt of the Gemini II mission. Before the first launch attempt, GLV–2 was exposed to a severe electrical storm while in its erector at the launch site. At that time, the direct substitution of GLV–3, then in vertical test at the contractor's facility, was contemplated. While this substitution was never made, the Air Force Gemini Program Office was able to identify, within 3 hours, all configuration differences between GLV–2 and GLV–3. Computer runs of released engineering, plus data packages describing changes involved in the substitution, were available for evaluation, and determination of required action was made within a total elapsed time of 5 hours. In another instance, the reprogramming of the Gemini VI–A and VII missions required the immediate determination of the compatibility of the aerospace ground equipment and launch complex 19 with the two launch vehicles. This compatibility was established overnight by computer interrogation. Months have been required to gather this kind of detailed configuration information on earlier programs. In addition to the uses mentioned previously, the methods of configuration management have been used to exercise total program control. The baseline for dollars is represented by the budget; the baseline for time is represented by the initial schedule; and for hardware, by drawings and specifications. By controlling all changes from this known posture, it has been possible to meet all of the program objectives.

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16. DEVELOPMENT OF THE GEMINI LAUNCH VEHICLE

By RICHARD C. DINEEN, *Director, Gemini Launch Vehicle System Program Office, Space Systems Division, Air Force Systems Command*

Summary

After selection of the Titan II intercontinental ballistic missile as the launch vehicle for the manned Gemini Program, NASA requested the Air Force Space Systems Division to direct the development and procurement of the Gemini launch vehicle. Ground rules specified that the modifications to the Titan II were to be minimal and should include only changes made in the interest of pilot safety, changes required to accept the Gemini spacecraft as a payload, and modifications and changes which would increase the probability of mission success. The configuration of the 11th production-model Titan II missile was used as a baseline for the Gemini launch vehicle.

Introduction

Reliability goals, failure-mode analyses, critical component searches, and other considerations, all made from the standpoint of pilot safety, had their impact in adapting the Titan II configuration to the Gemini launch vehicle. The decisions and guidance necessary to accomplish this adaptation were done through regular technical direction meetings with the contractors, and through monthly management seminars to review technical, schedule, and budgetary status. Interface between NASA and the McDonnell Aircraft Corp. was accomplished by monthly coordination meetings conducted by the Gemini Program Office. Stringent criteria were applied to all engineering investigations in order to make the best possible use of time and money.

Other management philosophies that contributed to the overall development were that the Gemini launch vehicle was to be manufactured on a separate production line, and the engines were to be manufactured as Gemini launch vehicle engines and not as a Titan II-family engine. Control of configuration, the institution of management and technical disciplines, and development of rigorous acceptance criteria were thus made possible for both the engines and the vehicle.

Most of the modifications to the Titan II were made in the interest of pilot safety, which consisted of improving the reliability of the launch vehicle through redundancy and uprating components, and coping with potential malfunctions. New criteria as well as a new system were developed to warn the crew of impending failures in their launch vehicle to permit them to make the abort decision. This malfunction detection system monitors selected parameters of vehicle performance, and displays the status of these parameters to the flight crew in the spacecraft. The redundant guidance-flight control system is automatically selected, by switchover, in the event the primary system malfunctions.

New drawings, new engineering specifications, and special procedures were developed for the total program. Strict configuration control and high-reliability goals were established at the beginning of the program. The following areas received special emphasis:

(1) Modifications to the vehicle subsystems.

(2) Pilot-safety program.

(3) Improved reliability of the vehicle.

(4) Reduction of the checkout time without degrading reliability.

(5) Evolution of guidance equations to meet Gemini requirements.

(6) Data comparison technique and the configuration-tab printout comparison used to insure that the launch of Gemini VI–A was accomplished with no degradation in reliability or no additional risk assumption.

(7) Gemini training, certification, and motivation programs.

Concluding Remarks

The excellent performance of the Gemini launch vehicle has enabled the flight crew to accomplish several important objectives including long-duration space flights and manned space rendezvous, and to perform extravehicular activity, all accompanied with a perfect safety record.

These accomplishments were climaxed by the rapid-fire launches of the Gemini VII and VI–A missions within a period of 11 days last December. This achievement was possible without a degradation in launch-vehicle reliability and without assumption of additional risks, because the Gemini-launch-vehicle program had imposed the strictest of disciplines throughout all phases of design, development, test, and launch activities. The data comparison technique was used for the launch vehicle and verified no degradation trends. It must be pointed out, however, that the short turnaround of Gemini launch vehicle 6 (GLV–6) could only be accomplished because of a thorough checkout on launch complex 19 in October 1965. The configuration of each vehicle was compared and checked against the complex by the configuration-tab printout. These techniques were also used on GLV–2 after the vehicle had been exposed to two hurricanes, and had experienced an electrical storm incident on the erector. After replacing all black-box components, the data comparison and the configuration-tab printout comparison techniques were used for assurance that the Gemini II could be safely launched.

The flight data of the seven Gemini launch vehicles launched to date have been carefully analyzed for anomalies. All systems have performed in a nominal manner, and the vehicle performance on all flights has never approached the 3-sigma-envelope outer limits. Of the 1470 instrumentation measurements taken during the 7 flights, not 1 has been lost. This is a particularly noteworthy achievement. These excellent flight results may, in general, be attributed to goals that were established for the Gemini-launch-vehicle system program at the outset.

The first of these goals is that the reliability, performance, and insertion accuracies of the launch vehicle must approach 100 percent. To date, the flight reliability of the launch vehicle is 100 percent—seven for seven. The safety margins of the launch vehicle have been maintained or improved, while the performance has improved approximately 14 percent.

The second goal is that the configuration of the launch-vehicle and test facilities must be rigidly controlled and yet retain the flexibility needed to react rapidly to program requirements. The configuration of the launch vehicle and facilities is vigorously controlled by a configuration-control board, chaired by the Program Director. By exercising strong configuration management, a first-article configuration inspection was completed on GLV–1 prior to the acceptance by the Government. The first-article configuration inspection was completed for launch complex 19 prior to the first manned launch. Configuration differences from vehicle to vehicle and engineering change effectivities are rapidly discernible by examination of the launch vehicle configuration-tab printout. Configuration management as implemented on the launch-vehicle program has guaranteed rather than hindered the capability to react immediately to changing requirements.

The third goal is that the launch vehicle to be used for manned flight must be accepted as a complete vehicle—no waivers, no shortages, no open modifications, all flight hardware fully qualified and supported with a full range of spares. The progress in achieving this goal has resulted in: no waivers on GLV–3, –5, and –6; no shortages of hardware since the delivery of GLV–2; and only one retrofit modification on GLV–5, three on GLV–6, and none on GLV–7. All flight hardware was fully qualified after the Gemini II mission. This qualification has only been possible by configuration disciplines, a realistic qualification test program, a closed-loop failure analysis system, and adequate spares inventory.

The final goal is that all personnel must be trained and motivated to achieve the 100-percent success goal. This goal is trying to disprove Murphy's law of the unavoidable mistake, but it has been demonstrated rather vividly that people and their mistakes are always present. There are procedure reviews, specialized training, and motivation to help preclude mistakes, but the fact that mistakes may occur must be recognized. The tail-plug and dust-cover incidents which occurred during the Gemini VI–A aborted launch are examples from which to learn. The philosophy of the pilot-safety program is not only to prevent mistakes, but to plan for mistakes and minimize their effect. The procedures and training have again been reviewed since the abort of the Gemini VI–A mission, and further reviews will be accomplished in the future, but it cannot be guaranteed that human mistakes will not again delay a launch. On the positive side of the ledger is the fact that planning included the systems to sense a malfunction and to prevent lift-off with a malfunctioning system.

One of the most valuable lessons of the Gemini launch-vehicle program has been that success is dependent upon the early establishment of managerial and technical disciplines throughout all phases of the program, with vigorous support of these disciplines by all echelons of management.

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FLIGHT OPERATIONS

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17. GEMINI MISSION SUPPORT DEVELOPMENT

By CHRISTOPHER C. KRAFT, JR., *Assistant Director for Flight Operations, NASA Manned Spacecraft Center, and* SIGURD SJOBERG, *Deputy Assistant Director for Flight Operations, NASA Manned Spacecraft Center*

Summary

The Gemini mission support operations have evolved from the basic concepts developed during Project Mercury. These concepts are being further developed during the Gemini Program toward the ultimate goal of supporting the Apollo lunar-landing mission.

Introduction

One of the points to be brought out during the course of this conference is that, just as Project Mercury was the forerunner to the Gemini Program, Gemini is the forerunner of the Apollo Program. Before the Gemini Program is concluded later this year, many of the flight systems and operational problems associated with the Apollo lunar-landing mission will have been explored and solved. The Gemini missions are adding to the general scientific and engineering experience in many areas, including spacecraft and launch-vehicle systems development, launch operations, flight-crew activities, and flight operations.

Mission Planning and Flight Support

To flight-operations personnel, the most important benefit of the Gemini flight program, which has already proved extremely useful in preparing for the Apollo missions, is the valuable experience that has been gained both in mission planning and in direct mission-operations activities. In particular, procedures have been developed and exercised for control of the precise inflight maneuvers required for rendezvous of two vehicles in space, and for providing ground support to missions of up to 14 days' duration. Considerable experience has been gained in the operational use of the Mission Control Center at Houston, Tex., and the tracking network, and in management of a large and widespread organization established to support the complex, worldwide mission-operations activities.

In preparing for the flight-operations support of the Gemini missions, the experience gained during Project Mercury has been very useful. Many of the basic flight-operations concepts and systems used in Project Mercury have been retained to support the Gemini and the Apollo missions. For example, the use of a worldwide network and control center involves operational concepts similar to those used in support of Project Mercury. Recovery operations are also similar, in many respects, to those developed for Mercury flights. On the other hand, there has been the requirement to augment or replace many of the original Mercury ground-support facilities and systems to meet the increased demands of the more complex Gemini and Apollo missions.

To insure maximum reliability and flexibility in the Gemini flights, it has also been necessary to expand the direct mission-support capabilities, particularly in the areas of flight dynamics and in real-time mission planning. Recovery operations have also been modified to provide maximum effective support at minimum resource expenditure.

The papers which follow will describe, in more detail, the mission support and recovery requirements and operations for the Gemini Program as they evolved through Project Mercury operational experience, and the progress we have made to date in supporting the Gemini missions. Of particular interest will be the extensive mission-planning activities and the development of the associated real-time operational computer programs. For example, the mission-planning effort is many times more extensive for a rendezvous mission than for the basic Mercury earth-orbital missions which, except for retrograde, had no inflight maneuvers.

The complexity of these activities, which stems both from consideration of operational constraints and from the capability for inflight maneuvering, ideally requires lead times of many months prior to the mission. In order to apply the experience gained from each mission to the following one, it has been necessary to provide flexibility in both the computer programs and the operational procedures for inflight control. This flexibility also provides the capability to perform real-time mission planning, which allows timely adjustments to the flight plan to accommodate eventualities as they occur during the mission.

The original Mercury Control Center at Cape Kennedy was inadequate to support the Gemini rendezvous and Apollo missions. A new mission control center was built with the necessary increased capability and flexibility and was located at the Manned Spacecraft Center, Houston, Tex. This location enhanced the contact of the flight-control people with the program offices in correlating the many aspects of mission planning to the flight systems and test programs as they were developed. The Mercury Control Center at Cape Kennedy, however, was modified to permit support of the early single-vehicle Gemini missions while the new mission control center was being implemented.

In the description of the Mission Control Center at Houston and the present tracking network, a number of innovations will be apparent. The most important innovations are: the staff support rooms, which provide support in depth to the flight-control personnel located at consoles within the mission operations control room; the simulation, checkout, and training systems, and the associated simulated remote sites, which provide the capability to conduct flight-controller training and full mission network simulations without deployment of personnel to the remote sites; and the remote-site data processors located at the network stations, which provide onsite data reduction for improved capability to perform real-time analysis of flight systems.

One of the most significant changes in the ground-support systems has been the use of automatic, high-speed processing of telemetry data, which has required a large increase in the Real Time Computer Complex. This capability, which was not available during Project Mercury, provides both control-center and flight-control personnel with selectable, detailed data in convenient engineering units for rapid, real-time analysis of flight-systems performance and status.

To the maximum extent possible, the Mission Control Center at Houston has been designed on a purely functional basis. In this manner, the data-handling and display systems are essentially independent of the program they support, and can be readily altered to support either Gemini or Apollo missions, as required.

Although the Gemini flight-control concepts are similar to those used for Project Mercury, the degree of flight-control support to the Gemini missions has not been as extensive as the support given to the Mercury missions. With increased flight experience and confidence in the performance of flight hardware, it is no longer necessary to provide the same minute-by-minute continuous support to the longer duration Gemini missions as was provided for the early Mercury missions. Extensive efforts are made, however, to insure that maximum ground support is provided during flight periods of time-critical activity, such as insertion, inflight maneuvers, retrofire, and reentry, and, of course, during the launch phase of the mission.

These activities require flight-operations support somewhat different from that for Mercury flights, in that multiple-shift operations are necessary both in the Mission Control Center and at the network stations. In general, three shifts of operations personnel are utilized in the Mission Control Center, and two shifts support the somewhat less active operations at the remote sites. Providing this flight support to multiple-vehicle, long-duration missions on a 24-hour basis requires many more flight-control personnel than were utilized in Project Mercury. However, careful consideration is given both to limiting these requirements and to streamlining flight-control readiness preparations as much as possible.

The phase-over to the Mission Control Center at Houston was conducted in an orderly fashion over a period of several missions, prior to the rendezvous mission, and was highly successful. The performance of the hardware and software of both the Mission Control Center and the network in supporting Gemini long-duration and rendezvous missions has been very satisfactory. As might be expected in a system as complex and widespread as this, operational failures did occur, particularly during long-duration missions, but they were very minor and extremely few. For the most part, the nature of these failures was such that, with the planned backup systems, the alternate routing of communications, and the alternate operational procedures, these problems were readily corrected with essentially no interruption or degradation in mission support. This basically trouble-free communications network would not have been possible without the cooperative and effective support of the Goddard Space Flight Center and the Department of Defense in developing the network and in managing its operation during mission periods.

Concluding Remarks

With the success of each mission, it becomes increasingly apparent that the flight-operations objectives of the Gemini Program are being fulfilled. The knowledge and experience in mission analysis and planning and in computer-program development and checkout are continuously expanding. Experience is increasing in the operation of the Mission Control Center and the network, and in the exercise of flight-control functions in support of increasingly more complex space-flight missions. This shakedown of operational systems and accumulation of flight experience continuously enhances the capability to more effectively plan for and provide support to the Apollo missions.

The performance of the total Government-industry organization involved in flight operations has been completely satisfactory. The mission-support preparations prior to each launch have been accomplished effectively. In particular, the concerted response by the entire team to the operational problems associated with the rapid preparations for the Gemini VII and VI–A missions in December 1965 and the unqualified success of these missions attest to the professional competence and personal diligence of the team.

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18. MISSION PLANNING

By WYENDELL B. EVANS, *Gemini Program Office, NASA Manned Spacecraft Center;* HOWARD W. TINDALL, JR., *Assistant Chief, Mission Planning and Analysis Division, NASA Manned Spacecraft Center;* HELMUT A. KUEHNEL, *Flight Crew Support Division, NASA Manned Spacecraft Center; and* ALFRED A. BISHOP, *Gemini Program Office, NASA Manned Spacecraft Center*

Summary

Project Mercury was a focal point for the development of the types of mission-planning techniques that are being used in the Gemini Program. The philosophies, mission-design requirements, and constraints used for Gemini follow, in many cases, the pattern established in the Mercury Program. This effort, in turn, will contribute directly to the Apollo and future space programs. The inclusion of the orbital attitude and maneuver system, the inertial guidance system, and the fuel-cell power system in the Gemini spacecraft provides a tremendous amount of flexibility in the types of missions that can be designed. This flexibility has required the development of a mission-planning effort which exceeds that required for Mercury missions by several orders of magnitude.

Introduction

The mission-planning activities for the Gemini Program can be categorized into four basic phases. First, the mission-design requirements were developed. These requirements influenced the systems configuration of the Gemini spacecraft and the modifications required for the target and launch vehicles. Second, design reference missions were established, which permitted the development of hardware specifications. Third, operational mission plans were developed for each flight, along with the formulation of mission logic in the ground control complex. This permits the fourth phase, real-time mission planning, to be used as circumstances require during a specific flight.

Mission-Planning Phases

Development of Mission-Design Objectives

In Gemini as in other space programs, launch vehicle performance has had a major influence on the design of the spacecraft and the development of mission plans. For example, early analyses showed that, due to spacecraft weight limitations, a source of electrical power lighter in weight than silver-zinc batteries was necessary for the long-duration missions. These analyses established the requirement for the development of a fuel-cell power system and influenced an early decision to plan the rendezvous missions for 2-day durations so they could be accomplished using battery power, should problems occur in fuel-cell development.

To satisfy the rendezvous objective, analyses established the requirement for the development of several new systems, including the radar, the digital command system, the inertial guidance system, and the orbital attitude and maneuver system.

The rendezvous objective required extensive analyses to establish the spacecraft maneuvering requirements and to optimize the launch window, orbit inclination, and target orbit altitude. In these analyses, the control of the out-of-plane displacement was a prime consideration.

Selecting a target orbit inclination that is slightly above the latitude of the launch site makes the out-of-plane displacement reasonably small for a relatively long period of time (fig. 18–1). By varying the launch azimuth so that the spacecraft is inserted parallel to the target-vehicle orbit plane, the out-of-plane displacement of the launch site at the time of launch becomes the maximum out-of-plane displacement between the two orbit planes. This variable launch-azimuth technique may also be used with guidance in yaw during second-stage powered flight to minimize the out-of-plane displacement. This is accomplished by biasing the launch azimuth of the spacecraft so that the launch azimuth is an optimum angle directed

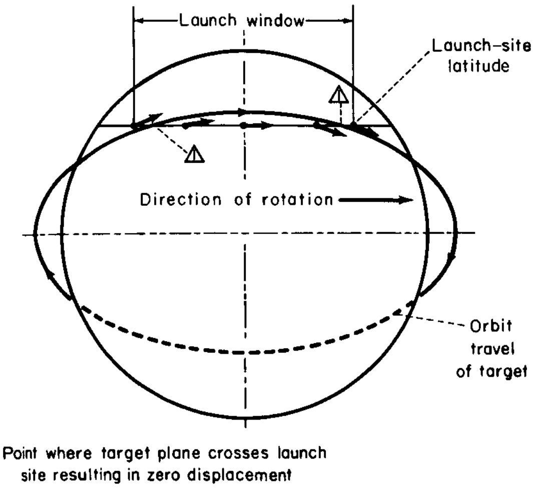


FIGURE 18-1.—Variable-azimuth launch technique.

toward the target-vehicle orbit plane. As a result, the out-of-plane distance is reduced prior to the initiation of closed-loop guidance during second-stage flight. The use of this technique is an effective way of using the launch-vehicle performance capability to control an out-of-plane displacement. However, since this technique requires additional launch-vehicle performance, a decision was made to also allocate spacecraft propellant for the correction of an out-of-plane displacement.

Analysis of launch vehicle insertion dispersions, ground tracking dispersions, and spacecraft inertial guidance dispersions established the spacecraft orbital-attitude-and-maneuver-system propellant-tankage requirement for rendezvous at 700 pounds, of which 225 pounds was allocated for an out-of-plane displacement correction. This amount of propellant would allow the spacecraft to correct an out-of-plane displacement of up to approximately 0.53°.

Launch times must be chosen so that the magnitude of the out-of-plane displacement does not exceed the spacecraft or launch-vehicle performance capabilities. By selecting an inclination of 28.87°, which is 0.53° above the launch-site latitude, and by using a variable-azimuth launch technique, the out-of-plane displacement can be controlled to within 0.53° for 135 minutes (fig. 18-2). With a maximum acceptable displacement of 0.53°, increasing the inclination to 30° reduces the plane window from one 135-minute window to two 33-minute windows (fig. 18-3). From these two curves it can be seen

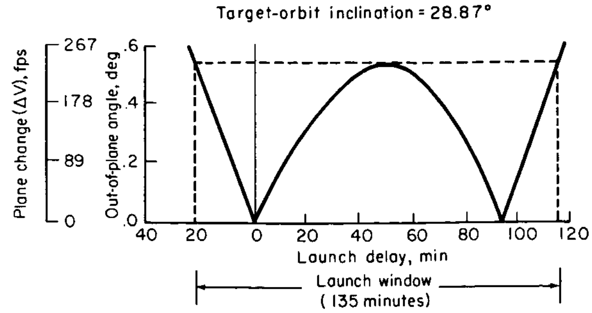


FIGURE 18-2.—Variable-azimuth launch window for target orbit inclination of 28.87°.

that the quantity of propellant required to provide a launch window of a given duration is very sensitive to target orbit inclination. With a maximum acceptable out-of-plane displacement of 0.53°, a target inclination of 28.87°, and a fixed-azimuth launch, the plane window is reduced to 17 minutes (fig. 18-4). The results of these analyses established the requirement to implement a variable launch azimuth guidance capability in both the spacecraft and launch vehicle and to establish the target orbit inclination at 28.87°.

The next parameter to be considered in this phase of mission planning was the desired orbit altitude for the rendezvous target vehicle. A near-optimum altitude would provide a zero phasing error simultaneously with the zero out-of-plane displacement near the beginning of the launch window on a once-per-day basis. This near-optimum condition for a target inclination of 28.87° occurs on a once-per-day basis at 99, 260, and 442 nautical miles. Because of launch-vehicle performance, the 260- and 442-nautical-

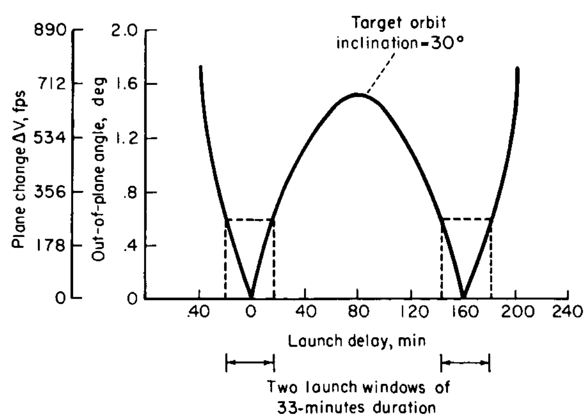


FIGURE 18-3.—Variable-azimuth launch windows for target orbit inclination of 30°.

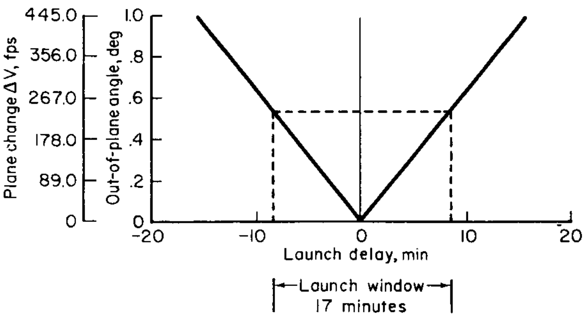


FIGURE 18-4.—Fixed-azimuth launch window for target orbit inclination of 28.87°.

mile orbits were not considered. The 99-nautical-mile orbit was not considered because of the relatively short lifetime of this orbit. Other altitudes—125, 150, 160, and 175 nautical miles—were evaluated. A rendezvous target orbit altitude of 161 nautical miles was selected. This altitude provided launch opportunities with zero phasing errors within the 135-minute launch window on a once-per-day basis, and provided near-optimum phasing conditions for the second day (fig. 18-5). The decision to select this altitude had an influence on the retro-rocket systems design and on the thermodynamic design of the spacecraft, the target vehicle, and the target docking adapter.

The selection of the Gemini insertion altitude was influenced by the launch-vehicle radio-guidance-system accuracies which are a function of the elevation angle at sustainer engine cutoff, of the spacecraft and the launch-vehicle second-stage exit-heating requirements, and of the launch vehicle performance capability. Based on an evaluation of these factors, an altitude of 87 nautical miles was established for the design requirement.

Establishment of Design Reference Missions

After the mission-design requirements were developed for the spacecraft, for the target vehicle, and for the launch vehicles, three basic types of design-reference missions were specified so that hardware development plans could be established for the airborne and ground systems. These types of mission were (1) unmanned ballistic for systems and heat protection qualification, (2) manned orbital 14-day with closed-loop guidance reentry, and (3) manned orbital rendezvous and docking with closed-loop guidance reentry. It is important to note that within

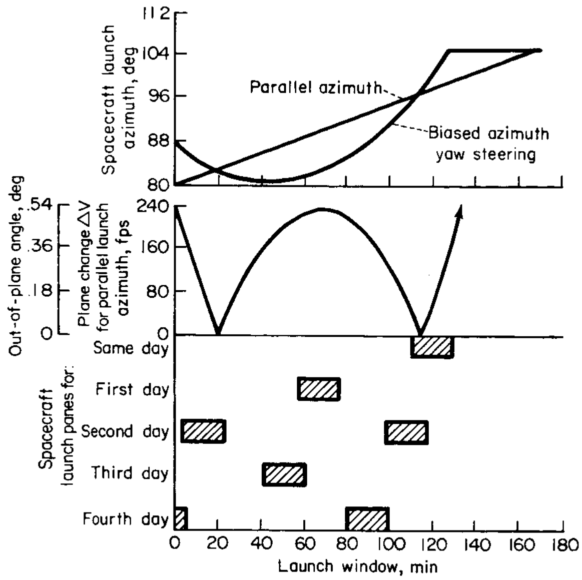


FIGURE 18-5.—Space-vehicle launch windows for rendezvous target orbit altitude of 161 nautical miles.

the framework of the long-duration and rendezvous missions, many other objectives can be accomplished, such as extravehicular activity and experiments.

Development of Operational Mission Plans

In the development of the detailed operational mission plans to satisfy the program objectives, the requirement has been to insure the highest probability of success by minimizing, within the limits of practicality, any degradation of the mission objectives resulting from systems failures or operational limitations. To accomplish this requirement, operational mission plans were developed which provided a logical buildup in the program objective accomplishment. The operational mission plans which were developed to accomplish this buildup are shown in table 18-I.

Qualification of the launch-vehicle and spacecraft systems was the primary objective of Gemini I and II. The objectives of Gemini III, the first manned flight, included the evaluation of spacecraft maneuvering in space, a requirement for the rendezvous missions; the qualification of the spacecraft systems to the level of confidence necessary for committing the spacecraft and crew to long-duration flight; the development of procedures necessary to conduct long-duration, rendezvous, and a closed-loop re-

TABLE 18–I.—*Operational Mission Objectives*

Objective / Mission	G–I	G–II	G–III	G–IV	G–V	G–VI	G–VII
Closed-loop reentry guidance:							
System qualification		●					
Procedure development			●				
Demonstration				○	○	○	○
EVA				●			
Long duration:							
System qualification	●	●	●		○		
Procedure development			○				
4 day				●			
8 day					●		
14 day							●
Rendezvous:							
System qualification	●	●	●	○	○		
Procedure development			○	○	○		
Rendezvous evaluation					○		
Rendezvous						●	
Experiments	0	0	3	13	17	3	20

● Primary objective.
○ Secondary objective.

entry; and the execution of three inflight experiments. The plans for Gemini IV included the first long-duration objective (4 days), extravehicular activity, further development of the rendezvous procedures, a demonstration of a closed-loop reentry, and the execution of 13 inflight experiments.

Gemini V, an 8-day flight, was the second step in the development of the long-duration capability. Other objectives planned for this flight were the final qualification of the rendezvous systems and procedures necessary for the Gemini VI mission, evaluation of the fuel-cell power system required for long-duration flights, the demonstration of the capabilities of the closed-loop reentry guidance, and the execution of 17 inflight experiments. Designating the primary objectives of the first five flights as nonrendezvous permitted the development of efficient checkout and launch procedures, a requirement for on-time launch. Early development of these procedures was mandatory to satisfy the rendezvous objective of the Gemini VI mission. The primary objective of Gemini VII, of course, was long duration (14 days). Three experiments were planned for Gemini VI and 20 experiments for Gemini VII. Plans for both of these flights included a demonstration of closed-loop reentry guidance.

The development of operational mission plans for implementing the mission objectives requires that extensive analyses be performed in the trajectory and flight-planning areas. In Gemini, detailed trajectory and flight planning has been found to be essential for mission success and must be done in such a way as to afford mission flexibility.

Trajectory Planning

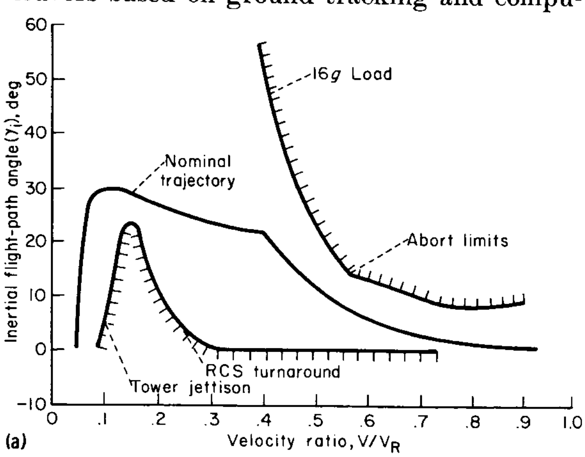
During Project Mercury, a major part of the trajectory-planning effort was spent in the development of the philosophy and techniques for monitoring the powered-flight trajectory, for determining when launch abort action was necessary, and for establishing go—no-go criteria for the acceptability of the orbit after the completion of launch-vehicle thrusting. These Mercury analyses were directly applicable to the Gemini Program. Generally, it was merely necessary to identify the most limiting trajectory criteria—that is, the trajectory conditions beyond which abort action is not safe due to such factors as exceeding spacecraft reentry heating, or aerodynamic load-design limits that were applicable to the Gemini spacecraft. The character of the resulting abort-limit lines used on the flight controller plotboards is very similar to that designed for Project Mercury (figs. 18–6(a) and 18–6(b)).

If a Mercury spacecraft failed to achieve orbit, only two possible courses of action were available: fire the retrorockets for an immediate abort, or do nothing. The maneuvering capability of the Gemini spacecraft provides a third, more desirable choice, which is using the orbital attitude and maneuver system as a third-stage propulsion system to achieve orbit (figs. 18–7(a) and 18–7(b)).

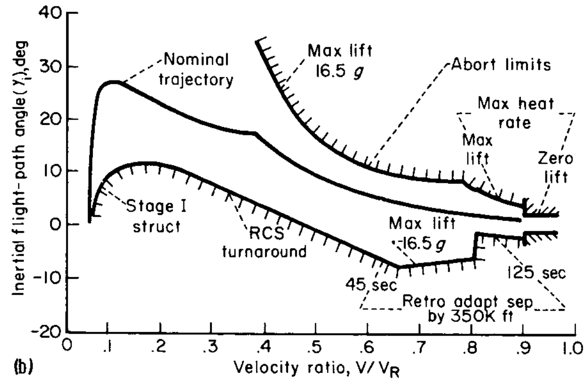
Abort actions or the use of orbital attitude and maneuver system into orbit has never been necessary; however, all possible contingency situations must have been analyzed, and corrective procedures developed.

The capabilities of the Gemini spacecraft provide a tremendous amount of flexibility in the types of missions which can be designed. This flexibility has allowed modification of mission plans both before and during an actual flight. For example, during the Gemini V mission, problems with the spacecraft electrical power system made it necessary to abandon the rendezvous evaluation pod test. The objectives of the test were accomplished, however. This was possible because mission-planning personnel conceived, planned, and set up the so-called phantom rendezvous and a spacecraft radar-to-ground transponder tracking test within a 1-day period during the 8-day flight. The phantom rendezvous, which involved a series of maneuvers based on ground tracking and computations, almost precisely duplicated the maneuvers planned for the midcourse phase of the Gemini VI flight. This series of maneuvers executed by the Gemini V flight crew were a milestone—the first in-orbit maneuvers carried out with the precision necessary for performing a space rendezvous. The near-perfect performance of the Gemini V spacecraft, flight crew, and the ground personnel verified the accuracy which could be expected during the rendezvous missions. Sufficient data were obtained from the spacecraft radar tracking test, and from the rendezvous evaluation pod test prior to its termination, to adequately flight-quality the spacecraft radar system for the Gemini VI mission.

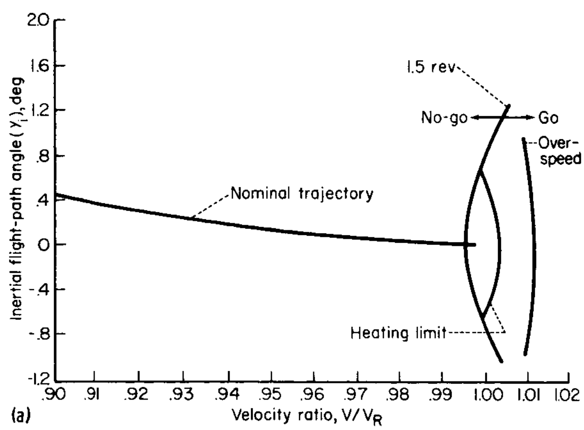
The changes made before the Gemini VII flight are well known. In order to utilize the Gemini VII spacecraft as a target for the Gemini VI-A mission, it was necessary to change the Gemini VII launch-azimuth and orbital-



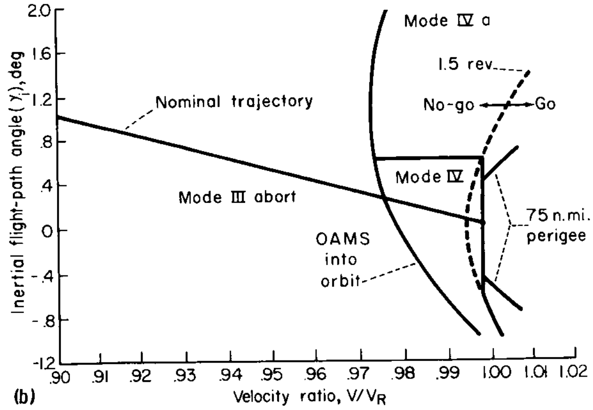
(*a*) Project Mercury.
FIGURE 18–6.—Abort limit lines launch trajectory monitoring.



(*b*) Gemini Program.
FIGURE 18–6.—Concluded.



(*a*) Project Mercury.
FIGURE 18–7.—Go—no-go criteria for acceptability of orbit after completion of thrust by launch vehicle.



(*b*) Gemini Program.
FIGURE 18–7.—Concluded.

insertion requirements. In addition, a radar transponder and acquisition lights were installed on spacecraft 7, and logic and computer programs were developed for selecting the Gemini VII in-orbit maneuvers required to arrive at the optimum conditions for rendezvous with a minimum expenditure of fuel. This was all accomplished within a 6-week period after the first Gemini VI launch attempt. It is interesting that, except for the development of a quick turnaround capability, the plan for Gemini VI–A was relatively unchanged. In fact, since the Gemini VII spacecraft was maneuvered precisely to the planned orbital inclination of 28.87° and altitude of 161 nautical miles, the Gemini VI–A mission was accomplished almost exactly as planned.

The point to be made here is that, to get the most out of each Gemini flight, the capability must exist to allow rapid response to changes in mission requirements. To provide this capability, a staff of experienced personnel must have carried out a wide variety of analyses and studies upon which they can quickly draw, both before and during the actual mission.

Flight planning.—The term "flight planning," as used in manned space flight, is the development of a schedule of inflight crew activities. Such a plan is required to insure that the most effective use is made of flight time. Detailed flight planning starts after mission objectives have been clearly defined and the trajectory profile has been established. The first task is to determine the exact operational procedures that are necessary to accomplish each of the mission activities. Operational procedures are developed by careful analyses and simulations. These analyses and simulations also establish the time, propellant, and electrical power that are required to accomplish each task. With these results, flight planning personnel can then establish the total quantity of consumables—propellant, electrical power, oxygen, food, and water—that will be necessary for a specific mission.

When all of the details of each mission have been worked out, plans for accomplishing the mission are documented in a flight plan. The flight plan provides a detailed schedule of the flight-crew and ground-station activities, checklists for normal and emergency procedures, a detailed procedure for conducting each planned activity, consumables allocations and nominal-usage charts, and an abbreviated schedule showing major events to be conducted throughout the flight. Figures 18–8(a) and 18–8(b) are samples of the detailed flight plan for the Gemini VII mission during the period from the lift-off through launch vehicle staging. Figure 18–9 is a sample of the abbreviated flight plan during the period from lift-off through the first 4 hours of flight, and figures 18–10(a) and 18–10(b) are examples of the procedures section showing the propellant usage summary and an operational test description.

The contents of the flight plan vary according to the mission. For example, for the Gemini VII flight, the detailed plan was written only through the launch vehicle station-keeping period because the remainder of the 14-day flight was preplanned to be conducted in real time. This approach was unique since, on previous missions, the complete flight plan was developed prior to launch, and real-time planning was adopted only when inflight anomalies occurred. On the Gemini VII mission, premission planning was oriented toward a general sequencing of the tests and experiments required in the flight in order to establish the required timelines. Detailed procedures for each crew activity were established for crew training; therefore, a majority of the real-time effort consisted of scheduling each activity. On Gemini VII this procedure proved to be quite satisfactory, and all objectives were accomplished except where equipment failure or the weather precluded completion of some activities.

Real-Time Mission Planning

Development of the mission design requirements, the operational mission plans, and documentation as previously mentioned is only part of the overall mission planning task. The next step is to make the plan work. This depends to a great extent on whether the launch vehicle and spacecraft perform as predicted. When an abnormal situation does arise, as during Gemini V, the planned activities must be rescheduled and, in some cases, compromised to make maximum use of the systems performance as it exists.

The necessity of being prepared to handle whatever contingency develops as the mission progresses has led to the development of a highly sophisticated and complex real-time flight-control computer program.

(a)

TIME HR:MIN:SEC	COMP	PLAT	CNTL MODE	ACTION COMMAND PILOT	ACTION PILOT
0:00:00	ASC	FREE		CNV-REPORT LIFT-OFF A-REPORT CLOCK START (EVENT TIMER)	
0:00:19	ASC	FREE		A-REPORT ROLL PROGRAM INITIATED	
0:00:20	ASC	FREE		A-REPORT ROLL PROGRAM COMPLETE	
0:00:23	ASC	FREE		A-REPORT PITCH PROGRAM INITIATED	
0:00:50	ASC	FREE		CNV-GIVE 50 SEC TIME-HACK FOR CHANGE TO DELAYED-LAUNCH MODE II	
				A-CONFIRM REPORTED CHANGE TO DELAYED-LAUNCH MODE II RELEASE 'D'-RING	A-RELEASE 'D' RING. UNCLIP KEYING SWITCH
				NOTE 'D'-RING STOWED AFTER INSERTION. CMD PILOT WILL USE THE KEYING SWITCH ON THE HAND CONTROLER	

MISSION	EDITION	DATE	STATION	AOS	LOS	TOTAL	REV	PAGE
GEMINI VII	FINAL	11/15/65	CNV	0:00:00	0:06:57	6:57	LAUNCH	1

(*a*) Lift-off through first 50 seconds.
FIGURE 18-8.—Example of detailed flight plan.

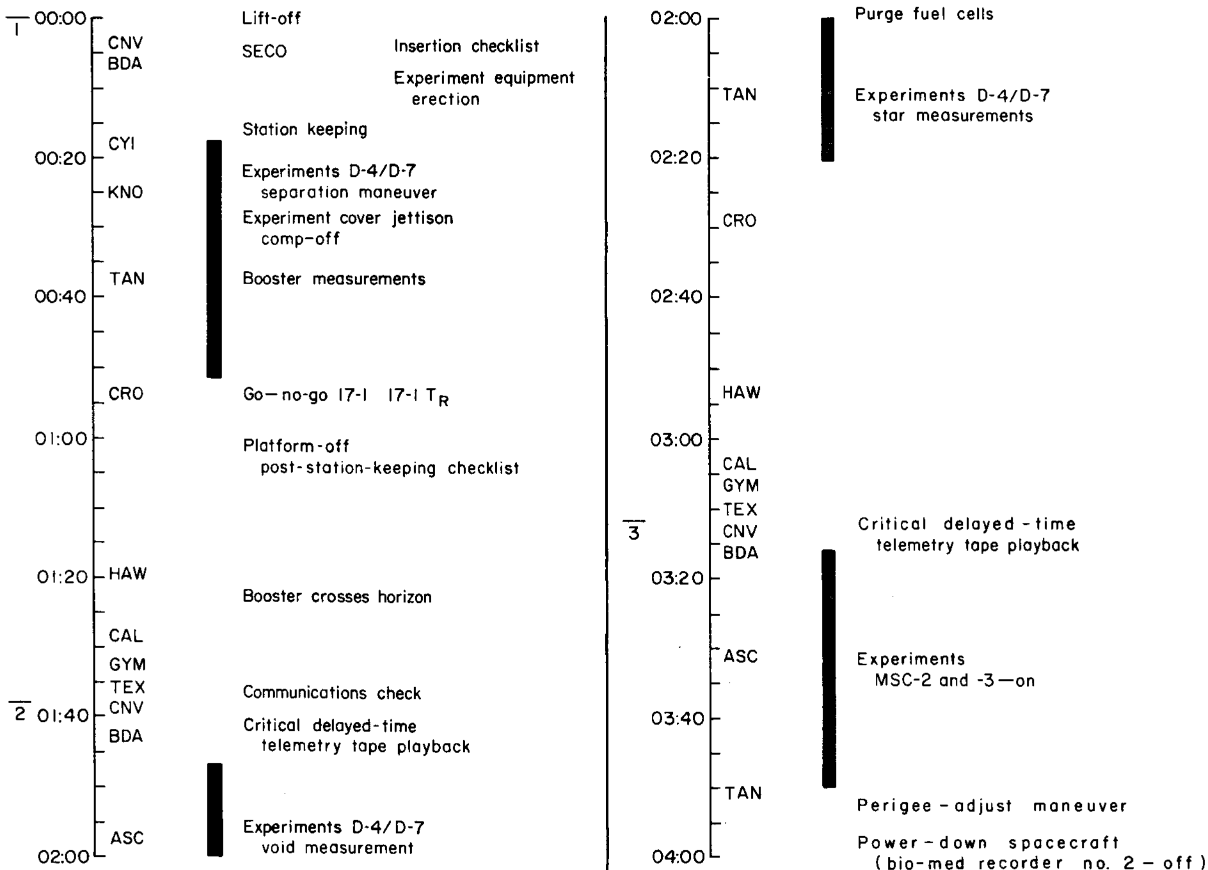
(b)

TIME HR:MIN:SEC	COMP	PLAT	CNTL MODE	ACTION	
				COMMAND PILOT	PILOT
0:01:00	ASC	FREE			A-REPORT CABIN PRESSURE HOLDING AT___PSID
0:01:40	ASC	FREE		CNV-REPORT CHANGE TO LAUNCH MODE II (70K FT) A-CONFIRM REPORTED CHANGE TO LAUNCH MODE II	
0:01:45	ASC	FREE			A-RESET DCS LIGHT. REPORT DCS UPDATE RECEIVED
0:02:15	ASC	FREE		A-REPORT STAGE II GO	
0:02:25	ASC	FREE			A-RESET DCS LIGHT. REPORT DCS UPDATE RECEIVED
0:02:35	ASC	FREE		STAGING NOTE ENGINE I LIGHTS-FLICKER ENGINE II LIGHT-OUT	
				A-REPORT STAGING STATUS CHECK 'G'-LEVEL FDI SCALE RANGE-HI	

MISSION	EDITION	DATE	STATION	AOS	LOS	TOTAL	REV	PAGE
GEMINI VII	FINAL	11/15/65	CNV	0:00:00	0:06:57	6:57	LAUNCH	2

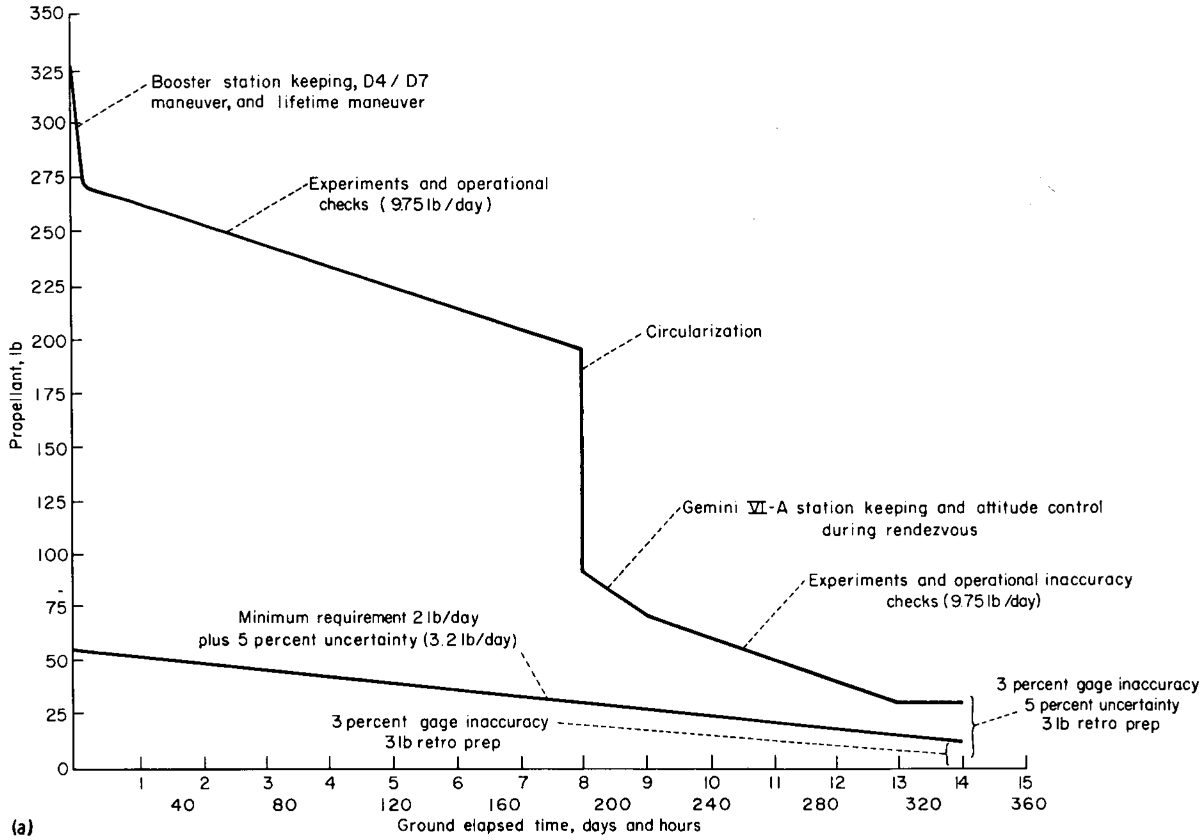
(*b*) One minute through 2 minutes 35 seconds after lift-off.

FIGURE 18-8.—Concluded.



Mission	Edition	Date	Time	Revolution	Page
Gemini VII	Final	November 15, 1965	00:00 to 04:00	1-3	1

FIGURE 18–9.—Example of abbreviated flight plan.



(*a*) **Estimated propellant usage for Gemini VII mission.**

FIGURE 18–10.—Example of procedures section of the flight plan.

(b)

RADAR TRANSPONDER TEST

Purpose

To verify calculated warm-up and cool-down curves for the transponder and as an operational check.

Spacecraft Systems Configuration

1. Reticle installed (for operational check)

2. AC POWER - ACME

3. ATTITUDE CONTROL - PULSE

Procedure

1. Temperature Check
TRANSPONDER - ON AT AOS
TRANSPONDER - OFF AT LOS

Note: 1. Check temperature every 12 hrs until temperature stabilizes, then every 24 hours.
2. Ground will monitor and plot the temperature trend.

2. Operational Check
TRANSPONDER - ON
Align spacecraft on radar located at Cape Kennedy.
TRANSPONDER - OFF after LOS.

Note: The operation check will be conducted on passes which occur at approximately VII lift-off plus 48 hours and VI-A lift-off minus 72 hours (total of 2 runs required).

Propellant Required

2 runs x 1 lb run = 2 lb

(b) Radar transponder test.
FIGURE 18-10.—Concluded.

19. MISSION CONTROL CENTER AND NETWORK

By HENRY E. CLEMENTS, *Chief, Flight Support Division, NASA Manned Spacecraft Center;* RICHARD L. HOLT, *Flight Support Division, NASA Manned Spacecraft Center; and* DOUGLAS W. CARMICHAEL, *Flight Support Division, NASA Manned Spacecraft Center*

Summary

As planning for the Gemini Program began, the capabilities of both the Mercury Control Center at Cape Kennedy, Fla., and the Manned Space Flight Network were reviewed and found inadequate to support the Gemini rendezvous missions. A new control center with expanded facilities was required to support the Gemini missions and the advanced flight programs of the future. Major modifications to the Manned Space Flight Network were also required. Equipment used in both systems was generally off the shelf, with proven reliability. Mission results have proved both support systems to be satisfactory.

Introduction

Project Mercury established the requirement for an effective ground-control capability for unmanned and manned space flights. During the Mercury flights, a control center remotely connected to a worldwide network of tracking stations repeatedly demonstrated its speed and efficiency in reacting to the anomalies encountered.

Mercury space flights, however, involved controlling only a single vehicle with no maneuvering capability. The Gemini Program, with its multiple-vehicle rendezvous and docking maneuvers and long-duration flights, required a ground control capable of processing and reacting to a vast amount of complex data on a real-time basis. Therefore, a new control facility was established that would support the Gemini Program and the future space flight programs.

The Manned Spacecraft Center at Houston, Tex., was chosen as the site for a new mission control center to be designated "MCC–H" (fig. 19–1). However, this control center could not be placed into operation in time to support the early nonrendezvous Gemini flights. To support this phase of the Gemini Program, the facilities of the Mission Control Center (MCC–K) at Cape Kennedy, Fla., were evaluated, and it was found that, with minor modifications, they would give sufficient support.

The new mission control center was designed to effect direction and control of the Gemini flights through the Manned Space Flight Network, which is a worldwide communications, tracking, and telemetry network. This network of stations had proved its operational capabilities through the Mercury flight program but, for the more complex missions of the Gemini Program, the network would require major modifications to all of its systems. The network had to have the capability to track two vehicles simultaneously and to provide dual command data based on orbital ephemeris, orbital plane changes, rendezvous maneuvers, and reentry control to the vehicles' computers. The amount of information generated during a Gemini flight was over 40 times the amount generated and transmitted to the control center during the most complex of the Mercury flights. The primary consideration in design efforts was reliability; the ground systems would have to support long-duration flights.

Existing schedules, reliability requirements, and monetary limits required that equipment going into the new control center be of a fully developed nature, and resulted in the control center being a consolidation of off-the-shelf equipment.

The Mission Control Center at Houston was designed to perform all known control and

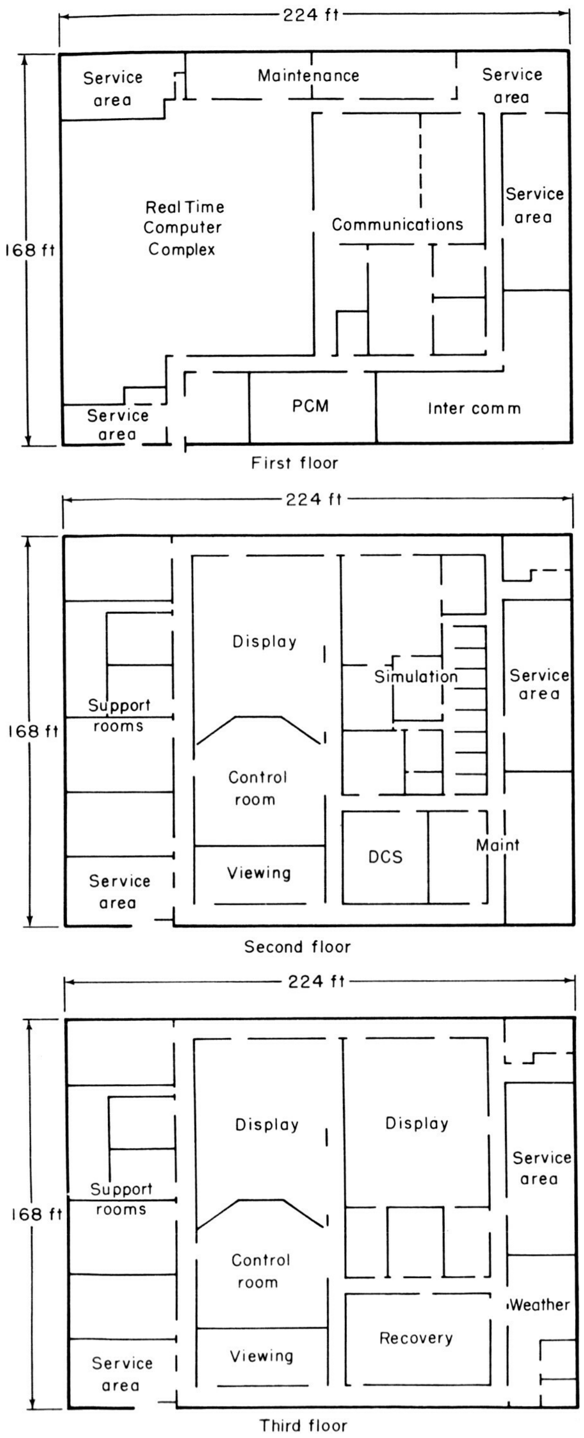


Figure 19–1.—Floor plan of Mission Control Center, Houston, Tex.

monitoring functions associated with manned space flight. The major requirements were—

(1) To direct overall mission conduct.

(2) To issue guidance parameters and to monitor guidance computations and propulsion capability.

(3) To evaluate the performance and capabilities of the space-vehicle equipment systems.

(4) To evaluate the capabilities and status of the spacecraft crew and life-support system.

(5) To direct and supervise activities of the ground-support systems.

(6) To direct recovery activities.

(7) To conduct simulation and training exercises.

(8) To schedule and regulate transmission of recorded data from sites.

(9) To support postmission analyses.

Development of Mission Control Center Equipment Systems

Real Time Computer Complex

The first three Gemini flights were controlled at the Mission Control Center at Cape Kennedy, but, as had been done during Project Mercury, the majority of real time computations were processed at the Goddard Space Flight Center (GSFC), Greenbelt, Md. The design of the Mission Control Center at Houston included a large increase in computer capacity to support actual and simulated missions. This increase was made necessary by the mounting number of mathematical computations required by the complex flight plans of the Gemini rendezvous missions.

The Real Time Computer Complex (fig. 19–2) was designed for data and display processing for actual and simulated flights. This computer complex consists of five large-capacity digital computers. These computers may be functionally assigned as a mission operations

Figure 19–2.—Real Time Computer Complex, Houston, Tex.

computer, a dynamic standby computer, a simulation operations computer, a ground support simulation computer, and a dynamic checkout computer; or they may be taken off-line and electrically isolated from the rest of the Real Time Computer Complex.

During a mission, the flight program is loaded into both a mission operations computer and a dynamic standby computer. This system allows the outputs of the computers to be switched, thus providing continued operation if the mission operations computer should fail. As the flight progresses, the vast amount of data received in the Real Time Computer Complex from the Manned Space Flight Network is translated into recognizable data displays that enable mission controllers to evaluate current mission situations and make real-time decisions.

During a mission, the remaining computers can be utilized for a follow-on mission simulation and development of a follow-on mission program.

Communications

The design of the Mission Control Center at Houston enables communications to enter and leave over commercial common-carrier lines, which are divided into five categories:

(1) Wideband data (40.8 kbps) lines handle only the transmission of telemetry data.

(2) High-speed data (2 kbps) lines carry command, tracking, and telemetry data.

(3) Teletype (100 words a minute) lines carry command, tracking, acquisition, telemetry, and textual message traffic.

(4) Video lines carry only television signals.

(5) Audio lines primarily handle voice communication between the Mission Control Center, the Manned Space Flight Network, and the spacecraft.

The Mission Control Center communications system (fig. 19–3) monitors all incoming or outgoing voice and data signals for quality; records and processes the signals as necessary; and routes them to their assigned destinations. The system is the terminus for all incoming voice communications, facsimile messages, and teletype textual messages, and it provides for voice communications within the control center. Telemetry data, routed through telemetry ground stations, are sent to the Real Time Computer Complex for data display and telemetry summary message generation. Some of the processed data, such as biomedical data, are routed directly to the display and control system for direct monitoring by flight controllers and specialists. Incoming tracking data are sent to the Real Time Computer Complex for generation of dynamic display data. Most command data and all outgoing voice communications, facsimile messages, and teletype textual messages originate within the system.

Display

The Mercury Control Center display capability required modification to support the Gemini flights. Additional flight controller consoles were installed with the existing Mercury consoles and resulted in increased video, analog, and digital display capability. The world map was updated, both in Gemini network-station position and instrumentation capability. A large rear-projection screen was installed for display of summary message data. A second large screen was provided for display

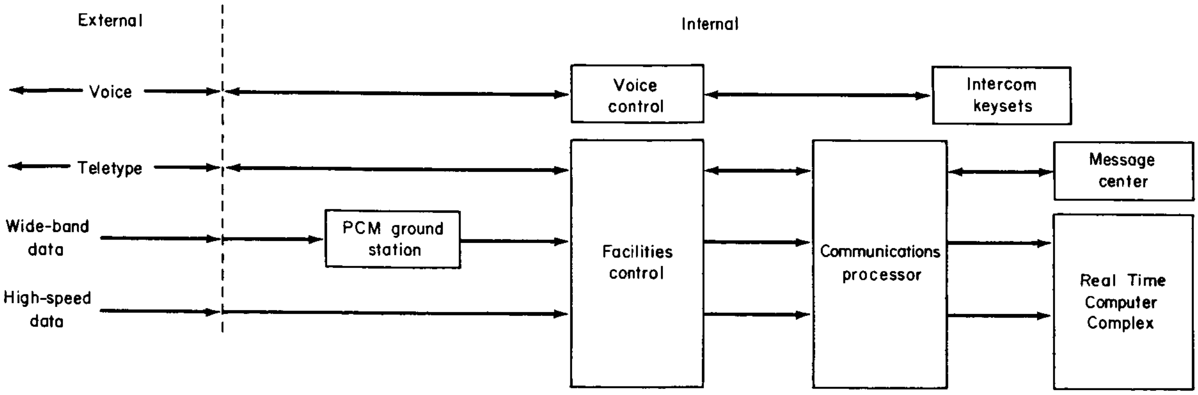


FIGURE 19–3.—MCC–H communications flow.

of flight rules, checklists, time sequences, or other group displays.

The implementation of the Mission Control Center at Houston provided major improvements in the amount and type of data displayed for real-time use by flight controllers. The display system utilizes various display devices, such as plotting, television, and digital, to present dynamic and reference information. Dynamic displays present real-time or near real-time information, such as biomedical, tracking, and vehicle systems data, that permits flight controllers to make decisions based on the most current information.

The display control system (fig. 19–4) is divided into five subsystems.

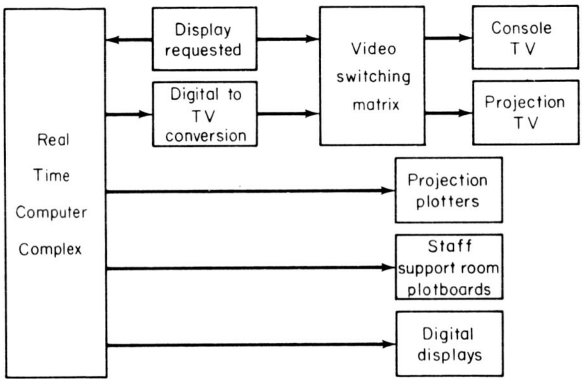


FIGURE 19–4.—MCC–H display/control subsystem.

Computer interface subsystem.—The computer interface subsystem and the real-time computer complex function together to provide the displays requested by flight controllers during actual or simulated missions. The interface subsystem detects, encodes, and transmits these requests to the real-time computer complex and, in turn, generates the requested displays, utilizing the output information from the computer complex.

Timing subsystem.—The timing subsystem generates the basic time standards and time displays used throughout the control center. The master instrumentation timing equipment utilizes an ultrastable oscillator and associated timing generators referenced to Station WWV and generates decimal, binary-coded decimal, and specially formatted Greenwich mean time for various individual and group displays. Standby battery power is provided for emergencies.

Television subsystem.—The television subsystem generates, distributes, displays, and records standard and high-resolution video information, using both digital and analog computer-derived data. A video switching matrix enables any console operator to select video from any of 70 input channels for display on his console TV monitor. The matrix accepts inputs from the 28 digital-to-TV converter channels, 11 opaque television channels, and other closed-circuit TV cameras positioned throughout the control center. Each console operator can also obtain hardcopy prints of selected television displays.

Group display subsystem.—The group display subsystem is made up of wall display screens in the Mission Operations Control Room (fig. 19–5). This system provides flight dynamics, mission status information, and reference data displayed in easily recognizable form. The system consists of seven projectors which project light through glass slides onto the large 10- by 20-foot screens. By selection of appropriate filters, the composite picture can be shown in any combination of seven colors.

Console subsystem.—The console subsystem is made up of consoles with assorted modules added to provide each operational position in the Mission Control Center with the required control and data display. The subsystem also provides interconnection and distribution facilities for the inputs and outputs of all these components, except those required for video and audio signals.

FIGURE 19–5.—Mission Operations Control Room, MCC–H.

Command

In the Gemini spacecraft, the amount of onboard equipment requiring ground control activation and termination has increased many times over that in the Mercury spacecraft. Project Mercury used radio tones for the transmission of command data; however, the number of available radio tones is limited by bandwidth and was found inadequate to support Gemini flights. Therefore, a digital system was subbit encoding is used to meet the Gemini command requirements.

At the Mission Control Center, the digital command system (fig. 19–6) can accept, vali-

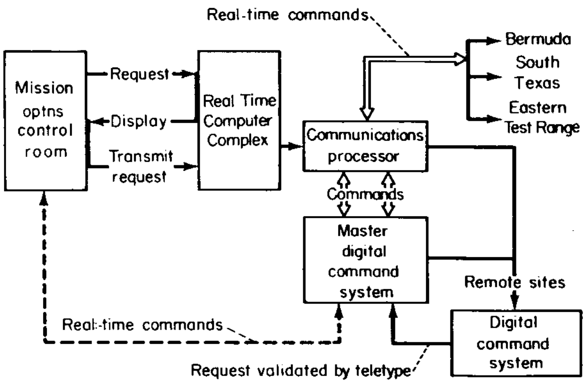


FIGURE 19–6.—Digital command system.

date, store (if required), and transmit digital command data through the real-time sites of the Manned Space Flight Network and to the remote sites equipped with digital command capabilities. The command data are transmitted to inflight vehicles or, at Cape Kennedy only, to a vehicle waiting to be launched. The system can also perform a simulated mode of operation similar to the operational modes.

Commands can be introduced into the digital-command control logic from the Real Time Computer Complex, from teletypewriter punched paper tape, or by manual insertion from the digital-command control consoles as remote control modules (located on the flight controller consoles).

Gemini Launch Data System

The Gemini launch data system was designed to provide the two Mission Control Centers with continuous command, radar, voice, and telemetry contact with the spacecraft from lift-off through orbital insertion. Inputs from three telemetry ground stations at Cape Kennedy are multiplexed with the downrange telemetry from the Eastern Test Range and are transmitted over wideband communciation lines to the Mission Control Center at Houston. In addition, real-time trajectory data can be sent to the Mission Control Center at Houston on high-speed communications lines.

Simulation Checkout and Training System

The simulation checkout and training system at the Mission Control Center in Houston allows the mission control team to perform either partial or total mission exercises. It provides for the development of mission operational procedures, the training of all personnel involved in controlling the mission, practicing the required interfaces between flight crew and mission control teams, and validation of support systems and control teams necessary during a mission.

Development of the Manned Space-Flight Network

If the requirements of the Gemini orbital and rendezvous missions were to be supported by the Manned Space Flight Network, major modifications of the network were necessary. Gemini missions required increased capability from all network systems, with exacting parameters and an exceedingly high reliability factor. To guarantee this reliability, the network was modified with proved systems that were constructed with off-the-shelf items of equipment. (See figs. 19–7 and 19–8.)

The network was required to provide the following functions necessary for effective ground control and monitoring of a Gemini flight from lift-off to landing:

(1) Communications between the network stations and the control center.

(2) Tracking and control of two vehicles simultaneously.

(3) Voice and telemetry communications with the spacecraft.

(4) Dual command data to two orbiting vehicles simultaneously.

(5) Reliability of all onsite systems for extended periods of time.

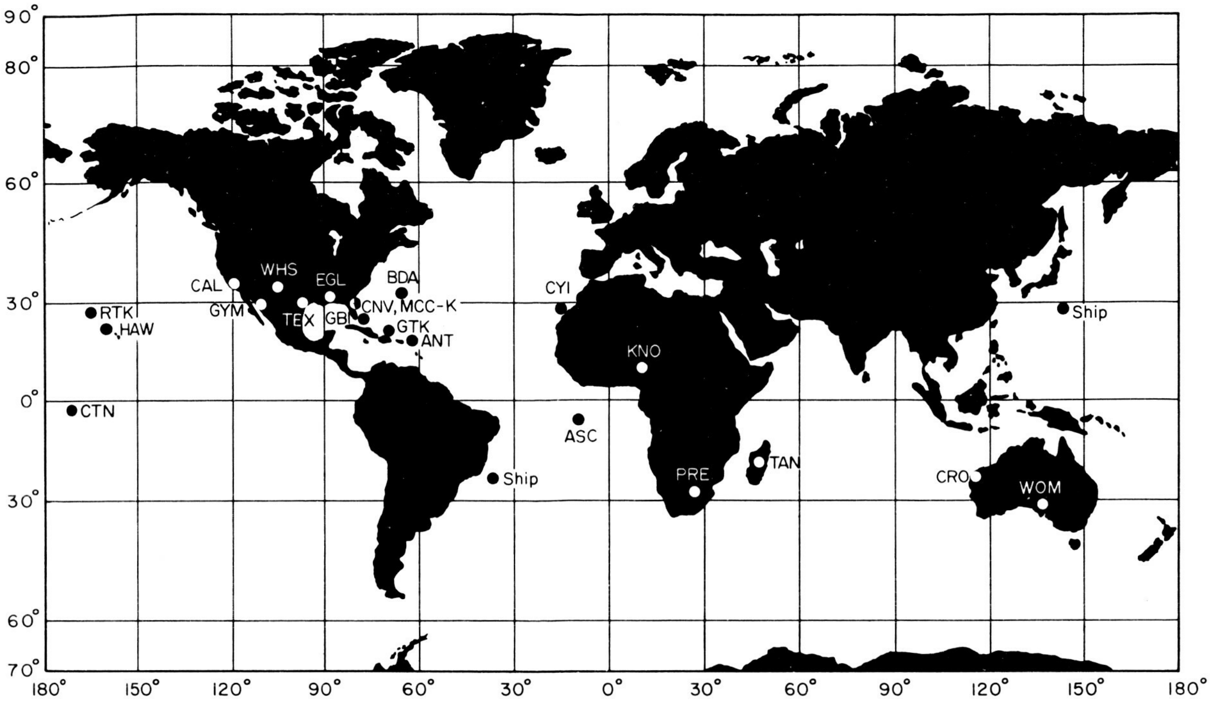


FIGURE 19–7.—Gemini network stations.

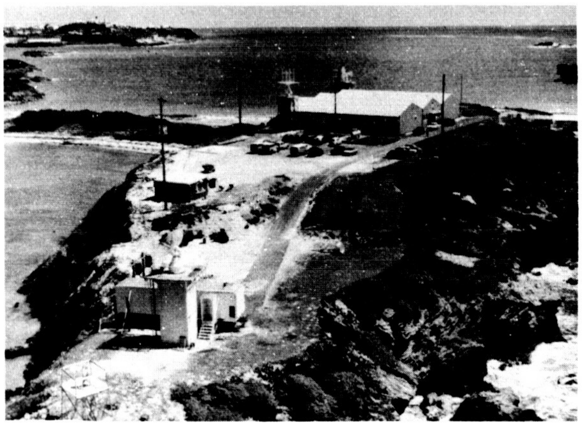


FIGURE 19–8.—Tracking station on Cooper's Island, Bermuda, West Indies.

Development of Network Equipment Systems

Radar

The network radar capability consists of the acquisition aid system and the radar tracking system.

Acquisition aid system.—For the rendezvous missions, the acquisition system must have the capability to acquire and accurately track two space vehicles simultaneously, as azimuth and elevation data from this system are used to aid the narrow-beam radars in rapidly acquiring their targets. Once the target is acquired, automatic tracking is possible, and no further acquisition assistance is required unless tracking is interrupted.

The spacecraft transmits telemetry signals in the 225- to 260-megacycle band. The signal is also used for target acquisition. The acquisition aid is a broad-beam-width antenna and does not require precise pointing to locate a target. It does, however, track with sufficient accuracy to provide pointing information to the narrow-beam radar.

The acquisition aid antenna provides not only a tracking and telemetry function but receives very-high-frequency voice communications from the orbiting spacecraft.

Although the target is normally first "seen" by the acquisition aid systems, the radar (C-band and S-band) search independently. At contact, all antennas can immediately be slaved to the system which makes acquisition first. This capability is provided by the acquisition bus system, which permits the operator of each individual system to know the status of all other antenna positions so that he can slave his equipment in azimuth and elevation to any other antenna.

Radar tracking system.—The radar tracking system provides the network and the control center with real-time information; that is, as soon as the radar has acquired the spacecraft, the operator enables a circuit and transmits the range, angle, and time data directly to the computers at the control center. These data are transmitted via teletype and high-speed data circuits.

The network radars consist of long-range, standard tracking radars that have been modified to meet manned space flight requirements. The network radar stations are equipped with

TABLE 19–I.—*Capabilities of Network Stations*

Station	Station symbol	Real-time telemetry to MCC–H	Acquisition aid	Radar	PCM telemetry ground station	Telemetry record	PAM telemetry (FM/FM) ground station	Flight controller display consoles	Digital command modulation	Radiofrequency command	Spacecraft communications (air-to-ground)	Voice	Teletype
Cape Kennedy	CNV												
Mission Control Center	MCC–K	X	X	X	X	X	X	X	X	X	X	X	X
Grand Bahama Island	GBI	X	X	X	X	X	X			X	X	[a]	X
Grand Turk Island	GTK	X	X	X	X	X				X	X	[a]	X
Bermuda	BDA	X	X	X	X	X	X			X	X	X	X
Antigua	ANT	X	X	X	X	X	X				X	[a]	X
Grand Canary Island	CYI		X	X	X	X	X	X	X	X	X	X	X
Ascension Island	ASC			X		X	X				X	[a]	X
Kano, Africa	KNO		X			X	X				X	X	X
Pretoria, Africa	PRE			X		X	X						X
Tananarive, Malagasy	TAN		X			X	X				X	X	X
Carnarvon, Australia	CRO		X	X	X	X	X	X	X	X	X	X	X
Woomera, Australia	WOM		X	X								X	X
Canton Island	CTN		X			X	X				X	X	X
Kauai Island, Hawaii	HAW		X	X	X	X	X	X	X	X	X	X	X
Point Arguello, Calif	CAL		X	X		X	X				X	X	X
Guaymas, Mexico	GYM		X	X	X	X	X	X			X	X	X
White Sands, N. Mex	WHS		X	X								X	X
Corpus Christi, Tex	TEX	X	X	X	X	X	X	X	X	X	X	X	X
Eglin, Fla	EGL		X	X		X	X					X	X
Wallops Island, Va	WLP		X	X	X	X	X	X	X	X	X	X	X
Coastal Sentry Quebec (ship)	CSQ		X		X	X	X	X	X	X	X	X	X
Rose Knot Victor (ship)	RKV		X		X	X	X	X	X	X	X	X	X
Goddard Space Flight Center	GSFC											X	X
Range Tracker (ship)	RTK			X		X					X	X	X

[a] Through Cape Kennedy Superintendent of Range Operations.

either S-band or C-band radars, or both. C-band radars operate on higher frequencies and afford greater target resolution or accuracy, while the S-band radars, operating at lower frequencies, provide excellent skin track capability.

The three principal types of radars used by the network stations (table 19–I) are the very long-range tracking (VERLORT), the FPQ–6 (the TPQ–18 is the mobile version), and the FPS–16. The S-band VERLORT has greater range capability (2344 nautical miles) than the C-band FPS–16; however, the FPS–16 has greater accuracy (±5 yards at 500 nautical miles). The C-band FPQ–6 has greater range and accuracy than the other two (±2 yards at 32 366 nautical miles).

Telemetry

Telemetry provides the flight controllers with the capability for monitoring the condition of the flight crew and of the spacecraft and its various systems.

To handle the tremendous flow of telemetry data required by Gemini rendezvous missions, eight of the network stations use pulse-code-modulated wideband telemetry instead of the frequency-modulated telemetry that was used during Project Mercury. The pulse-code-modulation data-transmission technique is used for exchanging all data, including biomedical data, between the spacecraft and the network tracking stations. Each station then selects and routes the biomedical data to the Mission Control Center in frequency-modulated form over specially assigned audio lines. Data are routed from the real-time sites in pulse-code-modulated form over wideband data and high-speed data lines to the Mission Control Center and in teletype summary form from the remote sites.

Remote-Site Data Processors

Associated with the telemetry systems are the remote-site data processors which help flight controllers keep up with the tremendous flow of information transmitted from the spacecraft. The controllers can select and examine specific types of data information on a real-time basis. The system automatically summarizes and prepares telemetry data for final processing at the Mission Control Center.

Command

The flight controllers must have some method of remote control of the onboard electronic apparatus as a backup to the flight crew. But, before the clocks, computers, and other spacecraft equipment can be reset or actuated from the ground, the commands must be encoded into digital language that the equipment will accept. This requirement led to development of the digital command system. Over 1000 digital commands can be inserted and stored in this system for automatic transmission to the space vehicles as required. Correctly coded commands can be inserted into the remote-site computers manually or by the control center via teletype data links. In addition, real-time commands can be transmitted through the command system from the control center.

Before the orbiting vehicles accept the ground commands, the correctness of the digital format must be verified. The information is then decoded for storage or for immediate use. Both the ground and spacecraft command systems have built-in checking devices to provide extremely high reliability. The space vehicles are able to accept and process over 360 different types of commands from the ground, as opposed to the 9 commands available with Mercury systems.

Communications

The Goddard Space Flight Center operates the overall NASA Communications Network (NASCOM) located around the world, and provides high-speed ground communications support for the agency's space flight missions. The Manned Space Flight Network uses a portion of the NASA Communications Network to tie together all network sites and the Mission Control Center with 173 000 miles of circuits, including 102 000 miles of teletype facilities, 51 000 miles of telephone circuits and more than 8000 miles of high-speed data circuits. Transmission rates over the network vary from 60 to 100 words per minute for teletype language to 2000 bits per second for radar data. The radio voice communications system at most stations consists of two ultrahigh frequency (UHF) receiving and transmitting systems and two high frequency (HF) transmitters and receivers for communications between the sites and the spacecraft.

Consoles

Five types of remote station consoles are included in the control rooms.

Maintenance and operations console.—The maintenance and operations console is used by the maintenance and operations supervisor. He is responsible for the performance of the personnel who maintain and operate the electronic systems at the station.

By scanning the panels, the maintenance and operations supervisor knows immediately the Greenwich mean time and the Gemini ground elapsed time since lift-off. Also available on the panel are pulse-code-modulated input/output displays, as well as controls with which the supervisor can select any preprogramed format that the pulse-code-modulation telemetry can receive.

On the right side of the maintenance and operations panel are status displays for the various electronic systems at the station. Through use of the internal voice loop, the supervisor can verify the RED or GREEN status of the systems.

Gemini and Agena systems monitor consoles.—Two consoles monitor Gemini and Agena systems. One console is the Agena systems monitor (to be used for rendezvous missions), and the other is the Gemini systems monitor. Identical in design, the two consoles display telemetered information and permit command of the vehicle events. Forty-five indicators on each console show vehicle parameters such as spacecraft attitude, fuel consumption, temperature, pressures, radar range, and battery current or supply. Meter alarm circuits generate audible signals whenever an indication exceeds the predetermined limits. To provide distinct signals for each console, the audible tones can be varied by adjustment of the oscillators.

Command communicator console.—The command communicator console is operated by the director of the flight control team and provides command control of certain spacecraft functions. In addition to having the displays and switches that the system consoles have, this console has nine digital clocks, including indicators for Greenwich mean time, ground elapsed time, and spacecraft elapsed time. Greenwich-mean-time concidence circuitry in the console allows presetting a time at which the time-to-retrofire (T_R) and the time-to-fix (T_F) clocks of the space vehicles will be automatically updated by the digital command system.

To convert telemetry information into teletype format, a pushbutton device is provided on the console. With this device, the Flight Director instructs the computer on which summary messages are to be punched on paper for teletype transmission.

Aeromedical monitor console.—The aeromedical console is monitored by one or two physicians. Displayed on this console are the physiological condition of the two orbiting astronauts and the operational condition of the onboard life-support systems.

As the Gemini spacecraft circles the earth, the console operators closely watch the fluctuations of four electronically multiplexed electrocardiogram (EKG) signals on the cardioscope. This display provides information concerning the heart functions of both astronauts.

As long as the spacecraft remains within tracking range of a station, the observers follow the electrocardiograms and blood pressures of the astronauts as charted on the aeromedical recorder. They also check the cabin pressure and oxygen consumption indicated on the dc meters, and they monitor the respiration and pulse rates of the astronauts.

Concluding Remarks

The performance of the Mission Control Centers at Houston and Cape Kennedy and the Manned Space Flight Network in supporting the Gemini Program has been completely adequate. In particular, the phase-over from the Mission Control Center at Cape Kennedy to the one at Houston during the early Gemini flights did not present any major problems. Operational failures did occur, particularly during long-duration missions. In all cases the redundancy and flexibility of the equipment have prevented any serious degradation of operational support.

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20. FLIGHT CONTROL OPERATIONS

By JOHN D. HODGE, *Chief, Flight Control Division, NASA Manned Spacecraft Center; and* JONES W. ROACH, *Flight Control Division, NASA Manned Spacecraft Center*

Summary

The objective of mission control is to increase the probability of mission success and to insure flight-crew safety. Any deviation from a nominal mission plan requires that a decision be made, and this decision may either increase the chances for mission success or jeopardize the overall mission, thereby affecting the life of the flight crew. In order to augment the analysis and decisionmaking capability, every mission control concept, function, procedure, and system must be designed and implemented with both crew safety and mission success as the primary objectives.

Introduction

Flight control is the portion of mission control pertaining primarily to the aspects of vehicle dynamics, orbital mechanics, vehicle systems operations, and flight crew performance. Flight control is defined as the necessary integration between the flight crew and ground-control personnel to accomplish manned space flights successfully.

At the beginning of Project Mercury, the flight control organization was established to provide ground support to the flight crew during all mission phases. This organization was responsible for the direction of mission operations, for insuring a greater margin of safety for the flight crew, and for assisting the flight crew with analyses of spacecraft systems. To accomplish the assigned tasks, flight-control-operations personnel must participate actively in all aspects of mission planning; they must have a good understanding of the spacecraft, launch vehicle, and ground systems operations; they must train operational personnel in near-mission simulations for the proper execution of planned and contingency activities; and they must evaluate postmission data for analysis and recommendations for improvement of future missions. The fundamental philosophy and objectives of flight control have remained constant since the inception of Project Mercury and have been a significant tool in the success of the Gemini Program. As the Gemini Program has progressed, flight control operations have been refined to provide a closer approach to optimum support during all mission phases.

Mission Planning

The success of the Gemini operations conducted thus far has been a function of extensive and thorough premission planning by flight-control personnel.

Mission Definition and Design

Specific mission activities normally begin with the receipt of the mission requirements approximately 2 years prior to the scheduled launch date. Each mission is constructed in relation to other missions to provide consistency and continuity in the overall program without unnecessary duplication of objectives. This advanced planning is necessary to provide the lead time for both the manufacture of the flight hardware and the construction and implementation of the ground support facilities. In this time period, the trajectory is designed, and the specific flight control plans and requirements are established. If, in the analysis leading to the design of the preliminary mission profile, a particular mission requirement is found to be incompatible, the requirement is compromised, and data supporting the decision are documented. As the mission plan and objectives become more clearly defined, the preliminary mission profile is updated and published as the preliminary trajectory working paper.

Flight Test Preparation

With the mission defined, the trajectory designed, and the flight and ground support hardware in production, flight-control personnel begin approximately a year of intensive preparation for the mission. This preparation includes the following:

(1) Detailed support requirements for the control center and tracking network are defined.

(2) Mission control documentation, such as mission rules, flight plans, procedures handbooks, and spacecraft and launch-vehicle schematics, are developed, reviewed, and refined.

(3) Real-time computer programs and the various operational trajectory profiles, including those for nominal, abort, and alternate cases, are prepared and checked out extensively.

(4) Landing and recovery plans are developed and tested for optimum support.

(5) Simulation training is provided to train the flight-control personnel and the flight crew to respond and support each other during all mission phases.

(6) Launch-vehicle and spacecraft tests are supported to obtain and review baseline data on systems interface operations for utilization during inflight analysis. Complementary to this Manned Spacecraft Center planning activity, the Goddard Space Flight Center and the Kennedy Space Center provide the necessary mission support for the Manned Space Flight Network and the launch complex, respectively.

Mission Execution

Real-time flight control activities begin with flight-control monitoring during the tests at the launch complex and with the launch countdown. To provide optimum mission support, all mission activities throughout the worldwide tracking network and the control center are integrated and keyed to the launch-complex-operations milestones.

The Flight Director and the remainder of the Mission Control Center flight control team assume mission responsibility at lift-off, and they monitor the launch trajectory and systems operations for possible deviations from the nominal. Immediate reaction by this team is required should a launch abort be necessary.

From the insertion go—no-go decision until recovery, flight control teams in the Mission Control Center and throughout the Manned Space Flight Network monitor the spacecraft systems operations, provide optimum consumables management, schedule flight-plan activities to accomplish mission objectives, monitor and compute trajectory deviations, and direct overall mission activities.

Postflight Analysis

After the mission has been completed, flight operations personnel are involved in a detailed postflight analysis and in a series of special debriefings conducted to evaluate their performance during the past mission so that operations during future flights can be improved.

Project Mercury Experience

At the conclusion of Project Mercury, an extensive review of the experience gained and the application of the experience toward the Gemini Program was initiated to provide more effective flight-control support.

The following concepts were used as a basic philosophy for the Gemini flight-control planning effort:

(1) Using one ground-control facility during all mission phases for positive mission control proved to be efficient and effective, and this centralized control philosophy was applied.

(2) A small nucleus of experienced flight control personnel was assigned to conduct the real-time mission activities and to train others to assume the same responsibilities for the expanding mission demands.

(3) The early Mercury Program developed real-time mission documentation through the process of reviewing every aspect of mission development for problem areas and solutions. These documents proved to be vital and effective tools for standardization of procedures and operational techniques of flight-control personnel. As the Gemini Program evolved, these documentation concepts were expanded and refined to meet the demands of the more difficult missions.

The following operational documents initiated in Project Mercury have further proved their value in the Gemini Program:

(1) Mission rules

(2) Flight plan

(3) Spacecraft systems schematics
(4) Remote-site and control-center procedures
(5) Integrated overall spacecraft countdown
(6) Trajectory working papers

The mission rules document is cited as an example of how a typical mission control document is developed. Other mission documentation has been developed in a similar fashion.

The primary objective of the mission rules document is to provide flight controllers with guidelines to expedite the decisionmaking process. These guidelines are based on an expert analysis of mission equipment configurations for mission support, of spacecraft systems operations and constraints, of flight-crew procedures, and of mission objectives. All these areas are reviewed and formulated into a series of basic ground rules to provide flight-crew safety and to optimize the chances for mission success. These mission rules are then put to the final test during an extensive series of premission simulations prior to the flight test. Some rules may be modified as a result of experience gained from simulations. To assure a consistent interpretation and a complete understanding of the guidelines, a semiformal mission-rules review is conducted with the primary and backup flight crews and with the flight-control teams prior to mission deployment. For final clarification and interpretation of the mission rules, all personnel are involved in a review conducted by the flight director and the flight-control teams 2 days before launch and during the terminal count on the final day.

Real-time simulation exercises were a necessary part of procedural development, mission rules evaluation, and flight-crew and flight-control-team integration.

Initial Gemini Development Problems

Flight-control personnel were faced with the responsibility of expanding their own knowledge to meet the greater demands of the more complex Gemini missions and ground-support equipment. Flight controllers found they needed to expand their technical backgrounds beyond those skills required in Project Mercury.

Mission control personnel found that computer processing was a necessity to handle the vast quantities of spacecraft and launch-vehicle telemetry and tracking information. The design of computer display formats for the new control center in Houston was a delicate task, requiring data of the proper type and quantity to aid, and not clutter, the evaluation and decisionmaking process. Personnel unfamiliar with computers and computer data processing had to master this new field to optimize the computer as a flight-control tool. To learn about computers, personnel interfaced directly with computer programers and witnessed the computer-subsystems testing to verify proper mission data flow. Remote-site teams began utilization of the remote-site processor computing system. They witnessed the advance in speed and accuracy available to them in telemetry and radar-data formatting and transmission to the Mission Control Center for evaluation. This was a vast improvement over Project Mercury operations, when spacecraft data were viewed on analog devices, and the selected values were recorded and transmitted to the control center by a low-speed teletype message. Remote-site and control-center personnel understood the importance of being able to use the computing facilities effectively. The flight controllers defined mission-control computing requirements at dates early enough to insert these requirements into the computer to be utilized for maximum mission support.

Some changes to the real-time computer program for the control center and the remote sites were necessary, due to adjustments in mission objectives and to mission control technique improvements. These changes posed some problems because the new requirements could not be integrated into the real-time computer system in the proper premission time period. In these instances, some off-line computing facilities have been utilized to fill in gaps, again without any compromise to flight-crew safety or mission safety. The flexibility inherent in the flight-control organization and its ground-support-facilities design played a vital role in the flight-control response to adjustments made in the mission objectives. During 1965, the decision to conduct Gemini missions with 2-month launch intervals required adjustments and flexibility at the launch sites and in the mission objectives as the launch date neared.

In July 1963 the question was asked as to how fast the flight-control organization could complete one mission and turn around to support the following mission. A preliminary study reported a complete turnaround time of 12 weeks would be required. But, as the entire Gemini effort gained more experience and confidence in its personnel and systems, the turnaround time shortened to launch minus 8 weeks, without compromising mission success or flight-crew safety. This allowed adequate time for debriefing and refinement of the previous mission control operation for the following flight.

To validate the expanded knowledge and procedural development necessary to interface flight-control personnel properly with their ground-support equipment, several plans were developed and executed.

A remote-site flight-control team traveled to the first Gemini tracking station available—Carnarvon, Australia. There, they developed and documented remote-site operations procedures. At the conclusion of this development, a Mission Control Center team went to the Mercury Control Center at the Kennedy Space Center to develop and document control-center operational guidelines. As each remote site became operational and was checked by remote-site teams, the developed procedures were reviewed and refined.

During October 1964, a week of network simulations was conducted with the Mission Control Center at the Kennedy Space Center and the new Gemini tracking network to integrate and test the developed procedures and to verify the correct mission information and data flow. These tests were conducted in near-mission-type exercises to train personnel for the first manned Gemini mission. They were scheduled so that adjustments to flight-control techniques could be accomplished prior to the scheduled launch date of the first manned Gemini mission.

Training exercises such as these and other simulations involving the flight crew and the flight-control teams were conducted to verify this important interface. The proficiency of the flight crews and of the flight-control team was the result of the numerous training exercises.

Results of these training and validation exercises were completely satisfactory and were put to further use by flight-control personnel involved with the development of the operating ground rules for the new mission control facility in Houston, Tex.

It became apparent that the new control center in Houston should be made available as soon as possible to support the more ambitious flight tests that were scheduled. The decision was made for this facility to support the Gemini II and III missions as a parallel and backup operation to the Kennedy Space Center. The success obtained from this support enabled the flight-control organization to use this new control center to direct and control the Gemini IV flight test, two missions ahead of the original schedule. There is no substitute for the real-time environment as an aid in assuring the readiness of a new facility. The support of these early missions undoubtedly enhanced the readiness and confidence level to support the later more complex missions.

The Mission Control Center at Houston contains the largest computing system of its type in the world. Along with other numerous automated systems, it enables flight-control personnel to work more effectively and to provide more efficient mission support. This major achievement was accomplished through an integrated team effort by NASA and its many support organizations.

Mission Control Decisions

Flight-control personnel follow a logical pattern in each decision determination. A logic diagram of the flight-controller decision-making process is shown in figure 20–1. This diagram traces the decision-making process from problem identification to data collection and correlation and to the recommended solution.

Anomalies or possible discrepancies are identified to flight control personnel in the following ways:

(1) Flight-crew observations.

(2) Flight-controller real-time observations.

(3) Review of telemetry data received from tape-recorder playbacks.

(4) Trend analysis of actual and predicted values.

(5) Review of collected data by systems specialists.

(6) Correlation and comparison with previous mission data.

(7) Analysis of recorded data from launch-complex testing.

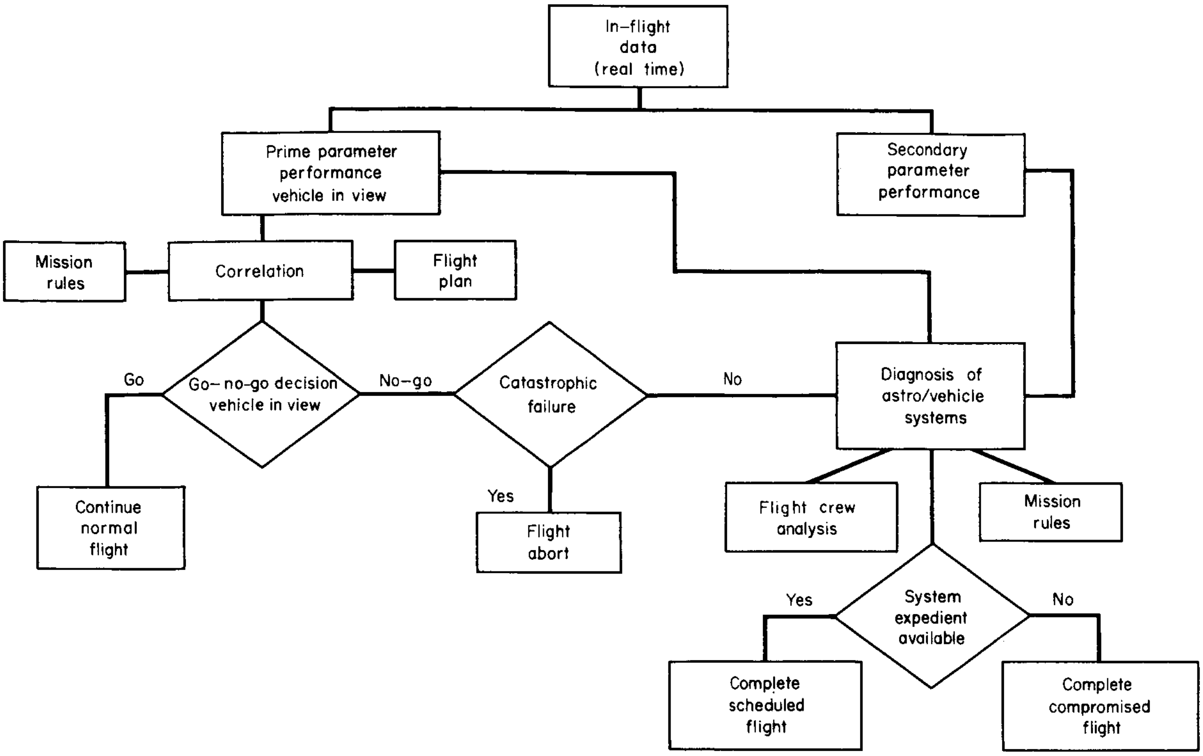

FIGURE 20-1.—The logic of flight-control decisions.

Flight-Control Mission Operations

The application of flight-control decision logic criteria is discussed in several Gemini flight test operations. Significant mission control operations activities are presented to illustrate several flight-control techniques and to show how support was provided during all mission phases.

Gemini III—Yaw Rates Caused by Water Evaporator

The Gemini spacecraft are equipped with a water evaporator to provide cooling when the space-radiator cooling is inadequate. The prime use of the water evaporator occurs during launch and the early portion of the first revolution, when the space radiator is ineffective due to the thermal effects of launch heating. The water evaporator is often referred to as the launch-cooling heat exchanger. The cooling principle employed in the water evaporator consists of boiling water around the coolant tubes at a low temperature and pressure, and venting the resultant steam overboard.

During the early part of the first revolution of Gemini III, the crew reported that the spacecraft was experiencing a yaw-left tendency for some reason. Prior to acquisition at the Carnarvon, Australia, tracking station, it was recommended to the flight director in Houston that the venting of the water evaporator could possibly produce a yaw-left to the spacecraft. There were no figures and calculations available at the time to support this theory. The theory was based on the fact that the water evaporator was known to be venting and that the vent port was located on the spacecraft in such a position that, if the thrust from the vent was sufficient, a yaw-left rate could be imparted to the spacecraft.

The water-evaporator theory was sound enough to eliminate any unnecessary concern with the onboard guidance and navigation system. Postflight analysis subsequently proved the theory to be valid. Although the yaw disturbance has been present on later missions, it has been expected and has caused no problems.

Gemini V—Reactant-Oxygen-Supply Tank-Heater Failure

During the countdown on Gemini V, the reactant-oxygen-supply tank was loaded with 182 pounds of oxygen and pressurized to 810 psia. At the beginning of the second revolution, the pressure had dropped from 810 to 450 psia under a heavy electrical load and after purging of both fuel-cell sections. The switch for the tank heater had been placed in the manual "on" position.

Over the Carnarvon tracking station, the pressure was reported to be 330 psia and dropping rapidly. At the Hawaii tracking station, approximately 20 minutes later, the oxygen pressure had fallen to 120 psia. It was determined at the time that the oxygen-supply heater had failed. In order to maintain the oxygen pressure, the spacecraft was powered down to 13 amperes, and by the fourth revolution the oxygen pressure had stabilized at 71.2 psia. This oxygen pressure was well below the minimum specification value for inlet pressure to the dual pressure regulators, and it was not known how long fuel cells would perform under these adverse conditions. The oxygen in the supply bottle was also on the borderline of being a two-phase mixture of liquid and gas, instead of the normal homogeneous fluid mixtures.

The performance of the fuel cells was monitored with special emphasis during the fourth and fifth revolutions to detect any possible degradation before the passing of the last planned landing area for the first 24-hour period. During this time, the orbit capabilities of the reentry batteries were reviewed in order to determine the maximum time that could be spent in orbit if a total fuel-cell failure occurred as a result of starvation of reactant oxygen. The maximum time was calculated to be 13 hours.

At the end of the fifth revolution, the flight crew were advised of a "go" condition for at least 16 revolutions. This decision was based on the following facts:

(1) Reactant-oxygen supply pressure had held steady at 71.2 psia for the fourth and fifth revolutions.

(2) There had been no noticeable voltage degradation.

(3) There had been no delta pressure warning light indications.

(4) Ground-test data indicated that no rapid deterioration of the fuel cells could be expected.

(5) There were 13 hours available on the reentry batteries.

This decision allowed flight-control teams to evaluate the fuel-cell operation for an additional 24 hours. The fuel cell reacted favorably during the next 24 hours, and another "go" decision was made at that time.

Gemini VI–A/VII Premission Planning

On October 28, 1965, 3 days after the first Gemini VI mission was canceled and approximately 6 weeks prior to the Gemini VII launch, the proposed Gemini VI–A/VII mission plan was presented to key flight control personnel for evaluation. From the initial review, the largest area of concern centered in the proper management of telemetry and radar data from two Gemini spacecraft. The ground system was configured to support one Gemini spacecraft and one Agena target vehicle for the Gemini VI mission. The major problem was how to utilize the system to support two Gemini spacecraft simultaneously without compromising mission success or flight-crew safety. Preliminary procedures for optimum data management were prepared and submitted in 3 days with the recommendation to support the Gemini VI–A/VII mission. Final plans and procedures were submitted 1 week later.

Real-time computer programs for the Gemini VI–A/VII missions were made available in five configurations by the Mission Control Center at Houston. Two remote-site computer programs, one for Gemini VII and one for Gemini VI–A, would match these five control center configurations to do the necessary computer processing and data routing. The Flight Director, through his control center staff, directed control center and remote sites of the proper configurations to provide the desired data for review by flight control personnel.

Control Center

The original Gemini VI computer program was operationally available and was used. The Agena portion of this program was bypassed, and certain processors were utilized to provide tracking data of spacecraft 7.

The following basic ground rules were established and followed as closely as practicable:

(1) Two basic computer programs would be utilized in five different configurations.

(2) Both computer programs would be capa-

ble of receiving manual inputs of spacecraft aerodynamic data.

(3) The Gemini VI–A program would contain the weight, reference area, and aerodynamics for spacecraft 6.

(4) The Gemini VII program would be identical to the Gemini VI–A program, with the following exceptions:

(a) It would process only spacecraft 7 telemetry.

(b) The spacecraft characteristics would initially be those of spacecraft 7.

(c) The Agena weight and area would be those of the Gemini VII spacecraft.

(d) The Agena thruster characteristics would reflect the spacecraft 7 aft-firing thrusters only.

Remote Sites

In a manner similar to that for the control center, certain basic guidelines were established and followed by remote-site personnel in the planning and execution of the combined Gemini VI–A/VII missions:

(1) Two remote-site data processor programs were written, one for Gemini VII and one for Gemini VI–A. The original Gemini VI remote-site data processor program was operational and was used. The Agena target vehicle portion of this program was bypassed, and the new Gemini VII program was obtained by recompiling the Gemini VI program with the spacecraft 7 calibration data.

(2) Two mission telemetry-data distribution frames would be provided. These telemetry-data-distribution-frame patchboards would switch and match the required spacecraft telemetry data to the proper flight control console. With these two patchboard arrangements and two remote-site data processor programs, remote tracking stations were capable of monitoring both spacecraft simultaneously.

At certain times the Gemini VII telemetry frequencies to be observed by ground control personnel were changed so that radiofrequency interference would be eliminated during launch preparation activities on Gemini VI–A at Cape Kennedy.

Since both spacecraft contained identical onboard command and telemetry systems, these systems had to be reviewed with the flight crews, and ground rules were established to eliminate any conflicts.

Orbital Activities

Gemini VII—Water in Space Suits

After the power-down of spacecraft 7 at the conclusion of the rendezvous with spacecraft 6, the flight crew reported water draining from their space-suit hoses when disconnecting the suits. At first this was thought to be condensate resulting from the chill-down of the spacecraft during the powered-down period. A cabin temperature survey reflected cabin humidity to be very high, approximately 90 percent. Over the Hawaii tracking station on the 167th revolution, the crew reported water was still draining from the suit hoses, and the onboard suit temperature gage was reading offscale on the low side. Although this was still thought to be condensate from the chill-down, there was a possibility the suit heat exchanger was flooded due to the water boiler (launch-cooling heat exchanger) being filled to the point that the differential pressure across the suit heat-exchanger plates was not sufficient to transfer water. The water boiler was not thought to be overfilled, since the evaporator pressure light was not on.

The result of the suit heat exchanger being flooded could indicate that the lithium hydroxide canister was being filled with water, which would inhibit its carbon-dioxide absorbing capabilities. Thus, the decision was made to dump the water boiler by boiling the water overboard. This was accomplished by bypassing the coolant around the space radiator and placing the cooling requirements on the water boiler.

Over the *Rose Knot Victor* tracking ship on the 168th revolution, the following procedure was voiced to the crew:

Time from lift-off, hr:min:sec	*Procedure*
268:33:00	Turn primary A pump on, B off; turn secondary A pump on, B off. Orient the spacecraft broadside to the sun. Start 8- to 10-degrees-per-second roll rate; maintain and select broadside orientation. Select radiator to bypass.
268:37:00	Turn evaporator heater on.
268:41:00	Select radiator flow.
268:42:00	Turn evaporator heater off. Turn primary A pump off, B on. Turn secondary A pump off, B on. Stop roll rate.

The above procedure was performed over the *Coastal Sentry Quebec* tracking ship on the 168th revolution. The Gemini VI-A flight crew reported large amounts of water actually vented from the water boiler. Approximately 2 hours later, the Gemini VII flight crew reported that the cabin was warm and dry, indicating that the suit heat exchanger was again operating properly and removing condensation. The development of this inflight test and the associated procedures was beyond the capability of the flight crew in the allowable time period.

Gemini VI–A—Accelerometer Bias Correction

During the first revolution of the Gemini VI-A spacecraft, it was apparent from the telemetry data that the X-axis accelerometer bias had shifted from the prelaunch value. The flight crew also noticed a discrepancy in the X-axis bias correction over the Carnarvon, Australia, tracking station when they performed their normal accelerometer bias check during the first revolution. The decision was made to update a new bias correction value via digital command load to the spacecraft computer over the United States at the end of the first revolution. Since a 24-second height-adjust burn was scheduled just after acquisition of signal over the United States, the bias correction was not uplinked until after completion of the burn. It was decided that the accuracy of the height-adjust burn was not critical enough to warrant updating prior to the burn. After the burn, the X-axis bias was updated as planned, and the value remained constant for the remainder of the mission. Correcting this bias constant made the execution of the remaining translational maneuvers more precise during the rendezvous phase and the remainder of the flight, including retrofire. This function of precisely accounting for the accelerometer bias is beyond the capability of the Gemini crew and must be performed by the flight control team. The requirement to update this constant was recognized by flight control personnel during the Gemini III mission. Requirements and procedures were developed to accomplish this task on the next spacecraft that required it.

Orbit Adjustments

The preflight mission plan called for the Gemini VII flight crew to perform a spacecraft phasing maneuver on the sixth day. This maneuver would provide an optimum Gemini VI–A launch opportunity on the ninth day for a rendezvous at the fourth apogee.

The preflight mission plan was not carried out because of the excellent turnaround progress at the launch site in preparation for the Gemini VI–A launch. To take advantage of this rapid turnaround progress, the decision was made to do a partial phasing maneuver on the third day, which would allow later orbit adjustments to optimize for either an eighth or ninth day launch of the Gemini VI–A flight. A posigrade burn of 12.4 feet per second was requested and accomplished, and subsequent tracking verified a normal spacecraft thruster burn. Again, a real-time mission plan change such as this is an example of the mandatory flexibility inherent in mission control operations. This flexibility permits a rapid response to take advantage of the situation as it unfolds.

Gemini III, V, and VI–A/VII Flight-Controller-Technique Summary

The most significant aspect of the items discussed has been the ability of the flight-control organization to identify the anomalies or requirements, to utilize the collected and available data, and to recommend solutions that enable the flight crew to accomplish the primary mission objectives. Without this extension of the flight-crew systems analysis, it is conceivable that several of the Gemini missions conducted thus far would have been terminated prematurely.

Concluding Remarks

The ability of the flight-control organization and the flight crew to work together as a team has greatly enhanced the success of the flight tests up to this point in the Gemini Program. This interface has been accomplished by numerous training exercises, by mission rules and procedures development, and by participation in system briefings between the flight crew and the flight-control personnel. Through this close relationship has developed the confidence level that must exist between the flight crews and the flight-control teams.

Experience gained from the Gemini Program up to this point is summarized as follows:

(1) During the launch, rendezvous, and reentry phases of a mission, the flight control task is primarily a flight-dynamics real-time problem. During the other mission phases, effective consumables management and flight-plan activities become more dominant.

(2) The orbital mission rules are immediate, short-term, or long-term decisions. Flight-control personnel do not normally participate in immediate decisions, as these are effected by the flight crew. Short-term and long-term decisions allow flight controllers time for data collection, review, analysis, and recommendations to accomplish mission objectives.

(3) Existing flight-vehicle instrumentation schemes are a design trade-off between systems complexity, payload capability, economics, and inflight systems management. Flight control personnel participate in flight-vehicle instrumentation configuration meetings to assure adequate malfunction-detection analysis and consumables management. In some instances, real-time computer operations are required to allow full use of the available data.

(4) During long-duration missions, detailed flight planning is not necessary except for the launch, rendezvous, extravehicular activity, and reentry phases of the flight tests. For extended missions, the remaining flight-plan activities must be arranged in a priority order and integrated into the flight plan at the appropriate times to accomplish the primary and secondary mission objectives.

(5) Experience gained during the testing phase of the program must be available for real-time use. Results of overstress testing are of particular importance in this area.

(6) The spacecraft mission simulator should be utilized primarily for procedural crew interface for launch and critical-mission-phase training, while development of computer-math models of flight vehicles is continued for detailed flight-controller training. This will eliminate a large computer programing effort and interface checkout on the mission simulator and also allow full utilization for flight-crew training.

(7) Communications satellites are effective systems in the accomplishment of manned space-flight operations. During the combined Gemini VI–A/VII missions, the *Coastal Sentry Quebec* tracking ship never lost communications while being supported by the communications satellite, Syncom III. In comparison, frequent loss was encountered over alternate routes during atmospheric transition periods.

(8) Advance planning and the inherent flexibility in both the facilities design and mission-control procedures allow for significant changes in mission objectives close to the launch date, if the basic configuration of the vehicle remains essentially constant.

(9) Flight-control support has been provided during all mission phases. During the Gemini VI–A/VII flight test, the flight-control team monitored and directed the Gemini VII spacecraft in its orbital activities while simultaneously accomplishing:

(a) A rendezvous simulation with the Gemini VI–A spacecraft at Cape Kennedy.

(b) Pad-support activities and the final launch countdown for the Gemini VI–A space vehicle.

(c) Simulations for the first Apollo mission from a different control room in the same control facility.

(10) Success in the proper and effective execution of mission control operations is a function of effective and thorough premission planning.

The basic experience learned thus far in the Gemini Program will be expanded and applied in appropriate areas for the remainder of the Gemini flight tests and for future programs in such a manner that the flight-control organization will continue to accomplish its assigned tasks.

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21. GEMINI POSTLANDING SYSTEMS TESTS AND RECOVERY OPERATIONS

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Summary

The recovery phase of the Gemini Program is discussed with consideration given to both postlanding systems and operations. The philosophy of systems operational evaluation, development, and validation prior to flight is presented, and the testing performed to support this philosophy is reviewed. The adequacy of this test program has been verified by the satisfactory performance to date, wherein all postlanding systems have performed as expected and wherein there have been no significant failures on actual flight missions.

Overall recovery operational support plans are summarized, and techniques are discussed for locating the spacecraft after landing and providing on-scene assistance and retrieval. The various landing situations encountered to date in the Gemini Program are presented, and the recovery activities reviewed. Landing distances from the recovery ship have varied from 11 to 91 nautical miles, and on-scene assistance times have varied from 12 to 50 minutes. Recovery operational support has been very satisfactory for all landing situations encountered. In addition, the operational flexibility provided by multiple landing areas has proved to be very valuable, in that it allowed the Gemini V mission to continue while a spacecraft electrical-power problem was being evaluated.

Introduction

The recovery phase of the Gemini Program is considered to encompass those activities from spacecraft landing through location and on-scene assistance and retrieval, together with the systems, plans, and procedures required for support during this period.

In the Gemini Program, postlanding systems, operational development, and testing were conducted in keeping with the basic philosophy that, insofar as possible, all systems and procedures would be validated in an operational test environment prior to flight. The systems include both those inherent in the spacecraft and those utilized by the operational support forces. Recovery operations in support of flight missions have been planned in keeping with the basic philosophy that a positive course of action would be preplanned for all possible landing situations, with the level of recovery support deployed into a given recovery area commensurate with the probability of landing in that particular area. Therefore, recovery forces are in position to support many different landing situations for each mission.

Postlanding Systems Testing

Utilizing experience gained in Project Mercury, the philosophy of conducting operational tests on the spacecraft, the spacecraft systems, and the support systems used in the postlanding and recovery mission phases received high emphasis during the periods prior to the first unmanned and the first manned flights. This operational testing supported several requirements: systems development under operational conditions; design verification and qualification; operational technique development; and recovery personnel training. Operational testing was carried out both under controlled test conditions requiring special facilities and also, where possible, under actual operational conditions representing very closely the environment to be expected in the actual mission landing and recovery areas. By this means, it was possible to identify many problem and potential problem areas on both the spacecraft and the spacecraft support systems, making it possible to redesign or change these systems before the flight missions. In potential problem areas where it was decided not to make system changes, the tests served to recognize the problem in sufficient depth to enable adequate operational procedures to be developed for most of the possible recovery situations.

From the spacecraft and spacecraft systems standpoint, the operational tests were carried out in the following basic areas:

(1) Spacecraft water stability (static and dynamic).

(2) Spacecraft structural integrity in the postlanding environment.

(3) Environmental-control-system postlanding testing.

(4) Postlanding electrical power testing.

(5) Spacecraft electronic communications and location-aid testing.

(6) Spacecraft postlanding habitability testing.

(7) Miscellaneous mechanical systems testing, visual location aids, etc.

Spacecraft support-systems and recovery-equipment operational development and testing were accomplished on the following:

(1) The auxiliary flotation device.

(2) The swimming interphone device.

(3) Airborne location receiver systems and tracking beacons.

(4) The survival beacon.

(5) The retrieval crane.

(6) Retrieval handling, and transportation dollies and cradles.

(7) Miscellaneous recovery equipment and line-handling devices.

(8) Launch-site surf retrieval equipment.

Operational techniques were developed for the following:

(1) Flight-crew egress.

(2) Recovery swimmer teams.

(3) Launch-site abort and recovery.

(4) At-sea retrieval.

(5) Postlanding safing and reentry-control-system deactivation.

Water Stability Testing

The Gemini spacecraft is designed to float in a nearly horizontal attitude after landing (fig. 21-1). Because of the small size and the basic circular cross section of the spacecraft, concern was expressed early in the program for the roll-stability characteristics, especially since the roll stability would greatly affect flight-crew egress techniques. There was potential danger of spacecraft flooding and sinking during egress, due to the low freeboard at the hatch-hinge line. Another concern with regard to water stability was in the pitch plane where the spacecraft originally had a nose-down trim attitude, also resulting in low freeboard at the hatch opening. Dynamic conditions, of course, tended to aggravate this condition. The potential hatch flooding problem was recognized early, and the spacecraft design included a sea curtain extending across the low-freeboard part of the hatch opening. This alone, however, was shown to be insufficient, and a combination of changes to the spacecraft configuration and operational techniques resulted from the early water-stability testing and egress-procedure development program. Spacecraft changes included the addition of extra flotation material in the reentry control system section, thus trimming the floating spacecraft to an approximately horizontal attitude in pitch. Initial design integration resulted in a spacecraft configuration that trimmed with an 18° list in the roll direction. This built-in list condition was retained and used to advantage by developing egress techniques in which the crewmembers egress one after the other from the high hatch.

FIGURE 21-1.—Gemini spacecraft postlanding flotation attitude.

Tight control of the postlanding center-of-gravity position was maintained throughout the spacecraft design and buildup phase, and spacecraft preflight measured center-of-gravity data

FIGURE 21-2.—Gemini spacecraft during water stability testing.

are checked against the water-stability data to insure satisfactory postlanding performance. Figure 21-2 shows the Gemini spacecraft during static water-stability tests.

Spacecraft At-Sea Testing

Early in the program, it was recognized that the Gemini spacecraft configuration, which called for almost all of the electrical and electronic systems to be packaged outside the pressure compartment, would present some special postlanding problems, since these systems and attendant cabling would be in flooded compartments after a water landing. Thus, the potential shorting and corrosive effects of salt water on all the equipment which was required to function after landing could have a distinct effect on both the safety and comfort of the flight crew and the successful conclusion of the recovery operation. The loss of electrical power to the electronic location beacon, for instance, could preclude, or at least make very difficult, the actual postlanding location of the spacecraft. This is especially the case for a contingency landing where the spacecraft would be in the water for a long period of time, and where the very nature of the contingency makes the location problem more difficult. The water and corrosion proofing of these essential postlanding systems called for stringent regard to detail design on the part of the system subcontractors, as well as close attention by the spacecraft contractor during electrical assembly. In addition, systems validation required realistic operational testing, with the spacecraft and the postlanding systems exactly like the configuration and installation of an actual flight spacecraft.

Gemini spacecraft static article 5 was provided for this testing. For all intents and purposes, this static article represented a flight spacecraft, complete with all systems required to operate in the landing and postlanding phases, and was equipped for manned at-sea testing. Static article 5 was later used for egress training and is still used for this purpose prior to each mission.

This test spacecraft was delivered by the contractor to the Manned Spacecraft Center in late December 1963. At the Manned Spacecraft Center, the spacecraft was extensively instrumented to allow all essential systems parameters to be monitored or recorded while the spacecraft was floating in the at-sea environment. In addition, biomedical instrumentation was installed so that test-subject safety could be determined at all times during manned tests. The instrumentation system called for remote monitoring and recording aboard the Manned Spacecraft Center test ship by the use of a floating cable to the spacecraft (fig. 21-3). For safety reasons, a line capable of lifting the spacecraft was provided as part of the connection from the ship.

In April 1964, static article 5 was placed in the Gulf of Mexico, 30 miles off Galveston, with two test subjects aboard for a postlanding test

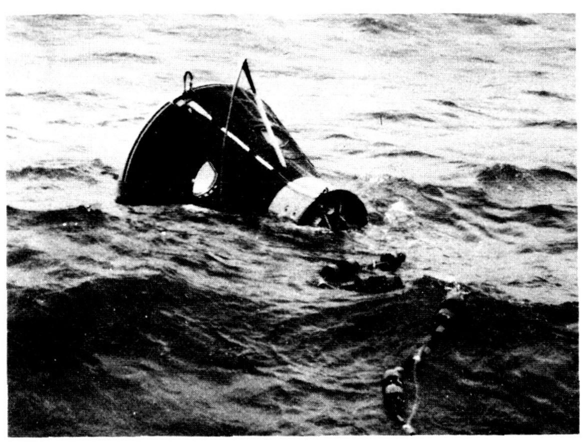


FIGURE 21-3.—Gemini static article 5 spacecraft undergoing at-sea tests to evaluate postlanding systems.

that was scheduled to last up to 36 hours. Wave heights of 5 to 6 feet and winds of 10 to 15 miles per hour existed at the time. These conditions were representative of the open-ocean conditions to be expected in recovery areas. Systems problems were encountered soon after the spacecraft was placed in the water; the first of these was the failure of the high-frequency antenna, which bent due to the wave-induced high rates of spacecraft motion. An abnormally high current drain was encountered in the electrical supply system, and, after approximately 1 hour, one of the two fans supplying air to the space suits failed. Pronounced seasickness of both test subjects was apparent within some 10 minutes after they entered the water, and suit ventilation from the postlanding environmental control system was found to be inadequate to provide crew comfort with suits on and hatches closed. This inadequacy existed even though the water temperature, air temperature, and solar heat load were less than that to be expected in daytime, subtropical recovery areas. The test was terminated after approximately 2 hours, primarily because of crew discomfort and worsening sea conditions.

The posttest systems failure analysis brought to light several areas of shorting in the electrical cabling installation, and corrosion problems on battery straps, electrical connectors, and spacecraft structural areas. The suit-fan failure was found to be caused by sea water entering the snorkel system, and this problem subsequently was solved after many at-sea tests with boilerplate spacecraft incorporating modified snorkel designs. Static article 5 was reworked during a 5-month period and made ready for another at-sea manned test with systems modified as necessary.

The at-sea test was repeated, with two astronauts as test subjects. This time, the test lasted 17 hours, and all spacecraft systems performed to specification except for a few problems of a very minor nature. Crew comfort remained generally inadequate throughout the test, even though the test environmental conditions were again less than to be expected in subtropical recovery areas. With space suits removed, test-subject comfort was improved, but no sequencing of the spacecraft environmental control system could be found that would provide adequate cooling with the hatches closed. All postlanding systems were tested during a test period that included aircraft ranging and homing runs on the ultra-high-frequency location beacon, and tests of the spacecraft high-frequency direction-finding system, using the U.S. Navy and Federal Communications Commission networks.

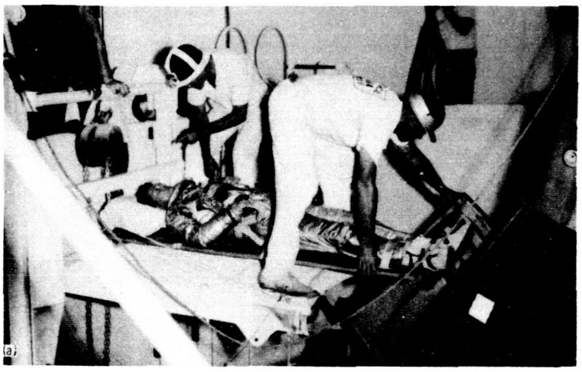
Subsequent manned at-sea tests were conducted to develop a technique to allow better cabin ventilation for crew comfort. It was found possible to open the high hatch a small amount even in relatively rough sea conditions, and this, in conjunction with suit removal, is the configuration that will be utilized in the event it becomes necessary for the flight crew to remain inside the spacecraft for long periods after a water landing.

Environmental-Water-Tank Tests

In the months just prior to the first manned flight, various degrees of concern existed relative to the ability of the flight crew to sustain the postlanding environment safely. The generally high heat levels to be expected inside the spacecraft cabin after reentry and landing, in conjunction with heat stress placed on the flight crew due to seasickness and possible dehydration, had to be considered in addition to any postflight problems caused by orthostatic hypotension. One of the limitations of operational testing is the difficulty in obtaining simultaneous occurrence at all desired environmental conditions. In order to gain a better feel for systems limitations in providing a habitable postlanding environment, a water-test-tank facility was built to provide for the following controlled environmental conditions:

(1) Air temperature at sea level.
(2) Humidity.
(3) Water temperature.
(4) Surface-wind simulation.
(5) Solar heat loading.
(6) Wave-induced spacecraft motion (by mechanical linkage).
(7) Spacecraft cabin reentry-heat pulse.

It was decided to conduct tests tailored to the actual postlanding environment to be expected in the Atlantic recovery area for the Gemini IV mission, which was the first long-duration flight in this program. In an effort to simulate the preconditioning effects of space flight, bed rest was determined to be the most practical method for the purpose of these tests. Three tests were conducted using the static article 5 spacecraft: the first, using two test subjects without preconditioning; the second, two other subjects who had received 4 days' bed rest preconditioning; and the third, using the original two test subjects with bed rest preconditioning. Figure 21–4(a) shows the suited test subjects being



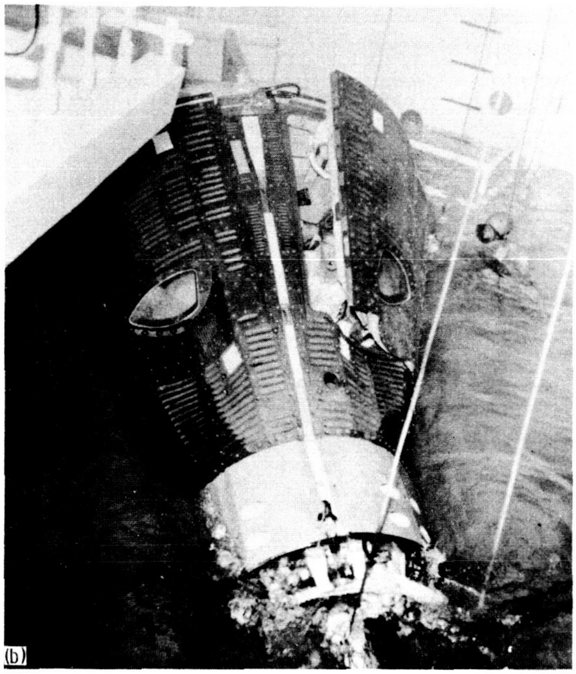
(*a*) Test subject being placed in spacecraft.
FIGURE 21–4.—Manned postlanding spacecraft habitability tests.

transferred to the spacecraft inside the test chamber. The transfer is made in this position in order not to compromise the preconditioning effects of horizontal bed rest.

The tests commenced at the simulated time-of-reentry heat pulse and progressed through the spacecraft change-to-landing attitude into an 18-hour postlanding phase, with the test crew egressing into life rafts at the end of the test. Figure 21–4(b) is a photograph taken during the postlanding test period. Biomedical data were taken before, during, and after the tests; and spacecraft systems data were monitored during the test. In general, the tests were considered successful in that the spacecraft system, together with the developed postlanding flight-crew procedures, was shown to be capable of maintaining adequate crew habitability for an acceptable postlanding period in a subtropical recovery environment. Thus, these tests added to the confidence level for postlanding operations on the Gemini IV and subsequent missions.

Retrieval Equipment

An aircraft carrier is used for spacecraft retrieval in the primary landing area, and de-



(*b*) Spacecraft during testing in a controlled environment.
FIGURE 21–4.—Concluded.

stroyers are primarily used in abort and secondary landing areas. A carrier has, as basic equipment, a crane capable of lifting weights well in excess of that of the Gemini spacecraft; hence, the carrier retrieval techniques followed closely those previously developed in the Mercury Program. Destroyers could retrieve the Mercury spacecraft with existing boat davits. However, the use of destroyers to retrieve the Gemini spacecraft presented a problem because the existing equipment on this type of ship cannot lift the spacecraft. Trade-off studies were made to determine the desirability and feasibility of providing all destroyers with a special lift capability, compared with use of destroyers only for crew retrieval and with the spacecraft remaining at sea until a ship with an inherent lift capability could arrive. The latter would have meant long delays in spacecraft retrieval time, especially in the abort landing areas. It was concluded that destroyers should be provided with the full capability of spacecraft retrieval, with the design goal of a simple retrieval crane which could be assembled on a destroyer's deck in a minimum of time and with little structural change to the ship. It was also decided at this time that the design should include the capability to retrieve the Apollo spacecraft, thus providing for a future requirement with an overall cost saving. Therefore, the Apollo spacecraft weight provided the main design criteria for all retrieval equipment presently used in the Gemini Program. Two types of lifting crane were designed, manufactured, and operationally tested aboard the NASA test-support vessel in the Gulf of Mexico. Both prototypes were next evaluated aboard a destroyer in the Atlantic, and one prototype, the davit rig, was selected for production manufacture. The davit rig basically consists of a crane capable of lifting 36 000 pounds, which is the Apollo retrieval weight plus 3g. The crane is mounted on the side of the destroyer fantail (fig. 21–5) and is fully power operated, providing spacecraft lift and power rotation of the retrieved spacecraft onto the deck. In addition, the design provides a power-operated holdoff arm which encircles the spacecraft during retrieval, preventing pendulum spacecraft motions due to rough seas. An important feature of the rig is that the entire control operation is accomplished by one man, thus avoiding difficult human coordination problems which are often a problem in rough sea operations. Destroyers have been modified with quickly detachable deck sockets in sufficient numbers to allow for Department of Defense scheduling flexibility in both the Pacific and Atlantic fleets. The entire davit crane can be installed or removed in approximately 4 hours.

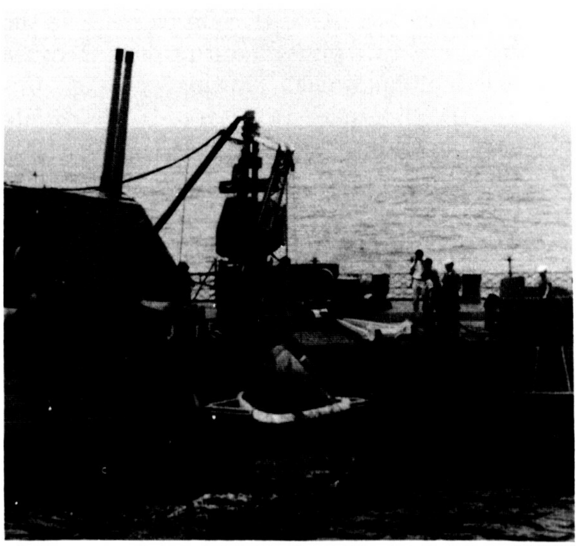


FIGURE 21–5.—Retrieval exercise by a destroyer utilizing the davit crane.

To obtain the best techniques, the other supporting retrieval equipment, such as special hooks, lines, dollies, and cradles, was designed and operationally tested in much the same manner as the davit rig.

Auxiliary Flotation Device

Recovery plans call for an auxiliary flotation device to be attached to the spacecraft as soon after landing as feasible. The device is installed by helicopter-deployed swimmer teams in the primary and launch-site landing areas or by pararescue personnel, deployed from fixed-wing aircraft, in other areas. Figure 21–6 shows the device attached to the spacecraft. Basically, the flotation device provides the following:

(1) Flotation to the spacecraft in case of leaks from structural damage, which could result in possible spacecraft loss because of sinking.

(2) A relatively stable work platform for the recovery personnel to provide any required assistance to the flight crew while awaiting retrieval.

The device is designed to be a form-fit to the spacecraft when inflated; thus, little or no relative motion exists between the spacecraft and the device. This provides a damping of spacecraft wave-induced dynamic motions without difficult load-point or fatigue problems. The design incorporates a redundant tube, installed within the external tube, and a second inflation system, as a backup to the primary external flotation tube.

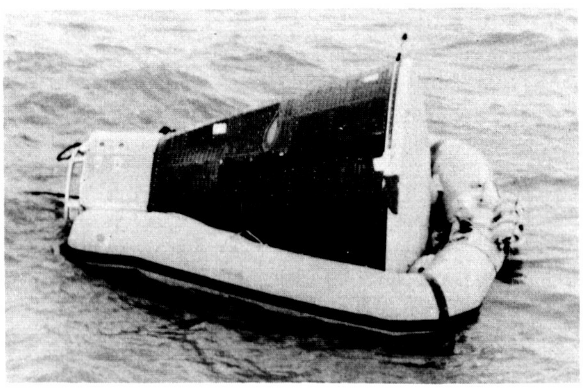


FIGURE 21–6.—Flotation collar installed on the spacecraft.

Development testing, airdrops, operational life tests, and installation techniques were accomplished in actual ocean environments.

Recovery Operations

The primary responsibility of the recovery forces is the rapid location and the safe retrieval of the spacecraft and the flight crew, and the collection, preservation, and return of information relating to the recovery operations, test data, and test hardware. This responsibility begins when the spacecraft and/or flight crew have been boosted relative to the launch pad.

Recovery plans and procedures are provided for all conceivable landing situations. For planning purposes, landing areas have been divided into planned landing areas and contingency landing areas. The planned landing areas are further divided into launch-site landing area, launch-abort (powered flight) landing area, periodic emergency landing area, and the nominal end-of-mission landing area. Any landing outside one of these planned landing areas is considered a contingency landing.

Department of Defense forces support all of these various landing situations. The level of support required is commensurate with the probability of a landing in the area and also with any special problems associated with such a landing.

Recovery Tasks

The various recovery tasks can be divided into three general categories. The first task is that of location. After the spacecraft has landed, the location of this landing may be determined by using tracking information from the Gemini network and then by computing a landing point from this information. Postlanding high-frequency-beacon signals are radiated from the spacecraft and ground-based high-frequency direction-finding stations are alerted for support in the event of a remote-area landing. In addition, the spacecraft is equipped with electronic location-aid beacons which operate in the ultra-high frequency range. This beacon is designed to radiate signals during and after landing. All landing areas are supported by aircraft having special receiver equipment compatible with the spacecraft beacons. Therefore, electronic homing by location aircraft is considered to be the primary means for recovery-force location finding, and considerable attention is given to the equipment and training devoted to this task. Visual location, once this aircraft homing has been accomplished, is assisted in the daytime by the presence of sea dye marker, which is dissipated from the spacecraft after landing, and at night by a flashing light.

Once the spacecraft has been located, the second phase begins, that of on-scene assistance. This on-scene assistance is provided by swimmers deployed either by helicopter or by fixed-wing aircraft. Each of these groups is equipped with the flotation collar which can be rigged on the spacecraft in order to provide for opening the spacecraft and rendering such assistance to the crew as may be needed.

The final phase of the recovery task is the retrieval of the crew and spacecraft and their return to the home base. This is accomplished in the primary landing area by using the inherent capabilities of the aircraft carrier to lift the spacecraft from the water. The crew may remain in the spacecraft for transfer to the recovery ship, or they may be transferred to the ship by helicopter earlier. Other ships, such as oilers and fleet tugs, regularly used in the recovery forces, also have an inherent capability of retrieving the spacecraft. Destroyers, which are also commonly used as recovery ships, do not have such an inherent capability and are fitted with the retrieval rig previously described.

Launch-Site Recovery

The launch-site landing area is that area where a landing would occur following an abort during the late portions of the countdown or during early powered flight. For planning purposes and considering all possible winds, it includes an area approximately 41 miles seaward of Cape Kennedy and 3 miles toward the Banana River from launch complex 19, with its major axis oriented along the launch azimuth (fig. 21–7). However, during the actual mission, the launch-site forces are concentrated on a relatively small corridor within this overall area. The corridor is determined by computing loci of possible abort landing points, utilizing the nominal launch trajectory and measured winds near launch time.

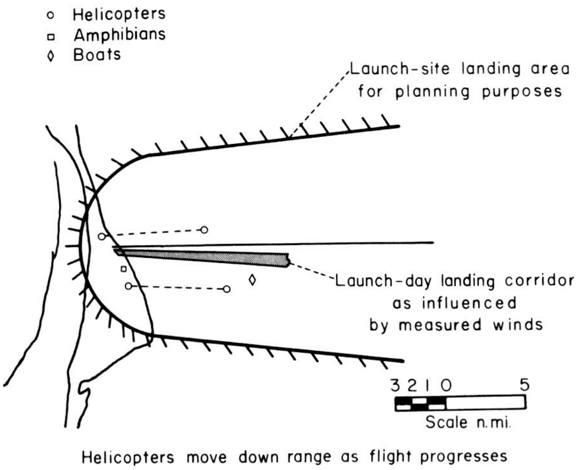


FIGURE 21-7.—Plan view of launch-site recovery area showing a typical force deployment.

Recovery problems in this area are unique and varied. Depending on the time of abort, the following situations can occur:

(1) Abort by seat ejection, followed by a landing on land or in the water just eastward of the launch pad.

(2) Abort by spacecraft, followed by seat ejection prior to landing because of the spacecraft impacting on land or in water too shallow for a safe landing.

(3) Abort by spacecraft, followed by a nominal deep-water landing in the spacecraft.

Decisions following abort in situations (2) and (3) are assisted by a ground observer who uses wind and tracking data in real time. This landing-position observer is prepared to advise the flight crew whether to remain with the spacecraft or to eject, following an abort during this critical time period. Because of the possibility of injury to the flight crew as a result of ejection-seat acceleration, launch-vehicle fire and toxic fumes, and landing in the surf or on obstructions, it is planned for the recovery forces to be capable of rapidly providing medical and other emergency first aid to the flight crew. In order to do this, a number of vehicles having unique capabilities are employed in the launch-site recovery area. The helicopter is the principal means of retrieval of the flight crew in a launch-site abort situation. The recovery forces are deployed in an excellent position to observe aborts in the launch-site area, and this visual observation is considered the primary method of location. However, assistance in location is available, if needed, in the form of information from a computer impact-prediction program. As a further backup, the flight crew's survival beacon is also activated following seat ejection, in order to provide an electronic location aid during parachute descent.

In addition to helicopters, the launch-site recovery force includes special amphibious vehicles and small boats so that all possible landing and recovery situations can be supported. Figure 21–8 shows a launch-site-recovery-force amphibian engaged in a surf recovery exercise. This launch-site recovery posture has been employed on all Gemini missions.

Suborbital Mission

The Gemini II flight was supported by 8 ships and 13 aircraft positioned along the ballistic ground track in such a way that they could reach any point in the area within 12 hours (fig. 21–9). At the planned landing point, an aircraft carrier with helicopter-borne swimmer teams was positioned to provide end-of-mission recovery capability. The aircraft were airborne along the ground track in order to provide on-scene assistance (flotation collar) and were capable of reaching the spacecraft within 4 hours of landing anywhere along the ground track or in the overshoot landing area.

Orbital Missions

The first manned Gemini flight was a three-orbit mission terminating in the West Atlantic area in the vicinity of Grand Turk Island (fig. 21–10). A total of 17 ships was employed to support the launch-abort landing areas and periodic emergency landing areas at the end of the first and second revolutions. A carrier and a destroyer having retrieval capability were pre-

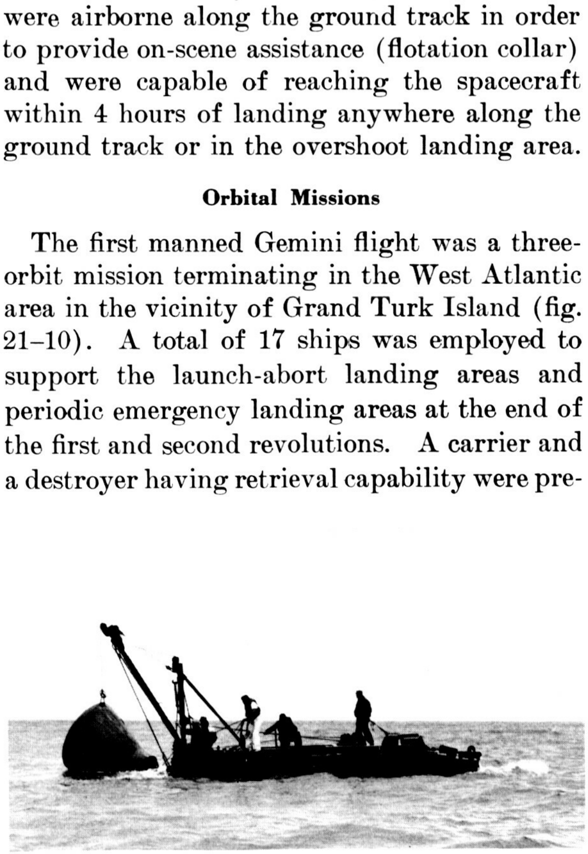


FIGURE 21–8.—Gemini surf retrieval vehicle.

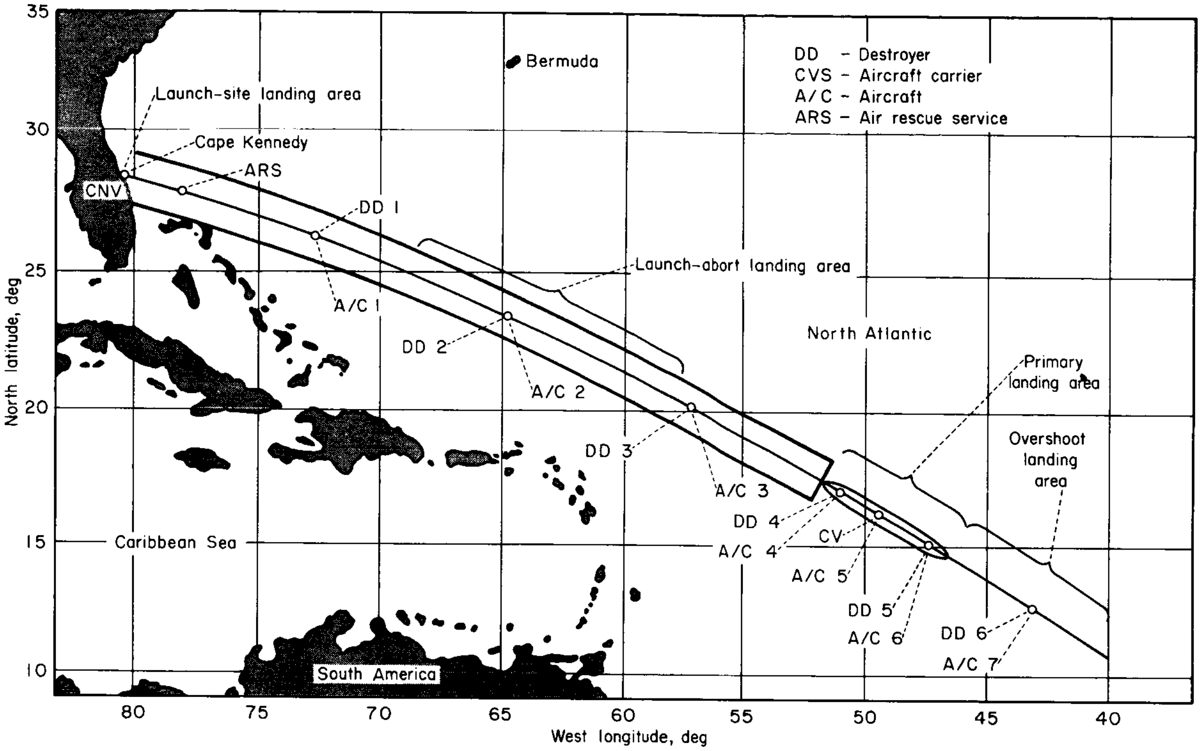


Figure 21–9.—Gemini II recovery-force deployment.

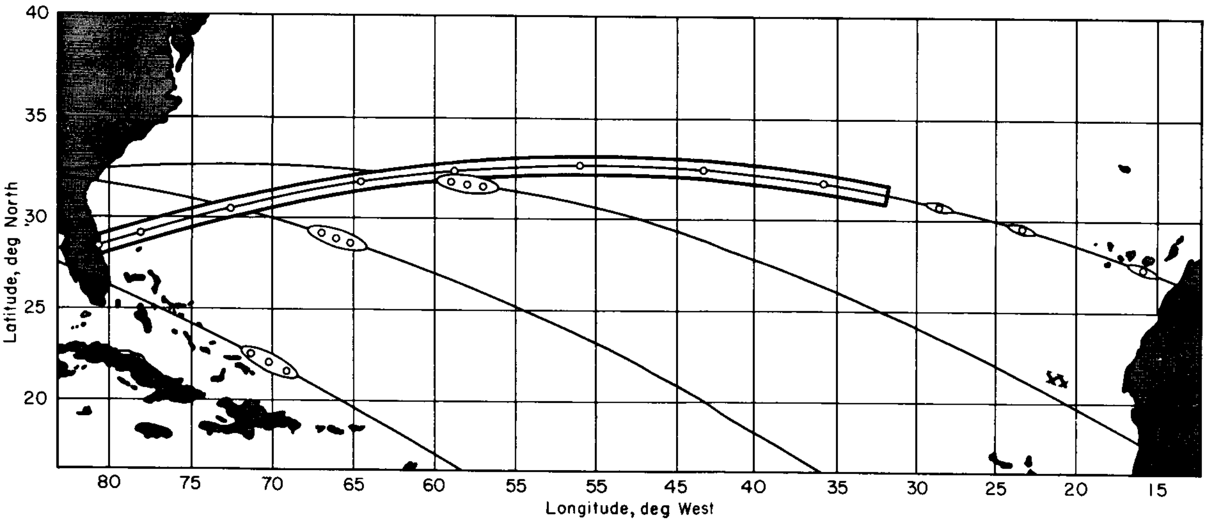


Figure 21–10.—Gemini III planned recovery area.

positioned in the end-of-mission landing area. Contingency forces consisted of aircraft located at stations around the world in such a way that they could reach any part of the worldwide ground track within 18 hours of a landing.

For long-duration missions, a recovery zone concept was adopted in which ships were placed in four zones around the world: West Atlantic, East Atlantic, West Pacific, and mid-Pacific. Landing areas were designated within these zones each time the ground track crossed the zone (fig. 21–11). One of the zones, the West Atlantic, was designated as the end-of-mission landing area and was supported by an aircraft carrier as well as destroyers. The other three zones were supported by destroyers and oilers.

Ships assigned to the launch-abort landing area were redeployed into the Atlantic landing zones after a successful launch. This distribution of recovery forces provided considerable flexibility in moving recovery forces in order to provide for changing aiming points resulting from variation in launch azimuth, to provide for precession of the ground tracks during the long-duration mission, and to take advantage of good weather conditions within the zone.

Contingency forces again consisted of aircraft deployed to staging bases around the world so that they could reach any point along the ground track within 18 hours of notification.

Primary Landing Area

In each case, the end-of-mission landing area was supported by an aircraft carrier with its special capability to provide a helicopter platform and an excellent facility for postflight activities. In addition, fixed-wing aircraft could be launched and recovered aboard in order to deliver personnel and data expeditiously. By providing carrier-borne helicopters with a location capability, it was possible to completely cover the terminal landing area with the carrier and its air group. Figure 21–12 shows the normal disposition of these aircraft in the vicinity of the carrier. One aircraft, desig-

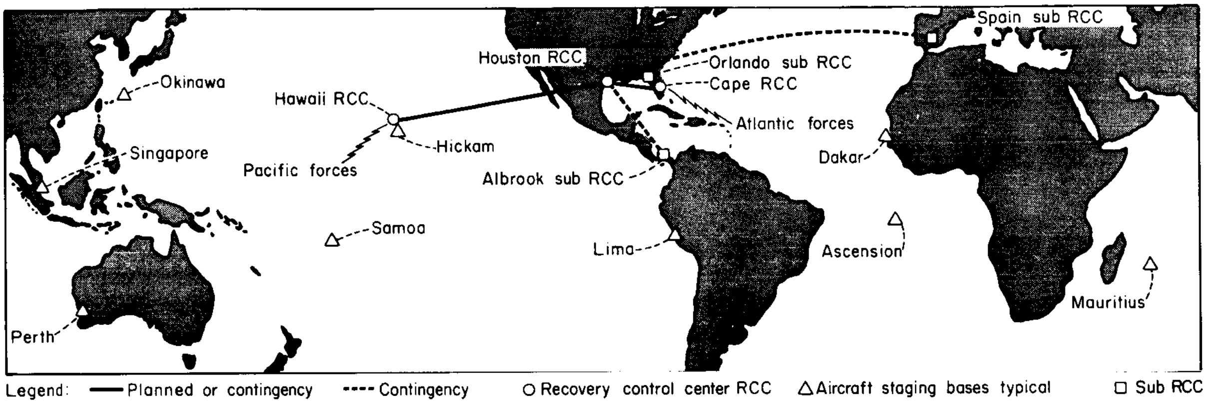


FIGURE 21–11.—Recovery control centers and typical contingency force staging bases.

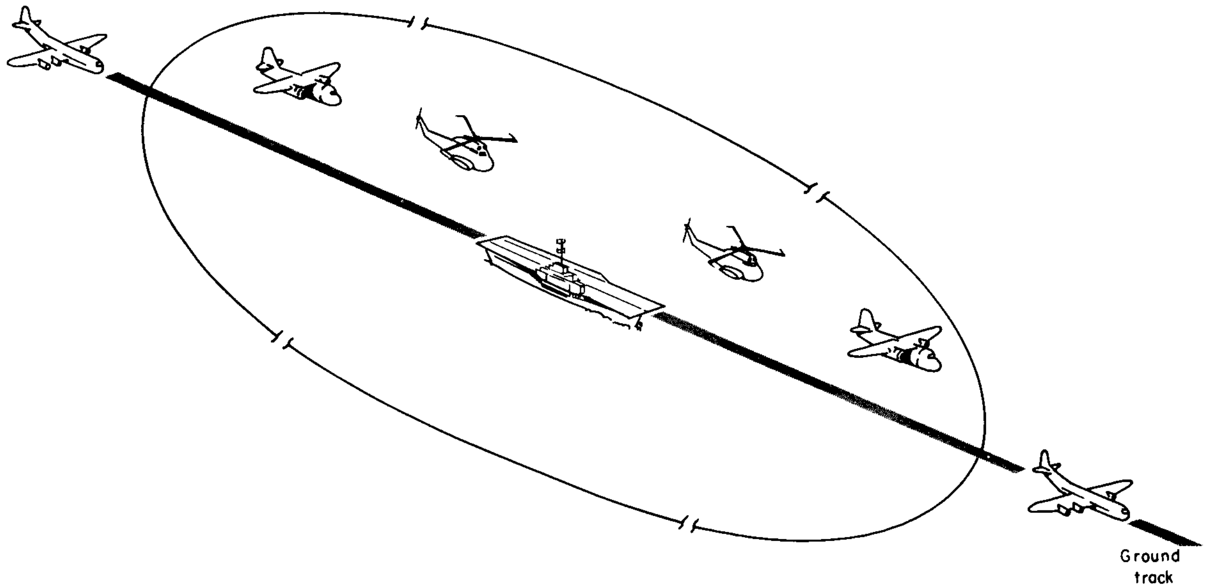


FIGURE 21–12.—Carrier and Aircraft positions in Primary landing area.

nated "Air Boss," served as an on-scene commander and air controller. After the search helicopters had located the spacecraft, swimmer helicopters were vectored-in to provide the on-scene assistance and to return the crew to the carrier, if desired. In addition, fixed-wing communications-relay aircraft relayed all radio transmissions in the recovery area back to the ship and to the various control centers on the beach.

The control of recovery forces is exercised through an arrangement of recovery control centers connected with the recovery forces through a worldwide communications network. These centers are depicted in figure 21-11. The primary interface between recovery and other mission operations activities occurs in the Mission Control Center at the Manned Spacecraft Center. The Mission Control Center also serves as the overall recovery control center.

Both planned and contingency recovery forces in the Atlantic area are controlled through the Recovery Control Center at Cape Kennedy, while Hawaii serves this function in the Pacific area. Contingency recovery forces in other command areas are controlled from recovery control centers in Europe for the Africa-Middle East area, in the Panama Canal Zone for the South American area, and in Florida for the North American area. These centers were established in order to take advantage of existing Department of Defense organizations and arrangements.

A summary of the Gemini Program recovery operations to date is presented in table 21-I. All landings have been in the primary recovery area, with the distance from the primary recovery ship varying from approximately 11 to 91 nautical miles, as shown.

It is significant to note that, although all landings have been in the nominal end-of-mission landing area in the Atlantic, the secondary landing areas in the Pacific were very beneficial during the 8-day Gemini V mission. During the early orbits in this mission, trouble developed with the spacecraft electrical-power source. Since the next several orbits did not pass through the primary landing area, the presence of these secondary recovery areas, with recovery forces on-station, allowed the flight to continue until the electrical-power problem could be evaluated. The electrical-power problem was eventually stabilized, and the mission was subsequently flown to its planned duration.

The primary recovery ship is positioned near the target landing point; therefore, the distances shown in table 21-I are a reasonable summary of landing accuracies to date. Postlanding recovery times are shown in the last three columns of table 21-I. In all landings, these times have been well within planning requirements, and the recovery force performance has been very satisfactory. Electronic aids were utilized in the location of the spacecraft for all but the Gemini VII flight, which landed within visual range of a deployed recovery aircraft. Even in this case the recovery aircraft was alerted to the near presence of the spacecraft by an electronic aid. In general, location techniques have proved very satisfactory and justify the close attention and training devoted to this phase of recovery.

For all Gemini missions, the landing area weather has been good, partially due to the fact that the target landing point is selected on the basis of forecasts and weather reconnaissance flights. On-scene assistance activities, including swimmer performance, has been very satisfactory, and the flotation collar has given no problems, again justifying the thorough operational evaluation and test program. Maximum exposure of the spacecraft systems to the unassisted postlanding environment has been 50 minutes, with most on-scene-assistance times being considerably less. Overall experience has tended to confirm the possibility of motion sickness and postlanding habitability problems. However, for the short times involved and for the weather conditions that have prevailed, no significant problems caused by the postlanding environment have been encountered.

All flight crews except the Gemini VI-A crew have been returned to the primary recovery ship by helicopter. The Gemini VI-A crew chose to remain with the spacecraft until it was retrieved by the recovery ship. Ship retrieval of the spacecraft has been nominal in all missions.

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TABLE 21–I.—*Recovery Operations Summary*

Flight	Location method	Description	Earth revolutions	Recovery forces		Weather		Distance from recovery ship to landing point, nautical miles	Event times after landing, minutes		
				Ships	Aircraft	Wind, knots	Wave height, feet		Flotation attached	Crew on ship	Spacecraft onboard
Gemini II	Electronic	Suborbital unmanned	1860 n. mi. downrange	8	13 aircraft, 9 helicopters	23	18	25	20	--------	90
Gemini III	Electronic	Orbital manned	3	17	44 aircraft, 11 helicopters	20	7	60	30	72	167
Gemini IV	Electronic	Orbital manned, 4 days	62	16	43 aircraft, 10 helicopters	13	4	48	20	57	136
Gemini V	Electronic	Orbital manned, 8 days	120	15	36 aircraft, 10 helicopters	8	3	91	50	91	235
Gemini VI–A	Electronic	Orbital manned, 1 day	16	14	38 aircraft, 10 helicopters	6	3	11	30	64	64
Gemini VII	Visual	Orbital manned, 14 days	205	14	38 aircraft, 10 helicopters	17	3	12	12	32	64

22. FLIGHT CREW PROCEDURES AND TRAINING

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Summary

Flight crew preparation activities outlined herein include initial academic training, engineering assignments, and mission training. Pilot procedures are discussed in conjunction with the simulation equipment required for development of crew procedures for the various phases of the Gemini mission. Crew activity summaries for the first five manned flights are presented, with a brief evaluation of the training effort.

Introduction

Because the Gemini operational concept takes full advantage of the pilots' control capabilities, crew preparation involves a comprehensive integration and training program. Some of the pilots participated in the design phase. All have followed their spacecraft and launch vehicle from the later stages of production through the many testing phases at the contractors' facilities and at Cape Kennedy. A wide variety of static and dynamic simulators have been used to verify design concepts and to provide subsequent training.

Procedures and Training Facilities

To better illustrate the crew activities, successive flight phases will be discussed in conjunction with the procedures and major training facilities involved.

Launch

During the launch phase, the flight crew monitors the launch vehicle performance and is given the option of switching to spacecraft guidance or of aborting the mission, in the event of anomalies in the launch vehicle or in the spacecraft performance. Figure 22–1 shows a view of the left cockpit with the launch-vehicle display, the guidance switch, and abort controls. By observing propellant tank pressures, engine-chamber-pressure status lights, and vehicle rates and attitudes, the command pilot can monitor the launch vehicle performance. If the flight crew observe excessive drift errors, they can actuate the guidance switch to enable the spacecraft guidance system to guide the launch vehicle. Launch-vehicle guidance failures, which cause rapid attitude divergence, automatically trigger the backup spacecraft guidance system.

The launch-abort procedures are divided into four discrete modes which are dependent on dynamic pressure, altitude, and velocity. Although the Gemini Mission Simulator provides the overall mission training, the Dynamic Crew Procedures Simulator (fig. 22–2) is the primary simulator used to develop launch-vehicle monitoring and abort procedures. Variations of $\pm 90°$ in pitch are used to simulate the changing longitudinal acceleration vector. Yaw and roll oscillations and launch acoustic noise-time histories are also programed to improve the simulation fidelity. The motion, noise, and cockpit displays are driven by a hybrid computer complex. Approximately 80 launch cases are simulated in the familiarization and training program.

Rendezvous

The primary rendezvous controls and displays are shown on the instrument panel in figure 22–3. The crew utilizes the "8-ball" attitude indicator for local vertical or inertial reference, flight director needles for computer and radar-pointing commands, digital readout of the radar range and angles through the computer console, and analog range and range-rate display. Orthogonal velocity increments, displayed on the left panel, present to the pilot the three velocities to be applied during the various rendezvous phases. All of these displays are used to accomplish closed-loop rendezvous.

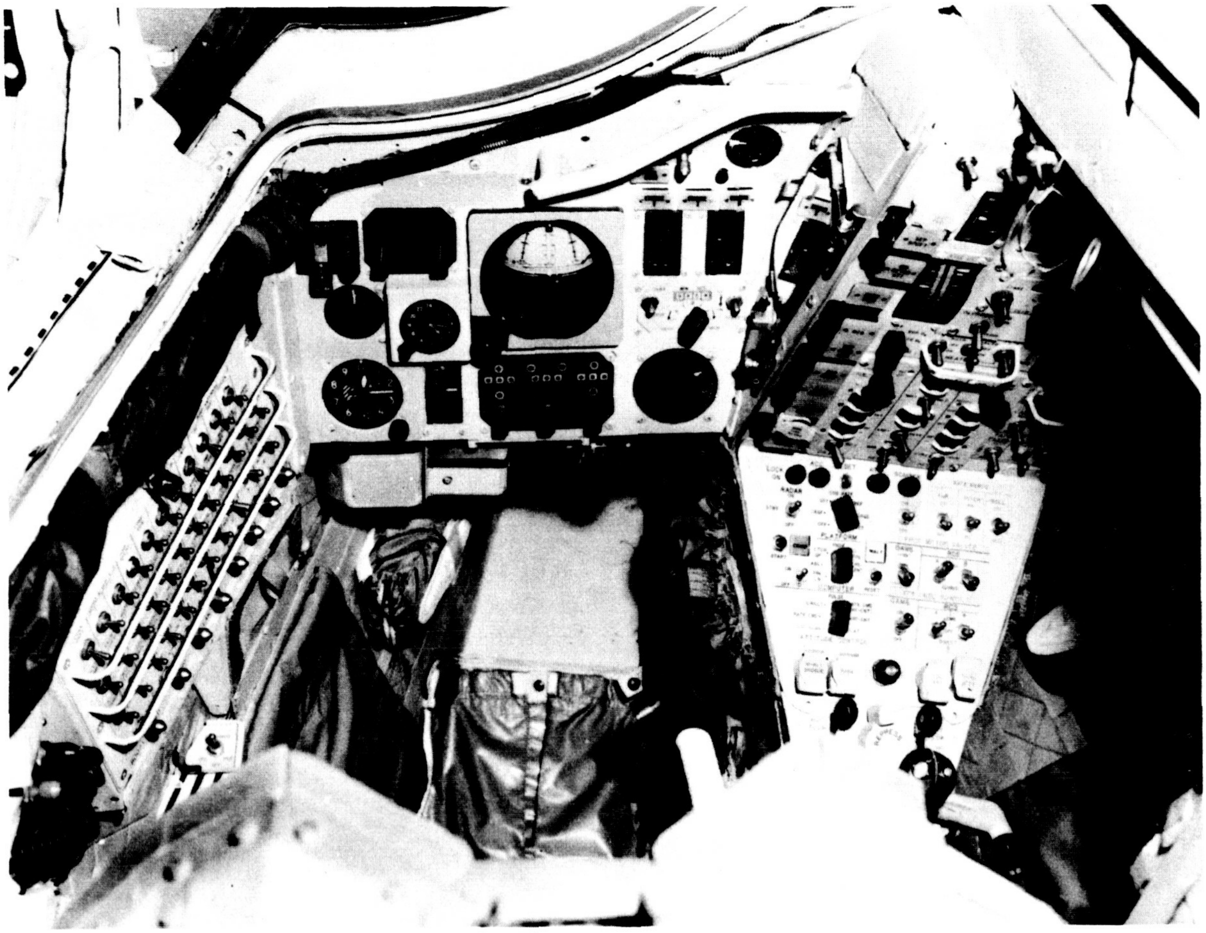


FIGURE 22-1.—Cockpit displays and controls normally accessible to the command pilot.

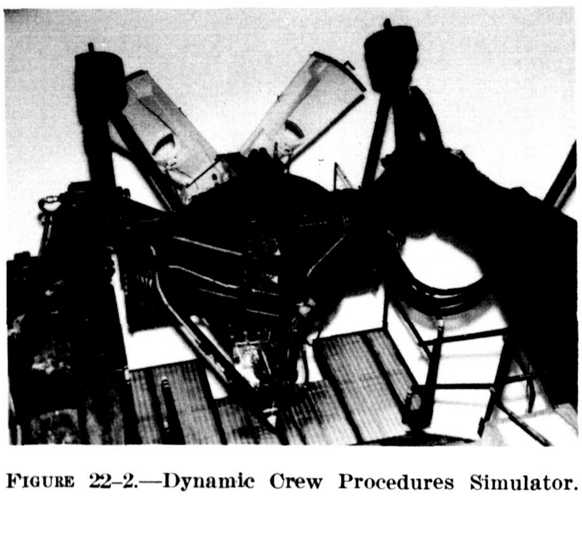
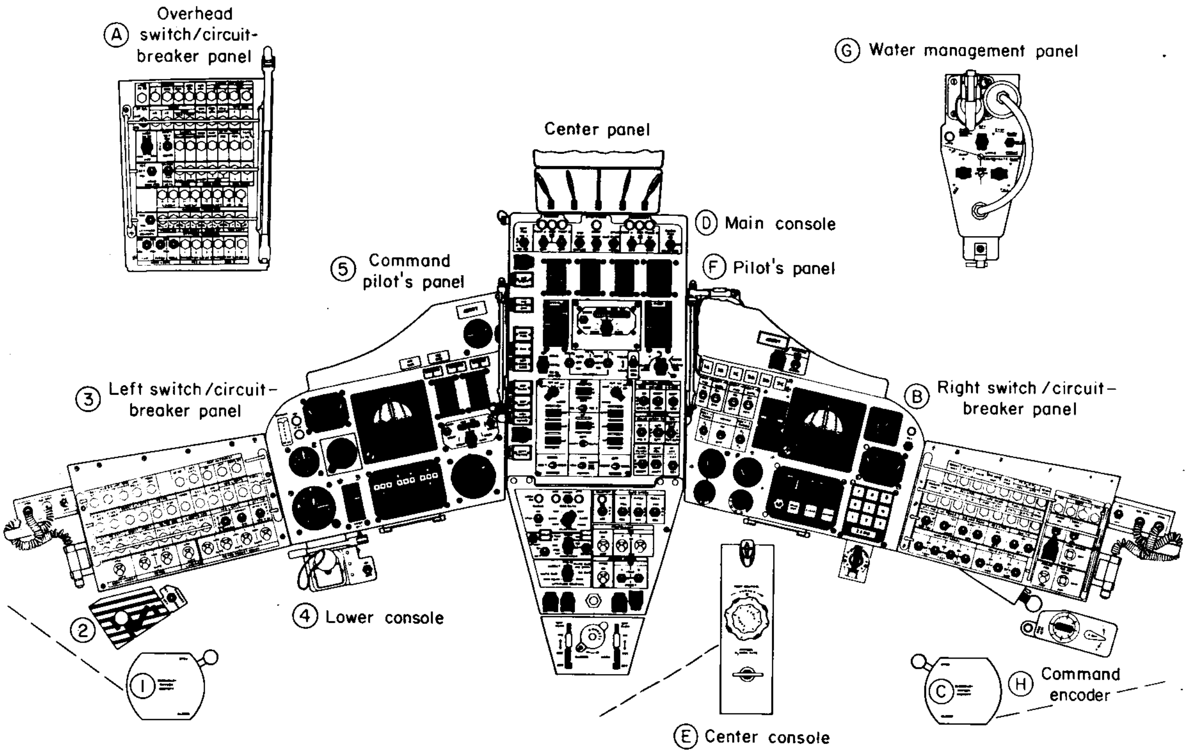


FIGURE 22-2.—Dynamic Crew Procedures Simulator.

A major portion of the rendezvous work, however, has been devoted to development of backup procedures. These backup procedures are required in the event of radar, computer, or inertial platform failures. The NASA and the spacecraft contractor have developed onboard charts which the pilot can use, with partial cockpit displays in conjunction with visual target observation, to compute the rendezvous maneuvers. To aid in the primary and backup rendezvous procedures, a collimated reticle is projected onto a glass plate in the left window (fig. 22-4). The brightness of the reticle is controlled by a rheostat. The pattern encompasses a 12° included angle. This device is used to aline the spacecraft on the target or starfield or to measure angular travel of the target over discrete time intervals.

Initial verification of the rendezvous procedures was accomplished on the engineering simulator (fig. 22-5) at the spacecraft contractor's plant. This simulator consists of a hybrid computer complex, a target and star display, and a crew station. Subsequent training was accomplished on the Gemini Mission Simulator



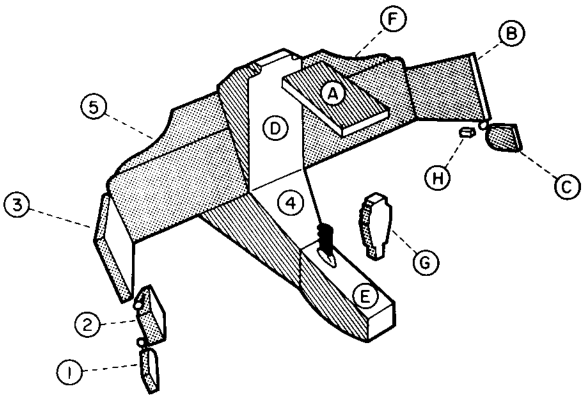


FIGURE 22-3.—Spacecraft instrument panel: (*1*) secondary oxygen shut-off (l.h.); (*2*) abort handle; (*3*) left switch/circuit-breaker panel; (*4*) lower console; (*5*) command pilot's panel; (*A*) overhead switch/circuit-breaker panel; (*B*) right switch/circuit-breaker panel; (*C*) secondary oxygen shut-off (r.h.); (*D*) main console; (*E*) center console; (*F*) pilot's panel; (*G*) water management panel; (*H*) command encoder.

(fig. 22-6), at the Manned Spacecraft Center. A second unit (fig. 22-7) is in the Mission Control Center facility at Cape Kennedy, Fla. The computer complex of both mission simulators consists of three digital computers with a combined storage capacity of 96 000 words. Six-degree-of-freedom computations are carried out during launch, orbit maneuvering either docked or undocked, and reentry. Maximum iteration rate for the six-degree-of-freedom equations is 20 cycles per second. Digital resolvers are incorporated to send analog signals to the various displays. Out-the-window visual simulation of the stars, the earth, and the rendezvous target are presented to each pilot through an infinity optics system. A spherical starfield is located within the crew-station visual display unit. The rendezvous target and the earth are generated remotely and are superimposed on the starfield scene by means of television, beam splitters, and mirrors within the crew-station display unit. Figures 22-8 and 22-9 shows an indication of the view available to the crew through the window of the simulator at Cape Kennedy. The rendezvous-target-vehicle scene is generated electronically, and the earth scene is televised from a filmstrip. The simulator at the Manned Spacecraft Center utilizes a 1/6-scale

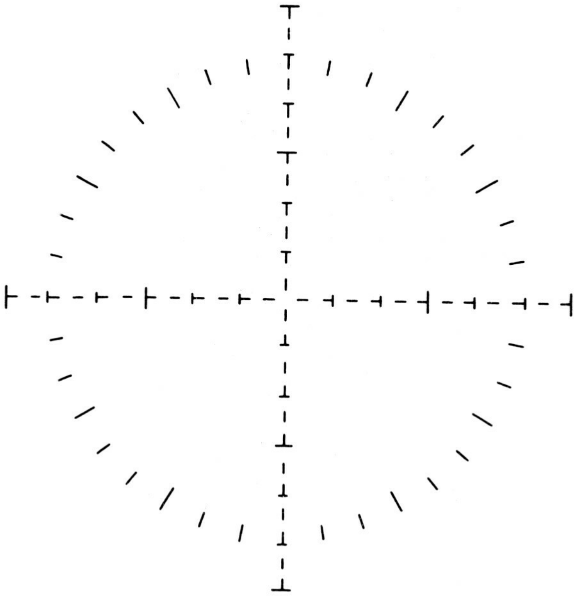


FIGURE 22-4.—Optical sight pattern.

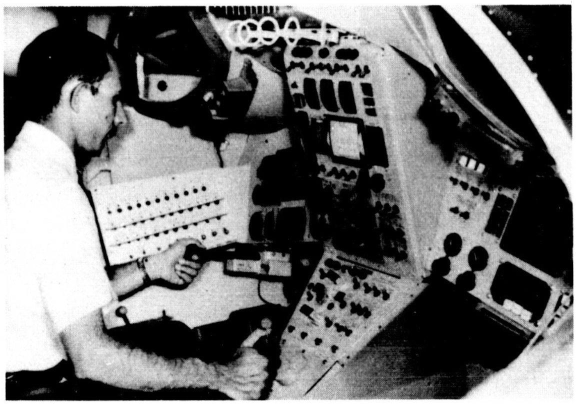


FIGURE 22-5.—Engineering Simulator.

FIGURE 22-6.—Mission Simulator at the Manned Spacecraft Center.

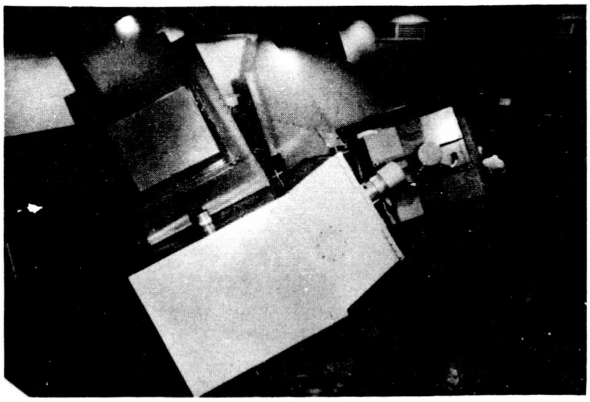


FIGURE 22-7.—Mission Simulator at the Kennedy Space Center.

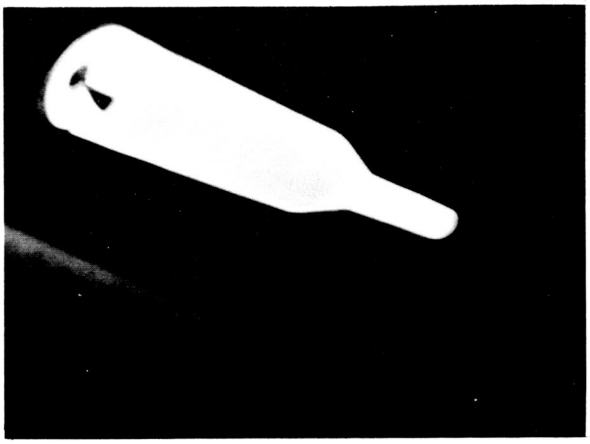


FIGURE 22-8.—Rendezvous target as seen through window of Mission Simulator at the Kennedy Space Center.

model of the rendezvous target vehicle and a gimbal-mounted television camera with air-bearing transport. The earth scene is a television picture of a 6-foot-diameter globe.

The crew stations for the simulators contain actual flight controls and displays hardware. The simulator at Cape Kennedy, which the crews utilize during the last 2 months prior to a flight, contains the exact cockpit stowage configuration in terms of operational equipment, experiments, cameras, and food. To provide additional crew comfort during the longer rendezvous simulations, the crew station was designed to pitch forward 30° from the vertical, thereby raising the crewman's head to the same level as his knees. Mission training is divided into segments so that no training period exceeds 4 hours. The simulator also generates approximately 300 telemetry signals which are transmitted to the worldwide communications and tracking network for use during integrated network simulations.

A part-task trainer which provides a full-scale dynamic simulation of the close-in formation flying and docking maneuvers is the Translation and Docking Simulator (fig. 22–10). The Gemini Agena target vehicle mockup is mounted on air-bearing rails and moves in two degrees of translation. The Gemini spacecraft is mounted in a gimbaled ring on another air-bearing track and incorporates the remaining four degrees of freedom. Cockpit controls activate a closed-loop control system consisting of an analog computer, servo amplifiers, and hydraulic servos. This simulator, located in the flight crew simulation building at Houston, has a maneuvering envelope defined by the size of the enclosure, which is 100 by 60 by 40 feet. Lighting configurations simulate day, night, and various spacecraft-target lighting combinations.

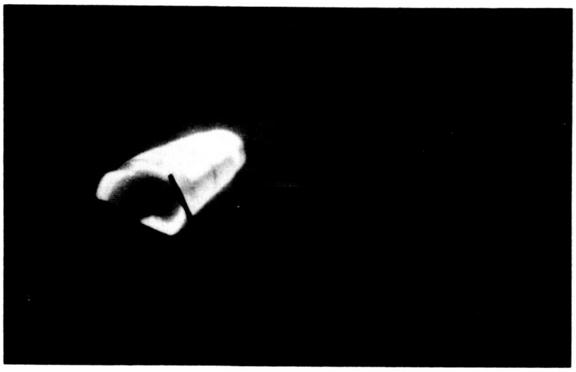


FIGURE 22–9.—View through window of Mission Simulator at the Manned Spacecraft Center.

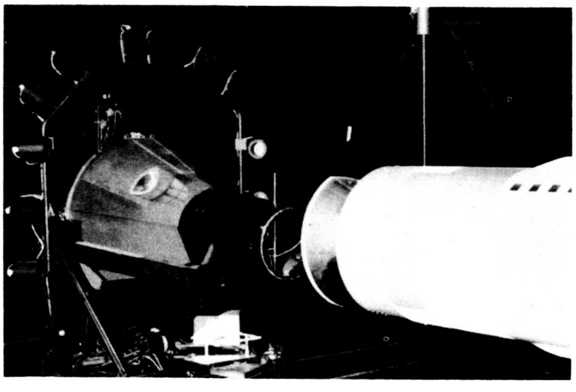


FIGURE 22–10.—Translation and Docking Simulator.

Retrofire and Reentry

The retrofire maneuver involves manual attitude control during solid retrorocket firing. The primary attitude reference is the "8-ball" attitude indicator. In the event of inertial platform or indicator failure, the window view of the earth's horizon and the rate gyro displays are used.

Associated with the retrofire maneuvers are the adapter separation activities. Approximately 1 minute prior to retrofire, the equipment adapter is separated to permit firing of the solid retrorockets, which are fixed to the retroadapter adjacent to the spacecraft heat shield. The equipment adapter is separated by three pilot actions: individual initiation of pyrotechnic guillotines for the orbital-attitude-and-maneuver-system lines, the electrical wiring, and then firing of the shaped charge which structurally separates the adapter from the spacecraft. After retrofire, the retroadapter separation is manually sequenced.

Reentry control logic is displayed to the pilots as roll commands in conjunction with down-range and cross-range errors. The down-range and cross-range error displays involve the pitch and yaw flight-director needles (fig. 22–3), which are used in a manner similar to the localizer and glide-slope display for an aircraft instrument-landing system. During the atmospheric deceleration portion of the reentry, the pilot must damp oscillations in pitch and yaw and, in addition, must control the roll in order to obtain proper lift-vector orientation. Good static and aerodynamic stability characteristics create a relatively easy damping task for the pilot.

Deployment of the drogue and the main parachutes is accomplished by the crew, based on altimeter readout and two discrete light indications which are triggered by separate barometric pressure systems.

The Gemini Mission Simulators have provided the majority of the training during the retrofire and reentry phase. Early familiarization and procedures development were conducted in the Gemini Part Task Trainer at the Manned Spacecraft Center, and in the engineering simulator at the spacecraft contractor's facility.

Systems Management

Overall management of spacecraft systems is similar to the concept used for aircraft. As shown in figure 22–3, the flight parameters are displayed directly in front of the pilots; the circuit breakers are located peripherally on the left, overhead, and right consoles; and the environmental control, fuel-cell heater, propulsion, communications, inertial platform, rate-gyro controls, and water-management panels are located on consoles between the pilots. The spacecraft separation, adapter separation, retro-rocket jettison, and deployment switches are guarded and interlocked with circuit breakers to prevent inadvertent operation during sleep periods, suit removal, and extravehicular operations.

The Agena control panel is located on the right side of the spacecraft. The pilot normally operates this control panel; however, by using a foot-long probe, called a swizzle stick, the simple toggle activities can be accomplished by the command pilot, even while he is wearing a pressurized suit.

Prior to the initial systems training on the Gemini Mission Simulator, six breadboard-type Gemini systems trainers are used for early familiarization. Figure 22–11 shows the electrical system trainer which portrays the control circuits and operational modes.

Extravehicular Activity

The crew procedures associated with extravehicular activity may be divided into two categories: first, preparation for extravehicular activity, which involves donning the specialized equipment; and second, flying the maneuvering unit and carrying out specific extravehicular tasks. Prior to egress, both crewmembers require approximately 2 hours of preparation for extravehicular activity. This activity includes removing the umbilical, the chest pack, and all other extravehicular equipment from stowage; then donning and checking out the equipment in the proper sequence. Each crewmember checks the life-support connections of the other crewman as each connection is made. Training for this phase of the extravehicular operation was carried out in specially prepared, static spacecraft mockups (fig. 22–12) located in the flight crew simulation building at the Manned Spacecraft Center, and in the Gemini Mission Simulator at Cape Kennedy. Also, training for egress and ingress and for extravehicular experiments is carried out under zero-gravity conditions in an Air Force KC–135 airplane (fig. 22–13) at Wright-Patterson Air Force Base. Spacecraft cockpit, hatches, and adapter section are installed in the fuselage for use during the aircraft flights. A 3-hour flight includes approximately 45 zero-g parabolas of 30 seconds'

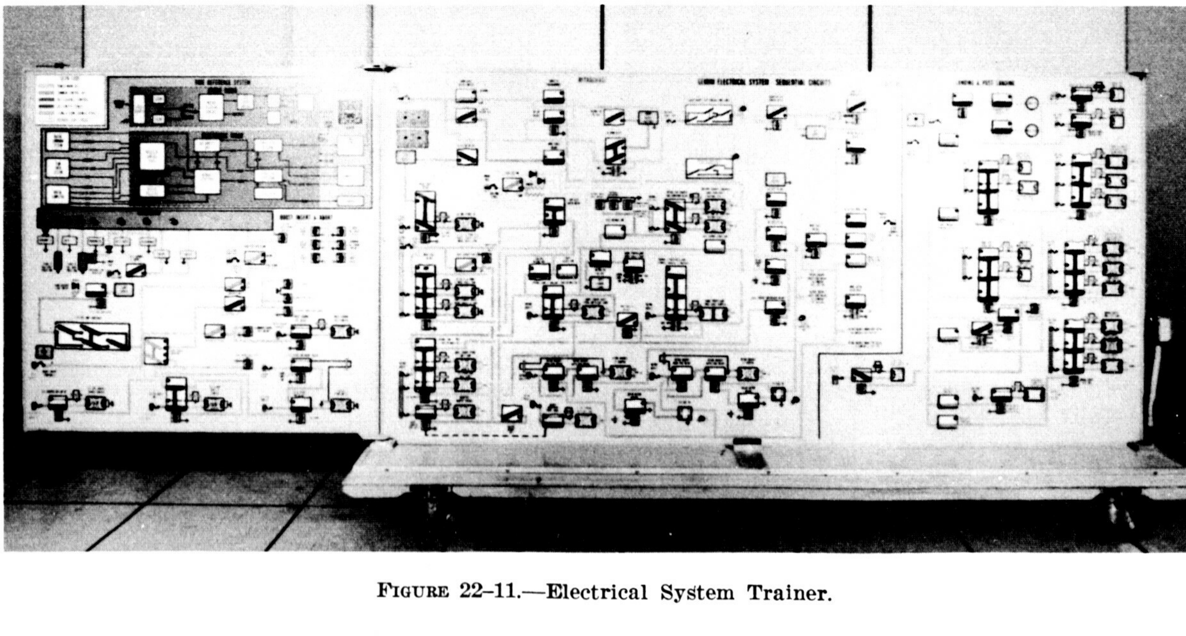


FIGURE 22–11.—Electrical System Trainer.

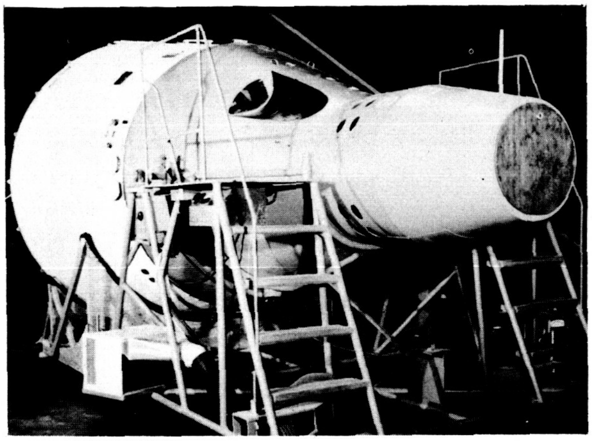


FIGURE 22-12.—Spacecraft mockup.

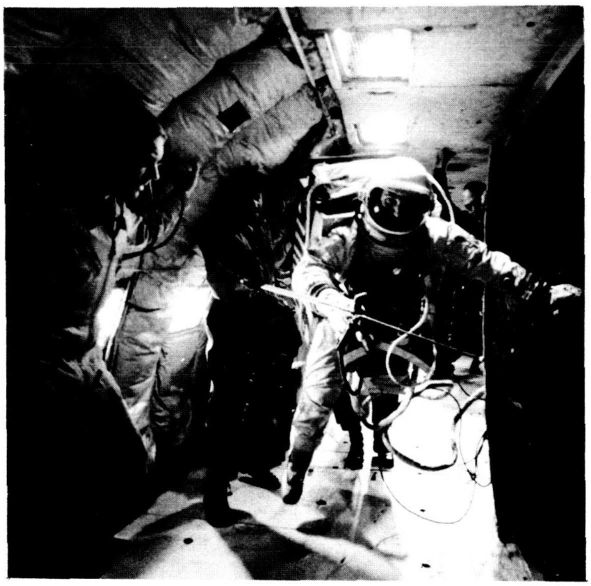


FIGURE 22-13.—Zero-g training in KC-135 airplane.

duration. The zero-g parabola involves a 45° pullup to 32 000 feet, then a pushover to zero-g with a minimum airspeed of 180 knots on top, followed by a gravity pitch maneuver to a 40° dive, after which a 2g pullout is accomplished with a minimum altitude of approximately 24 000 feet and an airspeed of 350 knots. The majority of the training for the extravehicular maneuvering procedures is carried out on three-degree-of-freedom simulators utilizing air bearings to achieve frictionless motion. Figure 22-14 shows a typical training scene, with the crewman in a pressurized suit practicing yaw control with a Gemini IV-type handheld maneuvering unit. The handheld unit (fig. 22-15) produces 2 pounds of thrust in either a tractor or pusher mode, as selected by a rocking trigger. The pilot directs the thrust with respect to his center of gravity to give a pure translation or to give a combination of translation and rotation. The low thrust level produces angular accelerations sufficiently low so that he can easily control his motion. Although the translation acceleration is also low, approximately 0.01g or ⅓ foot per second per second, this is sufficient thrust to give a velocity of 2 feet per second with a 6-second thrust duration. This general magnitude of velocity will accomplish most foreseeable extravehicular maneuvers.

FIGURE 22-14.—Three-degree-of-freedom air-bearing simulator.

In addition to the launch-abort training discussed previously, other contingency training includes practicing parachute and emergency egress procedures. Figure 22-16 shows parachute training activity which familiarizes the pilots with earth and water landings while wearing Gemini suits and survival equipment. This simplified parachute procedure involves a running takeoff and a predeployed parachute attached to a long cable which is towed by truck or motor launch.

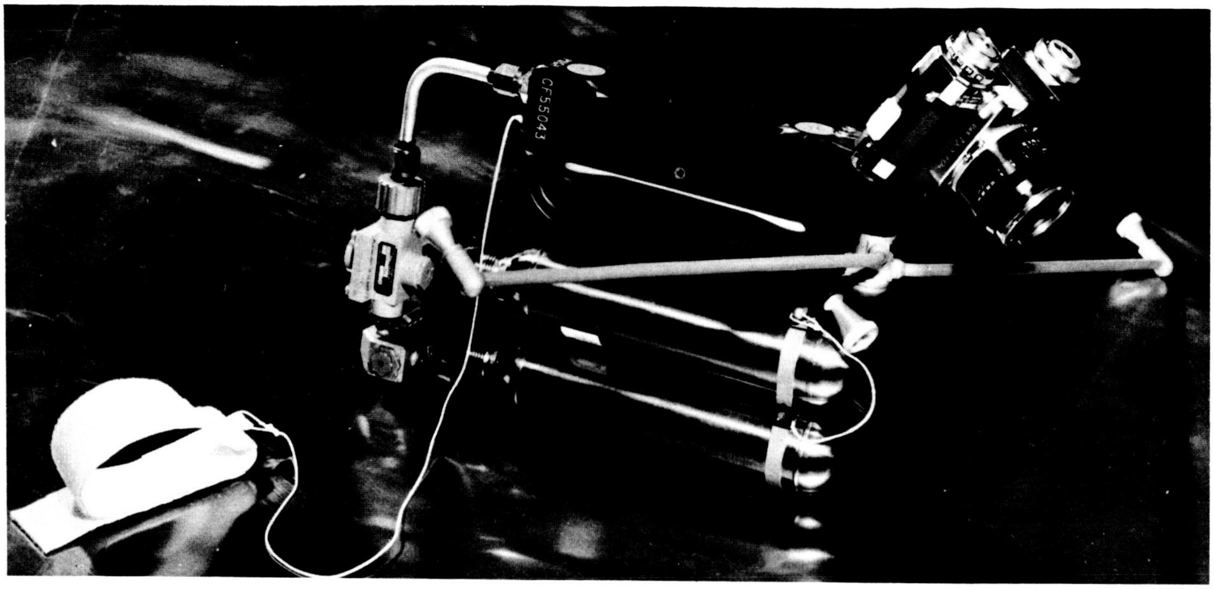


FIGURE 22-15.—Handheld maneuvering unit.

FIGURE 22-16.—Parachute training.

FIGURE 22-17.—Egress training.

Each crew undergoes an egress training session (fig. 22-17) in the Gulf of Mexico. Spacecraft systems procedures, egress techniques, water survival, and helicopter-sling techniques are rehearsed.

Flight Crew Preparation

Thirteen pilots were assigned as prime and backup crewmembers during the first five manned flights. As a partial indication of experience, their military aircraft pilot-rating date, total flight time, and assignment date to the astronaut program are listed in table 22-I. Considering that military aircraft ratings are achieved approximately 1 year after the start of flight training, their pilot experience ranges from 13 to 18 years; total aircraft flight time in high-performance aircraft varies from approximately 3000 to 5000 hours; and active affiliation with the NASA manned space-flight program varies from 20 months to nearly 7 years, at the time of launch. It is of interest to note that the man with the lowest flight time has also flown the X-15 rocket research airplane. They all obtained engineering degrees prior to or during the early stages of their engineer-pilot career. Age within the group ranges from 34 years to 42 years. All have undergone a three-part space-flight preparation program.

TABLE 22–I.—*Gemini Flight Crew Experience Summary*

Mission	Crew	Pilot rating date	Aircraft time, hours	Astronaut program	Flight date
Gemini III	Grissom	1951	4500	4/59	3/23/65
	Young	1954	3540	10/62	3/23/65
	Schirra	1948	3830	4/59	3/23/65
	Stafford	1953	4540	10/62	3/23/65
Gemini IV	McDivitt	1952	3450	10/62	6/ 3/65
	White	1953	4100	10/62	6/ 3/65
	Borman	1951	4940	10/62	6/ 3/65
	Lovell	1954	3550	10/62	6/ 3/65
Gemini V	Cooper	1950	3620	4/59	8/21/65
	Conrad	1954	3460	10/62	8/21/65
	Armstrong	1950	2760	10/62	8/21/65
	See	1953	3960	10/62	8/21/65
Gemini VI–A	Schirra [a]				12/15/65
	Stafford [a]				12/15/65
	Grissom [b]				12/15/65
	Young [b]				12/15/65
Gemini VII	Borman [c]				12/ 4/65
	Lovell [c]				12/ 4/65
	White [d]				12/ 4/65
	Collins	1953	3620	2/64	12/ 4/65

[a] Gemini III backup crew.
[b] Gemini III prime crew.
[c] Gemini IV backup crew.
[d] Gemini IV pilot.

The initial training phase involved a 6-month academic program, as shown in table 22–II. This particular curriculum was presented to the February 1964 group of astronauts. Because of the dual Gemini/Apollo training requirement, the curriculum is somewhat more comprehensive than the courses given to the first two groups.

TABLE 22–II.—*Astronaut Academic Program Basic Curriculum*

Course:	*Class hours*
Geology I	80
Geology II (laboratory—fieldwork)	80
Astronomy (laboratory—planetarium)	30
Math review	20
Flight mechanics	50
Basic aerodynamics	36
Aerodynamics	20
Rocket propulsion	34
Computers	16
Inertial systems	16
Navigational techniques	30
Guidance and control	34
Communications	12
Spacecraft control systems laboratory—simulations	16
Physics of the upper atmosphere and space	18
Basic physiology	32
Flight physiology and environmental systems	34
Meteorology	10
Total	568

The second phase of crew preparation involves assignment to engineering specialty areas. A typical breakdown of engineering categories is as follows:

(1) Launch vehicles
(2) Flight experiments and future programs
(3) Pressure suits and extravehicular activity
(4) Environmental control system, radiation protection, and thermal control
(5) Spacecraft, Agena, and service module propulsion
(6) Guidance and navigation
(7) Communications and tracking
(8) Electrical, sequential, and fuel cell systems
(9) Mission planning
(10) Crew safety, launch operations
(11) Landing and recovery systems

(12) Crew station integration
(13) Space vehicle simulators

The duration of this second phase, which extends to flight assignment date, varied from 8 months to 6 years. The Mercury flight assignment periods were included in phase II of Gemini flight preparation. All pilots, and in particular the Mercury-experienced crews, made many contributions to the design and operational concepts for the Gemini spacecraft.

The final phase begins with flight assignment and occurs approximately 6 months prior to launch date. At the start of this final phase, a detailed training plan is formulated by the training personnel and the assigned flight crew. A typical training schedule is summarized in figure 22-18. The assigned crews begin with detailed systems reviews using the systems trainers at the Manned Spacecraft Center, and actual participation in systems checkout activity at the spacecraft contractor's plant.

Training on the Gemini Mission Simulator starts about 3 months prior to launch. This training is carried out concurrently with all the other preparation activities. The initial training on the simulator is carried out at the Manned Spacecraft Center. Approximately 6 weeks prior to launch, the flight crew moves to Cape Kennedy in order to participate in the final spacecraft checkout and to continue training on the mission simulator.

Training time spent by the flight crews on the trainers and in the major areas is summarized in table 22-III. Differences in the time spent by the crews in the various activities are indicative of the type of missions and objectives.

In preparation for the first manned flight, a considerable number of hours were spent by the crews in the spacecraft systems activities at the spacecraft contractor's plant and with the spacecraft at Cape Kennedy. The extensive number of experiments carried out during the Gemini V and VII missions are reflected by the time spent in the preparation phase. For the first planned docking mission on Gemini VI, the prime crew spent 25 hours in the Translation and Docking Simulator, developing the control procedures for both formation flying and for docking.

Evaluation of Training

Although the adequacy of the astronaut training is difficult to measure, it is important that the value of the training facilities and activities

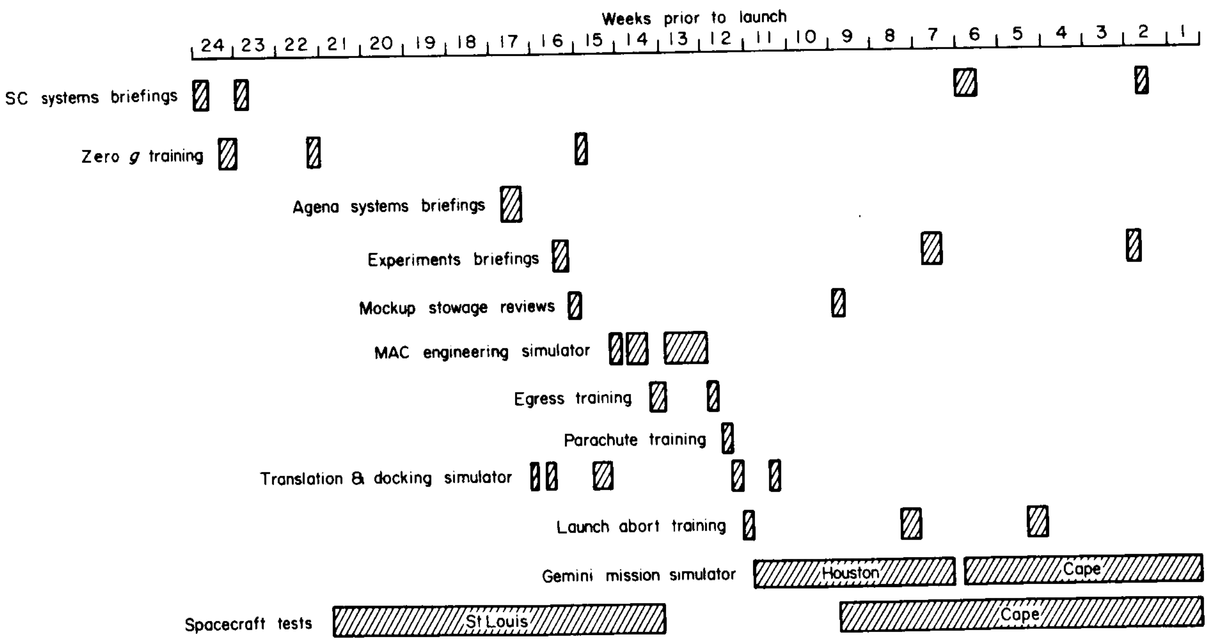


FIGURE 22-18.—Flight crew training schedule.

TABLE 22–III.—*Gemini Flight Crew Training Summary*

[Hours]

Training phase	Gemini III		Gemini IV		Gemini V		Gemini VI–A		Gemini VII	
	Prime	Backup	Prime	Backup	Prime	Backup	Prime[a]	Backup	Prime[b]	Backup
Mission simulator	118	82	126	105	107	110	107	76	113	114
Launch vehicle simulator	17	15	22	22	15	16	6	8	6	7
Docking simulator	1	5	6	6	2	12	25	17	4	4
Spacecraft systems tests and briefings	233	222	120	120	122	128	93	91	150	160
Experiments training	2	2	50	50	150	150	23	22	100	100
Egress and parachute training	18	15	23	23	12	12	6	6	9	13

[a] Prime crew on Gemini VI was backup on Gemini III.
[b] Prime crew on Gemini VII was backup on Gemini IV.

be examined at this point in the program. Comments made by the crews regarding their training are summarized as follows:

(1) Gemini mission simulator
 (a) Most important single training
 (b) Visual simulation invaluable
 (c) High fidelity required
 (d) Accurate crew station/stowage
(2) Spacecraft systems tests and briefings
 (a) Active participation in major spacecraft tests necessary
 (b) Briefings essential
(3) Contingency training
 (a) Egress and parachute training required
 (b) Launch-abort training essential

The crews were unanimous in their assessment of the importance of the Gemini Mission Simulator. The out-the-window visual simulation did not become fully operational until Gemini VI training at Cape Kennedy. The crews agree that this visual simulation is invaluable, particularly for the rendezvous training. Fidelity of hardware and software has been of utmost importance and should not be compromised. Practice in stowing and unstowing all the necessary cockpit gear, together with the operation of the total spacecraft systems, could be done only in the Gemini Mission Simulator, and this practice was found to be essential in establishing final cockpit procedures.

Although the time spent in the spacecraft tests and associated briefings varied with the crews, all crewmembers agreed that, without this participation and insight gained into the systems operation, the mission objectives could not have been carried out as they were.

Training for contingencies is considered by all as an essential part of the training for a flight. Water egress, as well as pad egress from the launch vehicle, is rehearsed by each pilot. Launch-abort training, both on the Dynamic Crew Procedures Simulator at the Manned Spacecraft Center, and the integrated network simulations on the Gemini Mission Simulator at Cape Kennedy, are believed to be very important.

Concluding Remarks

Extension of Gemini mission objectives from the initial three-orbit systems-verification flight to the long-duration missions with rendezvous and extravehicular activities have required a corresponding increase in the scope of simulation capability. The equipment which has been developed plus the experience gained on the simulators and in flight will provide a broad base from which to provide training for future Gemini flights as well as future programs.

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23. SPACECRAFT LAUNCH PREPARATION

By WALTER J. KAPRYAN, *Resident Manager, Gemini Program Office, NASA Kennedy Space Center, and* WILEY E. WILLIAMS, *Manager, Gemini/LEM Operations, NASA Kennedy Space Center*

Summary

This paper presents a general résumé of Gemini spacecraft launch preparation activities. It defines basic test philosophy and checkout ground rules. It discusses launch site operations involving both industrial area and launch complex activities. Spacecraft test flow is described in detail. A brief description of scheduling operations and test procedures is also presented.

Introduction

In order to present the story of spacecraft launch preparation planning for the Gemini Program in its proper perspective, it is pertinent to first outline basic test philosophy and to discuss briefly the experience gained during the Mercury Program, because early Gemini planning was very heavily influenced by that experience. However, as will be pointed out later, actual Gemini experience has permitted some deviation from the ground rules established on the basis of Mercury Program experience.

The major tenets of the NASA test philosophy have been that, in order to produce a flight-ready vehicle, it is necessary to perform a series of comprehensive tests. These involve (1) detailed component level testing, (2) detailed end-to-end individual systems tests, and (3) complete end-to-end integrated tests of the spacecraft systems and between the spacecraft and its launch vehicle wherein the intent is to simulate as closely as practical the actual flight sequences and environment. This sequence of testing begins at the various vendors' plants, with predelivery acceptance tests, progresses through the prime contractor's facility, wherein a complete spacecraft systems test operation is performed, and concludes with the launch site operation. All data are cross-referenced so that the testing at each facility adds to and draws from the results obtained at each of the other facilities.

Test experience during the Mercury Program showed that it was necessary to perform extensive redundant testing in order to expose weak components, to assist in determining design deficiencies, and to continue developing reliability information. The plan that evolved was that, to a large extent, all prime contractor's in-plant tests would be repeated at the launch site. Further, due to the physical arrangement of systems within the spacecraft, it was generally necessary to invalidate more than one system when replacing a faulty component. This, of course, introduced additional testing. Finally, because special aerospace-ground-equipment (AGE) test points were not used, it was necessary to disconnect spacecraft wiring in order to connect test cables. When the wiring was finally connected for flight, additional validation testing was required.

Consideration of these factors on the Mercury program led to the following ground rules for early Gemini launch preparation planning:

(1) Spacecraft design would be of modular form so that simultaneous parallel work and checkout activities could be performed on several modules.

(2) Spacecraft equipment would be arranged for easy accessibility to expedite cabling operations so that component replacement would invalidate only the system affected.

(3) Aerospace-ground-equipment test points would be incorporated on the spacecraft and spacecraft components to minimize the need for disconnecting spacecraft wiring in order to monitor system parameters.

(4) The ground equipment would be designed so that problems could be isolated to the black-box level without requiring component removal from the spacecraft.

(5) The ground equipment to be used at the prime contractor's facility and at the launch site would be identical, where practical, so that test data could be more reliably compared than was possible in the Mercury program.

(6) The complete spacecraft systems test operation at the prime contractor's facility would be repeated at the Kennedy Space Center until such time that experience established no further need for these tests.

As the Gemini Program progressed toward its early operational phase, overall test planning underwent considerable review. The aforementioned ground rules were reexamined repeatedly and evaluated on the basis of the current status of qualification and development testing of Gemini spacecraft equipment. It soon became apparent that the state of the art had advanced to the extent that Gemini equipment was better than Mercury equipment, and some of the redundant testing planned for Gemini could be eliminated. Judicious reduction of redundant testing was very desirable from the standpoint of cost, manpower requirements, schedules, and wear and tear on the spacecraft systems and the test equipment. Accordingly, a decision was made to eliminate the complete repeat of the inplant spacecraft systems test operation at the launch site. However, in order to have a trained Gemini checkout team at the launch site, a special task force comprised of experienced test personnel was organized and sent to the prime contractor's facility for the purpose of participating in the spacecraft systems test operation on at least the first two all-systems spacecraft. At the conclusion of these tests this team returned to the launch site with these spacecraft.

Launch Site Preparation

Industrial Area Activity

The first Gemini spacecraft having all systems installed was spacecraft 2, and, by the time of its delivery to the Kennedy Space Center, the launch-site preparation plan had basically evolved into its present form. All launch-site testing would be performed at the launch complex. Except for special requirements, no spacecraft testing would be performed in the industrial area. Industrial area activity would be confined to only those functions which should logically be performed away from the launch complex, and to preparing the spacecraft for its move to the launch complex. Typical spacecraft industrial area activity is as follows:

(1) Receiving inspection.

(2) Cleanup of those miscellaneous manufacturing activities not performed at the prime contractor's facility, and incorporation of late configuration changes.

(3) Pyrotechnic installation.

(4) Fuel-cell installation.

(5) Flight-seat installation.

(6) Rendezvous and recovery section buildup.

(7) Weight and balance.

(8) Manufacturing cleanup and inspection.

(9) Preparations for movement to the launch complex.

In addition to these typical activities, complete end-to-end propulsion system verification tests were performed with spacecraft 2 and 3. These tests included static firing of all thrusters. They were performed primarily to provide an early end-to-end checkout of the servicing procedures and equipment prior to their required use at the launch complex. A further benefit derived from these tests was the completion of development and systems testing on Gemini hypergolic systems to the point that these specific systems could be committed to flight with a high degree of confidence. A demonstration cryogenic servicing was also performed on spacecraft 2. Spacecraft 3, the first manned Gemini spacecraft, received a communications radiation test at the Kennedy Space Center radar range. This test exercised spacecraft communications in a radiofrequency environment that more closely simulated the actual flight environment than was possible at any other available facility. The remaining nonrendezvous spacecraft did not undergo any systems tests in the industrial area. For the first two rendezvous spacecraft, a radiofrequency and functional-compatibility test between the spacecraft and the target vehicle was also performed at the radar range (fig. 23–1). This particular test is basically a proof-of-design test, and the need for its continuation is being reviewed.

Launch-Complex Operations

A chart of typical launch-complex test operations is presented as figure 23–2. Testing be-

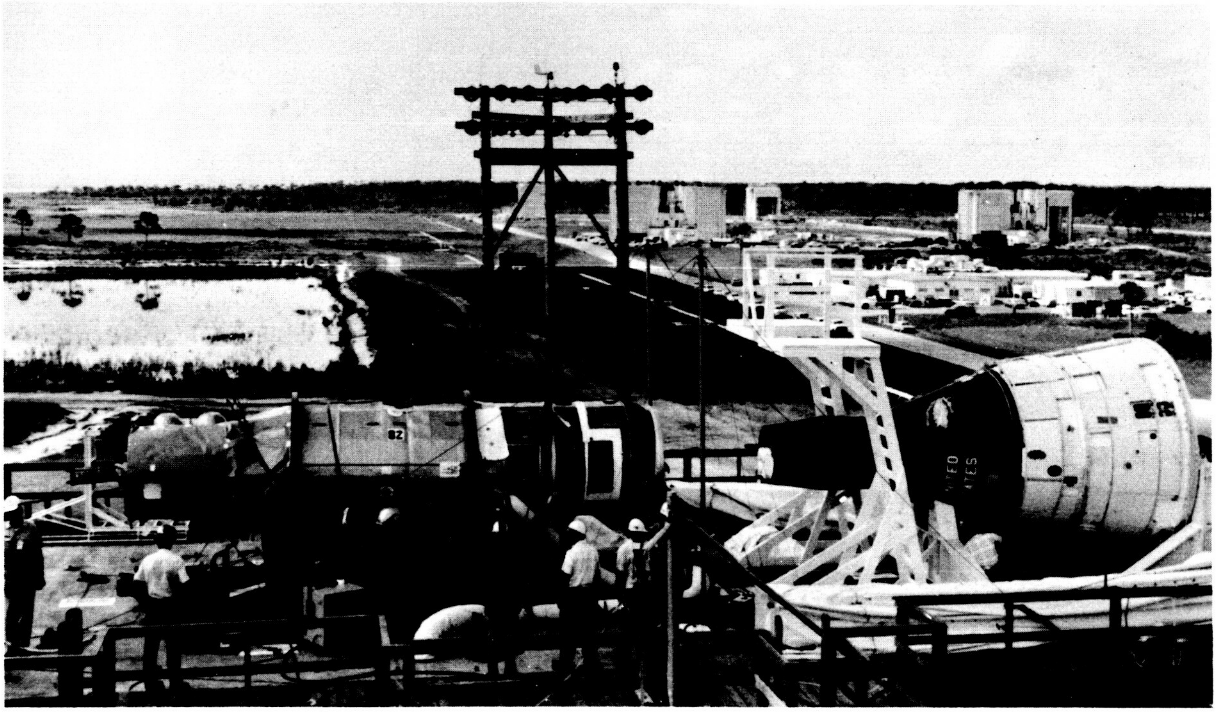

FIGURE 23–1.—Spacecraft and Gemini Agena target vehicle undergoing tests at radar range.

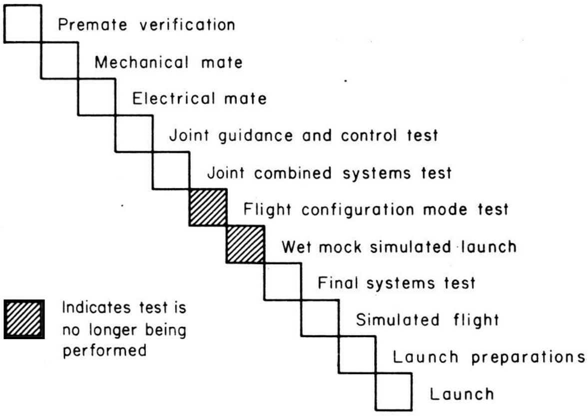

FIGURE 23–2.—Spacecraft test operations performed at launch complex.

gins with premate verification, which consists of thoroughly testing spacecraft systems down to the black-box level. The first fuel-cell activation is performed at this time. Data obtained are compared with data from the spacecraft systems tests at the prime contractor's facility and predelivery acceptance tests at the vendors' plants. The intent of this testing is to integrate the spacecraft with the launch complex and to get a last detailed functional look at all systems, especially those within the adapter, prior to performing mechanical mate and the assumption of integrated tests with the launch vehicle. Typical cabling configurations are shown in the next two figures; figure 23–3 shows the reentry module, and figure 23–4 shows the adapter. Following the successful completion of premate verification, the spacecraft and launch vehicle are mechanically mated. This operation is performed under the direction of a mechanical interface committee, which verifies that all clearances and physical interfaces are in accordance with the specifications.

Following mechanical mate, electrical-interface tests between the spacecraft and the launch vehicle are conducted to functionally or electrically validate the interface. All signals capable of being sent across the interface are tested in all possible modes and redundant combinations. Following the electrical mate, the joint guidance and control tests are performed. These tests consist largely of ascent runs involving primary guidance and switchover to secondary guidance. During these tests, such items as secondary static gains, end-to-end phasing, and switchover fade-in discretes are also checked for specification performance.

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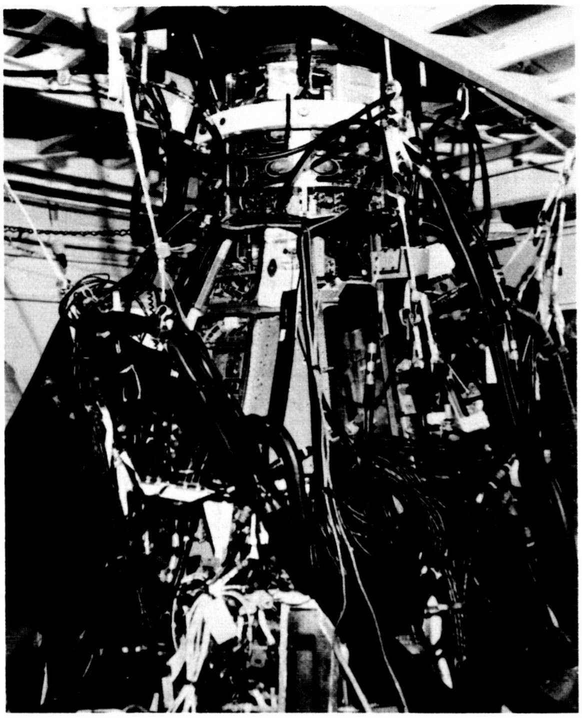

FIGURE 23-3.—Spacecraft reentry section with cables attached for systems test at launch complex.

Following the joint guidance and control tests, a joint combined systems test is performed. The purpose of the joint combined systems test is to perform a simulated mission. It is normally performed in three parts:

(1) Part 1 consists of exercising all abort modes and command links, both radiofrequency and hardline.

(2) Part 2 consists of an ascent run through second-stage engine cutoff, wherein there is a switchover from primary to secondary guidance.

(3) Part 3 consists of a full-blown simulated mission and involves a normal ascent on primary guidance, orbit exercises applicable to the specific mission, and rendezvous and catchup exercises. Finally, retrofire with a complete reentry to landing is simulated. Suited astronauts are connected to the environmental control system during this test. Thus, the joint combined systems test is a comprehensive, functional, integrated test of the entire space vehicle and serves as the first milestone for alerting the worldwide network and recovery forces to prepare to man their stations for launch.

FIGURE 23-4.—Spacecraft adapter assembly with cables attached for systems test at launch complex.

Following the joint combined systems test, a flight configuration mode test has been performed. This test simulates an ascent run as close as possible to the true launch environment. For this test, all of the ground equipment was disconnected, all launch vehicle and spacecraft umbilicals were pulled in launch sequence, and the total vehicle was electrically isolated from the launch complex. All monitoring of systems performance was through cabin instrumentation and telemetered data. This test unmasked any problems that may have been obscured by the presence of the aerospace ground equipment and demonstrated systems performance in flight configuration. A test such as this was very valuable to the Gemini Program in its earlier phases; however, now that the program has reached its present phase of stabilized and proved flight and ground equipment configuration, the value of the test is somewhat diminished. For that reason, beginning with Gemini VII the flight configuration mode test was no longer being performed. However, since certain sequential functions cannot be demonstrated without umbilical eject, the umbilical-pull portion of this test has been retained and has been incorporated as an additional sequence of one of the other test days.

The wet mock simulated launch is a dress rehearsal of the launch operation itself. Both launch vehicle and spacecraft are serviced and prepared exactly as though they were to be launched. The complete countdown is rehearsed and runs to T−1 minute. Astronaut ingress is performed exactly the same as on launch day. This operation actually includes all launch preparation functions and starts on F−3 day. This test is primarily an operational demonstration on the part of the launch team and serves as the second major milestone of an impending launch. This test, too, is of greatest value in the early operational phases of a program. As the program progresses, the wet mock simulated launch provides diminishing returns. The last spacecraft for which a complete wet-mock-simulated launch was performed was spacecraft 6 prior to its first launch attempt. It is doubtful that any further complete wet-mock-simulated launches will occur.

For the rendezvous phase of the program, a simultaneous launch demonstration is being performed in lieu of the wet-mock-simulated launch. This test is a coordinated countdown of the Atlas-Agena and the Gemini space vehicles. It simulates an Atlas-Agena launch and the first orbit of the Agena. As during wet mock simulated launch, the spacecraft and Gemini launch vehicle count runs to T−1 minute. The simultaneous launch demonstration, however, does not include the servicing of any of the vehicles, nor does it include the precount and midcount. It is being performed closer to launch than was the wet-mock-simulated launch and will be discontinued when experience shows it to be no longer necessary.

The deletion of the wet-mock-simulated launch improves the launch-complex schedule by several days, and also eliminates the requirement for an early mechanical mate. Since the erector is lowered during wet-mock-simulated launch, the spacecraft must be mechanically mated to the launch vehicle for this test. Therefore, its elimination permits integrated testing to continue while demated, by the utilization of an electrical interface jumper cable. Thus, any activities requiring access into the spacecraft adapter can be performed much later in the sequence of launch-complex operations than was heretofore possible. Spacecraft 8, for example, is not scheduled to be mechanically mated until after the completion of final systems test.

Following the wet-mock-simulated launch, final spacecraft systems tests are performed. They encompass the same scope as during premate verification. These tests provide final detailed component-level data prior to launch. At this time, all data are closely scrutinized for any trends indicating degraded performance. Following the final systems test, the final simulated flight is conducted. This test is very similar to the joint combined systems test. The runs are identical, and suited astronauts participate. One important additional function performed during this test is to utilize high-energy squib simulators during appropriate sequencing functions involving pyrotechnics. Thus, all pyrotechnic circuits experience electrical loads just as though actual squibs were being fired. The simulated flight is the last major test of the spacecraft prior to launch. Immediately after the simulated flight, final launch preparations begin, leading to the precount on F−3 day. The primary purpose of the precount is to perform power-on stray voltage checks prior to making final flight hookup of spacecraft pyrotechnics.

Following the precount, final servicing operations begin, and the spacecraft buttoning-up process starts. On F−1 day the midcount is performed. At this time the spacecraft is remotely powered up in order to demonstrate the safety of the pyrotechnic configuration. The fuel cells are activated during the midcount and remain powered up through launch.

The final countdown is started early on launch day and is of 6 hours' duration. During the count, an abbreviated check of all systems is made and is timed to be completed prior to the schedule target vehicle launch so that during the critical time period following that launch, a minimum of test activity is required. This approach has put us in the posture of being exactly on time at T−0 for the two complete rendezvous countdowns thus far.

The sequence of testing just described provides for several distinct milestones for gaging test progress, and it also provides for the logical resumption of testing in the event a test recycle is required, as was the case during the Gemini VI mission. Following the inflight failure of the Agena target vehicle and the subsequent decision to attempt a double spacecraft rendezvous, spacecraft 6 was removed from the launch complex and essentially placed in bonded storage. Immediately after the launch of spacecraft 7, spacecraft 6 was returned to the launch complex. Testing resumed with final systems test, included the final simulated flight, and concluded with the launch. Thus, in a matter of days, a complete new set of test data was obtained and correlated with the data from the previous more-extended spacecraft 6 checkout operation and permitted the spacecraft to be launched with a high degree of confidence. It goes without saying that the Gemini launch vehicle test plan was equally flexible, or the rapid recycle could never have been performed.

The waterfall chart shown in figure 23–2 does not, of course, represent all of the spacecraft test activity at the launch complex. For example, for the Gemini II and III missions an extensive electrical-electronic interference investigation was conducted. Special instrumentation was installed to monitor the critical spacecraft and launch vehicle interface circuits. The performance of these tests basically added another joint combined systems test to the flow plan. Also, cabin-leak rates must be determined for all spacecraft. This chart does not present any experiment test activity, which for some missions is of significant magnitude. In general, these activities are scheduled on a parallel basis with other activities, but at times they do add serially to the schedule.

A significant portion of the effort expended at the launch complex is not directly related to the performance of tests. For example, the following servicing operations are required:

(1) Hypergolic and pressurant servicing of the propulsion system.

(2) Cryogenic servicing for the fuel cells and the environmental control system.

(3) Servicing of secondary oxygen.

(4) Replacement of the lithium hydroxide canister within the environmental control system.

(5) Sterilizing and servicing of the water management system.

Certain experiments also have special servicing requirements and crew-station stowage exercises are required, to name but a few of the nontest functions being performed. The incorporation of a few configuration changes must also be anticipated. In order to project realistic launch dates, sufficient allowances must be provided in the overall launch-complex schedule for all of these activities.

Scheduling

For a normal mission operation, launch-complex test activities are scheduled on a two-shift, 5-day-week basis. The third shift and weekends are utilized for shop-type activity and troubleshooting, as required. The weekend also serves as a major contingency period in the event of failure to maintain schedules during the normal workweek. Daily scheduling meetings are held, during which all test and work activities are scheduled for the ensuing 24 hours. Scheduling on this basis has resulted in meeting projected launch schedules for most missions, and has enabled management to make realistic long-range program commitments. The only spacecraft for which there has been any significant differences between projected and actual schedules is spacecraft 2. Much of this discrepancy can be accounted for by the fact that it was the first spacecraft to use the complete launch complex. During the operations for spacecraft 2, there were many launch-complex problems, primarily associated with electrical shielding and grounding. Test procedures reflected the early stage of the program and also required significant refinement. The lessons learned with spacecraft 2 have enabled subsequent spacecraft to progress substantially on or ahead of schedule.

Test Procedures

All significant test operations are performed utilizing formal test procedures. Every step of the test is defined in the procedure. All procedures and the data obtained are certified as having been accomplished by inspection personnel. Any deviations to these procedures are documented in real time and are also certified by inspection. The program, therefore, has a complete documented file of every important spacecraft test performed at the Kennedy Space Center since the inception of the program.

Spacecraft testing in the Gemini Program is a joint NASA/contractor effort. The tests are conducted for the NASA by the contractor, with the NASA lead engineers working closely with their contractor counterparts. This method of operation provides a system of built-in checks and balances and enables the NASA management to keep fully aware of test progress so that necessary management decisions can be readily made. This method of operation has contributed significantly to the success of manned space-flight programs to date.

Concluding Remarks

Experience with the Gemini Program has demonstrated the basic soundness of the early program planning. Further, the Gemini Program has benefited greatly from Project Mercury experience. For example, the more realistic qualification requirements for Gemini equipment have reduced the incidence of equipment failures significantly over that of the Mercury Program. This factor has contributed to a test environment requiring much less repeat testing. The fact that the program was successfully able to eliminate the repeat of the spacecraft systems test operation at the launch site reduced spacecraft operations at the launch site from a projected 125 working days to approximately 45 working days at the present phase of the program. Spacecraft test plans are continually being reevaluated from the standpoint of still further streamlining. Gemini ground equipment has provided a much greater capability to monitor systems performance in detail so that the spacecraft can be committed to launch with ever greater confidence. Greater equipment accessibility has also contributed significant time savings. The net result has been a test flexibility that has enabled the program to accelerate schedules when necessary, and has enabled the program to recover from the catastrophic target vehicle flight of last October 25 with a rapid recycle and the highly successful rendezvous in space during Operation 76. This experience is evidence of a maturing manned space-flight effort. Extension of this experience should contribute significantly to more efficient utilization of money and manpower in future space programs.

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24. SPACECRAFT LAUNCH-SITE PROCESSING

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Summary

In this report, the data of interest with regard to the processing of the Gemini spacecraft are analyzed. The time required for processing any particular spacecraft is dependent not only upon the tests required but also upon the number of manufacturing tasks, the number of tasks that can be worked concurrently, and the amount of time available. The effort required to accomplish modifications, replacements, and repairs is accomplished in parallel with other activities and does not directly affect the schedule.

The influence of discrepancies found during testing and the number of discrepancies per testing hour can be predicted. In addition, such other parameters as the number of processing tasks and the number of testing shifts have been suitably combined with other factors into a mathematical model for predicting the number of days required at launch complex 19 at Cape Kennedy, Fla.

Introduction

The time required to complete the launch-pad processing of a Gemini spacecraft depends on several factors, such as testing, modification, part replacement, servicing time, and post-testing activities. Data on these factors have been analyzed and combined into a mathematical model which serves as a basis for predicting the launch-pad processing time required before a Gemini spacecraft can be launched from Cape Kennedy, Fla. Monitoring of the elements of the mathematical model provides a means of evaluating performance.

This model has been prepared by the Spacecraft Operations Analysis Branch at the Kennedy Space Center, using the following sources of data:

(1) Spacecraft test and servicing procedures from the spacecraft prime contractor.
(2) Inspection reports.
(3) The spacecraft test conductor's log.
(4) Daily activity schedules.
(5) Meeting attendance.
(6) Systems engineering reports.
(7) Operating personnel.

Clarification of the source material was obtained from systems engineers and spacecraft test conductors.

Spacecraft Schedule Performance

A comparison of schedules with performance (table 24–I) shows that spacecraft 2 was the only spacecraft that did not meet the planned checkout schedule. However, the spacecraft can be considered a special case for analysis purposes, since it was the first to use the new test facilities and flight hardware. This is supported by the fact that 102 aerospace-ground-equipment interim discrepancy records were recorded, as compared with 36 spacecraft interim discrepancy records. An interim discrepancy record is prepared whenever a problem is encountered on either ground equipment or on the spacecraft. The spacecraft discrepancies did not contribute significantly to the schedule slippage.

The original schedule for spacecraft 5 was exceeded by 15 days. This was caused by a 13-day extension due to several effects other than spacecraft testing, interim discrepancy records, troubleshooting, servicing, or modification, and is not included in this discussion. There was also a 2-day slip in the launch of spacecraft 5 caused by a countdown scrub.

Analysis of Spacecraft Processing Factors

Effects of Major Spacecraft Tests

The original checkout schedule consisted of 10 major tests. Later, four of the tests were combined into two, leaving eight major tests. The data from these tests form the basis for this phase of the evaluation.

TABLE 24-I.—*Scheduled Versus Actual Testing Time*

Planned test schedule, days				Actual performance, days				
Spacecraft	Prepad [a]	Pad [b]	Total	Prepad [a]	Pad [b]	Countdowns		
						1st	2d	3d
2	16	42	58	28	53	81	122	
3	24	53	77	31	47	78		
4	12	48	60	10	51	61		
5	7	43	50	7	56	63	65	
6	30	53	83	36	47	83	131	[c] 134
7	21	36	57	21	36	57		

[a] Testing before the spacecraft is installed on the launch vehicle at launch complex 19.

[b] Testing after the spacecraft is installed on the launch vehicle.

[c] The third countdown for spacecraft 6 required an additional 51 days—38 prepad days and 13 pad days.

The majority of the scheduled tests were accomplished in the time allotted. Reruns of test sequences and troubleshooting were, on occasion, accomplished in times other than that scheduled, but in the majority of cases this testing and troubleshooting were done in parallel with the daily work schedule.

Only a minor portion of the troubleshooting was performed in serial time, which is time that delays completion of a particular task. Analysis of test preparation, testing, and troubleshooting times revealed that—

(1) Serial troubleshooting time can be estimated as 0.2 shift for each shift of testing.

(2) The test times (table 24-II and fig. 24-1) for individual tests provide a good basis for future planning.

(3) The time used for test preparation will increase as the time allotted increases.

(4) Five shifts were required, on the average, for spacecraft 3 through 7 serial troubleshooting time.

Figure 24-1 shows the distribution of the test and serial troubleshooting times. The data in this figure have been combined according to the test sequence evolution and are displayed on the basis of major tests.

Effects of Spacecraft Discrepancies

The original spacecraft test sequence consisted of 10 major tests. On spacecraft 4, the electrical interface and integrated validation

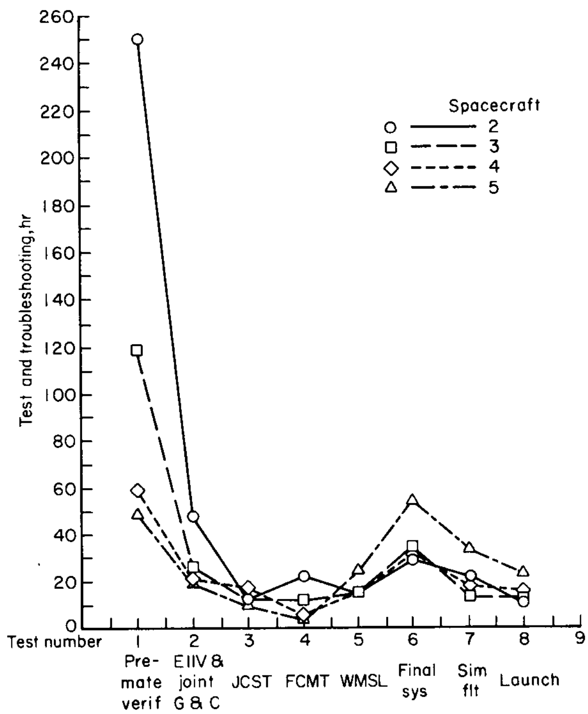

FIGURE 24-1.—Test and troubleshooting time for individual tests.

test and the joint guidance and control test were combined and performed as one test. On spacecraft 5, the premate systems test and the premate simulated-flight test were combined to form the premate verification test. As a result, the test sequence has evolved to the eight major tests shown in table 24-II.

TABLE 24-II.—*Spacecraft Performance Summary*

Test	Spacecraft	SEDR No.	Interim discrepancy records				Setup time		Testing time		Serial troubleshooting time		Total				Modification time	Discrepancies	Replacement items
													Setup plus testing time		Testing plus troubleshooting				
			Spacecraft	AGE [a]	Unclassified	Total	Shifts	Hours	Shifts	Hours	Shifts	Hours	Shifts	Hours	Shifts	Hours			
1. (a) Premate systems test	2	453	7	53	0	60	9	72	6.6	53	10.6	85	26.2	210	17.2	138			
	3	--------	2	28	8	38	9	72	8.3	66	2.5	20	19.8	158	10.8	86			
	4	--------	7	10	11	28	12	96	4.8	38.5	.1	1	16.9	135.5	4.9	39.5			
(b) Premate simulated fit	2	454	11	25	5	41	0	0	6.5	52	7.5	60	14	112	14	112			
	3	--------	1	5	4	10	0	0	2.6	21	1.5	12	4.1	33	4.1	33			
	4	--------	1	1	6	8	1.5	12	2.5	20	0	0	4.0	32	2.5	20			
(c) Premate verification	5	453	11	18	14	43	6	48	2.9	23	3.3	26	12.2	97	6.2	49			
	6	--------	6	14	7	27	7	56	4.8	38.5	.9	7	12.7	101.5	5.7	45.5			
	7	--------	13	7	8	28	9	72	5.5	44	.1	1	14.6	117	5.6	45			
2. (a) Electrical interface and integrated validation	2	456	1	4	0	5	1	8	3.2	25.5	.5	4	4.7	37.5	3.7	29.5			
	3	--------	0	1	1	2	1	8	1.1	9	.8	6	2.9	23	1.9	15			
(b) Joint guidance and control	2	464	2	1	1	4	1.5	12	2.1	17	.3	2	3.9	31	2.4	19			
	3	--------	0	1	1	2	1.5	12	1.5	12	0	0	3.0	24	1.5	12			
(c) Electrical interface and integrated validation and joint guidance and control	4	456	1	1	0	2	6	48	2.7	21.5	0	0	8.7	69.5	2.7	21.5			
	5	--------	0	3	0	3	2	16	2.3	18.5	.3	2.5	4.6	37	2.6	21			
	6	--------	0	1	0	1	5	40	2.3	18.5	0	0	7.3	58.5	2.3	18.5			
	7	--------	1	3	2	6	6	48	2.4	19.5	.4	3	8.8	70.5	2.8	22.5			
3. Joint combined system test	2	457	4	4	4	12	1	8	1.4	11	.1	.50	2.5	19.5	1.5	11.5			
	3	--------	2	5	1	8	1	8	1.3	10	.6	5	2.9	23	1.9	15			
	4	--------	5	5	2	12	6	48	1.5	12	.7	6	8.2	66	2.2	18			
	5	--------	4	7	3	14	1.5	12	1.5	12	0	0	3.0	24	1.5	12			
	6	--------	3	1	2	6	3	24	1.1	9	0	0	4.1	33	1.1	9			
	7	--------	5	4	2	11	6	48	1.4	11	.1	1	7.4	60	1.5	12			
4. Flight configuration mode test	2	459	1	1	1	3	2	16	2.0	16	.9	7	4.9	39	2.9	23			
	3	--------	1	2	0	3	1.5	12	.8	6	.8	6	3.1	24	1.6	12			
	4	--------	1	0	0	1	1	8	.6	5	.4	3	2.0	16	1.0	8			
	5	--------	1	2	0	3	2	16	.6	5	.2	1.3	2.8	22.3	.8	6.3			
	6	--------	0	1	1	2	6	48	1.4	11.5	0	0	7.4	59.5	1.4	11.5			
	7	--------	N/A																
5. Wet mock simulated launch	2	458	5	3	1	9	9	72	1.3	10	.6	5	10.9	87	1.9	15			
	3	--------	5	7	5	17	9	72	1.9	15	0	0	10.9	87	1.9	15			
	4	--------	0	5	4	9	9	72	1.9	15	0	0	10.9	87	1.9	15			
	5	--------	7	7	0	14	9	72	2.6	20.5	.4	3	12.0	95.5	3.0	23.5			
	6	--------	2	8	5	15	13	104	2.6	21	.8	6	16.4	131	3.4	27			
	7	--------	N/A	--------	--------	--------	--------	--------	--------	--------	--------	--------	--------	--------	--------	--------			

[a] Aerospace ground equipment.

TABLE 24-II.—*Spacecraft Performance Summary*—Continued

Test	Spacecraft	SEDR No.	Interim discrepancy records				Setup time		Testing time		Serial troubleshooting time		Total				Modification time	Discrepancies	Replacement items
													Setup plus testing time		Testing plus troubleshooting				
			Spacecraft	AGE [a]	Unclassified	Total	Shifts	Hours	Shifts	Hours	Shifts	Hours	Shifts	Hours	Shifts	Hours			
6. Final systems test	2	460	2	2	2	6	1	8	3.1	25	0.8	6	4.9	39	3.9	31			
	3		2	4	1	7	2	16	3.0	24	1.1	9	6.1	49	4.1	33			
	4		5	5	3	13	3	24	3.6	28.5	0	0	6.6	52.5	3.6	28.5			
	5		6	15	4	25	4	32	4.6	37	2.5	20	11.1	87	7.1	57			
	6		7	6	4	17	11	88	4.4	35	.4	3	15.8	126	4.8	38			
	7		8	2	3	13	2	16	3.9	31	0	0	5.9	4.7	3.9	31			
7. Simulated flight	2	461	0	6	1	7	1.5	12	2.0	16	.6	5	4.1	33	2.6	21			
	3		4	2	2	8	1	8	1.4	11	.5	4	2.9	23	1.9	15			
	4		6	2	6	14	3	24	2.2	17.5	.1	1	5.3	42.5	2.3	18.5			
	5		4	6	4	14	4	32	3.4	27.5	.9	7.5	8.3	67	4.3	33			
	6		3	9	1	13	9	72	2.9	23	1.9	15	13.8	110	4.8	38			
	7		2	4	1	7	3.5	28	2	16	.4	3	5.9	47	2.4	19			
8. Launch	2	463	3	3	1	7	10.5	84	1.4	11	0	0	11.9	95	1.4	11			
	3		0	0	0	0	10.5	84	1.6	12.5	.1	.5	12.2	97	1.7	13			
	4		1	0	1	2	10.5	84	1.6	13	.1	1	12.2	98	1.7	14			
	5		3	3	3	9	10.5	84	2.3	78	.4	3.5	13.2	105.5	2.7	21.5			
	6		2	3	0	5	10.5	84	1.7	13.3	0	0	12.2	97.3	1.7	13.3			
	7		2	4	2	8	10.5	84	2.2	17.5	0	0	12.7	101.5	2.2	17.5			
Total	2		36	102	16	134	36.5	292	29.6	236.5	21.6	174.5	87.7	703	51.2	411	98	327	42
	3		17	55	23	95	36.5	292	23.5	186.5	7.9	62.5	67.9	541	31.4	249	99	278	20
	4		27	29	33	89	52.0	416	21.4	170.5	1.5	12	74.9	598.5	22.9	182.5	129	218	22
	5		36	61	28	125	39	312	20.2	161.5	8	63.8	67.2	537.3	28.1	225.3	85	258	44
	6		23	43	20	86	64.5	516	21.2	169.8	4	31	89.7	716.8	25.2	200.8	83	332	42
	7		31	24	18	73	37	296	17.4	139	1	8	55.4	443	18.4	147	89	266	46

Of the total interim discrepancy records occurring in a test sequence, 31 to 40 percent occurred during the first test of the sequence. The wide range of interim-discrepancy-record occurrence (28 to 60) in the initial test is caused by modifications made on the test complex between missions and by methods which were, as yet, insufficient for verifying that the complex is in optimum operational condition. In this analysis, the first test has been deleted to avoid biasing the test average.

Table 24–III shows the average number of interim discrepancy records experienced by each spacecraft, exclusive of the first test. The high incidence of these records for spacecraft 2 was expected. The averages for spacecraft 3, 4, 6, and 7 are considered normal (accumulative average: 8.8). However, the high average experienced on spacecraft 5 was not anticipated. It is attributed to the large increase in ground equipment and unclassified interim discrepancy records which occurred during the last three tests; prior to those tests, the number of these records had been no higher than predicted. The high incidence of records for spacecraft 5 might also be attributed to a normal life breakdown of the ground equipment.

TABLE 24–III.—*Interim Discrepancy Record Summary by Spacecraft to First Countdown*

Spacecraft	Total tests	Average IDR[a] per test with first test deleted	Percent AGE[b] and unclassified IDR[a]
2	10	10.4	77
3	10	6.3	82
4	9	7.6	70
5	8	11.7	71
6	8	8.4	71
7	6	9.0	60

[a] Interim discrepancy record.
[b] Aerospace ground equipment.

Future spacecraft operations groups can benefit from spacecraft 5 experience. A sharp increase in the occurrence of interim discrepancy records indicated the need to start an investigation.

An analysis of test interim discrepancy records revealed that—

(1) Ground equipment and unclassified interim discrepancy records comprise approximately 70 percent of the total.

(2) The incidence of the interim discrepancy records and the amount of serial troubleshooting time are not directly related. This indicates that most of the interim-discrepancy-record tasks do not restrict further testing and are resolved in parallel with other activities.

(3) An analysis of the interim discrepancy records with respect to their occurrence in a test sequence (fig. 24–2) shows that 0.6 to 1.8 of these records per hour of testing can be expected for the first test of a series and 0.5 per hour of testing thereafter.

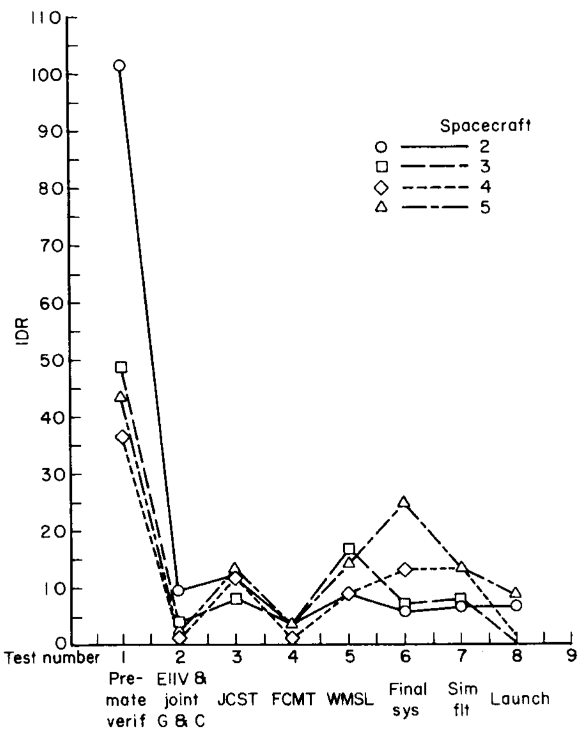

FIGURE 24–2.—Occurrence of interim discrepancy records for individual tests.

Effects of Spacecraft Modifications

Table 24–IV shows the modification times and the number of mission preparation sheets required on spacecraft 2 through 7 at the Kennedy Space Center. The mission preparation sheet is an engineering work order required for all manufacturing and testing accomplished on the spacecraft at the Kennedy Space Center. Thus far, modifications have been accomplished in parallel with scheduled testing and manu-

facturing and have not added serial time to the schedule. The number of the mission preparation sheets required to effect modifications on spacecraft 4 through 7 was 14 percent of the total required and 19 percent of the total required at the launch site. This shows that modifications are only a minor portion of the overall manufacturing and testing effort.

TABLE 24–IV.—*Modification and Mission-Preparation-Sheet Summary to First Countdown*

Spacecraft	Modification shifts	Modification MPS[a]	MPS[a] worked on pad	Total MPS[a] worked at launch site
2	98			
3	99		183	249
4	129	34	207	275
5	85	40	242	290
6	81	33	180	280
7	89	46	190	229

[a] Mission preparation sheet.

Effects of Spacecraft Parts Replacement

Of approximately 216 items replaced on spacecraft 2 through 7, 74 were classified as major items. The major items replaced (table 24–V) as a result of launch-site testing represent only 9.8 percent of the total number replaced at the Kennedy Space Center. The remaining 90.2 percent are a result of testing at the prime contractor's plant, component qualification testing, or experience gained from preflight testing or inflight performance of previous spacecraft.

TABLE 24–V.—*Item-Replacement History*

Spacecraft	Total items replaced	Major items replaced	Items replaced as a result of major tests
2	42	9	7
3	20	6	2
4	22	7	3
5	44	18	4
6	42	18	2
7	46	16	4
Total	216	74	22

Statistical Analysis of Overall Test Data

The data on testing, shown in table 24–II, were analyzed to determine functional relationships that could be used to plan and project spacecraft processing schedules. At corresponding points in a testing sequence, a high correlation (0.94) exists between the accumulative number of interim discrepancy records and the accumulative hours of testing and troubleshooting (fig. 24–3). From this relationship, the testing and troubleshooting time for a test sequence can be projected if the accumulative number of interim discrepancy records can be estimated.

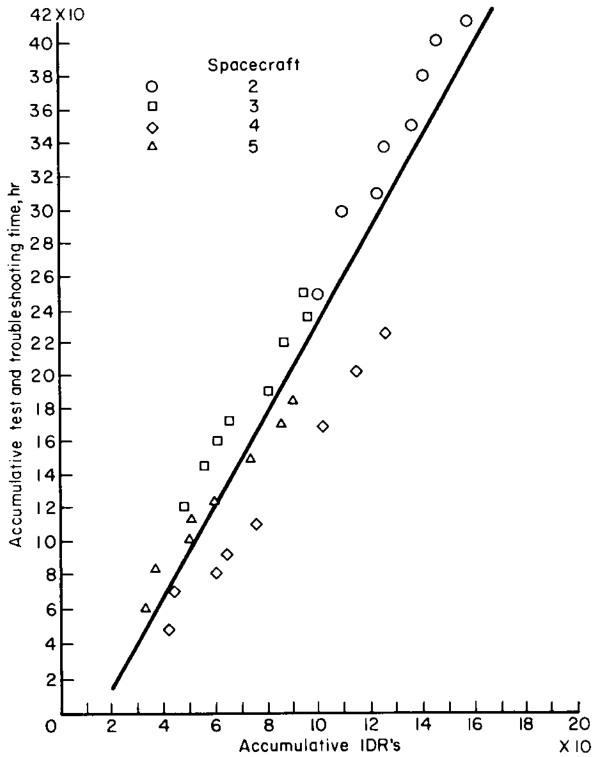

FIGURE 24–3.—Test and troubleshooting accumulative time compared with accumulative interim discrepancy records.

A method of estimating total interim discrepancy records reveals that a relationship (correlation: 0.88) exists between the test sequence and the accumulative number of these records. For example, the trend line shown in figure 24–4 is translated so that it passes through the estimated number of 27 interim discrepancy records for the first test on spacecraft 6. From the trend line, the projected value for 8 tests was

82 interim discrepancy records. From this forecast and from figure 24–3, a projection of 190 hours of testing and troubleshooting time was made for spacecraft 6. The actual result was 200 hours of testing and troubleshooting, with 86 interim discrepancy records recorded.

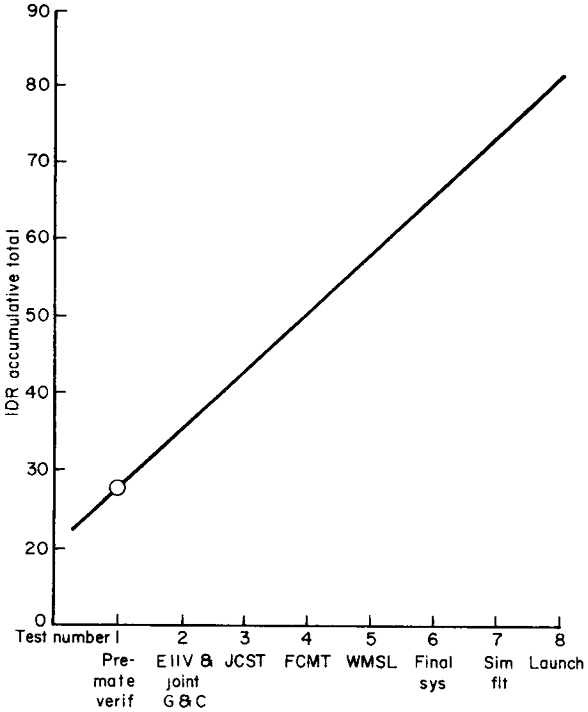

FIGURE 24–4.—Projection of accumulative quantity of interim discrepancy records.

Mathematical Model for Prediction of Processing Times

Assessment of Work Load

An examination of the mission-preparation-sheet logs and the daily schedules for spacecraft 3 through 7 led to the conclusion that nontesting tasks are virtually unaffected by testing. That is, during any given testing period, many nontesting tasks can be performed. Although the number of the mission preparation sheets has increased, no corresponding increase has been noted in the number of working shifts on the launch pad, indicating that there has been a steady improvement in the number of tasks that can be worked concurrently. Figures 24–5 and 24–6 present a synthesis of these observations.

Prediction Model

The spacecraft processing time required at launch complex 19 can be reduced to a mathematical model. The model consists of the following elements:

(1) The number of tasks performed during each work shift. These tasks can be categorized as—

(a) Major tests.

(b) Discrepancy records and squawks (minor discrepancies not involving a configuration change).

(c) Servicing.

(d) Troubleshooting.

(e) Parts replacement and retesting.

(f) Modification and assembly.

(2) The total number of mission preparation sheets.

(3) The actual number of shifts worked.

Tables 24–VI through 24–X and figures 24–5 and 24–6 summarize launch-pad histories of spacecraft 3 through 7. The difference in testing and troubleshooting times between these tables and table 24–II exists because table 24–II is based on serial troubleshooting time.

For the purpose of this study, the term "work unit" is defined as one task per work shift. Thus, in a given shift, as many as five mission preparation sheets could be processed using five work units. Discrepancy records and squawks have not been given the same consideration as the mission preparation sheets. Normally, one work unit has been found to equal six discrepancy records and squawks in any combination. Figure 24–7 shows a history of work units and work shifts required for spacecraft 3 through 7.

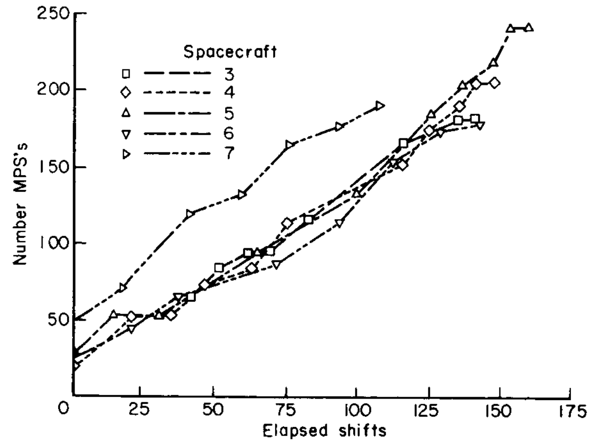

FIGURE 24–5.—Accumulative quantity of mission preparation sheets compared with elapsed shifts.

TABLE 24-VI.—*Work Summary for Spacecraft 3*

Task	Dates, 1965	Shifts		Test	Mission preparation sheets	Discrepancy records and squawks	Troubleshooting	Mission preparation sheets release
		Available	Used					
Premate verification test	2/05–2/17	37	34	24	103	24	12.5	29
Electrical interface and integrated validation; joint guidance and control	2/17–2/19	8	8	6	30	1	1.5	63
	2/20–2/21	6	6	0	17	1.5	0	--------
Joint combined systems test	2/22–2/25	10	10	3	36.5	8	2.5	83
Propellant servicing	2/25–2/27	8	6	5.5	24	1.5	0	93
Flight configuration mode test	2/28–3/08	12	9	3	40	7.5	1	99
Wet-mock-simulated launch	3/04–3/08	14	14	11	47.5	3.5	1	116
System test	3/08–3/15	21	21	6.1	107.5	15.5	1.5	134
Simulated flight	3/15–3/18	10	10	3	49	4	2	169
Launch	3/19–3/23	13.5	13.5	12.5	31.5	4.5	0	176
Total	------------	139.5	131.5	74.1	486.0	71.0	22.0	183

TABLE 24-VII.—*Work Summary for Spacecraft 4*

Task	Dates, 1965	Shifts		Test	Mission preparation sheets	Discrepancy records and squawks	Troubleshooting	Mission preparation sheets release
		Available	Used					
Premate verification test	4/15–4/23	25	19	20	78.5	4	7.5	20
Electrical interface and integrated validation; joint guidance and control	4/24–4/27	12	6	8.5	29	1.5	2.5	52
Joint combined systems test	4/27–4/30	11	11	8.5	46	1.5	2.5	55
Propellant servicing	5/01–5/06	16	10	8	30	3	1	72
Flight configuration mode test	5/06–5/07	4	4	2	11.5	0	0	87
	5/07–5/10	7	7	0	20.5	2	0	--------
Wet-mock-simulated launch	5/10–5/13	11	11	11	24.5	2	1.5	114
	5/14–5/23	29	26	0	132	9.5	1	--------
System test	5/23–5/26	9	9	6.6	46	4.5	0	158
Simulated flight	5/26–5/30	10.5	10.5	5.5	45.5	2	0	173
Launch	5/30–6/03	12.5	12.5	12.5	39.5	2	0	192
Total	------------	147.0	126.0	82.6	503.0	32	16.0	207

TABLE 24–VIII.—*Work Summary for Spacecraft 5*

Task	Dates, 1965	Shifts		Test	Mission preparation sheets	Discrepancy records and squawks	Trouble-shooting	Mission preparation sheets release
		Available	Used					
Premate verification	6/28–7/02	15	15	12. 5	95. 5	3. 0	3. 0	28
Electrical interface and integrated validation; joint guidance and control	7/03–7/08	17	11	4. 5	32	1. 5	2. 0	51
Joint combined systems test	7/08–7/12	12	9	3	33. 5	2	2. 0	56
Flight configuration mode test	7/08–7/12	12	12	3	56. 5	3. 5	0	65
	7/12–7/16	9	6	0	19	0	0	--------
Wet-mock-simulated launch	7/20–7/22	12	12	12	20	2	2. 5	91
	7/23–7/29	21	18	0	114. 5	11	0	--------
Propellant servicing	7/30–8/01	9	9	6. 5	40	2	0	136
	8/02–8/07	18	18	0	135. 5	11	0	--------
System test	8/08–8/12	12. 5	12	11. 1	114. 5	7. 5	5	188
Simulated flight	8/12–8/14	8. 5	8. 5	8. 5	29	2	2	207
Launch	8/14–8/19	14	14	13. 5	74. 5	7. 5	0	220
Total	------------	160. 0	145. 5	74. 6	764. 5	53. 0	16. 5	242

TABLE 24–IX.—*Work Summary for Spacecraft 6 to First Countdown*

Task	Dates, 1965	Shifts		Tests	Mission preparation sheets	Discrepancy records and squawks	Trouble-shooting	Mission preparation sheets release
		Available	Used					
Premate verification	9/09–9/15	21	18	11. 5	90. 5	6. 5	1	45
Electrical interface and integrated validation	9/16–9/16	3	3	0	15	5	0	--------
Joint guidance and control	9/17–9/21	14	11	7. 5	32	4	0	65
Joint combined systems test	9/21–9/23	10	10	4. 5	22. 5	5	0	--------
Manufacturing	9/24–9/30	21	12	0	46	3. 5	0	--------
Flight configuration mode test	10/01	3	3	7. 5	9. 5	2. 5	--------	89
Wet-mock-simulated launch	10/02–10/07	18	15	15. 5	35. 5	7	--------	--------
Demate	10/08	3	3	0	11	3	--------	115
Final systems, electrical interface and integrated validation; joint guidance and control	10/09–10/15	20	17	15. 5	76	15	1	157
Simulated flight and special impact prediction test	10/15–10/20	16	13	12	39	6	2	175
Launch	10/21–10/25	14	14	11	29	4	0	180
Total	------------	143	122	85	406	61. 5	5	--------

TABLE 24-X.—*Work Summary for Spacecraft 7*

Task	Dates, 1965	Shifts		Test	Mission preparation sheets	Discrepancy records and squawks	Troubleshooting	Mission preparation sheets release
		Available	Used					
Premate verification	9/30–10/04	18	18	14.5	61.5	8	0.1	71
Electrical interface and integrated validation	10/05–10/12	24	24	8.4	181.5	16	.4	120
Joint combined systems test	10/13–10/15	9	9	7.4	42	5	.1	122
Manufacturing	10/16–10/18	9	9	0	50	6	0	143
Final systems	10/19–10/23	15	15	5.9	62	11	0	165
Simulated flight	10/24–10/29	18	15	5.5	48.5	6	.5	178
Launch	10/30–11/04	14	14	12.7	48	5	0	190
Total		107	104	54.4	493.5	57	1.0	190

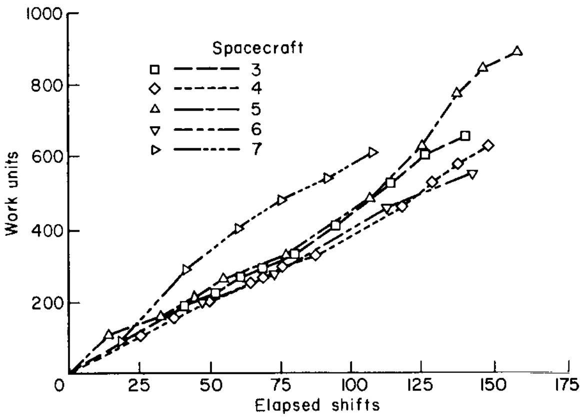

FIGURE 24-6.—Accumulative quantity of work units compared with elapsed shifts.

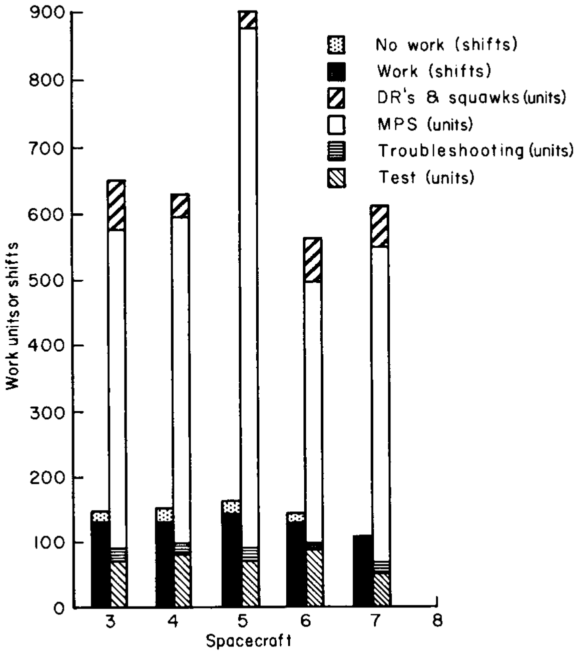

FIGURE 24-7.—Total work units and shifts required for each spacecraft.

The number of workdays necessary to process a Gemini spacecraft at the launch complex can be established using the following formula:

$$PD=\frac{\alpha(\text{number of mission preparation sheets})+\beta(\text{testing shifts})}{3\gamma}$$

where

PD=Total work required at the launch complex

$\alpha=\frac{\text{Nontest work units}}{\text{Nontest mission preparation sheets}}$ (Manufacturing mission-preparation-sheet performance factor)

$\beta=\frac{\text{Testing shifts}+\text{troubleshooting shifts}}{\text{Testing shifts}}$ (Testing factor)

$\gamma=\frac{\text{Total work units}}{\text{Total shifts worked}}$ (Overall work rate factor)

Figure 24–8 is a plot of α, β, and γ for spacecraft 3 through 7. These curves are the important factors used in predicting future spacecraft performance and processing time, as well as determining the present performance of a spacecraft being processed.

If no radical changes occur in spacecraft processing at the launch complex, the chart would infer that the following can be expected on the average:

(a) For every testing work shift, 0.2 of a troubleshooting shift can be expected.

(b) A nontest mission-preparation-sheet task will require three work shifts to accomplish.

(c) Approximately 5.75 tasks can be in progress concurrently.

These are, of course, estimates based on average figures. An examination of the data shows that as many as 10 tasks per shift have been worked concurrently on occasion; also, certain mission preparation sheets can be completed in less than one work shift. However, the use of total available data, rather than isolated cases, yields a better understanding of the factors and the relationships that affect overall processing time.

For example, the Spacecraft Operation Analysis Branch at Kennedy Space Center made the following predictions for spacecraft 7 using the process estimators:

(1) Based on an 8-test schedule, the predicted number of mission preparation sheets was less than 200, and the estimated number of work units was 672.

(2) Based on a 6-test schedule, the predicted number of mission preparation sheets was 190, and the number of work units was estimated at 580.

(3) For the 6-test schedule, 190 mission

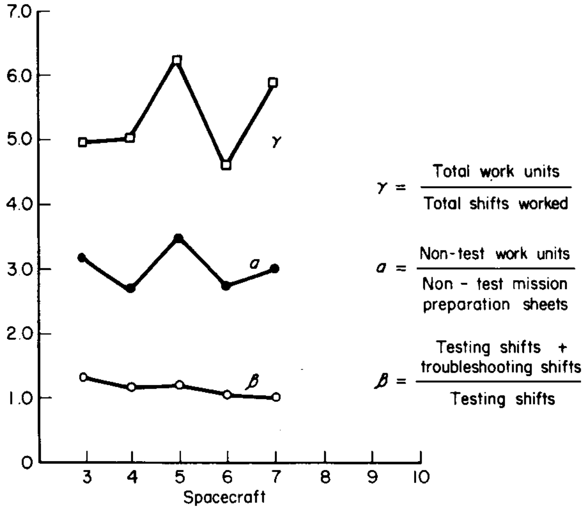

FIGURE 24–8.—Spacecraft processing estimators.

preparation sheets were recorded, and 607 work units were used.

The predicted versus the actual workload data was within a nominal 5 percent.

Analysis of Mission Preparation Sheets

The number of mission preparation sheets and the resulting workload account determine the spacecraft processing time. Table 24–XI shows the incidence of preparation sheets for spacecraft 3 through 5 at the launch pad. The daily completion rate of the preparation sheets is shown in table 24–XII.

The differences in completion rates by location and spacecraft were expected. Spacecraft 3 underwent hypergolic servicing and static firing before it went to the launch complex, with a resulting low daily completion rate of the preparation sheets. Spacecraft 4 through 7, however, were available prior to installation on the launch complex. All five spacecraft

TABLE 24-XI.—*Mission Preparation Sheets for Spacecraft 3, 4, and 5*

Spacecraft	Testing	Servicing	Replacement	Manufacturing	Open [a]	Unclassified [b]
3	26	41	14	83	15	4
4	41	31	29	97	0	9
5	44	44	51	89	7	12

[a] Mission preparation sheets released but not completed at the end of the spacecraft hoisting operation at the launch pad.

[b] Mission preparation sheets not identified as testing, servicing, replacement, or manufacturing.

TABLE 24-XII.—*Mission-Preparation-Sheet Daily Completion Rate*

Spacecraft	Prepad MPS [a] [b]	Pad MPS [a] [c]	Overall MPS [a]
3	2	3. 9	3. 2
4	6. 8	4. 6	4. 5
5	5. 4	4. 3	4. 5
6	2. 8	3. 8	4. 5
7	1. 8	5. 3	4. 0

[a] Mission preparation sheet.

[b] Testing before the spacecraft is installed on the launch vehicle at launch complex 19.

[c] Testing after the spacecraft is installed on the launch vehicle.

were subject to the same contraints of testing at the launch complex, and the difference in the rate of preparation sheet completion is attributed to a reduced workload and improved planning.

The total number of elapsed days has been used in the computation of the daily completion rates (table 24-XII) of the preparation sheets. If a comparison is to be made between these figures and those from the estimators used in the prediction model, an adjustment must be made for days not worked. This adjustment results in an increase from 4.6 to 5.0 days for spacecraft 4, and an increase from 4.3 to 5.0 days for spacecraft 5. Using the estimators from figure 24-8, the daily completion rates for mission preparation sheets are computed to be 5.5 to 5.3 for these spacecraft.

Concluding Remarks

The processing of Gemini spacecraft, from their arrival at the Kennedy Space Center through launch, is summarized as follows:

(1) Preparing for testing, testing, and troubleshooting constitute a maximum of 15 percent of the total processing work units. This constitutes an average of 57 percent of the scheduled work shifts.

(2) The number of interim discrepancy records, or problems resulting from testing, increases in direct proportion to the testing.

(3) All spacecraft met their schedules except spacecraft 2, when new test facilities were used for the first time.

(4) The time used for test preparation, as well as for total processing, tends to be the time allotted for these activities.

(5) To date, the time required for spacecraft modification and parts replacement has not directly affected any launch date because these activities have been accomplished in parallel with other scheduled work.

(6) The mathematical model provides an estimate for the processing time for future spacecraft.

(7) Monitoring of the process estimators provides an evaluation of the present processing of the spacecraft.

(8) A definite pattern in the occurrence of aerospace-ground-equipment interim discrepancy records has been established. Any significant increase from the normal pattern should be used as an indicator to start an investigation.

(9) The number of mission preparation sheets released against a spacecraft affects the total processing time. On the average, 1 day of processing time is required to complete five preparation sheets.

(10) To realize an accelerated processing schedule, consideration of the number of nontest work tasks is as important as consideration of the number of tests to be performed.

D

MISSION RESULTS

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25. MAN'S RESPONSE TO LONG-DURATION FLIGHT IN THE GEMINI SPACECRAFT

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Summary

The biomedical data from the Gemini III through VII missions support the conclusion that man is able to function physiologically and psychologically in space and readapt to the earth's 1-g environment without any undue symptomatology. It also appears that man's response can be projected into the future to allow 30-day exposures in larger spacecraft.

Introduction

When contemplating such titles as "4 Days in June," "8 Days in August," and "14 Days in December," it is difficult to realize that just 2 years ago, only an uncertain answer could be given to the question, "Can man's physiology sustain his performance of useful work in space?" This is particularly true on this great day for space medicine when man has equaled the machine.

Prior to our first manned space flight, many people expressed legitimate concern about man's possible response to the space-flight environment. This concern was based upon information obtained from aircraft experience and from conjecture about the effects of man's exposure to the particular environmental variables known to exist at that time. Some of the predicted effects were anorexia, nausea, disorientation, sleeplessness, fatigue, restlessness, euphoria, hallucinations, decreased g-tolerance, gastrointestinal disturbance, urinary retention, diuresis, muscular incoordination, muscle atrophy, and demineralization of bones. It will be noted that many of these are contradictory.

This Nation's first probing of the space environment was made in the Mercury spacecraft which reached mission durations of 34 hours. The actual situation following the completion of the Mercury program may be summarized as follows:

No problem: Launch and reentry acceleration, spacecraft control, psychomotor performance, eating and drinking, orientation, and urination.

Remaining problems: Defecation, sleep, and orthostatic hypotension.

This first encounter with the weightless environment had provided encouragement about man's future in space, but the finding of orthostatic hypotension also warned that there might be some limit to man's exposure. The reported Russian experiences strengthened this possibility. No serious gross effects of simple exposure to the space-flight environment had been noted, but the first hint was given that the emphasis should shift to careful methods for observing more subtle changes. These findings influenced the planning for the Gemini mission durations, and the original plan was modified to include a three-revolution checkout flight, followed by an orderly approximate doubling of man's exposure on the 4-day, 8-day, and 14-day missions which have been completed. It was felt that such doubling was biologically sound and safe, and this has proved to be the case. The U.S. manned space-flight missions are summarized in table 25–I.

This plan required the use of data procured from one mission for predicting the safety of man's exposure on a mission twice as long.

Medical Operational Support

The Gemini mission operations are complex and require teamwork in the medical area, as in all others. Space-flight medical operations have consisted, in part, of the early collection of baseline medical data which was started at

TABLE 25-I.—*U.S. Manned Space Flights*

Astronauts	Launch dates	Duration, hr: min
Shepard	May 5, 1961	00:15
Grissom	July 21, 1961	00:15
Glenn	Feb. 20, 1962	4:56
Carpenter	May 24, 1962	4:56
Schirra	Oct. 3, 1962	9:14
Cooper	May 15, 1963	34:20
Grissom Young	Mar. 3, 1965	4:52
McDivitt White	June 3, 1965	96:56
Cooper Conrad	Aug. 21, 1965	190:56
Borman Lovell	Dec. 4, 1965	330:35
Schirra Stafford	Dec. 15, 1965	25:21

the time of the original selection of the astronauts and which has been added to with each exposure to the simulated space-flight environment during spacecraft testing. Physicians and paramedical personnel have been trained to become a part of medical recovery teams stationed in the launch area and at probable recovery points in the Atlantic and Pacific Oceans. Flight surgeons have been trained and utilized as medical monitors at the various network stations around the world, thus making possible frequent analysis of the medical information obtained in flight. A team of Department of Defense physician-specialists has also been utilized to assist in the detailed preflight and postflight evaluations of the condition of the flight crews. Without the dedicated help of all of these personnel functioning as a team, the conduct of these missions would not have been possible (fig. 25-1).

A high set of standards has been adhered to in selecting flight crews. This has paid off very well in the safety record obtained thus far. The difficult role that these flight crews must play,

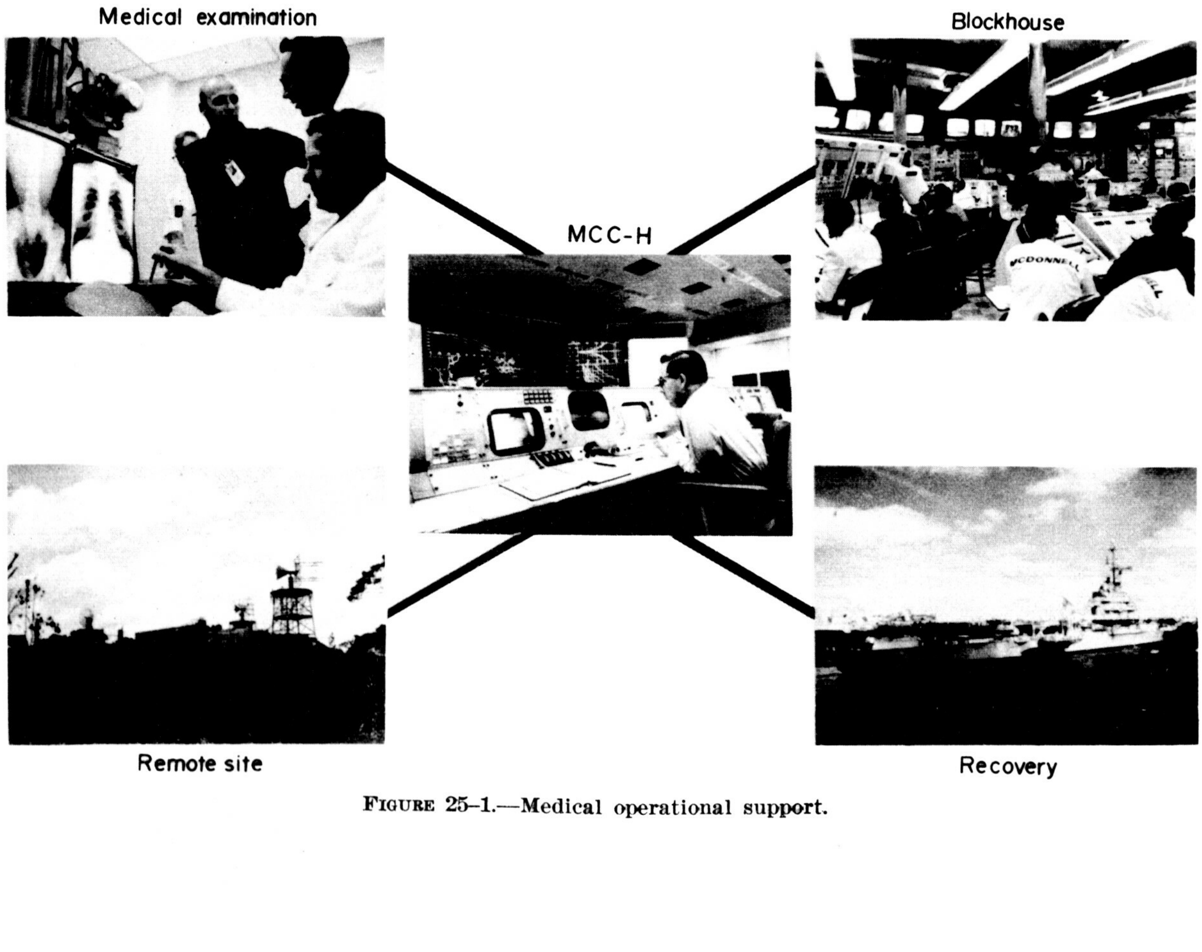


FIGURE 25-1.—Medical operational support.

both as experimenters and as subjects, deserves comment. From a personal point of view, the simpler task is to be the experimenter, utilizing various pieces of equipment in making observations. On these long-duration missions, the crews have also served as subjects for medical observations, and this requires maximum cooperation which was evidenced on these flights.

Data Sources

Physiological information on the flight crews has been obtained by monitoring voice transmissions; two leads of the electrocardiogram, a sternal and an axillary; respiration by means of an impedance pneumograph; body temperature by means of an oral thermistor; and blood pressure. These items make up the operational instrumentation, and, in addition, other items of bioinstrumentation are utilized in the experiments program. Also, some inflight film footage has been utilized, particularly during the extravehicular exercise on the 4-day mission. The biosensor harness and signal conditioners are shown in figure 25–2. A sample of the telemetered data, as received at the Mission Control Center, is shown in figure 25–3. These data were taken near the end of the 8-day flight, and it can be seen that the quality is still excellent. The Gemini network is set up to provide real-time remoting of medical data from the land sites to the surgeon at the Mission Control Center. If requested, the medical data from the ships can be transmitted immediately after each spacecraft pass. The combined Gemini VI–A and VII mission posed a new problem in monitoring, in that it required the simultaneous monitoring of four men in orbit. The network was configured to do this task, and adequate data were received for evaluation of both crews.

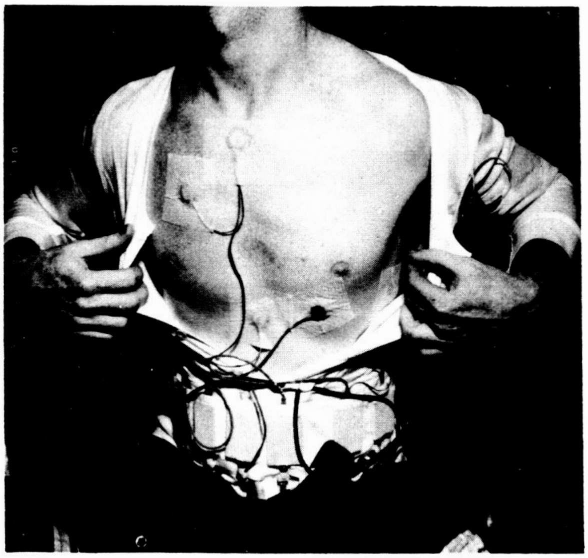


FIGURE 25–2.—Biosensor harness and signal conditioners.

It must be realized that this program has involved only small numbers of people in the flight crews. Thus, conclusions must be drawn from a minimum amount of data. Individual variability must be considered in the analysis of any data. Aid is provided in the Gemini Program by having two men exposed to the same conditions at the same time. Each man also serves as his own control, thus indicating the importance of the baseline data.

Preflight Disease Potential

As missions have become longer, the possibility of an illness during flight has become greater, particularly in the case of communicable diseases to which the crew may have been exposed prior to launch. The difficult work schedules and the stress imposed by the demands of the prelaunch period tend to create fatigue unless watched carefully, and thus become an additional potential for the development of flu-like diseases. They also preclude any strict isolation. On each of the Gemini missions a potential problem, such as viral upper respiratory infections or mumps exposure, has developed during the immediate preflight period, but the situation has been handled without hampering the actual mission. No illness has developed in the flight crews while in orbit. However, strenuous effort must be exerted toward protecting the crew from potential disease hazards during this critical period.

Denitrogenation

The 5-psia cabin pressure and the 3.7-psia inflated suit pressure create the potential for the development of dysbarism, and this was particularly true on the 4-day mission which involved extravehicular activity. Care has been taken to denitrogenate the crews with open-loop breathing of 100 percent oxygen for at least 2 hours prior to launch. No difficulty has been experienced with this procedure.

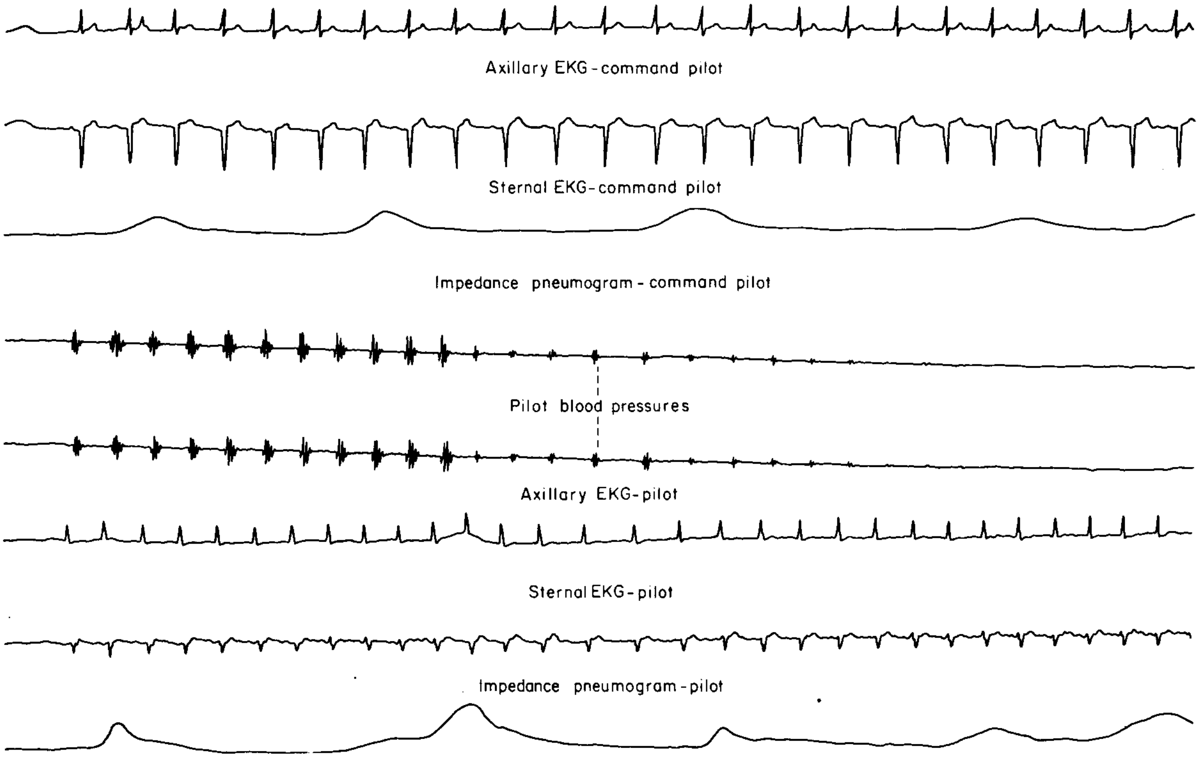


FIGURE 25–3.—Sample of biomedical data.

Preflight Exercise

The crews have used various forms of exercise to maintain a state of physical fitness in the preflight period. The peak of fitness attained has varied among the crewmembers, but they all have been in an excellent state of physical fitness. They have utilized running and various forms of activity in the crew-quarters gymnasium in order to maintain this state. Approximately 1 hour per day has been devoted to such activity.

Space-Flight Stresses

There has been a multiplicity of factors acting upon man in the space-flight environment. He is exposed to multiple stresses which may be summarized as: full pressure suit, confinement and restraint, 100 percent oxygen and 5-psia atmosphere, changing cabin pressure (launch and reentry), varying cabin and suit temperature, acceleration g-force, weightlessness, vibration, dehydration, flight-plan performance, sleep need, alertness need, changing illumination, and diminished food intake. Any one of these stresses will always be difficult to isolate. In a sense, it could be said that this is of only limited interest, for the results always would represent the effects of man's exposure to the total space-flight environment. However, in attempting to examine the effects of a particular space-flight stress, such as weightlessness, it must be realized that the responses observed may indeed be complicated by other factors such as physical confinement, acceleration, dehydration, or the thermal environment.

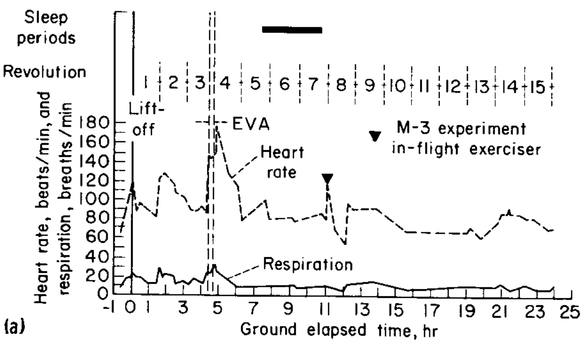
Heart Rate

On all missions, the peak elevations of heart rates have occurred at launch and reentry. The peak rates observed during the launch and reentry are shown in table 25–II. These detailed timeline plots of heart and respiratory rates demonstrate the peak responses associated with particular activities required by the flight plan, as was noted during the Mercury missions (fig. 25–4 (a) and (b)). As the mission durations have become longer, it has been necessary to compress the heart-rate data from the Gemini VII mission to the form shown in figure 25–5 (a) and (b). Such a plot demonstrates the di-

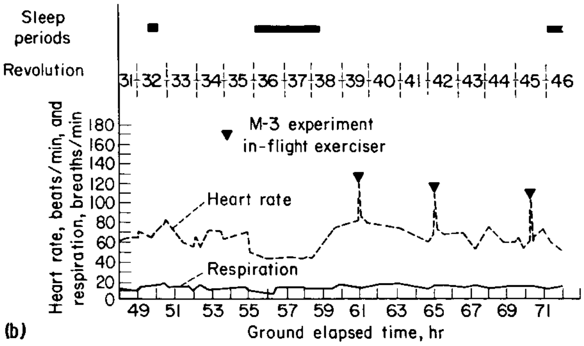
TABLE 25–II.—*Peak Heart Rates During Launch and Reentry*

Gemini mission	Peak rates during launch, beats per minute	Peak rates during reentry, beats per minute
III	152	165
	120	130
IV	148	140
	128	125
V	148	170
	155	178
VI–A	125	125
	150	140
VIII	152	180
	125	134

urnal cycles related to the nighttime and the normal sleep periods at Cape Kennedy, Fla. In general, it has been noted that there has been a decrease in the heart rate from the high levels at launch toward a rather stable, lower baseline rate during the midportion of the mission. This is altered at intervals since the heart has responded to demands of the inflight activities in a very normal manner throughout the mission. The rate appears to stabilize around the 36- to 48-hour period and remain at this lower level until two or three revolutions before retrofire. The anticipation and the activity associated with preparation for retrofire and reentry cause an increase in the heart rate for the remainder of the flight. The electrocardiogram has been very helpful in observing the response to the sleep periods when heart rates have frequently been observed in the forties and some in the high thirties. The graphing of such rates by mini-



(*a*) From lift-off to 24 hours ground elapsed time.
FIGURE 25–4.—Physiological measurements for Gemini IV pilot.



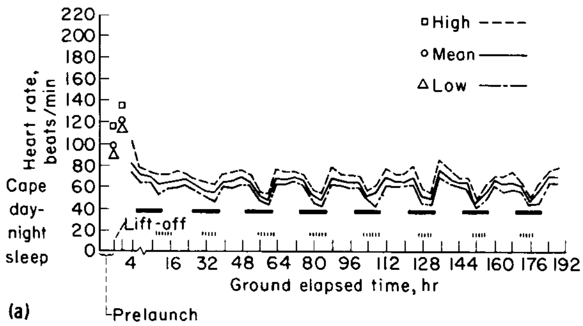
(*b*) From 48 to 72 hours ground elapsed time.
FIGURE 25–4.—Concluded.

mum, maximum, and mean has also been helpful in determining the quality of sleep. If the crewmen have awakened several times to check the condition of spacecraft controls and displays, there is a noted spread between the maximum and minimum rates.

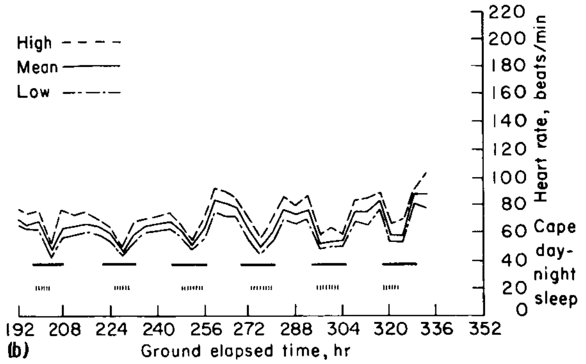
During the extravehicular operation, both crewmen noted increased heart rates. The pilot had a heart rate of 140 beats per minute while standing in the open hatch, and this rate continued to climb during the extravehicular activity until it reached 178 beats per minute at spacecraft ingress. Future extravehicular operations will require careful attention to determine the length of time these elevated rates are sustained.

Electrocardiogram

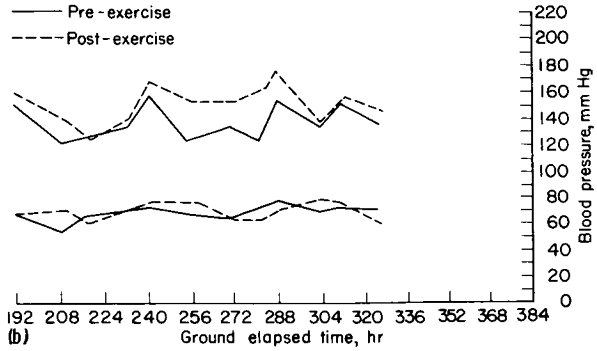
The electrocardiogram has been observed on a real-time basis, with a series of detailed measurements being taken during the Gemini VII flight. The electrocardiogram has also been evaluated postflight, and the only abnormalities of note have been occasional, and very rare, pre-



(*a*) From lift-off to 192 hours ground elapsed time.
FIGURE 25–5.—Physiological measurements for Gemini VII pilot.



(b) From 192 to 352 hours ground elapsed time.
FIGURE 25–5.—Concluded.

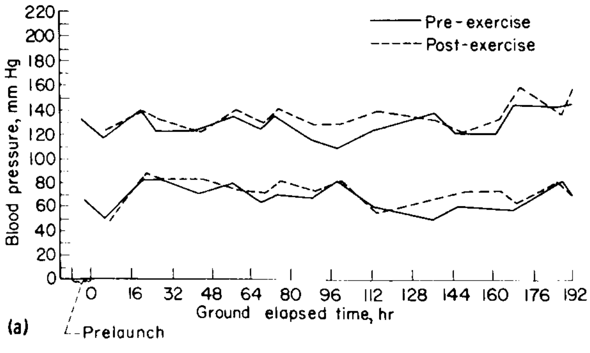


(b) From 192 to 384 hours ground elapsed time.
FIGURE 25–6.—Concluded.

mature auricular and ventricular contractions. The detailed analyses have shown no significant changes in the duration of specific segments of the electrocardiogram which are not merely rate related. On each of the long-duration missions, a special experiment has involved observation of the relationship of the Q-wave to the onset of mechanical systole, as indicated by the phonocardiogram. These data, in general, have revealed no prolongation of this interval with an increase in the duration of space flight.

Blood Pressures

The blood pressure values were determined three times in each 24 hours during the 4- and 8-day missions, and two times each 24 hours on the 14-day mission. These determinations were made before and after exercise on the medical data passes. The only truly remarkable thing in all blood pressures to date has been the normalcy with a lack of significant increase or decrease with prolonged space flight (fig. 25–6 (a) and (b)). The blood pressures have varied



(a) From lift-off to 192 hours ground elapsed time.
FIGURE 25–6.—Blood pressure measurement.

with heart rate, as evidenced by the 201 over 90 blood pressure obtained after retrofire during one of the missions. This was accompanied by a heart rate of 160, however, and is felt to be entirely normal.

Some blood pressures of particular interest were those determined on the 4-day mission: (1) just after retrofire and while the crew was still in zero g; (2) just before the transition to two-point suspension on the main parachute, which places the crew at about a 45° back angle; (3) just after the transition to two-point suspension; and (4) with the spacecraft on the water and the crew in a sitting position. All of these pressures were in the same general range as the inflight blood pressures and were all certainly normal, demonstrating no evidence of hypotension.

Body Temperature

The oral thermistor was used with each medical data pass, and all body temperatures recorded have been within the normal range. Occasional spurious readings were noted on the oral thermistor when it got misplaced against the body, causing it to register.

Respiratory Rates

Respiratory rates during all of the long-duration missions have tended to vary normally along with heart rate. Hyperventilation has not occurred in flight.

Inflight Exercise

An exercise consisting of 30 pulls on a bungee cord has been utilized to evaluate cardiovascular response on all of these missions. No significant difference in the response to this calibrated exercise load has been noted through the 14-day flight. In addition to these programed exercise response tests, the bungee cord has been utilized for additional exercise periods. Daily during the 14-day mission, the crew performed 10 minutes of exercise, including the use of the bungee cord for both the arms and the legs, and some isometric exercises. These 10-minute periods preceded each of the three eating periods.

Sleep

A great deal of difficulty was encountered in obtaining satisfactory sleep periods on the 4-day mission. Even though the flight plan was modified during the mission in order to allow extra time for sleep, it was apparent postflight that no long sleep period was obtained by either crewman. The longest consecutive sleep period appeared to be 4 hours, and the command pilot estimated that he did not get more than 7½ to 8 hours' good sleep in the entire 4 days. Factors contributing to this lack of sleep included: (1) the firing of the thrusters by the pilot who was awake; (2) the communications contacts, because the communications could not be completely turned off; and (3) the requirements of housekeeping and observing, which made it difficult to settle down to sleep. Also the responsibility felt by the crew tended to interfere with adequate sleep.

An attempt was made to remove a few of these variables on the 8-day mission and to program the sleep periods in conjunction with normal nighttime at Cape Kennedy. This required the command pilot to sleep from 6 p.m. until midnight eastern standard time, and the pilot to sleep from midnight until 6 a.m., each getting a 2-hour nap during the day. This program did not work out well due to flight-plan activities and the fact that the crew tended to retain their Cape Kennedy work-rest cycles with both crewmen falling asleep during the midnight to 6 a.m. Cape Kennedy nighttime period. The 8-day crew also commented that the spacecraft was so quiet that any communication or noise, such as removing items attached with Velcro, produced an arousal reaction.

On the 14-day flight, the flight plan was designed to allow the crew to sleep during hours which generally corresponded to nighttime at Cape Kennedy. There was a 10-hour period established for this sleep (fig. 25–7), and it worked out very well with their normal schedule. In addition, both crewmen slept at the same time, thus obviating any arousal reactions from the actions of the other crewmember. The beginning of the scheduled rest and sleep period was altered to move it one-half hour earlier each night during the mission in order to allow the crew to be up and active throughout the series of passes across the southern United States. Neither crewman slept as soundly in orbit as he did on the earth, and this inflight observation was confirmed in the postflight debriefing. The pilot seemed to fall asleep more easily and could sleep more restfully than the command pilot. The command pilot felt that it was unnatural to sleep in a seated position, and he continued to awaken spontaneously during his sleep period and would monitor the cabin displays. He did become increasingly fatigued over a period of several days, then would sleep soundly and start his cycle of light, intermittent sleep to the point of fatigue all over again. The cabin was kept quite comfortable during the sleep periods by the use of the Polaroid screen and some foil from the food packs on the windows. The noise of the pneumatic pressure cuff for Experiment M–1 did interfere with sleep on both the 8- and 14-day missions. The crew of the 4-day flight were markedly fatigued following the mission. The 8-day crew were less so, and the 14-day crew the least fatigued of all. The 14-day crew did feel there was some irritability and loss of patience during the last 2 days of the mission, but they continued to be alert and sharp in their responses, and no evidence of performance decrement was noted.

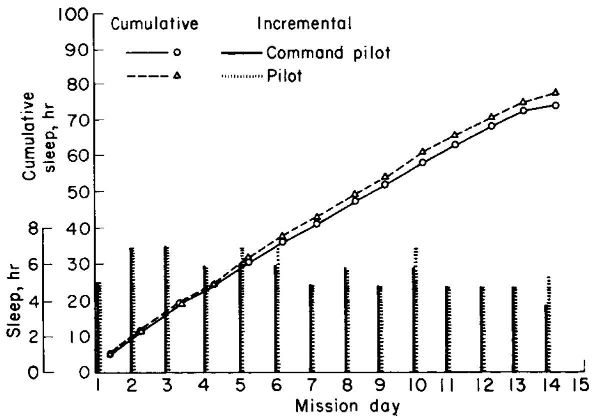


FIGURE 25–7.—Sleep data for Gemini VII flight crew.

Food

The diet has been controlled for a period of 5 to 7 days before flight and, in general, has been of a low residue. The Gemini VII crew were on a regulated calcium diet of a low-residue type for a period of 12 days before their 14-day mission. The inflight diet has consisted of freeze dehydrated and bite-size foods. A typical menu is shown in table 25–III. The crew are routinely tested with the inflight menu for a period of several days before final approval of the flight menu is given. On the 4-day flight, the crew were furnished a menu of 2500 calories per day to be eaten at a rate of four meals per day. They enjoyed the time that it took to prepare the food, and they ate all the food available for their use. They commented that they were hungry within 2 hours of ingesting a meal and that, within 4 hours after ingesting a meal, they felt a definite physiological need for the lift produced by food.

TABLE 25–III.—*Typical Gemini Menu*

[Days 2, 6, 10, and 14]

	Calories
Meal A:	
Grapefruit drink	83
Chicken and gravy	92
Beef sandwiches	268
Applesauce	165
Peanut cubes	297
	905
Meal B:	
Orange-grapefruit drink	83
Beef pot roast	119
Bacon and egg bites	206
Chocolate pudding	307
Strawberry cereal cubes	114
	829
Meal C:	
Potato soup	220
Shrimp cocktail	119
Date fruitcake	262
Orange drink	83
	684
Total calories	2418

These findings were in marked contrast to the 8-day mission where each crewmember was furnished three meals per day for a caloric value of 2750. Again these meals consisted of one juice, two rehydratable food items, and two bite-size items. The 8-day crew felt no real hunger, though they did feel a physiological lift from the ingestion of a meal. They ate very little of their bite-size food and subsisted principally on the rehydratable items. A postflight review of the returned food revealed that the average caloric intake per day varied around 1000 calories for this crew. Approximately 2450 calories per day was prepared for the 14-day mission and including ample meals for 14⅔ days. Inflight and postflight analyses have revealed that this crew actually consumed about 2200 calories per day.

Water Intake

There has been an ample supply of potable water on all of these missions, consisting of approximately 6 pounds per man per day. Prior to the 4-day and 8-day missions, the water intake was estimated by calibrating a standard mouthful or gulp for each crewman; then, during the flight, the crew would report the water intake by such measurements. On the 4-day mission, the water intake was less than desired in the first 2 days of the mission but increased during the latter part of the flight, varying from 2.5 to 5.0 pounds in a 24-hour period. The crew were dehydrated in the postrecovery period. On the 8-day mission, the crew did much better on their water intake, averaging 5.2 to 5.8 pounds per 24 hours, and they returned in an adequately hydrated state.

For the 14-day mission, the water dispensing system was modified to include a mechanism whereby each activation of the water dispenser produced ½ ounce of water, and this activated a counter. The number of counts and the number of ounces of water were laboriously logged by the crew. It has been obvious that the crewmen must be reminded of their water intake, and when this is done they manage very well. The 14-day crew were well hydrated at the time of their recovery, and their daily water intake is presented in figure 25–8.

Waste Disposal

A urine collection device has been utilized on each of the Gemini missions and has been modified according to need and experience. On the 14-day flight, for the first time, the system permitted the collection of urine samples. Prior to this time, all of the urine was flushed overboard. The system shown in figure 25–9 al-

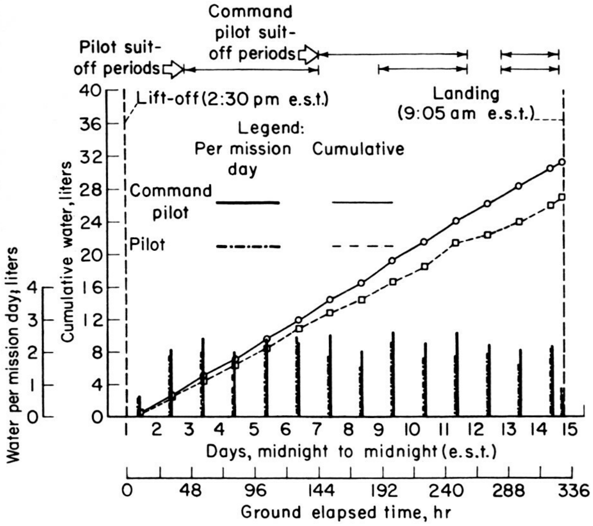


Figure 25–8.—Water intake per day for Gemini VII flight crew.

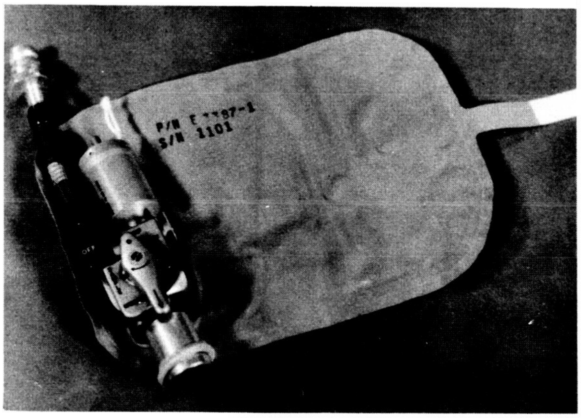


Figure 25–9.—Urine collection device.

lowed for collection of a 75-cc sample and the dumping of the remainder of the urine overboard. The total urine volume could be obtained by the use of a tritium-dilution technique.

The handling of fecal waste has been a bothersome inflight problem. Before the mission, the crews eat a low-residue diet, and, in addition, on the 8-day and 14-day missions, they have utilized oral and suppository Dulocolax for the last 2 days before flight. This has proved to be a very satisfactory method of preflight preparation. The fecal collection device is shown in figure 25–10.

The sticky surfaces of the bag opening can be positioned much easier if the crewman is out of the space suit, as occurred during the 14-day flight. The system creates only a minimum amount of difficulty during inflight use and is an adequate method for the present missions. On the 14-day flight, the system worked very well and allowed the collection of all of the fecal specimens for use with the calcium-balance experiment.

Bowel habits have varied on each of the three long-duration missions, as might be expected. Figure 25–11 lists the defecations recorded for these three missions, and the longest inflight delay before defecation occurred was 6 days on the 14-day mission. The opportunity to measure urine volume on the 14-day flight has been of particular interest, as it had been anticipated a diuresis would occur early in the flight. Figure 25–12 shows the number of urinations per day and the urine volume as determined from the flowmeter utilized on the 14-day mission. The accuracy of these data will be compared with that from the tritium samples.

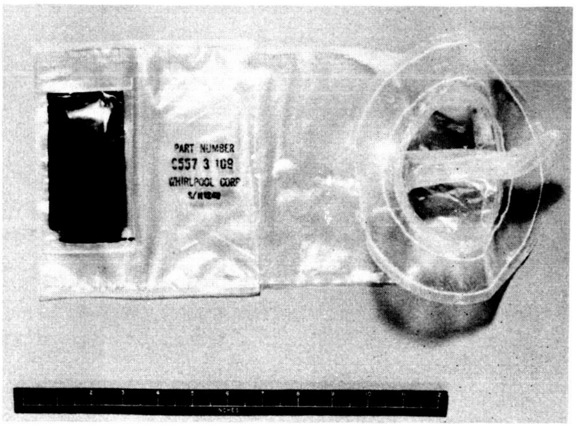


Figure 25–10.—Fecal bag.

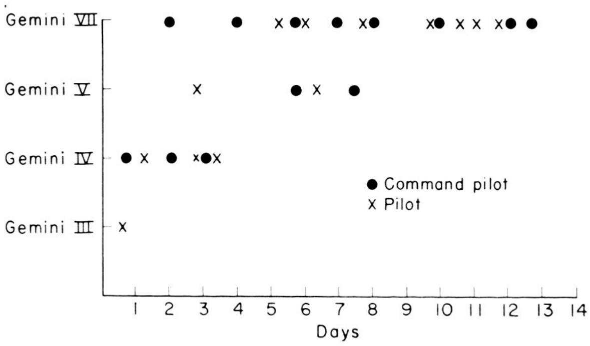


Figure 25–11.—Inflight defecation frequency.

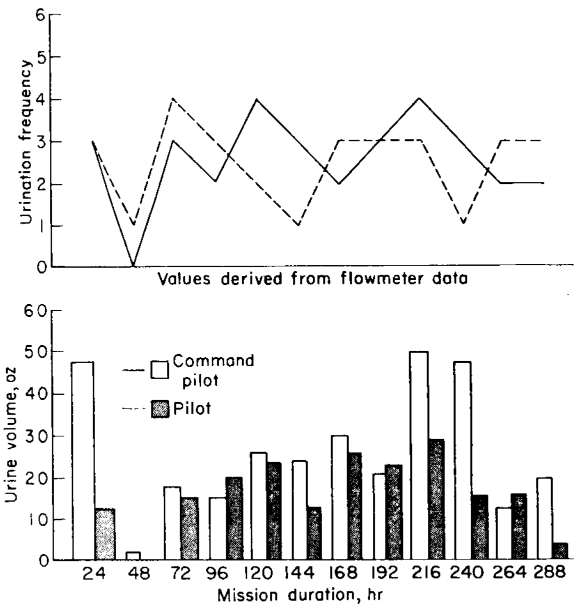


FIGURE 25-12.—Urine volume and urination frequency of Gemini VII flight crew.

Medications

Medications in both injectable and tablet forms have been routinely provided on all flights. The basic policy has continued to be that a normal man is preferred and that drugs are used only if necessary. A list of the supplied drugs is shown in table 25-IV, and the medical kit is shown in figure 25-13. The injectors may be used through the suit, although to date none have been utilized. The only medication used thus far has been dexedrine, taken prior to reentry by the Gemini IV crew. The dexedrine was taken to insure an adequate state of alertness during this critical mission period. In spite of the minimal use of medications, they must be available on long-duration missions, and each crewmember must be pretested to any drug which may potentially be used. Such pretesting of all of the medications listed in table 25-IV has been carried out with each of the crews.

TABLE 25-IV.—*Gemini VII Inflight Medical and Accessory Kits*

(a) Medical kit

Medication	Dose and form	Label	Quantity
Cyclizine HCl	50-mg tablets	Motion sickness	8
d-Amphetamine sulfate	5-mg tablets	Stimulant	8
APC (aspirin, phenacetin, and caffeine)	Tablets	APC	16
Meperidine HCl	100-mg tablets	Pain	4
Triprolidine HCl	2.5-mg tablets	Decongestant	16
Pseudoephedrine HCl	60-mg tablets		
Diphenoxylate HCl	2.5-mg tablets	Diarrhea	16
Atropine sulfate	0.25-mg tablets		
Tetracycline HCl	250-mg film-coated tablet	Antibiotic	16
Methylcellulose solution	15-cc in squeeze-dropper bottle	Eyedrops	1
Parenteral cyclizine	45-mg (0.9-cc in injector)	Motion sickness	2
Parenteral meperidine HCl	90-mg (0.9-cc in injector)	Pain	2

(b) Accessory kit

Item	Quantity
Skin cream (15-cc squeeze bottle)	2
Electrode paste (15-cc squeeze bottle)	1
Adhesive disks for sensors	12 for EKG, 3 for phonocardiogram leads
Adhesive tape	20 in.

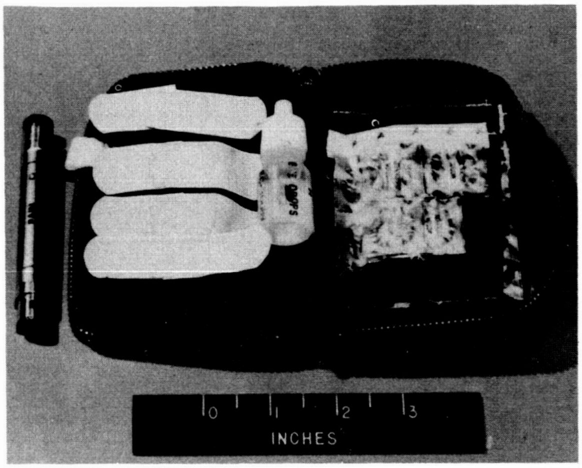


FIGURE 25–13.—Medical kit carried onboard the spacecraft.

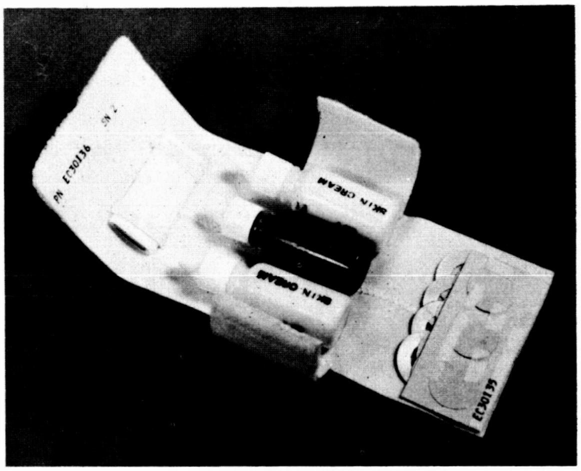


FIGURE 25–14.—Medical accessory kit carried onboard the spacecraft.

On the 14-day mission, a medical accessory kit, shown in figure 25–14, was carried to allow the reapplication of medical sensors should they be lost during the flight. The kit contained the sensor jelly, and the Stomaseal and Dermaseal tape for sensor application. In addition, the kit contained small plastic bottles filled with a skin lotion, which was a first-aid cream. During the 14-day mission, this cream was used by both crewmen to relieve the dryness of the nasal mucous membranes and was used occasionally on certain areas of the skin. During the mission, the lower sternal electrocardiogram sensor was replaced by both crewmen, and excellent data were obtained after replacement.

Psychology of Flight

Frequent questions are asked concerning the ability of the crewmembers to get along with one another for the long flight periods. Every effort is made to choose crewmembers who are compatible, but it is truly remarkable that none of the crews, including the long-duration crews, have had any inflight psychological difficulties that were evident to the ground monitors or that were discussed in postflight debriefings. They have had some normal concerns for the inherent risks of space flight. They were well prepared for the fact that 4, 8, and 14 days in space in such a confined environment as the Gemini spacecraft would not be an easy task. They had trained well, done everything humanly possible for themselves, and knew that everyone connected with the program had done everything possible to assure their stay. There is some normal increased tension at lift-off and also prior to retrorocket firing. There was some normal psychological letdown when the Gemini VII crew saw the Gemini VI–A spacecraft depart after their rendezvous. However, the Gemini VII crew accepted this very well and immediately adjusted to the flight-plan activity.

A word should be said about overall crew performance from a medical point of view. The crews have performed in an exemplary manner during all flights. There has been no noted decrease in performance, and the fine control tasks such as reentry and, notably, the 11th-day rendezvous during the Gemini VII mission have been handled with excellent skill.

Additional Inflight Observations of Medical Importance

The crews have always been busy with flight-plan activity and have felt that their days were complete and full. The 14-day crew carried some books, occasionally read them in the presleep period, and felt they were of value. Neither crewman completed a book. Music was provided over the high-frequency air-to-ground communications link to both the 8-day and the 14-day crews. They found this to be a welcome innovation in their flight-plan activity.

The crews have described a sensation of fullness in the head that occurred during the first 24 hours of the mission and then gradually disappeared. This feeling is similar to the increase of blood a person notes when hanging on parallel bars or when standing on his head. There was no pulsatile sensation in the head and no obvious reddening of the skin. The exact cause of this condition is unknown, but it may be related to an increase of blood in the chest area as a result of the readjustment of the circulation to the weightless state.

It should be emphasized that no crewmembers have had disorientation of any sort on any Gemini mission. The crews have adjusted very easily to the weightless environment and accepted readily the fact that objects will stay in position in midair or will float. There has been no difficulty in reaching various switches or other items in the spacecraft. They have moved their heads at will and have never noticed an aberrant sensation. They have always been oriented to the interior of the spacecraft and can orient themselves with relationship to the earth by rolling the spacecraft and finding the horizon through the window. During the extravehicular operation, the Gemini IV pilot oriented himself only by his relationship to the spacecraft during all of the maneuvers. He looked repeatedly at the sky and at the earth and had no sensations of disorientation or motion sickness at any time. The venting of hydrogen on the 8-day flight created some roll rates of the spacecraft that became of such magnitude that the crew preferred to cover the windows to stop the visual irritation of the rolling horizon. Covering the windows allowed them to wait for a longer period of time before having to damp the rates with thruster activity. At no time did they experience any disorientation. During the 14-day flight, the crew repeatedly moved their heads in various directions in order to try to create disorientation but to no avail. They also had tumble rates of 7° to 8° per second created by venting from the water boiler, and one time they performed a spin-dry maneuver to empty the water boiler, and this created roll rates of 10° per second. On both occasions they moved their heads freely and had no sensation of disorientation.

The crews of all three long-duration missions have noted an increased g-sensitivity at the time of retrofire and reentry. All the crews felt that they were experiencing several g when the g-meter was just beginning to register at reentry. However, when they reached the peak g-load, their sensations did not differ from their centrifuge experience.

Physical Examination

A series of physical examinations have been accomplished before each flight in order to determine the crewmembers' readiness for mission participation, and also after each flight to evaluate any possible changes in their physical condition. These examinations normally have been accomplished 8 to 10 days before launch, 2 days before launch, on launch morning, and immediately after the flight and have been concluded with daily observations for 5 to 10 days after recovery. These examinations thoroughly surveyed the various body systems. With the exception of items noted in this report, there have been no significant variations from the normal preflight baselines. The 14-day crew noted a heavy feeling in the arms and legs for several hours after recovery, and they related this to their return to a 1-g environment, at which time their limbs became sensitive to weight. In the zero-g condition, the crew had been aware of the ease in reaching switches and controls due to the lack of weight of the arms. The 8-day crew also reported some heaviness in the legs for several hours after landing. Both the 8-day and 14-day crews reported some muscle stiffness lasting for several days after recovery. This was particularly noted in the legs and was similar to the type of stiffness resulting from initial athletic activity after a long period of inactivity.

On all missions there has been minimum skin reaction surrounding sensor sites, and this local irritation has cleared rapidly. There have been a few small inclusion cysts near the sternal sensors. In preparing for the 8-day flight the crews bathed daily with hexachlorophene for approximately 10 days before the flight. In addition, the underwear was washed thoroughly in hexachlorophene, and attempts were made to keep it relatively free of bacteria until donning. The 14-day crew showered daily with a standard hexachlorophene-containing soap and also used Selsun shampoos for a 2-week period. Following the 8-day and 14-day missions, the crewmembers' skin was in excellent condition. The 8-day flight crewmembers did have some dryness and scaling on the extremities and over the sensor sites, but, after using a skin lotion for several days, the condition cleared rapidly. The 14-day crewmembers' skin did not have any dryness and required no treatment postflight. After their flight, the 8-day crew had some marked dandruff and seborrheic lesions of the scalp which required treatment with Selsun for a period of time. The 14-day crew had virtually no dandruff in the postflight examination, nor was it a problem during flight.

The crew of the 14-day mission wore new lightweight space suits and, in addition, removed them for a portion of the flight. While significant physiological differences between the suited and unsuited crewman were difficult to determine, it was noted that the unsuited crewman exercised more vigorously, slept better, and had higher urine output because fluid was not being lost as perspiration. The excellent general condition of the crewmembers, particularly their skin condition, is to a large extent attributable to the unsuited operations.

Bacterial cultures were taken from each crewmember's throat and from several skin areas before and after the long-duration missions. The numbers of bacteria in the throat flora were reduced, and there was an increase in the fecal flora in the perineal areas. All fungal studies were negative. These revealed no significant difference in the complexity of the microflora. No significant transfer of organisms between crewmembers has been noted, and there has been no "locking in" of floral patterns through 14 days.

Postflight ear, nose, and throat examinations have consistently been negative, and caloric examinations before and after each flight have been normal. On each of the long-duration missions, the crews have reported nasal drying and stuffiness, and this has been evident by the nasal voice quality during voice communication with the surgeon at the Mission Control Center. This symptom has lasted varying amounts of time but has been most evident in the first few days of the mission. The negative postflight findings have been of interest in view of these inflight observations. The crews have reported they found it necessary to clear their ears frequently in inflight. Some of this nasal and pharyngeal congestion has been noted in the long-duration space cabin simulator runs in a similar environment. It may be related to dryness, although the cabin humidity would not indicate this to be the case, or another cause might be the pure oxygen atmosphere in the cabin. It may also be related to a possible change in blood supply to the head and thorax as a result of circulatory adaptation to weightlessness.

The oral hygiene of the crewmembers has been checked closely before each flight and has been maintained inflight by the use of a dry toothbrush and a chewable dental gum. This technique provided excellent oral hygiene through the 14-day flight.

Weight

A postflight weight loss has been noted for each of the crewmembers; however, it has not increased with mission duration and has varied from 2.5 to 10 pounds. The majority of the loss has been replaced with fluid intake within the first 10 to 12 hours after landing. Table 25-V shows the weight loss and postflight gain recorded for the crewmen of the long-duration flights.

TABLE 25-V.—*Astronaut Weight Loss*

Gemini mission	Command pilot weight loss, lb	Pilot weight loss, lb
III	3	3. 5
IV	4. 5	8. 5
V	7. 5	8. 5
VI-A	2. 5	8
VII	10	6

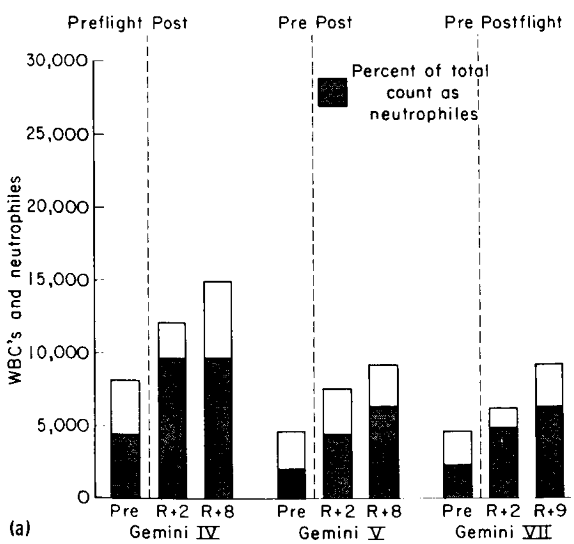
Hematology

Clinical laboratory hematologic studies have been conducted on all missions, and some interesting findings have been noted in the white-blood-cell counts. The changes are shown in figure 25-15 (a) and (b). It can be seen that on the 4-day flight there was a rather marked absolute increase in white blood cells, specifically neutrophiles, which returned to normal within 24 hours (though not shown in the figure). This finding was only minimally present following the 8-day flight and was noted again following the 14-day flight. It very likely can be explained as the result of an epinephrine response. The red-cell counts show some postflight reduction that tends to confirm the red-cell mass data to be discussed.

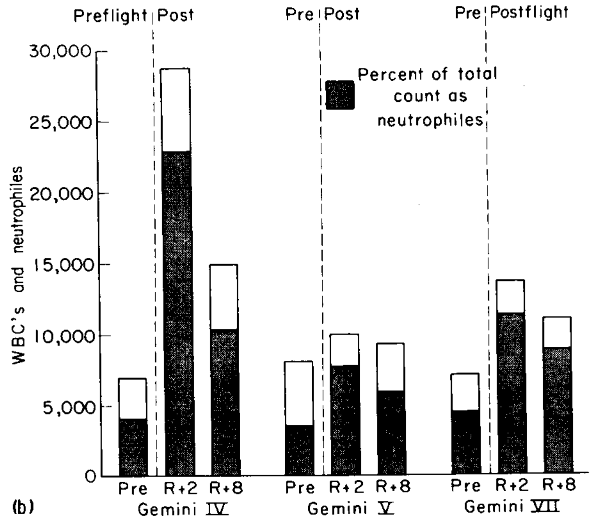
Urine and blood chemistry tests have been performed before and after each of the missions, and the results may be seen in tables 25–VI and 25–VII. The significant changes noted will be discussed in the experiments report.

Blood Volume

On each of the long-duration flights, plasma volume has been determined by the use of a technique utilizing radio-iodinated serum albumin. On the 4-day mission, the red-cell mass was calculated by utilizing the hematocrit determination. Analysis of the data caused some concern as to the validity of the hematocrit in view of the dehydration noted. The 4-day mission data showed a 7- and 15-percent decrease in the circulating blood volume for the two crewmembers, a 13-percent decrease in plasma volume, and an indication of a 12- and 13-percent decrease in red-cell mass, although it had not been directly measured. As a result of these findings, red cells were tagged with chromium 51 on the 8-day mission in order to get an accurate measurement of red-cell mass while continuing to utilize the radio-iodinated serum albumin technique for plasma volume. The



(*a*) Command pilots.
FIGURE 25–15.—White blood cell response.



(*b*) Pilots.
FIGURE 25–15.—Concluded.

chromium-tagged red cells also provided a measure of red-cell survival time. At the completion of the 8-day mission, there was a 13-percent decrease in blood volume, a 4- to 8-percent decrease in plasma volume, and a 20-percent decrease in red-cell mass. These findings pointed to the possibility that the red-cell mass decrease might be incremental with the duration of exposure of the space-flight environment. The 14-day flight results show no change in the blood volume, a 4- and 15-percent increase in plasma volume, and a 7- and 19-percent decrease in red-cell mass for the two crewmembers. In addition to these findings, the red-cell survival time has been reduced. All of these results are summarized in figure 25–16. It can be concluded that the decrease in red-cell mass is not incremental with increased exposure to the space-flight environment. On the 14-day flight, the maintenance of total blood volume, by increasing plasma volume, and the weight loss noted indicated that some fluid loss occurred in the extracellular compartment but that the loss had been replaced by fluid intake after the flight. The detailed explanation of the decreased mass is unknown at the present time, and several factors, including the atmosphere, may be involved. This loss of red cells has not interfered with normal function and is generally equivalent to the blood withdrawn in a blood-bank donation, but the decrease occurs over a longer period of time, and this allows for adjustment.

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TABLE 25-VI.—*Gemini VII Urine Chemistries*

[All dates 1965]

Determination	Command pilot							Pilot						
	Preflight	Postflight						Preflight	Postflight					
	Nov. 23 and Dec. 1	Dec. 18		Dec. 20		Dec. 21		Nov. 23 and Dec. 1	Dec. 18		Dec. 20		Dec. 21	
		Measured	Percent of preflight	Measured	Percent of preflight	Measured	Percent of preflight		Measured	Percent of preflight	Measured	Percent of preflight	Measured	Percent of preflight
Sodium, $\frac{\text{meg}}{\text{24 hr}}$	143	95	66	182	127	150	105	150	76	51	94	63		
Potassium	71	118	166	93	131	90	127	70	60	86	89	127		
Chlorine	141	89	63	168	119	145	103	141	67	48	73	52		
Calcium, $\frac{\text{mg}}{\text{24 hr}}$	228	269	118	260	114	210	92	184	89	48	105	57		
Phosphate	1131	2133	188	936	83	978	86	1200	996	83	1345	112		
17-hydroxycorticosteriods	7.7	18.6	241	7.3	95	9.1	118	6.2	11.3	183	8.1	130	8.2	132
Ephinephrine, $\frac{\mu\text{g}}{\text{24 hr}}$	7.8	16.4	210	(a)		(a)		10.2						
Norepinephrine	50.3	103.0	204	(a)		(a)		42.7						
Aldosterone, $\frac{\mu\text{g}}{\text{24 hr}}$	26	75	288			28	108	26	47	181			60	230
Creatine, $\frac{\text{mg}}{\text{24 hr}}$	2035	3297	162	1380	68	2070	102	2230	2003	90	2225	100		

a Not significant.

TABLE 25-VII.—*Gemini VII Blood Chemistry Studies for Command Pilot*

Determination	Preflight		Postflight			
	Nov. 24 and Nov. 25, 1965	Nov. 30 and Dec. 2, 1965	Dec. 18, 1965		Dec. 19, 1965	Dec. 20 and Dec. 21, 1965
			11:30 a.m., e.s.t.	6:20 p.m., e.s.t.		
Blood urea nitrogen, mg percent	19	16	16	20	25	18
Bilirubin, total mg percent	.4	.2	.3	----------	.3	.4
Alkaline phosphatase (B–L units)	1.7	2.0	1.7	----------	----------	----------
Sodium, meq/liter	147	146	138	140	144	143
Potassium, meq/liter	4.7	5.4	4.1	4.7	4.7	4.9
Chloride, meq/liter	103	103	100	102	103	106
Calcium, mg percent	9.0	9.2	8.6	9.2	9.0	9.2
Phosphate, mg percent	3.2	3.7	4.0	3.2	3.1	3.6
Glucose, mg/100 ml, nonfasting	71	90	98	----------	----------	----------
Albumen, g percent	4.6	4.73	5.16	----------	4.5	4.6
Alpha 1, g percent	.23	.26	.08	----------	----------	----------
Alpha 2, g percent	.40	.39	.40	----------	----------	----------
Beta, g percent	.63	.84	.72	----------	----------	----------
Gamma, g percent	1.03	.97	.72	----------	----------	----------
Total protein, g percent	6.9	7.2	7.1	7.6	7.0	7.1
Uric acid, mg percent	6.8	6.6	4.6	6.0	5.9	6.0

Tilt Studies

The first abnormal finding noted following manned space flight was the postflight orthostatic hypotension observed on the last two Mercury missions. Study of this phenomenon has been continued in order to develop a better appreciation of the physiological cost of manned space flight. A special saddle tilt table, shown in figure 25–17, has been used, and the tilt-table procedure has been monitored with electronic equipment providing automatic monitoring of blood pressure, electrocardiogram, heart rate, and respiration. The procedure consists of placing the crewman in a horizontal position for 5 minutes for stabilization, tilting to the 70° head-up position for 15 minutes, and then returning to the horizontal position for another 5 minutes. In addition to the usual blood pressure and pulse rate determinations at minute intervals, some mercury strain gages have been used to measure changes in the circumference of the calf. On the 4-day, 8-day, and 14-day missions there were no symptoms of faintness experienced by the crew at any time during the landing sequence or during the postlanding operation. Abnormal tilt-table responses, when compared with the preflight baseline tilts, have been noted for a period of

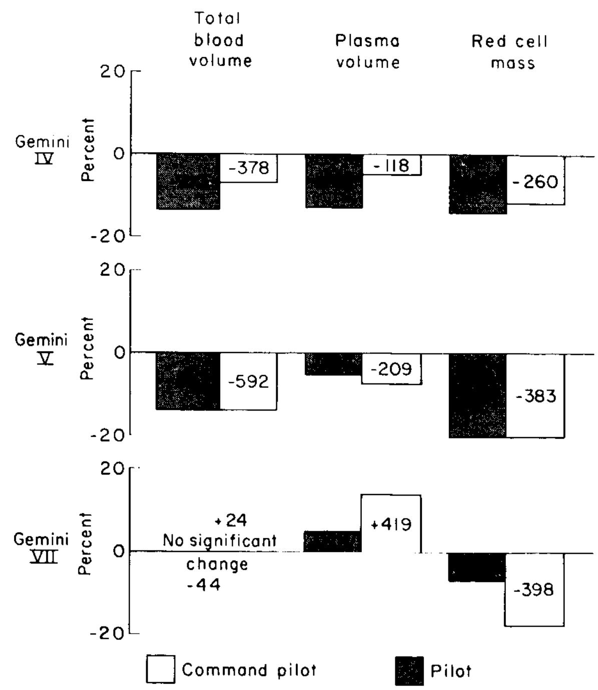


FIGURE 25–16.—Blood volume studies.

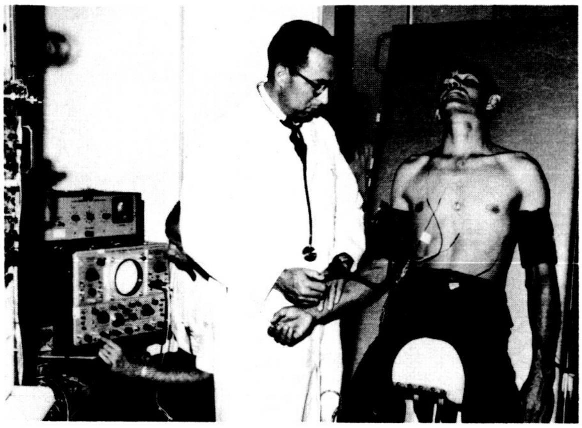


FIGURE 25–17.—Tilt-table test.

48 to 50 hours after landing. Typical initial postlanding tilt responses are graphed for the 4-day and 8-day mission crews in figures 25–18 through 25–21. A graph of the percentage increase in heart rate from baseline normal to that attained during the initial postflight tilt can be seen in figure 25–22. All of the data for Gemini III through VI–A fell roughly on a linear curve. The projection of this line for the 14-day mission data would lead one to expect very high heart rates or possible syncope. It was not believed this would occur. The tilt responses of the 14-day mission crew are shown in figures 25–23 and 25–24.

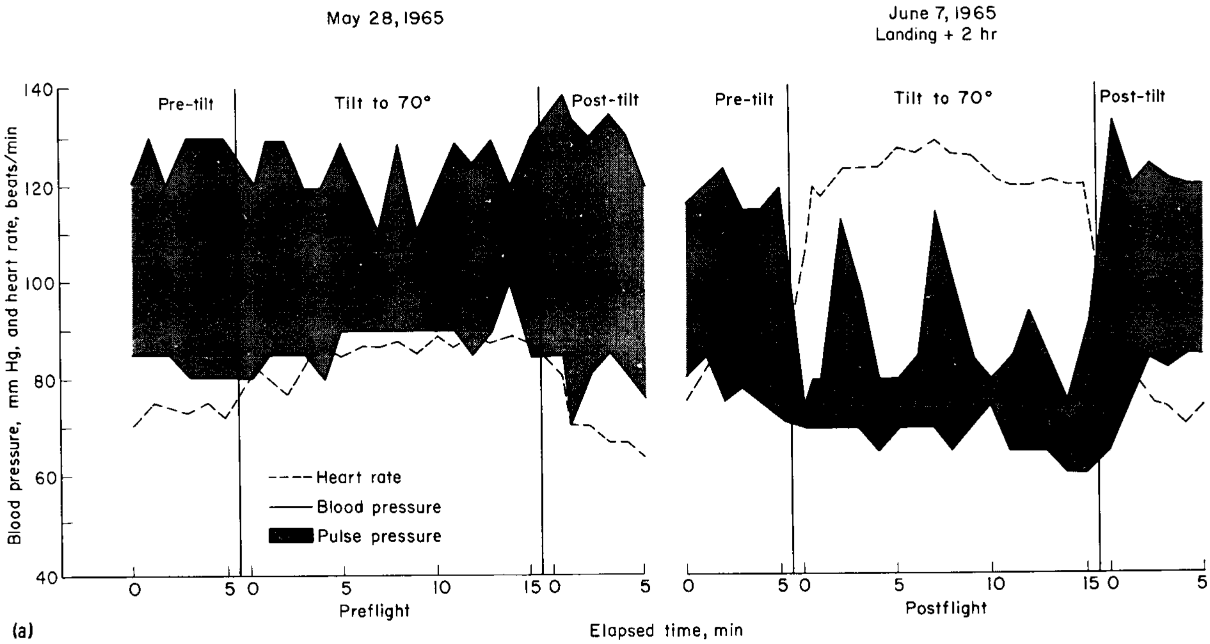
The response of the command pilot is not unlike that of previous crewmen, and the peak heart rate attained is more like that seen after 4 days of space flight. The tilt completed 24 hours after landing is virtually normal. The pilot's tilt at 1 hour after landing is a good example of individual variation, for he had a vagal response, and the heart rate, which had reached 128, dropped, as did the blood pressure, and the pilot was returned to the horizontal position at 11 minutes. Subsequent tilts were similar to previous flights, and the response was at baseline values in 50 hours. When these data are plotted on the curve in figure 25–22, it will be noted that they more closely resemble 4-day mission data. There has been no increase in the time necessary to return to the normal preflight tilt response, a 50-hour period, regardless of the duration of the flight. The strain-gage data generally confirm pooling of blood in the lower extremities during the period of roughly 50 hours that is required to readjust to the 1-g environment. The results of these studies may be seen in figure 25–25.

Bicycle Ergometry

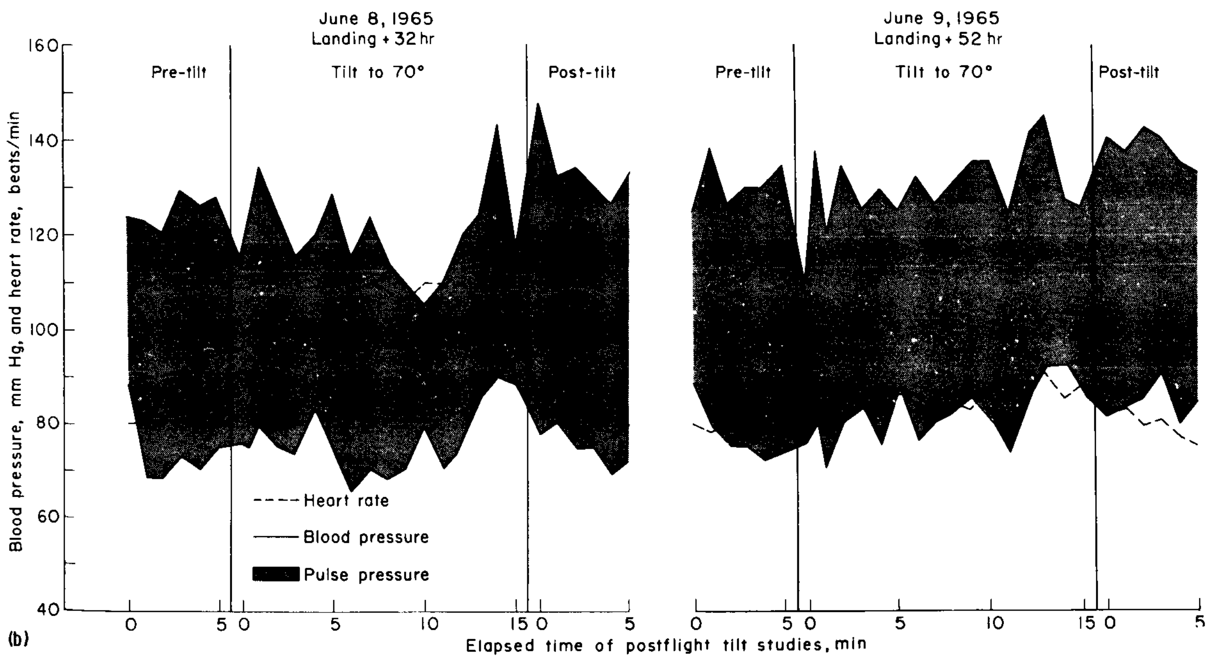
In an effort to further assess the physiologic cost of manned space flight, an exercise capacity test was added for the 14-day mission. This test utilized an electronic bicycle ergometer pedaled at 60 to 70 revolutions per minute. The load was set at 50 watts for 3 minutes and increased by 15 watts during each minute. Heart rate, respiration rate, and blood pressure were recorded at rest and during the last 20 seconds of each minute during the test. Expired air was collected at several points during the test, which was carried to a heart rate of 180 beats per minute. Postflight results demonstrated a decrease in work tolerance, as measured by a decrease in time necessary to reach the end of the test, amounting to 19 percent on the command pilot and 26 percent on the pilot. There was also a reduction in physical competence measured as a decrease in oxygen uptake per kilogram of body weight during the final minute of the test.

Medical Experiments

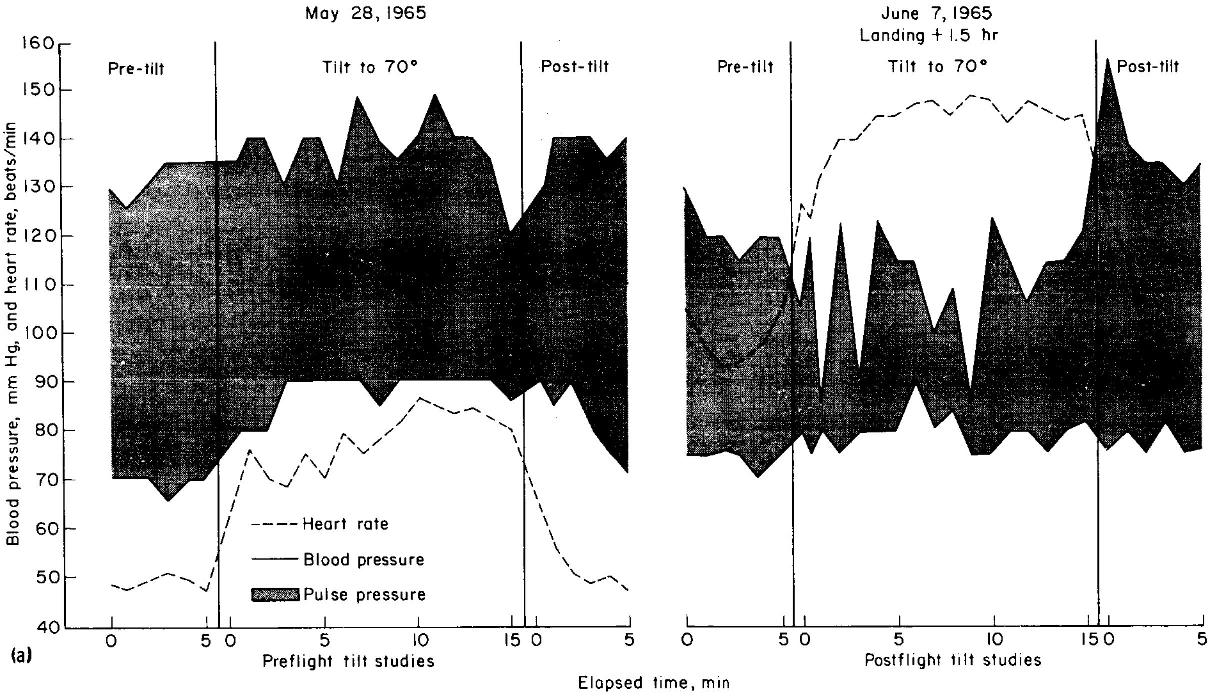
Certain procedures have been considered of such importance that they have been designated operationally necessary and have been performed in the same manner on every mission. Other activities have been put into the realm of specific medical experiments in order to answer a particular question or to provide a particular bit of information. These investigations have been programed for specific flights. An attempt has been made to aim all of the medical investigations at those body systems which have indicated some change as a result of our earlier investigations. Thus, attempts are not being made to conduct wide surveys of body activity in the hope of finding some abnormality, but the investigations are aimed at specific targets. A careful evaluation is conducted on the findings from each flight, and a modification is made to the approach based upon this evaluation in both the operational and experimental areas. Table 25–VIII shows the medical experiments which have been conducted on the Gemini flights to date.



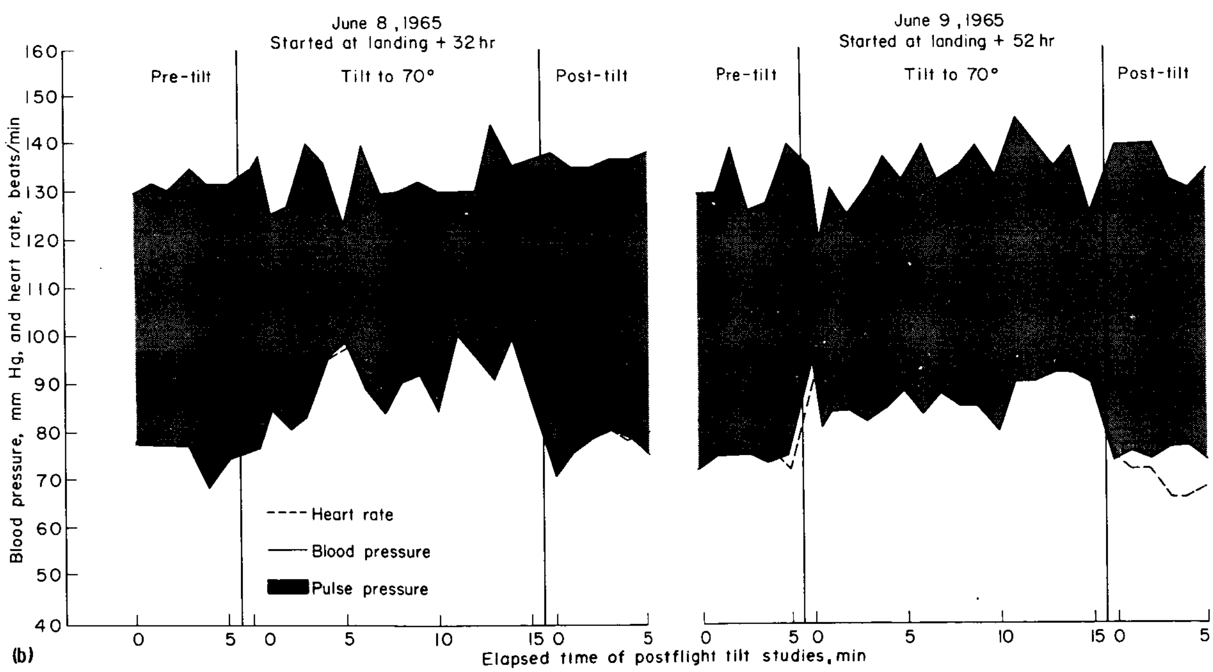
(*a*) Studies conducted preflight and at 2 hours after landing.
FIGURE 25-18.—Tilt-table studies of Gemini IV command pilot.



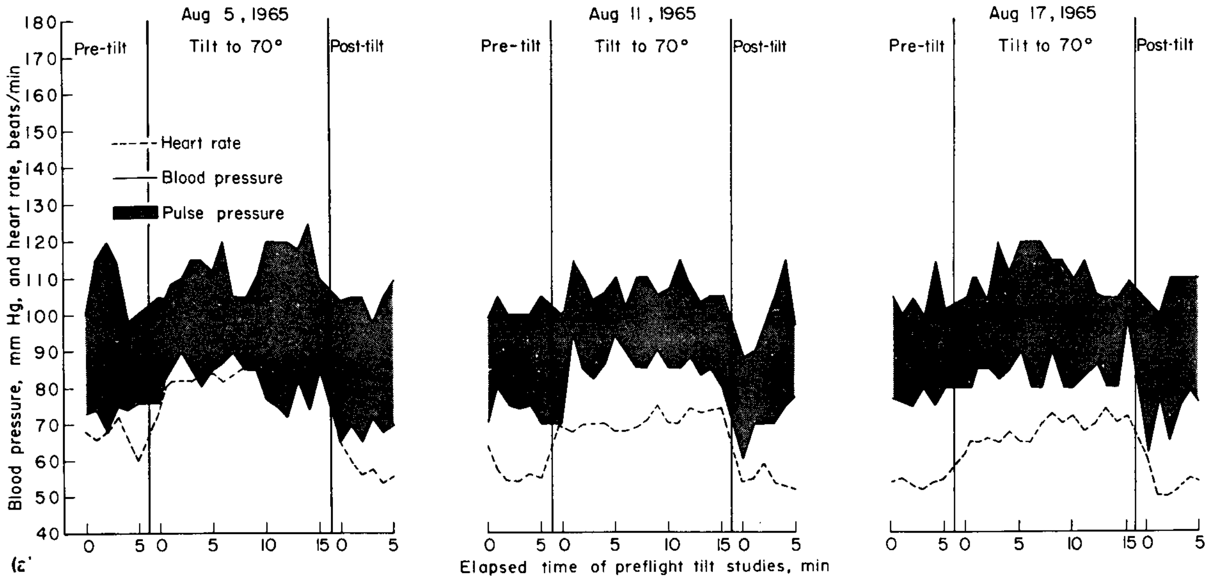
(*b*) Studies conducted 32 hours and 52 hours after landing.
FIGURE 25-18.—Concluded.



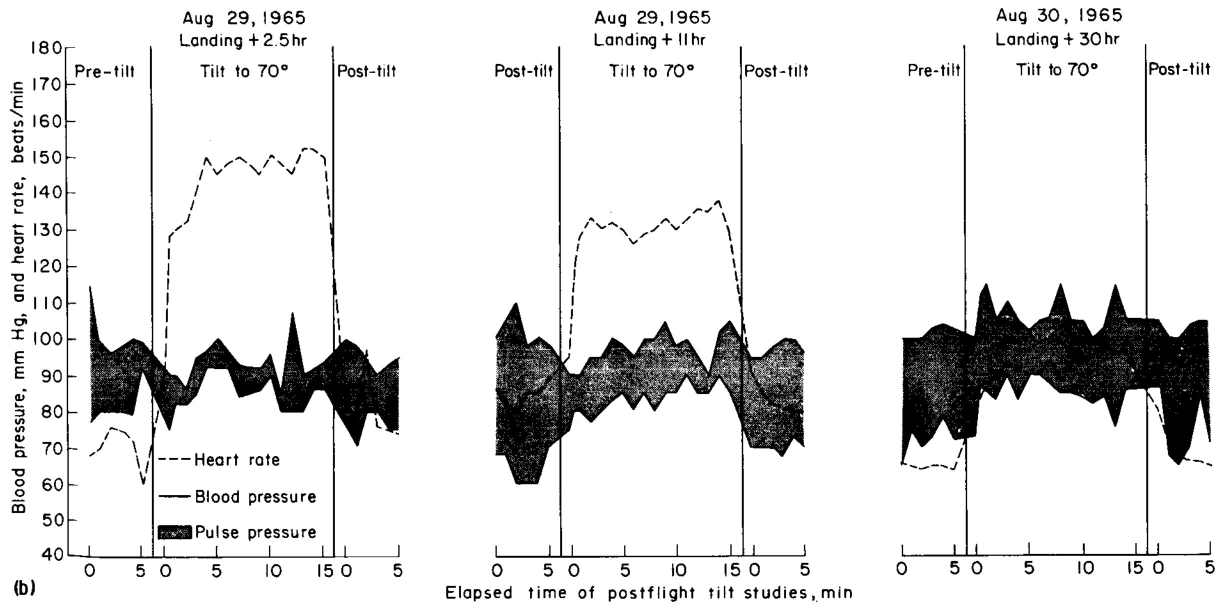
(*a*) Studies conducted preflight and 1.5 hours after landing.
FIGURE 25–19.—Tilt-table studies of Gemini IV pilot.



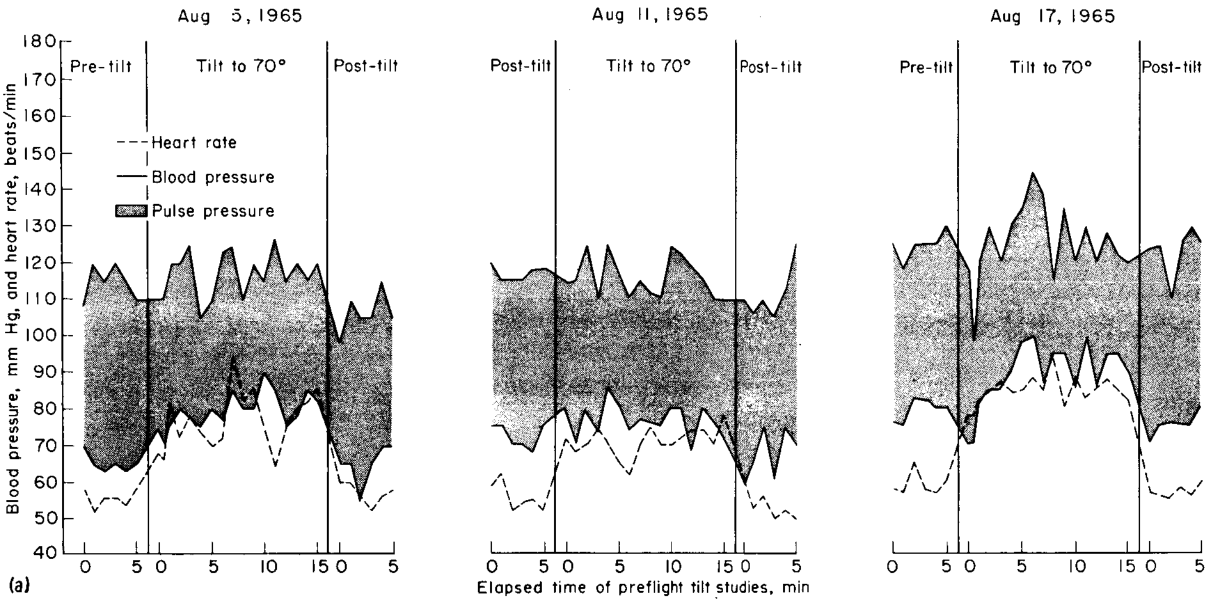
(*b*) Studies conducted 32 hours and 52 hours after landing.
FIGURE 25–19.—Concluded.



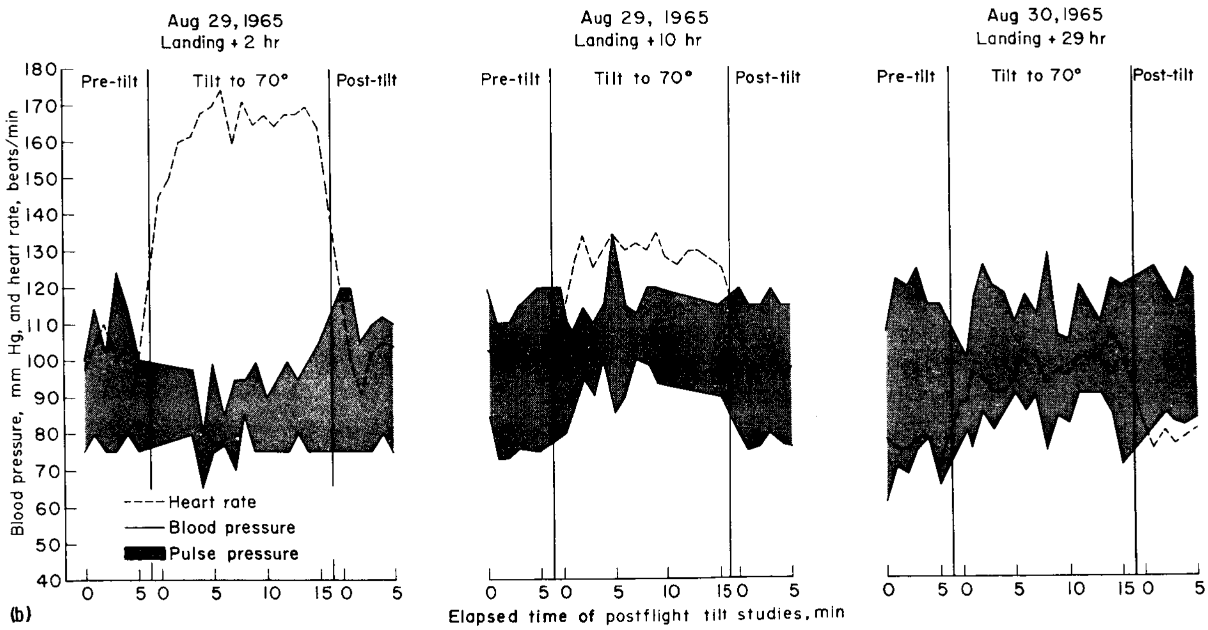
(*a*) Preflight.
FIGURE 25–20.—Tilt-table studies of Gemini V command pilot.



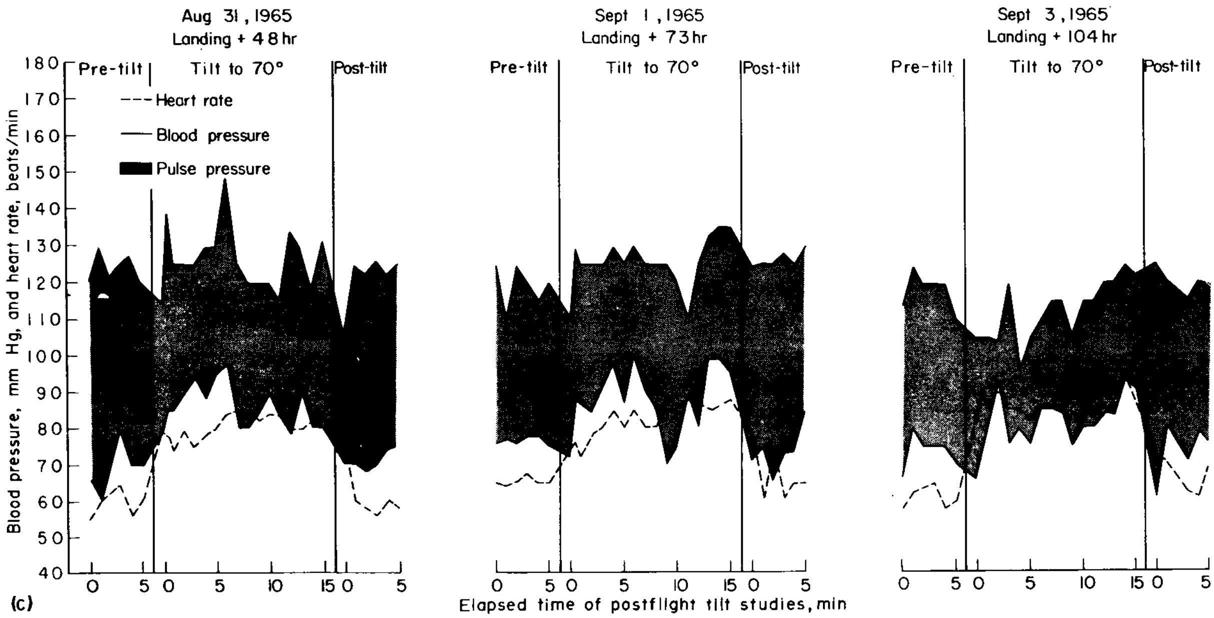
(*b*) Postflight.
FIGURE 25–20.—Concluded.



(*a*) Preflight.
FIGURE 25-21.—Tilt-table studies of Gemini V pilot.



(*b*) Studies conducted at 2, 10, and 29 hours after landing.
FIGURE 25-21.—Continued.



(c) Studies conducted at 48, 73, and 104 hours after landing.
FIGURE 25-21.—Concluded.

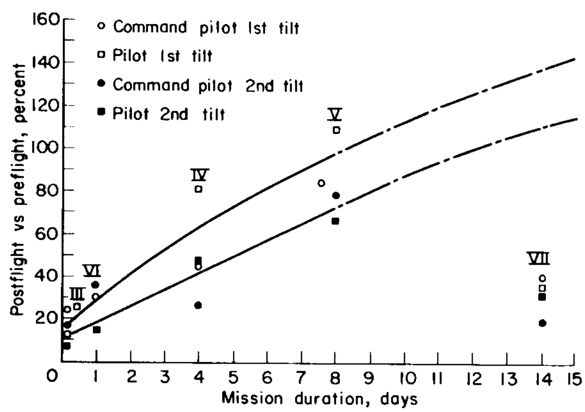
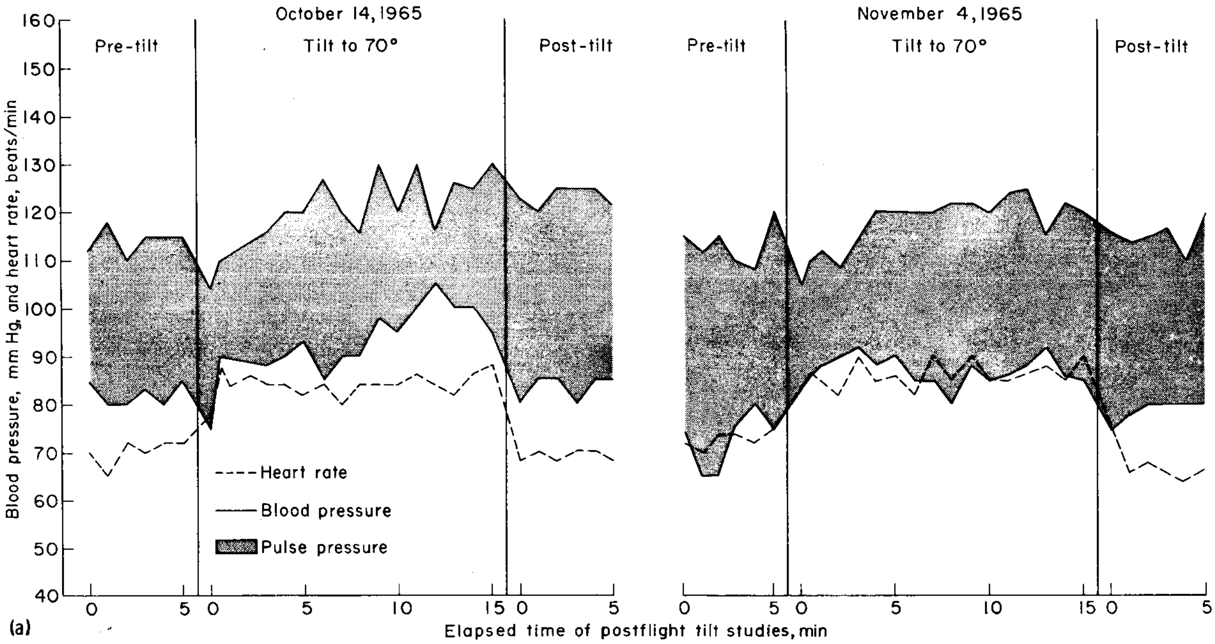
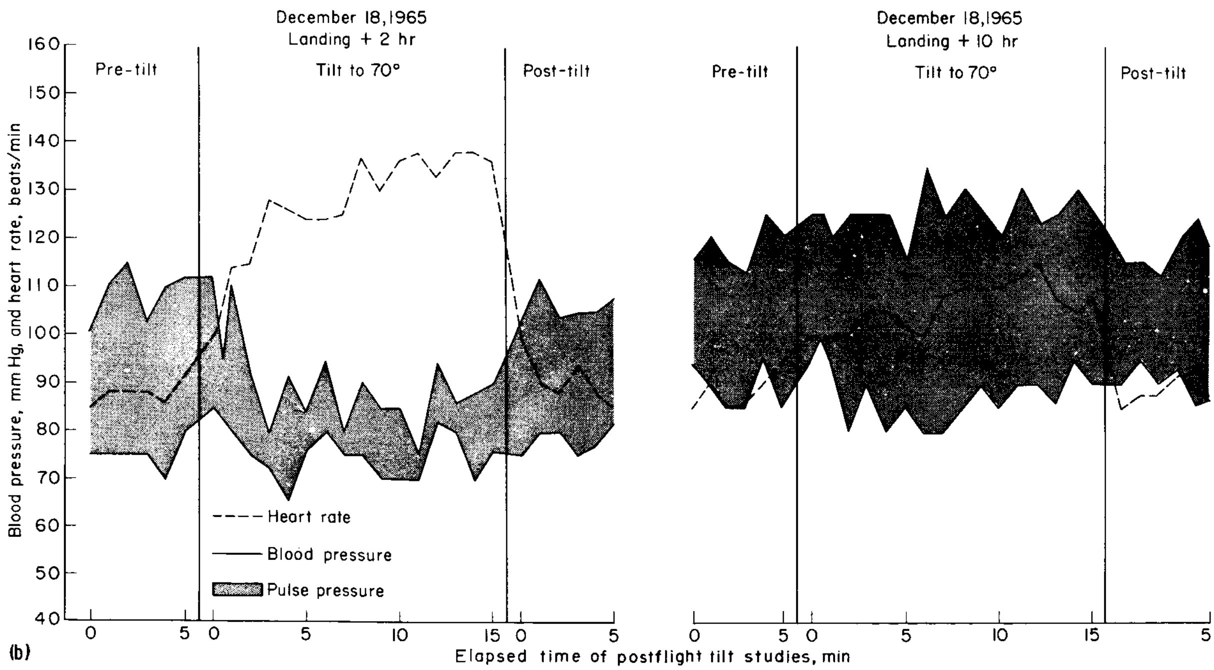


FIGURE 25-22.—Heart-rate tilt response compared with mission duration.



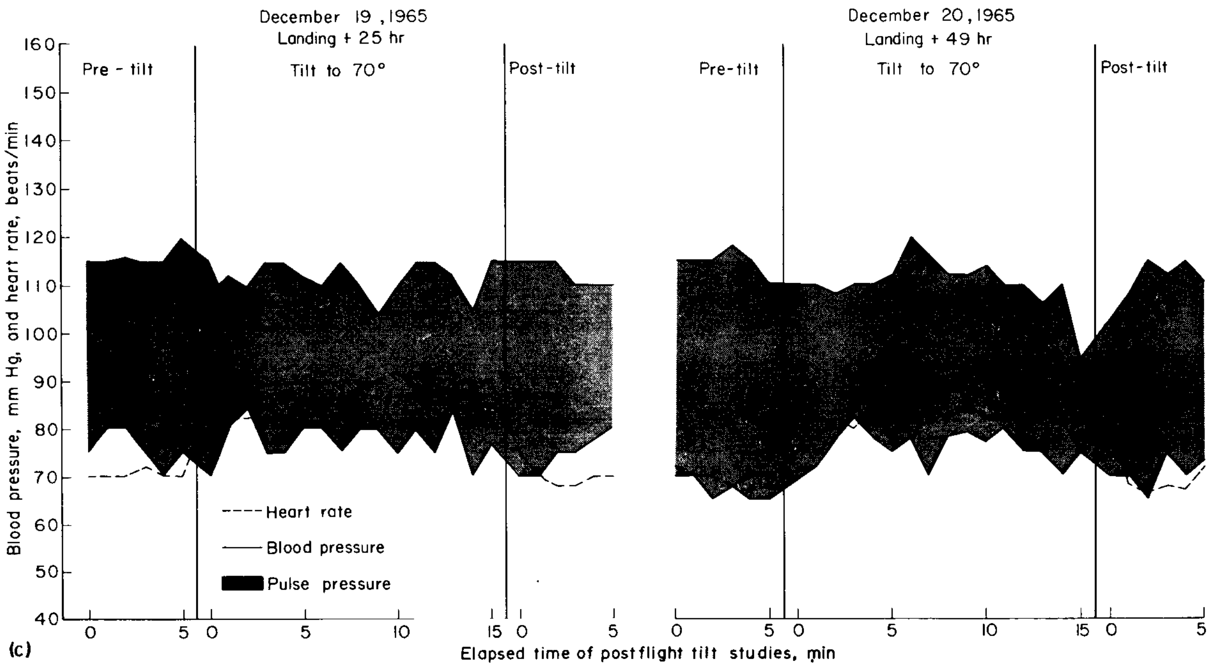
(*a*) **Preflight.**

FIGURE 25-23.—Tilt-table studies of Gemini VII command pilot.

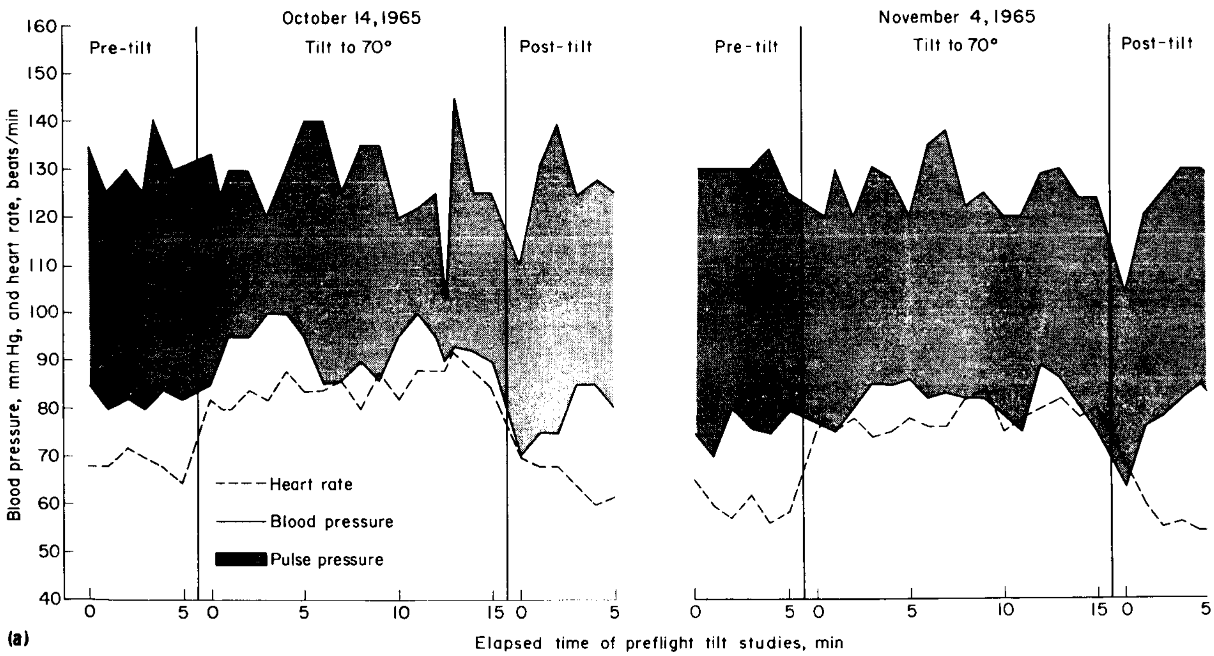


(*b*) **Studies conducted at 2 and 10 hours after landing.**

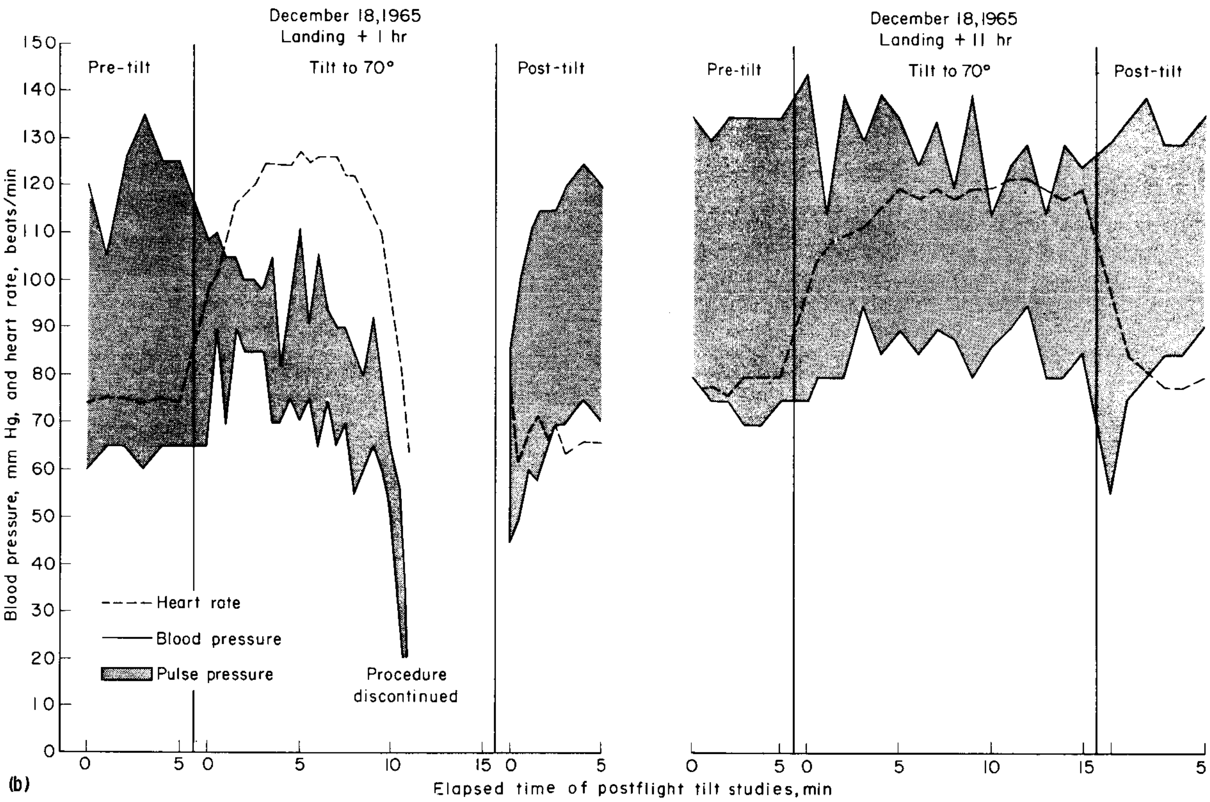
FIGURE 25-23.—Continued.



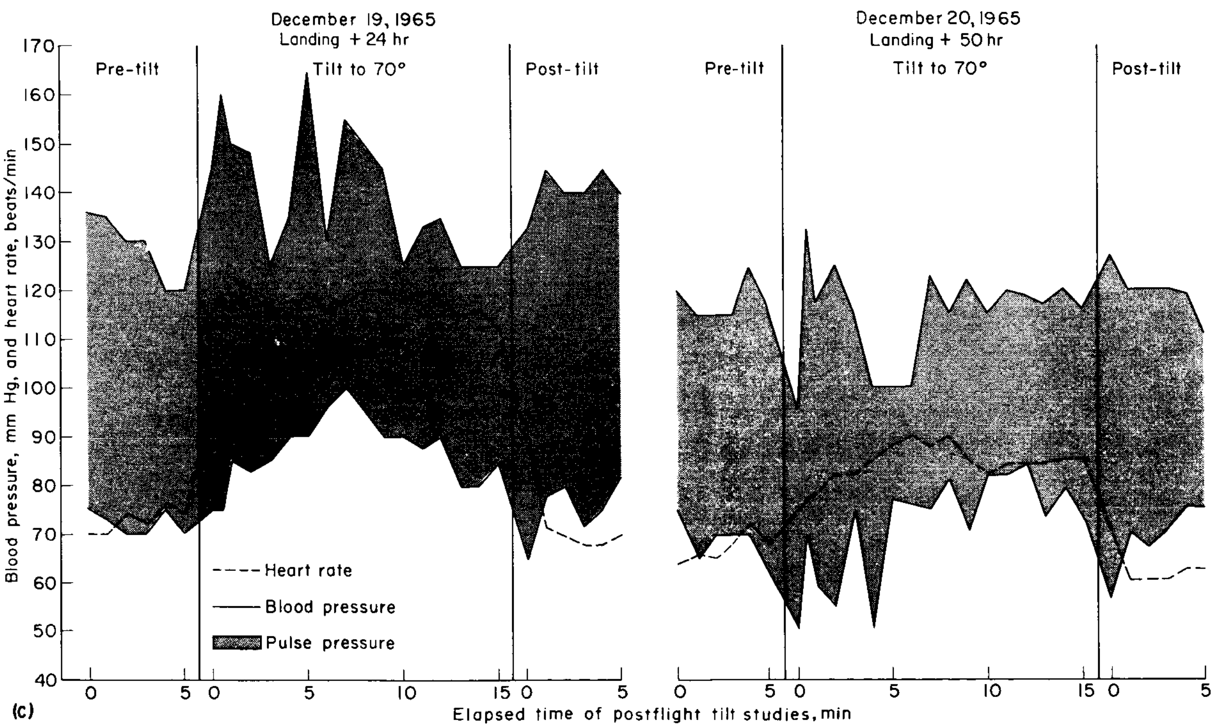
(*c*) Studies conducted at 25 and 49 hours after landing.
FIGURE 25-23.—Concluded.



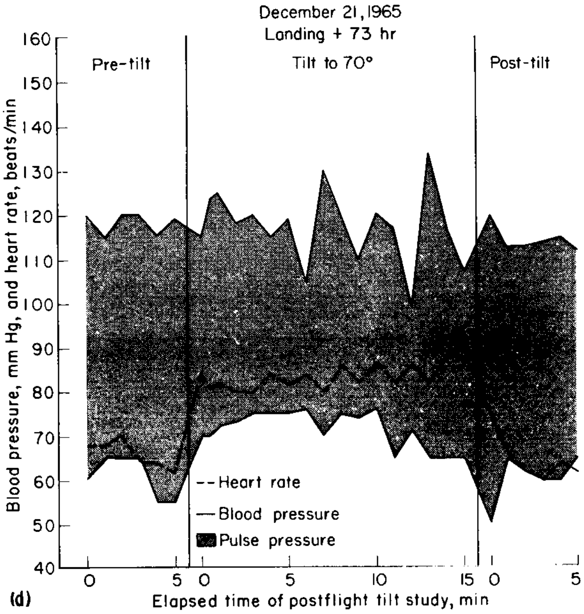
(*a*) Preflight.
FIGURE 25-24.—Tilt-table studies of Gemini VII pilot.



(*b*) Studies conducted at 1 and 11 hours after landing.
FIGURE 25–24.—Continued.



(*c*) Studies conducted at 24 and 50 hours after landing.
FIGURE 25–24.—Continued.



(*d*) Study conducted at 73 hours after landing.
FIGURE 25-24.—Concluded.

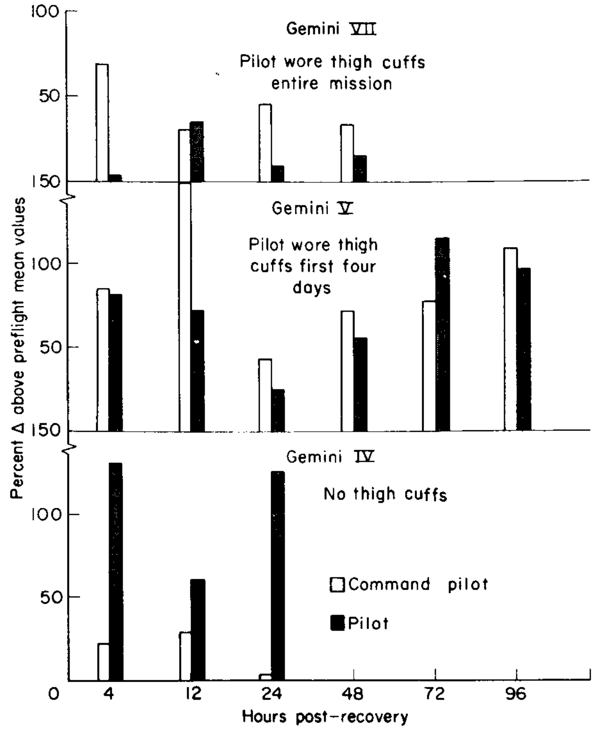


FIGURE 25-25.—Leg volume changes during postflight tilt-table studies.

TABLE 25-VIII.—*Medical Experiments on Gemini Long-Duration Missions*

Code	Short title	Gemini IV, 4 days	Gemini V, 8 days	Gemini VII, 14 days
M-1	Cuffs	----------	X	X
M-2	Tilt table	Include as medical operations procedure		
M-3	Exercise tolerance	X	X	X
M-4	Phonocardiogram	----------	X	X
M-5	Body fluids	----------	----------	X
M-6	Bone densitometry	X	X	X
M-7	Calcium and nitrogen balance study	----------	----------	X
M-8	Sleep analysis	----------	----------	X
M-9	Otolith function	----------	X	X

Radiation

The long-duration flights have confirmed previous observations that the flight crews are exposed to very low radiation dose levels at orbital altitudes. The body dosimeters on these missions have recorded only millirad doses which are at an insignificant level. The recorded doses may be seen in table 25-IX.

Concluding Remarks

A number of important medical observations during the Gemini flights have been made without compromising man's performance. It can be stated with certainty that all crewmen have performed in an outstanding manner and have adjusted both psychologically and physiologically to the zero-g environment and then readjusted to a 1-g environment with no undue symptomatology being noted. Some of the findings noted do require further study, but it is felt that the experience gained through the 14-day Gemini VII mission provides great confidence in any crewman's ability to complete an 8-day lunar mission without any unforeseen psychological or physiological change. It also appears that man's responses can be projected into the future to allow 30-day exposures in larger spacecraft. The predictions thus far have been valid. Our outlook to the future is extremely optimistic, and man has shown his capability to fulfill a role as a vital, functional part of the spacecraft as he explores the universe.

TABLE 25–IX.—*Radiation Dosage on Gemini Long-Duration Missions*

[In millirads]

Mission	Command pilot	Pilot
Gemini IV [a]	38.5 ± 4.5	42.5 ± 4.7
	40.0 ± 4.2	45.7 ± 4.6
	42.5 ± 4.5	42.5 ± 4.5
	45.0 ± 4.5	69.3 ± 3.8
Gemini V [a]	190 ± 19	140 ± 14
	173 ± 17.3	172 ± 17.2
	183 ± 18.3	186 ± 18.6
	195 ± 19.5	172 ± 17.2
Gemini VII [b]	178 ± 10	98.8 ± 10
	105 ± 10	215 ± 15
	163 ± 10	151 ± 10

[a] Values are listed in sequence: left chest, right chest, thigh, and helmet.

[b] Values are listed in sequence: left chest, right chest, and thigh.

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26. DATA ANALYSIS AND REPORTING

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Summary

The acquisition of vast quantities of data combined with a need to evaluate and quickly resolve mission anomalies has resulted in a new approach to data reduction and test evaluation. The methodology for selective reduction of data has proved effective and has allowed a departure from the traditional concept that all test data generated must be reduced. Real-time mission monitoring by evaluation engineers has resulted in a judicious selection of flight segments for which data need to be reduced. This monitoring, combined with the application of compression methods for the presentation of data, has made it possible to complete mission evaluations on a timely basis.

Introduction

Data reduction and flight test evaluation plans for the Gemini Program were conceived in 1963, and implementation began with the first unmanned qualification flight in April 1964. The objective of these plans was to insure swift but thorough mission evaluations, consistent with the schedule for Gemini flights.

Data Processing

The quantity of data to be made available during each Gemini flight had a significant effect on the planning for data reduction. Table 26-I shows the impossible data-reduction task on the spacecraft alone that confronted the data processors in the planning stage. Obviously, even if all of these data were reduced, the manpower and time could not be afforded to examine it. Gemini is not being flown to provide information on its system, but rather for studying the operational problems associated with space flight. However, the inevitable system problems that occur must be recognized and corrected. Overall system performance was stressed in the selection of parameters to be measured. This action, however, succeeded only in reducing the data acquisition to what is shown in table 26-I. In developing the overall Gemini data reduction and evaluation plans, two main questions had to be answered: (1) Where would the data be reduced? (2) How much of the orbital telemetry data could be processed effectively?

TABLE 26-I.—*Gemini Flight Data Production Rate*

Each second:	
Real time	51 200 bits
Delayed time	5120 bits
Each revolution:	
Delayed-time analog	2 000 000 data points
Delayed-time events	4 000 000 interrogations
Gemini V (8-day mission):	
Delayed-time analog	250 000 000 data points
Tabulations required	1 000 000 pages
Plots required	750 000 pages

A review was initiated to study the experience gained during Project Mercury and to determine the reduction capabilities that existed within the various Gemini organizations, or that would exist in the near future. The data reduction plan that emerged from this review was documented in a Gemini Data Reduction and Processing Plan. A summary of where the telemetry data were to be reduced is shown in table 26-II.

Recognizing that all data from the first, second, and third missions could be reduced and analyzed, it was decided to do just that and to develop the approach for data reduction and analyses for later missions from that experience. It rapidly became apparent that selective data reduction and analyses would be necessary. It was decided that key systems engineers from the appropriate organizations—such as the spacecraft contractor or his subcontractor, the target vehicle contractor, the Air Force, and NASA—should closely monitor the flight by using the real-time information facilities in the Mission Control Center at Houston and the facility at the Kennedy Space Center. This close monitoring of engineering data would permit the selection of only those segments of the mission data necessary to augment or to verify the real-time information for postflight evaluation. All the data for periods of high activity covering dynamic conditions such as launch, rendezvous, and reentry would be reduced and analyzed. Any further data reduction would be accomplished on an as-required basis. The outcome of these plans is shown in table 26–III. The percentage of flight data processed for postflight evaluation was substantially decreased after the first manned, three-orbit flight.

Reduction Operations

Even with the reduced percentage of flight data processed, the magnitude of the task cannot be discounted. Table 26–IV shows the data processing accomplished in support of the postflight evaluation of the 8-day Gemini V mission. More than 165 different data books were produced in support of the evaluation team. For this mission, the Central Metric Data file at the Manned Spacecraft Center received 4583 data items.

TABLE 26–II.—*Telemetry Data Processing Plan*

Mission	Computer-processed data			Kennedy Space Center
	Manned Spacecraft Center	McDonnell Aircraft Corp.	Air Force	
Gemini I	Backup, spacecraft	Prime, spacecraft	Launch vehicle	Quick-look oscillographs, spacecraft and launch vehicle
Gemini II	Prime, spacecraft	Backup, spacecraft	Launch vehicle	Quick-look oscillographs, spacecraft and launch vehicle
Gemini III through Gemini VII	Launch and orbit, spacecraft	Reentry, spacecraft	Launch vehicle	Quick-look computer plots: Launch; Real-time, spacecraft; Delayed-time, spacecraft (Cape Kennedy passes)

TABLE 26–III.—*Postflight Data Reduction for Mission Evaluation*

Mission	Data available	Data reduced
Gemini I	Launch plus 3 revolutions	All
Gemini II	Launch, flight and reentry	All
Gemini III	Launch, reentry, 3 revolutions	All
Gemini IV	Launch, reentry, 62 revolutions	Launch, reentry, 29 revolutions
Gemini V	Launch, reentry, 120 revolutions	Launch, reentry, 39 revolutions
Gemini VII	Launch, reentry, 206 revolutions	Launch, reentry, 41 revolutions, 14 station passes
Gemini VI–A	Launch, reentry, 16 revolutions	Launch, reentry, 9 revolutions, 3 station passes

Very few data reduction centers have grown as fast as the one at the Manned Spacecraft Center. Just 4 years ago this Center was only a field of grass, and, today, combining the Mission Control Center and the Computation and Analysis Division computer complexes, it houses one of the largest data processing and display capabilities in the world. Figure 26–1 shows a floor plan and some of the major devices employed for data processing in the Computation and Analysis Building.

It became very clear during the evaluation of the first three flights that it would be impossible to plot or tab all of the selected data from the longer duration flights. Computers can look at volumes of data in seconds, but they require many hours to print data in a usable form. Many more tedious hours are required to manually scan the data for meaningful information. Recognizing these facts, the data processing programs were revised to include compression methods of the presented data. These methods include presentation of the mean value over a specified time interval along with the maximum and minimum values during the interval or presentation of only data that go beyond a predetermined value of sigma. Also possible is the presentation of only the data falling outside a predetermined band having a variable mean as a function of time or as a function of other measured or predetermined values. Smoothing and wild-point editing may also be applied in a judicious manner. An example might be the presentation of all valid points of the fuel-cell voltage-current curve falling outside a predetermined band. This involves bus voltage multiplied by the sum of the stack currents in a section along a predetermined degradation curve for given values of total section current.

Systems evaluation during the flight for selection of requirements, combined with compression methods for data processing, made possible the processing of the mass of recorded data for support of the mission evaluation team on a schedule consistent with the Gemini Program requirements.

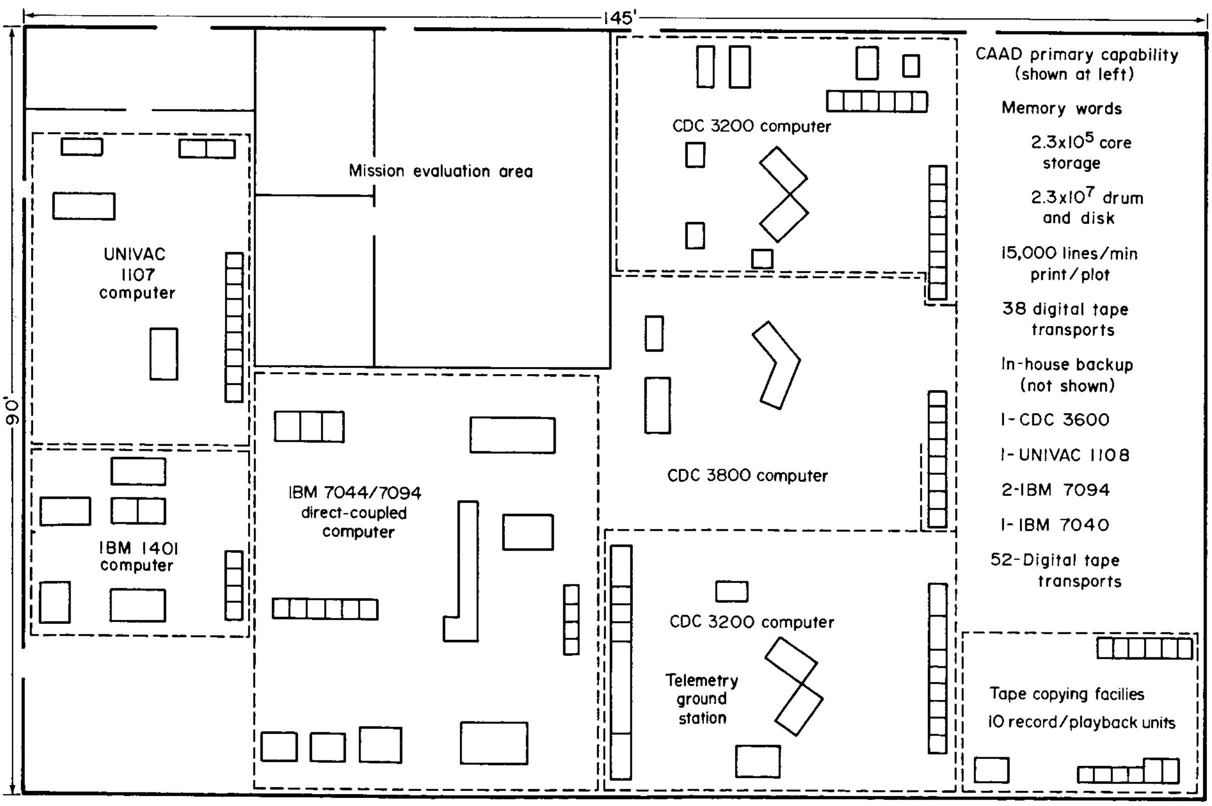


Figure 26–1.—Data reduction facilities of the Computation and Analysis Division.

TABLE 26-IV.—*Gemini V Reduction Task*

Telemetry tapes processed:	
Delayed-time data	55 tapes
Real-time data	16 tapes
Time edit analysis	129 tapes
Time history presentation:	
Plots (selected parameters)	14 revolutions
Tabulations (selected parameters)	15 revolutions
Statistical plots	15 revolutions
Statistical tabulations	30 revolutions
Event tabulations	30 revolutions
Ascent phase special computations:	
Computer word time correction	All
Aerodynamic parameters	All
Steering deviations	All
Angle of attack	All
Orbital phase special computations:	
Ampere-hour	24 revolutions
Orbital attitude and maneuver system propellant remaining	6 revolutions
Orbital attitude and maneuver system thruster activity	3 revolutions
Experiment MSC-1	90 minutes of flight
Coordinate transformation	20 minutes of flight
Reentry phase:	
Lift-to-drag ratio	All
Angle of attack	All
Reentry control system propellant remaining	All
Reentry control system thruster activity	All

Postmission Evaluation

Evaluation Planning

Plans were begun in the fall of 1963 for the postflight evaluation of the Gemini missions. This early planning culminated in the Gemini Program Mission Evaluation and Reporting Plan, which documented the procedures for mission evaluation and outlined the format of the report.

The most important consideration of these plans was to assure that evaluation was completed and a report generated for each mission in sufficient time to apply the knowledge gained to the next mission. Optimum use of personnel and time was required. It was obvious that the personnel responsible for the design, testing, and qualification of the vehicle and its systems, and those personnel responsible for conduct of the flight were the most knowledgeable and, therefore, the most logical personnel to accomplish the evaluation. It was decided to utilize these personnel rather than a separate evaluation organization. The most important criteria in the selection of team personnel were that they be intimately familiar with their subject or system and that they be cognizant of mission events that affected that subject or system.

The reporting organization shown in figure 26-2 consists of a management staff including a team manager, a chief editor, a deputy chief editor, an editorial staff, and a data support group. In addition, a senior editor for each major section of the report and a managing editor for the launch and target vehicles sections are assigned from the organization primarily responsible for the subject reported. The team is program oriented, cutting across line and contractor organizations, operating independently of normal administrative lines of authority, and reporting directly to the Gemini Program Manager. While serving on the evaluation team, members are relieved of their regular duties to the maximum extent possible but are released as soon as their report section is approved. The sequence of reporting is shown in table 26-V.

TABLE 26-V.—*Gemini Mission Reports, Sequence of Reporting*

Report	Type	Distribution schedule
Launch summary	Teletype	Lift-off + 2 hours
Special TWX	Teletype	Each 24 hours and when significant event occurs
Mission summary	Teletype	End-of-mission + 6 hours
Interim mission	Teletype	End-of-mission + 5 days
Final mission	Printed	End-of-mission + 35 days
Supplementary mission	Printed	As defined by mission report

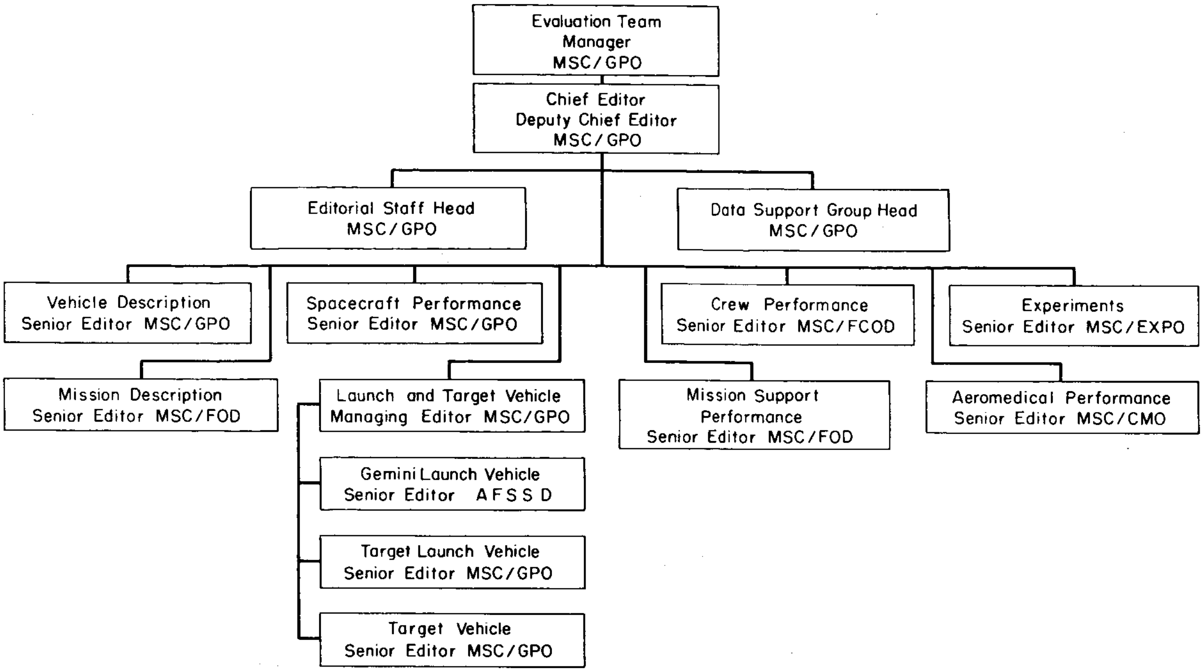


FIGURE 26-2.—Gemini Mission Evaluation Team organization.

Operations During the Mission

Team operations during the mission have been modified as requirements for change have become obvious with experience. Initially, team members had no evaluation-team function to perform during the mission. However, as the missions became more complex, a requirement for mission monitoring became evident. Team members had to follow the mission closely in order to optimize and expedite the evaluation. The experience gained on longer flights indicated a need for system specialists to act as consultants to the flight controllers. Again, the personnel who were most capable of providing this support were those who were instrumental in the design, test, or operation of the systems. A large number of these personnel had been working on the evaluation team, and the two functions were consolidated. During the mission, this flight monitoring and evaluation effort is continuously provided to the flight director. The consultant-team concept has proved to be very effective and has been used many times in support of the flights. Working around the unexpected drop in fuel-cell oxygen supply pressure on Gemini V and restoring the delayed-time telemetry recorder to operational status on the same flight are examples of this support.

Report Development During the Postmission Period

One of the most important evaluation functions for the team is to obtain the observations of the flight crew and to discuss performance characteristics with them. This must be accomplished quickly and effectively, and a high degree of organization is required. As soon as possible after the mission ends, the onboard flight log is microfilmed and sent to the Manned Spacecraft Center where it is reproduced and copies distributed to team members. Voice transcriptions of recorded onboard and air-to-ground conversations are expedited and disseminated. A schedule for debriefing of the flight crew is approved in advance of the mission and rigidly followed. Table 26–VI shows a typical schedule for debriefing the flight crew at the end of a mission.

Within a period of 2 weeks, each mission report author must accomplish the following tasks: examine all necessary data; define data reduction requirements; read technical debriefing; read air-ground and onboard voice transcripts; read crew flight log; attend systems debriefing; correlate findings with other team members; submit special test requests for failure analysis; and prepare report section. Evaluation cutoff dates are assigned and firmly adhered

TABLE 26-VI.—*Gemini Typical Postflight Crew Debriefing Schedule*

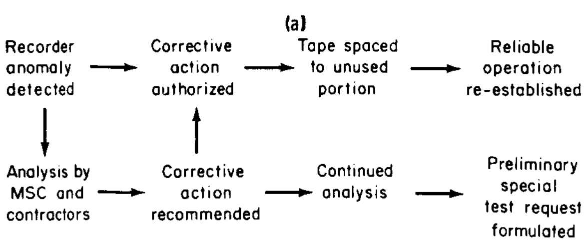
[Numbers are days after recovery]

Medical examinations	Immediately after recovery
Technical debriefing, medical examinations	1, 2, 3, and 4
Management and project debriefing	5
Technical debriefing, photograph identification	6
Prepare pilot's section of mission report	7
Systems debriefing	8
Scientific debriefing	9
Final debriefing	10

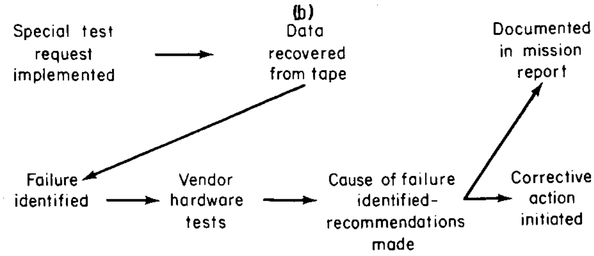
to in order to optimize manpower utilization. Problems not resolved within this allotted period are assigned to specific NASA or contractor organizations for resolution and documentation in supplementary reports.

A postflight inspection is conducted on the spacecraft after each mission. This inspection is expanded as a result of special test requests generated during the mission evaluation. A representative of the evaluation team is assigned to insure that the postflight inspection and testing of each spacecraft are coordinated with the mission evaluation effort. This representative submits daily reports by teletype to the mission evaluation team.

The evaluation required to formulate and implement corrective action is begun at the earliest possible moment. Figure 26-3 shows a typical reaction to an inflight failure which occurred in the following manner. Starting with the telemetry tape dump during revolution 30, poor quality data were received by the worldwide network stations. As a result of mission evaluation team consultation with the spacecraft contractor, the tape recorder vendor, and the flight controllers, a decision was made to record data for both revolutions 46 and 47 and then dump only the



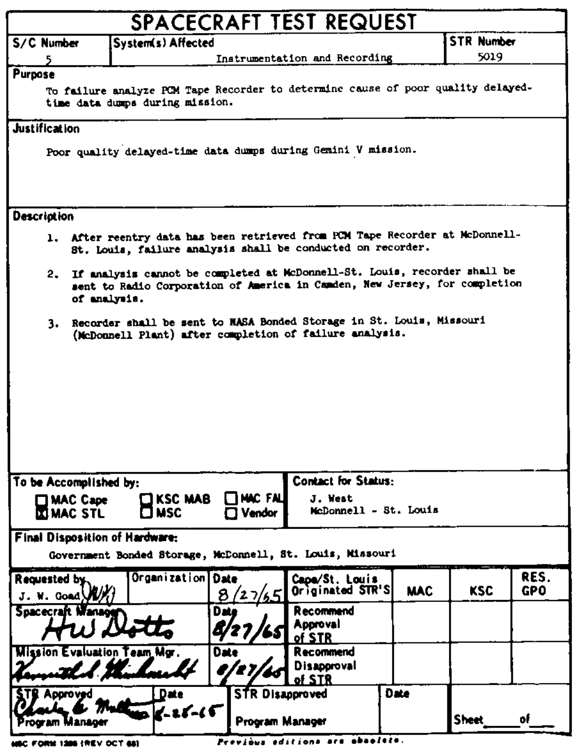
(*a*) Activities during mission.
FIGURE 26-3.—Gemini V PCM recorder anomaly check.



(*b*) Postflight activities.
FIGURE 26-3.—Concluded.

revolution 47 data. In this manner, operation of the recorder over a new portion of the magnetic tape was started, and good quality data were obtained for the remainder of the mission.

After recovery of the spacecraft, the Spacecraft Test Request, shown in figure 26-4, expedited removal of the recorder and its delivery to the contractor's plant. First priority was given to recovery of the last orbit and reentry data from the recorder before a failure analysis was begun. With a mission evaluation team member and personnel from the contractor, vendor, and resident quality assurance office in attendance, the recorder was opened, and the failure isolated to flaking of oxide from the tape. The recorder was then sent to the vendor's fa-



SPACECRAFT TEST REQUEST

S/C Number	System(s) Affected	STR Number
5	Instrumentation and Recording	5019

Purpose
To failure analyze PCM Tape Recorder to determine cause of poor quality delayed-time data dumps during mission.

Justification
Poor quality delayed-time data dumps during Gemini V mission.

Description
1. After reentry data has been retrieved from PCM Tape Recorder at McDonnell-St. Louis, failure analysis shall be conducted on recorder.
2. If analysis cannot be completed at McDonnell-St. Louis, recorder shall be sent to Radio Corporation of America in Camden, New Jersey, for completion of analysis.
3. Recorder shall be sent to NASA Bonded Storage in St. Louis, Missouri (McDonnell Plant) after completion of failure analysis.

To be Accomplished by: ☐ MAC Cape ☒ MAC STL ☐ KSC MAB ☐ MSC ☐ MAC FAL ☐ Vendor

Contact for Status: J. West, McDonnell - St. Louis

Final Disposition of Hardware: Government Bonded Storage, McDonnell, St. Louis, Missouri

Requested by J. W. Goad	Organization	Date 8/27/65	Cape/St. Louis Originated STR'S	MAC	KSC	RES. GPO
Spacecraft Manager		Date 8/27/65	Recommend Approval of STR			
Mission Evaluation Team Mgr.		Date 8/27/65	Recommend Disapproval of STR			
STR Approved, Program Manager		Date 8-28-65	STR Disapproved, Program Manager	Date	Sheet ___ of ___	

MSC FORM 1288 (REV OCT 63) *Previous editions are obsolete.*

FIGURE 26-4.—Spacecraft Test Request form.

cility for additional tests to determine the cause of the flaking. It was discovered that the flaking was caused by an epoxy having been inadvertently splashed on one of the rollers during the final record/playback head-alinement procedure. This epoxy had softened the binder used to adhere the iron oxide to the tape base, and the iron oxide had peeled away from the tape. The vendor duplicated the failure mode, and the results of the tests and the recommended corrective action were submitted in a failure analysis report to the spacecraft contractor. As a reply to the NASA Spacecraft Test Request, the contractor reported the findings and the corrective action to be taken.

Figure 26–5 is the actual schedule of work for the Gemini V mission evaluation and is typical for all missions. Despite the rapidity with which the report is completed, the formalized content and presentation format, implemented by a well coordinated and motivated team, has resulted in a series of mission evaluation reports which are thorough and timely.

The completion of the mission evaluation within a time frame compatible with the relatively short interval between missions is a notable accomplishment. A concentrated effort by the most knowledgeable specialists has been expended to reveal all anomalies, to find their cause, and to formulate corrective action in a timely manner. The evaluation is not considered complete, however, until all the facts and figures from each mission have been thoroughly documented for future reference.

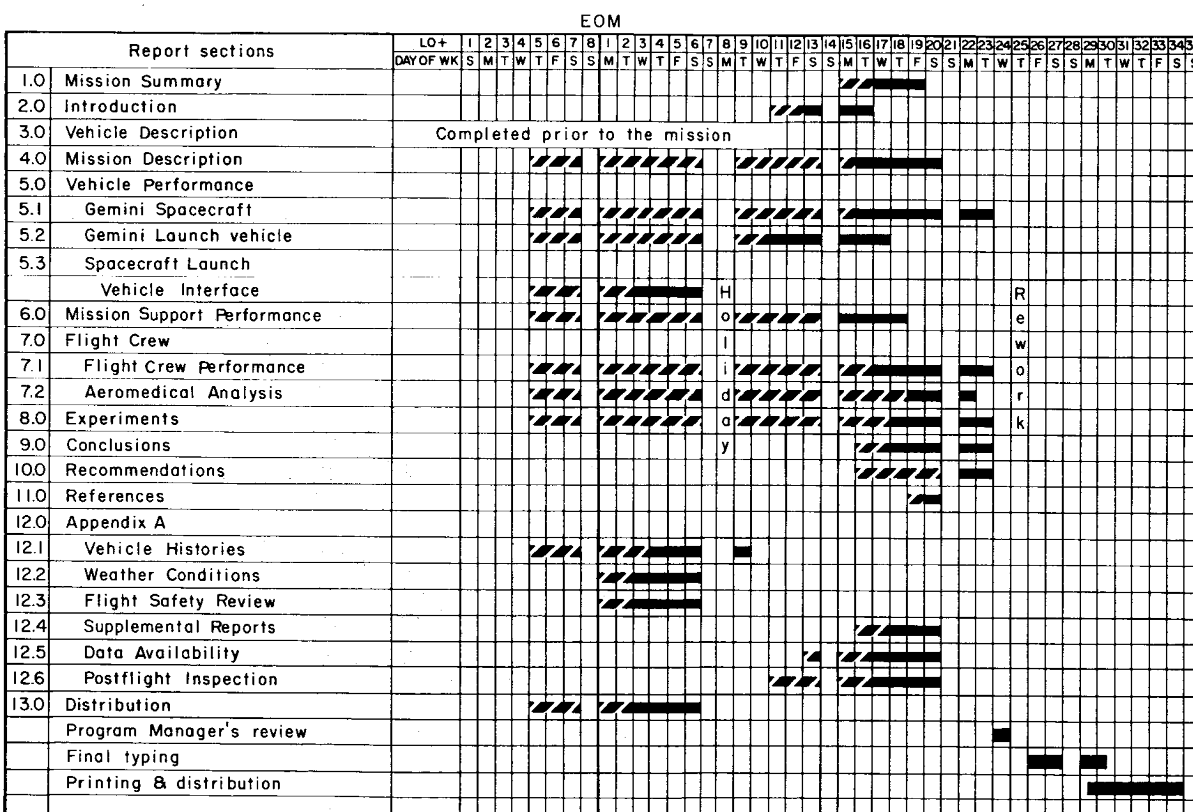


FIGURE 26–5.—Gemini mission reporting schedule.

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27. ASTRONAUTS' REACTIONS TO FLIGHT

By VIRGIL I. GRISSOM, *Astronaut, Astronaut Office, NASA Manned Spacecraft Center;* JAMES A. MCDIVITT, *Astronaut, Astronaut Office, NASA Manned Spacecraft Center;* L. GORDON COOPER, JR., *Astronaut, Astronaut Office, NASA Manned Spacecraft Center;* WALTER M. SCHIRRA, *Astronaut, Astronaut Office, NASA Manned Spacecraft Center; and* FRANK BORMAN, *Astronaut, Astronaut Office, NASA Manned Spacecraft Center*

Summary

The Gemini spacecraft was designed to make use of man's ability to function in the space environment. The extravehicular activity carried out during the Gemini IV flight demonstrated that an astronaut could maneuver and work outside his spacecraft. Man's capabilities in space were further demonstrated with the successful rendezvous between Gemini VI–A and VII.

Very few anomalies occurred during the first five manned Gemini flights, and most of the planned experiments were performed successfully. The flight crews have been well pleased with the Gemini spacecraft. Even though the cabin is small, the crews have been able to operate effectively and efficiently.

Introduction

The pilot's role in manned space flight has changed somewhat from the days of Project Mercury. Initially, man's reactions and his capabilities in a space environment were two of the big unknowns, but Project Mercury proved man to be both adaptable and capable. Therefore, the Gemini spacecraft was designed to use the pilot as the key system in its operation.

Preflight and Launch

When chosen for a specific mission, a flight crew is immediately faced with two tasks: training for the flight, and checkout of the spacecraft. The emphasis in these areas has changed from concentrating the major effort on spacecraft testing and checkout for the Gemini III mission to concentrating on training for the Gemini VI–A and VII missions. This was a natural evolution in that Gemini III was the first mission to use the new spacecraft for a manned flight, and the flight plan was designed to check out the spacecraft systems. The crews of the Gemini VI–A and VII spacecraft had high confidence in their vehicles through their association with previous missions, but they had difficult flights to accomplish since the emphasis was on operational mission requirements.

The schedule on launch day has greatly improved since the Mercury flights. For the Mercury flight, MR–4, the pilot was awakened at 1:10 a.m. and manned the spacecraft at 3:58 a.m. The Gemini launch is usually between the rather gentlemanly hours of 9 a.m. and 11 a.m. Also, the interval between crew awakening and insertion into the spacecraft has been shortened. However, it has not yet been possible to shorten the time between crew insertion and lift-off, although it is recognized that efficiency is increased by shortening the interval between the time that the crew awakes refreshed from a good night's sleep and the time of lift-off. This increased efficiency is especially helpful during the early, critical phase of the flight when the crewmembers are becoming adjusted to their new environment. After long periods in the spacecraft (90 minutes or more) the pilots become uncomfortable from lying on their backs in the Gemini ejection seat. The back, neck, and leg muscles tend to become cramped and fatigued.

The pilots concentrate during the last few days prior to a flight on the details of the flight plans, the status of the spacecraft, and both normal and emergency operational procedures. During this period, the backup crew and the flight-crew director endeavor to keep the crew from being disturbed by anything not connected with the operation of the mission.

Some experiments do place heavy burdens on the crew at this time, and an attempt should be made to avoid adding to the crew's workload during this period. A typical example of one of the heavy prelaunch activities was the preparation for the medical experiment M–7 by the Gemini VII flight crew. The preparation involved a rigid diet, complete collection of all body wastes, and two controlled distilled-water baths each day. The diet went well; the food was well prepared and tasty; however, the collection of body wastes was difficult to integrate with other activities, because the waste could only be collected at the places most frequented by the flight crew, such as the launch complex, the simulator, and the crew quarters. Fortunately, the fine cooperation of the M–7 experimenters resulted in a minimum number of problems.

Even though some of the flight crews, especially the Gemini V crew, had a comparatively limited time to prepare for their missions, they were well trained in all phases and were ready to fly on launch day.

During the prelaunch period, the backup crew is used extensively in the checkout of the spacecraft, and, at the same time, this crew must prepare to fly the mission. But their prime responsibility, by far, is spacecraft testing and monitoring.

Powered Flight

All flight crews have reported lift-off as being very smooth. The Gemini VI–A crew indicated that they could tell the exact moment of lift-off by the change in engine noise and vibration, and all crews agree that vertical motion is readily apparent within seconds of lift-off. Even without clouds as a reference, it is easy to determine when the launch-vehicle roll program starts and ends.

The noise level is quite low at lift-off, increasing in intensity until sonic speed is reached. At that time, it becomes very quiet and remains quiet throughout the remainder of powered flight.

With one exception, the launch has been free from any objectionable vibration. On the Gemini V flight, longitudinal oscillations, or POGO, were encountered. The crew indicated that the vibration level was severe enough to interfere with their ability to read the instrument panel. However, POGO lasted only a few seconds and occurred at a noncritical time.

The second stage of the launch vehicle ignites prior to separation from the first stage. This causes the flame pattern to be deflected and apparently to engulf the second stage and the spacecraft. The crew of Gemini VI–A indicated that the flame left a residue on the exterior of the window, and every crew has reported a thin film on the outside of the window. The pilot of Gemini VI–A noted that a string of cumulus clouds was very white and clear prior to staging and that the clouds were less white and clear afterward, indicating that the port window obscuration could have occurred during staging.

The horizon is in full view during second-stage flight while the radio guidance system is guiding the launch vehicle. Each correction that the guidance system initiates can be readily observed by the crew. It would appear that, given proper displays and an automatic velocity cutoff, the crew could control the launch vehicle into a satisfactory orbit.

Second-stage engine cutoff is a crisp event. The g-level suddenly drops from approximately 7 to zero, and in no case has any tail-off been felt by the crews.

The powered-flight phase has been closely duplicated on the dynamic crew procedures simulator trainer at the Manned Spacecraft Center. After the first flight, the vibration level and the sounds were changed to correspond with what the pilots actually heard during launch. The simulation has such fidelity that there should be no surprises for the crew during any portion of powered flight.

Orbit Insertion

The insertion into orbit has been nominal for every flight. The separation and turnaround of the spacecraft and the operation of the onboard computer have been as planned.

At spacecraft separation and during turnaround, there is quite a bit of debris floating all around the spacecraft. Some of these small pieces stay in the vicinity for several minutes.

During insertion, the aft-firing thrusters cannot be heard, but the acceleration can be felt. The firing of the attitude and translation thrusters can be heard, and the movement of the spacecraft is readily apparent.

System Operation

Inflight Maneuvering

The flight crews have found the pulse-control mode to be excellent for fine tracking, and the fuel consumption to be negligible. The direct mode was needed and was most effective when large, rapid attitude changes were required. However, the use of the direct and also the rate-command mode is avoided whenever possible because of the high rate of fuel consumption. Rate command is a very strong mode, and it is relatively easy to command at any desired rate up to full authority. It is the recommended mode for the critical tasks, such as retrofire and translation burns, that are beyond the capability of the platform mode.

The platform mode is a tight attitude-hold control mode. It has the capability of holding only two indicated attitudes on the ball display—zero degrees yaw and roll, and zero or 180 degrees pitch. But the platform mode can be caged and the spacecraft pointed in any direction and then the platform released. This gives an infinite number of attitudes. It is the recommended mode for platform alinement and for retrograde or posigrade translation burns. The horizon-scan mode is a pilot-relief mode and is used when a specific control or tracking task is not required. It is better than drifting flight because it controls the spacecraft through a wide dead band in pitch and roll, although it has no control of yaw. Drifting flight is perfectly acceptable for long periods of time, as long as the tumbling rates do not become excessive (5° per second or more). Spacecraft control with the reentry control system is very similar to that of the orbital attitude and maneuver system. Slightly more authority is available with the orbital attitude and maneuver system than with both rings of the reentry control system. This results in some tendency to overcontrol and waste fuel. Actually the one-ring reentry control system operation is satisfactory for most tasks. All pilots used both rings for retrofire, but some used only one ring for reentry. The reentry rate-command mode has not been used by any crew except that of Gemini IV. The automatic reentry mode also has not been employed.

Two orbital maneuvers during the flight of Gemini VII were accomplished in a spacecraft powered-down configuration. This means they were without the platform, the computer, and the rate needles. The yaw attitude was established by using a star reference obtained from ground updates and the celestial chart. Roll-and-pitch attitudes were maintained with respect to the horizon, which was visible to the night-adjusted eye. The pilot made the burns, maintaining attitude on the star with attitude control and rate command, while the command pilot timed the burn. No unusual difficulty was encountered when performing the no-platform maneuvers, and the crew considered this procedure acceptable.

For this long-duration flight, it was found desirable to adhere to the same work-rest cycle that the crew was used to on the ground. To support this schedule, both crewmembers slept simultaneously, except during the first night. The ground was instructed not to communicate except for an emergency.

The Gemini IV mission was a good test of the life-support systems for extravehicular activity. Preparations for extravehicular activity started during the first revolution and continued into the second. Extravehicular activity demonstrated that man can work in a pressurized suit outside the spacecraft and can use a maneuvering unit to move from one point to another. The maneuvering unit used short bursts of pulse mode. During extravehicular activity, the pilot used the spacecraft as a visual, three-dimension orientation reference. At no time did the pilot experience disorientation. The pilot made general observations and investigated tether dynamics. Control with the tether was marginal, but it was easy to return to the hatch area using the tether. When the pilot pushed away, the spacecraft pitched down at rates of 2° per second from the resultant force, and the pilot moved perpendicular to the surface of the spacecraft. It was difficult to push away from the surface of the spacecraft at an angle. After the pilot had reentered the spacecraft, the hatch was to be closed, but the latch handle malfunctioned. However, the pilot had been trained thoroughly in both the normal and failure modes of the hatch and was able to close it successfully.

Life-Support Systems

The bite-size foods for the crews were not as appetizing as had been expected. The rehydratable foods were good and were preferred to the bite-size foods. Preparing and consuming the meal takes time and must be done with care. The food is vacuum-packed to eliminate any waste volume, but this capability does not exist when the crew is trying to restow the empty food bags. Thus, they have a restowage problem. Most of the food is in a semiliquid form, and any that remains in the food bags is a potential source of free moisture in the cabin. The water has been good and cold. Even so, there seems to be a tendency to forget to drink regularly and in sufficient quantities.

On the first long-duration mission, the crewmen had a difficult time sleeping when scheduled. The spacecraft is so quiet that any activity disturbed the sleeping crewman. For the later missions, the crewmembers slept simultaneously, when it was possible.

Defecation is performed carefully and slowly; the whole procedure is difficult and time consuming, but possible. A major problem for long-duration flights was the storage of waste material. It was normally stowed in the aluminum container which held the food. It was necessary that a thorough housekeeping and stowage job be done every day. Otherwise, the spacecraft would have become so cluttered that it would have been difficult for the crewmen to find anything.

The Gemini VII crewmen wore the G5C space suit, which is 8 to 10 pounds lighter than the normal suit. This suit contains no bumper material and has only two layers of nylon and rubber. The G5C space suit includes a zipper-type hood, which is designed to be worn over an ordinary pilot helmet.

For the Gemini VII mission, fully suited operations were conducted during launch, rendezvous, and reentry. When the hoods were on, there was considerable noise in the intercom system because of the airflow in the hood. Visibility while wearing the hood was acceptable during orbital flight, but during reentry vision was somewhat obscured and the command pilot removed his hood. When fully suited, the crew found it difficult to see the night horizon and to observe and operate switches in the overhead and water-management panels. In the partially suited configuration, which was maintained for approximately 2 days, there was a loss in suit cooling efficiency, and some body areas did not receive sufficient cooling. Intercommunication was improved with the hoods off, but mobility was restricted because of the hood being on the back of the head. On the second day, the pilot removed his suit, and his comfort was definitely improved. Ventilation was adequate, and the skin was kept dry. In the suit-off configuration, there was increased mobility. It was easier to exercise, unstow equipment, and perform other operations. It took approximately 20 minutes to remove the suit, including the time required to place the plugs in the suit openings in case emergency donning was required. During the sixth day of the mission, both pilots had their suits off. One apparent improvement was that all crews on the long-duration flights felt a need to exercise. Even though exercise periods were scheduled regularly, most crews requested more frequent and longer periods of exercise.

System Management

One of the crew's prime functions is to monitor and control the spacecraft's various systems. This requires a thorough knowledge of the details of each system, as well as how to operate the system in any failure mode. It is true that the ground complex has much more information concerning the operation of systems than the crew does, and they have a staff of experts for each system. But, unfortunately, the crew is in contact with the ground stations for only a small percentage of the flight. The crew must be prepared to rapidly analyze problems and make the correct decisions in order to complete the mission safely. Every flight has had an example of this. Gemini III had the dc-dc converter failure and suspected fuel leak; Gemini IV experienced a computer memory alteration; and Gemini V experienced fuel-cell oxygen-supply degradation while performing the rendezvous evaluation pod experiment. Gemini VI–A probably had the most difficult problem of all. The shutdown on the pad occurred in a manner that it had not considered during training. Gemini VII had flight control and fuel-cell problems. These are the times that it pays to have a well-trained crew onboard.

Visual Sightings

The Gemini III crew were surprised at the flame that appeared around the spacecraft during staging. During the remainder of the flight, the Gemini III crew observed thruster firings, Northern and Southern Hemisphere constellations, and the town of Mexicali, Mexico.

The Gemini IV crew were impressed at the clarity with which objects could be seen from directly overhead. Roads, canals, oil tanks, boat wakes, and airfields could be seen. The moon was a bright light; however, the stars close to it as well as the stars of the seventh magnitude could be seen. When the spacecraft passed from darkness into light, the airglow was clearly observed, and the planets seemed to increase in brightness. Meteors could be seen as they burned in the earth's atmosphere below the orbital flight path.

The Gemini VI–A crew made some very accurate visual sightings which have been reported in the presentation of the rendezvous.

The Gemini VII crew tracked their launch vehicle during the station-keeping exercise by using the acquisition lights on the launch vehicle, but they could not estimate the range. The spacecraft docking lights were turned on, but they did not illuminate the launch vehicle. As the time approached for rendezvous, spacecraft 6, at a range of approximately 2 to 3 miles, appeared to the Gemini VII flight crew like a point of reflected light against the dark earth background just before sunset. At approximately 0.5-mile range, thruster firings could be seen as thin streams of light shooting out from the spacecraft.

All crews reported that accurately tracking an object on the ground is an easy task. The difficult part is identifying and acquiring the target initially. It requires that the ground transmit accurate acquisition times and pointing angles. Also, a careful preflight study of maps and aerial photographs aids in early identification.

Experiments

Experiments and their results are covered in other papers. But, the point should be made here that, for the crew to successfully complete any experiment, they must have a thorough understanding of what the experimenter is attempting to do. And, even more important, they must have equipment available at an early date to use in their training. One of the biggest problems is getting the actual flight equipment to work well in its environment. A ground rule has been established that all flight gear, experimental and operational, must be available and in the spacecraft for the altitude chamber test.

Retrofire and Reentry

During the Gemini III mission, a reentry control system plume-observation test was conducted. Because the reentry control system yaw thrusters obstruct the view of the horizon at night, a nightside retrofire would be impossible when using the horizon or stars as a reference. When the retroadapter was jettisoned, there was an audible noise. Jettisoning could be felt, and there was debris around the spacecraft. During reentry the spacecraft was stable, and there were no difficulties in damping out the oscillations.

During the Gemini IV reentry, the rate-command system provided excellent control, and the attitudes were held within ±1 degree. The reentry rate command with the roll gyro turned off was used so that the hand controller did not have to be held deflected in roll for the entire reentry. The spacecraft rolled about its longitudinal axis at the beginning of reentry, and, after aerodynamics started to take effect, the spacecraft rolled about its trim axis and reentered in a wide spiral.

The Gemini V crew performed retrofire during the middle of the night, using the attitude ball as a reference. At retrofire, the outside appeared to be a fireball. The command pilot reported that it felt as though the spacecraft were going back west, and the pilot reported that he felt that he was going into an inside loop.

The Gemini VI–A crew also performed their retrofire at night and did not see the horizon until just before the 400 000-foot-altitude point because of losing their night visual adaptation.

The Gemini VII crew had communications problems during retrofire, since the vented air noise in the helmets hindered good communications. During reentry, the command pilot had to remove his hood because it interfered with his vision of the horizon.

Landing and Reentry

The drogue parachute is normally deployed at 50 000 feet to stabilize the spacecraft prior to main parachute deployment. After deployment, the spacecraft appears to oscillate about 20° to 30° on each side. The onboard recordings indicated that these oscillations have never exceeded ±10°.

Main-parachute deployments take place in full view of the crew, and it is quite a beautiful and reassuring sight. Up to this point, all events have been quite smooth, with all loads being cushioned through line stretching and reefing. But, changing from the single-point attitude to the landing attitude causes quite a whip to the crew. After the Gemini III flight, all crews have been prepared, and there have been no problems.

The impact of landing has varied from a very soft impact to a heavy shock. The amount of spacecraft swing, and at what point during the swing the landing occurs, changes the landing loads. The amount of wind drift, the size of the waves, and the part of the wave contacted also vary the load. Even the hardest of the landings has not affected crew performance.

Concluding Remarks

In conclusion, the flight crews have been well pleased with the Gemini spacecraft. Even though the cabin volume is very limited, they have been able to operate effectively and efficiently.

28. GEMINI VI–A RENDEZVOUS MISSION PLANNING

By EDGAR C. LINEBERRY, *Mission Planning Analysis Division, NASA Manned Spacecraft Center*

Summary

This paper discusses the mission planning effort for the Gemini VI–A mission which applied directly to rendezvous. Included are a discussion of the basic design criteria and a brief history of the considerations which led to the selection of the particular Gemini VI–A mission plan. A comparison between the nominal and actual flight trajectories is also presented.

Introduction

The basic Gemini VI–A mission design criteria were, in effect, quite simple. Consideration was given almost exclusively to the development of a plan which would provide the highest probability of mission success. The desire was to develop a plan which could routinely depart from the nominal in response both to trajectory dispersions and to spacecraft systems degradation, while minimizing dispersed conditions going into the terminal phase of rendezvous. More specifically, the plan would provide flexibility without introducing undue complexity; that is, the flight controllers would have the capability, in the event of dispersed conditions, to select alternate maneuver sequences that would not be dissimilar to the basic maneuver sequence.

Selection of the Basic Mission Plan

Prior to the selection of the Gemini VI–A mission plan, three significantly different plans (fig. 28–1) were analyzed to the extent necessary to permit a realistic choice consistent with the desired flexibility criteria. The first of these was the tangential mission plan. The salient feature of this plan was a final tangential approach to the target vehicle, preceded by several orbits during which midcourse maneuvers would be commanded from the ground. The last maneuver in the ground-controlled sequence would be designed to place the spacecraft on an intercept trajectory. The onboard system would be utilized to correct this final trajectory to effect rendezvous. The second plan investigated the coelliptic plan, utilized the same midcourse-maneuver sequence as the tangential plan, except that the final maneuver in the ground-controlled sequence would be designed to place the spacecraft in an orbit with a constant differential altitude below the target orbit. The onboard system in this plan would be utilized to establish an intercept trajectory departing from the coelliptic orbit. The third plan which was investigated incorporated a rendezvous at the first spacecraft apogee. In effect, a nominal insertion would place the spacecraft on

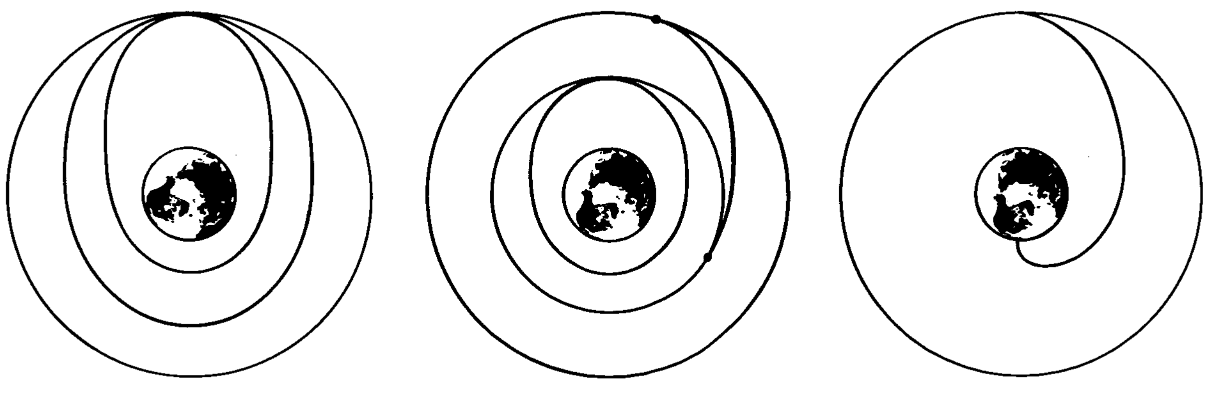

FIGURE 28–1.—Rendezvous mission plan development.

an intercept trajectory, and the onboard system would be utilized to correct for dispersed conditions, thereby placing the spacecraft on a final intercept trajectory.

As can be seen, two of these three plans incorporated a parking-orbit mode of operation prior to the establishment of a final intercept trajectory, whereas the third plan incorporated a direct intercept mode. Based upon various analyses conducted for the plans, a recommendation was made to adopt the coelliptical mission plan. Two major considerations, as well as a number of lesser ones, influenced this recommendation.

First of all, the mission plan for rendezvous at first apogee was eliminated as a contender, as compared with the other plans, for the Gemini VI–A mission because of its increased spacecraft propellant requirements for reasonable trajectory dispersions. Secondly, the terminal-phase initiation conditions of the coelliptical plan afforded a certain advantage over the tangential plan. Without going into detail, the basic desired feature of the coelliptical plan is that the relative terminal-phase trajectory of the spacecraft with respect to the target is not particularly affected by reasonable dispersions in the midcourse maneuvers. On the other hand, it is grossly affected when initiating from the tangential approach. More simply stated, the coelliptical approach affords a standardized terminal-phase trajectory, yielding obvious benefits in the establishment of flight-crew procedures and training. Another benefit derived from this plan is that the rendezvous location can be controlled to provide the desired lighting conditions. As a consequence of these advantages, the coelliptical mission plan was selected.

Terminal-Phase Considerations

The above discussion leads naturally to a consideration of the terminal phase, because it was this portion of the mission plan which governed the plan selection. These considerations also dictate the targeting conditions of the preterminal-phase midcourse activity controlled by the ground. The most basic consideration was to provide a standardized terminal-phase trajectory which was optimized for the backup procedures—that is, those procedures developed for use in the event of critical systems failure. It was possible to optimize the trajectory for the backup procedures with no degradation of the primary inertial-guidance-system closed-loop rendezvous-guidance technique.

Since it is possible to select any particular transfer trajectory to serve as a standard, extensive analyses were performed to provide a transfer trajectory with certain desired characteristics. It was desired, first of all, that the transfer initiation maneuver for a nominal coelliptical trajectory be alined along the line of sight to the target. This procedure has the obvious advantage of providing the crew with an excellent attitude reference for this critical maneuver, should it be needed. The second characteristic desired in the transfer trajectory was a compatibility between the closed-loop guidance mode and the final steering and braking performed manually by the flight crew. Based upon the transfer initiation criteria, the desired feature in the resultant trajectory would be a situation in which the nominal trajectory would create low inertial line-of-sight rates during the time period prior to and including braking. Such a trajectory would be consistent with the steering technique utilized by the flight crew to null the line-of-sight rate to zero. The analyses resulted in a choice of 130° orbital travel of the target vehicle between the terminal-phase initiation and braking. As can be seen in figure 28–2, the 130° transfer trajectory not only satisfies the second desired characteristic, but also fulfills a third desired condition, in that the approach of the spacecraft, relative to the target, is from below, thus assuring a star background which could be utilized as an inertial reference.

After the selection of the transfer trajectory, the differential altitude between the two orbits was the next decision point. Analyses were

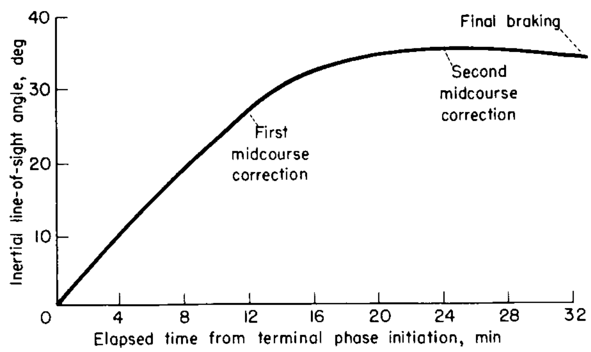

FIGURE 28–2.—Gemini 130° transfer trajectory.

carried out and resulted in a decision to utilize a 15-nautical-mile differential altitude between the orbits of the two vehicles. This choice resulted from a trade-off between a desire to be close enough to insure visual acquisition of the target prior to terminal-phase initiation, and a desire to minimize the influence of dispersions in the previous midcourse maneuvers on the desired location of terminal-phase initiation. Figure 28–3 shows that the effect of dispersions on the terminal-phase initiation time increases as the differential altitude is decreased. For the selected differential altitude of 15 nautical miles, the 3-sigma dispersion of the timing of the terminal-phase initiation maneuver is on the order of ±8 minutes. Factors governing the choice of the desired lighting condition for terminal-phase initiation cannot be considered here; however, the decision was made for the nominal initiation time to be 1 minute into spacecraft darkness. This condition and the selected differential altitude of 15 nautical miles established the targeting conditions for the ground-controlled maneuvers at the time of the coelliptical maneuver.

Ground-Control Midcourse-Phase Considerations

As previously noted, the intention was to provide a plan as insensitive to dispersions and spacecraft systems degradation as possible. This led to the provision of three spacecraft

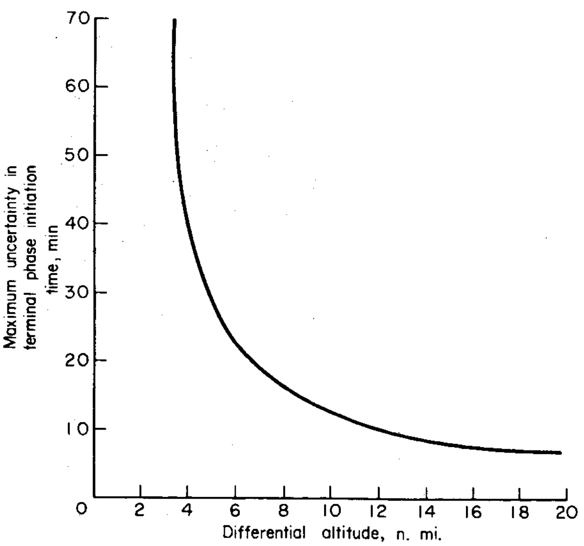

FIGURE 28–3.—Terminal phase maneuver time dispersion analysis.

revolutions in the nominal plan, with preestablished maneuver points to compensate for any of the dispersions likely to occur either in target altitude and ellipticity or in spacecraft insertion. Emphasis was given to minimizing the demands of this phase of the mission on the spacecraft propulsion system. Because the propulsion requirements for the terminal rendezvous phase could increase significantly from degraded systems performance, it was imperative that the maximum amount of spacecraft propulsion capability exist at the time those activities were initiated. These decisions were reflected in the following mission plan characteristics:

(1) Maneuvers were carried out with the Gemini VII spacecraft to provide the best possible launch opportunities and optimum orbital conditions for rendezvous.

(2) The Gemini launch vehicle was targeted to provide a differential altitude of 15 nautical miles between the two orbits at first spacecraft apogee. The launch vehicle was targeted also to launch the spacecraft into the target plane; that is, launch-vehicle guidance was utilized to fly a dog-leg launch trajectory in order to minimize spacecraft propulsion requirements in orbit for making a plane change.

(3) During the first orbit the flight crew were left free of rendezvous activity. This period of time was used for spacecraft systems checks. It was also used by the Mission Control Center—Houston to determine the precise spacecraft 6 orbit.

(4) Ground tracking, computation and display, and command capability were provided to carry out the ground-controlled midcourse maneuvers.

Since it was necessary to plan for nonnominal situations such as delayed lift-off, a real-time mission planning capability was implemented in the Mission Control Center. This capability consisted of various computer-driven displays which would permit the flight controllers to assess any particular situation and select a maneuver sequence which was compatible with the mission constraints.

Comparison Between Prelaunch Nominal and Actual Gemini VI–A Mission Trajectories

Prior to launch of the Gemini VI–A spacecraft, the maneuver plan selected consisted of two nonzero maneuvers: (1) A phase-adjustment maneuver to be performed at the second spacecraft apogee to raise the perigee to approximately 117 nautical miles; and (2) the coelliptical maneuver to be made at the third spacecraft apogee. However, in order to account for insertion dispersions, two additional maneuver points were established: (1) a height-adjustment maneuver to be made at first spacecraft perigee following first apogee; and (2) a plane-change maneuver to be performed at a common node following the phase-adjustment maneuver. Since the launch vehicle was targeted to achieve the correct differential altitude and plane location, these two maneuvers were nominally zero.

Ground network tracking during the first orbit revealed an underspeed condition at insertion, as well as a small out-of-plane condition. This can be seen in figure 28–4. Whereas the targeted condition for first apogee was a differential altitude of 15 nautical miles, the actual value which resulted was approximately 23 nautical miles. Consequently, the height-adjustment maneuver at first perigee (fig. 28–5) was 14 feet per second. The additional refinement of the spacecraft orbit following the height-adjustment maneuver indicated that a second height adjustment would be required, and the maneuver sequence was altered to include this adjustment at the second spacecraft perigee. The phase-adjustment maneuver to be performed at second spacecraft apogee was adjusted accordingly (fig. 28–6). Because of the underspeed condition at insertion, the Gemini VI–A spacecraft was actually catching up too fast; therefore, during the phase-adjustment maneuver at second apogee, the prelaunch nominal value of 53 feet per second was changed to 61 feet per second. This maneuver adjusted the catchup rate to establish the correct phasing condition at the time of the coelliptical maneuver.

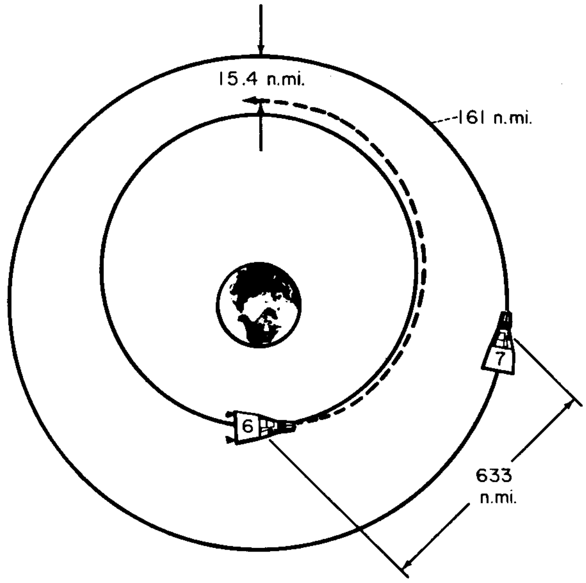

FIGURE 28–5.—Gemini VI–A first adjustment.

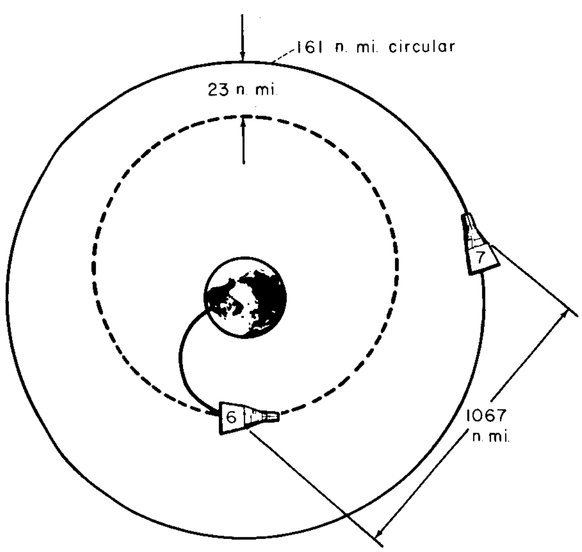

FIGURE 28–4.—Gemini VI–A insertion.

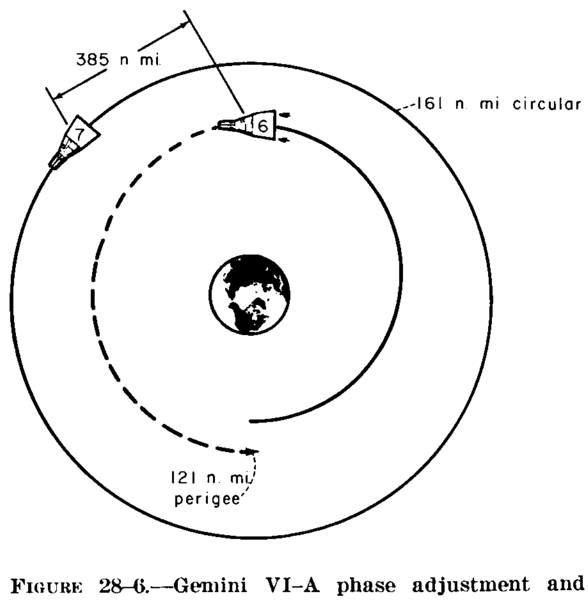

FIGURE 28–6.—Gemini VI–A phase adjustment and plane change maneuvers (common node) at second apogee.

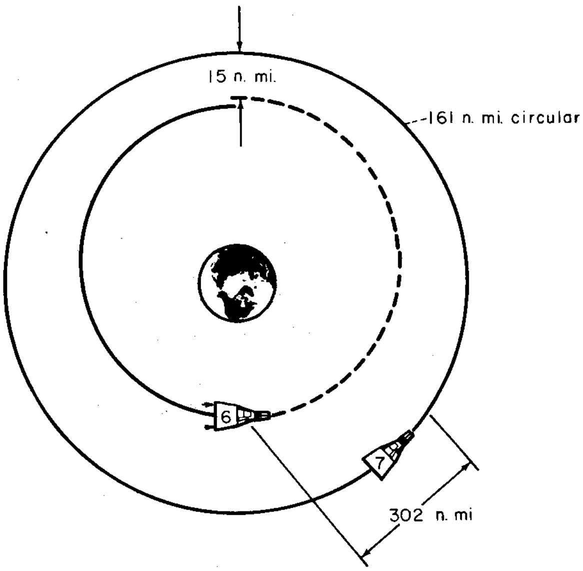

FIGURE 28–7.—Gemini VI–A second height adjustment maneuver at second perigee.

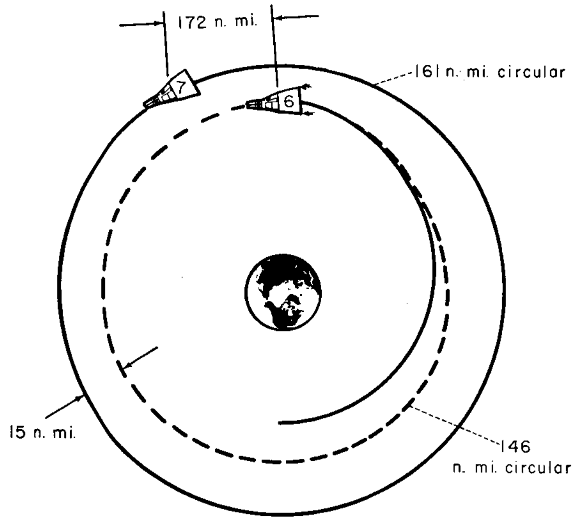

FIGURE 28–8.—Gemini VI–A coelliptical maneuver at third apogee.

Following the phase-adjustment maneuver, a plane change of 34 feet per second was performed to place the Gemini VI–A spacecraft in the plane of the Gemini VII spacecraft. At the next spacecraft perigee, the second height-adjustment maneuver of 0.8 foot per second was performed to correctly adjust the differential altitude to 15 nautical miles (fig. 28–7). At the third spacecraft apogee, a coelliptical maneuver of 43 feet per second was performed (fig. 28–8). Following this maneuver, radar tracking indicated a downrange-position error of approximately 2 miles at the time of the coelliptical maneuver, so that the actual downrange displacement was approximately 172 nautical miles, compared with the desired value of 170 nautical miles. The result, as determined on the ground, was a predicted slip of approximately 2 minutes in the terminal-phase-initiation maneuver. This information, as well as a ground-computed terminal-phase-initiation maneuver, was passed to the flight crew to serve as a comparative value with onboard computations.

Concluding Remarks

The discussion dealing primarily with the terminal-phase portion of the mission will be discussed in the following paper. The Gemini VI–A mission-planning effort resulted in the successful rendezvous with the Gemini VII spacecraft.

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29. RENDEZVOUS OF GEMINI VII AND GEMINI VI–A

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Summary

A description of the rendezvous techniques, procedures, and flight data charts developed for the Gemini VI–A mission is presented in this paper. The flight data charts and crew timeline activities were developed over an 8-month period.

Successful rendezvous is critically dependent on the presentation to the flight crew of sufficient information developed onboard the spacecraft. The Gemini VI–A flight crew used this information to evaluate the rendezvous progress by several different methods and made critical decisions based on their evaluation. The system combination found most effective in making these evaluations was the range-rate data from the radar, and the angle data from the platform.

Introduction

The Gemini spacecraft was designed to use four subsystems in determining the rendezvous maneuver and presenting information to the crew. These subsystems are the radar, computer, platform, and cockpit displays. In all cases, the crew has independent operational control over each system and performs the function of selecting how these systems will be integrated.

The Gemini VI–A rendezvous flight plan was based on the use of flight data displayed to the crew in a manner to allow monitoring and backup for each spacecraft maneuver. The philosophy of maximum manual backup capability begins with the mission profile in which a coelliptical spacecraft-catchup orbit is employed prior to initiation of rendezvous. This permits use of a standard transfer change in velocity (ΔV) in both magnitude and direction, with the time of initiation determined by the elevation angle of the target line of sight above the local horizontal. Thus, the transfer maneuver varies only because of dispersions in the catchup orbit, and these are corrected by angle and range measurements.

The discussions that follow apply to that time period from the start of circularization thrusting to a point where the Gemini VI–A spacecraft was within 100 feet of the Gemini VII spacecraft, and had no attitude rates and less than 0.5-foot-per-second relative velocity in all translational axes (station keeping). Although the closed-loop guidance technique is considered the primary method to accomplish rendezvous, backup guidance techniques were developed to assure rendezvous in the event of equipment failures. Accordingly, the procedures are presented for both the closed-loop guidance technique and the backup guidance techniques required in the event of radar, computer, or platform failure. In addition, flight data charts were developed specifically for the Gemini VI–A mission. These charts provide a means for determining the proper transfer maneuver and midcourse corrections, for monitoring the performance of closed-loop guidance, and for the calculation of the required backup maneuvers in the event of equipment malfunctions or failures.

Optical tracking of the target is a mandatory requirement in case a radar or platform failure is encountered. Thus the day-night cycle becomes an increasingly important parameter for the rendezvous mission. Lighting conditions for the terminal-phase maneuver were investigated after the coelliptical mission plan, involving a 130° transfer trajectory, was developed. At an altitude of 161 nautical miles, the target is in daylight for 55 minutes and in darkness for 36 minutes. The lighting conditions, displayed in figure 29–1, are planned so that the crew can track the target by reflected sunlight just prior to transfer to obtain data for the transfer maneuver. During the transfer maneuver and all subsequent maneuvers, the crew tracks the target's artificial lighting with respect to the stars for inertial angular measurement or uses platform angles when the optical sight is boresighted on the target. The braking maneuver occurs just as the target becomes lighted at sunrise. Thus it can be seen that the rendezvous initiation is normally planned to occur at 1 minute after sunset and the braking maneuver to occur at a range of 3000 feet when the target is starting to be illuminated by sunlight.

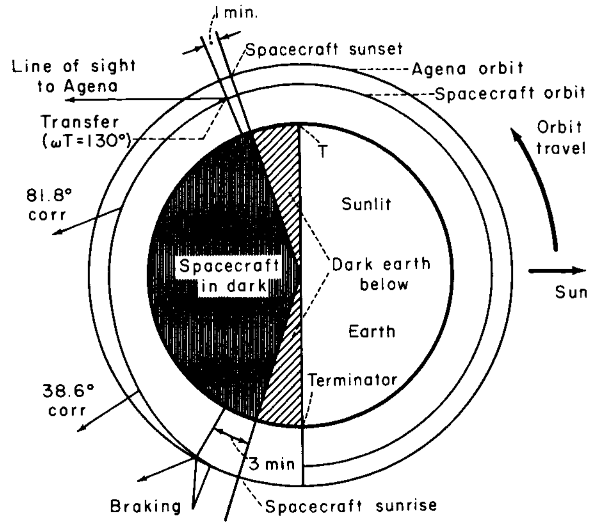


FIGURE 29-1.—Terminal-phase lighting conditions.

Closed-Loop Rendezvous Procedures

Closed-loop rendezvous procedures are presented in the left column of figure 29-2; they are listed in the exact order that the crew performs them. Cockpit responsibility is assigned by the

(a)

NOMINAL	RADAR FAILURE	COMPUTER FAILURE	PLATFORM FAILURE
INITIATION CUE - ANGLE/MDU OUTPUT	INITIATION CUE - ANGLE/MDU INPUT COMPUTER - CATCH-UP AT FAILURE ZERO ADD 25, 26, 27	INITIATION CUE - "8" BALL	INITIATION CUE - RANGE (MDU) OUTPUT COMPUTER - CATCH-UP AT FAILURE
0:00 APPLY CIRCULARIZATION TRANS (C) START GET (P) AT 0,0,0 ATT, APPLY THRUST TO ZERO READOUTS FROM ADD 80,81,82 GO TO RDR ACQ ATT, ACQUIRE LOCK-ON FDR - RDR FDM - ATT	0:00 APPLY CIRCULARIZATION TRANS (C) START GET (P) AT 0,0,0 ATT, APPLY THRUST TO ZERO ADD 80, 81, 82 (C) ZERO ADD 25, 26, 27 (P) COMPUTER - RNDZ/CTCH-UP (P) FDR - COMP FDM - ATT	0:00 APPLY CIRCULARIZATION TRANS (C) START GET (P) GO TO RDR ACQ ATT, ACQUIRE LOCK-ON FDR - RDR FDM - ATT/RATE	0:00 APPLY CIRCULARIZATION TRANS (C) START GET (P) GO TO RDR ACQ ATT, ACQUIRE LOCK-ON FDR - RDR FDM - ATT/RATE
ATT CNTL - PULSE MAN CONT - OFF	ATT CNTL - PULSE MAN CONT - OFF	ATT CNTL - PULSE MAN CONT - OFF	ATT CNTL - PULSE MAN CONT - OFF
SET E.T. TO 4:00 BORESIGHT ON AGENA BY NULLING FDI'S (C)	SET E.T. TO 4:00 (C)	SET E.T. TO 4:00 BORESIGHT ON AGENA BY NULLING FDI'S (C)	SET E.T. TO 4:00 BORESIGHT ON AGENA BY NULLING FDI'S (C)
4:00 ON MARK (P) START E.T. UP (C) COMPUTER - RNDZ (P)	4:00 ON MARK (P) START E.T. UP (C)	4:00 ON MARK (P) START E.T. UP (C)	4:00 ON MARK (P) START E.T. UP (C)
NOTE READ Θ (59) AND R (69) (EACH 100 SEC PT) (P) INPUT WT = 83:13000; ΔWT = 93:04820 (P) VERIFY ADD 83, 93, 54, 53, 24, 92 IF REQ (P)	NOTE CONTROL Θ TO NOMINAL UNTIL AGENA IS VISIBLE, THEN CONTROL S/C TO KEEP AGENA AT CENTER OF RETICLE.	NOTE WHEN VISIBLE, CONTROL S/C TO KEEP AGENA AT TOP OF RETICLE. MONITOR RANGE ON R - Ṙ METER (C) MONITOR "8" BALL (P)	NOTE WHEN VISIBLE, CONTROL S/C TO KEEP AGENA AT TOP OF RETICLE. READ R (69) (EACH 100 SEC PT) MONITOR RANGE ON R - Ṙ METER (C) WHEN R = 43.00 N.M. READ EVERY 10 SEC
NOTE Θ WHICH EXCEEDS 20.1° AND CIRCLE IT. IF CIRCLED Θ IS NEARER 20.1° THAN 21.4°, LABEL IT PT A. IF NOT, LABEL IT PT B AND THE PREVIOUS ONE PT A. ADD 3:20 TO PT A TIME TO OBTAIN TIME OF PT C. CALCULATE GET RESET TIME BY ADDING 4:30 TO PT C TIME (P).	MONITOR Θ (59) EVERY 100 SEC WHEN Θ = 19°, READ Θ (59) EVERY 10 SEC RECORD TIME WHEN Θ = 20.1° (59) (LABEL POINT A)	WHEN ATT BALL READS 15.5° (P) SELECT STAR PATTERN ON MARK (P) HOLD STARS FIXED IN RETICLE (C) START WATCH (P) ON MARK (P) READ Δα OVER 01:40 (C)	WHEN R ≤ 41.00 N.M. SELECT STAR PATTERN ON MARK (P) HOLD STARS FIXED IN RETICLE (C) START WATCH AND READ R (69) (P) ON MARK FROM (P) READ Δα OVER 01:40 (C) READ R (69) (P)
3:20 AFTER PT A (PT C), READ Θ (59) AND R (69) (P) AFTER PT C, START COMP - PUSH CALCULATE ΔṘ & ΔΘ̇ CORR (P)	ADD 3:20 AND RECORD Θ (59) AT THIS TIME (LABEL POINT C) (P)	CALCULATE UP/DOWN ΔV CORR (P) FWD ΔV NOMINAL	CALCULATE UP/DOWN AND FWD ΔV (P)
	INPUT: 25:00284; 26:90147; 27:00000 (P) NULL FDI'S (COMP) (ATT) START COMP - PUSH (C) CALCULATE UP/DOWN ΔV CORR (P) FWD ΔV NOMINAL SET UP/DOWN IVI BY MAN KNOBS (C)	BORESIGHT ON AGENA (C)	MONITOR R (69) EVERY 10 SEC (P) BORESIGHT ON AGENA (C)
ATT CNTL - RATE CMD MAN CONT - ON (C)	ATT CNTL - RATE CMD MAN CONT - ON (C)	ATT CNTL - RATE CMD MAN CONT - ON (C)	ATT CNTL - RATE CMD MAN CONT - ON (C)
COMP LITE ON NOMINALLY 03:50 AFTER PT C, THEN START THRUST TO ZERO IVI (C) (S/C BORESIGHT ON AGENA)	WHEN AGENA IN CENTER OF RETICLE (Θ = 27.4°) START THRUST TO ZERO IVI (C)	WHEN BALL READS 27.5° (P) START THRUST (C)	WHEN R = 32.96 (P) START THRUST (C)

(*a*) Determination of terminal phase initiation.
FIGURE 29-2.—Closed-loop and backup rendezvous procedures.

(b)

	NOMINAL	RADAR FAILURE	COMPUTER FAILURE	PLATFORM FAILURE
ON	0:00 RESTART GET AT CALC TIME (P) AFTER THRUST MAN CONT - OFF ATT CNTL - PULSE (C) SET E.T. TO 02:00 & STBY	0:00 AT END OF THRUST, GET = 0 AND START (P) MAN CONT - OFF FDM - RATE CNTL AGENA TO CENTER OF RETICLE ATT CNTL - PULSE (C) ZERO ADDRESS 26, THEN 25 COMP - RNDZ/CTCH-UP (P) SET E.T. TO 02:00 & STBY	0:00 AT END OF THRUST, GET = 0 AND START (P) MAN CONT - OFF FDM - ATT/RATE CNTL AGENA TO TOP OF RETICLE ATT CNTL - PULSE (C) SET E.T. TO 02:00 & STBY	0:00 AT END OF THRUST GET = 0 AND START (P) MAN CONT - OFF FDM - ATT/RATE CNTL AGENA TO TOP OF RETICLE ATT CNTL - PULSE (C) SET E.T. TO 02:00 & STBY
		1:00 TRACK TARGET (C) READ θ (59) (P)	1:00 ON MARK (P) HOLD STARS FIXED IN RETICLE (C)	1:00 ON MARK (P) HOLD STARS FIXED IN RETICLE (C)
ON	2:00 ON MARK (P) START E.T. UP (C)	2:00 ON MARK (P) START E.T. UP (C)	2:00 ON MARK (P) START E.T. UP (C)	2:00 ON MARK (P) START E.T. UP (C) READ R (69) (P)
	3:00 READ R (69) (P)			
	4:00	4:00 READ θ (59) CALCULATE UP/DOWN ΔV CORRECTION START COMP - PUSH (P) MAN INSERT CORR INTO IVI'S	4:00 ON MARK (P) READ $\Delta\alpha$ (C) CALCULATE ΔV CORRECTION (P)	4:00 ON MARK (P) READ $\Delta\alpha$ (C) READ R (69) CALCULATE UP/DOWN AND FWD/AFT ΔV CORRECTION (P)
ON	5:00	ATT CNTL - RATE CMD MAN CONT - ON	ATT CNTL - RATE CMD MAN CONT - ON	ATT CNTL - RATE CMD MAN CONT - ON
OFF	ZERO ADD 25, 26, 27 (P) ENCDR - ON SEND CMD 270 (SPIRAL ANT SEL) ENCDR - OFF (P)	BORESIGHT ON AGENA #1 CORR THRUST RADIALLY ASAP (C)	BORESIGHT ON AGENA #1 CORR THRUST RADIALLY ASAP (C)	BORESIGHT ON AGENA #1 CORR THRUST ASAP (C)
		MAN CONT - OFF ATT CNTL - PULSE (C) COMP - RNDZ/CATCH-UP (P) BORESIGHT ON AGENA (C)	MAN CONT - OFF ATT CNTL - PULSE (C) ENCDR - ON SEND CMD 270 (SPIRAL ANT SEL)(P) ENCDR - OFF CNTL AGENA TO TOP OF RETICLE (C)	MAN CONT - OFF ATT CNTL - PULSE (C) ENCDR - ON SEND CMD 270 (SPIRAL ANT SEL)(P) ENCDR - OFF CNTL AGENA TO TOP OF RETICLE (C)
	7:00 READ θ (59) (P)	7:00 READ θ (59) (P)	7:00 ON MARK (P) HOLD STARS FIXED IN RETICLE (C)	7:00 ON MARK (P) HOLD STARS FIXED IN RETICLE (C) 8:00 READ R (69) (P)
OFF	10:00 READ θ (59) (P) 10:20	10:00 READ θ (59) CALCULATE UP/DOWN ΔV CORRECTION START COMP - PUSH (P) MAN INSERT CORR INTO IVI'S	10:00 ON MARK (P) READ $\Delta\alpha$ (C) COMP ΔV CORRECTION (P)	10:00 ON MARK (P) READ $\Delta\alpha$ (C) READ R (69) (P) CALCULATE UP/DOWN - FWD/AFT ΔV CORRECTION (P)
ON	ATT CNTL - RATE CMD MAN CONT - ON	ATT CNTL - RATE CMD MAN CONT - ON	ATT CNTL - RATE CMD MAN CONT - ON	ATT CNTL - RATE CMD MAN CONT - ON
	BORESIGHT ON AGENA 11:40 82° CORR APPLY THRUST (C) WHEN IVI STOP COUNTING UP. READ R (69) (P)	BORESIGHT ON AGENA #2 CORR THRUST RADIALLY ASAP (C)	BORESIGHT ON AGENA #2 CORR THRUST RADIALLY ASAP (C)	BORESIGHT ON AGENA #2 CORR THRUST ASAP (C)
ON	MAN CONT - OFF ATT CNTL - PULSE (C)	MAN CONT - OFF ATT CNTL - PULSE (C) COMP - RNDZ/CATCH-UP (P) BORESIGHT ON AGENA (C)	MAN CONT - OFF ATT CNTL - PULSE (C) CNTL AGENA TO TOP OF RETICLE (C)	MAN CONT - OFF ATT CNTL - PULSE (C) CNTL AGENA TO TOP OF RETICLE (C)

(*b*) Determination of 82° correction.
FIGURE 29–2.—Continued.

letters C for command pilot and P for pilot. The procedures start with the initiation of the circularization maneuver. The stopwatch feature of the clock, which is located on the pilot's instrument panel, is started and is used throughout the remainder of the rendezvous phase as the basic time reference for crew procedures. The event timer, which is located on the command pilot's instrument panel, is synchronized to the pilot's time and is used as a reference for the command pilot's critical events.

At 4 minutes after the circularization maneuver, the event timer is synchronized, and the computer is switched to the rendezvous mode. The command pilot controls the spacecraft attitude to boresight on the target, while the pilot verifies the pertinent computer constants, and, at the specific times requested by the charts, he records elevation angle and range to the target vehicle. This is continued until the initiation cue is reached.

The initiation cue was selected to provide the thrust application along the elevation angle of the line of sight to the target vehicle. Two of the reasons for this decision were that radar lock-on could be maintained continuously, and, secondly, that this provided a convenient pointing reference for use during the thrusting maneuver. The reasons were valid whether radar pointing commands or the optical sight was used. An additional procedural advantage to this technique was that it was not necessary for the command pilot to switch his flight director reference from radar to computer during the rendezvous. However, this approach meant that, in most cases, the command pilot would have some small velocity components to thrust-out individually in the lateral and vertical axes. This disadvantage was deemed an insufficient reason to sacrifice a reference which could be the same for all modes of operation.

After the initiation point is determined, the pilot initiates the closed-loop guidance sequence by depressing the START COMP button. The pilot then calculates the thrust required for the transfer maneuver from the flight data recorded on the charts. The data used are pitch angle and range. The purpose of this calculation is to check the onboard computer solution and to provide backup data in case a system should fail.

After the initiation point for transfer has been selected and backup solutions have been calculated, the pilot then determines when the clock is to be resynchronized with the onboard computer.

When the START COMP button is depressed, the required change in velocity is presented on a cockpit display. When the START COMP light comes on, the command pilot applies thrust to bring the displayed velocity values to zero and, at the same time, maintains boresighting on the target. This event completes the transfer maneuver. At the previously described time, the pilot resets the stopwatch to zero to synchronize it with the computer for the remainder of the rendezvous.

After the transfer maneuver, the command pilot remains boresighted on the target vehicle, and between the 3- and 5-minute period the computer collects radar data at intervals of 20 seconds to be used for the first midcourse cor-

(c)

NOMINAL	RADAR FAILURE	COMPUTER FAILURE	PLATFORM FAILURE
13:00	13:00 READ θ (59) (P)	13:00 ON MARK (P) HOLD STARS FIXED IN RETICLE (C)	13:00 ON MARK (P) HOLD STARS FIXED IN RETICLE (C) 14:00 READ R (69) (P)
15:00 READ R (69) (P) 16:00	16:00 READ θ (59) CALCULATE UP/DOWN CORRECTION START COMP - PUSH (P) MAN INSERT CORR INTO IVI'S	16:00 ON MARK (P) READ $\Delta\alpha$ (C) READ $\dot{R}$ FROM METER (C) CALCULATE ΔV CORRECTION (P)	16:00 ON MARK (P) READ $\Delta\alpha$ (C) READ R (69) (P) CALCULATE UP/DOWN - FWD/AFT ΔV CORRECTION (P)
	ATT CNTL - RATE CMD MAN CONT - ON	ATT CNTL - RATE CMD MAN CONT - ON	ATT CNTL - RATE CMD MAN CONT - ON
17:00	#3 CORR BORESIGHT ON AGENA THRUST RADIALLY ASAP (C)	#3 CORR BORESIGHT ON AGENA THRUST RADIALLY ASAP (C)	#3 CORR BORESIGHT ON AGENA THRUST ASAP (C)
ZERO ADD 25, 26, 27 (P)	MAN CONT - OFF ATT CNTL - PULSE (C)	MAN CONT - OFF ATT CNTL - PULSE (C)	MAN CONT - OFF ATT CNTL - PULSE (C)
	COMP - RNDZ/CATCH-UP (P) BORESIGHT ON AGENA (C)	CNTL AGENA TO CENTER OF RETICLE	CNTL AGENA TO CENTER OF RETICLE
19:00 READ θ (59) (P)	19:00 READ θ (59) (P)	19:00 ON MARK (P) HOLD STARS FIXED IN RETICLE (C)	19:00 ON MARK (P) HOLD STARS FIXED IN RETICLE (C) 20:00 READ R (69) (P)
22:00 READ θ (59) (P) 22:20	22:00 READ θ (59) CALCULATE UP/DWN ΔV CORRECTION START COMP - PUSH (P) MAN INSERT CORR INTO IVI'S	22:00 ON MARK (P) READ $\Delta\alpha$ (C) READ $\dot{R}$ FROM METER (C) CALCULATE ΔV CORRECTION (P)	22:00 ON MARK (P) READ $\Delta\alpha$ (C) READ R (69) (P) CALCULATE UP/DOWN - FWD/AFT ΔV CORRECTION (P)
ATT CNTL - RATE CMD MAN CONT - ON	ATT CNTL - RATE CMD MAN CONT - ON	ATT CNTL - RATE CMD MAN CONT - ON	ATT CNTL - RATE CMD MAN CONT - ON
BORESIGHT ON AGENA 23:40 34° CORR APPLY THRUST (C)	#4 CORR BORESIGHT ON AGENA THRUST RADIALLY ASAP (C)	#4 CORR BORESIGHT ON AGENA THRUST RADIALLY ASAP (C)	#4 CORR BORESIGHT ON AGENA THRUST ASAP (C)
WHEN IVI'S STOP COUNTING UP. READ R (69) (P) COMP - CATCH-UP (C) ZERO ADD 25, 26, 27 (P) START COMP - PUSH BORESIGHT ON AGENA (C)	COMP RNDZ/CATCH-UP PUSH START COMP (P) AFTER THRUST, BEGIN LINE OF SIGHT NULLING	AFTER THRUST, BEGIN LINE OF SIGHT NULLING AND RANGE AND RANGE RATE MONITORING (C)	AFTER THRUST, BEGIN LINE OF SIGHT NULLING AND RANGE AND RANGE RATE MONITORING (C)
IF REQ CAGE & FREE PLAT NULL LOS MOTIONS (C)			
$40 > \dot{R} > 25$ AT R = 15,000 FT AT 3,000 FT, REDUCE $\dot{R}$ TO 4 FT/SEC (C) AT 500 FT, DOCKING LT - ON (P) AT 100 FT, REDUCE $\dot{R}$ TO 1/2 FT/SEC (C)	26:30 REMOVE 15 FT/SEC WHEN APPROACHING TARGET VISUALLY: BRAKE AS NECESSARY AT 500 FT, DOCKING LT - ON (P) AT 100 FT, REDUCE $\dot{R}$ TO 1/2 FT/SEC (C)	$40 > \dot{R} > 25$ AT R = 15,000 FT AT 3,000 FT, REDUCE $\dot{R}$ TO 4 FT/SEC (C) AT 500 FT, DOCKING LT - ON (P) AT 100 FT, REDUCE $\dot{R}$ TO 1/2 FT/SEC (C)	$40 > \dot{R} > 25$ AT R = 15,000 FT AT 3,000 FT, REDUCE $\dot{R}$ TO 4 FT/SEC (C) AT 500 FT, DOCKING LT - ON (P) AT 100 FT, REDUCE $\dot{R}$ TO 1/2 FT/SEC (C)
AT 50 FEET, RDR - OFF (P) ENCDR - ON CMD 250 (ACQ LTS - OFF) WHEN REQ (P)	ENCDR - ON CMD 250 (ACQ LTS - OFF) WHEN REQ (P)	AT 50 FEET, RDR - OFF (P) ENCDR - ON CMD 250 (ACQ LTS - OFF) WHEN REQ (P)	AT 50 FEET, RDR - OFF (P) ENCDR - ON CMD 250 (ACQ LTS - OFF) WHEN REQ (P)

(c) Determination of 34° correction, and braking.

FIGURE 29–2.—Concluded.

rection. During this time, the pilot interrogates the computer to obtain the necessary data to analyze closed-loop guidance and trajectory parameters. This information is recorded on a monitor sheet (fig. 29–3). When the radar data collection is completed by the computer at 5 minutes, the START COMP light goes off, indicating that the computer is sequencing to the next part of its program. The crew now has an option of alining the platform during the next 5 minutes 20 seconds or of ignoring it. Their decision is based upon premission rules regarding accuracy requirements of the platform. The pilot then takes certain data from the computer in order to obtain the change in velocity requirements for a backup solution to the first midcourse maneuver. The first midcourse correction occurs at a point in the trajectory where 81.8° central angle travel of the target remains until intercept. Just prior to the first midcourse maneuver, the spacecraft must be boresighted for a final radar data collection by the computer. As soon as this occurs, the required velocities for the first midcourse correction are displayed. The command pilot then applies thrust to drive the displays to zero. Upon the completion of thrusting, the first midcourse correction is complete, and the identical cycle is repeated for the second midcourse correction which occurs at 33.6° central angle travel to go to rendezvous. This maneuver corresponds to a time of 23 minutes 40 seconds after the midpoint of the transfer maneuver.

When the second correction has been completed, the computer is switched from the rendezvous mode to the catchup mode. This allows radar data to the computer to be updated every one-eighth second. From this point in the trajectory, the target motion with respect to the stars should be essentially zero; therefore, the command pilot is required to note any motion of the target vehicle with respect to the celestial

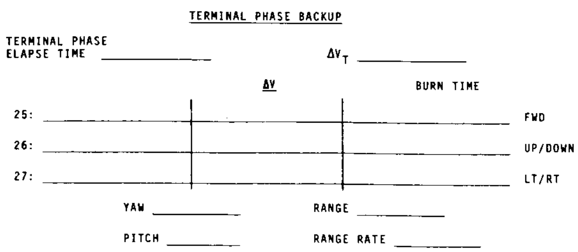


FIGURE 29–3.—Terminal phase backup monitor sheet.

background and null the motion. The pilot, meanwhile, is continuously monitoring pitch angle, range, and range rate to determine trajectory characteristics and is assisting the command pilot by giving him position reports and providing backup information. Braking thrust at the termination of rendezvous is applied as a function of range. The nominal range for initiation of braking is 3000 feet, and at 1500 feet the range rate is reduced to 4 feet per second.

Backup Procedures

Columns 2, 3, and 4 on figures 29–2 through 29–4 present the sequence of the backup rendezvous procedures in the event of radar, computer, or platform failure. It should be noted that the procedures and the arrangement of the procedures were specifically tailored to insure that an orderly transfer could be made in the event of system failure. Four midcourse corrections are used in the backup procedures, while only two are used in closed-loop guidance. The increased number was required to detect a trajectory error as early as possible and to make the appropriate corrections. The second and fourth backup measurements provide a check of the first and second closed-loop maneuvers. An optical sight with a collimated reticle was one of the essential pieces of hardware to implement the backup procedures. This sight was used to track the target and measure inertial angular rates.

Radar Failure

A radar failure eliminates range and range rate from the analog meter and the computer. In this event, the initiation cue is based upon line-of-sight elevation angle. The spacecraft is controlled to a specified pitch attitude of 27.4° using the flight director indicators, and the target vehicle is visually observed. Visual observation is a mandatory requirement unless thrusting is initiated on ground-calculated time. When the target passes through the center of the reticle, thrusting is initiated. Once again the nominal change in velocity is applied along the line of sight, and a correction normal to the line of sight is based upon the measured change in the elevation angle as read from the computer. The intermediate corrections rely upon this capability to read elevation angle from the computer to enable the pilot to calculate corrections normal to the line of sight. Since ranging information is not available, a small braking maneuver is selected by time, and the final braking thrust is not applied until the command pilot can visually detect size growth of the target vehicle.

Computer Failure

A computer failure precludes the use of accurate elevation or pitch angle as an initiation cue. The reference then used to provide this cue is the attitude indicator ball. Loss of the computer also prevents use of the velocity displays. The transfer thrusting application is therefore based on the nominal change in velocity along the line of sight and a calculated change normal to the line of sight. The calculation is based on the change from nominal of the inertial elevation angle. The first two intermediate corrections are based only upon the variation of the inertial elevation angle from nominal, using the optical reticle as the measuring device and the celestial background as the inertial reference. The last two corrections include range-rate data from the analog meter. The pilot uses the stopwatch feature of his wristwatch to measure the time of thrust in each axis which corresponds to the required change in velocity.

Platform Failure

In the event of a platform failure, the initiation cue is ranged obtained from the computer. The initial transfer and the four intermediate corrections are based upon deviations in the change of range and inertial elevation angle from the nominal. The change in inertial elevation angle is measured by using the optical reticle. The reticle pattern and markings were designed to insure the required accuracy for this measurement. The procedures for the trajectory from the end of the fourth backup midcourse maneuver to termination of rendezvous are the same as previously discussed under closed-loop rendezvous procedure.

Flight Charts

The flight charts are an extension of the Gemini V charts and were tailored for the Gemini VI-A mission. The Gemini V charts were developed specifically for the planned exercise with the rendezvous evaluation pod. The Gemini VI-A charts have been refined considerably from Gemini V charts due to experience gained from simulations and crew training. Figure 29-3 is the form used for recording the ground-computed termination phase initiation. Figure 29-4 is the form used for recording data necessary to monitor the trajectory and for the determination of the proper point for transfer. Figure 29-5 is used to determine the initial thrusting required for transfer as a check on the closed-loop solution and as a backup in case of a system failure. Figure 29-6 is used to calculate intermediate corrections in the backup procedures and to check the closed-loop solution for the two midcourse maneuvers. All measurements and thrust applications are made orthogonally with respect to an axis system oriented along the spacecraft axes. The spacecraft X-axis is alined with the line of sight to the target. Figure 29-7 is the monitor sheet used for closed-loop guidance. Figure 29-8 is a curve used to determine separation from the target vehicle using only range from the computer.

Figure 29-9 is a polar plot of the nominal Gemini VI-A trajectory from the circularization maneuver to termination of rendezvous. Nominal range, range rates, elevation angles, and ground elapsed times are provided at various points along the trajectory.

Discussion of the Gemini VI-A Rendezvous

The closed-loop guidance technique was used satisfactorily during the Gemini VI-A rendezvous mission. The radar range data that were read from the computer were highly accurate throughout the entire maneuver and provided the crew with the necessary information to monitor the trajectory, shown in figure 29-10(a). Radar range-rate data from the analog meter showed close correlation to computed data with less than 3-feet-per-second difference, and was limited in accuracy only by the meter markings and readability. Angle data after the circularization maneuvers were slightly erratic in value (fig. 29-10(b)). The pilot noted that the closed-loop guidance solutions appeared to give values near the nominal and was concerned primarily with the way this anomaly would affect the selection of the correct angle to push the START COMP button during the transfer maneuver.

(a)

GT–6 RENDEZVOUS FLIGHT CHARTS
NOMINAL AND ACTUAL CONDITIONS – CIRCULARIZATION TO TERMINAL INITIATION

AFTER SWITCHING COMP TO RENDEZVOUS MODE, PERFORM THE FOLLOWING:

VERIFY r_T: 54 73082
T: 53 53776
$1/A_T$: 24 12690
RLO: 92 00000

INPUT ωT: 83 13000
Δ: 93 04820

RDR DATA POINTS	TIME FROM NSR INITIATE MIN:SEC	θ NOM DEG	θ ACTUAL ADD 59 DEG	R NOM N.M.	R ACTUAL ADD 69 N.M.	ΔR ACTUAL N.M.	ΔR NOM N.M.		ΔV_I NOM FPS	ΔV_I ACTUAL ADD 71 FPS	ΔV_T NOM FPS
1	4:00	5.4		136.09			2.60				
2	5:40	5.5		133.49			2.60				
3	7:20	5.7		130.89			2.60				
4	9:00	5.8		128.29			2.60				
5	10:40	6.0		125.69			2.60				
6	12:20	6.2		123.09			2.60				
7	14:00	6.3		120.49			2.60				
8	15:40	6.5		117.89			2.60		230.0		518
9	17:20	6.7		115.29			2.60		222.1		502
10	19:00	6.9		112.69			2.60		214.2		486
11	20:40	7.1		110.09			2.60		206.3		470
12	22:20	7.3		107.49			2.60		198.4		454
13	24:00	7.5		104.89			2.60		190.5		438
14	25:40	7.7		102.30			2.60		182.6		422
15	27:20	7.9		99.71			2.59		184.7		406
16	29:00	8.2		97.12			2.59		176.9		390
17	30:40	8.5		94.53			2.59		169.1		374
18	32:20	8.8		91.94			2.59		161.3		358
19	34:00	9.1		89.35			2.59		153.5		342
20	35:40	9.4		86.76			2.59		145.7		327

(*a*) Between 4 minutes and 35 minutes 40 seconds from coelliptical maneuver (NSR).
FIGURE 29–4.—Transfer maneuver monitor sheet.

The backup solution calculated from the flight data charts indicates that an angle bias existed. The fact that range and range rate prior to transfer were exactly nominal led to a belief that elevation angle and elevation angle rate also should have been nominal. This difference may have been partly due to a platform alinement. The cause of the remainder of the difference has not been determined. This effect caused the crew to transfer one data point later than the nominal point, and also indicated that the two spacecraft were less than the nominal 15-nautical-mile vertical separation. This in turn led to an erroneous change in velocity solution to be calculated along the line of sight for the backup procedure.

The ground-calculated backup solution showed close agreement with the closed-loop data. In subsequent missions, however, ground solutions will not be available for some rendezvous transfers; hence, the requirement will continue to exist to provide the crew with an independent onboard method of calculating transfer velocities.

The target-center polar plot was used to provide backup verification. The data are correct for direction and generalized for magnitude of the thrust vector. The five values that were available to the crew for the transfer solution are shown in table 29–I.

It was noted by the pilot, immediately after the final backup calculation, that the 23-foot-per-second solution along the line of sight

(LOS) was in error, as the data from points prior to this gave 32 feet per second. As noted in table 29–I, the polar plot and the change in range-change ($\Delta\Delta R$) solutions indicate that 32 feet per second should be applied along the line of sight. The ground-calculated solution was additional verification of this number. Had the computer failed to arrive at a solution or given an erroneous value, sufficient information existed onboard from the polar plot and $\Delta\Delta R$ method to correctly determine that the transfer change in velocity was in fact 32 feet per second along the line of sight. This was the change in velocity that the crew would have applied in case of a failure mode. This problem highlights the fact that the crew must have ample onboard methods to correctly interpret and execute the transfer maneuver.

(b)

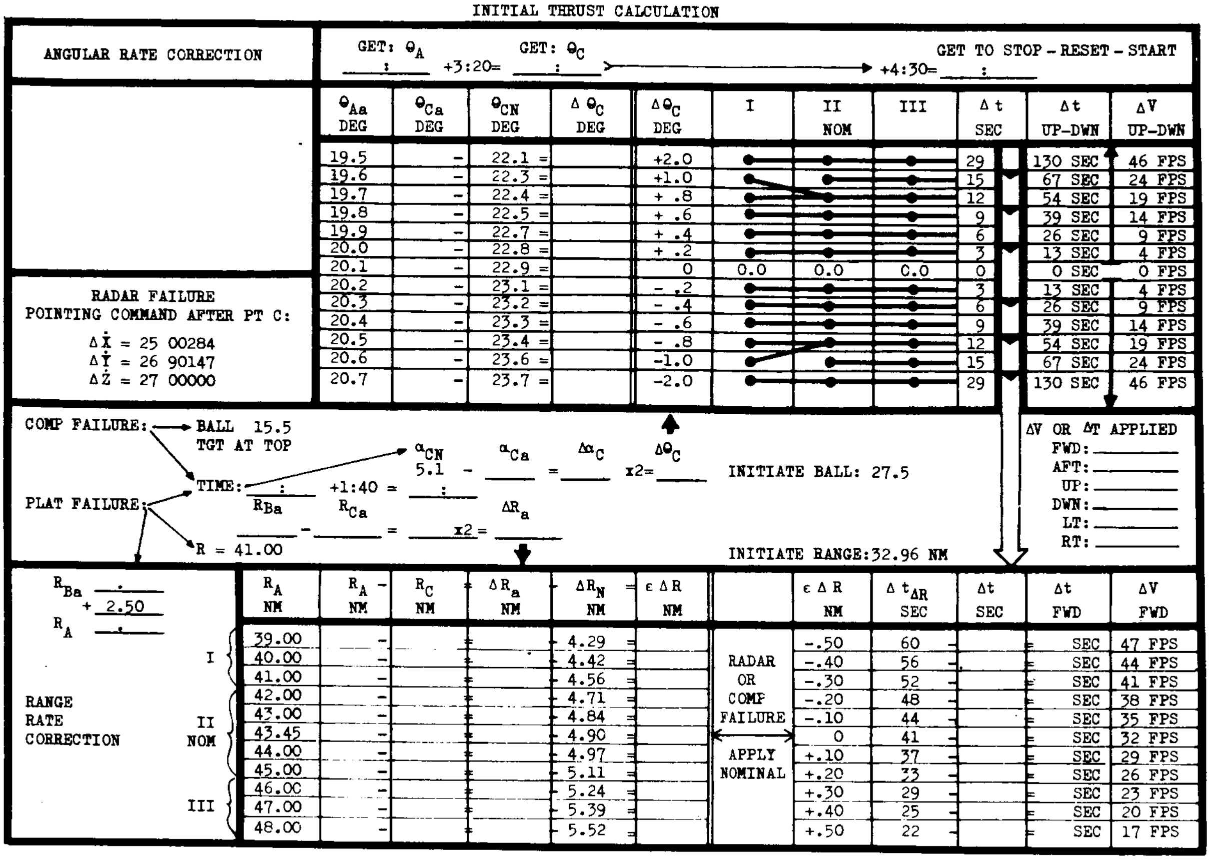
RDR DATA POINTS	TIME FROM NSR INITIATE MIN:SEC	Θ NOM DEG	Θ ACTUAL ADD 59 DEG	R NOM N.M.	R ACTUAL ADD 69 N.M.	ΔR ACTUAL N.M.	ΔR NOM N.M.		ΔV_T NOM FPS	ΔV_T ACTUAL ADD 71 FPS	ΔV_T NOM FPS
21	37:20	9.7		84.18			2.58		137.9		311
22	39:00	10.0		81.60			2.58		130.2		296
23	40:40	10.4		79.02			2.58		122.5		281
24	42:20	10.8		76.44			2.58		114.8		265
25	44:00	11.2		73.87			2.57		107.1		249
26	45:40	11.7		71.30			2.57		99.5		234
27	47:20	12.2		68.73			2.57		92.0		219
28	49:00	12.7		66.17			2.56		84.5		204
29	50:40	13.3		63.61			2.56		77.1		189
30	52:20	13.9		61.06			2.55		69.9		174
31	54:00	14.5		58.52			2.54		62.8		159
32	55:40	15.3		55.98			2.54		56.1		145
33	57:20	16.1		53.45			2.53		49.7		131
34	59:00	16.9		50.93			2.52		43.9		118
35	00:40	17.9		48.43			2.50		38.9		106
36	02:20	19.0		45.93			2.50		35.0		95
37	04:00	A 20.1		43.45			2.48		32.6		86
38	05:40	B 21.4		40.99			2.46		32.0		80
39	07:20	C 22.9		38.55			2.44		33.3		75

(b) Between 37 minutes 20 seconds and 1 hour 7 minutes 20 seconds from coelliptical maneuver (NSR).

FIGURE 29–4.—Concluded.

TABLE 29–I.—*Transfer Solution Values*

Thrust	Closed-loop	Backup charts	Ground	Polar plot	$\Delta\Delta R$
Along line of sight	31 ft/sec forward	23 ft/sec forward	32 ft/sec forward	32 ft/sec forward	32 ft/sec forward
Normal line of sight	4 ft/sec up	2 ft/sec up	2 ft/sec up	0 ft/sec	0 ft/sec
Lateral line of sight	1 ft/sec right	-------------	2 ft/sec left	-------------	


GT-6 RENDEZVOUS FLIGHT CHARTS
INITIAL THRUST CALCULATION

ANGULAR RATE CORRECTION

GET: θ_A ___:___ +3:20= GET: θ_C ___:___ → +4:30= ___:___ GET TO STOP – RESET – START

θ_{Aa} DEG	θ_{Ca} DEG	θ_{CN} DEG	$\Delta\theta_C$ DEG	$\Delta\theta_C$ DEG	I	II NOM	III	Δt SEC	Δt UP–DWN	ΔV UP–DWN
19.5	–	22.1 =		+2.0				29	130 SEC	46 FPS
19.6	–	22.3 =		+1.0				15	67 SEC	24 FPS
19.7	–	22.4 =		+ .8				12	54 SEC	19 FPS
19.8	–	22.5 =		+ .6				9	39 SEC	14 FPS
19.9	–	22.7 =		+ .4				6	26 SEC	9 FPS
20.0	–	22.8 =		+ .2				3	13 SEC	4 FPS
20.1	–	22.9 =		0	0.0	0.0	C.0	0	0 SEC	0 FPS
20.2	–	23.1 =		– .2				3	13 SEC	4 FPS
20.3	–	23.2 =		– .4				6	26 SEC	9 FPS
20.4	–	23.3 =		– .6				9	39 SEC	14 FPS
20.5	–	23.4 =		– .8				12	54 SEC	19 FPS
20.6	–	23.6 =		–1.0				15	67 SEC	24 FPS
20.7	–	23.7 =		–2.0				29	130 SEC	46 FPS

RADAR FAILURE
POINTING COMMAND AFTER PT C:

$\Delta\dot{X}$ = 25 00284
$\Delta\dot{Y}$ = 26 90147
$\Delta\dot{Z}$ = 27 00000

COMP FAILURE: → BALL 15.5
TGT AT TOP

PLAT FAILURE:

TIME: ___:___ +1:40 = ___:___

α_{CN} 5.1 – α_{Ca} ____ = $\Delta\alpha_C$ ____ x2= $\Delta\theta_C$ ____ INITIATE BALL: 27.5

R_{Ba} ____ – R_{Ca} ____ = ____ x2 = ΔR_a ____

R = 41.00

INITIATE RANGE: 32.96 NM

ΔV OR ΔT APPLIED
FWD: ____
AFT: ____
UP: ____
DWN: ____
LT: ____
RT: ____

R_{Ba} ___.___
+ 2.50
R_A ___.___

RANGE
RATE
CORRECTION

	R_A NM	R_A – NM	R_C NM	ΔR_a NM	ΔR_N NM	$\epsilon\Delta R$ NM		$\epsilon\Delta R$ NM	$\Delta t_{\Delta R}$ SEC	Δt SEC	Δt FWD	ΔV FWD
I	39.00	–			4.29 =			–.50	60 –		= SEC	47 FPS
I	40.00	–			4.42 =		RADAR	–.40	56 –		= SEC	44 FPS
I	41.00	–			4.56 =		OR	–.30	52 –		= SEC	41 FPS
II NOM	42.00	–			4.71 =		COMP	–.20	48 –		= SEC	38 FPS
II NOM	43.00	–			4.84 =		FAILURE	–.10	44 –		= SEC	35 FPS
II NOM	43.45	–			4.90 =			0	41 –		= SEC	32 FPS
II NOM	44.00	–			4.97 =		APPLY	+.10	37 –		= SEC	29 FPS
II NOM	45.00	–			5.11 =		NOMINAL	+.20	33 –		= SEC	26 FPS
III	46.00	–			5.24 =			+.30	29 –		= SEC	23 FPS
III	47.00	–			5.39 =			+.40	25 –		= SEC	20 FPS
III	48.00	–			5.52 =			+.50	22 –		= SEC	17 FPS

FIGURE 29–5.—Initial thrust calculation sheet.

A significant problem developed when the Gemini VII spacecraft went into darkness. The Gemini VI–A crew was not able to acquire it visually until a range of 25.7 nautical miles, when the spacecraft's docking light became faintly visible. The observed light was not sufficient to provide tracking for the first two backup midcourse corrections. The flashing acquisition lights were not seen until 14.5 nautical miles because the apparent intensity of the docking light was much greater. The crew had previously been briefed that the acquisition light should be visible for tracking at a range of 30 nautical miles.

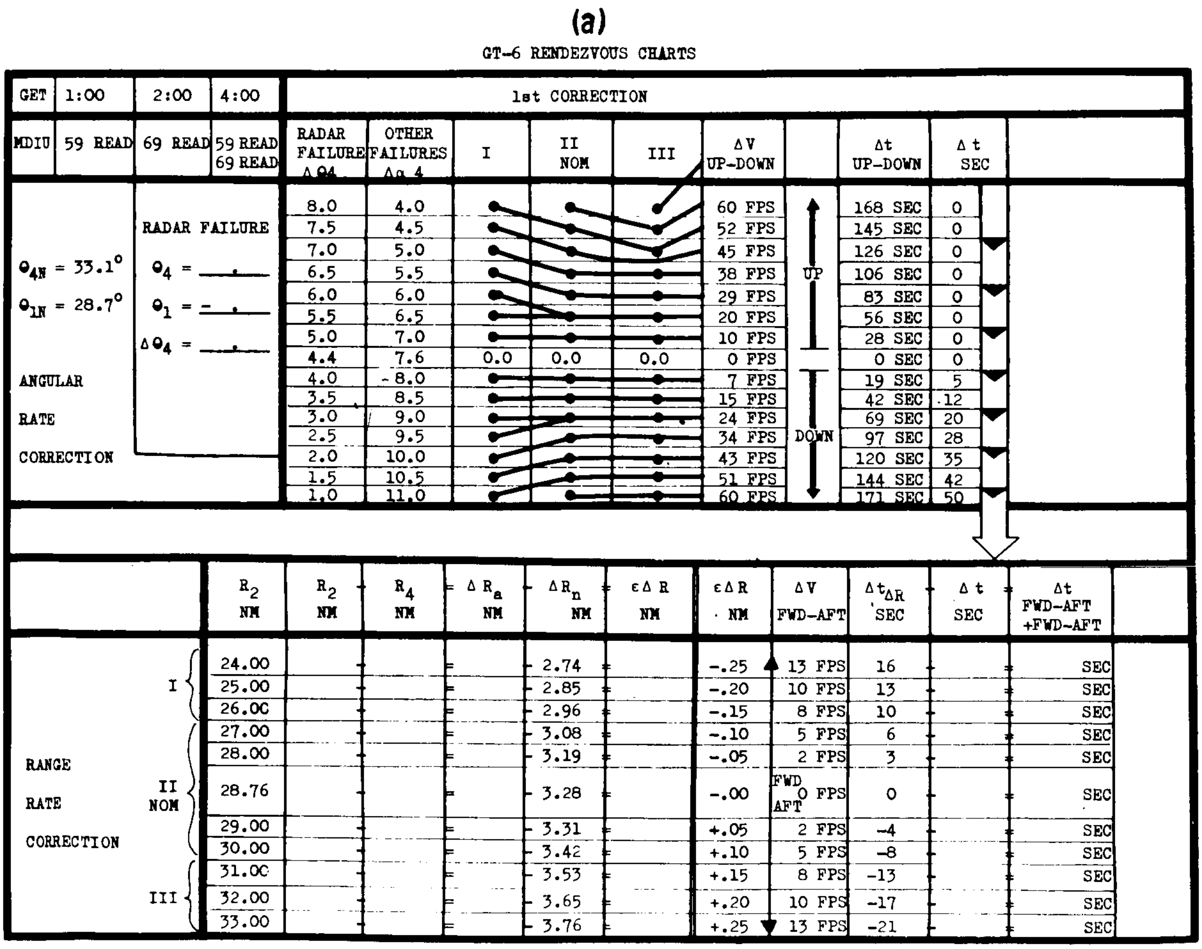
The platform alinement performed during the period from 5 to 10 minutes after transfer precluded any backup solution to the first midcourse maneuver. The backup solution for the second midcourse maneuver was calculated and requested 6 feet per second up, versus 3 feet per second up, and 4 feet per second forward for the closed loop (table 29–II). The backup solution would have been adequate to provide an intercept with the Gemini VII spacecraft.

After the second midcourse correction, the computer was switched into the catchup mode and the pilot recorded pitch angle and range data at 1-minute time intervals. The command pilot nulled the inertial angular rate by thrusting toward the target vehicle whenever it exhibited motion with reference to the stars.

The target vehicle became illuminated in sunlight at approximately 0.74 nautical mile. Braking was initiated at 3000 feet and completed at 1500 feet, at which time the range rate had been reduced to 7 feet per second. The end of the rendezvous occurred and station keeping was initiated when the Gemini VI–A spacecraft was directly below the Gemini VII spacecraft at a distance of 120 feet.

TABLE 29-II.—*Midcourse Maneuver Values*

Thrust	Closed-loop	Backup charts	Polar plot
(a) First midcourse maneuver			
Along line of sight	7 ft/sec forward	Not available due to computer program	5 ft/sec forward
Normal line of sight	7 ft/sec up	Not available due to platform alinement	5 ft/sec up
Lateral line of sight	5 ft/sec left	Not calculated	Not calculated
(b) Second midcourse maneuver			
Along line of sight	4 ft/sec forward	Not available due to computer program	5 ft/sec forward
Normal line of sight	3 ft/sec up	6 ft/sec up	5 ft/sec up
Lateral line of sight	6 ft/sec right	Not calculated	Not calculated



(*a*) First correction maneuver.
FIGURE 29-6.—Intermediate correction calculation sheets.

(b)

GT-6 RENDEZVOUS FLIGHT CHARTS

GET	7:00	8:00	10:00	2nd CORRECTION								
MDIU	59 READ	69 READ	59 READ 69 READ	RADAR FAILURE $\Delta\theta_{10}$	OTHER FAILURES $\Delta\alpha_{10}$	I	II NOM	III	ΔV UP-DOWN		Δt UP-DOWN	Δt SEC
		RADAR FAILURE		9.5	2.5				42 FPS	UP	118 SEC	0
$\theta_{10K} = 44.1°$		θ_{10}: ___ . ___		9.0	3.0				36 FPS		101 SEC	0
$\theta_{7N} = 38.1°$		θ_7 :- ___ . ___		8.5	3.5				30 FPS		85 SEC	0
ANGULAR		$\Delta\theta_{10}$: ___ . ___		8.0	4.0				24 FPS		69 SEC	0
RATE				7.5	4.5				18 FPS		51 SEC	0
CORRECTION				7.0	5.0				12 FPS		32 SEC	0
				6.5	5.5				6 FPS		16 SEC	0
				6.0	6.0	0.0	0.0	0.0	0 FPS		0 SEC	0
				5.5	6.5				6 FPS		16 SEC	5
				5.0	7.0				12 FPS		32 SEC	9
				4.5	7.5				18 FPS		51 SEC	15
				4.0	8.0				24 FPS	DOWN	69 SEC	20
				3.5	8.5				30 FPS		85 SEC	25
				3.0	9.0				36 FPS		101 SEC	29
				2.5	9.5				42 FPS		118 SEC	34

		R_8 NM	R_8 NM	R_{10} NM	ΔR_a NM	ΔR_n NM	$\epsilon\Delta R$ NM	$\epsilon\Delta R$ NM	ΔV FWD-AFT	$\Delta t_{\Delta R}$ SEC	Δt SEC	Δt FWD-AFT +FWD -AFT
	I	17.00				2.43		-.25	13 FPS	16		SEC
		17.50				2.50		-.20	10 FPS	13		SEC
		18.00				2.57		-.15	8 FPS	10		SEC
RANGE		18.50				2.65		-.10	5 FPS	6		SEC
RATE		19.00				2.72		-.05	2 FPS	3		SEC
CORRECTION	II NOM	19.37				2.77		.00	FWD 0 FPS AFT	0		SEC
		20.00				2.86		+.05	2 FPS	-4		SEC
		20.50				2.93		+.10	5 FPS	-8		SEC
	III	21.00				3.00		+.15	8 FPS	-13		SEC
		21.50				3.08		+.20	10 FPS	-17		SEC
		22.00				3.15		+.25	13 FPS	-21		SEC

(*b*) Second correction maneuver.

FIGURE 29–6.—Continued.

Status of Gemini Rendezvous Procedures and Charts

Numerous modifications to the Gemini VI–A procedures and flight data charts have been made for the Gemini VIII mission. In addition, possible changes are contemplated for subsequent missions. A format change in the charts was indicated by usage of the Gemini V and VI–A charts. The charts used for the backup transfer, as well as the four intermediate correction charts, have been changed to a nomograph presentation. This allows the user to interpolate directly without calculation, as in the case of the present charts. In addition, this presentation provides a far greater expansion of the data and limits than was possible with the tabular format. This was not critical with the present charts and mission requirements, but future applications may require a much greater flexibility; thus it was deemed advisable to change from this standpoint.

The calculations required have been changed to make them additive only, rather than additive or subtractive. The concept of the intermediate correction charts for monitoring and backup has also been changed. Previously, the charts were designed using a reference trajectory with a perfect intercept of the target. When an error in the trajectory was noted, the present charts tried to force the trajectory back to nominal; consequently, the rendezvous trajectory was shifted, and rendezvous was obtained earlier or later, depending on the error. This approach is sufficient to complete rendezvous but does not constrain the target's total central angle travel to 130°; therefore, the time to rendezvous becomes a variable. The new charts provide that the backup procedures present the same calculated corrections as the closed-loop guidance, and further insure that the same total central angle travel is obtained.

Changes to the computer program and readout capability have decreased crew workload and have increased ability to obtain key parameters at the required times. These items are instantaneous range, range rate, and pitch angle. Range and pitch angle were formerly available only at specified intervals and defined times in the programing sequence. Range rate had to be calculated from range points. Monitoring of the closed-loop guidance previously has been restricted to only certain time intervals, due to inability to obtain these parameters. The crew will now have access to these values over a greatly extended time period. This change greatly enhances monitoring of the closed-loop guidance and provides far greater latitude in developing procedures which are more consistent with operational constraints. This point should not be overlooked in the design of future space applications.

The flight director attitude displays were marked in a manner whereby the reading accuracy could be read to only ±2° in most areas and to ±5° when the spacecraft was within ±30° of 90° pitch. The displays are presently being re-marked to 1° increments and will provide reading accuracy to within ±0.5° at all pitch angles. This new marking will provide accurate angle measurements for the transfer maneuver and for midcourse corrections in case of computer failure.

Concluding Remarks

The closed-loop rendezvous guidance system performed satisfactorily. The radar range in-

(c)

GT-6 RENDEZVOUS FLIGHT CHARTS

GET	13:00	14:00	16:00	3rd CORRECTION							
MDIU	59 READ	69 READ	59 READ 69 READ	RADAR FAILURE $\Delta\theta_{16}$	OTHER FAILURES $\Delta\alpha_{16}$	I	II NOM	III	ΔV UP-DOWN	Δt UP-DOWN	Δt SEC
$\theta_{16N} = 59.4°$		RADAR FAILURE		12.0	0				28 FPS	80 SEC	0
$\theta_{13N} = 51.0°$		θ_{16} = ___ . ___		11.5	.5				24 FPS	68 SEC	0
		θ_{13} = ___ . ___		11.0	1.0				20 FPS	56 SEC	0
		$\Delta\theta_{16}$ = ___ . ___		10.5	1.5				16 FPS UP	44 SEC	0
ANGULAR				10.0	2.0				12 FPS	33 SEC	0
RATE				9.5	2.5				8 FPS	22 SEC	0
CORRECTION				9.0	3.0				4 FPS	11 SEC	0
				8.4	3.6	0.0	0.0	0.0	0 FPS	0 SEC	0
				8.0	4.0				3 FPS	8 SEC	2
				7.5	4.5				6 FPS	16 SEC	5
				7.0	5.0				9 FPS	25 SEC	7
				6.5	5.5				12 FPS DOWN	34 SEC	10
				6.0	6.0				16 FPS	44 SEC	13
				5.5	6.5				20 FPS	55 SEC	16
				5.0	7.0				24 FPS	69 SEC	20

		R_{14} NM	R_{14} NM	R_{16} NM	ΔR_a NM	ΔR_n NM	$\epsilon\Delta R$ NM	$\epsilon\Delta R$ NM	ΔV FWD-AFT	$\Delta t_{\Delta R}$ SEC	Δt SEC	Δt FWD-AFT +FWD -AFT	$\dot{R}$ ANALOG 16:00
RANGE RATE CORRECTION	I	9.00				1.58		-.25	13 FPS	16		SEC	85
	I	9.50				1.66		-.20	10 FPS	13		SEC	88
	I	10.00				1.75		-.15	8 FPS	10		SEC	90
	II NOM	10.50				1.84		-.10	5 FPS	6		SEC	93
	II NOM	11.00				1.93		-.05	2 FPS	3		SEC	96
	II NOM	11.76				2.06		.00	FWD 0 FPS AFT	0		SEC	98
	II NOM	12.00				2.10		+.05	2 FPS	-4		SEC	100
	II NOM	12.50				2.19		+.10	5 FPS	-8		SEC	103
	III	13.00				2.28		+.15	8 FPS	-13		SEC	106
	III	13.50				2.36		+.20	10 FPS	-17		SEC	108
	III	14.00				2.45		+.25	13 FPS	-21		SEC	111

(c) Third correction maneuver.
FIGURE 29-6.—Continued.

(d)

GT-6 RENDEZVOUS FLIGHT CHARTS

GET	19:00	20:00	22:00	4th CORRECTION								
MDIU	59 READ	69 READ	59 READ 69 READ	RADAR FAILURE $\Delta\theta_{22}$	OTHER FAILURES $\Delta\alpha_{22}$	I	II NOM	III	ΔV UP-DOWN		Δt UP-DOWN	Δt SEC
		RADAR FAILURE		18.5	-6.5				30 FPS		84 SEC	0
				17.5	-5.5				25 FPS		72 SEC	0
$\theta_{22N} = 80.7^\circ$		θ_{22}: ____.__		16.5	-4.5				20 FPS		56 SEC	0
				15.5	-3.5				15 FPS	UP	42 SEC	0
$\theta_{19N} = 69.2^\circ$		θ_{19}:-____.__		14.5	-2.5				10 FPS		30 SEC	0
				13.5	-1.5				6 FPS		18 SEC	0
		$\Delta\theta_{22}$:____.__		12.5	- .5				3 FPS		9 SEC	0
ANGULAR				11.5	+ .5	0.0	0.0	0.0	0 FPS		0 SEC	0
				10.5	+1.5				3 FPS		9 SEC	3
RATE				9.5	+2.5				6 FPS		18 SEC	5
				8.5	+3.5				10 FPS		30 SEC	9
CORRECTION				7.5	+4.5				15 FPS	DOWN	42 SEC	12
				6.5	+5.5				20 FPS		56 SEC	16
				5.5	+6.5				25 FPS		72 SEC	21
				4.5	+7.5				30 FPS		84 SEC	25

		R_{20} NM	R_{20} NM	R_{22} NM	ΔR_a NM	ΔR_n NM	$\epsilon\Delta R$ NM	$\epsilon\Delta R$ NM	ΔV FWD-AFT	$\Delta t_{\Delta R}$ SEC	Δt SEC	Δt FWD-AFT +FWD -AFT	$\dot{R}$ ANALOG 22:00
	I	4.00				0.86		-.25	13 FPS	16		SEC	51
	I	4.50				0.97		-.20	10 FPS	13		SEC	54
	I	5.00				1.08		-.15	8 FPS	10		SEC	56
	II NOM	5.50				1.18		-.10	5 FPS	6		SEC	59
RANGE	II NOM	6.00				1.29		-.05	2 FPS	3		SEC	62
RATE	II NOM	6.32				1.36		.00	FWD 0 FPS AFT	0		SEC	64
CORRECTION	II NOM	7.00				1.51		+.05	2 FPS	-4		SEC	66
	II NOM	7.50				1.61		+.10	5 FPS	-8		SEC	69
	III	8.00				1.72		+.15	8 FPS	-13		SEC	72
	III	8.50				1.83		+.20	10 FPS	-17		SEC	74
	III	9.00				1.94		+.25	13 FPS	-21		SEC	77

(*d*) **Fourth correction maneuver.**
FIGURE 29–6.—Concluded.

formation obtained through the computer was very accurate and provided good data to monitor the closed-loop solution. The angle data obtained were slightly erratic and had a possible bias prior to the transfer maneuver. The angle data alone would provide a poor basis on which to base a rendezvous maneuver.

The backup charts and the polar plot gave the crew good information on the rendezvous trajectory and provided a means to complete the rendezvous maneuver in case system failures were encountered.

A continuously updated local-horizontal reference on the platform is highly desirable. The flight director attitude indicator that is referenced to local horizontal provides the flight crew an excellent reference for both the closed-loop and the backup guidance systems.

The optical sight is a mandatory piece of equipment for backup guidance techniques.

The acquisition lights used on Gemini VII were unsatisfactory and precluded optical tracking for transfer and the first two backup midcourse corrections. The lights should provide adequate means of tracking the target at the initiation of the transfer maneuver.

Orientation of the rendezvous phase was optimally located to present the most favorable lighting conditions for target acquisition and tracking, and use of the star background for measurements and braking. These considerations are a requirement for future missions.

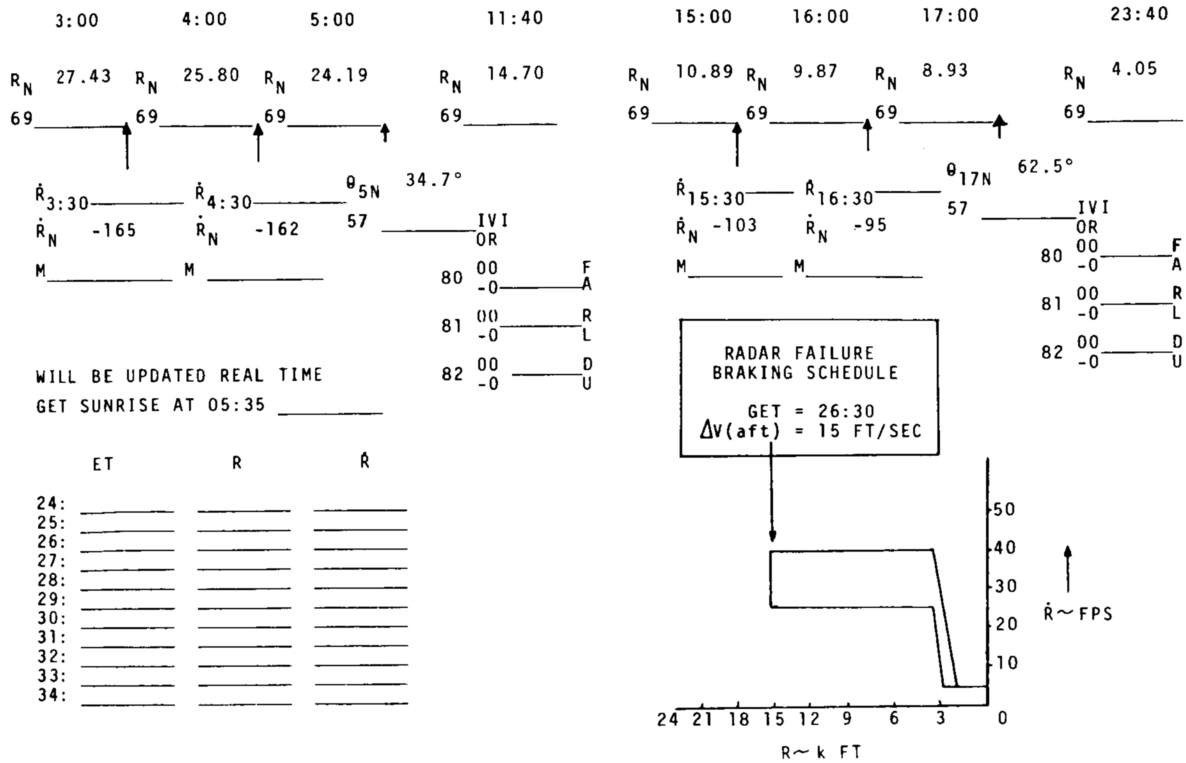


FIGURE 29-7.—Closed-loop intermediate correction monitor sheet.

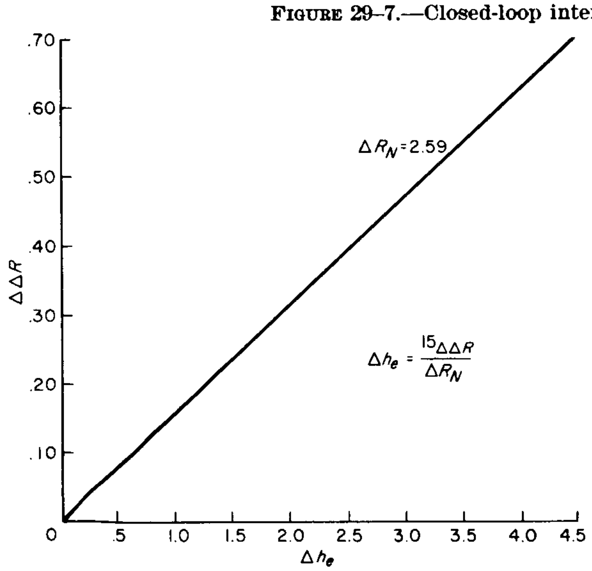


FIGURE 29-8.—Separation determination sheet.

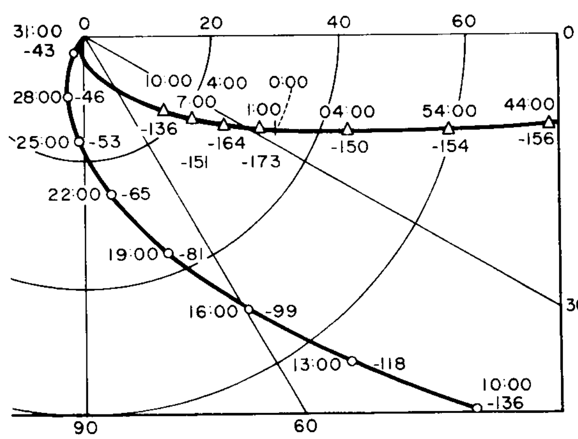
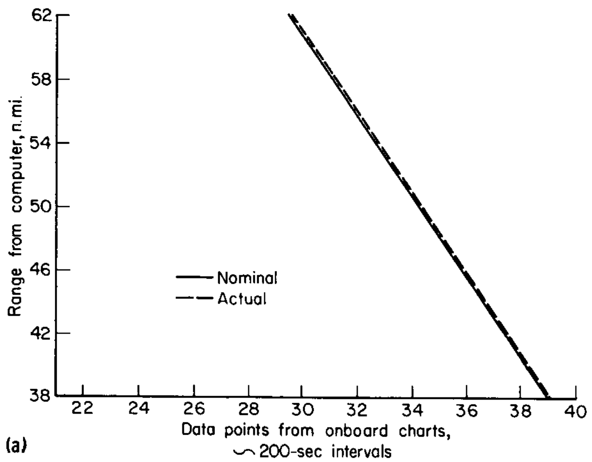
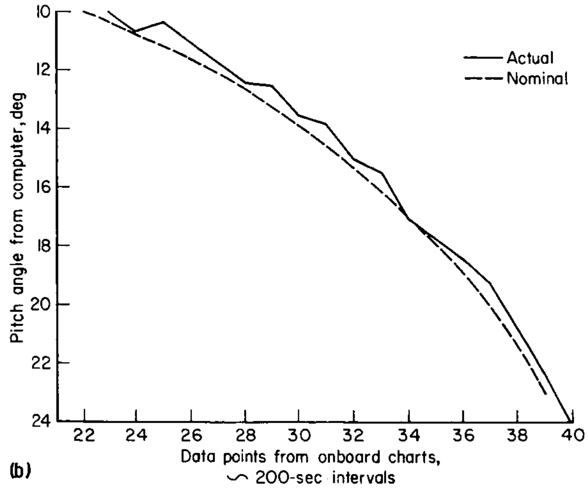


FIGURE 29-9.—Polar plot of nominal Gemini VI-A trajectory.



(*a*) Range versus time output.
FIGURE 29-10.—Gemini VI-A onboard data.



(*b*) Angle versus time computer output.
FIGURE 29-10.—Concluded.

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CONCLUDING REMARKS

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30. CONCLUDING REMARKS

By James C. Elms, *Deputy Associate Administrator for Manned Space Flight, NASA*

The preceding papers presented an interim report of the Gemini Program at its midpoint, and describe the objectives, designs, missions, and accomplishments to date—in short, a detailed report of a successful program. The major goal of the U.S. space program is to make this country conclusively and emphatically preeminent in space. The Nation is indeed proud of the Gemini Program's contributions, which include long-duration space flight, rendezvous, extravehicular activities, experiments, and the demonstration of active control of reentry to achieve a precise landing point. All the accomplishments have significantly contributed to the basic technology and to a better understanding of the space environment. These contributions will continue to be made throughout the remainder of the Gemini Program. The rapid increase in flight duration to 4 days, then 8 days, and finally 14 days, the extravehicular activities, the rapid turnaround, the accomplishment of major events on schedule in spite of adversity, all demonstrate the greatly increased capability of NASA, and are made even more meaningful by the policy of encouraging the world to observe the program. Much has been said about real-time flight planning, which has proved to be a requirement in the Gemini Program and which the Gemini team has been able to satisfy. The performance of the combined team of the Department of Defense, the contractors, NASA, and other Government agencies in planning and executing the Gemini VI–A and VII missions is an example of real-time management. This is a capability that will serve the Nation well in future missions. Gemini, in addition to being a giant step bridging the gap between Mercury and Apollo, is providing a means of program qualification for Apollo itself, and will continue to do so.

At the close of the Mercury Program, NASA had demonstrated that man could live in the weightless state for 1½ days, perform his job satisfactorily, and return unharmed. However, it is a long way from 1½ days to the 8 days required for the lunar trip. There were some optimists, not the least of whom were the astronauts themselves, but as recently as 1 year ago, diverse medical opinions existed as to the consequences of prolonged weightlessness, and many were greatly concerned. The Gemini Program produced the necessary evidence to prove that weightlessness would not be a limiting factor in the lunar program. As was discussed, the more sophisticated medical experiments which are planned for the remainder of the Gemini Program and for the Apollo Program will examine the total body system functions rather than simply gross postflight changes. This will provide necessary information regarding the possible effects of flights of much longer duration than the lunar landing mission.

The Gemini Program, because of the successful rendezvous mission, has also gone a long way toward removing the second constraint on the lunar landing program, that of rendezvous and docking. The successful rendezvous, as well as the long-duration flight, not only proved that man can survive weightlessness but demonstrated once and for all the vital role played by the astronauts in the performance of those missions. Because development of the rendezvous and docking techniques is of vital importance to the Apollo missions, subsequent Gemini flights are being tailored to simulate the constraints that will be imposed by the rendezvous of the lunar excursion module and the command and service modules in lunar orbit. The Gemini VII/VI–A rendezvous was conducted under ground direction in the initial phase, and by the crew using the onboard radar-computer system for the terminal phase. It has always been considered necessary to back up any rendezvous systems with optical techniques and equipment. In Apollo missions, where lives may depend upon successful rendezvous, the importance of simple reliable techniques cannot be overemphasized. Future Gemini missions will continue to evaluate these backup techniques. Several re-rendezvous and docking exercises on each mission will explore the relative effects of light and darkness as well as the effects of stars and earth background on vital acquisition and tracking of a rendezvous target. In spite of the great contributions already made to their program, the Apollo personnel are vitally interested in what will be learned in the remaining five Gemini missions.

What has Gemini contributed to other programs? An obvious example is the transfer of technology to the Manned Orbital Laboratory Program. This is a bit of reverse lend-lease to the Department of Defense as a partial repayment for the excellent support NASA has received and will continue to receive in the Gemini Program. In addition to Gemini's medical experiments, NASA has made a modest start in the development and performance of experiments and other disciplines. This has begun to stimulate the interest required to take full advantage of the capability of this program, and the Apollo Program which follows, to carry more advanced experiments.

Extravehicular activity has and will continue to increase our knowledge of man's ability to work in space outside the spacecraft itself. One result is the increased capability to perform useful experiments in space which will reduce the requirement for carrying equipment in the spacecraft or having it immediately available to the crew from inside the spacecraft. We can begin formulating plans for activities which will require resupply of personnel and life-support equipment or performance of maintenance on unmanned equipment.

NASA is halfway through the Gemini flight program. You have read a very optimistic series of presentations because the results have been excellent to date. In order to reach this halfway point in such an enthusiastic mood, NASA has had to solve many problems along the way. It cannot be overemphasized how hard this Gemini team has had to work to make it look so easy. You can be assured that it has not been a "piece of cake."

A word of general caution must be added in closing. The success of the manned space program to date is no guarantee in itself of future successes. As the Nation builds, step by step, the total capability in space, continued full support and even harder work than in the past will be required. A major setback could still require reassessment of the ability to meet goals on schedule. The Nation is now truly at the beginning of a major adventure in the exploration of space, but still has a long way to go.

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PART II

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31. EXPERIMENTS PROGRAM SUMMARY

By R. O. Piland, *Manager, Experiments Program Office, NASA Manned Spacecraft Center,* and P. R. Penrod, *Experiments Program Office, NASA Manned Spacecraft Center*

Introduction

The successful completion of the Mercury Program had shown without reservation that man can function ably as a pilot-engineer-experimenter for periods up to 34 hours in weightless flight. It was thus a primary objective of the Gemini Program to explore man's capabilities in an extension of these rules which would encompass both increased duration and complexity. Man's proved effectiveness as a scientific observer from the vantage point of orbital flight was amply supported by the capabilities of the Gemini spacecraft in the areas of scientific equipment accommodation, fuel budget and system for accurate attitude control, and habitability for extended missions. All of the above, in context with the planned mission profiles, afforded an unprecedented opportunity for the conduct of a comprehensive program of inflight experiments. From the very inception of the Gemini Program, therefore, there was a parallel and concerted effort by the National Aeronautics and Space Administration to seek out and foster the generation of suitable experiments from all sources. Among others, these would include educational institutions, varied U.S. Government agencies, NASA field centers, the Department of Defense, and industry laboratories.

The resultant complement of experiments included those of medical, scientific, and technological significance. The total program is summarized in table 31–I which shows, for each experiment, the numerical identification, name, principal investigator, principal-investigator organizational affiliation, and flights to date. It is noted that a total activity of 54 experimental efforts has so far been included in the flight program. By way of information, it is anticipated that the remainder of the Gemini Program (missions VIII through XII) will include some 56 experimental flight activities, which are similarly identified on table 31–I. Since final flight assignment has not been made, flight distribution is not shown.

It is also apparent that the concentration of experiments has been on the longer-duration missions. This, of course, is due to the inherent influence of time, which permits a larger data yield for time-sensitive parameters, repeated contacts with preselected subjects, and increased potential for objects of opportunity. Of major significance, however, was the increased crew time available for the operation of equipment and participation in experimental protocol. It should also be emphasized that planning on a programwide basis permits the scheduling of experiments on multiple flights if these additional data points with the associated continuity in time and procedures are particularly significant. Finally, more ambitious mission objectives such as crew extravehicular activities and rendezvous-and-docking permitted the programing of experiments which extend beyond the cabin confines of a single spacecraft, even beyond the limitations of a single mission.

Procedures

In order to most effectively take advantage of the capabilities described above, the procedures which are summarily defined here were employed.

Experiment proposals received were evaluated by NASA within the framework of the following major considerations:

(1) Scientific, technical, or biomedical merit
(2) Effect on safety of flight
(3) Extent of changes required to spacecraft
(4) Mission compatibility
(5) State of readiness and qualification of equipment
(6) Degree of crew participation
(7) Attitude-control fuel budget
(8) Weight and volume
(9) Instrumentation and electrical power

TABLE 31-I.—*Gemini Experiments*

[January 14, 1966]

Experiment number	Title	Principal investigator	Affiliation	Mission no.—					
				III	IV	V	VI	VII	VIII to XII
M-1	Cardiovascular conditioning	L. F. Dietlein	NASA-MSC			X		X	
M-3	Inflight exerciser	L. F. Dietlein	NASA-MSC		X	X		X	
M-4	Inflight phonocardiogram	L. F. Dietlein	NASA-MSC		X	X		X	
M-5	Bioassays body fluids	L. F. Dietlein	NASA-MSC					X	X
M-6	Bone demineralization	Pauline Mack	Texas Woman's University		X	X		X	
M-7	Calcium balance study	Whedon	National Institutes of Health					X	
M-8	Inflight sleep analysis	R. Adey and P. Kellaway	Baylor Medical School					X	X
M-9	Human otolith function	A. Graybiel	U.S. Navy			X		X	
MSC-1	Electrostatic charge	P. E. Lafferty	NASA-MSC		X	X			
MSC-2	Proton electron spectrometer	J. Marbach	NASA-MSC		X			X	
MSC-3	Tri-axis flux-gate magnetometer	W. D. Womack	NASA-MSC		X			X	X
MSC-4	Optical communication	D. Lilly	NASA-MSC					X	
MSC-5	Lunar UV spectral reflectance	R. C. Stokes	NASA-MSC						X
MSC-6	Beta spectrometer	J. Marbach	NASA-MSC						X
MSC-7	Bremsstrahlung spectrometer	R. S. Lindsey	NASA-MSC						X
MSC-8	Color patch photography	J. R. Brinkmann	NASA-MSC						X
MSC-10	Two-color earth's limb photography	M. Peterson	MIT		X				
MSC-12	Landmark contrast measurement	C. E. Manry	NASA-MSC					X	X
T-1	Reentry communications	L. C. Schroder	NASA-Langley	X					
T-2	Manual navigation sightings	D. Smith and B. Greer	NASA-Ames						X
D-1	Basic object photography	AF Avionics Lab	Wright-Patterson AFB			X			
D-2	Nearby object photography	AF Avionics Lab	Wright-Patterson AFB			X			
D-3	Mass determination	Air Force Field Office	NASA-MSC (DOD)						X
D-4	Celestial radiometry	AF Cambridge Lab	USAF-Hanscom Field			X		X	
D-5	Star occultation navigation	AF Avionics Lab	Wright-Patterson AFB					X	X
D-6	Surface photography	AF Avionics Lab	Wright-Patterson AFB			X			
D-7	Space object radiometry	AF Cambridge Lab	USAF-Hanscom Field			X		X	
D-8	Radiation in spacecraft	AF Weapons Lab	Kirkland AFB		X		X		
D-9	Simple navigation	AF Avionics Lab	Wright-Patterson AFB		X			X	
D-10	Ion-sensing attitude control	AF Cambridge Lab	USAF-Hanscom Field						X
D-12	Astronaut maneuvering unit	Air Force Field Office	NASA-MSC (DOD)						X
D-13	Astronaut visibility	S. Duntley	University of California			X		X	

D-14	UHF-VHF polarization	Naval Research Lab	U.S. Navy						X
D-15	Night image intensification	AF Avionics Lab	Wright-Patterson AFB						X
D-16	Power tool evaluation	Air Development Center	U.S. Navy						X
S-1	Zodiacal light photography	E. P. Ney	University of Minnesota			X			X
S-2	Sea urchin egg growth	R. S. Young	NASA-Ames	X					
S-3	Frog egg growth	R. S. Young	NASA-Ames						X
S-4	Radiation and zero g on blood	M. Bender	Atomic Energy Commission	X					X
S-5	Synoptic terrain photography	P. Lowman	NASA-Goddard		X	X	X	X	X
S-6	Synoptic weather photography	K. M. Nagler	U.S. Weather Bureau		X	X	X	X	X
S-7	Cloud top spectrometer	F. Saiedy	Natl. Environ. Sat. Center			X			X
S-8	Visual acuity	S. Duntley	University of California					X	
S-9	Nuclear emulsion	M. Shapiro and C. Fichtel	NRL and Goddard						
S-10	Agena micrometeorite collection	Dr. D. Hemenway	Dudley Observatory						X
S-11	Airglow horizon photography	H. Friedman	Naval Research Lab.						X
S-12	Micrometeorite collection	C. Hemenway	Dudley Observatory						X
S-13	UV astronomical camera	K. Henize	Dearborn Observatory						X
S-26	Ion wake measurement	D. Medved	Electro-Optical Systems, Inc.						X
	Total: 49 experiments			3	11	17	3	20	56

Having selected experiments which were in concert with the criteria in the above areas, the principal investigators for the proposed experiments were "contracted" by NASA to design, develop, qualify, and deliver flight equipment in accordance with the Gemini Program management and design criteria. Included also is the requirement to establish the necessary experiment protocol and support the preflight, flight, and postflight activities associated with the particular experiment.

Activities in the immediate preflight interval are variable and somewhat unique to the experiment. Crew familiarization with objectives and training in procedures are the responsibility of the principal investigators, and the principal investigator was required to define and assist as required in implementation. Similarly, where baseline data on crew physiological parameters are required, the principal investigator has an equivalent responsibility. Preparation and state of readiness of special ground targets or ground-located participating equipment is a principal-investigator task. Participation in final crew briefings, equipment checks, and NASA-sponsored press conferences is required.

During the flight, principal-investigator availability for consulting on real-time adjustment of experimental procedures is essential. Also, the manning and operation of ground targets and participating equipment sites are required.

Postflight activities include participation in the scientific debriefing of the crew. A summary compilation of experimental results is required for incorporation in the mission report during the immediate postflight interval. It is NASA policy to sponsor, within 90 days after flight, a public report of the experimental results in the degree of reduction and analysis that exists at the time. A final publication of results is required when data analysis is complete and conclusions are firmly established.

Summary Results

The results of the experiments included in the Gemini VI–A and VII missions that had a significant data yield will be reported in detail by the respective principal investigators later in this series of papers. In the cases where those experiments had flown previously, the total experimental results will be reflected. The results of experiments included on previous missions which were not included on VI–A and VII have been reported previously by the principal investigators but will be summarily reviewed here. References 1 and 2 contain experiment evaluations for the Gemini III, IV, and V missions, respectively. (A complete listing of reference material used by the principal investigators in the publication of their results is not repeated here but is concurrently recognized.)

The following synopsis is derived, for the most part, from the above references. It is emphasized that some of the results are tentative. In some cases the experimenters have not completed their analysis of the data. Moreover, a number of the experiments are repeated on several missions, and the total experiment is not complete until all missions have been conducted and the results correlated and analyzed.

S–1 Zodiacal Light Photography

Data from the Mercury Program had shown conclusively that experiments on extraterrestrial light could be performed above 90 kilometers without airglow contamination. The S–1 experiment flown on the Gemini V mission, then, was to address the following questions:

(a) What is the minimum angle from the sun at which the zodiacal light could be studied without twilight interference?

(b) Can the gegenschein be detected and measured above the airglow layer?

The experiment was successfully completed, and it demonstrated that approximately 16° is the smallest elongation angle at which zodiacal light may be studied without external occulting. Photographic results appear to show the gegenschein, the first time such efforts have been successful. Its center appears to have an angular size of about 10° and is within a very few degrees of the anti-sun direction. There is no evidence of the westerly displacement which might be expected if the phenomena resulted from a cometlike dust tail of the earth.

This single set of data (ref. 1) is interesting but does not establish firm conclusions, especially with respect to the source of the gegenschein. The experiment is to be flown on subsequent Gemini missions for additional data on these two, plus other dim light phenomena.

S-2 Sea Urchin Egg Growth

The objective of the S-2 experiment was to evaluate the effects of subgravity fields on fertilization, cell division, differentiation, and growth of a relatively simple biological system.

Inasmuch as the experimental results were negated by a mechanical failure of the inflight equipment, equipment description and experimental protocol are not included in detail.

S-4 Zero G and Radiation Effects on Blood

Biological effects of the types usually associated with radiation damage have been observed following space flight. These effects include mutation, production of chromosome aberrations, and cell killing. This could be due to either or both of two things: effects of the heavy-primaries component of radiation which is not available for test in terrestrial laboratories, or synergistic interaction between radiation and "weightlessness" or other space flight parameters. The S-4 experiment was to explore such possibilities.

The procedure was to irradiate a thoroughly studied biological material with a known quality and quantity of radiation during the zero-g phase of flight. This, with concurrent and equivalent irradiation of a duplicate ground-located control sample, would yield a comparative set of data and would be evidence of synergism, if it existed, between the radiation administered and some space flight parameter. Since chromosomal aberration is one of the best known effects of radiation, it was selected as a suitable response for the study.

The equipment operated properly, and the experimental procedures were successfully completed (ref. 2). The lack of aberrations in the postflight blood samples from the crew makes the possibility of residual effects of radiation encountered on such a space flight very unlikely, at least on genetic systems. The yield of single-break aberrations (deletions) for the inflight sample was roughly twice that seen in the ground control and previous samples. All physical evidence contradicts the possibility of variant radiation doses to the ground control and flight samples. It appears then that some space-flight parameter does interact synergistically with radiation. Although this effect is not large from the point of view of radiation cytogenics, it is of interest. Further experiments will be necessary in order to confirm the synergistic effect and to determine just which space-flight parameter or parameters are involved, as well as the mechanism of the action.

S-7 Cloud-Top Spectrometry

Tiros weather satellites have provided meteorologists with information on geographic distribution of cloudiness and a qualitative indication of cloud types. Meteorologists are further interested in cloud altitudes because altitude is indicative of the dynamic and thermodynamic state of the atmosphere on which weather forecasts are based. Basically, the method of the S-7 experiment consists of comparing the cloud's radiance in the oxygen A-band at 7600 angstroms (Å), with its radiance in an atmospheric window outside the band. The ratio will show the absorption or transmission of oxygen in the atmosphere above the cloud top.

The objective of the experiment was to test the feasibility of measuring cloud altitude by this method. As a correlation and calibration technique, concurrent cloud-top measurement by civilian and military aircraft was programed.

During the flight of Gemini V, 26 spectrographic observations were obtained on various cloud types, some for low clouds over the west coast of Baja California, some for relatively high clouds on a tropical storm in the Eastern Pacific, and some for tropical storm Doreen. From the data yield, it is quite apparent, qualitatively, that transmission in the oxygen band for high clouds is much larger than that for low clouds. The results (ref. 1) prove the feasibility of the cloud-altitude measurement from a spacecraft by this method. Already, system design requirements are being formulated for a more sophisticated second-generation weather satellite instrument.

D-1 Basic Object Photography, D-2 Nearby Object Photography, D-6 Surface Photography

The purpose of Experiments D-1, D-2, and D-6 was to investigate man's ability to acquire, track, and photograph objects in space and objects on the ground from earth orbit. These three experiments used the same equipment, and the experiment numbers primarily designate the type of object which served as the aiming point. In D-1 the aiming points were celestial bodies and the rendezvous evaluation pod (REP) at relatively long photographic range. The D–2 designated the short-range tracking and photographing of the REP, and the D–6 aiming points were objects on the ground.

Since investigation of acquisition and tracking techniques was the primary objective of these experiments, two acquisition modes and three tracking modes were employed using commercially available equipment.

On the Gemini V flight (ref. 1), D–1 was accomplished using celestial bodies as aiming points. Distant photography of the REP, however, was not possible because of spacecraft electrical-power difficulties which developed after REP ejection. The planned D–2 close-range photography of the REP was not possible for the same reason. The D–6 terrestrial photography was accomplished within the limitations dictated by weather conditions and by spacecraft electrical power and thruster conditions. The photographs obtained were significant only as an element of the data to be used in the evaluation of techniques. The other elements of data were time-correlated position and pointing information, atmospheric conditions, sun angle, exposure settings, and astronauts' flight logs and verbal comments.

D–5 Star Occultation Navigation

The objectives of the D–5 experiment were to determine the usefulness of star occultation measurements for space navigation, and to determine a horizon density profile to update atmospheric models for horizon-based measurement systems.

Knowledge of the time of occultation of a known star by a celestial body, as seen by an orbiting observer, determines a cylinder of position whose axis is the line through the star and the body center, and whose radius is equal to the occulting body radius. The times of six occultations provide information to uniquely determine all orbital parameters of the orbiting body. Determination of these times of occultation by the earth is difficult because of atmospheric attenuation of the star light. The star does not arbitrarily disappear but dims gradually into the horizon. Measurement of the percentage of dimming with respect to the altitude of this grazing ray from the star to the observer provides a percentage altitude for occultation. That is, the star can be assumed to be occulted when it reaches a predetermined percentage of its unattenuated value. The procedure for the D–5 experiment provides the means of measuring this attenuation with respect to time in order to determine the usefulness of the measurements for autonomous space navigation. In addition, the measurements would provide a density profile of the atmosphere which could be used to update the atmospheric model for this system and to refine models used for other forms of horizon-based navigation, orbit prediction, and missile launches.

Results of this experiment were negative due to a malfunction of the experimental hardware. A postflight analysis identified the source of failure. Corrective action has been implemented, and the experiment will be flown again later in the program.

D–8 Radiation in the Gemini Spacecraft

Prerequisite to successful completion of future manned-space-mission planning is the availability of data on the radiation environment and its shielding interactions. The D–8 experiment was for the purpose of gaining reliable empirical dosimetry data to support the above activities.

The quantitative and qualitative characterizations of the radiation levels associated with the Gemini mission originated, in the main, with those energetic protons and electrons present in the inner Van Allen belt and encountered each time the spacecraft passed over the South Atlantic Anomaly.

Instrumentation consisted of both active and passive dosimetry systems. The active instrument included tissue-equivalent chambers with response characteristics which match closely that of soft muscle. An active sensor was placed in a fixed location in the spacecraft, and another portable unit was used for survey purposes. Meticulous calibration of the instruments and inflight adherence to experimental protocol lend confidence in the validity of results (ref. 2). The average dose rate for all "non-anomaly" revolutions analyzed was found to be 0.15 millirad per hour.

Dose-rate data obtained from the South Atlantic Anomaly region shows a rapid and pronounced rise in magnitude over the cosmic levels; that is, rises of two orders in magnitude, or to more than 100 millirads per hour average. This is associated with an average "anomaly" transit time of 12 minutes.

The five passive dosimetry packages were to ascertain both total accumulated dose and the intensity of radiation causing it. They were located in areas of maximum, minimum, and intermediate shielding. Preflight investigation of the extraneous effects of onboard sources revealed this to be less than 1 millirad per day; therefore, all recorded data could be considered cosmic in nature.

There was a very good correlation between the integrated dose readings from the active and the passive dosimeters located in the same area. The difference was only 12 percent for the discharge ionization chamber. The variations that do exist are for known reasons, which will permit generation of suitable correction factors for the passive devices so that they can provide a reliable assessment of radiation dose on future missions.

D–9 Simple Navigation

The objective of the D-9 experiment was to demonstrate the utility of a technique for manual navigation during space flight. Considerable efforts prior to flight had been devoted to reducing the very complex orbital determination mathematics to a rather simple model which could be exercised by the use of tables or a simple handheld analog computer. The solution derived consisted of dividing the normally used six-degree-of-freedom analysis into two separate and distinct three-degree-of-freedom problems. The first would determine the size and shape of the orbit, and the second would yield in-orbit orientation. All of the data to support these calculations could be derived using a simple handheld sextant for making the necessary celestial and horizon observations.

The role this experiment has in the program is simple procedures and technique development. The equipment and experimental protocol have been reported previously and are described in reference 1. A detailed accounting of the sightings made is not included here, but on both Gemini IV and VII the procedures were successfully completed, the data yield was up to expectations, and only detailed analysis is required to arrive at the final conclusion. In summary, the basic concept was demonstrated to be feasible; however, the stability of the observables, specifically horizon determination on which system accuracy depends, needs further investigation.

MSC–1 Electrostatic Charge

The objective of the MSC-1 experiment was to establish a definition of the electrostatic potential on an orbiting Gemini spacecraft. This would permit calculation of the energy available for an electrical discharge between the Gemini spacecraft and another space vehicle.

The field readings on Gemini IV (ref. 2) were extremely large compared with what was expected; however, the data gave no reason to suspect any electrical or mechanical malfunction of the equipment. Investigations were initiated to determine whether the apparent electric field was due to some cause other than a true field at the surface of the spacecraft. A test series confirmed that the instrument was responsive to radiated radiofrequency energy and to charged plasma-current particles. The Gemini V instrument was modified to shield the sensor from electric fields terminating on the spacecraft. However, readings obtained on Gemini V were as high as those from Gemini IV. Investigations are continuing to identify the extraneous source of sensor stimuli. One hypothesis which is supported from a number of standpoints is enhanced ionospheric charged-particle concentrations resulting from outgassing of the spacecraft. Correlation with day/night cycle (thermal gradients), operation of the water boiler, fuel-cell purging, and mission time profile lends emphasis to this.

MSC–4 Optical Communications

The objectives of the MSC-4 experiment were to evaluate an optical communications system, to evaluate the crew as a pointing element, and to probe the atmosphere using an optical coherent radiator outside the atmosphere.

Inasmuch as unfavorable cloud conditions and operating difficulties for ground-based equipment all but negated a data yield, no significant discussion is included here. It was shown, however, that the laser beacon is visible at orbital altitudes, and static tests have shown that adequate signal-to-noise ratios can be obtained.

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MSC–10 Two-Color, Earth Limb Photography

The plans for guidance and navigation for the Apollo mission require observation of the earth, potentially its limb, in order to make a navigational fix. In this case, a precise definition of the observable limb is essential. The uncertain state of the lower atmosphere, with its tropospheric storms and the accompanying clouds, prompts a consideration of observing higher levels of the atmosphere that have a satisfactory predictability.

On the Gemini IV earth limb photographs, primary attention was given to the comparison of the terrestrial elevation of the blue above the red portion of each photographed limb. The profiles of the blue are more regular than the red in their brighter parts. Comparative values of the peak radiances, blue and red, of the limbs vary by nearly 50 percent. This is preliminary, and work still remains to evaluate the densitometric photography data in order to judge the validity of scattering theory to account for the blue limb profiles. (Detailed accounting is included in ref. 2.)

MSC–12 Landmark Contrast Measurement

The objective of the MSC–12 experiment was to measure the visual contrast of landmarks against their surroundings. These data were to be compared to calculated values of landmark contrast in order to determine the relative visibility of these landmarks when viewed from outside the atmosphere. The landmarks are potentially a source of data for the onboard Apollo guidance and navigation equipment.

This experiment depended on photometric data to be obtained by the photometer included in the D–5 equipment complement. As noted earlier, a malfunction of the photometer was experienced, which negated a data yield from this experiment.

T–1 Reentry Communication

The T–1 experiment was conducted during the Gemini III mission to determine whether water injection into the flow field around the spacecraft is effective in maintaining communications links during the reentry portion of the flight.

Attenuation levels were measured at ultra high frequency (UHF) and C-band frequencies with and without water injection. UHF signals which had been blacked out were restored to significant levels by high flow rate injection. The C-band signal was enhanced by medium to high flow rates. The recovered UHF signal exhibited an antenna pattern beamed in the radial direction of injection from the spacecraft. Postflight analysis shows that the UHF recovery agrees very well with injection penetration theory. More optimum antenna locations and injection sites should minimize the problem of resultant signal directionality. (Ref. 1 contains a detailed report.)

Conclusion

It is felt that the inflight experiments completed to date have been very successful and clearly indicate the desirability of fully exploiting the capabilities of subsequent spacecraft designs and missions for the conduct of an experiments program. Accordingly, the following programs are in effect:

(1) The remainder of the Gemini Program will reflect a continued emphasis on the conduct of inflight experiments. Certain of these will be an extension of a series which has already begun on missions III through VII. Others will be introduced as new experiments, some of which are of considerably increased complexity. As noted earlier, some 56 experimental activities are included.

(2) A series of experiments is being incorporated in Apollo earth-orbital flights.

(3) A lunar-surface experiments package is being developed for deployment on the lunar surface during a lunar-landing mission.

(4) An experiments pallet for Apollo service module accommodation of a heavier, more sophisticated payload is being developed.

(5) An extensive airplane flight-test program for remote-sensor development has been developed.

The results of these and similar programs should contribute immeasurably to the related technologies as well as to the basic and applied sciences.

References

1. Manned Space Flight Experiments Symposium, Gemini Missions III and IV, Oct. 18, 1965. (NASA publication.)

2. Manned Space-Flight Experiments Interim Report, Gemini V Mission, Jan. 6, 1966. (NASA publication.)

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PHYSICAL SCIENCE EXPERIMENTS

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32. GEOASTRONOMICAL OBSERVATIONS

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Introduction and Summary

The manned Mercury orbital flights conducted from February 6, 1962, to May 16, 1963, established the following general features through visual observations by the astronauts:

(1) The night airglow band, centered some 90 kilometers above the earth, is visible at all times on the nightside of the earth. Visual measurements were made of the altitude, width, and luminance of the airglow (ref. 1) and were confirmed by rocket observations.

(2) As seen through the spacecraft window, the faintest stars observed at night, even under relatively ideal conditions, were described as of the fifth magnitude.

(3) With no moon, the earth's horizon is visible to the dark-adapted eye. The earth's surface is somewhat darker than the space just above it, which is filled with the diffuse light of airglow, zodiacal light, integrated starlight, and resolved stars.

(4) With the aid of starlight but no moon, zodiacal light, airglow, clouds, and coastlines are just visible to the dark-adapted eye.

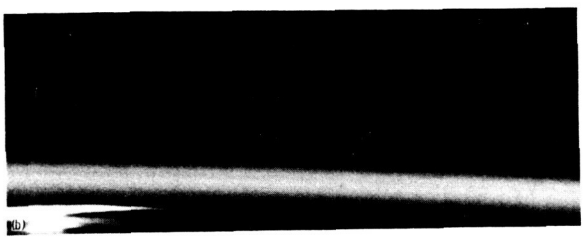
(5) With moonlight reflected on the earth, the horizon is still clearly defined, but, in this case, the earth is brighter than the background of space. Indeed, with moonlight, the clouds can be seen rather clearly, and their motion is distinct enough to provide a clue to the direction of the motion of the spacecraft.

(6) The night sky (other than in the vicinity of the airglow band and horizon) appears quite black, with the stars as well-defined points of light which do not twinkle. Lights on the earth do twinkle when viewed from above the atmosphere.

(7) The zodiacal light was successfully observed by Cooper in the last of the Mercury flights but was not seen during the previous Mercury flights, presumably because of the cabin lights which could not then be extinguished.

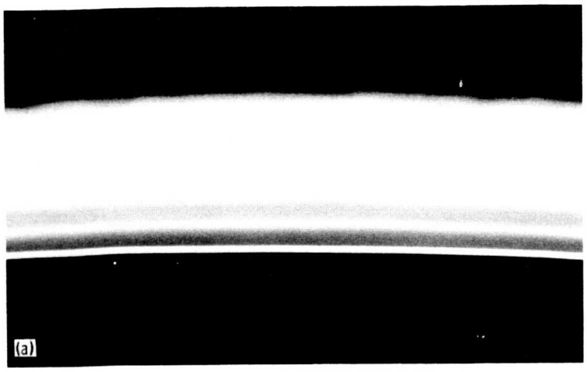
(8) A "high airglow" was observed on one occasion on the nightside by both Schirra and Cooper. Schirra described this as a brownish "smog-appearing" patch which he felt was higher and wider than the normal nightglow layer. Schirra observed this patch while over the Indian Ocean, and Cooper while over South America. It is possible that this phenomenon may have been a tropical 6300 angstroms (Å) atomic oxygen emission, first reported by Barbier and others (ref. 2).

(9) Twilight is characterized by a brilliant, banded, multicolored arc which exists along the horizon in both directions from the position of the sun. On MA-8, during twilight an observation was made, for the first time, of a very remarkable scene. The scene is shown in figure 32-1(a), which is a black-and-white reproduction of a color painting. The painting was made from Schirra's description (refs. 3 and 4) of a series of blue bands. Figure 32-1(b) is a black-



(*a*) Painting made from a MA-8 description of blue bands.

FIGURE 32-1.—Banding in the twilight horizon zone.



(*b*) Print from 16-mm color film exposed on Gemini IV.
FIGURE 32–1.—Concluded.

and-white reproduction of one of many frames of color, 16-mm movie film taken by McDivitt and White during Gemini IV. These color photographs were the first physical proof of the bands seen by Schirra, which had also been visually observed by Cooper during MA–9 (ref. 4).

(10) Finally, during the Mercury flights, the following phenomena were not observed:

(a) Vertical structure in the nightglow
(b) Polar auroras
(c) Meteors
(d) Comets

From the Gemini flights, additional information was derived which included:

(1) Specific information on day and night star sightings.

(2) Observations of aurora australis from Gemini IV and VII.

(3) Meteors were first observed by the Gemini IV crew and again by the Gemini VII crew.

(4) Vertical structure in the night airglow was first observed and noted in the logbook by Gemini IV crewmen.

In the following sections, more detailed discussions of these observations are given.

Observation of Stars

Nighttime

Information on star sightings at nighttime from the Gemini spacecraft indicates that, on the average, crews can generally observe stars slightly fainter than the sixth magnitude. The most objective evidence of this to date was reported by the Gemini VI–A and VII crews

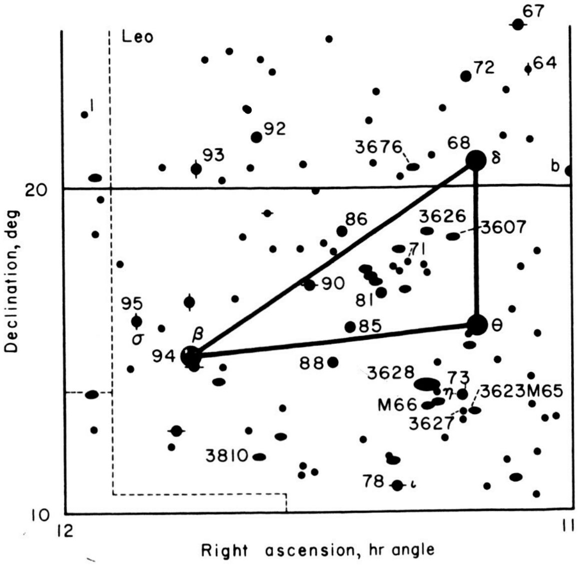


FIGURE 32–2.—Data on nighttime star observations by the Gemini VI–A flight crew.

through simple tests. Both Gemini VI–A crewmembers counted the number of stars they could see within the triangle Denebola and δ and *θ* Leonis shown in figure 32–2. The command pilot reported seeing two stars, and the pilot saw three. Referring to figure 32–2, this report indicates that at the moment of observation the command pilot could see to a magnitude between 6.00 and 6.05, while the pilot could see to a value greater than 6.05. Figure 32–3 is a test card, carried aboard the Gemini VII space-

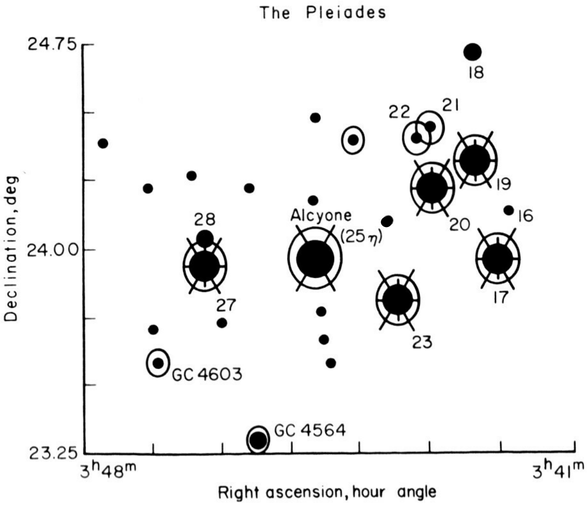


FIGURE 32–3.—Data on nighttime star observations by the Gemini VII flight crew.

craft, showing the area of the Pleiades with the crew's markings of observed stars. For purposes of this report, the stars shown here are identified in more detail than on the original card used by the crew so that a comparison can be made between the crew's markings and the accompanying list of identified stars and their magnitudes. The command pilot observed stars down to magnitudes in the range of 6.26 to 6.75, while the pilot could see to at least 4.37. Except for the pilot's observation, these compare well with less objective, but nevertheless important, sightings by the Gemini IV crew who carried a card showing the relative locations and magnitudes of stars in more than five well-known constellations in their nighttime sky. The constellation Corona Australis provided the most stringent test, with stars identified down to 5.95 magnitude. Both members of the crew reported that they could easily see all the stars on their card as well as fainter stars, whose brightness they estimated to be in the order of the seventh magnitude. All crews have made subjective comment that the number of nighttime stars seen from the spacecraft was greater than the number seen from their ground-based observations, and about the same or perhaps a little more than from a high-flying jet aircraft. The reports varied within this range from individual to individual during scientific debriefings of Gemini flight crews.

In the interest of accuracy and precision, it must be noted that even the best of these reported tests contain some subjectivity. A vigorous analysis of these results is simply not possible because of the many unknowns that have a great bearing on the results. Therefore, it seems appropriate at this time to briefly review the variable parameters whose value and/or constancy must be assumed in the absence of precise supporting data on values and on test procedures.

The end instrument in these tests is the human eye itself—a device whose extreme adaptability and whose variability makes its response characterization very difficult to ascertain. The subjectivity of results is also reinforced by the psychophysical nature of studies in vision.

Figure 32-4 shows a collection (refs. 5 and 6) of relationships which have a bearing on nighttime vision. Precise experiments concerning brightness sensitivity required a detailed knowledge of such parameters as—

(1) Retinal position of the image.

(2) Contrast between point source image and background.

(3) Degree of dark adaptation.

(4) Duration of point source exposure.

(5) Relative movement of the image (induced by subject or spacecraft).

(6) Color or hue of the image.

In most cases these parameters are composite functions that can be divided into even more detailed variables.

Several purely physical parameters associated with sightings from the Gemini spacecraft also have a great bearing on the end results. The effect of the transmission, absorption, and scattering of light as it passes through the triple-layered windowpanes is not completely known. In addition, each crewman has noted deposits on the spacecraft window, primarily on the outermost of the six surfaces. These deposits can be greatly restrictive to vision. Astronaut Lovell's results, which were two star magnitudes fainter than his associate's, are tentatively accredited to a more severe case of material deposition. Although the effect of this on light transmission—so important when dealing with very low light levels—is not known, its effect of light scattering during Gemini V and VII has been well documented by the visual acuity experimenters in section 34 of this report. However, during the nighttime the fraction of interior spacecraft light scattered and reflected into the crewmen's line of vision can present the most significant degradation to seeing, even with bright moonlight (either direct or reflected from the earth) incident on the heavily coated outer window surfaces. The problem of undesirable internal light, which is sometimes unavoidable for operational reasons, is clearly shown in figure 32-5. This is a nighttime photograph of the moon taken as part of the Gemini VII Dim Light Study reported separately. Although full information is not yet available, it should be noted that the photograph is a time exposure with the light inte-

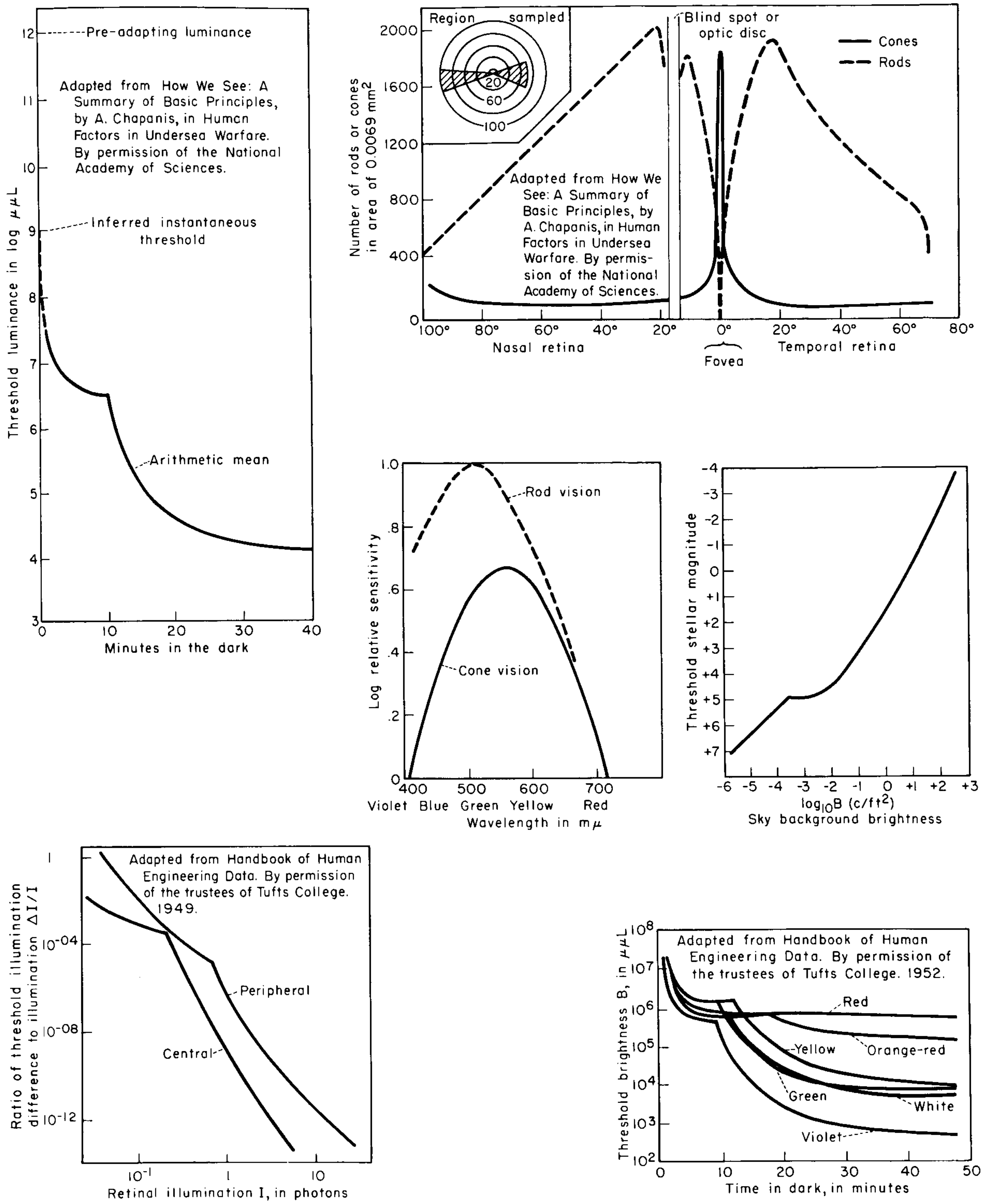


FIGURE 32-4.—Collection of important parameters in vision.

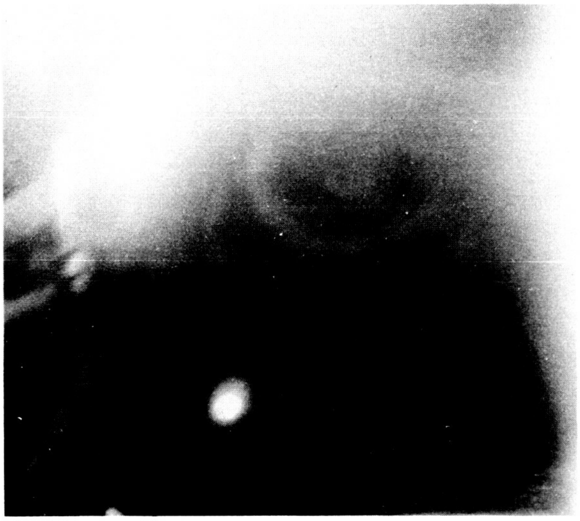


FIGURE 32-5.—Time exposure of moon with scattering and internal light reflections.

grated over several seconds. Thus, it does not necessarily represent the visual scene that would be apparent to the crew, but does exemplify a limiting factor in nighttime star observations by contrast reduction and interference with the low level of dark adaptation required.

Daytime

The sighting of stars in the daytime (when the sun is above the horizon as viewed from the spacecraft) has been difficult. Most of the difficulty comes from scattered sunlight and earthlight on the spacecraft window. Even sunlight or earthlight illuminating the interior of the spacecraft through the window other than the viewing window (in the shade) makes visual observations of stars difficult, if not impossible.

Stars were definitely observed in daylight in several instances. Two of these occurred in Gemini V and VI-A. In a paper being prepared by E. P. Ney, W. F. Huch, C. Conrad, and L. G. Cooper, evidence is given that first and second magnitude stars were seen in the daytime sky. This occurred when proper precautions were taken during the performance of the S-1 experiment.

In a paper under preparation by D. F. Grimm, W. M. Schirra, and T. P. Stafford, the sightings of stars in the daytime prior to and during rendezvous exercises are analyzed. Briefly, from the data on the observations of various stars in Orion, it is concluded that Schirra was able to see stars as faint as the fourth magnitude. This is deduced from his observation of several stars in the Sword of Orion. The subject of visibility of stars and planets during twilight has been treated comprehensively by Tousey and Koomen (ref. 5). As a result of that work, the current analyses from the Gemini flights, and from future flights where photometric observations are made simultaneously with visual observations of known stars, a rather complete analysis will be possible.

Observations of the Aurora Australis

The fact that the Mercury and Gemini orbits have been confined within geographic latitudes of about ±32° means that observation of the polar aurora should be infrequent. The zone where auroras are most frequently observed is some 23° from the geomagnetic pole, thus at a geomagnetic latitude of about 67°. The fact that the geomagnetic pole is approximately 11° from the geographic pole means that the auroral zone occurs at geographic latitudes in the range of 56° to 78°. The dip of the horizon from the spacecraft is significant—for example, about 17° for a spacecraft 150 nautical miles (278 kilometers) above the earth's surface. Thus, a spacecraft at such a height, at its extreme geographic latitude, affords line-of-sight visibility to the apparent horizon to 49° geographic latitude, only 7° from the auroral zone.

The auroral zone is not "well behaved" and actually affords a more favorable circumstance for spacecraft auroral observation than the preceding general discussion implies. Just to the south of western Australia (fig. 32-6), the auroral zone comes as far north as 51° S, which means that the southern horizon for a spacecraft at 150 nautical miles in this region, namely 49° S, is only about 2° from the auroral zone. It is well to recall that auroras, though they statistically occur more frequently in the auroral zone, do not occur exclusively in this region. Furthermore, the location of the auroral zone moves toward the equator during periods of geomagnetic activity. During times of geomagnetic storms, auroras become visible very far from the so-called auroral zone, and are even

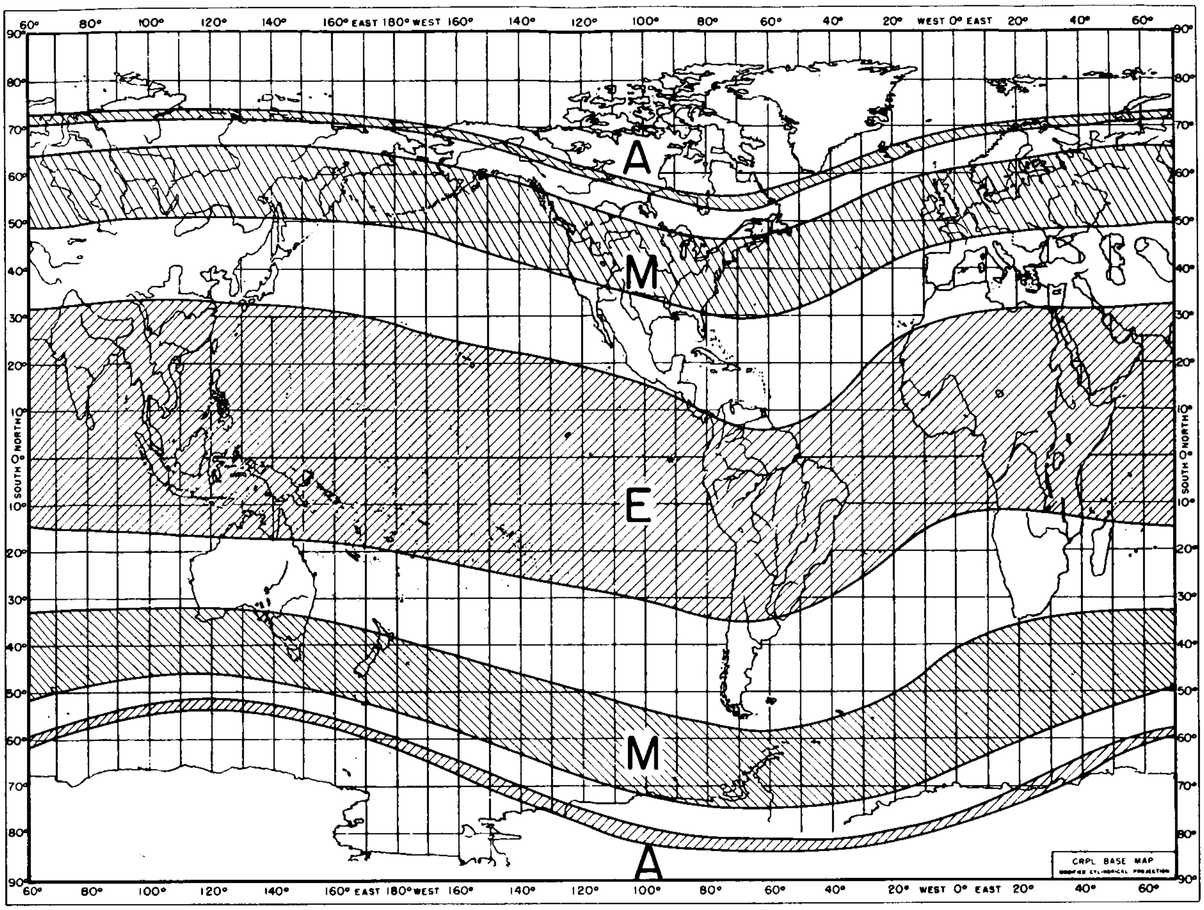


FIGURE 32-6.—Auroral map as seen from earth.

seen in the southern parts of the United States. The significant point in this discussion is that for the Gemini flights the combination of circumstances favors the observation of auroras to the south of the Australia region. The favorable factors for auroral observation are: (1) the apogee is near the southern extreme latitude, thus giving the maximum dip of the horizon; (2) the orbits are such that the spacecraft nights occur at longitudes near the general longitude of Australia; and (3) the southern auroral zone has its most equatorward excursion just south of Australia.

This report includes data from three separate flights in which auroral sightings to the south of Australia were noted by astronauts. During the Gemini IV flight, McDivitt and White saw an aurora in the form of auroral sheets projected against the earth. (See ref. 4, pp. 4 and 5, for a general description of what they saw.) Specifically, on June 4, 1965, at 17:24:37 Greenwich mean time (G.m.t.), at a spacecraft altitude of 151.41 nautical miles, at $-31.89°$ geocentric latitude, $-32.06°$ geodetic latitude, and 104.19° longitude, and with dip-of-horizon of $-16.75°$, the latitude of the southern horizon is $-48.81°$, very close to the best observing latitude in this region. Concerning this sighting, Astronaut White notes "the unusual display (June 4, 1965, 17 h. 24 m.) of night airglow combined with some northern-lights-type effect. The airglow looks lit up way out on the horizon." Some "spacecraft nights" later, McDivitt remarks:

I see the same sort of curve of lights like the northern lights except they are below us. I saw them another time. They were great big long lines . . . looks like arcs parallel to direction of flight path, and they extend from just below the airglow in the earth's horizon up a little past the top of the airglow, the same thing I saw the other night except not quite as bright as it was then.

The crew of Gemini V described a similar phenomenon in the same general location. During the 2-week flight of Gemini VII, the crewmen made a sketch of an auroral arc which was well defined between their apparent horizon and the airglow layer. Their sketch is reproduced as figure 32–7.

Meteors

A brief comment on the astronauts' meteor observations made during the early Gemini flights is given in reference 4. That Gemini V had the expectation of seeing a good many meteors can be seen from the Hourly Plots of Meteor Counts for July and August 1965 (fig. 32–8; also see ref. 7). Actually, the August meteors show more than a tenfold increase over the rest of the year. The crew's estimate of the number seen during the Gemini V flight is given in table 32–I. A much smaller number of meteors was observed during the flights of Gemini VII and VI–A (see table 32–I). This was expected, as shown in figure 32–9 (also see ref. 9), since the number of December meteors is greatly reduced as compared with the peak for the year, which occurs in August.

The number of meteors seen by the crew is a function of a number of factors, including the time interval in which they are observing (which may or may not include the actual peak of a shower), the nature of the Gemini window (their approximate angle of view is 50°), and the condition of that window (which will determine the limiting magnitude of the meteors seen). The Gemini VII pilot reported that his window was smudged, probably due to the staging process. Thus, only the bright meteors, within the rather small angle of view afforded by the spacecraft window, would catch the pilot's attention. So it is not surprising that so few meteors were reported during Gemini VII in spite of the pilot's attention to specific observation of them. Observation of meteors during Gemini VI–A was very much a chance

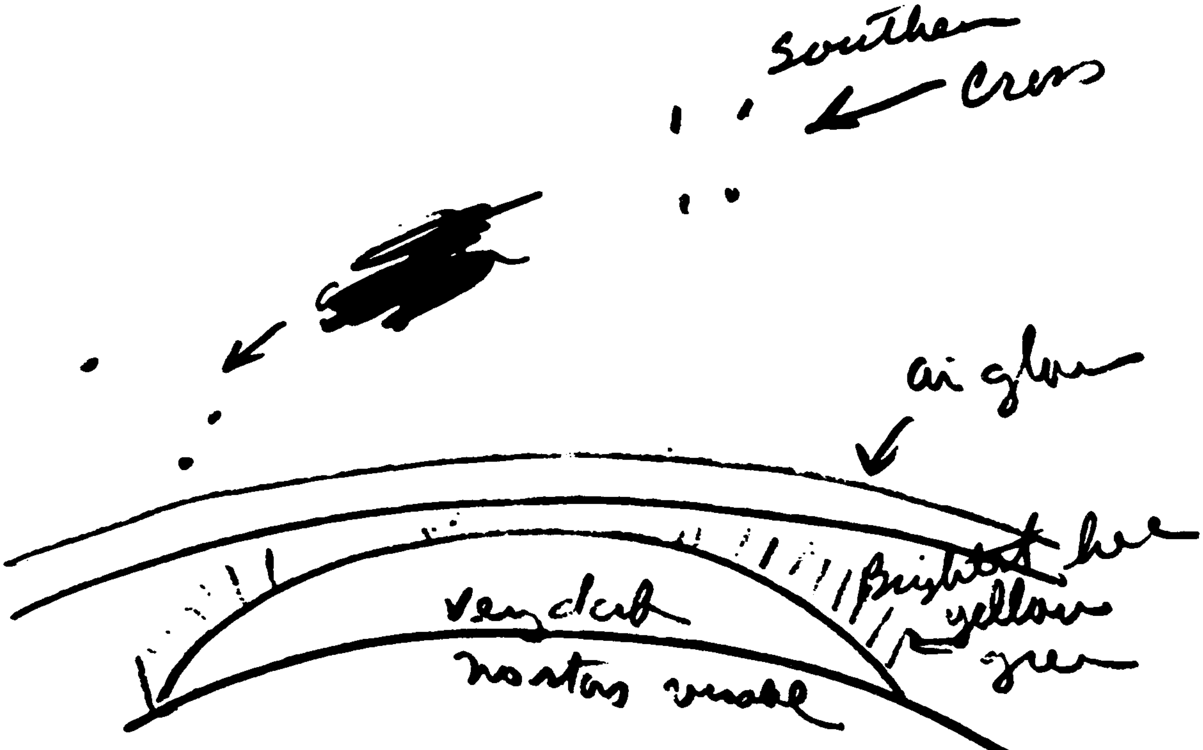


FIGURE 32–7.—Auroral arc as sketched by Gemini VII crewmen.

TABLE 32–I.—*Meteors Observed During Gemini Flights*

Flight no.	Date of flight (1965)	Duration	Phase of moon	Meteor shower [a]	Approximate date of maximum of shower [a]	Count reported by crew
III	Mar. 23	9 hr	Last quarter, Mar. 25			None
IV	June 3–7	4 days	First quarter, June 6			Many (no number given)
V	Aug. 21–28	8 days	Last quarter, Aug. 20	Perseids	Aug. 10 (Aug. 9–14) [b]	Numerous (20/hr estimated) [c]
VII	Dec. 4–18	14 days	First quarter, Dec. 1; last quarter, Dec. 15	Geminids	Dec. 11, 12 (Dec. 9–12)	3 total; [d] 1 in 30-minute observation interval
VI–A	Dec. 15	24 hr	Last quarter, Dec. 15	Geminids		1 fireball

[a] See ref. 8.
[b] See ref. 9.
[c] The times of observation of 5 or more meteors are recorded on the onboard tape. Several of these were noted at the same time as lightning flashes.
[d] From the pilot's description, these were probably Geminids.

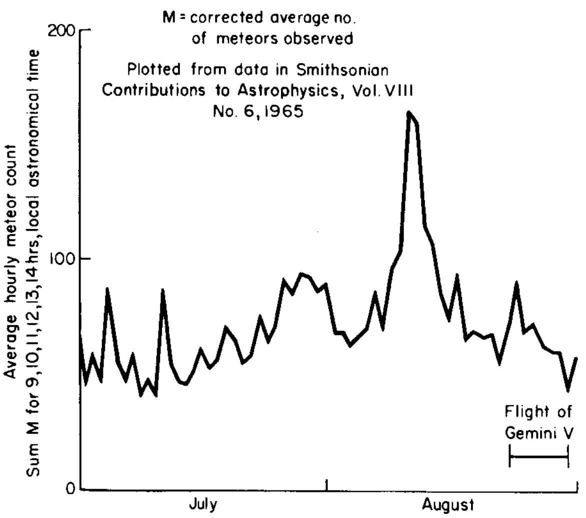


FIGURE 32–8.—Average hourly count of meteors during July and August.

situation since no interval of concentrated observation of them was possible on that rendezvous flight. The brightness of the moon, going through full phase during Gemini VII, may also have interfered with meteor observations. Although the peak of the Geminids meteor shower definitely occurred during the flight of Gemini VII, the crewmen probably were not observing during that period, which would last only a few hours. Another factor might be the presence of frequent lightning flashes, which could distract the crewmen's attention and hamper their dark adaptation.

It is possible that crewmen may count numerous meteors on some future flight when they happen to, or plan to, observe near the maximum of a meteor swarm.

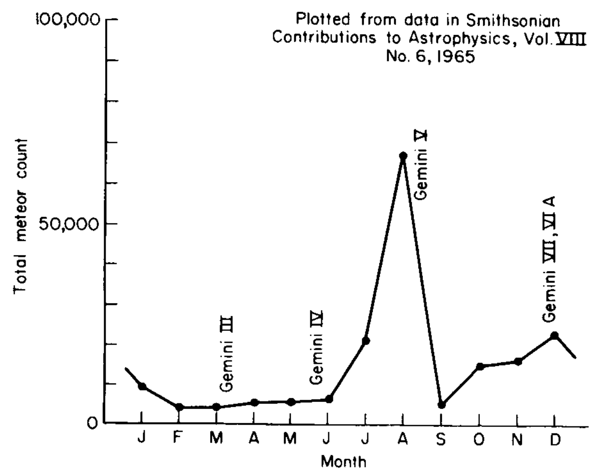


FIGURE 32–9.—Monthly meteor count.

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33. DIM LIGHT PHOTOGRAPHY

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Introduction and Summary

For the Gemini VI and VII missions, plans were made to perform photography (on an opportunity basis) of a variety of dim-light phenomena with existing onboard cameras using "operational" film. Eastman No. 2475 film was selected for the morphological photography of Comet Ikeya-Seki. This work had been intended for Gemini VI as originally scheduled for October 25, 1965, just 5 days after perihelion passage of the comet. This investigation was brought about by a number of factors including the following:

(1) Previous, unaided eye observations by Mercury and Gemini astronauts which suggested the possibility and desirability of recording certain phenomena on film.

(2) An unusual event such as the newly discovered Comet Ikeya-Seki.

(3) The need to obtain additional information on airglow, for example, to assist in interpretation of results from an unmanned satellite, the first of the polar orbiting geophysical observatory series.

(4) The desire to obtain information on night cloud cover to assist in the design of future weather satellites.

(5) The desire to obtain information on the level of the luminance (brightness) of the day sky.

(6) The wish to study the earth's atmosphere by means of twilight limb photography, etc.

Another consideration, particularly in the case of the Gemini VII mission, was that during a 14-day mission, there might be sufficient time to exploit a number of observational possibilities. It was recognized that considerations of the mission requirements, operational procedures, and the scheduled experiments with the attendant fuel and time usage would probably preclude the performance of many of the dim-light photographic tasks. Nevertheless, it was determined that it would be useful to have an onboard checklist of subtasks and written related material that would permit maximum ultilization of the camera equipment and film allocated to the flights, should time and fuel become available. A reproduction of the detailed information written for the astronauts is available from the authors.

Other factors behind this type of investigation included:

(1) A study of the ease with which an observation or an experiment could be synthesized onboard (provided certain basic equipment was available to the crewmembers—in this case a flexible camera, interchangeable lens, a variety of black-and-white and color film, and some optical filters) based on phenomena observed by the crewmembers or transmitted to them from the ground. The information transmitted, in turn, could come either as a result of ground, rocket, or satellite observations, or as a spontaneous need to obtain some knowledge from the spacecraft.

(2) Additional experience which might benefit related experiments such as stellar spectroscopy and airglow photography which are definitely selected for the later Gemini missions.

(3) The further advancement of the acquisition of data on the optical environment of a manned satellite.

(4) The desire to continue to give the crewmen the opportunity to bring back objective information to support and add to their visual observations.

(5) The wish to obtain information to help define future experiments as to design, procedure, scheduling, interference, and complexity.

This report should be considered only as a progress report, inasmuch as at this writing all the onboard voice recordings are not available for study, and there has been insufficient time to analyze the recorded briefings and to identify and analyze the film with a densitometer.

The specific phenomena for possible study and photography during the missions included: (1) twilight scene, (2) night cloud cover, (3) sunlit airglow, (4) day-sky background, (5) night airglow, edge-on, (6) aurorae, (7) meteors, (8) lightning, (9) artificial lighting, (10) galactic survey, (11) zodiacal light and gegenschein, and (12) comets.

Formal briefings and training of the crewmembers for this study were minimal, which was both possible and necessary for several reasons. Except for three narrow-bandpass filters, this study used only onboard equipment, with which the crew were familiar. Even the use of lens filters was not new, since a minus blue haze filter was onboard for use in terrestrial photography. The crewmembers had been exposed to information about dim light phenomena briefly on several occasions during their basic training in astronomy and atmospheric physics. This had been reinforced during discussions and debriefing sessions with previous crewmembers, and Astronaut Schirra had observed some of these phenomena directly during his MA-8 mission. Because this study was approved and inserted into the flight plan at a late date, due to its low priority in a very busy schedule of events, and because the investigators (as well as the crews) did not wish to add a disorganizing influence late in the planning, the investigators chose properly to omit a formal briefing. Instead, the crewmembers were provided with written material and checklists to acquaint them with the specific operational tasks and inflight judgments required to obtain data and to respond quickly to ground requests as opportunities arose during the flight.

Photographs taken and identified at this time (February 6, 1966) included:

(1) Black-and-white as well as color shots of the twilight scene.

(2) A series showing night cloud cover where the illumination was the sum of lunar, airglow, zodiacal, and stellar light.

(3) Lightning.

(4) Airglow, edge-on.

(5) Thrusters.

(6) The Gemini VII spacecraft from Gemini VI-A.

(7) Probably the third stage of a Minuteman rocket and possibly its reentry vehicle.

Many tasks were not performed because of fuel- and weather-related scheduling problems. It is emphasized here that all the approved experiments reported elsewhere were properly accorded higher priority.

Description

A fuller description of all the phenomena listed in the introduction for possible photography has been prepared by the authors (ref. 1). For brevity, only those tasks for which there was an opportunity to photograph from Gemini VI-A or Gemini VII are given here. However, for ready reference and illustration, the checklist placed onboard is reproduced as figure 33-1. The exposures shown were based on an American Standards Association (ASA) value of several thousands for the Eastman 2475 film, using data reported by Hennes and Dunkelman, 1966 (ref. 2).

It is emphasized that the tasks and procedures were related to the approved onboard cameras, which included:

(1) Hasselblad (70-mm film) with 80-mm (f/2.8) lens and 250-mm (f/5.6) telephoto lens.

(2) Movie/sequence Maurer 16-mm camera.

For dim-light photography, faster lenses would have been desirable. Nevertheless, in some cases, it was still considered reasonable to use these relatively slow lenses, with the highest speed film available, for survey purposes.

Results

Reproductions of three photographs, whose analysis has recently begun, are shown on the following pages. Figure 33-2 is a photograph of the Gemini VII spacecraft taken from Gemini VI-A during the rendezvous exercise. Most of the illumination was furnished from the Gemini VI-A docking light, since the moon was in the last quarter and produced an illuminance of only 10 percent of full moonlight. Figure 33-3 is a photograph, from a 140-nautical-mile slant angle, of a Minuteman missile reentering the earth's atmosphere showing the

DIM LIGHT PHOTOGRAPHY

CODING:
1 = HASSELBLAD 2 = 16 MM MAURER
3 = 2475 B & W 4 = SO 217 COLOR
A = 80 MM LENS B = 250 MM LENS
C = F-STOP 2.8 D = F-STOP 5.6
X = 75 MM LENS Y = 1 FPS, 1/50

1.TWILIGHT BANDS: POST-SUNSET OR PRE-SUNRISE

EQUIP		1	2	3	4	5	6
CODE 14B	F T	11 1/500	5.6 1/500	5.6 1/125	5.6 1/30	5.6 1/8	5.6 1/2
CODE 24XY	F T	32 10	22 10	16 10	11 10	8 10	5.6 10

7	8	9	10
5.6 2	5.6 10	5.6 30	5.6 120
4 10	2.5 210		

REVERSE ORDER OF SEQUENCE FOR PRE-SUNRISE: HORIZON JUST ABOVE SUNSET IN LOWER LEFT OR RIGHT CORNER

2.NIGHT CLOUD COVER: CODE 13AC, TRACK CLOUDS

CONDITIONS VS TIME	1	2	3	4
NO MOON	8	16	-	-
QUARTER MOON	1/4	1/2	1	2
FULL MOON	1/30	1/15	1/8	1/4

3 SUNLIT AIRGLOW: CODE 13AC

NO	SUBJECT/COND	SET	1	2	3
1	sunset +60 SEC HORIZON SCENE, SET POINT CENTERED IN LOWER PART OF PHOTO	T FTR	1/8 6300	1/8 6225	1/8 5300
		-	4	5	6
		T FTR	10 5300	10 6225	10 6300
2	SMOST SET/RISE +/-60 SOUTH HZ	(SAME TIME/FILTER SEQUENCE AS NO. 1)			
3	NORTH OR SOUTH HORIZON AT MID NIGHT ±10 MIN	SET	1	2	3
		T FTR	10 6300	10 6225	10 5300
4	NMOST SET/RISE +/-60 NORTH HZ	(SAME TIME/FILTER SEQUENCE AS NO. 1)			
5	SUNRISE -60 SEC AT HORIZON	(REVRS TIME/FILTER SEQUENCE AS NO. 1)			

4.DAY SKY BACKGROUND: CODE 13AC, WINDOW SHADED FROM SUN & EARTHSHINE - POINT CAMERA TOWARD SKY, 3 EXP; 5, 30 120 SEC

5.NIGHT AIRGLOW EDGE-ON: CODE 13AC - 5 EXP; 1/2, 1, 2, 4, 8 SEC WITH HORIZON IN FIELD

6.AURORAE: CODE 13AC TWO TYPES OF AURORA

BRIGHT	1/8	1/2	2
DIM	1	4	15

7.METEORS: CODE 13AC

TOTAL COUNT	30	120	300
INDIVIDUAL RECORD	AS REQUIRED		

8.LIGHTING: USE WITH BOTH CODES: 13AC 13BD

TOTAL COUNT	10	30	120	300
INDIVIDUAL RECORD	AS	REQUIRED		

DO WITH METEORS

9.ARTIFICIAL LIGHTING: CODE 13AC, 1/8, 1/2; AND CODE 13BD, 1/4, 1 SEC

10.GALACTIC SURVEY: CODE 13, HOLD +1/2 DEG

SUBJECT/COND	1	2	3	4	5	6	7
ORION OR PEG ASUS, MOON 45 DEG AWAY	80 2.8 60	80 2.8 120	80 2.8 240	80 5.6 120	80 5.6 240	250 5.6 120	250 5.6 240
CODE AC, EACH EXP 90 SEC, DO EACH ZODIACAL CONSTELLATION TO WITHIN 15° OF SUN							

11.ZODIACAL LIGHT & GEGENSCHEIN: CODE 13AC

5-10 MIN INTO DARK

ZODIACAL	1/16	1/4	1	3	5
GEGENSCH	10	30	60	120	-

12.COMET: CODE 13AC OR 13BD IF PHOTOS TAKEN

FOLLOWING LIST OF KEY WORDS/PHRASES AS REF

TIME HACK STAR TRANSITS	GLARE & LIGHTING
ANGULAR MEASUREMENTS	LAYERS/STREAK/THICK-
LOCATE POSITION	NESS/SEPERATION/HUE/
ADJACENT STARS/PLANETS	COLOR/BRIGHTNESS/
ESTIMATE ATTITUDE/RATES	EDGE FEATURES/COUNT

FIGURE 33–1.—Crew inflight checklist for dim-light study.

FIGURE 33–2.—Gemini VII spacecraft as photographed at night by Gemini VI–A flight crew.

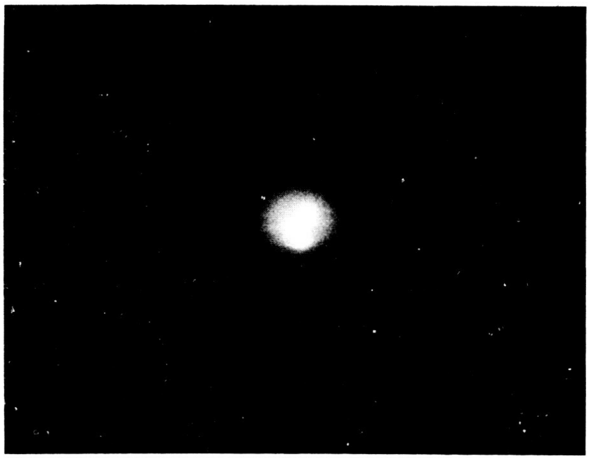

FIGURE 33–3.—Reentering Minuteman missile as photographed by Gemini VII flight crew.

FIGURE 33-4.—Nightglow, moonlit earth and clouds, and lightning in clouds as photographed by Gemini VII flight crew.

glow from the third-stage rocket and possibly its reentry vehicle. Figure 33-4 is one of a series of scenes showing night cloud cover. The exposure was 8 seconds at a lens setting of f/2.8 and was taken when the moon was almost full. The night airglow is seen in the original film as a rather faint but distinctly visible layer. When comparing this photograph with those taken of the night airglow from a rocket (ref. 2), it is difficult to explain the faint layer when taking into account the apertures, time, and film. An analysis is in progress to determine whether the exposure here is effectively less than f/2.8. The bright-appearing cloud just to the right of the center is believed to be caused by lightning.

Certain new experiments, or at least modifications or additions to those already scheduled for later manned flights, were identified. Among these are:

(1) Photographic and spectroscopic studies of the twilight scene in order to study aerosol heights and composition.

(2) Photographic and/or photoelectric luminance (brightness) of the day-sky background (related to the difficulties of seeing stars in the daytime) and otherwise making physical observations during the daytime phase. (As an example, the S-1 experiment planned for Gemini VIII will include at least one exposure to obtain data on the day sky.)

(3) Further studies of night cloud cover.

(4) Planetary spectrophotography.

(5) Photoelectric measurements to support both visual estimates and photographic exposures for phenomena too dim for "standard" exposure meters.

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34. EXPERIMENT S–8/D–13, VISUAL ACUITY AND ASTRONAUT VISIBILITY

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Summary

Preflight, inflight, and postflight tests of the visual acuity of the members of the Gemini V and Gemini VII crews showed no statistically significant change in their visual capability. Observations of a prepared and monitored pattern of rectangles made at a ground site near Laredo, Tex., confirmed that the visual performance of the astronauts in space was within the statistical range of their respective preflight thresholds, and that laboratory visual acuity data can be combined with environmental optical data to predict correctly man's limiting visual capability to discriminate small objects on the surface of the earth in daytime.

Introduction

Reports by Mercury astronauts of their sighting small objects on the ground prompted the initiation of a controlled visual acuity experiment which was conducted in both Gemini V and Gemini VII. The first objective of Experiment S–8/D–13 was to measure the visual acuity of the crewmembers before, during, and after long-duration space flights in order to ascertain the effects of a prolonged spacecraft environment. The second objective was to test the use of basic visual acuity data, combined with measured optical properties of ground objects and their natural lighting, as well as of the atmosphere and the spacecraft window, for predicting the flight crew's limiting naked-eye visual capability to discriminate small objects on the surface of the earth in daylight.

Inflight Vision Tests

Inflight Vision Tester

Throughout the flights of Gemini V and Gemini VII, the visual performance of the crewmembers was tested one or more times each day by means of an inflight vision tester. This was a small, self-contained, binocular optical device containing a transilluminated array of 36 high-contrast and low-contrast rectangles. Half of the rectangles were oriented vertically in the field of view, and half were oriented horizontally. Rectangle size, contrast, and orientation were randomized; the presentation was sequential; and the sequences were nonrepetitive. Each rectangle was viewed singly at the center of a 30° adapting field, the apparent luminance of which was 116 foot-lamberts. Both members of the flight crew made forced-choice judgments of the orientation of each rectangle and indicated their responses by punching holes in a record card. Electrical power for illumination within the instrument was derived from the spacecraft.

The space available between the eyes of the astronaut and the sloping inner surface of the spacecraft window, a matter of 8 or 9 inches, were important constraints on the physical size of the instrument. The superior visual performance of all crewmembers, as evidenced by clinical test scores, made it necessary to use great care in alining the instrument with the observer's eyes, since the eyes and not the instrument must set the limit of resolution. In order to achieve this, the permissible tolerance of decentering between a corneal pole and the corresponding optical axis of the eyepiece was less than 0.005 of an inch. This tolerance was met by means of a biteboard equipped with the flight crewmember's dental impression to take advantage of the fixed geometrical relation between his upper teeth and his eyes. Figure 34–1 is a photograph of the inflight vision tester.

Selection of the Test

The choice of test was made only after protracted study. Many interacting requirements were considered. If, for example, the visual capabilities of the astronauts should change during the long-duration flight, it would be of prime importance to measure the change in such a way that man's inflight ability to recognize, classify, and identify landmarks or unknown objects on the ground or in space could be predicted. These higher-order visual discriminations depend upon the quadratic content of the difference images between alternative objects, but virtually all of the conventional patterns used in testing vision yield low-precision information on this important parameter. Thus, the prediction requirement tended to eliminate the use of Snellen letters, Landolt rings, checkerboards, and all forms of detection threshold tests.

The readings must not go off-scale if visual changes should occur during flight. This requirement for a broad range of testing was not readily compatible with the desire to have fine steps within the test and yet have sufficient replication to insure statistically significant results.

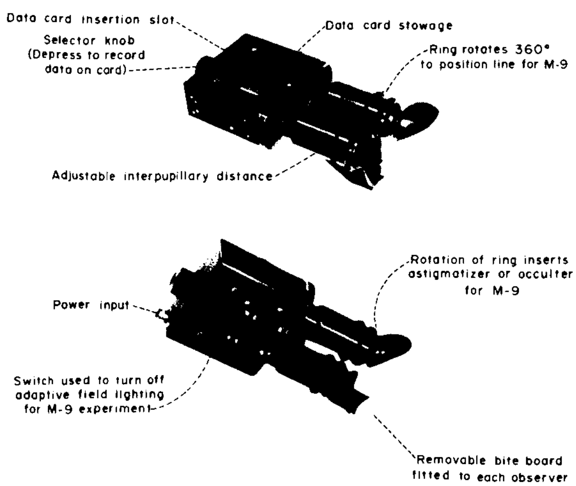

FIGURE 34–1.—Inflight vision tester.

It was also deemed desirable that the pattern chosen for the inflight vision tester should be compatible with that used on the ground where search contamination of the scores must be carefully avoided; this consideration made any conventional detection threshold test undesirable. The pattern on the ground was within sight for at least 2 minutes during all usable passes, but variations due to atmospheric effects, geometrical foreshortening, directional reflectance characteristics, etc., made it necessary to select a test which could be completed in a 20-second period centered about the time of closest approach.

The optimum choice of test proved to be the orientation discrimination of a bar narrow enough to be unresolved in width but long enough to provide for threshold orientation discrimination. The size and apparent contrast of all of the bars used in the test were sufficient to make them readily detectable, but only the larger members of the series were above the threshold of orientation discrimination. These two thresholds are more widely separated for the bar than for any other known test object. The inherent quadratic content of the difference image between orthogonal bars is of greater magnitude than the inherent quadratic content of the bar itself. Interpretation of any changes in the visual performance of the astronauts is, therefore, more generally possible on the basis of orientation discrimination thresholds for the bar than from any other known datum.

Rectangles in the Vision Tester

The rectangles presented for viewing within the inflight vision tester were reproduced photographically on a transparent disk. Two series of rectangles were included, the major series set at a contrast of −1 and the minor series set at about one-fourth of this value. The higher contrast series constituted the primary test and was chosen to simulate the expected range of apparent contrast presented by the ground panels to the eyes of the crewmen in orbit. The series consisted of six sizes of rectangles. The sizes covered a sufficient range to guard against virtually any conceivable change in the visual performance of the astronauts during the long-duration flight. The size intervals were small enough, however, to provide a sufficiently sensitive test.

The stringent requirements imposed by conditions of space flight made it impossible to use as many replications of each rectangle as was desirable from statistical considerations. After much study, it was decided to display each of the six rectangular sizes four times. This compromise produced a sufficient statistical sample to make the sensitivity of the inflight test comparable to that ordinarily achieved with the most common variety of clinical wall chart. This sensitivity corresponds roughly to the ability to separate performance at 20/15 from performance at 20/20. It was judged that this compromise between the sensitivity of test and the range of the variables tested was the proper one for this exploratory investigation.

A secondary test at lower contrast was included as a safeguard against the possibility that visual performance at low contrast might change in some different way. With only 12 rectangles assignable within the inflight vision tester for the low-contrast array, it was decided to use only 3 widely different rectangle sizes, presenting each of these sizes 4 times.

Because of the accelerated launch schedule of Gemini V, it was not possible to use the flight instrument for preflight experiments. These data were, therefore, obtained with the first of the inflight vision testers (serial no. 1), while the last instrument to be constructed (serial no. 5) was put aboard the spacecraft. The two instruments were optically identical except for their 12 low-contrast rectangles, which measured a contrast of −0.332 and −0.233, respectively. In Gemini VII all of the reported data (preflight, inflight, and postflight) were obtained with serial no. 5 tester.

Analysis of Correct Scores in Gemini V

A comparison of the correct scores made by the Gemini V crewmembers on the ground (preflight) and in space (inflight) can be used to ascertain whether their observed visual performance differed in the environments or changed during the 7-day mission. The correct scores from the low-contrast and high-contrast series in the vision tester are shown for both crewmembers in figure 34–2. The results of standard statistical tests applied to these data are shown in tables 34–I through 34–IV.

Comparisons between preflight and inflight data are given in tables 34–I and 34–II. All Student's t tests show no significant difference in means. All Snedecor's F tests show no significant difference in variances at the 0.05 level, with the exception of Cooper's high-contrast comparison, which shows no significant difference at the 0.01 level.

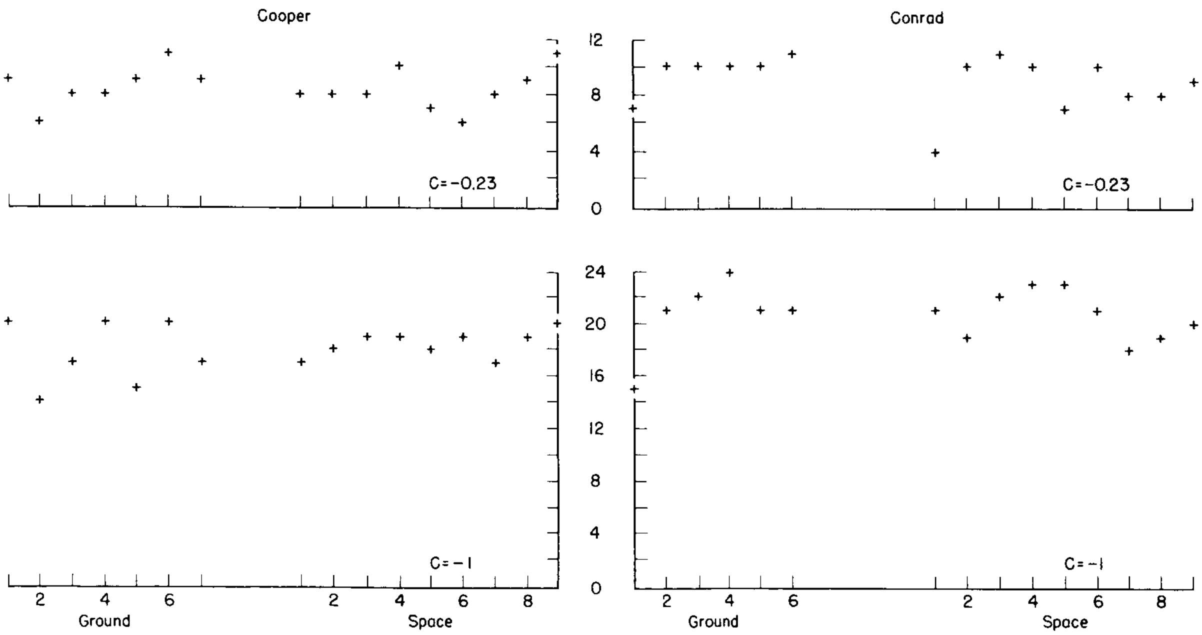

FIGURE 34–2.—Correct vision-tester scores for Gemini V flight crew.

Comparisons between the inflight data at the beginning of the mission with that at the end are made in tables 34–III and 34–IV. All Student's t tests and Snedecor's F tests show no significant differences at 0.05 level, with the exception of the F test on Conrad's low-contrast comparison, which shows no significant contrast at 0.01 level.

TABLE 34–I.—*Vision Tester (Ground Versus Space*

Cooper	$C=-1$		$C=-0.23$	
	Ground	Space	Ground	Space
Number	7	9	7	9
Mean	17. 6	18. 4	8. 6	8. 3
Standard deviation	2. 3	. 96	1. 3	1. 4
t	0. 96		0. 31	
$t_{0.05}$	2. 14		2. 14	
F	6. 12		1. 02	
$F_{0.05}$	3. 58		3. 58	
$F_{0.01}$	6. 37		----------	

TABLE 34–II.—*Vision Tester (Ground Versus Space)*

Conrad	$C=-1$		$C=-0.23$	
	Ground	Space	Ground	Space
Number	7	9	7	9
Mean	20. 7	20. 7	9. 7	8. 6
Standard deviation	2. 7	1. 7	1. 2	2. 0
t	0		1. 13	
$t_{0.05}$	2. 14		2. 14	
F	2. 79		2. 43	
$F_{0.05}$	3. 69		4. 82	

These statistical findings support the null hypothesis advanced by many scientists before the Gemini V mission was flown.

Analysis of Correct Scores in Gemini VII

A comparison of the correct scores made by the Gemini VII crewmembers on the ground (preflight) and in space (inflight) can be used to ascertain whether their observed visual performance differed in the environments or changed during the 14-day mission. The correct scores from the low-contrast and high-contrast series in the vision tester are shown for both crewmembers in figure 34–3. The results of standard statistical tests applied to these data are shown in tables 34–V through 34–VIII.

Comparisons between preflight and inflight data are given in tables 34–V and 34–VI. All Student's t tests show no significant difference in means. All Snedecor's F tests show no significant difference in variances at the 0.05 level, with the exception of Borman's low-contrast comparison, which shows a weekly significant difference at the 0.01 level.

TABLE 34–III.—*Vision Tester (Inflight Trend)*

Cooper	$C=-1$		$C=-0.23$	
	First 4	Last 4	First 4	Last 4
Number	4	4	4	4
Mean	18. 2	18. 8	8. 5	8. 5
Standard deviation	. 83	1. 1	. 87	1. 8
t	0. 68		0	
$t_{0.05}$	2. 45		2. 45	
F	1. 73		4. 33	
$F_{0.05}$	9. 28		9. 28	

TABLE 34–IV.—*Vision Tester (Inflight Trend)*

Conrad	$C=-1$		$C=-0.23$	
	First 4	Last 4	First 4	Last 4
Number	4	4	4	4
Mean	21. 3	19. 5	8. 8	8. 75
Standard deviation	1. 5	1. 1	2. 8	. 83
t	1. 64		0	
$t_{0.05}$	2. 45		2. 45	
F	1. 96		11. 19	
$F_{0.05}$	9. 28		9. 28	
$F_{0.01}$	----------		29. 5	

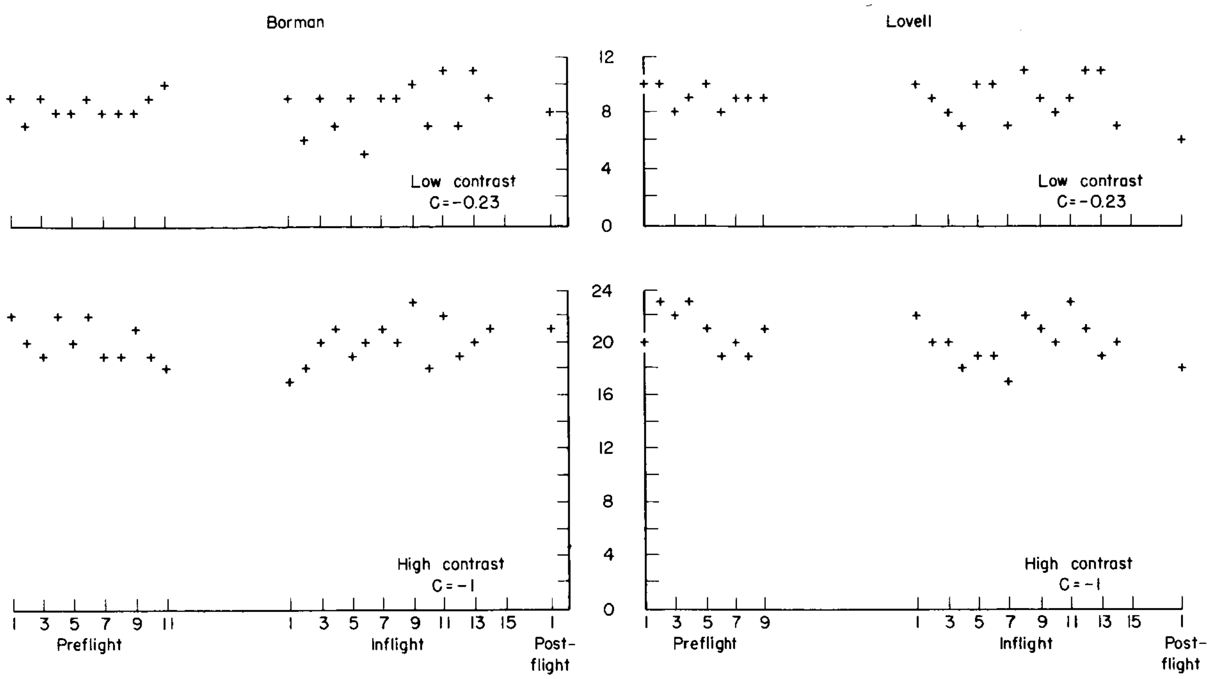

FIGURE 34-3.—Correct vision-tester scores for Gemini VII flight crew.

TABLE 34-V.—*Vision Tester (Ground Versus Space)*

Borman	$C=-1$		$C=-0.23$	
	Ground	Space	Ground	Space
Number	11	14	11	14
Mean	20. 0	19. 9	8. 45	8. 4
Standard deviation	1. 3	1. 6	. 78	1. 7
t	0. 12		0. 017	
$t_{0.05}$	2. 07		2. 07	
F	1. 49		4. 74	
$F_{0.05}$	2. 89		2. 89	
$F_{0.01}$	4. 66		4. 66	

Comparisons between the inflight data at the beginning of the mission with those at the end are made in tables 34–VII and 34–VIII. All Student's t tests and Snedecor's F tests show no significant difference at 0.05 level, with the exception of the F test on Borman's low-contrast comparison, which shows no significant contrast at the 0.01 level.

These statistical findings provide additional support for the null hypothesis advanced by many scientists before the Gemini missions were flown. Examination of the sensitivity of the test must be considered next. This topic is treated in the following paragraphs.

Preflight Physiological Baseline

Design of the inflight vision tester, as well as the ground sighting experiments described in subsequent paragraphs and the interpretation of the results from both experiments, required that a preflight physiological baseline be obtained for both crewmembers. For this purpose a NASA van was fitted out as a portable vision research laboratory, moved to the Manned

TABLE 34-VI.—*Vision Tester (Ground Versus Space)*

Lovell	$C=-1$		$C=-0.23$	
	Ground	Space	Ground	Space
Number	9	14	9	14
Mean	20. 9	20. 0	9. 1	9. 1
Standard deviation	1. 4	1. 6	74	1. 4
t	1. 29		0. 073	
$t_{0.05}$	2. 08		2. 08	
F	1. 17		3. 64	
$F_{0.05}$	3. 26		3. 26	
$F_{0.01}$	5. 62		5. 62	

TABLE 34-VII.—*Vision Tester (Inflight Trend)*

Borman	$C=-1$		$C=-0.23$	
	First 5	Last 5	First 5	Last 5
Number	5	5	5	5
Mean	19. 0	20. 0	8. 0	9. 0
Standard deviation	1. 4	1. 4	1. 3	1. 8
t	1. 00		0. 91	
$t_{0.05}$	2. 31		2. 31	
F	1. 00		2. 00	
$F_{0.05}$	6. 39		6. 39	

TABLE 34-VIII.—*Vision Tester (Inflight Trend)*

Lovell	$C=-1$		$C=-0.23$	
	First 5	Last 5	First 5	Last 5
Number	5	5	5	5
Mean	19. 8	20. 4	8. 8	9. 2
Standard deviation	1. 3	1. 5	1. 2	1. 6
t	0. 60		0. 40	
$t_{0.05}$	2. 31		2. 31	
F	1. 27		1. 88	
$F_{0.05}$	6. 39		6. 39	

Spacecraft Center at Houston, Tex., and operated by Visibility Laboratory personnel. Figure 34–4 is a cutaway drawing of this research van. The astronauts, seated at the left, viewed rear-screen projections from an automatic projection system located in the opposite end of the van. Each astronaut participated in several sessions in the laboratory van, during which they became experienced in the psychophysical techniques of the rectangle orientation discrimination visual task. A sufficiently large number of presentations was made to secure a properly numerous statistical sample. The astronauts' forced-choice visual thresholds for the discrimination task were measured accurately and their response distributions determined so that the standard deviations and confidence limits of their preflight visual performance were determined.

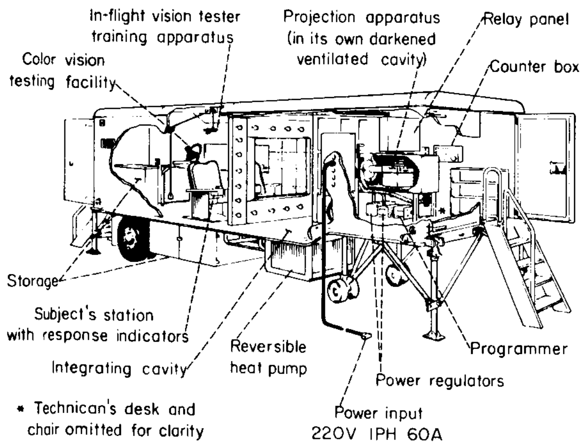

FIGURE 34–4.—Vision research and training van.

Figure 34–5 is a logarithmic plot of the Gemini V pilot's preflight visual thresholds for the rectangle orientation discrimination task. In this figure the solid angular subtense of the rectangles is plotted along the horizontal axis because both the inflight vision tester and the ground observation experiments used angular size as the independent variable. The solid line in this figure represents the forced-choice rectangle orientation threshold of the pilot at the 0.50 probability level. The dashed curves indicate the $-\sigma, +\sigma$, and $+2\sigma$ levels in terms of contrast. The six circled points in the upper row indicate the angular sizes of the high-contrast ($C=-1$) rectangles presented by the inflight vision tester. The three circled points of the middle and lower rows show the angular sizes of the low-contrast rectangles used in the preflight unit (serial no. 1) and the flight unit (serial no. 5), respectively.

The separate discriminations recorded on the record cards in the inflight vision tester can be used to determine a threshold of angular size.

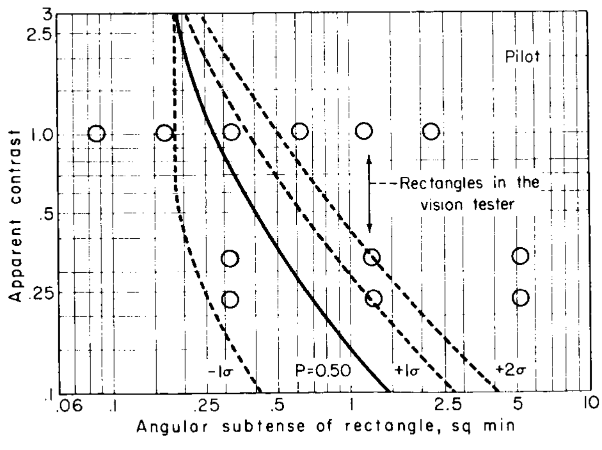

FIGURE 34–5.—Logarithmic plot of preflight visual thresholds.

These thresholds and corresponding statistical confidence limits derived with the aid of figure 34–5 are plotted for the high- and low-contrast tests of the Gemini V command pilot in figures 34–6 and 34–7, and for the Gemini V pilot in figures 34–8 and 34–9. Corresponding thresholds and confidence limits for the vision tester data secured by the Gemini VII command pilot are shown in figures 34–10 and 34–11. Similar data secured by the Gemini VII pilot are shown in figures 34–12 and 34–13.

These eight figures also support the null hypothesis, and their quantitative aspect constitutes a specification of the sensitivity of the test. Thus, as planned, variations in visual performance comparable with a change of one line on a conventional clinical wall chart would have been detected. Preflight threshold data can, therefore, be used to predict the limiting visual acuity capabilities of astronauts during space flight, if adequate physical information concerning the object and its background, atmospheric effects, and the spacecraft window exists. A test of such predictions was also carried out and is described in the following paragraphs.

Ground Observations

The crews of both Gemini V and Gemini VII observed prepared and monitored rectangular patterns on the ground in order to test the use of basic visual acuity data, combined with measured optical properties of ground objects and their natural lighting, the atmosphere, and the spacecraft window, for predicting the limiting naked-eye visual capability of astronauts to discriminate small objects on the surface of the earth in daylight.

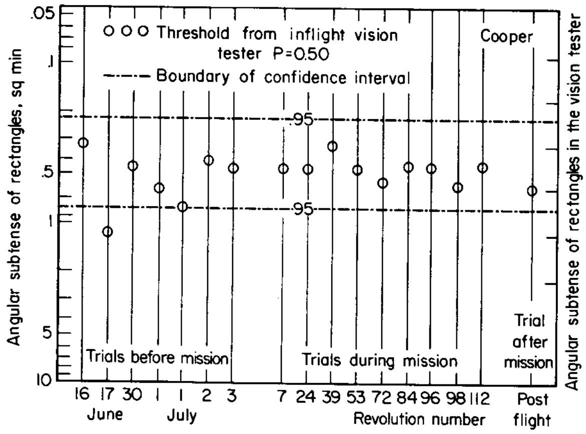

FIGURE 34–6.—Gemini V command pilot's rectangle discrimination thresholds, $C=-1$.

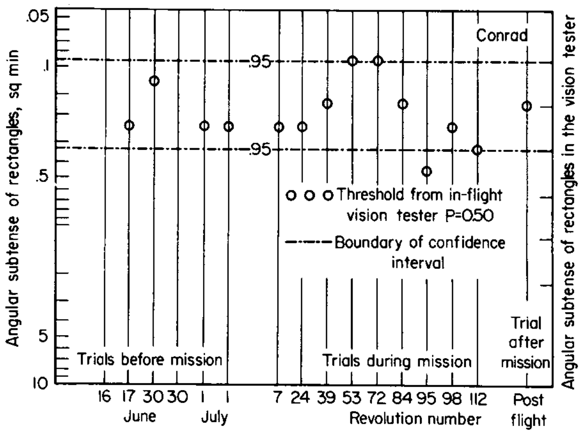

FIGURE 34–8.—Gemini V pilot's rectangle discrimination thresholds, $C=-1$.

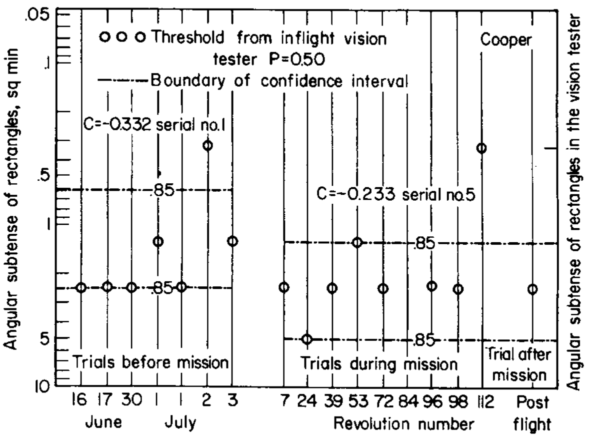

FIGURE 34–7.—Gemini V command pilot's rectangle discrimination thresholds.

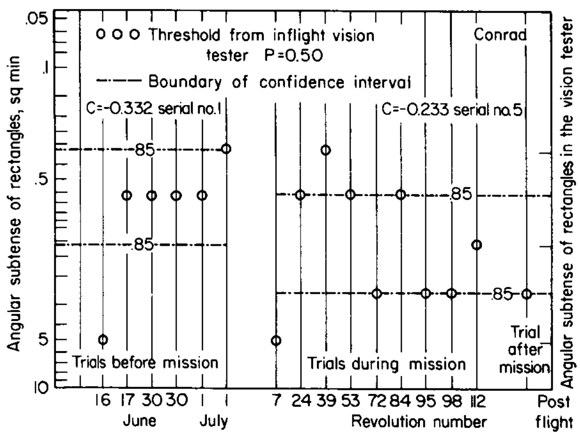

FIGURE 34–9.—Gemini V pilot's rectangle discrimination thresholds.

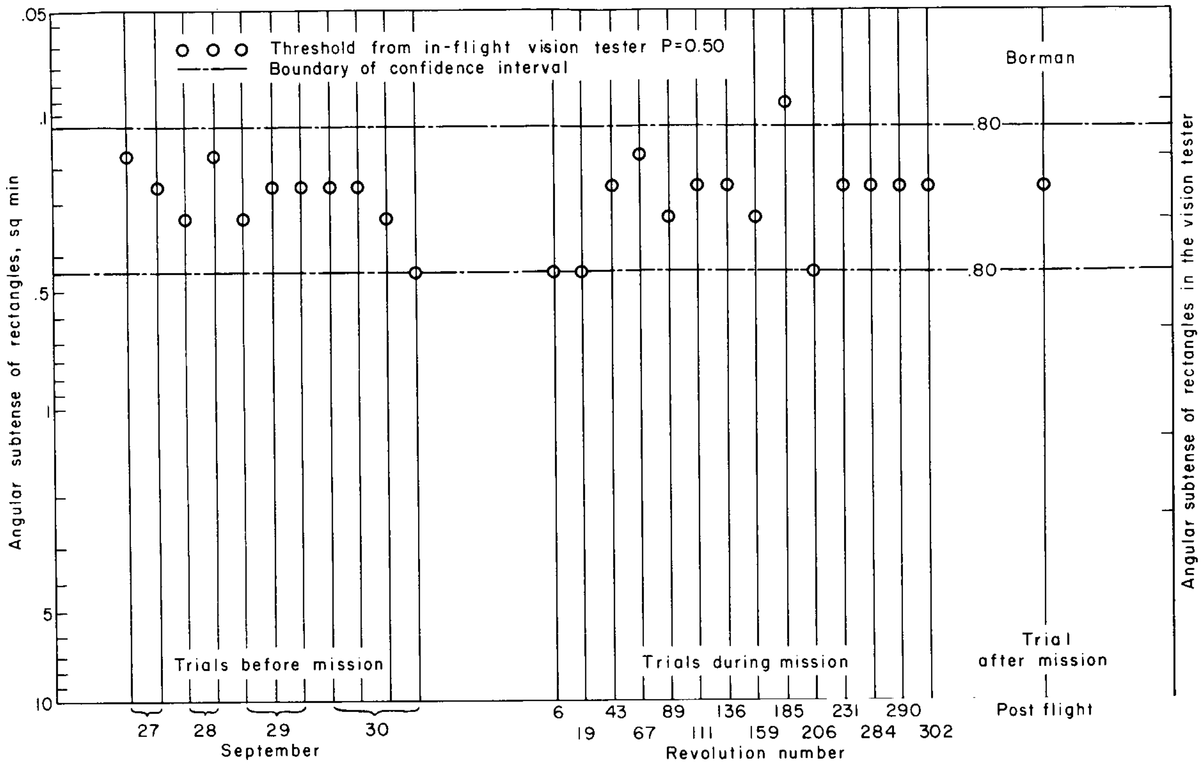

Figure 34–10.—Gemini VII command pilot's rectangle discrimination thresholds, $C = -1$.

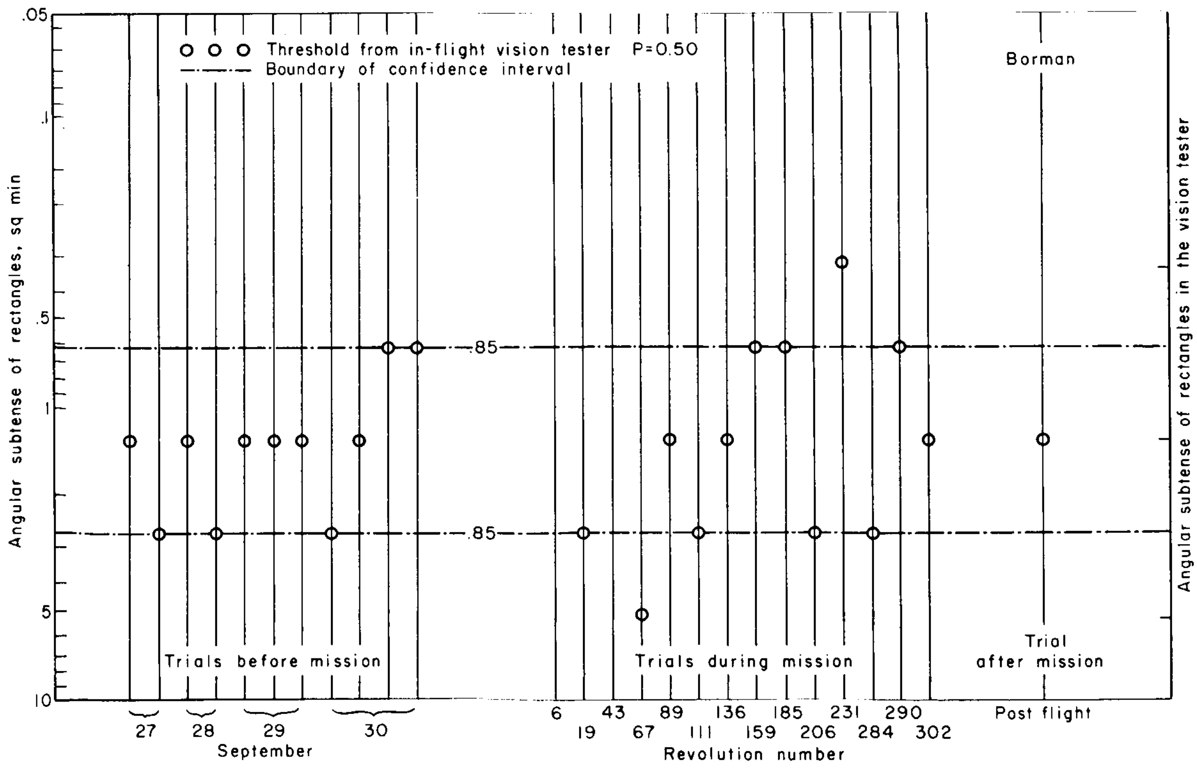

Figure 34–11.—Gemini VII command pilot's rectangle discrimination thresholds, $C = -0.233$.

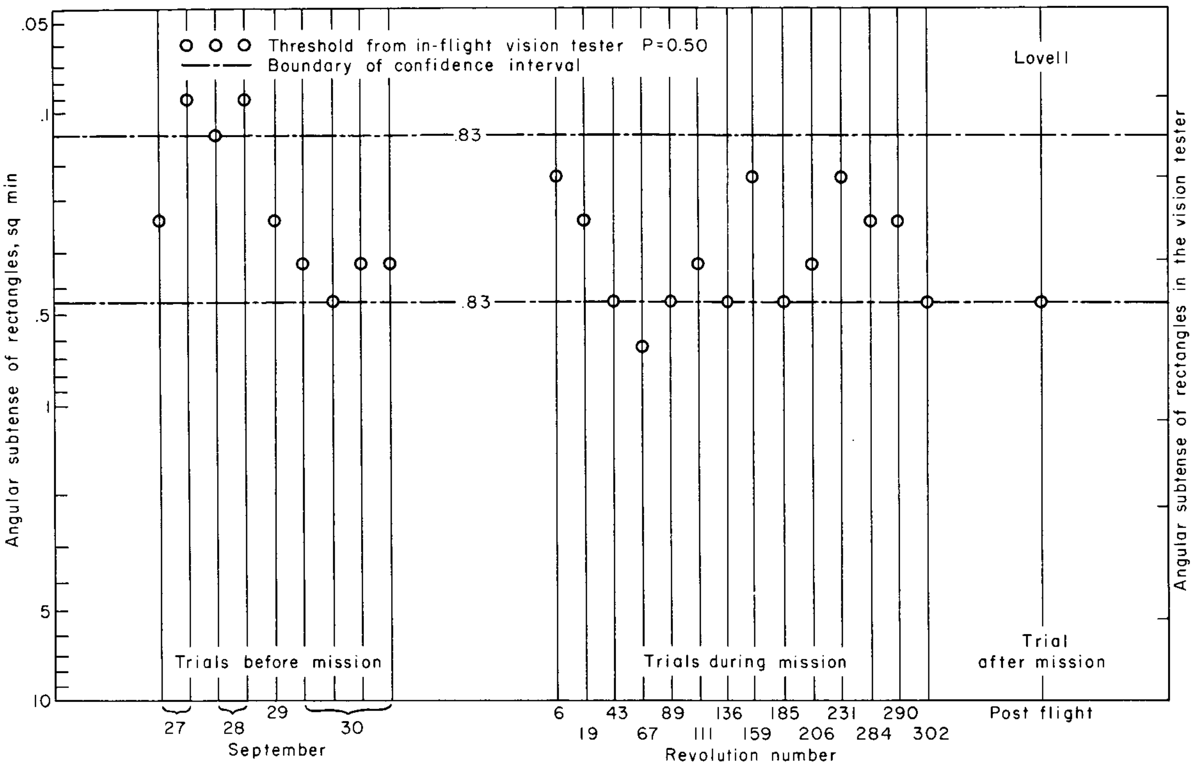

FIGURE 34–12.—Gemini VII pilot's rectangle discrimination thresholds, $C=-1$.

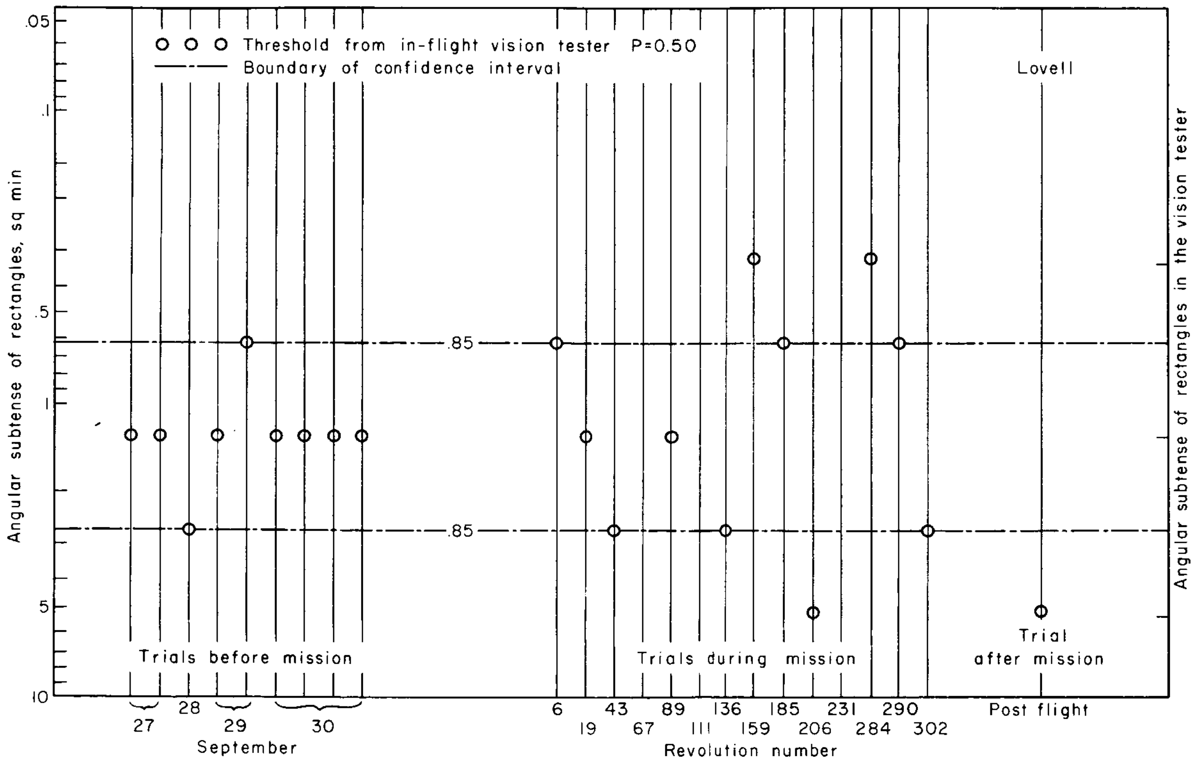

FIGURE 34–13.—Gemini VII pilot's rectangle discrimination thresholds, $C=-0.233$.

Equipment

The experimental equipment consists of an inflight photometer to monitor the spacecraft window, test patterns at two ground observation sites, instrumentation for atmospheric, lighting, and pattern measurements at both sites, and a laboratory facility (housed in a trailer van) for training the astronauts to perform visual acuity threshold measurements and for obtaining a preflight physiological baseline descriptive of their visual performance and its statistical fluctuations. These equipments, except the last, are described in the following paragraphs.

Spacecraft window photometer.—A photoelectric inflight photometer was mounted near the lower right corner of the pilot's window of the Gemini V spacecraft, as shown in figure 34–14, in order to measure the amount of ambient light scattered by the window into the path of sight at the moment when observations of the ground test patterns were made. The photometer (fig. 34–15) had a narrow (1.2°) circular field of view, which was directed through the pilot's window and into the opening of a small black cavity a few inches away outside the window. The photometric scale was linear and extended from approximately 12 to 3000 foot-lamberts. Since the apparent luminance of the black cavity was always much less than 12 foot-lamberts, any reading of the inflight photometer was ascribable to ambient light scattered by the window. Typical data acquired during passes of Gemini V over the Laredo site are shown in figure 34–16. This in-

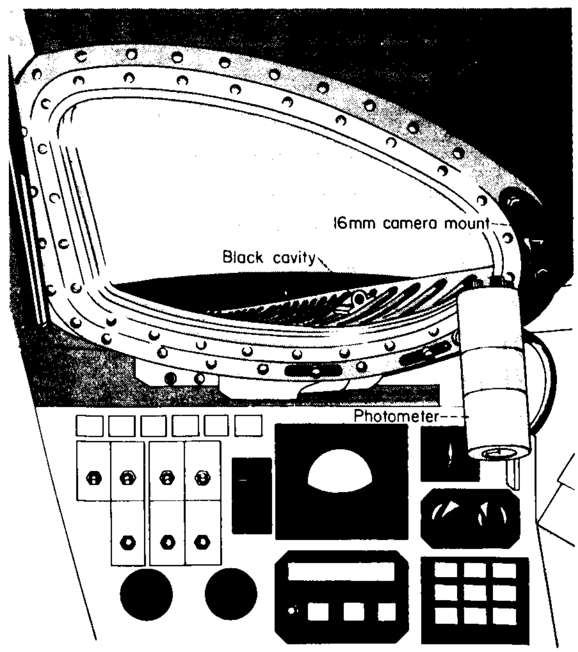

Figure 34–14.—Location of inflight photometer.

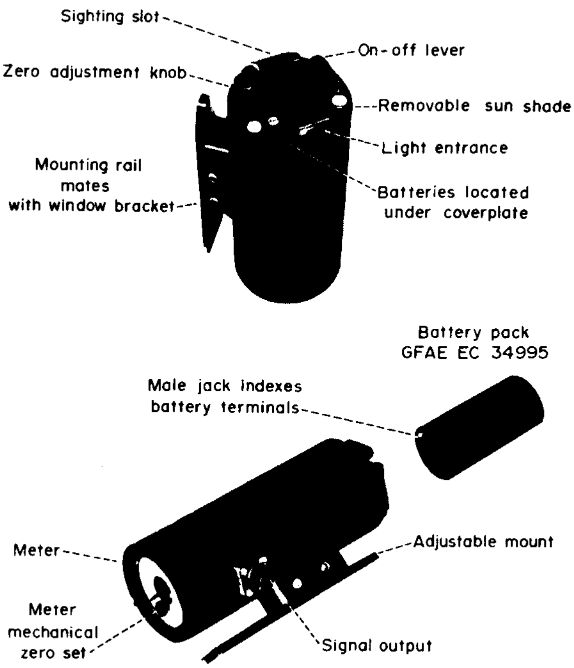

Figure 34–15.—Inflight photometer components.

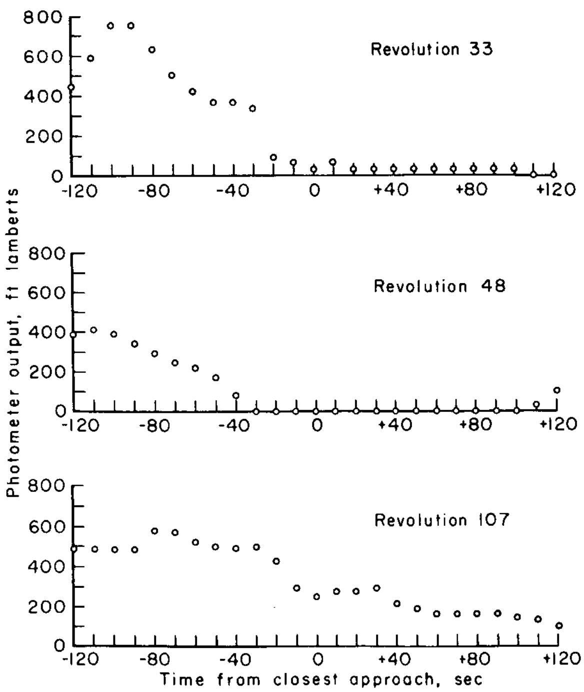

Figure 34–16.—Photometer data for Laredo, Tex., ground observation site.

formation, combined with data on the beam transmittance of the window and on the apparent luminance of the background squares in the ground pattern array, enabled the contrast transmittance of the window at the moment of observation to be calculated. Uniformity of the window could be tested by removing the photometer from its positioning bracket and making a handheld scan of the window, using a black region of space in lieu of the black cavity. A direct-reading meter incorporated in the photometer enabled the command pilot to observe the photometer readings while the pilot scanned his own window for uniformity. A corresponding scan of the command pilot's window could be made in the same way. Data from the photometer were sent to the ground by real-time telemetry. Electrical power for the photometer was provided entirely by batteries within the instrument.

Ground observation sites.—Sites for observations by the crew of Gemini V were provided on the Gates Ranch, 40 miles north of Laredo, Tex. (fig. 34–17), and on the Woodleigh Ranch, 90 miles south of Carnarvon, Australia (figs. 34–18 and 34–19). At the Texas site, 12 squares of plowed, graded, and raked soil 2000 by 2000 feet were arranged in a matrix of 4 squares deep and 3 squares wide. White rectangles of Styrofoam-coated wallboard were laid out in each square. Their length decreased in a uniform logarithmic progression from 610 feet in the northwest corner (square number 1) to 152 feet in the southwest corner (square number 12) of the array. Each of the 12 rectangles was oriented in one of four positions (that is, north-south, east-west, or diagonal), and the orientations were random within the series of 12. Advance knowledge of the rectangle orientations was withheld from the flight crew, since their task was to report the orientations. Provision was made for changing the rectangle orienta-

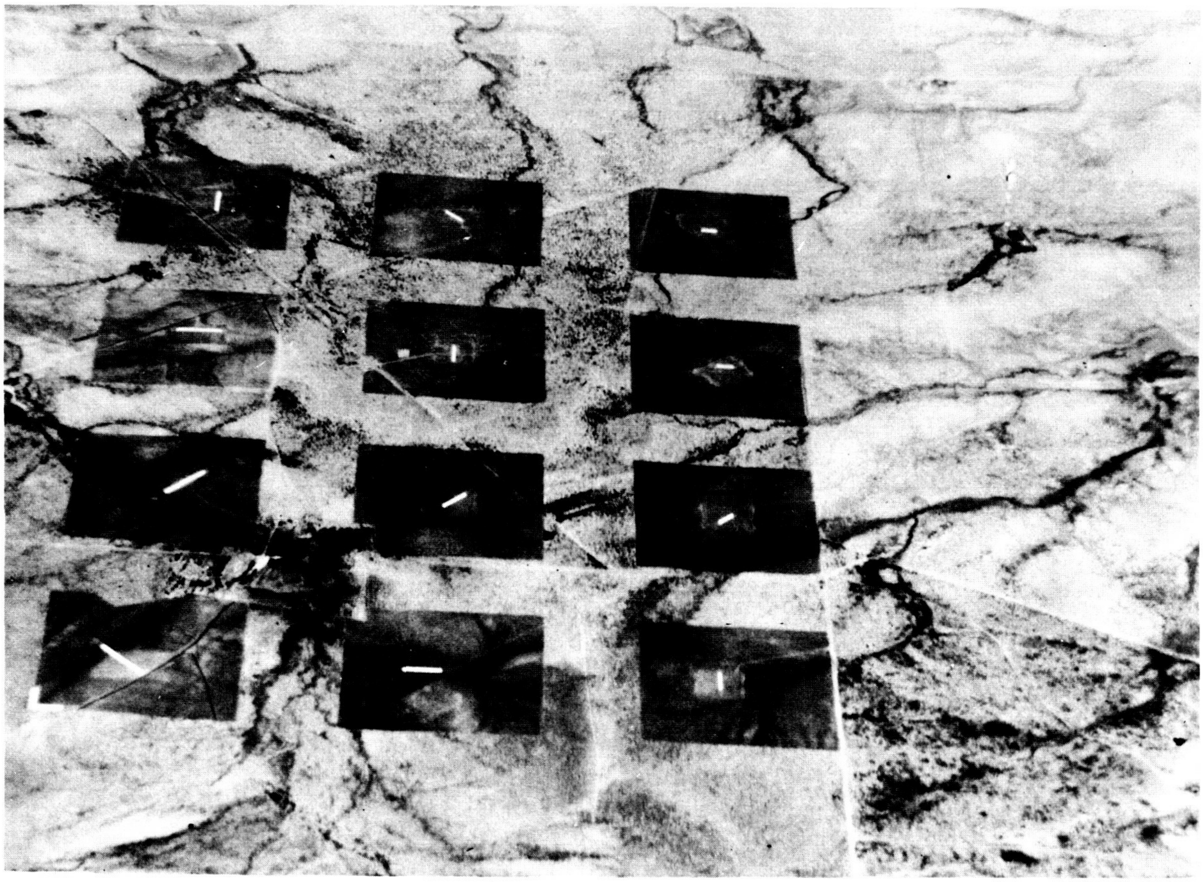

FIGURE 34–17.—Aerial photograph of Gemini V visual acuity experiment ground pattern at Laredo, Tex.

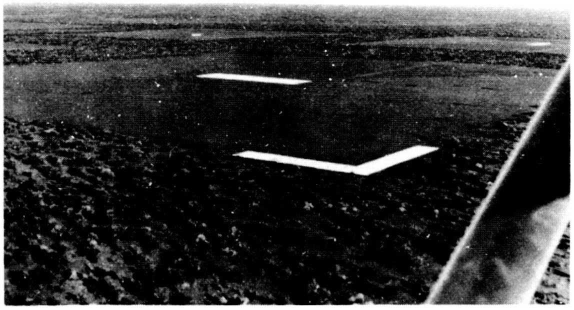

FIGURE 34–18.—Aerial photograph of the Gemini V visual acuity ground observation pattern at Carnarvon, Australia.

tions between passes and for adjusting their size in accordance with anticipated slant range, solar elevation, and the visual performance of the astronauts on preceding passes. The observation site in Australia was somewhat similar to the Texas site, but, inasmuch as no observations occurred there, the specific details are unnecessary in this report.

The Australian ground observation site was not manned during Gemini VII because the afternoon time of launch precluded usable daytime overpasses there until the last day of the mission. The 82.5° launch azimuth used for Gemini VII prevented the use of an otherwise highly desirable ground site in the California desert near the Mexican border. Weather statistics for December made the use of the Texas site appear dubious, but no alternative was available. The afternoon launch made midday passes over this site available on every day of the mission. Experience gained on Gemini V pointed to the need for a more prominent orientation marking. This was provided by placing east-to-west strips of crushed white limestone 26 feet wide and 2000 feet long across the center of each of the four north background squares in the array. Thus, only eight test rectangles were used in a 2 by 4 matrix on the center and south rows of background squares, as shown in figure 34–20. The largest and smallest rectangles were of the same size as those used in Gemini V.

Instrumentation.—Instrumentation at both ground sites consisted of a single tripod-mounted, multipurpose, recording photoelectric

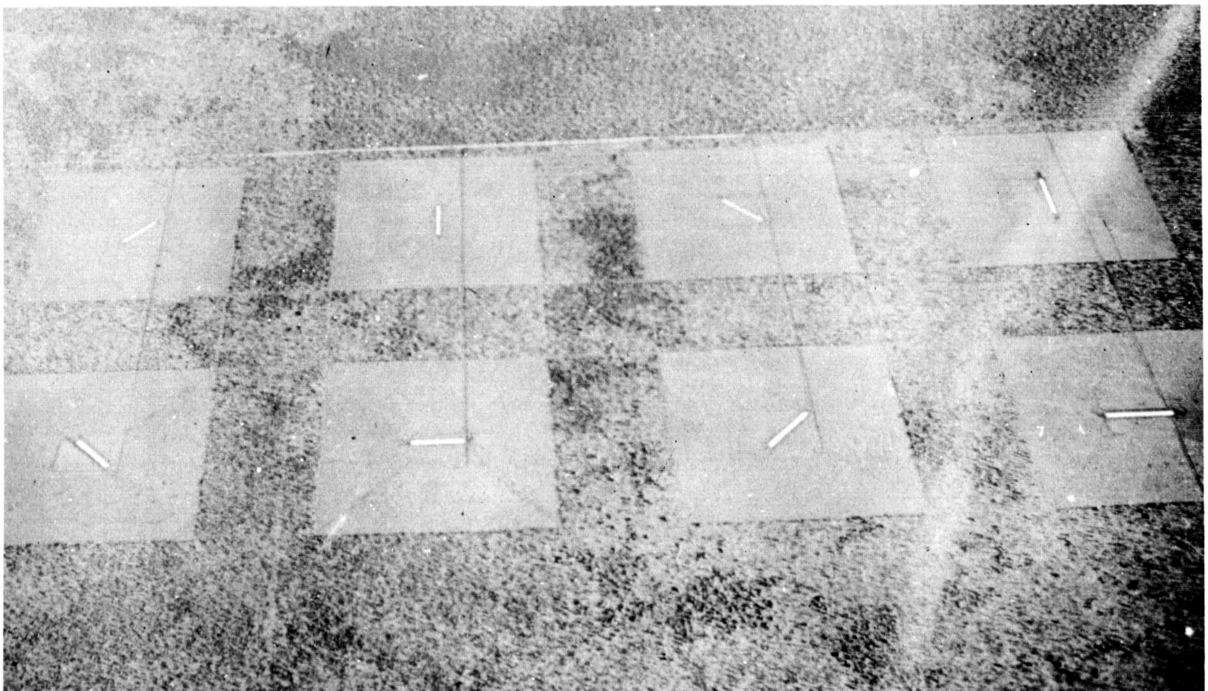

FIGURE 34–19.—Aerial photograph of the Gemini V visual acuity experiment ground pattern at Carnarvon, Australia.

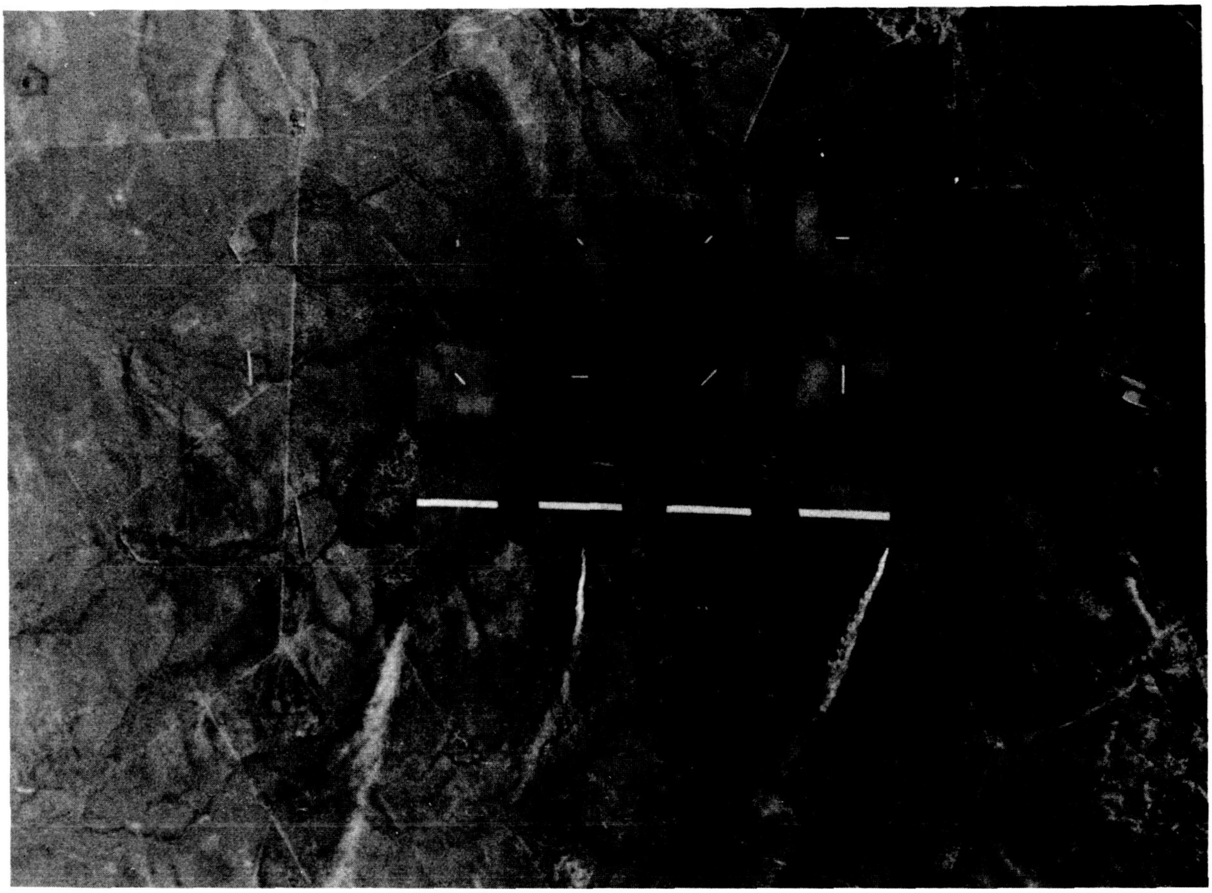

FIGURE 34–20.—Visual acuity experiment ground pattern at Laredo, Tex., as photographed by the Gemini VII flight crew during revolution 17.

photometer (figs. 34–21 and 34–22) capable of obtaining all the data needed to specify the apparent contrast of the pattern as seen from the spacecraft at the moment of observation. The apparent luminance of the background squares needed for evaluation of the contrast loss due to the spacecraft window was also ascertained by this instrument. A 14-foot-high mobile tower, constructed of metal scaffolding and attached to a truck, supported the tripod-mounted photometer high enough above the ground to enable the plowed surface of the background squares to be measured properly. This arrangement is shown in figures 34–23 and 34–24.

Observations in Gemini V

Observation of the Texas ground-pattern site was first attempted on revolution 18, but fuel-cell difficulties which denied the use of the platform were apparently responsible for lack of acquisition of the ground site.

The second scheduled attempt to see the pattern near Laredo was on revolution 33. Acquisition of the site was achieved by the command pilot but not by the pilot, and no readout of rectangle orientation was made.

At the request of the experimenters, the third attempt at Laredo, scheduled originally for revolution 45, was made on revolution 48 in order to secure a higher sun and a shorter slant range. Success was achieved on this pass and is described in the following section.

Unfavorable cloud conditions caused the fourth scheduled observation at the Texas site, on revolution 60, to be scrubbed. Thereafter, lack of thruster control made observation of the ground patterns impossible, although excellent weather conditions prevailed on three scheduled occasions at Laredo (revolutions 75,

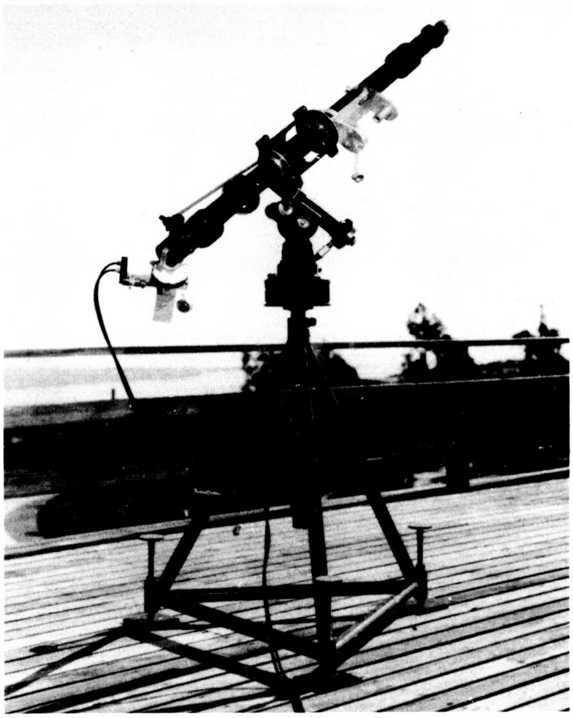

FIGURE 34–21.—Ground-site tripod-mounted photoelectric photometer.

92, and 107) and once at the Australian site (revolution 88). Long-range visual acquisition of the smoke markers used at both sites was reported in each instance, but the drifting spacecraft was not properly oriented near the closest approach to the pattern to enable observations to be made. A fleeting glimpse of the Laredo pattern during drifting flight on revolution 92 enabled it to be photographed successfully with hand cameras. Another fleeting glimpse of the pattern was also reported on revolution 107.

Results of Observations in Gemini V

Quantitative observation of ground markings was achieved only once during Gemini V. This observation occurred during revolution 48 at the ground observation site near Laredo, Tex., at 18:16:14 Greenwich mean time (G.m.t.) on the third day of the flight. Despite early acquisition of the smoke marker by the command pilot and further acquisition by him of the target pattern itself well before the point of closest approach, the pilot could not acquire the markings until the spacecraft had been

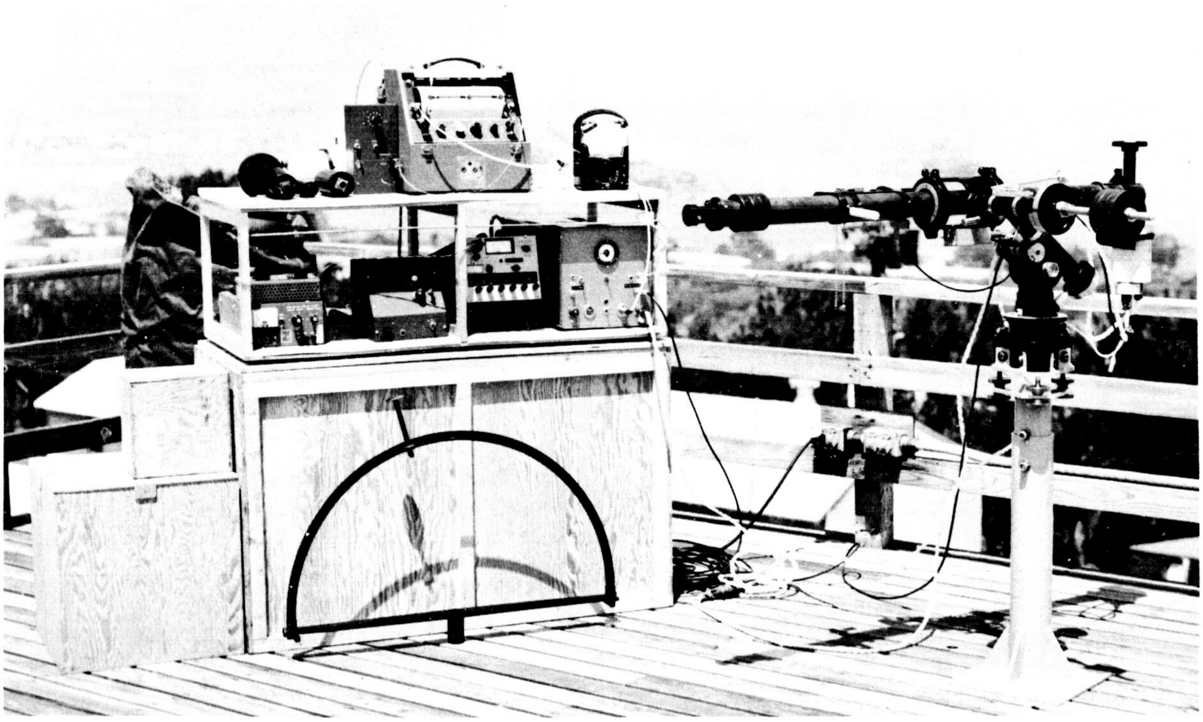

FIGURE 34–22.—Ground-site photoelectric photometer with recording unit.

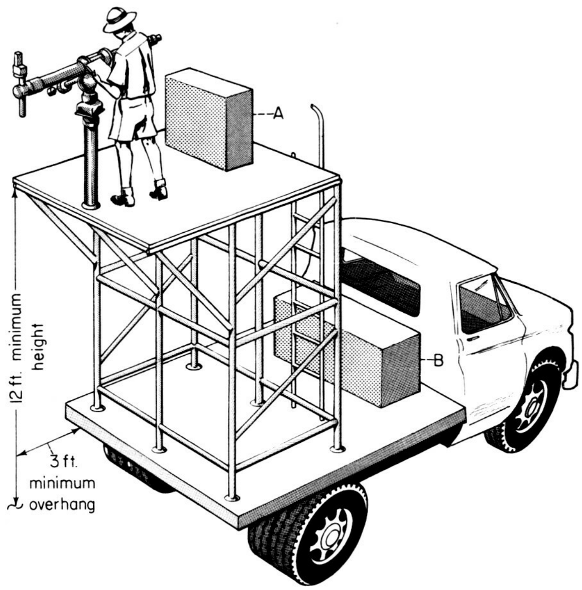

FIGURE 34-23.—Ground-site photoelectric photometer mounted on a truck.

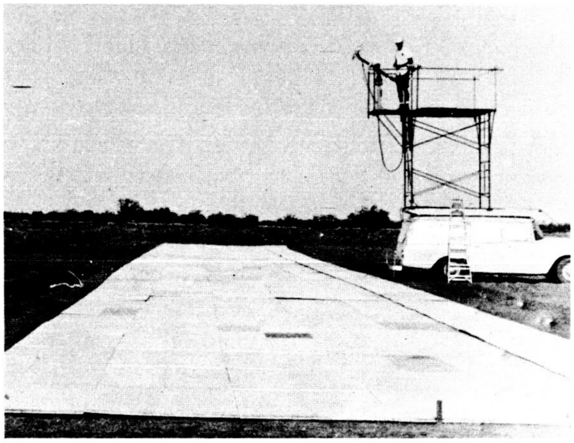

FIGURE 34-24.—Photograph of truck-mounted photoelectric photometer.

turned to eliminate sunlight on his window. Telemetry records from the inflight photometer show that the pilot's window produced a heavy veil of scattered light until the spacecraft was rotated. Elimination of the morning sun on the pilot's window enabled him to make visual contact with the pattern in time to make a quick observation of the orientation of some rectangles. It may be noted that, during approach, the reduction of contrast due to light scattered by the window was more severe than that due to light scattered by the atmosphere.

An ambiguity exists between the transcription of the radio report made at the time of the pass and the written record in the flight log. The writing was made "blind" while the pilot was actually looking at the pattern; it is a diagram drawn in the manner depicted in the Gemini V flight plan, the Mission Operation Plan, the Description of Experiment, and other documents. The orientation of the rectangles in the sixth and seventh squares appears to have been correctly noted. The verbal report given several seconds later correctly records the orientation of the rectangle in the sixth square if it is assumed that the spoken words describe the appearance of the pattern as seen from a position east of the array while going away from the site.

Despite the hurried nature of the only apparently successful quantitative observation of a ground site during Gemini V, there seems to be a reasonable probability that the sighting was a valid indication of the pilot's correctly discriminating the rectangles in the sixth and seventh squares. Since he did not respond to squares 8 through 12, it can only be inferred that his threshold lay at square 6 or higher.

Tentative values of the apparent contrast and angular size of the sixth and seventh rectangles at the Laredo site at the time of the observation are plotted in figure 34-25. The solid line rep-

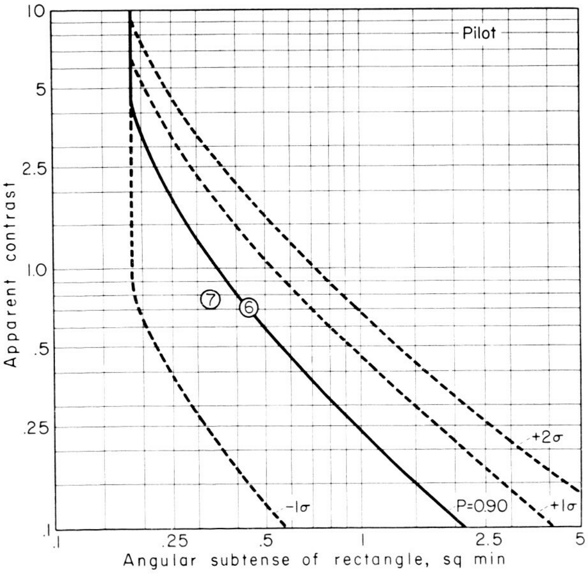

FIGURE 34-25.—Apparent contrast compared with angular size of the sixth and seventh rectangles for revolution 48 of the Gemini V mission.

resents the preflight visual performance of Astronaut Conrad as measured in the vision research van. The dashed lines represent the 1- and 2-sigma limits of his visual performance. The positions of the plotted points indicate that his visual performance at the time of revolution 48 was within the statistical range of his preflight visual performance.

Observations in Gemini VII

Observations of the Texas ground-pattern site were made on revolutions 16, 17, and 31 under very favorable weather conditions. Heavy clouds blanketed the site throughout the remainder of the mission, however, and no further observations of the site were possible. Contamination of the outer surface of the pilot's window made observation of the ground pattern difficult and the result uncertain. The contamination, which was observed to have occurred during launch, was mapped during revolution 19 by means of a window scan with the inflight photometer in the manner described in an earlier section. Figure 34–26 shows some numerical results of this scan, and figure 34–27 is a photograph of a shaded pencil sketch intended to portray the appearance of the window deduced from the telemetered scan curves. Comparison of this sketch with a similar one made by the pilot during flight shows good correlation.

Figures 34–26 and 34–27 show that the command pilot's window was not measurably contaminated on its inboard side. Successful observations of the ground pattern were made by the command pilot through this clear portion of his window on revolutions 17 and 31. No direct sunlight fell on the window during those observations.

Results of Observations in Gemini VII

The results of observations by the command pilot on revolutions 17 and 31 of Gemini VII are shown in figure 34–28. These observations

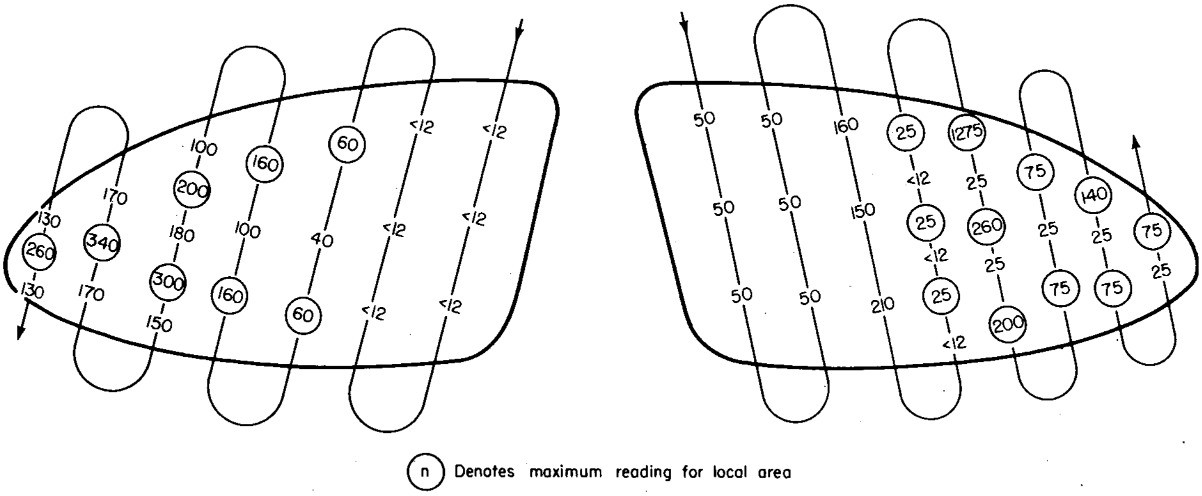

FIGURE 34–26.—Numerical results of window scan.

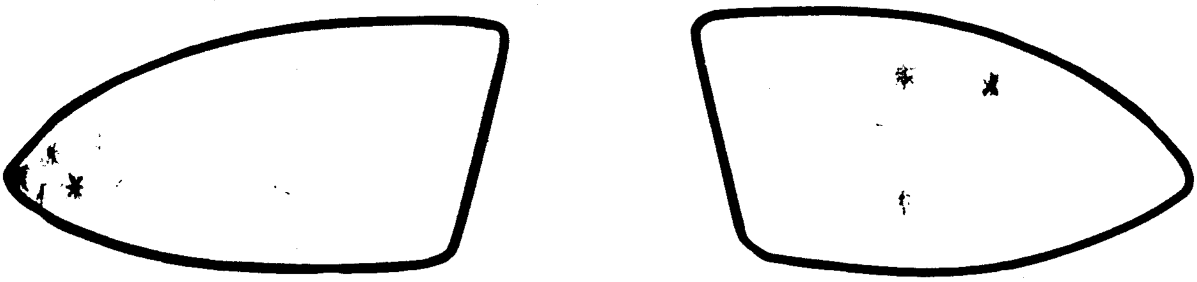

FIGURE 34–27.—Photograph of shaded pencil sketch of window contamination.

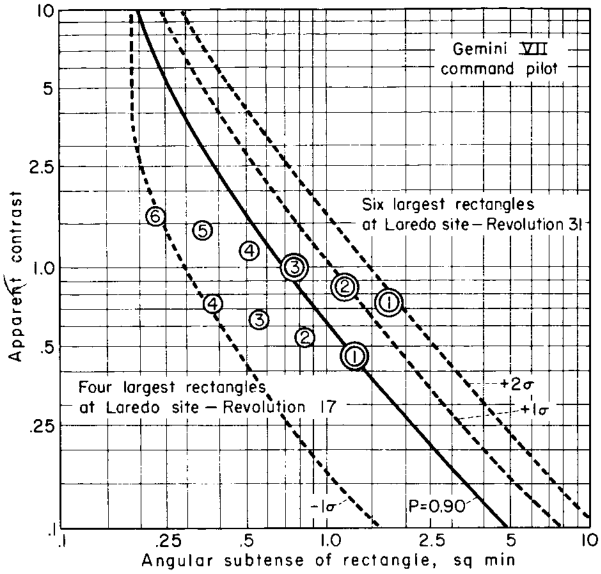

FIGURE 34–28.—Apparent contrast compared with angular size of rectangles.

occurred at 27:04:49 and 49:26:48 ground elapsed time (g.e.t.) on the second and third days of the flight, respectively.

In figure 34–28 the circled points represent the apparent contrast and angular size of the largest rectangles in the ground pattern. Apparent contrast was calculated on the basis of measured directional luminances of the white panels and their backgrounds of plowed soil, of atmospheric optical properties measured in the direction of the path of sight to the point of closest approach, and of a small allowance for contrast loss in the spacecraft window based upon window scan data and readings of the inflight photometer at the time of the two observations. Angular sizes and apparent contrast were both somewhat larger for revolution 31 than for revolution 17 because the slant range was shorter and because the spacecraft passed north of the site, thereby causing the background soil to appear darker, as can be noted by comparing figure 34–20 with figure 34–29. The orientations of those rectangles indicated by double circles were reported correctly, but those represented by single circles were either reported incorrectly or not reported at all.

The solid line in figure 34–28 represents the preflight visual performance of Borman as measured in the vision research van. The dashed lines represent the $-\sigma$, $+\sigma$, and $+2\sigma$ contrast limits of his visual performance. The positions of the plotted points indicate that his visual performance was precisely in accordance with his preflight visual thresholds.

Conclusions

The stated objectives of experiment S–8/D–13 were both achieved successfully. Data from the inflight vision tester show that no change was detected in the visual performance of any of the four astronauts who composed the crews of Gemini V and Gemini VII. Results from observations of the ground site near Laredo, Tex., confirm that the visual performance of the astronauts during space flight was within the statistical range of their preflight visual performance and demonstrate that laboratory visual data can be combined with environmental optical data to predict correctly the limiting visual capability of astronauts to discriminate small objects on the surface of the earth in daylight.

FIGURE 34-29.—Visual acuity experiment ground pattern at Laredo, Tex., as photographed by the Gemini VII flight crew during revolution 31.

35. EXPERIMENT S–5, SYNOPTIC TERRAIN PHOTOGRAPHY

By PAUL D. LOWMAN, JR., Ph. D., *Laboratory for Theoretical Studies, NASA Goddard Space Flight Center*

Introduction

The S–5 Synoptic Terrain Photography experiment was successfully conducted during the Gemini VI–A and VII missions. The purpose of this report is to summarize briefly the methods and results of the experiment. Interpretation of the large number of pictures obtained will, of course, require considerable time, and a full report is not possible now. As in previous reports, representative pictures from the missions will be presented and described.

Gemini VI–A

The purpose of the S–5 experiment in Gemini VI–A was, as in previous Gemini missions, to obtain high-quality color photographs of selected land and near-shore areas for geologic, geographic, and oceanographic study. The oceanographic study is an expansion of the scope of the experiment undertaken at the request of the Navy Oceanographic Office. The camera, film, and filter (Hasselblad 500C, Planar 80-mm lens, Ektachrome SO–217, and haze filter) were the same as used on previous flights. Camera preparation and loading were done by the Photographic Technology Laboratory, Manned Spacecraft Center, as was preliminary identification of the pictures.

The experiment was very successful, especially in view of the changes in mission objectives made after the experiment was planned. About 60 pictures useful for study were obtained. Areas covered include the southern Sahara Desert, south-central Africa, northwestern Australia, and several islands in the Indian Ocean.

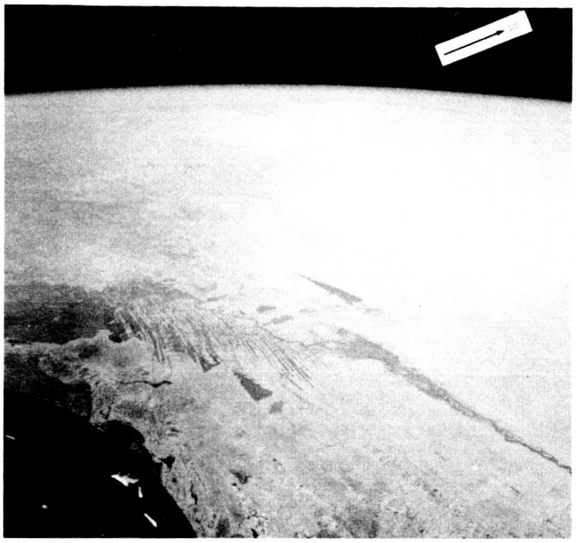

FIGURE 35–1.—Niger River and vicinity of Tombouctou, Mali (view looking northwest).

Figure 35–1, one of a continuous series taken during the 15th revolution, shows a portion of central Mali including the Niger River and the vicinity of Tombouctou. The Aouker Basin and part of the southwestern Sahara Desert are visible in the background. The picture furnishes an excellent view of what are probably stabilized sand dunes (foreground), such as sand dunes which are no longer active and have been partly eroded (ref. 1). These dunes probably represent a former extension of the arid conditions which now characterize the northern Sahara. This photograph and others in the series should prove valuable in the study of the relation of the stabilized dunes to active dunes and to bedrock structure.

Figure 35–2 shows the Air ou Azbine, a plateau in Niger. The dark, roughly circular masses are Cenozoic lava flows on sandstones and schists (ref. 2). The crater at the lower left would appear to be of volcanic origin in view of its nearness to lava flows, but Raisz (ref. 2) indicates this area to be capped by sandstone. The picture gives an excellent view of the general geology and structure of the uplift as a whole.

Figure 35–3, one of several extremely clear pictures of this region, was taken over Somalia in the vicinity of the Ras Hafun (the cape at left). The area is underlain by Cenozoic

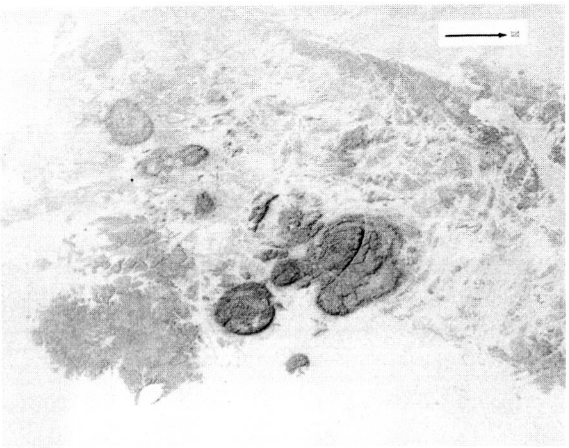

FIGURE 35–2.—Air ou Azbine, volcanic plateau in Niger.

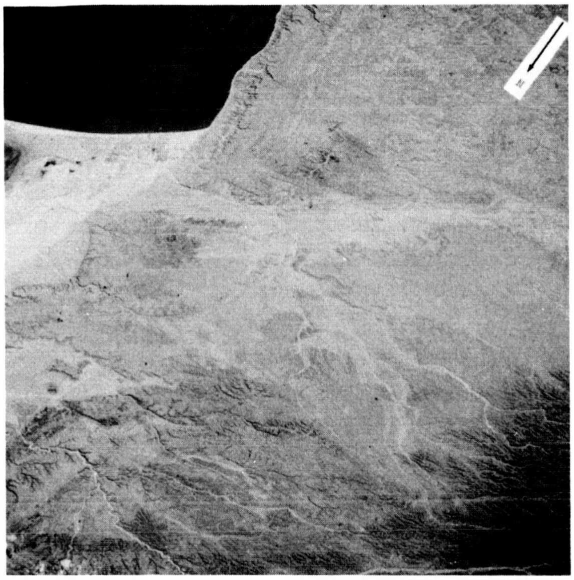

FIGURE 35–3.—Indian Ocean coast of Somalia, with Ras Hafun at left (north at bottom).

marine and continental sedimentary rock (ref. 3), and appears to be relatively recently emerged. As such, it furnishes an excellent opportunity to study development of consequent drainage, since much of the area is in a youthful stage of geomorphic development.

Figure 35–4 shows several lakes in the portion of the Rift Valley south of Addis Ababa, Ethiopia. Considerable structural detail is visible, such as the presumably fracture-controlled drainage on the east side of the Rift Valley. In addition, several areas of volcanic rock can be distinguished. This photograph may be helpful in testing Bucher's suggestion

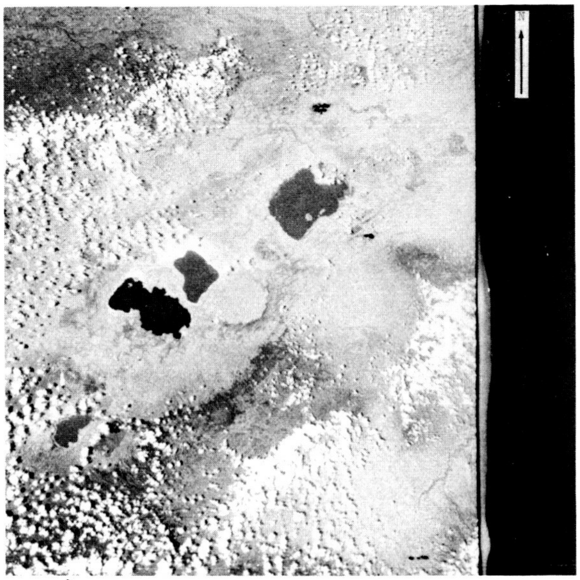

FIGURE 35–4.—Lakes in the Rift Valley, Ethiopia, south of Addis Ababa.

(ref. 4) that vulcanism in the Rift Valley is independent of structure. This area is in any event of great geologic interest and is a prime subject of study during the Upper Mantle Project (ref. 5).

Gemini VII

The scope of the terrain photography experiment (S–5) was considerably expanded for the Gemini VII mission because of the much greater mission length, and the greater amount of film capacity available. Requests had been received for photography of a number of specific areas from Government agencies, such as the U.S. Geological Survey, and from universities, and these were incorporated into the flight plan. The Hasselblad 500C and Ektachrome SO–217 again were the major equipment items, but, in addition, a Zeiss Sonnar 250-mm telephoto lens and Ektachrome infrared, type 8443, film were carried.

The experiment was highly successful. Approximately 250 pictures usable for geologic, geographic, and oceanographic purposes were obtained, covering parts of the United States, Africa, Mexico, South America, Asia, Australia, and various ocean areas. However, two major difficulties hampered the experiment. First, the cloud cover was exceptionally heavy over many of the areas selected. Second, a deposit was left on the spacecraft windows, apparently from second-stage ignition; this deposit seriously degraded a number of the pictures. The large number of usable pictures obtained is a tribute to the skill and perseverance of the crew.

Figure 35–5 is one of a series taken over the southern part of the Arabian peninsula. The series provides partial stereoscopic coverage. The area shown, also photographed during the Gemini IV mission, is the Hadramawt Plateau with the Hadramawt Wadi at lower right. The plateau is underlain by gently dipping marine shales (Geologic Map of the Arabian Peninsula, U.S. Geological Survey, 1963) deeply dissected in a dendritic pattern. Several interesting examples of incipient stream piracy are visible, in which streams cutting headward intersect other streams. (All are, of course, now dry.)

Figure 35–6 was taken over Chad, looking to the southeast over the Tibesti Mountains. This photograph was specifically requested to investigate geologic features discovered on Gemini IV photographs (ref. 6). One of these features is the circular structure at far left center. Although probably an igneous intrusion, such as a laccolith, its similarity to the Richat structures suggests that an impact origin be considered. Another structural feature whose significance is currently unknown is the series of concentric lineaments at far left. These are probably joints emphasized by wind and stream erosion, and may be tensional fractures associated with the epeirogenic uplift of the Tibesti massif. In addition to these structures, considerable detail can be seen in the sedimentary, igneous, and metamorphic rocks of the western Tibestis. The large circular features are calderas, surrounded by extensive rhyolite or ignimbrite deposits (ref. 7).

Figure 35–7, since it was taken with the 250-mm lens, is of considerable interest in evaluating the usefulness of long-focal-length lenses. The area covered is the Tifernine Dunes (ref. 2) in south-central Algeria. Despite the longer focal length, the region included in the picture is about 90 miles from side to side because of the camera tilt. The picture provides a synoptic view of the dune field and its relation to surrounding topography, which should prove valuable in studies of dune formation.

Figure 35–8 shows a portion of the Erg Chech in west-central Algeria, looking to the southeast. The dark ridges at the lower left are the Kahal Tabelbala and Ougarta, folded Paleozoic sandstones, limestones, and schists (ref. 8), separated by the Erg er Raoui, a dune field. Of considerable interest is the variety of dunes in the lower right. At least two major directions of dune chains at high angles to each other are visible, suggesting a possible transition from transverse to longitudinal dunes.

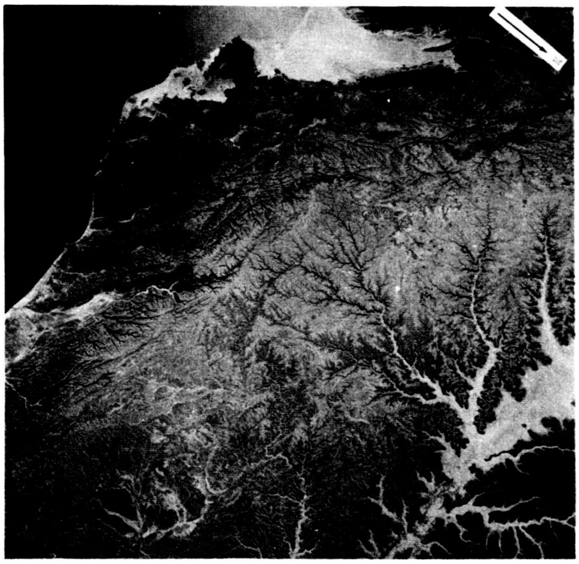

FIGURE 35–5.—Nearly vertical view of the Hadramawt Plateau, south coast of the Arabian Peninsula (north to right).

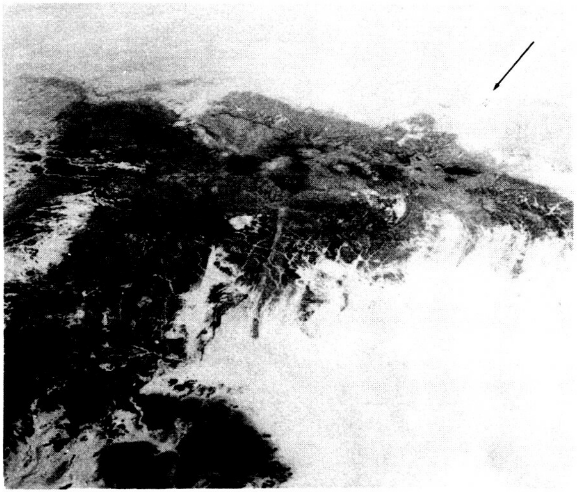

FIGURE 35–6.—Tibesti Mountains, Chad (view looking to southeast).

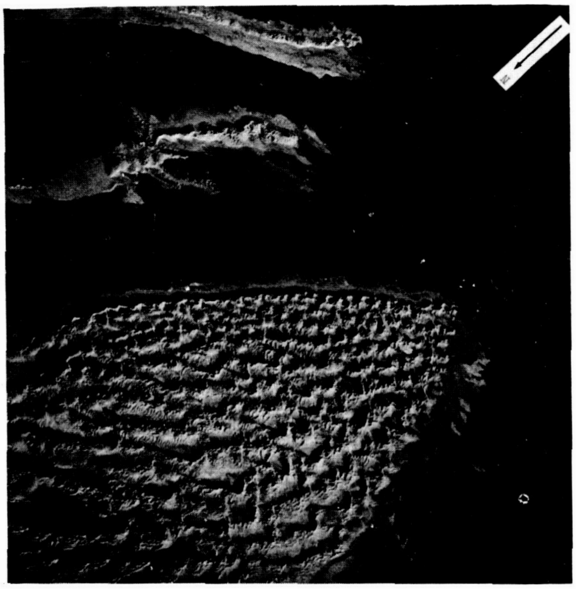

FIGURE 35–7.—Tifernine dune field, Algeria (view looking to southeast).

FIGURE 35–8.—Part of the Erg Chech, Algeria, and the Erg er Raoui (view looking to southeast).

The value of such photographs in the study of sand dune formation and evolution is obvious.

Figure 35–9 is one of several taken with color infrared film, used for the first time in scientific terrain photography on this flight. Despite the obscuration of the window caused by the previously mentioned deposit and the artifacts at right, the picture demonstrates strikingly the

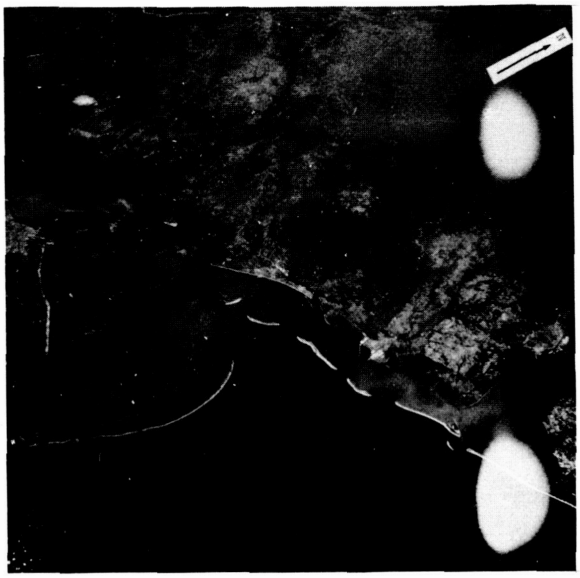

FIGURE 35–9.—Black-and-white of color photograph taken with infrared film over Gulf of Mexico (view looking northwest over Mobile Bay-New Orleans coast).

potential value of this type of film for hyper-altitude photography.

The area shown in figure 35–9 includes the Gulf coast of Alabama, Mississippi, and Louisiana; Mobile Bay is at lower right, and Lake Ponchartrain and New Orleans at far left. The arc at left center is the Chandeleur Island chain. The picture is notable for several reasons. First, the infrared sensitivity provides considerable haze-penetrating ability, as had been expected from the behavior of black-and-white infrared films flown on rockets (ref. 9). This is shown by the fact that highways can be distinguished at slant ranges of about 200 miles (at upper left: probably Interstate 55 and Route 190). Other cultural features include additional highways, the bridge carrying Interstate 59 across the east end of Lake Ponchartrain (the causeway, however, is not visible), and the Mississippi River-Gulf outlet canal (the white line crossing the delta parallel to the left border).

Many color differences can be seen in the Gulf of Mexico and adjoining inland waters. There appears to be considerable correspondence between water color and depth, as suggested in a report being prepared by R. F. Gettys. For example, the dark tonal boundary just above the spacecraft nose (lower left) may outline the 60-fathom contour as shown on Coast and Geodetic Chart 1115. Also, the tone contours just east of the Mississippi Delta at lower left correspond roughly to the depth of water between the delta and Breton Island. However, it is probable that temperature of the water and overlying air influence the color response of this film, and more detailed analysis is needed.

Considerable color detail is visible in land areas. Differences are probably the expression of vegetation rather than soil or geologic units, since the expected color response (for example, red replacing green) is present on the color prints. It is obvious, from this and adjoining pictures, that much more color discrimination is possible with color infrared film than with conventional color film. This fact is of great importance for the application of hyperaltitude photography to range management, forestry, and agriculture, since terrain photography on previous Gemini flights has shown that the color response of conventional color film in green wavelengths is poor, probably due to atmospheric scattering.

Summary

The following results have been achieved during the terrain photography on the Gemini IV and VII missions:

(1) New areas not previously photographed have been covered.

(2) Coverage of previously photographed areas has been extended or improved.

(3) The value of color infrared film in hyperaltitude photography has been demonstrated.

(4) The effectiveness of moderately long focal lengths has been demonstrated.

The experiment on both missions has been highly successful, despite the difficulties encountered.

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36. EXPERIMENT S-6, SYNOPTIC WEATHER PHOTOGRAPHY

By KENNETH M. NAGLER, *Chief, Space Operations Support Division, Weather Bureau, Environmental Science Services Administration, and* STANLEY D. SOULES, *National Environmental Satellite Center, Environmental Science Services Administration*

Summary

The weather photography experiment conducted in the Gemini IV, V, VI-A, and VII missions resulted in a total of nearly 500 high-resolution color photographs showing clouds. Many of these illustrate interesting meterological features on a scale between that obtainable from surface or aircraft views, and that obtainable from operational weather satellites.

Description

The S-6 weather photography experiment represents an effort to get a selection of high-resolution color photographs of interest to the meteorologist.

The pictures obtainable from the altitude of the Gemini flights provide details on a scale between that of views from the ground or aircraft and that from weather satellites. When the Gemini photographs are taken approximately vertically, every cloud is plainly visible over an area approximately 100 miles square. At oblique angles, much larger areas can be seen in considerable detail. Such views are illustrative of, and can assist in, the explanation of various meteorological phenomena. Also, they are an aid in the interpretation of meteorological satellite views, which are sometimes imperfectly understood.

The equipment for the experiment has been relatively simple. It consists of the Hasselblad camera (Model 500C, modified by NASA) with a haze filter on the standard Zeiss Planar 80-mm f/2.8 lens. The film (70-mm) has been for the most part Ektachrome MS (SO-217), although one roll of Anscochrome D-50 film was used on the Gemini V flight. Also, the infrared Ektachrome film used on Gemini VII primarily for other purposes yielded some meteorologically interesting pictures.

The procedures for conducting the experiment were essentially the same on the four missions. Well in advance of the flights, a number of meteorologists (primarily from the National Environmental Satellite Center and the Weather Bureau) were questioned as to the types of cloud systems they would like to see, and as to what particular geographical areas were of interest. Several months before each flight, the aims of the experiment were discussed in detail with the flight crew. A number of specific types of clouds were suggested as possibilities for viewing on each mission.

The mission plans were arranged so that the pilots could devote part of their time to cloud photography over the preselected areas. On the day preceding each launch, the pilots were briefed on interesting features likely to be seen on their mission. During the mission, areas of interest were selected from time to time from weather analyses and from Tiros pictures. When operationally feasible, this information was communicated to the crew from the Manned Spacecraft Center at Houston, Tex., in time for them to locate and to photograph the clouds in question, provided this did not interfere with their other duties. So long as fuel was available for changing the attitude of the spacecraft for this purpose, the pilots were able to search for the desired subjects. Otherwise, they could take pictures only of those scenes which happened to come into view.

Results

In all, close to 500 high-quality pictures containing clouds or other meteorologically significant information were taken by the crews on Gemini IV, V, VI-A, and VII missions. Many of the aims of the experiment were realized; naturally, with the variety and the infrequent occurrence of some weather systems, and with the crew's other activities and constraints, some meteorological aims were not realized.

The results of the Gemini IV and Gemini V missions have been discussed previously by Nagler and Soules (refs. 1, 2, and 3).

Before mentioning specific features of interest, it should be pointed out that many views, while not scientifically significant, do illustrate cloud systems of many types in color and with excellent resolution. These make a valuable library for educational and illustrative purposes. Some of the categories of meteorologically interesting views obtained on these Gemini flights are described below.

Organized Convective Activities

In all of the flights there were views illustrating cloud fields which resulted from organized convection under a variety of meteorological conditions. These included the cumulus cloud streets, long lines of cumulus clouds parallel to the windflow, as illustrated in figure 36-1. Also, some scenes show a broad pattern of branching cumulus streets. Another type of convection pattern, occurring when there is little shear throughout the cloud layer, is the cellular pattern. In these patterns, sometimes the rising motion, as indicated by the presence of clouds, is in the center of the cells with descending motion near the edges, as in figure 36-2; and sometimes the circulation is in the opposite sense.

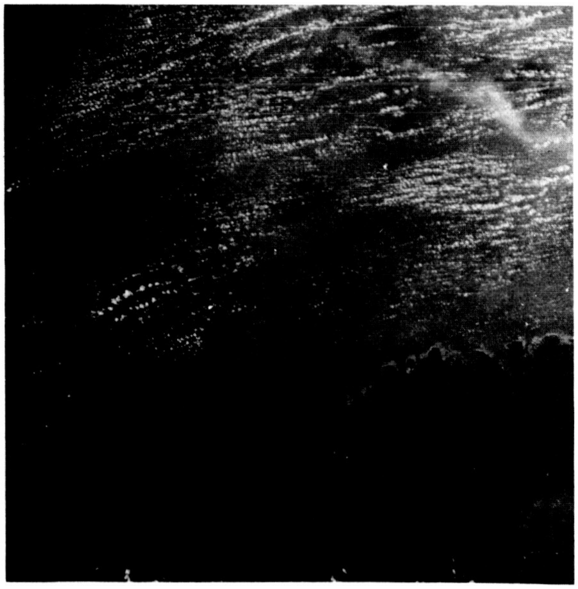

FIGURE 36-1.—Typical cumulus cloud streets in the South Atlantic Ocean near the mouth of the Para River, Brazil. Photographed by Gemini VII flight crew at 19:53 G.m.t., December 12, 1965.

Eddy Motions

Vortices induced by air flowing past islands or coastal prominences have also been photographed on the Gemini flights. Figure 36-3 shows a vortex of the latter type. Views of such eddies on successive passes, to show how they move and change, were not obtained and remain a goal for future missions.

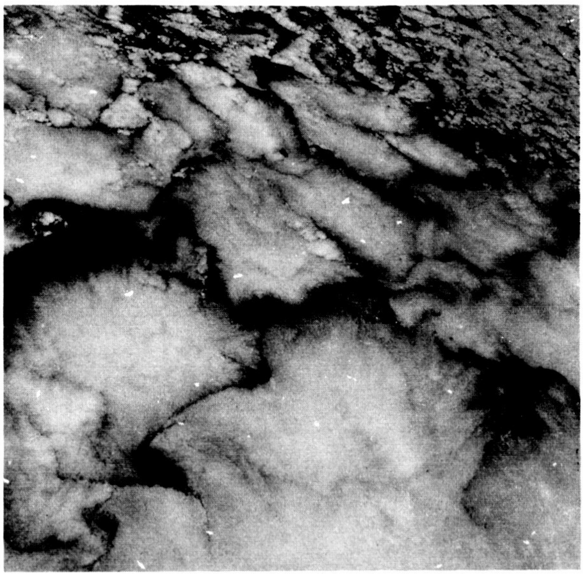

FIGURE 36-2.—Cellular cloud patterns over the Central North Pacific Ocean, showing small vortices along the boundaries. Photographed by Gemini IV flight crew at 22:29 G.m.t., June 4, 1965.

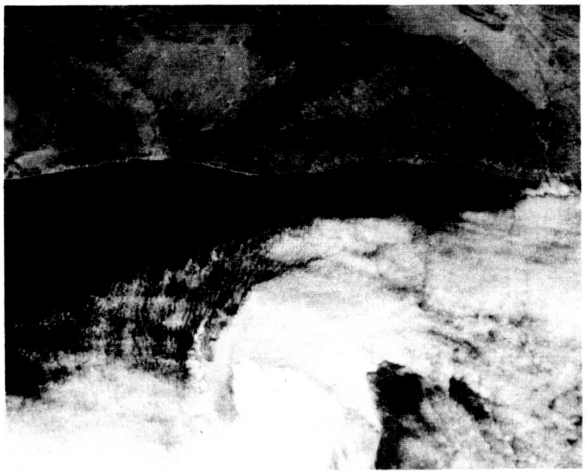

FIGURE 36-3.—Vortex in stratocumulus clouds off Morocco, induced by strong northeasterly winds flowing past Cape Rhir just north of this scene. Photographed by Gemini V flight crew at 10:25 G.m.t., August 26, 1965.

Tropical Storms

Views of tropical storms are naturally of interest to the meteorologist. A number of such views were obtained, ranging from small incipient disturbances to mature storms.

Daytime Cloudiness Over Land

Many of the pictures illustrate, as do many meteorological satellite pictures, the nature of cumulus clouds over land areas during the daytime. Of particular interest in this regard are the views of Florida (figs. 36–4, 36–5, and 36–6) obtained on three successive passes approximately 90 minutes apart. These show the changes and movements of such clouds.

Cirrus Clouds Relative to Other Cloud Decks

Sometimes on meteorological satellite views the determination as to whether the clouds present are high (cirrus) or lower (altostratus or stratus) clouds is a difficult one. The suggestion is often present that dark areas on such pictures may be shadows of cirrus clouds on lower decks. Sometimes, by their orientation, the long dark lines present give an indication of the direction of the winds at the cirrus level, since cirrus clouds in the strong wind core of the upper troposphere (jetstream) frequently occur in long bands parallel to the winds. In the Gemini VI–A and VII flights, several examples of such cirrus shadows on lower clouds were obtained, one of which is shown in figure 36–7.

FIGURE 36–4.—View of Florida showing cumulus clouds over the land, the first of three views of this area taken on successive passes. Photographed by Gemini V flight crew at 15:31 G.m.t., August 22, 1965.

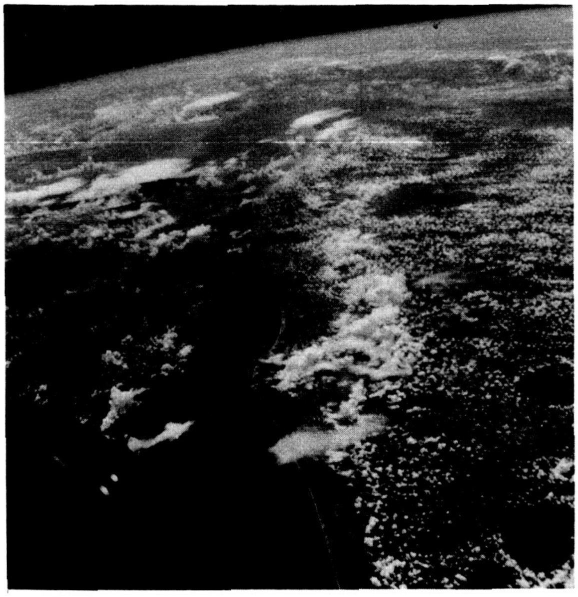

FIGURE 36–5.—Florida, the second of three views of this area, showing increased cumulus cloud development along a line just inland from the east coast. Photographed by Gemini V flight crew at 17:07 G.m.t., August 22, 1965.

FIGURE 36–6.—Florida, the third of three views taken on successive passes showing that the cumulus activity had developed to the cumulonimbus (thunderstorm) stage just inland in the Cape Kennedy area. Photographed by Gemini V flight crew at 18:38 G.m.t., August 22, 1965.

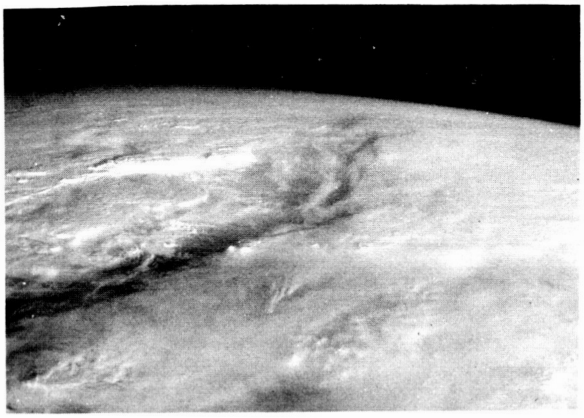

FIGURE 36–7.—Cirrus shadows on lower cloud layers, over the North Atlantic Ocean. Photographed by Gemini VI flight crew at 10:38 G.m.t., December 16, 1965.

Other Phenomena

Pictures of features other than clouds, often obtained from the S–5 synoptic terrain photography experiment which uses the same camera and film as S–6, sometimes are of interest in meteorology and related fields. For example, smoke from forest fires or from industrial sources may indicate the low-level wind direction and may yield quantitative information on the stability of the lower atmosphere. Sand dunes of various types are of interest to those working on the relationship between winds and deposition patterns. One of many dune scenes is shown in figure 36–8. Similarly, the configuration of bottom sand in some shallow water areas can be related to motions in the ocean. Figure 36–9 is one of several views of the ocean bottom in the Bahama Islands area. Also, the differences in the reflectivity of wet and dry soils can be related to the occurrence of recent rainfall (ref. 4). Figure 36–10 shows the dark area resulting from heavy rains in the previous 24 hours.

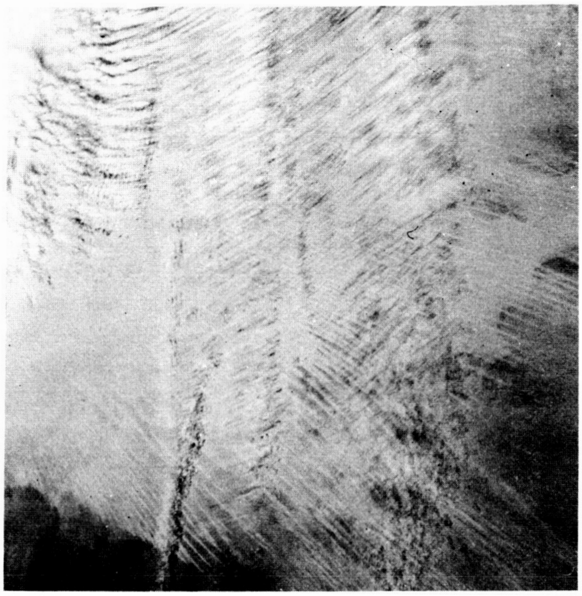

FIGURE 36–8.—Seif dunes in the northwestern Sudan, with a banded cloud structure above, one of a number of views of dune formations taken on the Gemini flights. Photographed by Gemini VII flight crew at 12:02 G.m.t., December 11, 1965.

Conclusion

In conclusion, through the skill of the crews of various Gemini missions, and the assistance of many NASA individuals working in the experiments program, a great many excellent, useful pictures of the earth's weather systems have been obtained; however, weather systems are extremely variable, and there remain a number of interesting views or combinations of views which it is hoped will be obtained on future manned space flights over regions of the earth, both within and outside the equatorial zone.

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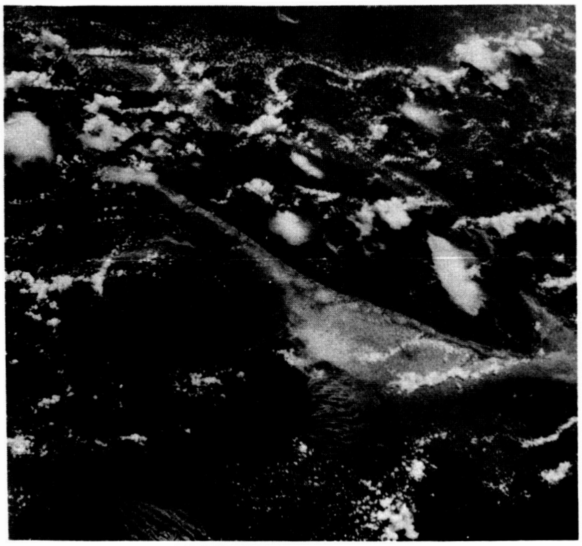

FIGURE 36-9.—Great Exuma Island in the Bahamas, showing the bottom configuration in the shallow water areas. Photographed by Gemini V flight crew at 18:39 G.m.t., August 22, 1965.

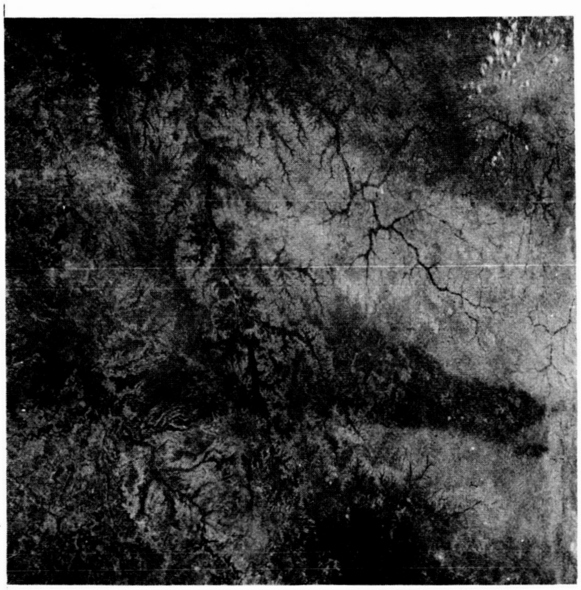

FIGURE 36-10.—Terrain shading in central Texas, caused by heavy rainfall the previous day. The highway prominent in the upper left corner connects Odessa and Midland. The stream in the center of the picture is the North Concho River along San Angelo. Photographed by Gemini IV flight crew at 17:46 G.m.t., June 5, 1965.

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37. EXPERIMENTS MSC–2 AND MSC–3, PROTON/ELECTRON SPECTROMETER AND TRI-AXIS MAGNETOMETER

By JAMES R. MARBACH, *Advanced Spacecraft Technology Division, NASA Manned Spacecraft Center, and* WILLIAM D. WOMACK, *Advanced Spacecraft Technology Division, NASA Manned Spacecraft Center*

Introduction

Experiments MSC–2 and MSC–3 were the first of a continuing series of measurements of particles and fields conducted by the Radiation and Fields Branch at the Manned Spacecraft Center (MSC) in support of its shield verification and dose prediction program for all manned spacecraft. The simultaneous measurement of the external radiation environment and the radiation dose received by the flight crew throughout a space mission serves to evaluate and perfect calculational techniques, whereby the dose to be received by the crew on any given mission can be estimated prior to that mission.

Instrumentation

The specific function of the MSC–2 and MSC–3 instrumentation was to respectively provide an accurate picture of the proton and electron intensities and energies, and the direction and magnitude of the earth's magnetic field during selected portions of the Gemini IV and Gemini VII missions. The MSC–3 experiment was actually flown in support of MSC–2 to provide the instantaneous direction of the earth's magnetic field relative to the spectrometer. This information was needed in the reduction of MSC–2 data since the particle intensities encountered are strongly directional with respect to the magnetic field. The Gemini IV mission employed a pulse height analyzer with plastic scintillator in an anticoincidence arrangement for the proton/electron measurement. Internal gain shifting techniques provided alternate measurements of the proton and electron environment every 13 seconds. The instrument monitored electrons of $0.4<E<8$ MeV and protons of $25<E<80$ MeV at fluxes between 0 and 3×10^5 particles/cm^2-sec. The MSC–3 experiment on Gemini IV utilized a tri-axial flux gate magnetometer to detect the direction and amplitude of the earth's magnetic field over the range of 0 to 60 000 gammas.

The Gemini VII spectrometer utilized the same pulse height analyzer technique as on Gemini IV except the anticoincidence scintillator was replaced with a thin dE/dx plastic wafer over the instrument entrance aperture. This modification allowed the measurement of protons of $5<E<18$ MeV instead of $25<E<80$ MeV. The electron range and flux-handling capability were the same as those on Gemini IV, and again protons and electrons were measured alternately in time. The Gemini VII magnetometer was identical to that on Gemini IV. Figures 37–1 through 37–5 show the instruments as employed on both spacecraft.

Gemini IV Data

Both experiments were operated at the same time throughout the Gemini mission and were scheduled for turn-on during passes that provided maximum coverage through the South

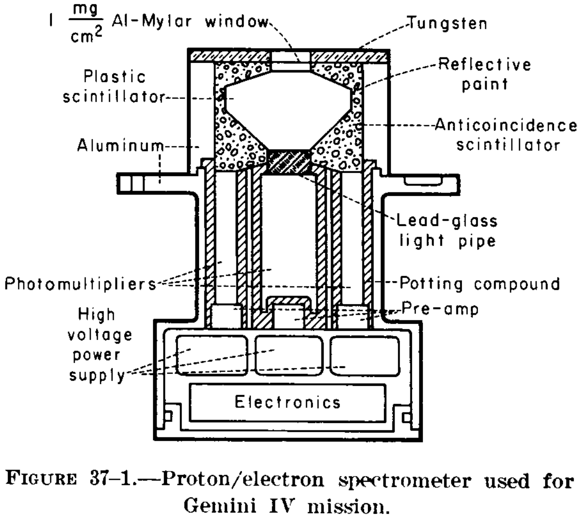

FIGURE 37–1.—Proton/electron spectrometer used for Gemini IV mission.

Anomaly Region between South America and Africa. This region (bounded approximately by 30° E and 60° W longitude and 15° S and 55° S latitude) is the only portion of the spacecraft trajectory that presents any significant proton and electron intensities.

Figure 37–6 is an intensity time history for a typical pass through the anomaly. This particular revolution has been converted to true omnidirectional flux and shows a peak counting

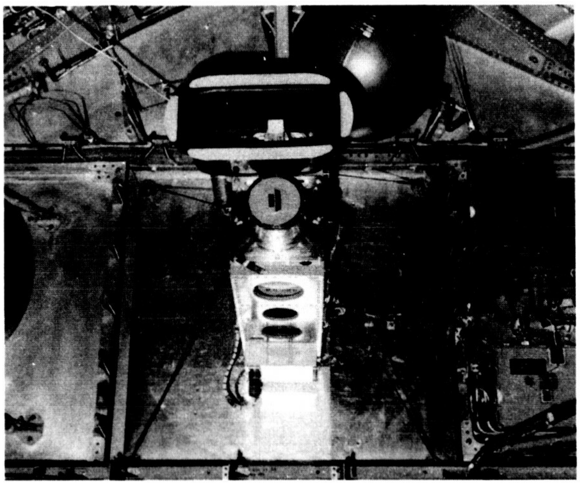

FIGURE 37–2.—Location of proton/electron spectrometer in Gemini IV spacecraft adapter assembly.

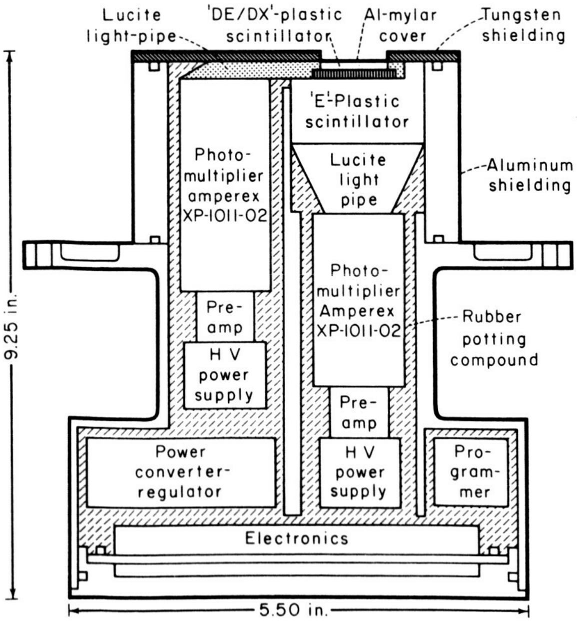

FIGURE 37–3.—Proton/electron spectrometer used for Gemini VII mission.

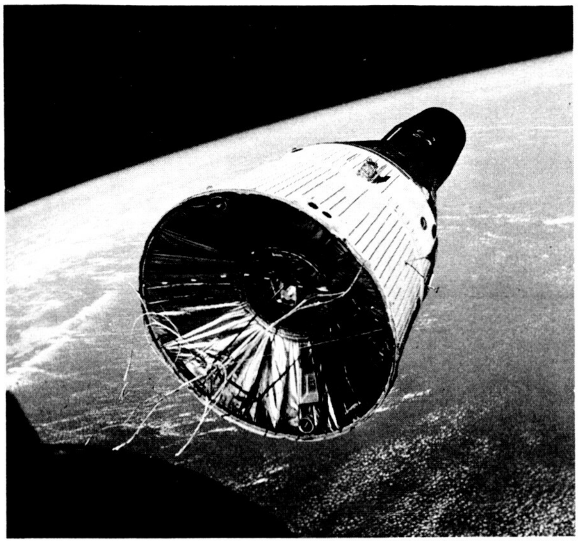

FIGURE 37–4.—Location of proton/electron spectrometer in Gemini VII spacecraft.

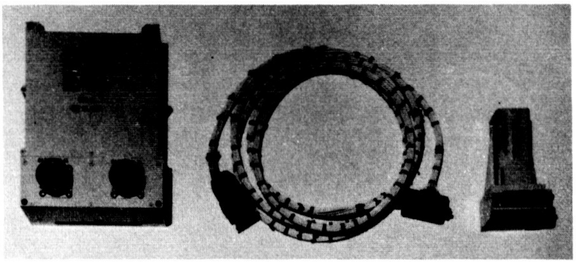

FIGURE 37–5.—Magnetometer used for Gemini IV and VII missions.

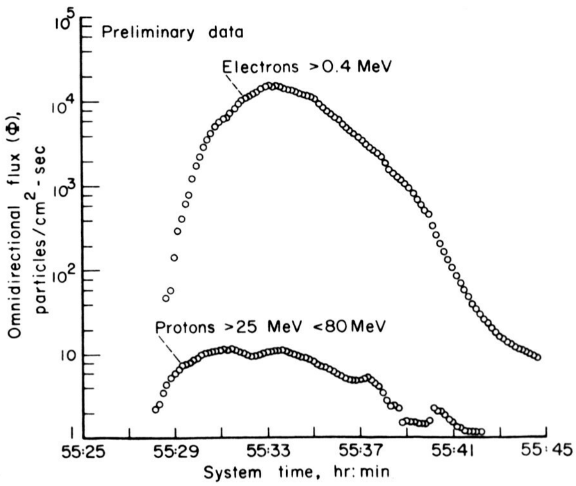

FIGURE 37–6.—Flux compared with time for revolution 36 of Gemini IV mission.

rate of about 10^4 electrons/cm²-sec and 10 protons/cm²-sec. Peak counting rates encountered never exceeded about 6×10^4 for electrons and 10^2 for protons. Figure 37–7 shows characteristic electron spectra observed through one anomaly pass. As is evident in the figure, the spectrum changes significantly through the anomaly. Figure 37–8 depicts the proton spectrum for the same pass. The change in shape here is much more subtle.

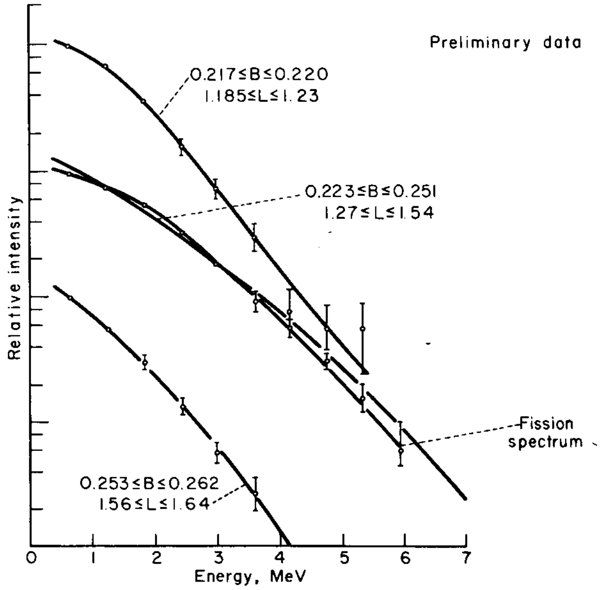

FIGURE 37–7.—Characteristic electron spectra for revolution 36 of Gemini IV mission.

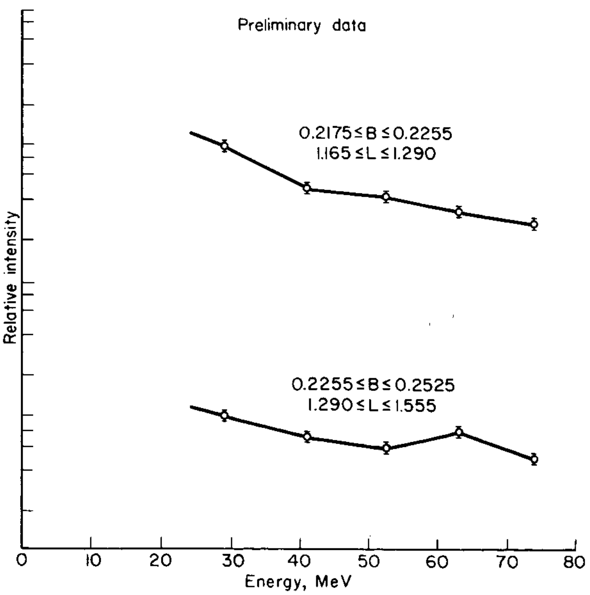

FIGURE 37–8.—Characteristic proton spectra for revolution 36 of Gemini IV mission.

Figure 37–9 is a plot of magnetometer data that were typical throughout most of the mission. The strongly varying direction of the field lines, with respect to the spacecraft during revolutions 7 and 22, was due almost entirely to the tumbling motion of the spacecraft, which was free to drift in pitch, roll, and yaw throughout most of the mission. Revolution 51 is a pass during which the pilot held pitch, roll, and yaw as close to zero as possible. Figure 37–10 shows the total field strength measured during revolution 51 as compared with the theoretical values predicted for this region using the computer technique of McIlwain. The difference is attributed to small errors in the measurement due to stray magnetic fields from the spacecraft. In order to check this assumption, the total field intensity values, as predicted by McIlwain, were assumed to be correct, and the three axes were appropriately corrected so that the measured total field agreed with the predicted values. These corrected values are also plotted in figure 37–10. Figure 37–11 is a plot of the total field direction as

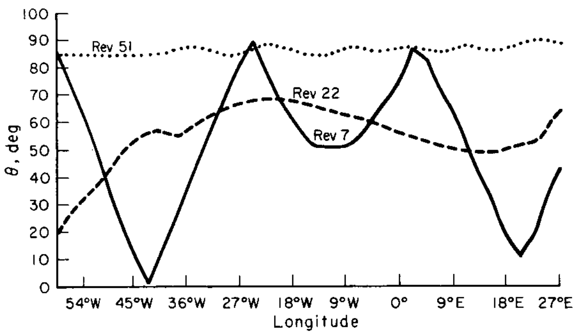

FIGURE 37–9.—Direction of magnetic field during Gemini IV mission.

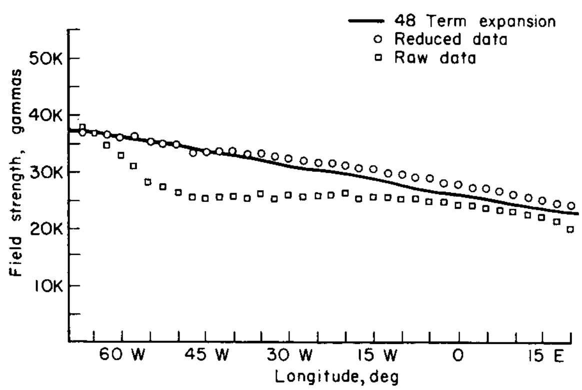

FIGURE 37–10.—Field strength measured during revolution 51 of Gemini IV mission.

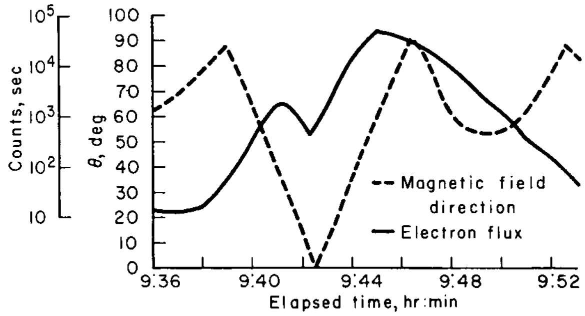

FIGURE 37-11.—Correlation of Experiment MSC-2 and Experiment MSC-3 data for revolution 7 of Gemini IV mission.

measured on revolution 7 with the correction included. The point where the spacecraft Z-axis is approximately parallel with the magnetic field correlates nicely with an observed dip in charged particle intensity as observed by the MSC-2 spectrometer. Since the flux incident on the spectrometer is at a minimum whenever the Z-axis of the spacecraft is alined with the magnetic field, this dip would be expected if, in fact, the corrected data were true.

Dose Calculations

In order to determine what intensities and spectra were encountered throughout the entire mission, the data in figure 37-6 were replotted in B and L coordinates. This plot, together with figures 37-7 and 37-8, was then used in the MSC-developed computer code to calculate what approximate dose should have been received by the crew for the entire mission. It should be noted that the B, L plots are based on one revolution only and, thus, provide only preliminary data with corresponding uncertainties in the dose estimates. The spectral data used are good to within about a factor of 2.

Data From Gemini VII

Very few data from the Gemini VII mission have been reduced so that little can be discussed at this time about the results. Quick-look, strip-chart data indicate the spectrometer was operating as expected insofar as the electron measurement is concerned. Proton data, however, appear to be somewhat erratic and are suspected, but a detailed analysis of more data is needed to determine if a true difficulty developed during the launch or orbit phase of the mission.

Several days prior to the Gemini VII launch, the magnetometer Z-axis detector was observed to have failed. Replacement of the sensor would have caused a slippage in the launch date, and it was decided that, based on the apparent reliability of the McIlwain total intensity values (as determined on Gemini IV), the needed directional data could be obtained using only two axes and the calculated total B values. Preliminary strip-chart data from the flight show the X- and Y-axes performed as expected.

Conclusions

The significant variation of the spectral shape of charged particles, particularly electrons, in manned spacecraft orbits points out the need for simultaneous inside/outside measurements during actual missions if significant correlations of measured and calculated dose are to be obtained.

The spectra measured indicate that a significant number of electrons are penetrating into the cabin, based on knowledge of the Gemini spacecraft shielding effectiveness. Although the dosimeters reflect very little accumulated dose due to electrons, it is difficult to determine how the gross difference in calculated and measured dose can be due entirely to inadequacies in the shielding calculations. A preliminary study of a spacecraft hatch has been made to determine its transparency to incident electrons. By placing the hatch in an electron beam, it was shown that its ability to shield electrons is less than what the shielding program predicts. Assuming that the rest of the spacecraft totally shields electron flux from the cabin, this investigation shows that sufficient electron penetration would occur through the spacecraft hatch area alone to produce a measurable electron dose in the crew compartment. It is possible that the design of the dosimeter packages is such that they are relatively insensitive to the expected electron dose levels. This is presently being investigated.

The possibility of error in either or both the calculational technique and the dosimeter system suggests that a sensitive electron spectrometer inside the spacecraft cabin would provide very valuable data. An effort is presently underway at MSC to modify the bremsstrahlung spectrometer experiment equipment (MSC–7), which is now scheduled for a later Gemini mission, to detect both electron flux as well as secondary X-rays. This technique and the associated results will be discussed in the experiment symposia following the flights in which the equipment is installed.

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38. EXPERIMENT D4/D7, CELESTIAL RADIOMETRY AND SPACE-OBJECT RADIOMETRY

By BURDEN BRENTNALL, *Air Force Systems Command Field Office, NASA Manned Spacecraft Center*

Summary

The study of the spectral irradiance of natural phenomena and manmade objects has been of increasing interest in recent years both to the scientific community and to the Department of Defense. The purpose of the Air Force D4/D7 experiment has been to obtain accurate measurements from space of emitted and reflected radiance from a comprehensive collection of subjects. The determination of threshold sensitivity values in absolute numbers, and the separation and correlation of specific targets with various backgrounds have been prime objectives.

This report is intended to provide a description of the equipment used on Gemini V and VII and its operations, and a discussion of the measurements made. Results will be discussed generally on a quantitative basis.

Experiment Description

Two interferometer spectrometers and a multichannel spectroradiometer were used as the sensing instruments in this experiment. The selection of the instruments and the particular detectors in the instruments was based upon the spectral bands to be investigated in each flight (fig. 38–1) and the nature of the intended measurements. The instrument characteristics (field of view and resolution, for example) were a compromise among optimization for a particular type of measurement, a need for a broad selection of spectral information, and the performance and other influencing characteristics of the spacecraft.

Since the D4/D7 experiment equipment is contained in several units, it will be reviewed first by component and then integrally as an experimental system aboard the Gemini space-

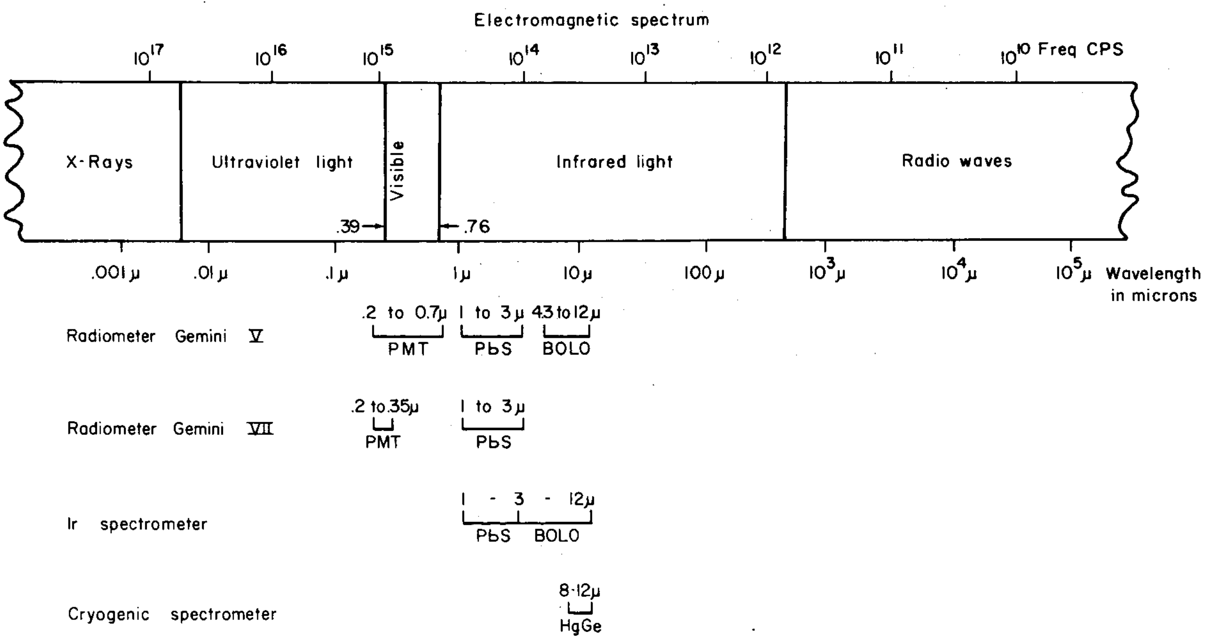


FIGURE 38–1.—Spectral bands to be investigated.

craft. After the system has been defined, operational aspects will be discussed.

D4/D7 Flight Equipment

Radiometer

One of the three measuring instruments used in this experiment was a multichannel, direct-current spectroradiometer. In this radiometer (fig. 38–2), the impinging energy is focused by the collecting optics, mechanically chopped and filtered to obtain specific bands of interest, and then received by the three detectors. The detector signals are then amplified and demodulated. The resultant signals are a function of energy intensity in a given spectral band.

The D4/D7 radiometer (fig. 38–3) was made by Block Engineering Associates, Cambridge, Mass. The radiometer instrument parameters for each flight are presented in table 38–I.

As a result of reviewing the Gemini V flight data, a decision was made to modify the Gemini VII radiometer to incorporate a more sensitive ultraviolet (UV) photomultiplier tube. An ASCOP 541F–05M tube was installed in place of the IP 28 flown on Gemini V, and the bolometer detector was eliminated to make room for the larger photomultiplier tube.

Thirteen signals were provided from the radiometer on Gemini V; 11 were provided on Gemini VII. The signals included detector temperatures, gain, filter wheel position, and analog signal output from the detectors.

Interferometer Spectrometer

The second sensing instrument was a dual-channel interferometer spectrometer (fig. 38–4). The interferometer section was patterned after the Michelson interferometer (fig. 38–5).

The beam splitter splits the optical path, sending part of the beam to the movable mirror M_1 and the other part to a fixed mirror M_2. As a result of the optical path changeability, the waves returning from the mirrors may be in phase (additive) or may be out of phase to some degree and have a canceling effect. The total effect is to produce cyclic reinforcement or interference with the wave amplitude at the detector at any given frequency. The frequency at the detector of this alternate cancellation and reinforcement is a function of the particular spectral energy wavelength λ, the optical retardation B of the mirror, and the time required to move the mirror (scan time) T.

Thus,

$$F_\lambda = \frac{B}{\lambda T}$$

The detector puts out an alternating-current signal which is the sum of the alternating-

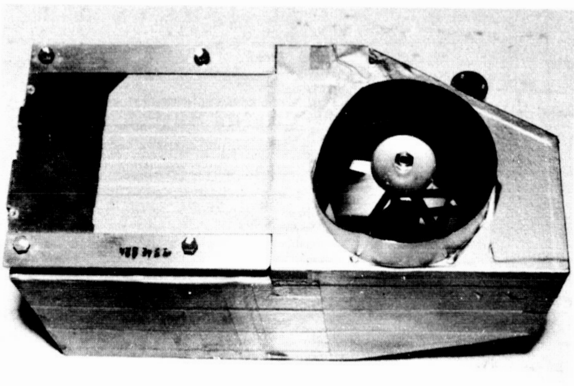


FIGURE 38–3.—Trichannel spectroradiometer.

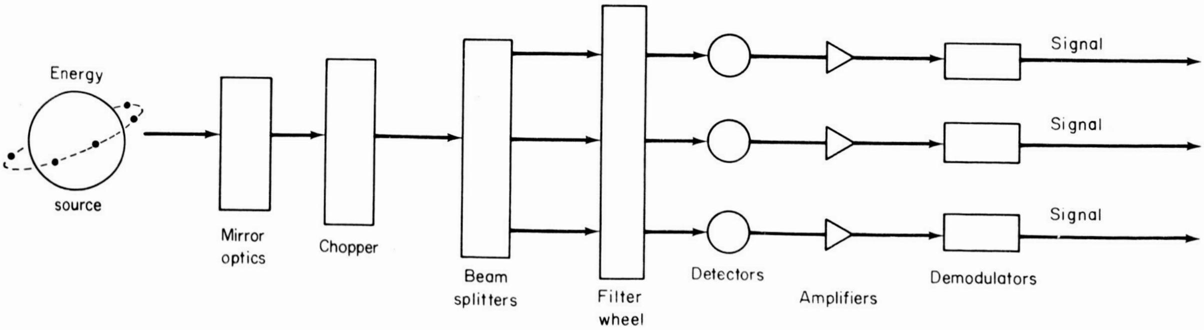


FIGURE 38–2.—Radiometer functional diagram.

TABLE 38-I.—*Radiometer Instrument Parameters*

Weight	17.5 lb		
Power input	14 watts		
Field of view	2°		
Optics	4 in. Cassegrain		
Detectors, Gemini V	Photomultiplier tube (IP 28)	Lead sulfide	Bolometer
Spectral band, μ	0.2–0.6	1.0–3.0	4–15
Nominal filter width, μ	0.03	0.1	0.3
Filters used, μ	0.22	1.053	4.30
	.24	1.242	4.45
	.26	1.380	6.00
	.28	1.555	8.0
	.30	1.870	9.6
	.35	2.200	15.0
	.40	2.820	
	.50		
	.60		
Dynamic range	10^5 in 4 discrete steps	10^3 log compressed	10^3 log compressed
Detectors, Gemini VII	Photomultiplier tube (ASCOP 541 F–05M)	Lead sulfide	
Spectral band, μ	0.2–0.35	1.0–3.0	
Nominal filter width, μ	0.03	0.1	
Filters used, μ	0.2200	1.053	
	.2400	1.242	
	.2500	1.380	
	2600	1.555	
	.2800	1.870	
	.2811	1.9000	
	.2862	2.200	
	.3000	2.725	
	.3060	2.775	
		2.825	
Dynamic range	10^5 in 4 discrete steps	10^3 log compressed	

current signals corresponding to all the wavelengths from the source. The amplitudes of the signals will vary directly with the source brightness at each wavelength. The output of the interferometer is then a complex waveform called an interferogram which is the Fourier transform of the incident radiation frequencies (fig. 38–6(a)). This transform is reduced to a plot of wavelength versus intensity by taking the inverse transform of the interferogram (fig. 38–6(b)). An interferogram made with the D4/D7 instrument is shown in figure 38–6(c) and an actual measurement on the California coast during Gemini V is shown in figure 38–6(d).

The D4/D7 interferometer spectrometer discussed here (and referred to nontechnically as the "uncooled" or "IR" spectrometer) contained a lead sulfide detector and a bolometer detector, thus providing correlative information to that of the spectroradiometer. This, too, was a Block Engineering instrument. Its parameters are listed in table 38–II. Data output from the instrument included the signals

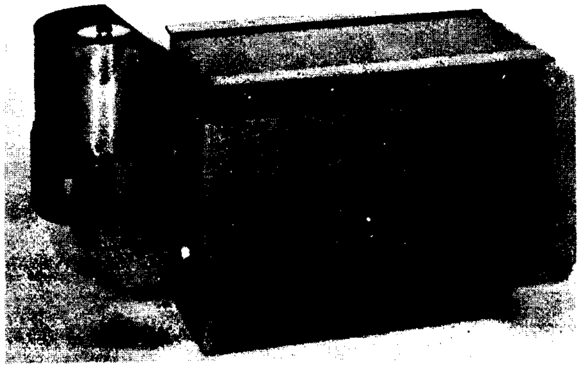


FIGURE 38-4.—Dual-channel interferometer spectrometer.

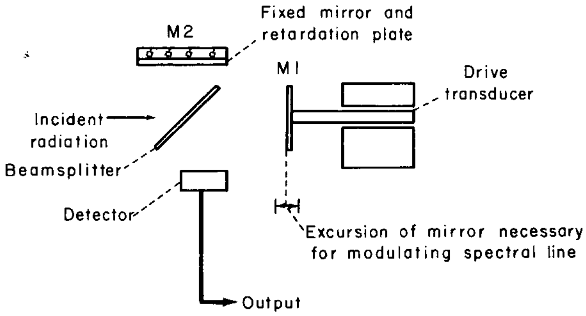
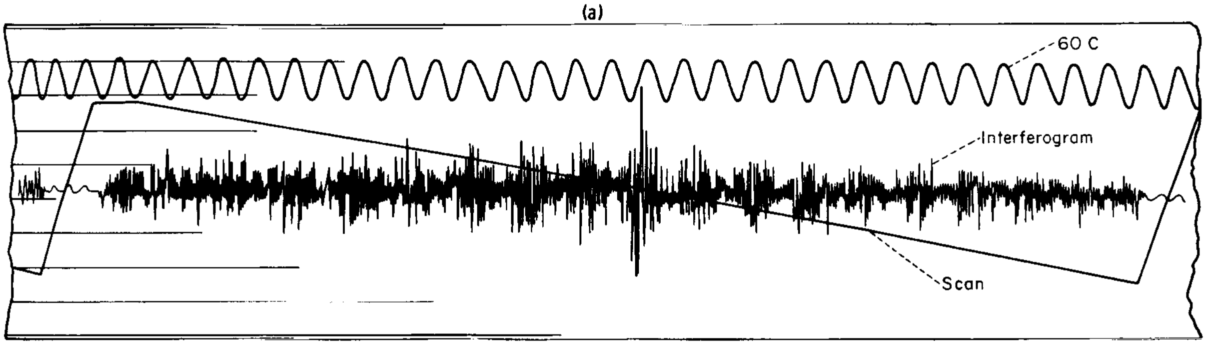


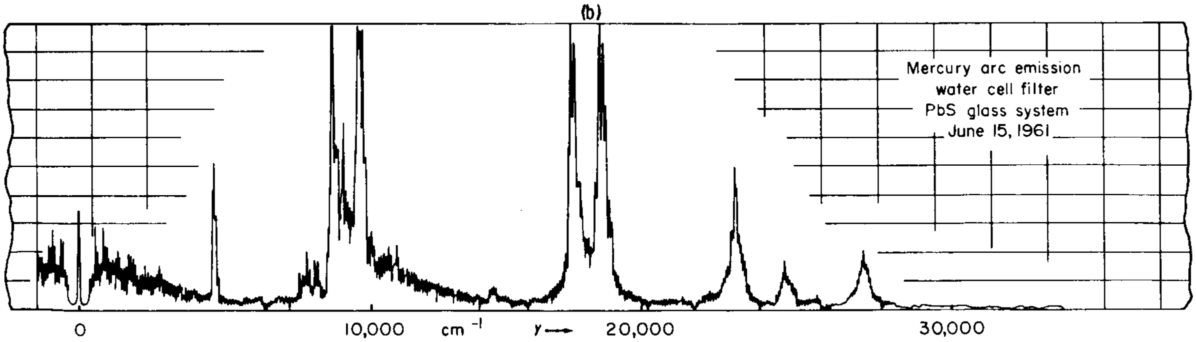
FIGURE 38-5.—Schematic of Michelson interferometer.

(a)



(*a*) Representation of an interferogram.
FIGURE 38-6.—Interferometer measurements.

(b)



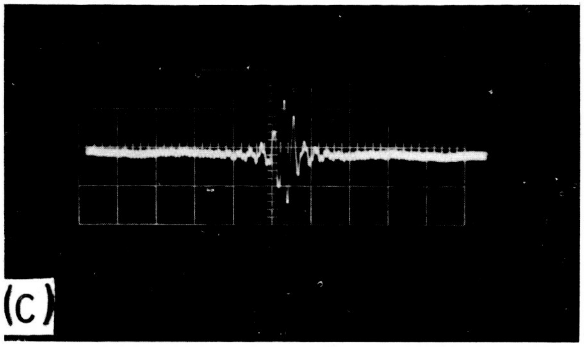
(*b*) Representation of an interferogram reduced to a spectrum.
FIGURE 38-6.—Continued.

from the two detectors, gain settings, detector temperatures, and automatic calibration source data. Lead-sulfide signal data were handled on a data channel-sharing basis with the detector output from the cryogenic spectrometer.

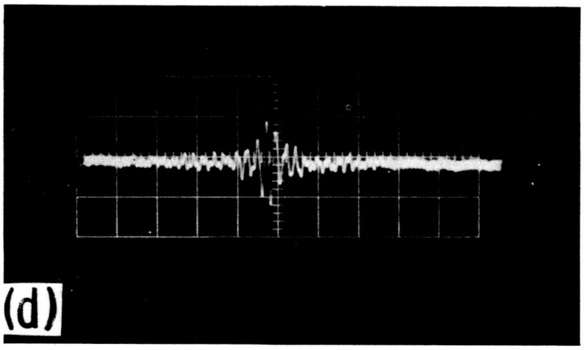
Cryogenic Interferometer Spectrometer

The cryogenic interferometer spectrometer is similar in operation to the IR spectrometer, although dissimilar in appearance (fig. 38-7). The principal difference is that the highly sensitive detector must be cryogenically cooled to make measurements in the region of interest (8 to 12 microns). The cooling is accomplished by immersing a well containing the detector, optics, and some of the electronics in liquid neon.

The cryogenic subsystem was made for Block Engineering by AiResearch Division of Garrett Corp. It was an open-cycle, subcritical, cryogenic cooling system which maintained the instrument well at a temperature of −397° F for a period of approximately 14 hours. Figure 38–8 shows an X-ray view of the cryogenic tank and instrument well. The parameters for the instrument are listed in table 38–III.



(*c*) Spectrometer interferogram, 2100° C calibration source.
FIGURE 38–6.—Continued.



(*d*) IR spectrometer interferogram during the Gemini V flight (California coastal land).
FIGURE 38–6.—Concluded.

TABLE 38–II.—*Parameters of the IR Spectrometer*

Weight	18.5 lb
Power input	8 watts
Field of view	2°
Optics	4 in. Cassegrain

Detectors	Lead sulfide	Bolometer
Spectral band, μ	1–3	3–15
Dynamic range	10^3 automatic gain changing	10^3 automatic gain changing

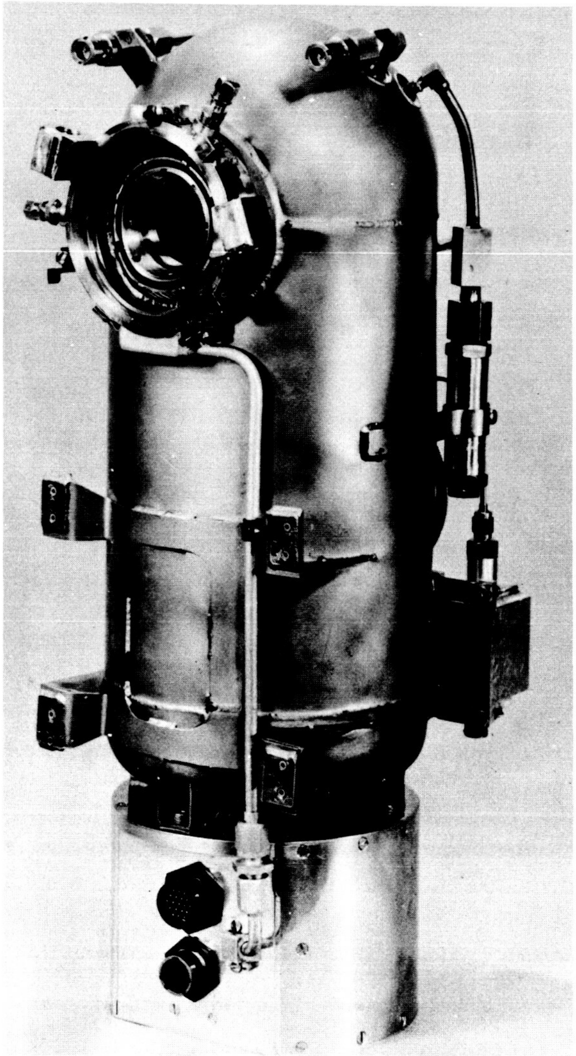


FIGURE 38–7.—Cryogenic interferometer spectrometer.

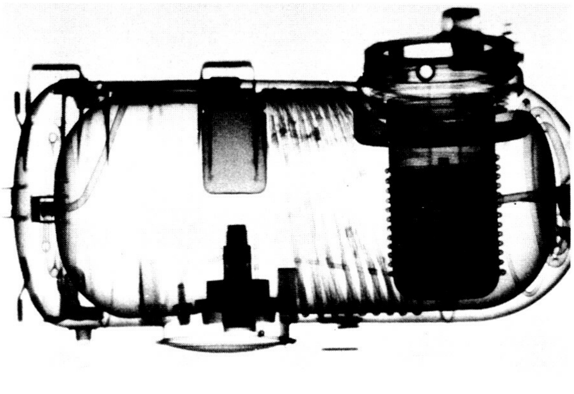


FIGURE 38–8.—X-ray view of cryogenic interferometer spectrometer.

TABLE 38–III.—*Parameters of the Cryogenic Interferometer Spectrometer*

Weight (with neon)	33.5 lb.
Power input	6 watts
Field of view	2°
Optics	4 in. Cassegrain
Detector	Mercury-doped germanium
Spectral band	8 to 12 microns
Dynamic range	10^3 automatic gain changing
Coolant	Liquid neon

Electronics Unit

The electronics unit used in conjunction with the three sensing devices contained the various circuits necessary for the experiment. The circuitry includes an electronic commutator, filter motor logic, variable control oscillators, mixer amplifier, clock pulse generator, and other secondary electronic circuitry.

Recorder Transport and Electronics

The D4/D7 experiment tape recorder was separated into two modules: the tape transport and the recorder electronics. This was done so that the recorder would fit into the available space on the Gemini reentry vehicle. The recorder provided 56 minutes of tape for three channels of data. It was not capable of dump, and data were stored and retrieved with the spacecraft.

Frequency-Modulation Transmitter and Antenna

In parallel with the recorder, the D4/D7 transmitter provided three channels of real-time frequency-modulated (FM) data to selected ground stations located around the earth. The transmitter, operating through an antenna extended from the pilot's side of the spacecraft, transmitted 2 watts on an assigned ultrahigh frequency.

Control Panel

The majority of the switches associated with the experiment were located on the pilot's main console (fig. 38–9). Additional functions were provided by a meter and some sequencing switches.

D4/D7 Experiment System

The experiment system consisting of the foregoing components was mounted in Gemini V

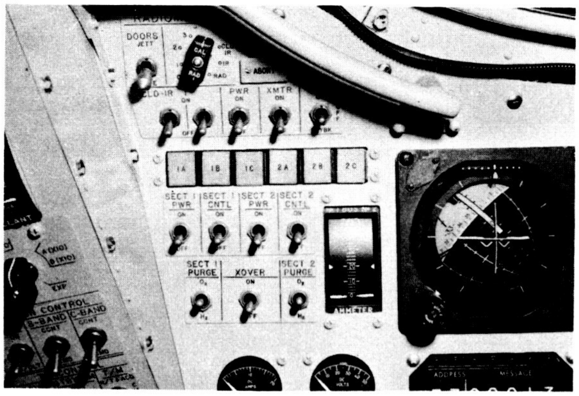


FIGURE 38–9.—Instrument panel for Experiment D4/D7.

and VII as shown in figure 38–10. The radiometer and spectrometers were mounted in the Gemini retroadapter section on swingout arms. After the spacecraft was in orbit, doors in the adapter were pyrotechnically opened, and the three sensing units swung through the openings into boresight alinement with the spacecraft optical sight. After the sensing units had been erected, the spacecraft was pointed at the desired area for measurement. Figure 38–11 shows the Gemini VII with the instruments extended. Gemini V was similar in appearance.

The data from the radiometer were telemetered through the spacecraft pulse code modulation (PCM) system. The data from the spectrometers were telemetered through the transmitter or routed to the recorder, or both were accomplished, if desired.

D4/D7 Mission Plan

The desired objectives for the D4/D7 measurements included the following:

	Microns
Earth backgrounds	0.2 to 12
Sky backgrounds	0.2 to 12
Sky-to-horizon spectral calibrations	8 to 12
Rocket exhaust plumes	0.2 to 3
Natural space phenomena (stars, moon, sun)	0.2 to 12
Manmade objects in space	0.2 to 12
Weather phenomena (clouds, storms, lightning)	0.2 to 10
Equatorial nadir-to-horizon spectral calibrations	8 to 10

Since the lifetime of the cryogenic neon in the cooled spectrometer was limited to 14 hours, 5

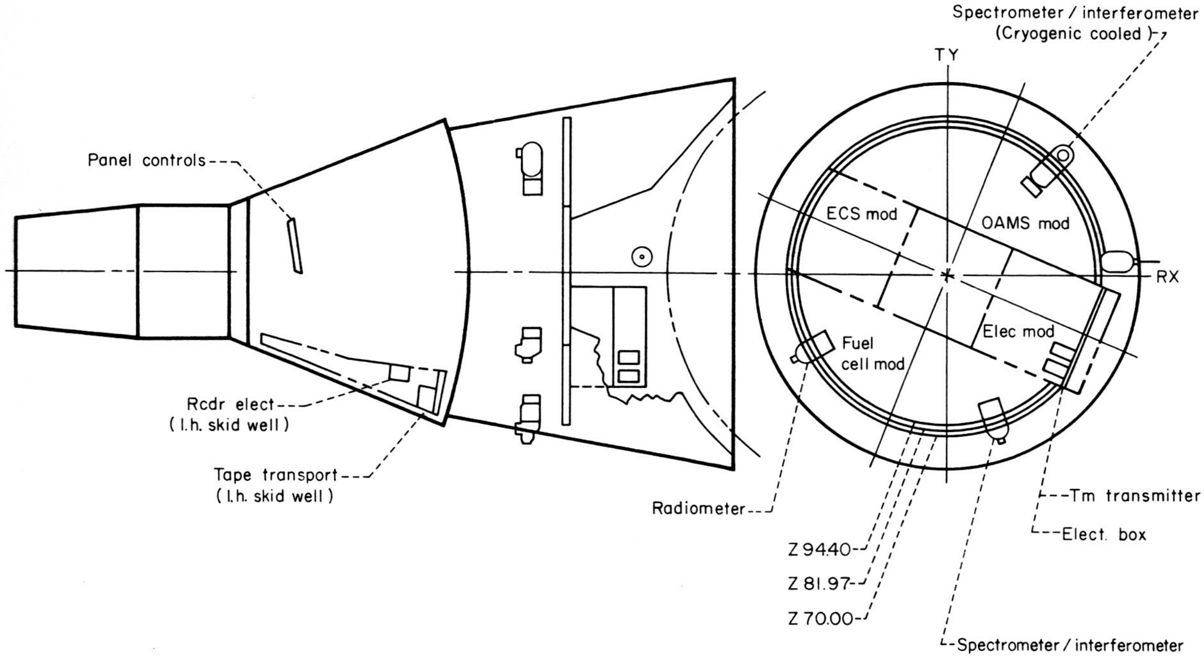


FIGURE 38-10.—Location of Experiment D4/D7 equipment in spacecraft.

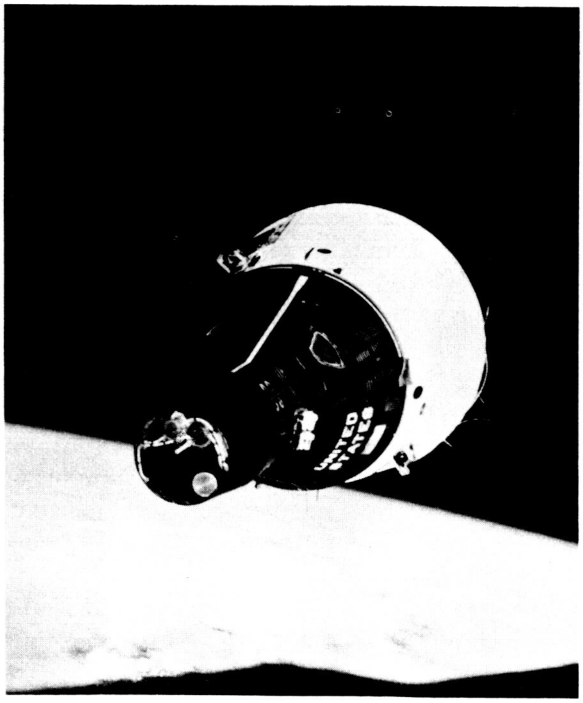


FIGURE 38-11.—Cryogenic spectrometer and radiometer erected on Gemini VII spacecraft.

of which would be spent on the launch pad, the measurements requiring the use of the cooled spectrometer were planned for the first few revolutions. The rocket-plume measurements were planned for those revolutions which brought the spacecraft closest to the firing site, yet as early or late in the day as feasible to minimize background radiation. The sun measurement was planned to be the final measurement, since calibration of the detectors might be affected. The remainder of the measurements, requiring real-time updating, were interspersed throughout the flight.

Results From Gemini V

Approximately 3 hours 10 minutes of D4/D7 data were gathered during the Gemini V flight. Twenty-one separate measurements were made, covering 30 designated subjects. The PCM and FM transmitted data amounted to 125 000 feet of magnetic tape.

Processing the data requires a great amount of time. The interferometer data must be run through a wave analyzer or a high-speed computer. The wave analyzer integrates 35 interferograms and gives the results in the form of Fourier coefficients in approximately 30 minutes. The computer takes about 2 hours to perform the transform on one interferogram. Over 10 000 interferograms were made during the Gemini V flight.

The PCM data are reduced in terms of filter settings and gain; then, calibration coefficients are applied. Both PCM and FM data are correlated with crewman comments and photography, where applicable.

From the foregoing, the magnitude of the data-reduction task can be seen. The data from D4/D7 on Gemini V are still in the process of reduction and, at the present time, are not available in sufficient amounts to be discussed qualitatively to any significant extent. All the PCM data from the radiometer have been reduced and are presently being correlated with the spectrometer data as they become available. The process of reducing the interferograms is presently 35 percent complete. The following is a list of the D4/D7 measurements made during the Gemini V flight:

Revolution	Location	Measurement
1	Carnarvon, Australia.	Operational readiness check of cryogenic spectrometer
2	Africa-Australia	Rendezvous evaluation pod (REP) measurements during darkness
14	Australia	Night water and night land measurements
16	Africa	Mountains and land with vegetation
16	Malagasy	Night water and night land measurements
16	Australia	Star measurement, Vega
16	Australia	Equipment alinement check
17	Australia	Moon irradiance measurement
31	Africa	Cloud blanket sweep, nadir-to-horizon
31/32	Florida	Land with vegetation
45	Australia	Night void-sky measurement
47	Australia	Zodiacal light
47	Australia	Star measurement, Deneb
47	California	Minuteman missile launch
51	Hawaii	Island measurement
61	New Mexico	Rocket sled firing
62	California	Minuteman missile launch
74	Africa	Water, land, mountains, desert
88	Africa	Desert
89	Africa	Mountains
103	Australia	Horizon-to-nadir scan

The equipment was erected and operationally verified over Carnarvon, Australia, during the first revolution.

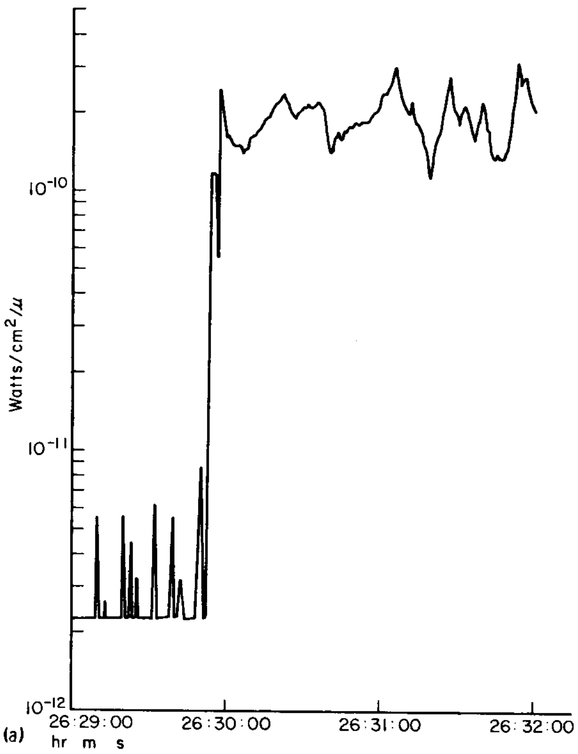
During the second revolution, the REP was ejected and measurements were made of its separation from the spacecraft during the spacecraft darkness period. The primary instrument for this measurement was the cryogenic spectrometer. The cover on the spectrometer was jettisoned when the REP was approximately 2500 feet away from Gemini V, and measurements were made during the remainder of the darkness period.

After 15 minutes of operation, the filter wheel on the radiometer ceased working and remained on filter settings of 4000 angstroms (Å), 2.2 microns, and 4.3 microns for the remainder of the flight. Since the interferometers still functioned satisfactorily, the restriction in radiometer data was not of major concern. The main loss of data was in the UV region—not covered by the spectrometers—where only the 4000 Å information was available. In playing the onboard D4/D7 recorder after its retrieval, it was discovered that no REP measurement data were recorded on the tape. This limited the information from the cryogenic spectrometer to the FM data received during the pass over Carnarvon. Review of the interferograms made at Carnarvon indicates that the signal was well above the noise level. Reduction is in process, and attempts are being made to separate the background signal and spacecraft radiance from the signal of the REP. This task is made more difficult by the lack of data from the onboard recorder.

Due to the date of the launch of Gemini V, the moon measurements had to be made on a partially illuminated moon. The radiometer data from this measurement can be seen in figures 38–12(a) and 38–12(b).

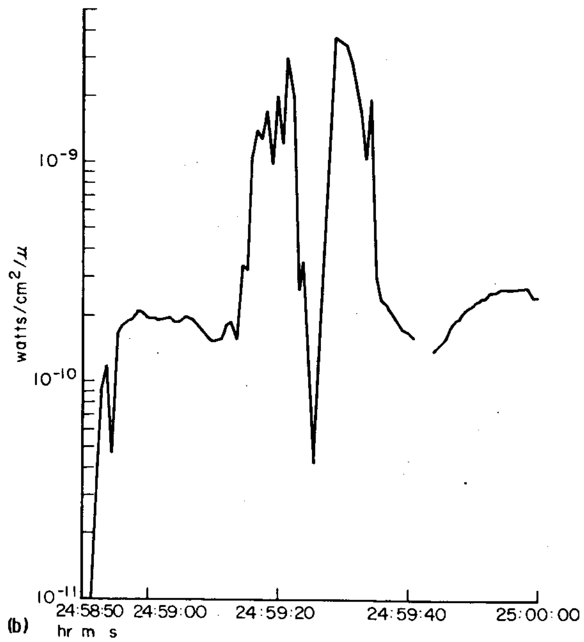
Quick-look information on the 4000 Å radiometer data on Vega and Deneb is excellent. The values on that spectrum band were slightly higher than those theoretically predicted. For example, the value for Vega was 1.2×10^{-11} watts per square centimeter per micron at 4000 Å.

An example of the IR spectrometer data can be seen in figure 38–13. This shows the return at 1.88 microns on the California land background.



(*a*) Moon measurements made during revolution 17, Gemini V mission.

FIGURE 38–12.—**Radiometer data from moon measurements (4000 Å).**



(*b*) **Moon measurements made during alinement check, revolution 16 of Gemini V mission.**

FIGURE 38–12.—Concluded.

Results From Gemini VII

The D4/D7 results from Gemini V did have some effect on the experiment on Gemini VII. Since there were only 4 months between the two flights, there was little time for data evaluation inputs to use for design modification. One modification, as previously noted, was made to the radiometer. Another modification, a switch guard on the recorder switch, was added to the instrument panel. Otherwise the experiment system was identical for both spacecraft.

The planned measurements to be made by Gemini VII were affected by the data gathered from Gemini V. Certain measurements were repeated where information in addition to that provided by Gemini V was desired. New measurements were added, based on the demonstrated ability shown by the crew and equipment on Gemini V.

Data gathered on the Gemini VII flight totaled 3 hours 11 minutes, which was almost the same as the amount gathered on Gemini V. There were 36 separate D4/D7 measurements made of 42 designated subjects.

The following is a list of the measurements made during the Gemini VII flight:

Revolution	Location	Measurement
1	Africa-Malagasy	Launch vehicle measurement and cooled spectrometer alinement check
1	Malagasy	Launch vehicle background measurement
1	Malagasy-Australia.	Launch vehicle measurement
2	Ascension	Void space measurement
2	Ascension	Star measurement—Rigel with cryogenic spectrometer
2	Ascension	Launch vehicle measurement
2	South Atlantic	Star measurement—Sirius with cryogenic spectrometer
2	Malagasy	Night sky-earth horizon calibration sweep with cooled spectrometer
6	Malagasy	Cryogenic lifetime check

Revolution	Location	Measurement
6	Hawaii	Cryogenic lifetime check
7	Hawaii	Cryogenic lifetime check
8	Ascension	Cryogenic lifetime check
15	Malagasy	Radiometer and IR spectrometer alinement check on nearly full moon
30	Malagasy	Star measurements—Betelgeuse and Rigel without cryogenic instrument
31	Florida	Polaris launch
32	Ascension	Milky Way
32	North America	Earth background—coastal, mountains, desert, land with vegetation
45	North America	Earth background—water, mountains, plains, coastal regions correlated with IR color-film photographs
49	Malagasy	Night airglow
49	Malagasy	Large fire on earth at night
59	Malagasy	Full moon measurement
59	Australia	Night land, water, cloud reflectance with full moon
59	Australia	Lightning at night
74	Africa	Cloud blanket sweep with camera correlation
75	Africa	Lightning at night
76	Ascension	Horizon-to-nadir scan
88	Africa	Desert
89	Malagasy	Celestial measurement—Venus
104	Australia	Night land and water
117/118	Florida	Gemini VI–A abort
148	New Mexico	Rocket sled firing
149	Pacific	Night measurement of Minuteman reentry
161/162	Florida	Gemini VI–A climb to orbit
166	Hawaii	Gemini VI–A station keeping
169	South America	Gemini VI–A separation burn
193	Texas	Sun measurement

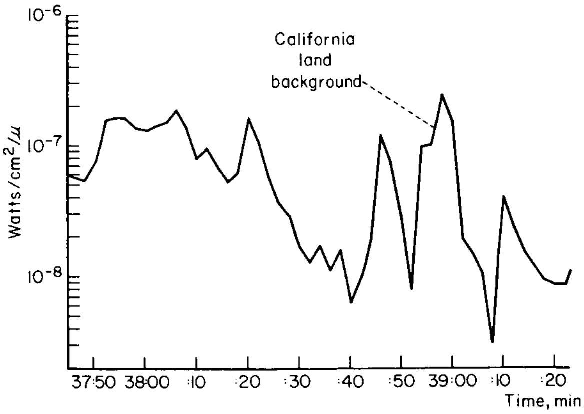


FIGURE 38–13.—Interferometer spectrometer data (1.88μ).

Nineteen minutes after Gemini VII lift-off the D4/D7 sensors were erected, and the equipment turned on. An 8-feet-per-second separation burn was made away from the launch vehicle at sunset, and measurements on the launch vehicle were begun. Cryogenic spectrometer measurements were made for the remainder of the night cycle as the spacecraft separated from the launch vehicle. Periodically during this period, launch vehicle background measurements were made, and, at one point, the launch vehicle was measured against a moon background.

During the second revolution, measurements were performed with the cryogenic spectrometer on void space, on the launch vehicle, and on the stars Rigel and Sirius. At the conclusion of the measurement on Sirius a slow pitch-down maneuver was made to the horizon. The purpose of this measurement was to do a night sky-to-horizon calibration sweep in the 8- to 12-micron region. The radiometer gave UV correlation data during this measurement.

Alinement of the radiometer and IR spectrometer was performed December 5, 1965, on a nearly full moon. Photographic coverage of the measurement objective was simultaneously obtained by a camera boresighted along the instrument axis (fig. 38–14). The equipment alinement was checked by the use of a meter in the center console. The crewmen boresighted the spacecraft on the moon and then made minor excursions in pitch and yaw to locate the aiming point for optimum signal return (fig. 38–15). This accounts for the dips in the curves seen on

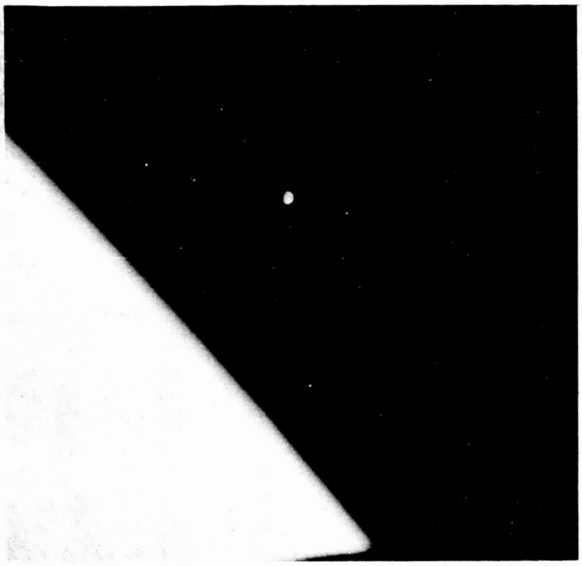


FIGURE 38–14.—Photograph of nearly full moon taken during alinement of radiometer and infrared spectrometer.

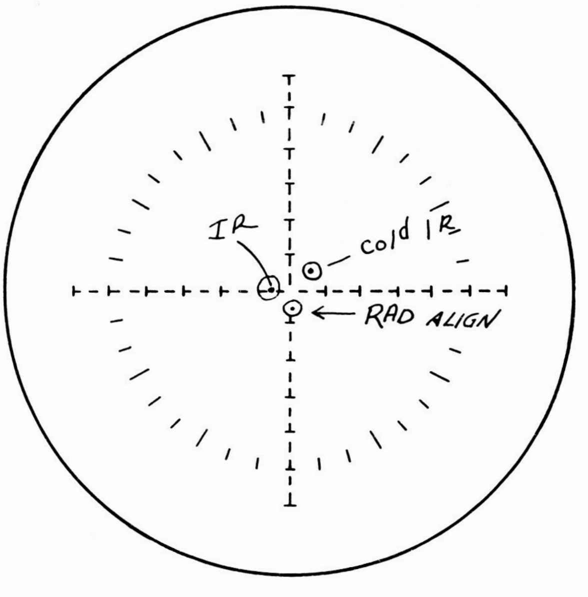


FIGURE 38–15.—Alinement pattern (as noted in flight logbook).

figures 38–16 and 38–17. The values of moon irradiance from 2000Å to 3060Å and 1 to 3 microns as measured by the radiometer on December 5 are shown in figures 38–18 and 38–19. The data show good correlation with the other instruments and with the measurements made at the full moon on December 8. As an illustration, a plot of the lead sulfide channel readings taken December 5 on the radiometer is compared with the lead sulfide channel readings on the IR spectrometer made on December 8 (fig. 38–20). The values taken on December 8 are slightly higher than those taken on December 5, as would be expected. Figure 38–21 shows the flight measurements from Gemini V on a predicted 25-day moon curve and those for Gemini VII against a full moon curve.

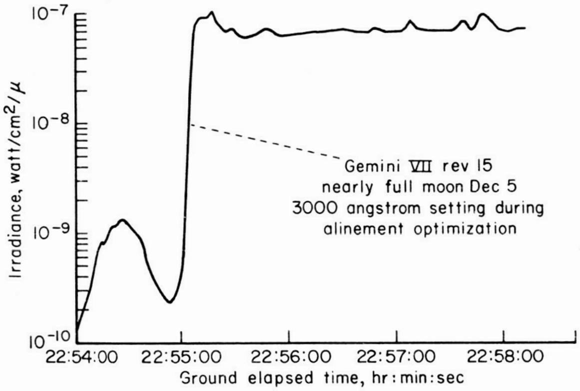


FIGURE 38–16.—Moon irradiance during alinement optimization (3000 angstrom setting).

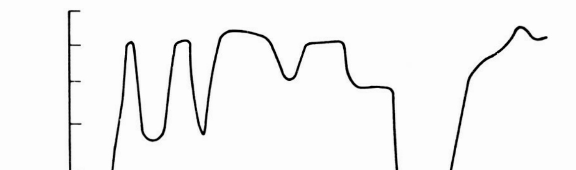


FIGURE 37–17.—Moon irradiance during alinement optimization (1.555 micron setting).

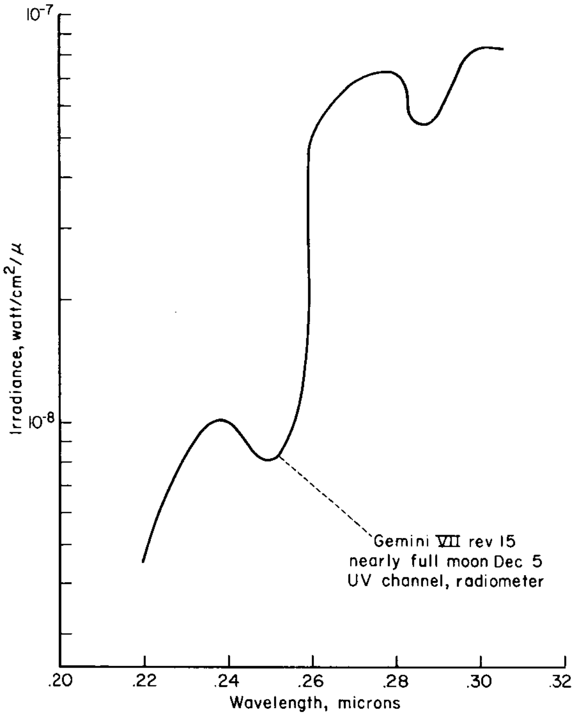


FIGURE 38-18.—Values of moon irradiance from 2000 to 3060 angstroms.

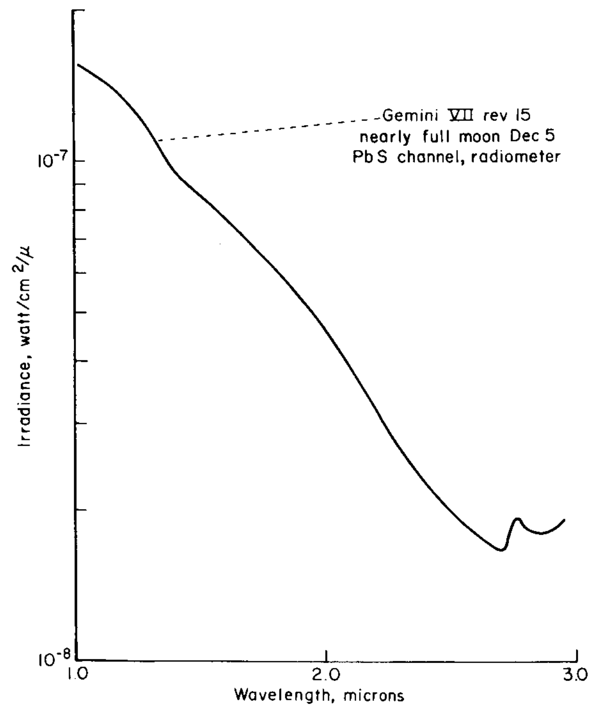


FIGURE 38-19.—Values of moon irradiance from 1 to 3 microns.

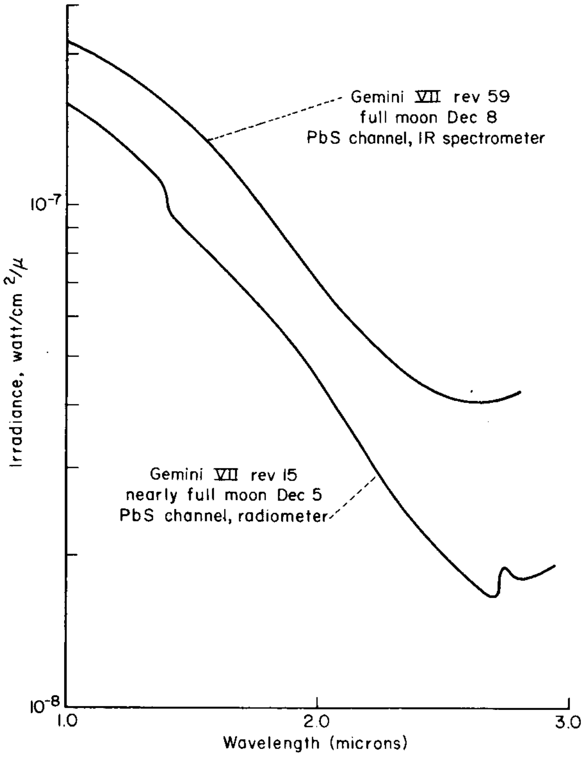


FIGURE 38-20.—Comparison of PbS channel readings on December 5 and December 8, 1965.

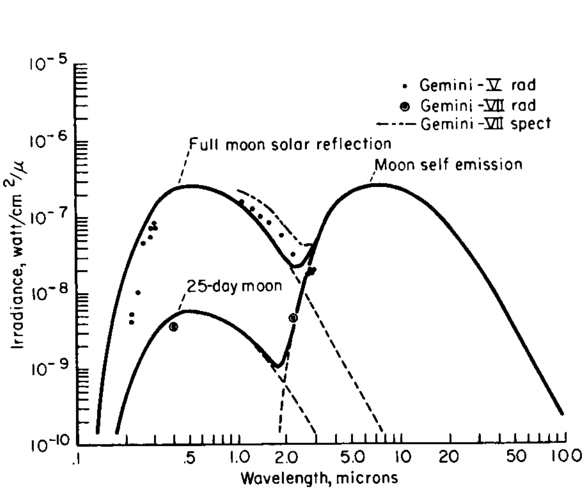


FIGURE 38-21.—Experiment D4/D7 lunar irradiance measurements during Gemini V and VII missions.

Throughout the measurements, a high degree of photograph and voice correlation was maintained. Figure 38-22 is a picture of a cloud bank measured during the cloud blanket sweep over Africa. Figure 38-23 is a photograph, made with IR film, of the Gulf coast during a D4/D7 land/water measurement. Photographic coverage was also accomplished during the Polaris launch, airglow measurement, Gemini VI–A retrograde maneuver, rocket sled run, and horizon-to-nadir calibration.

During the flight all of the sensing equipment functioned perfectly. The experiment recorder operated intermittently during the first two revolutions and operated satisfactorily thereafter. The recorder difficulty caused no serious loss of data, however, since vital parts of the measurements were scheduled over experiment ground receiving stations. The transmitter worked well throughout the flight.

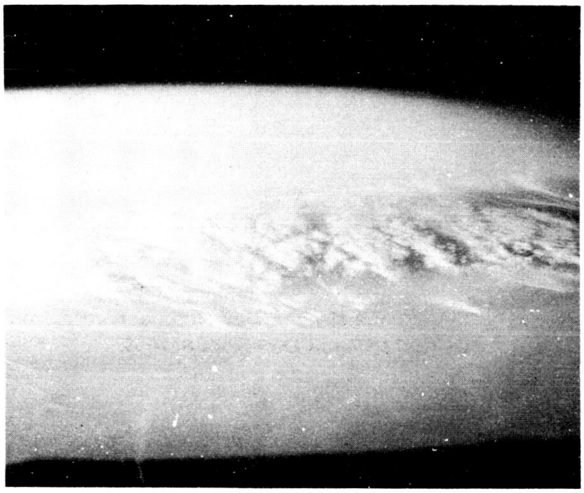


FIGURE 38–22.—Cloud formation photographed during infrared cloud blanket sweep.

FIGURE 38–23.—Photograph of Gulf Coast taken during Experiment D4/D7 background measurements.

Crewman performance during the flight was outstanding. In addition to performing all scheduled measurements, several targets of opportunity (for example, a ground fire and lightning) were measured on the crewman's initiative.

In addition to the acquisition of a large amount of significant radiometric data, several adjunct pieces of information were obtained. First, the alinement check after Gemini VII was in orbit showed that ground alinement between the optical sight and D4/D7 equipment in the adapter was valid within 0.5°. Concern had been expressed that alinement under 1-g conditions and shifting at the heat shield interface with the adapter during launch might cause some problems. Second, the cryogenic lifetime for the cooled spectrometer—nominally 14 to 15 hours under quiescent 1-g conditions—was essentially unchanged by subjection to launch environment and then zero-g conditions. The system was a subcritical, open-cycle, liquid-neon system in a fixed-wall Dewar flask. It operated for 8 hours 50 minutes in space after 5 hours of ground operation awaiting lift-off. Globularization of the neon due to weightlessness caused no perturbations in the operating characteristics of the cryogenic system.

Finally, it is to be noted that frost or snow can be seen in pictures of Gemini VII in roughly an oval pattern aft of the cryogenic spectrometer. This frost was still on the spacecraft some 10 days after the cryogen had been depleted, which is interesting in view of the sublimation characteristics of a hard vacuum.

In conclusion, because the data processing is so slow and because there has been so much to correlate, there are few results yet available. The voice annotations, photographic coverage, and debriefing comments are contributing significantly to the meaning and correlation of the data.

Man's contributions in the choice of targets, mode of equipment operation, and ability to track selectively with the spacecraft have been unique in giving the flexibility necessary to accomplish such a diverse group of radiometric measurements.

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B

MEDICAL SCIENCE EXPERIMENTS

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39. EXPERIMENT M-1, CARDIOVASCULAR CONDITIONING

By Lawrence F. Dietlein, M.D., *Assistant Chief for Medical Support, Crew Systems Division, NASA Manned Spacecraft Center; and* William V. Judy, *Crew Systems Division, NASA Manned Spacecraft Center*

Introduction

Ground baseline studies in support of Experiment M-1 indicated that leg cuffs alone, when inflated to 70 to 75 millimeters of mercury for 2 out of every 6 minutes, provided protection against cardiovascular "deconditioning" which was occasioned by 6 hours of water immersion (ref. 1). Four healthy, male subjects were immersed in water to neck level for a 6-hour period on two separate occasions, 2 days apart. Figures 39-1, 39-2, 39-3, and 39-4 indicate that 6 hours of water immersion resulted in cardiovascular "deconditioning," as evidenced by cardioacceleration in excess of that observed during the control tilt and by the occurrence of syncope in two of the four subjects. The tilt responses following the second period of immersion, during which leg cuffs were utilized, revealed that a definite protective effect was achieved. Cardioacceleration was less pronounced, and no syncope occurred.

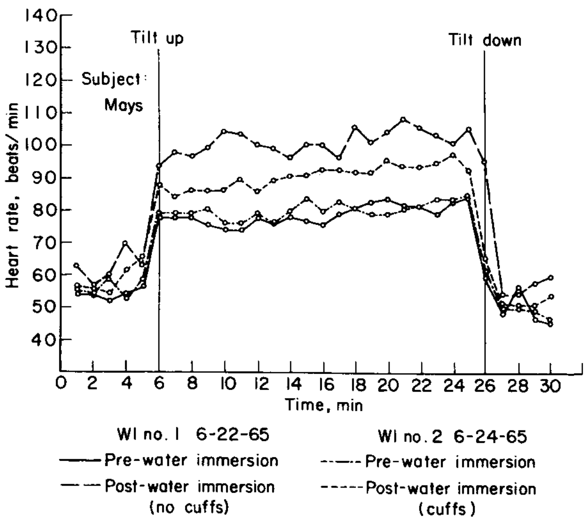

FIGURE 39-1.—Six-hour water immersion studies, first subject.

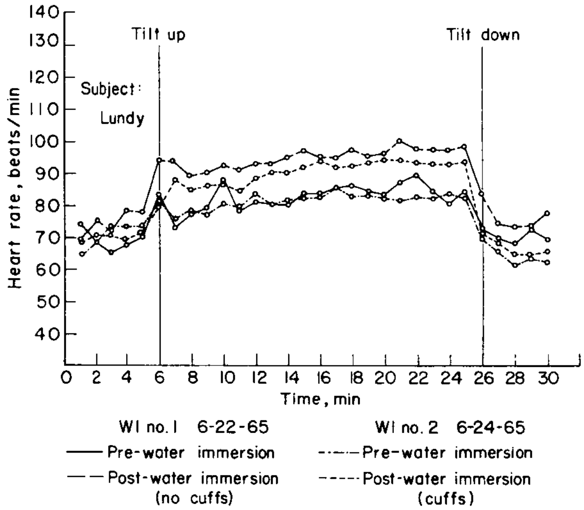

FIGURE 39-2.—Six-hour water immersion studies, second subject.

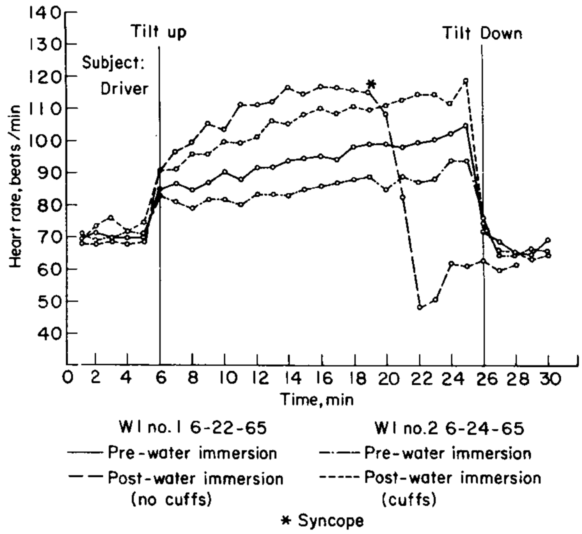

FIGURE 39-3.—Six-hour water immersion studies, third subject.

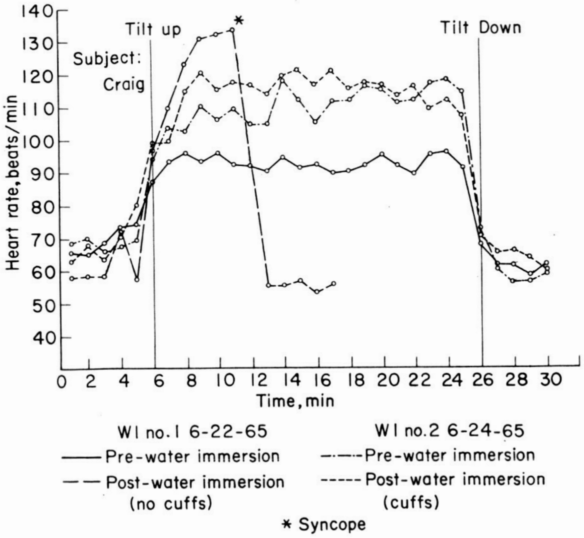

FIGURE 39-4.—Six-hour water immersion studies, fourth subject.

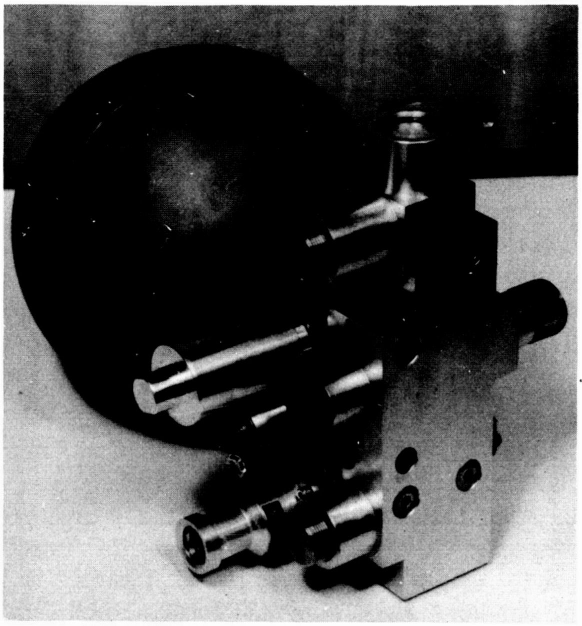

FIGURE 39-5.—Cardiovascular reflex conditioning system.

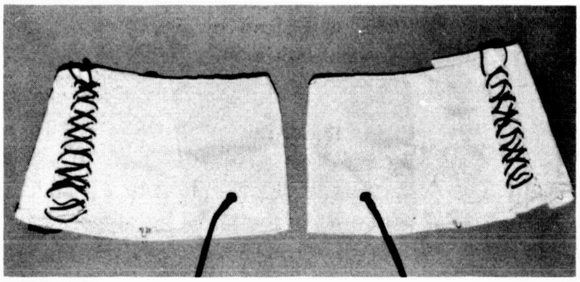

FIGURE 39-6.—Cardiovascular conditioning pneumatic cuffs.

The physiological mechanisms responsible for the observed efficacy of the cuff technique remain obscure. One might postulate that the cuffs prevent thoracic blood volume overload, thus inhibiting the so-called Gauer-Henry reflex with its resultant diuresis and diminished effective circulating blood volume. Alternatively, or perhaps additionally, one might postulate that the cuffs induce an intermittent artificial hydrostatic gradient (by increasing venous pressure distal to the cuffs during inflation) across the walls of the leg veins, mimicking the situation that results from standing erect in a 1-g environment and thereby preventing the deterioration of the normal venomotor reflexes. Theoretically, this action should lessen the pooling of blood in the lower extremities and increase the effective circulating blood volume upon return to a 1-g environment following weightlessness or its simulation. The precise mechanism, or mechanisms, of action must await further study.

Equipment and Methods

The equipment used in Experiment M-1 consisted of a pneumatic timing or cycling system and a pair of venous pressure cuffs (figs. 39-5 and 39-6). The cycling system was entirely pneumatic and alternately inflated and deflated the leg cuffs attached to the pilot's thighs. The system flown on Gemini V (fig. 39-7) consisted of three basic components:

(1) A pressurized storage vessel charged with oxygen to 3500 psig.

(2) A pneumatic control system for monitoring the pressurized storage vessel.

(3) A pneumatic oscillator system for periodically inflating and deflating the leg cuffs.

The equipment flown on Gemini VII was almost identical to that used on Gemini V and was supplied with oxygen pressure from the spacecraft environmental control system. The pneumatic venous pressure cuffs were formfitted to the proximal thigh area of the pilot. The cuffs consisted essentially of a 3- by 6-inch bladder enclosed in a soft nonstretchable fabric. The bladder portion of each cuff was positioned on the dorsomedial aspect of each thigh. The lateral surface of the cuffs consisted of a lace adjuster to insure proper fit.

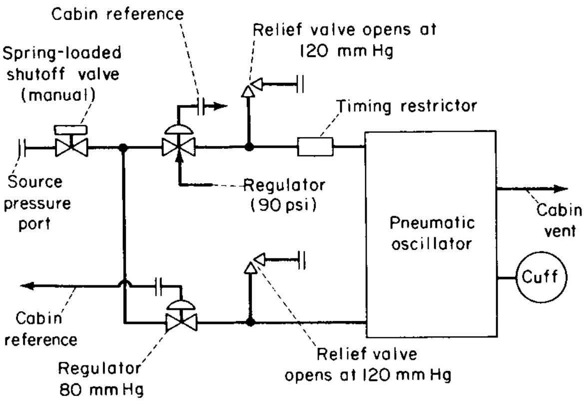

FIGURE 39–7.—Schematic diagram of cardiovascular reflex conditioner.

Results

The Cardiovascular Conditioning Experiment (M–1) was flown on the Gemini V and VII missions. The pilots for these missions served as experimental subjects; the command pilots were control subjects. The experiment was operative for the first 4 days of the 8-day Gemini V mission, and 13.5 days of the Gemini VII mission.

Prior to these missions, each crewmember was given a series of tilt-table tests. These control tilts are summarized in table 39–I, the numerical values indicated being mean values for the three control tilts. The results of six consecutive postflight tilts for the Gemini V command pilot and pilot are summarized in figures 39–8 and 39–9. Figure 39–10 summarizes the heart-rate change during the initial postflight tilt expressed as a percent of the preflight value for all the Gemini flights to date. The results of four consecutive postflight tilts for the Gemini VII command pilot are indicated in figures 39–11 through 39–14, and for the Gemini VII pilot in figures 39–15 through 39–18. Figure 39–19 summarizes the Gemini VII tilt-table data.

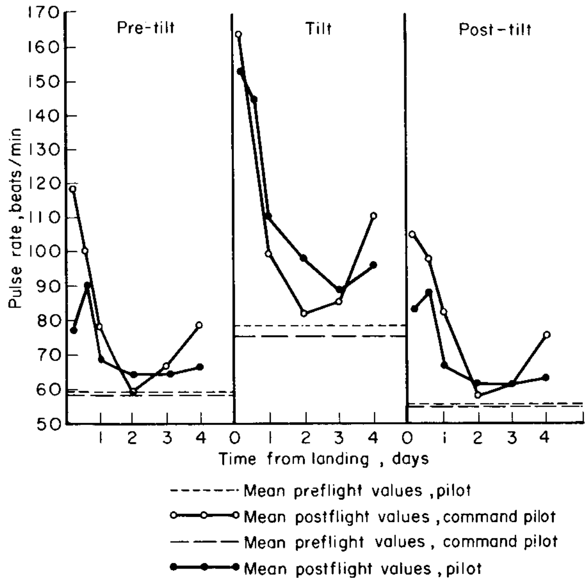

FIGURE 39–8.—Summary of pulse rate during tilt-table studies of Gemini V flight crew.

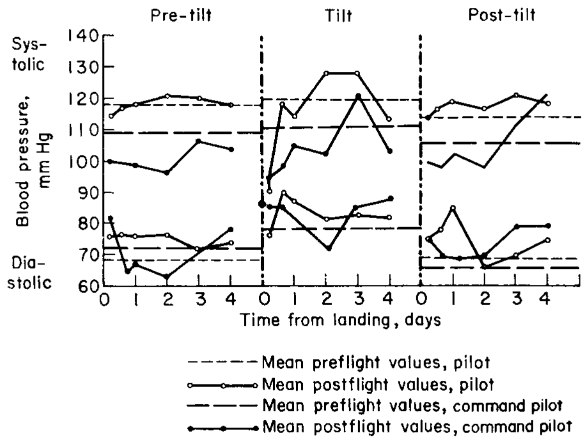

FIGURE 39–9.—Summary of blood pressure during tilt-table studies for Gemini V flight crew.

The crewmembers for both the Gemini V and VII missions exhibited increased resting pulse rates during the first 12 to 24 hours after recovery. Resting pulse rate changes for both crews are indicated as deviations from the preflight mean values in table 39–II.

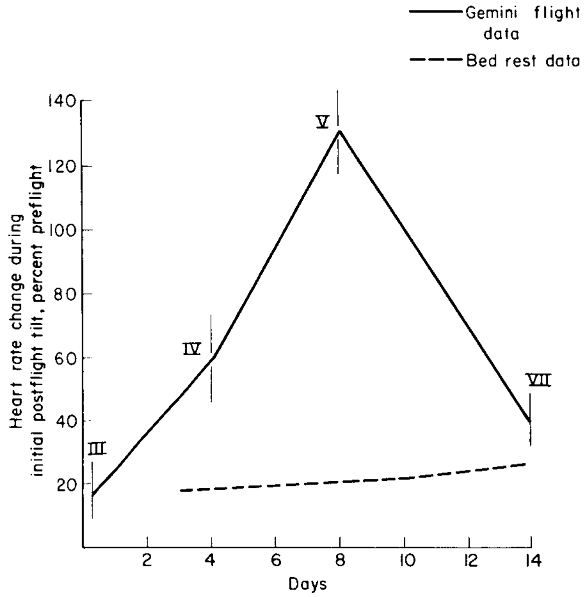

FIGURE 39-10.—Pulse-rate change after Gemini missions compared with bed-rest data.

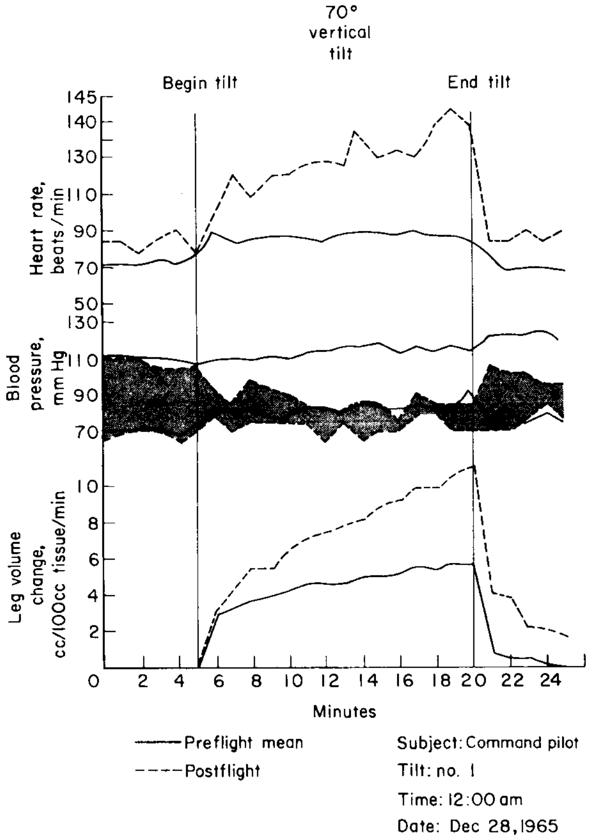

FIGURE 39-11.—Data from first tilt-table study of Gemini VII command pilot.

The Gemini V crew exhibited a higher postflight mean resting pulse rate than did the Gemini VII crew, with a maximal difference of 12-fold (pilot's) occurring 2 to 4 hours after recovery. This elevated resting pulse rate gradually returned to the preflight levels. The Gemini VII crew exhibited a slight increase in postflight mean resting pulse rate over preflight levels; these values returned to preflight levels approximately 24 hours after recovery. The crewmembers for both Gemini V and VII exhibited changes in their resting systolic and diastolic blood pressures after the missions. These values are indicated as deviations from the preflight mean values in table 39-III.

All crewmembers had a decreased resting systolic blood pressure 2 to 4 hours after recovery. The Gemini V command pilot and the Gemini VII pilot maintained a lower-than-preflight systolic pressure throughout the postflight test period. All crewmembers exhibited a decreased resting diastolic blood pressure during each postflight tilt test except during the first and last tilts for the Gemini V command pilot, and during the second tilt for the Gemini VII pilot. Daily changes in resting blood pressures are indicated in figures 39-9 and 39-19 as deviations from the preflight mean values.

TABLE 39-I.—*Summary of Tilt-Table Tests*

Subject	Mission	Pretilt		70° vertical tilt			Posttilt		
		Pulse rate	Blood pressure	Pulse rate	Blood pressure	Δ leg volume, percent	Pulse rate	Blood pressure	Δ leg volume, percent
Command pilot	V	58	109/72	75	111/79	+3.0	55	108/62	+0.3
	VII	59	117/68	78	120/79	+2.7	56	115/64	+.2
Pilot	V	73	110/72	87	114/81	+4.5	70	113/76	+.4
	VII	72	131/75	84	126/84	+4.4	70	123/73	+.5

TABLE 39-II.—*Change in Mean Resting Heart Rate*

[Data in beats per minute [a]]

Subject	Mission	Hours after recovery					
		2-4	8-12	24-30	48-56	72-80	90-104
Command pilot	V	+21	+32	+10	+6	+6	+9
	VII	+10	+8	−2	−1	----------	----------
Pilot	V	+59	+41	+18	0	+12	+19
	VII	+4	+9	+5	−5	----------	----------

[a] Positive values are above the preflight mean; negative values are below the preflight mean.

TABLE 39-III.—*Change in Mean Resting Blood Pressure*

[Data in mm of mercury [a]]

Subject	Mission	Hours after recovery											
		2-4 [b]		8-12 [b]		24-30 [b]		48-56 [b]		72-80 [b]		96-104 [b]	
Command pilot	V	−9	+10	−10	−8	−10	−3	−13	−9	−3	−3	−5	+6
	VII	−3	−3	+11	+9	+2	−3	+5	−5	-----	-----	-----	-----
Pilot	V	−3	−8	0	−9	+1	−8	+4	−9	+3	−3	+1	−6
	VII	−8	−4	−7	−2	−4	−4	−14	−5	-----	-----	-----	-----

[a] Positive values are above the preflight mean; negative values are below the preflight mean.
[b] Left value is systolic; right value is diastolic.

During the postflight tilts, all the Gemini V and VII crewmembers exhibited increased pulse rates. Highest rates were observed during the tilts performed 2 to 4 hours after recovery. Pulse rate increases over preflight mean values for each postflight tilt are indicated in table 39-IV.

TABLE 39-IV.—*Change in Mean Tilt Heart Rate*

[Data in beats per minute [a]]

Subject	Mission	Hours after recovery					
		2-4	8-12	24-30	48-56	72-80	90-104
Command pilot	V	+79	+69	+35	+14	+13	+21
	VII	+40	+19	+2	+4	----------	----------
Pilot	V	+86	+55	+21	+4	+11	+32
	VII	+28	+33	+34	+2	----------	----------

[a] Positive values are above the preflight mean; negative values are below the preflight mean.

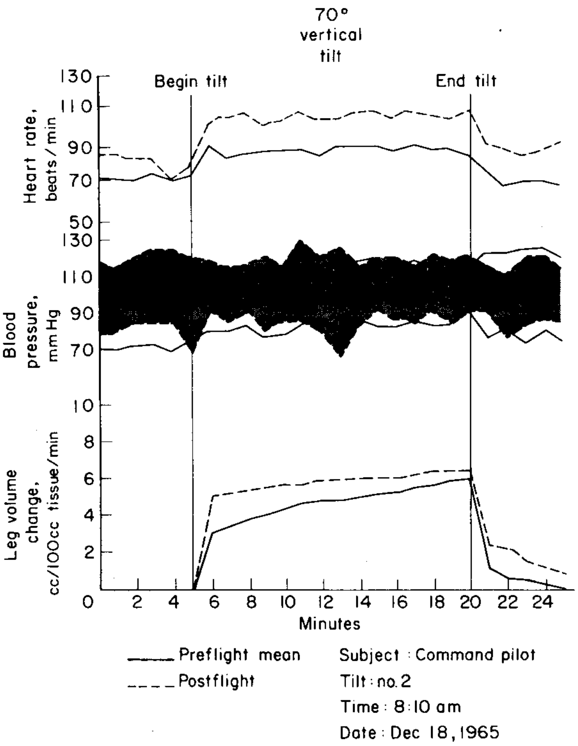

FIGURE 39-12.—Data from second tilt-table study of Gemini VII command pilot.

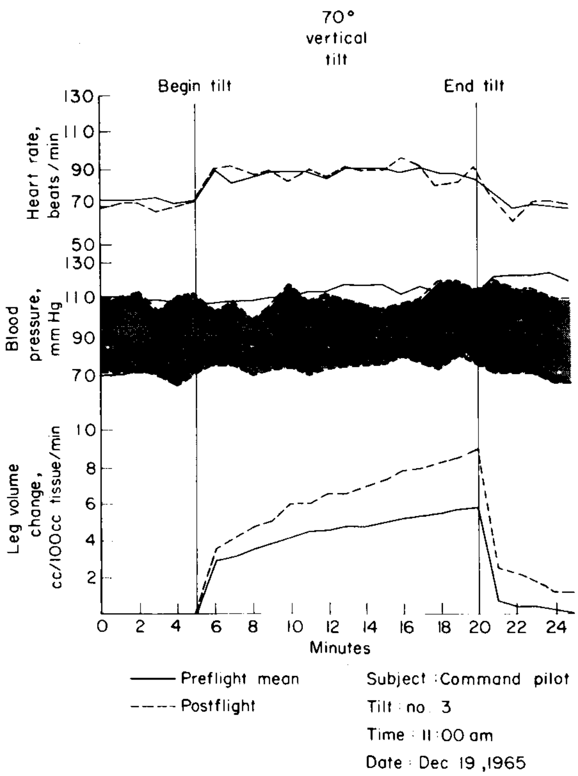

FIGURE 39-13.—Data from third tilt-table study of Gemini VII command pilot.

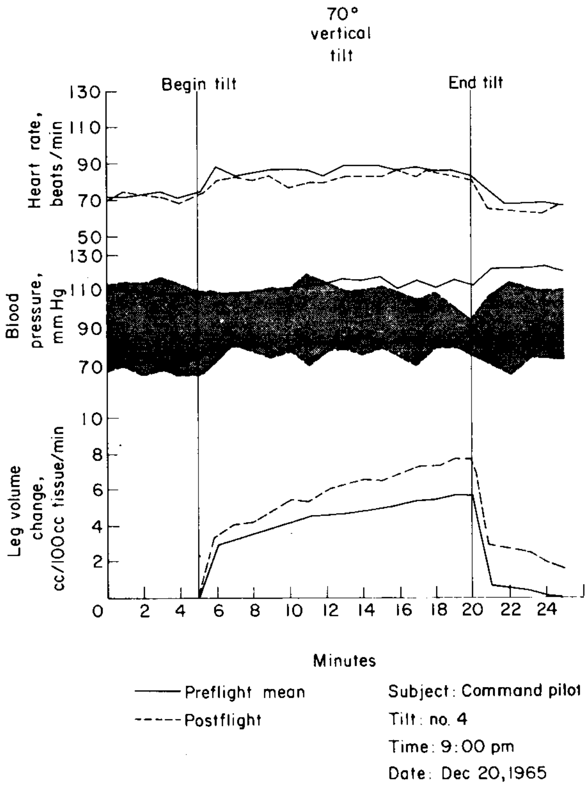

FIGURE 39-14.—Data from fourth tilt-table study of Gemini VII command pilot.

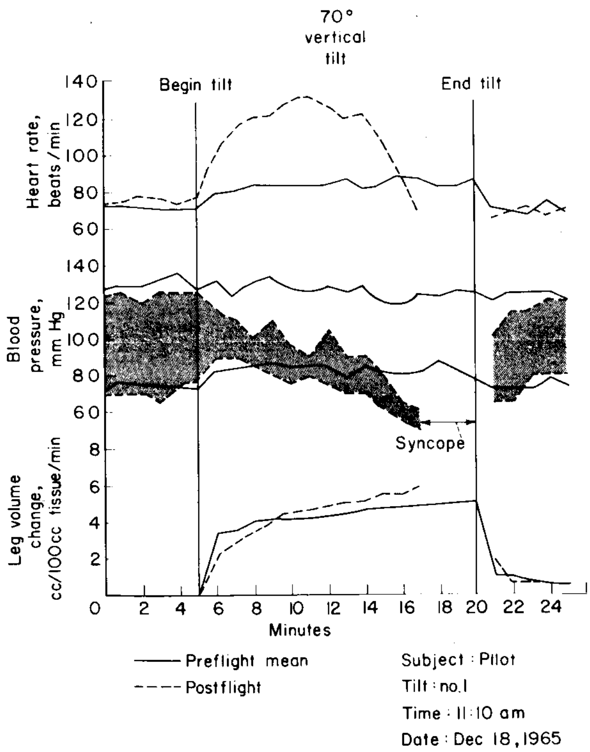

FIGURE 39-13.—Data from third tilt-table study of Gemini VII pilot.

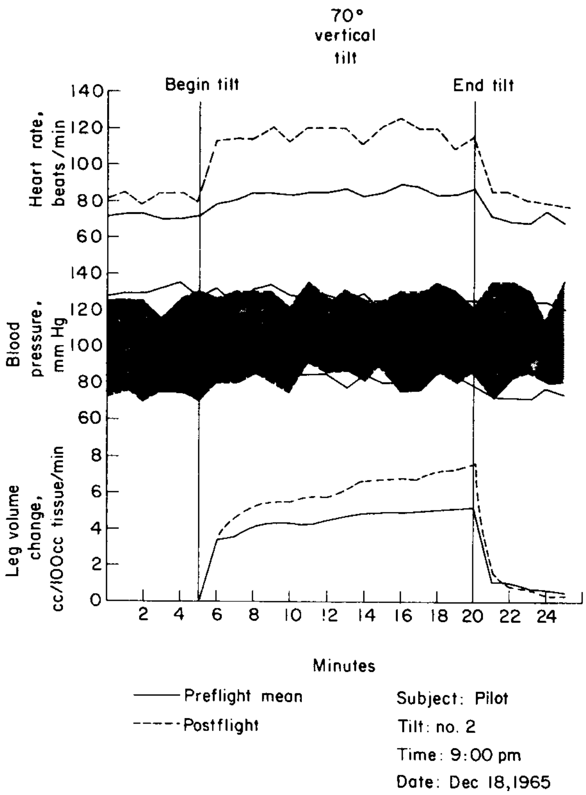

FIGURE 39–16.—Data from second tilt-table study of Gemini VII pilot.

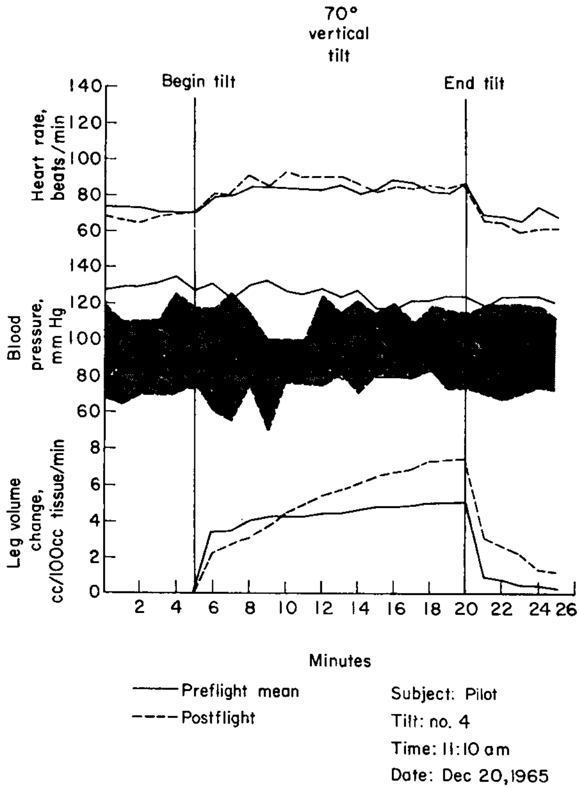

FIGURE 39–18.—Data from fourth tilt-table study of Gemini VII pilot.

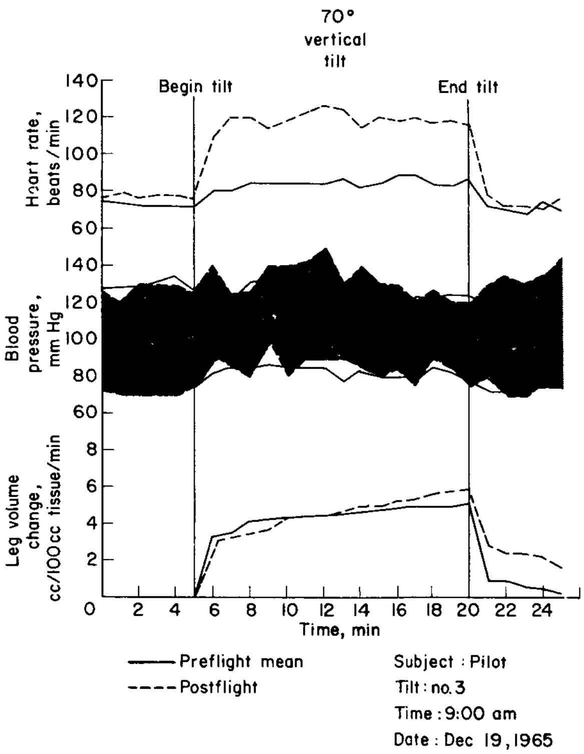

FIGURE 39–17.—Data from third tilt-table study of Gemini VII pilot.

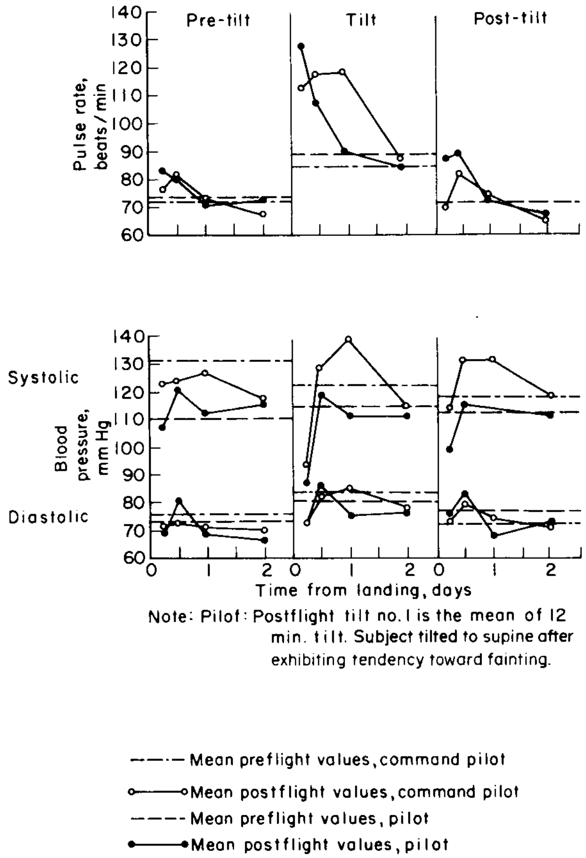

FIGURE 39-19.—Summary of tilt-table study for Gemini VII flight crew.

The Gemini V crew had a twofold greater increase in pulse rate than did the Gemini VII crew during the first two postflight tilts. Although the Gemini VII crew had a smaller increase in pulse rate during the tilt procedures, the Gemini VII pilot had to be returned to the supine position at the end of 12 minutes during the first tilt. This syncopal response was of the vasodepressor type and is illustrated in figure 39-15. This untoward experience on the first tilt procedure may account for his increased pulse rate during the second and third tilts. The pulse rates of all crewmembers decreased during succeeding tilts to near preflight levels (figs. 39-8 and 39-19).

All crewmembers exhibited narrowed pulse pressures during the first postflight tilt (compared with the preflight tilt and the postflight resting values). The Gemini V crew also exhibited a marked pulse pressure narrowing during the second (8 to 12 hours) postflight tilt. The Gemini V command pilot maintained a low systolic pressure during the third and fourth tilts, whereas the Gemini V pilot returned to normal preflight levels after the second postflight tilt. The Gemini VII crew revealed no marked pulse pressure narrowing during their second, third, or fourth postflight tilts. The changes in systolic and diastolic pressures for both crews are indicated as deviations from the preflight mean values in table 39-V.

During the postflight recovery phase, the blood pressure values for the Gemini V and VII crewmembers returned to near pretilt resting levels (figs. 39-8 and 39-19). Leg volume changes during the postflight tilts indicate that the pilots who wore the pneumatic cuffs did indeed pool significantly less blood in their legs during the tilts than did the command pilots. These values are indicated at percent increase above the preflight control values in table 39-VI.

TABLE 39-V.—*Changes in Mean Tilt Blood Pressure*

[Data in mm of mercury [a]]

Subject	Mission	Hours after recovery											
		2-4 [b]		8-12 [b]		24-30 [b]		48-56 [b]		72-80 [b]		96-104 [b]	
Command pilot	V	−16	+6	−13	+6	−6	+2	−9	−7	+11	+7	−8	+9
	VII	−27	−8	+5	+4	−3	−6	−4	−5	-----	-----	-----	---
Pilot	V	−20	−3	−12	+11	+6	+9	+8	+2	+8	+4	+7	+3
	VII	−33	−11	+2	−2	+6	+1	−12	−11	-----	-----	-----	---

[a] Positive values are above the preflight mean; negative values are below the preflight mean.
[b] Left value is systolic; right value is diastolic.

TABLE 39-VI.—*Change in Leg Blood Volume (cc/100cc Tissue per Minute)*

[Data in percent change above preflight mean]

Subject	Mission	Hours after recovery					
		2–4	8–12	24–30	48–56	72–80	96–104
Command pilot	V	89	149	44	73	78	111
	VII	71	31	47	33		
Pilot	V	87	73	25	57	117	97
	VII	2	36	9	15		

Although the Gemini VII pilot exhibited a vasodepressor type syncope during his first postflight tilt, he did not pool an excessive amount of blood in his legs (2 percent above the preflight control value). In addition, despite the fact that the V and VII command pilots pooled similar quantities of blood in their legs during the first postflight tilt, they differed considerably in the volume pooled during the remaining tilts. These differences, as well as those of the Gemini V pilot, may be a reflection primarily of differences in the state of hydration.

Changes in total blood volume, plasma volume, and red cell mass were determined before and after flight. Radioactive isotope (I^{125}, Cr^{51}) techniques were utilized in these measurements. The results are indicated as percent changes in table 39-VII.

TABLE 39-VII.—*Change in Intravascular Volume*

[Data in percent [a]]

Subject	Mission	Total blood volume	Plasma volume	Red cell mass
Command pilot	V	−13	−8	−20
	VII	0	+15	−19
Pilot	V	−12	−4	−20
	VII	0	+4	−7

[a] Positive values are above the preflight mean; negative values are below the preflight mean.

The Gemini VII crew sustained a 4- to 15-percent increase in plasma volume during the 14-day mission, whereas the Gemini V crew lost 4 to 8 percent of their plasma volume during the 8-day mission. Both crews lost 7 to 20 percent of their red cell mass. The Gemini VII pilot, however, sustained only a 7-percent decrease as compared with the 19- to 20-percent decrease of the other crewmembers. The decrease in red cell mass and the increase in plasma volume of the Gemini VII crew offset each other to give a net zero-percent change in total blood volume, whereas the reduction in plasma volume and the red cell mass of the Gemini V crew contributed to the measured 13-percent decrease in total blood volume. These changes in total blood volume may reflect, in part, the state of hydration of the Gemini V crew, but this is not true in the case of the Gemini VII crew. The postflight changes in body weight are indicated in table 39-VIII.

TABLE 39-VIII.—*Nude Body Weight Changes*

[Negative values indicate weight loss]

Subject	Mission	Pounds
Command pilot	V	−7.5
	VII	−10.0
Pilot	V	−8.5
	VII	−6.5

The Gemini V command pilot and pilot sustained a 7.5- and 8.5-pound loss in body weight, respectively. The Gemini VII command pilot and pilot lost 10.0 and 6.5 pounds, respectively. These values are similar to those observed after previous missions of shorter duration.

Discussion

The flight conditions operative during the Gemini VII mission were notably different from those of the Gemini V flight. These variables or differences were of sufficient magnitude that a comparison of the M-1 results on the two missions is difficult, if not impossible. Gemini VII was decidedly different from previous Gemini flights in that the Gemini VII crew did not wear their suits during an extensive portion of the 14-day flight. Their food and water intake was more nearly optimal than in previous flights; this assured better hydration and electrolyte balance, and the Gemini VII exercise regimen was more rigorous than that utilized on previous flights. These variables, in addition to the usual individual variability always present, preclude any direct comparison of M-1 results on the two missions. This is particularly true since the pulsatile cuffs were operative during only the first half of the 8-day Gemini V mission. The Gemini VII pilot's physiological measurements should be compared only with those of the command pilot who served as the "control" subject.

It is indeed true that the postflight physiological responses of the Gemini VII crew were vastly different from, and generally improved over, those observed in the Gemini V crew. It is difficult, however, to determine which of the previously mentioned variables were responsible for the observed improvement. This improvement is perhaps best shown in figure 39-8, which depicts the change in heart rate during the initial postflight tilts expressed as a percentage change with respect to the preflight value. The responses of the Gemini VII crew were far superior to the responses observed in the Gemini IV and V crews, and they were very nearly comparable to the response following 14 days of recumbency.

Additional comparisons between the Gemini VII and V crews may be summarized as follows:

(1) The Gemini VII crew exhibited less increase in postflight mean resting pulse rate (4 and 10 beats per minute versus 21 and 59 beats per minute).

(2) The Gemini VII crew exhibited signs of orthostatic intolerance for only 24 hours postflight; the Gemini V crew exhibited these signs for 24 to 48 hours.

(3) The Gemini VII crew pooled less blood in their lower extremities during all postflight tilts.

(4) The Gemini VII crew exhibited less pronounced changes in intravascular fluid volumes in the postflight period as shown in the following:

(a) Total blood volume: 0 percent versus 13 percent

(b) Plasma volume: +15 percent and +4 percent versus −8 percent and −4 percent.

(c) Red cell mass: −19 percent and −7 percent versus −20 percent and −20 percent.

(5) The Gemini VII crew lost 10.0 pounds (command pilot) and 6.5 pounds (pilot) during their flight, while the Gemini V crew lost 7.5 and 8.5 pounds, respectively.

(6) The Gemini VII crew regained less body weight during the first 24 hours postflight (40 percent and 25 percent versus 50 percent).

The physiological findings in the Gemini V crew have been previously reported (ref. 2) and will only be summarized here.

(1) The pilot's resting pulse rate and blood pressure returned to preflight resting levels within 48 hours after recovery; the command pilot required a somewhat longer period.

(2) The pilot's pulse pressure narrowed during tilt and at rest was less pronounced than that of the command pilot.

(3) The pilot's plasma volume decreased 4 percent, and the command pilot's decreased 8 percent.

(4) The pilot's body weight loss was 7.5 pounds; the command pilot's was 8.5 pounds.

(5) The pooling of blood in the legs of the pilot was generally less than that observed in the command pilot.

The observed differences between the Gemini V command pilot and pilot probably reflect only individual variability and cannot be construed as demonstrating any protective effect of the pulsatile thigh cuffs. The Gemini V tilt data are summarized in figures 39-9 and 39-10.

Tilt-table data are graphically presented in figures 39-11 through 39-14 for the command pilot and in figures 39-15 through 39-18 for the pilot. All the Gemini VII tilt data are summarized in figure 39-19. During the first postflight tilt, the pilot exhibited signs of vasodepressor syncope; the procedure was interrupted, and the pilot was returned to the supine position. This episode occurred despite the fact that there was no evidence of increased pooling of blood in the lower extremities. In subsequent tilts, the pilot exhibited no further signs of syncope or impending syncope. It is of significance that this episode of syncope occurred despite the fact that the measured blood volume of both crewmembers was unchanged from preflight levels.

It would seem possible that this syncopal episode was the result of sudden vasodilitation with pooling of blood in the splanchnic area, diminished venous return, diminished cardiac output, and decline in cerebral bloodflow.

As previously mentioned, there was no diminution in the total blood volume of either crewmember after the mission. The pilot's plasma volume increased 4 percent; the command pilot's increased 15 percent. The pilot's red cell mass decreased 7 percent; the command pilot's, 19 percent. The pilot lost 6.5 pounds (nude body weight) during the mission and replaced 25 percent of this loss during the first 24 hours after recovery. The command pilot lost 10.0 pounds and replaced 40 percent of this value within the first 24 hours following recovery.

The pilot's subsequent tilts revealed a moderate cardioacceleration during tilts 2 and 3, with normal pulse pressure and insignificant pooling of blood in the lower extremities (figs. 39–16, 39–17, and 39–18). The command pilot exhibited moderate cardioacceleration, marked narrowing of the pulse pressure, and increased pooling of blood in the lower extremities during the initial postflight tilt. Subsequent tilts revealed a rather rapid return to normal of heart rate and pulse pressure, but a greater tendency to pool blood in the legs than was observed in the pilot.

Conclusions

On the basis of the preflight and postflight data, it must be concluded that the pulsatile cuffs were not effective in lessening postflight orthostatic intolerance. This conclusion is based not on the occurrence of syncope during the pilot's first tilt, but rather on the higher heart rates observed during subsequent tilts, as compared with the control subject. It is well established that syncope in itself is a poor indicator of the extent or degree of cardiovascular deconditioning.

The pulsatile cuffs appeared to be effective in lessening the degree of postflight pooling of blood in the lower extremities as judged by the strain gage technique.

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40. EXPERIMENT M-3, INFLIGHT EXERCISE—WORK TOLERANCE

By LAWRENCE F. DIETLEIN, M.D., *Assistant Chief for Medical Support, Crew Systems Division, NASA Manned Spacecraft Center; and* RITA M. RAPP, *Crew Systems Division, NASA Manned Spacecraft Center*

Summary

The response of the cardiovascular system to a quantified workload is an index of the general physical condition of an individual. Utilizing mild exercise as a provocative stimulus, no significant decrement in the physical condition of either of the Gemini VII crewmembers was apparent. The rate of return of the pulse rate to preexercise levels, following inflight exercise periods, was essentially the same as that observed during preflight baseline studies.

Objective

The objective of Experiment M-3 was the day-to-day evaluation of the general physical condition of the flight crew with increasing time under space flight conditions. The basis of this evaluation was the response of the cardiovascular system (pulse rate) to a calibrated workload.

Equipment

The exercise device (figs. 40-1 and 40-2) consisted of a pair of rubber bungee cords attached to a nylon handle at one end and to a nylon foot strap at the other. A stainless-steel stop cable limited the stretch length of the rubber bungee cords and fixed the isotonic workload of each pull. The device could be utilized to exercise the lower extremities by holding the feet stationary and pulling on the handle. Flight bioinstrumentation (fig. 40-3) was utilized to obtain pulse rate, blood pressure, and respiration rate. These data were recorded on the onboard biomedical magnetic tape recorder and simultaneously telemetered to the ground monitoring stations for real-time evaluation.

Procedure

The device used in Gemini VII required 70 pounds of force to stretch the rubber bungee cords maximally through an excursion of 12

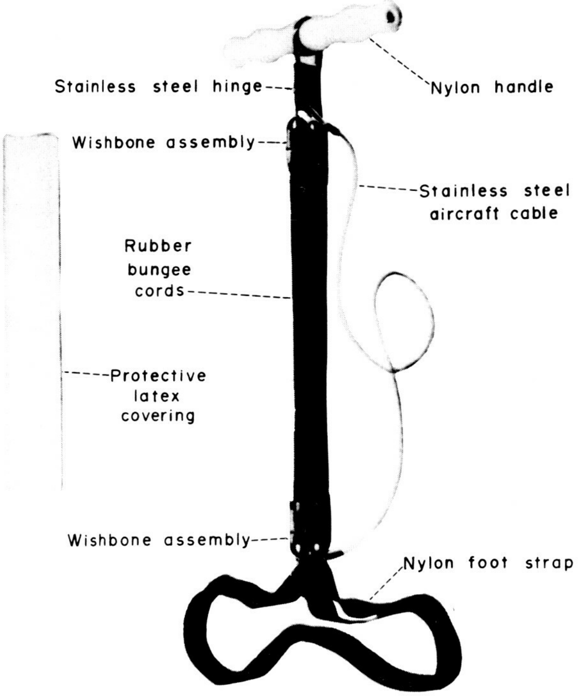

FIGURE 40-1.—Inflight exerciser major components.

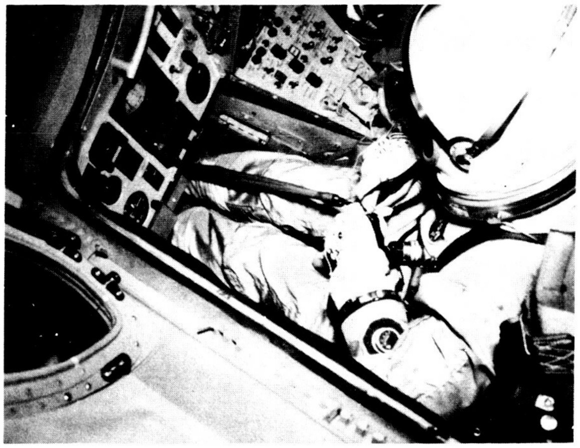

FIGURE 40-2.—Inflight exerciser in use.

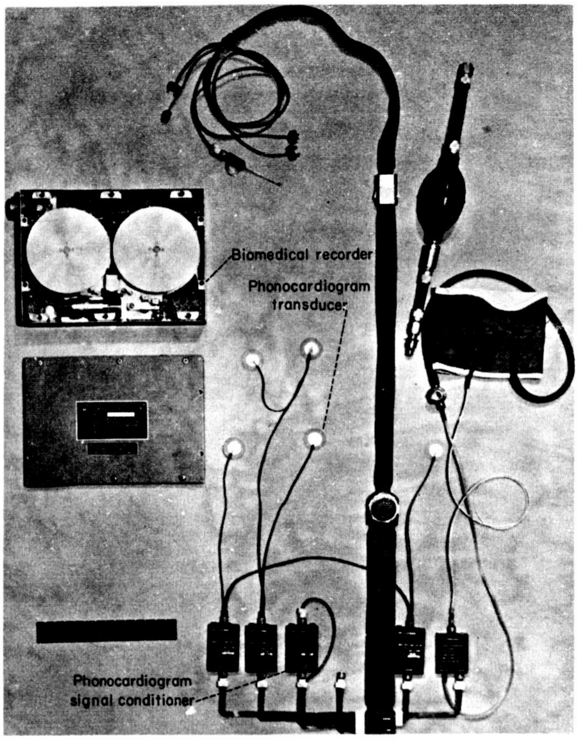

FIGURE 40-3.—Biomedical and communications harness used during Gemini IV mission.

inches. Exercise periods lasted for 30 seconds, during which time the astronaut stretched the bungee cords through a full excursion once per second. Exercise periods (crew status reports) were scheduled twice daily for each crewmember. Additional isometric-isotonic exercises were performed by each astronaut approximately three times daily. Blood pressure measurements were obtained before and after each exercise period (crew status report).

Results

The flight crew performed the exercises as scheduled. Heart rates were determined by counting 15-second periods for 2 minutes before and following exercise, as well as the first and last 15-second periods during each exercise. Comparison of 1-g preflight exercise periods with succeeding periods also revealed little difference in heart-rate response. Inflight responses to exercise are graphically illustrated in figure 40-4. Heart rates are plotted for the command pilot and pilot before, during, and following exercise. Both astronauts exhibited a moderate rise in pulse rate during exercise, with a rapid return to near preexercise levels within 1 minute following exercise. Similar M-3 results have been previously reported for the Gemini IV and Gemini V crews (refs. 1 and 2).

Representative preexercise and postexercise blood pressures are illustrated in figures 40-5 and 40-6 for the command pilot. The systolic values tended to be slightly higher following exercise. Diastolic values were more variable, but generally tended to be slightly higher following exercise. Samples of telemetered physiological data obtained during a typical inflight exercise are illustrated in figure 40-7.

Conclusions

The M-3 experiment on Gemini VII was successfully performed. On the basis of the data obtained during this mission, the following conclusions appear warranted:

(1) The response of the cardiovascular system to a calibrated workload is relatively constant for a given individual during space flights lasting 14 days.

(2) The crewmembers are able to perform mild-to-moderate amounts of work under the conditions of space flight and within the confines of the Gemini spacecraft. This ability continues essentially unchanged for missions up to 14 days.

(3) Using a variant of the Harvard Step Test as an index, no decrement in the physical condition of the crew was apparent during the 14-day missions, at least under the stress of the relatively mild workloads imposed in this experiment.

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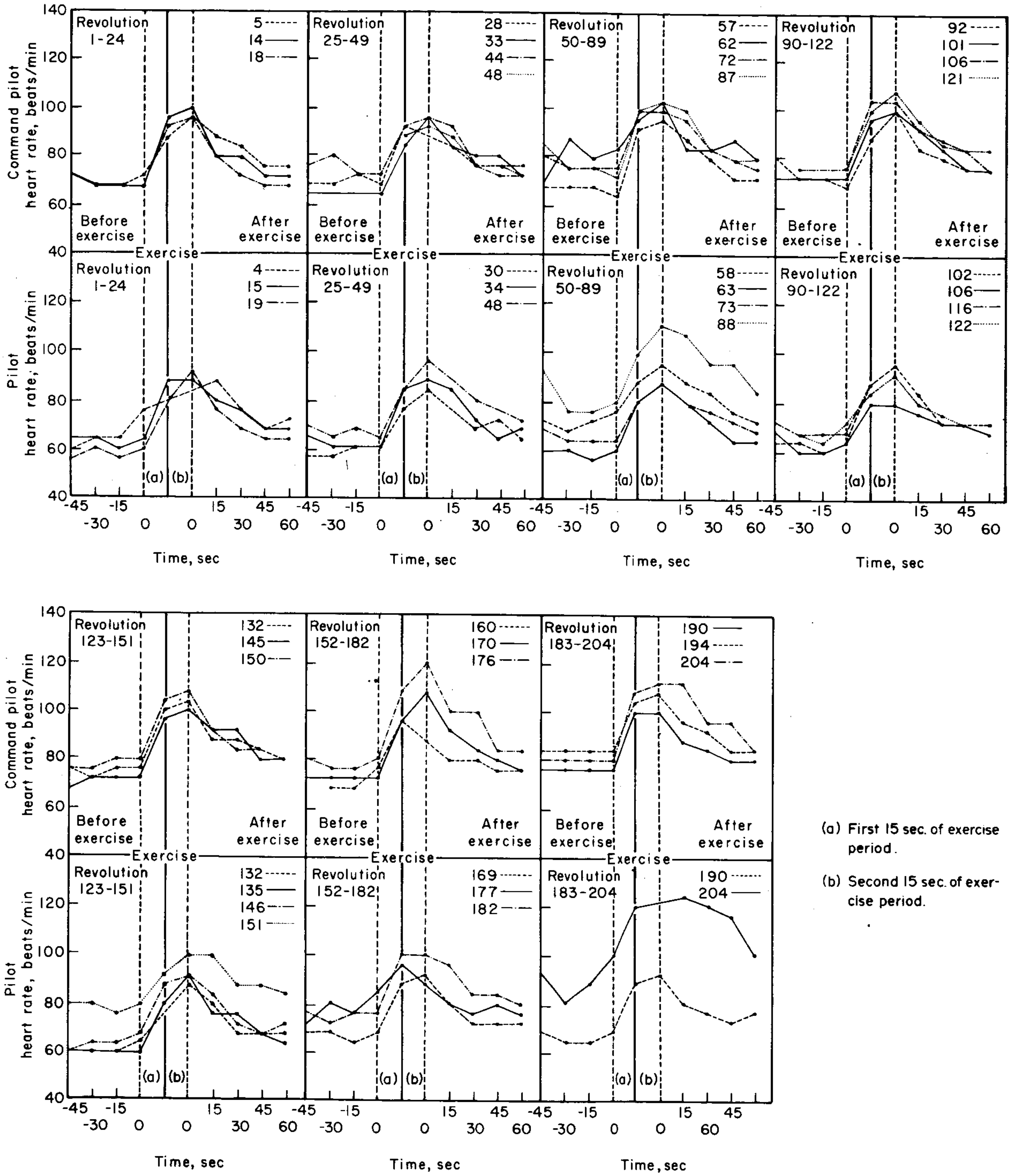

FIGURE 40-4.—Inflight responses to exercise.

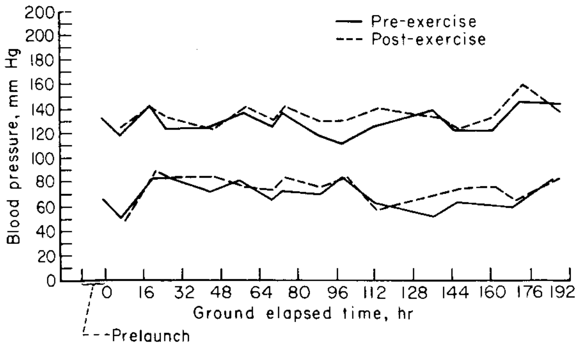

FIGURE 40–5.—Blood pressure of Gemini VII command pilot from lift-off through 192 hours ground elapsed time.

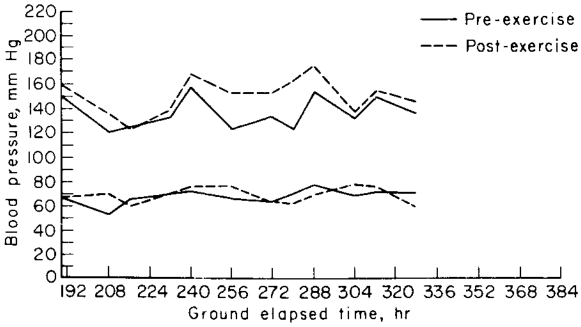

FIGURE 40–6.—Blood pressure of Gemini VII command pilot from 192 through 322 hours ground elapsed time.

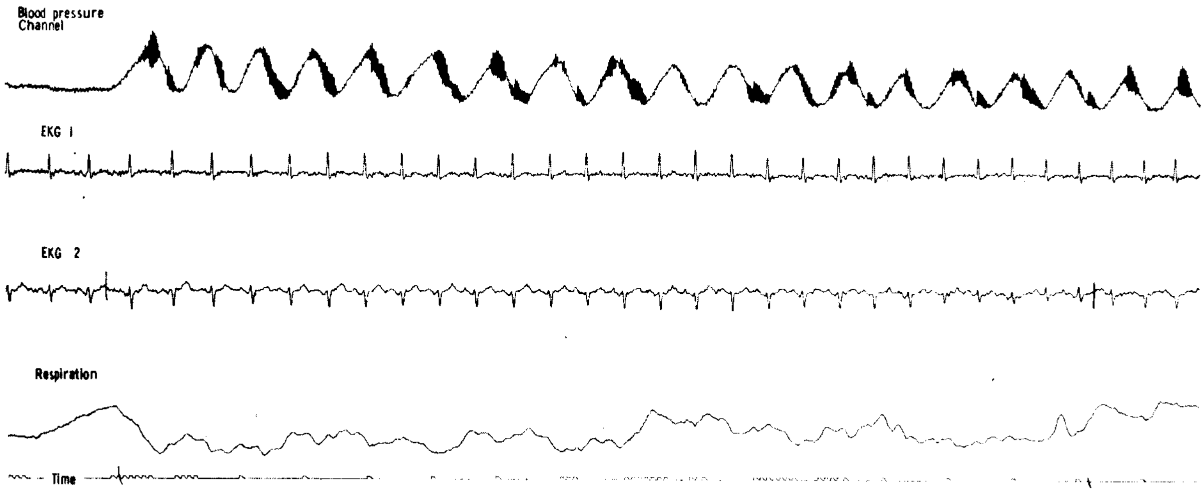

FIGURE 40–7.—Sample of telemetered physiological data during inflight exercise. (Recorder speed, 25 mm/sec.)

41. EXPERIMENT M–4, INFLIGHT PHONOCARDIOGRAM—MEASUREMENTS OF THE DURATION OF THE CARDIAC CYCLE AND ITS PHASES DURING THE ORBITAL FLIGHT OF GEMINI V

By LAWRENCE F. DIETLEIN, M.D., *Assistant Chief for Medical Support, Crew Systems Division, NASA Manned Spacecraft Center, and* CARLOS VALLBONA, M.D., *Texas Institute of Rehabilitation and Research, Baylor University College of Medicine*

Summary

Simultaneous electrocardiographic and phonocardiographic records were obtained from both Gemini V crewmembers. Analysis of these data revealed:

(1) Wide fluctuations of the duration of the cardiac cycle within physiological limits throughout the mission.

(2) Fluctuations in the duration of electromechanical systole that correlated with changes in heart rate.

(3) Stable values for electromechanical delay (onset of QRS to onset of first heart sound) throughout the mission, with shorter values observed at the peak heart rates recorded during lift-off and reentry.

(4) Higher values for the duration of systole and for electromechanical delay in the command pilot than in the pilot, suggesting preponderance of cholinergic influences (vagal tone) in the command pilot.

(5) Evidence of adrenergic reaction (sympathetic tone) at lift-off, at reentry, and in the few hours that preceded reentry.

Objective

The objective of Experiment M–4 was to measure the electrical and mechanical phases of the cardiac cycle of both astronauts throughout the flight of Gemini V in order to gain information on the functional cardiac status of flight crewmembers during prolonged space flights.

Equipment

The experimental equipment system consisted of three distinct parts, including the following: (1) a phonocardiographic transducer; (2) an electrocardiographic signal conditioner (preamplifier and amplifier); and (3) an onboard biomedical tape recorder.

The transducers and signal conditioners were housed within the Gemini pressure suit. The phonocardiographic sensor was applied parasternally in the left-fourth intercostal space of each flight crewmember. Electrodes for the detection of the electrocardiographic signals were applied in the usual location for the manubrium-xiphoid (MX) lead.

The phonocardiographic transducer used on Gemini V was identical with that used in Gemini IV (ref. 1). It consisted of a 7-gram piezoelectric microphone 1 inch in diameter and 0.200 inch in thickness (fig. 41–1), and was developed by the Bioinstrumentation Section of the Crew Systems Division. The transducer or sensor responds to the translational vibrations imparted to the chest wall with each contraction of the heart. The sensor was secured to the chest wall of each astronaut by means of a small disk of doublebacked adhesive. A 10-inch length of

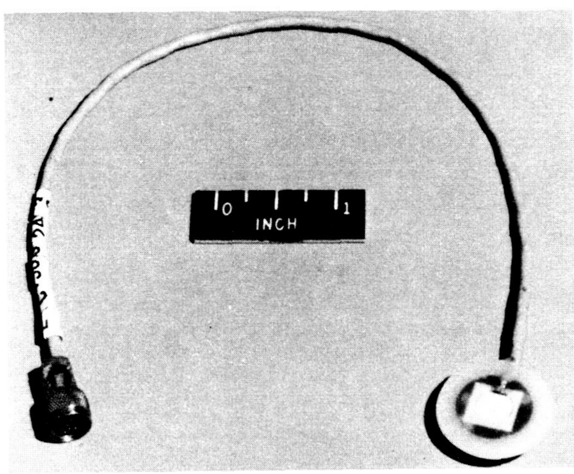

FIGURE 41–1.—Phonocardiogram transducer.

flexible 0.10-inch-diameter shielded cable transmitted the phonocardiographic signal to the Gemini signal conditioner (fig. 41–2) housed in a pocket of the undergarment. The phonocardiographic signal was then relayed from the signal conditioner output to the suit bioplug and thence to the biomedical magnetic tape recorder (fig. 41–3).

The electrocardiogram and the phonocardiogram of each astronaut were recorded simultaneously throughout the mission. The recording procedure was entirely passive and did not require active participation on the part of the flight crewmembers.

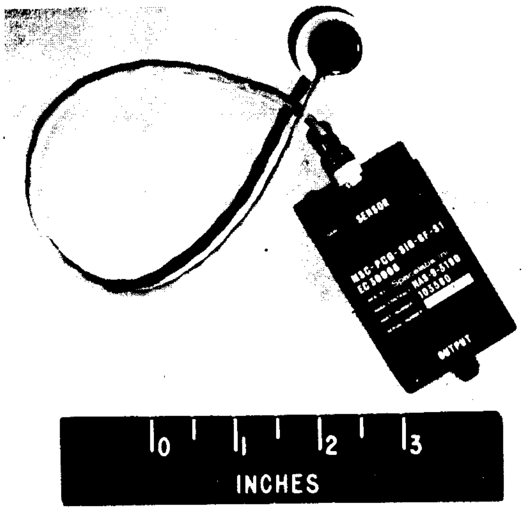

FIGURE 41–2.—Phonocardiograph system.

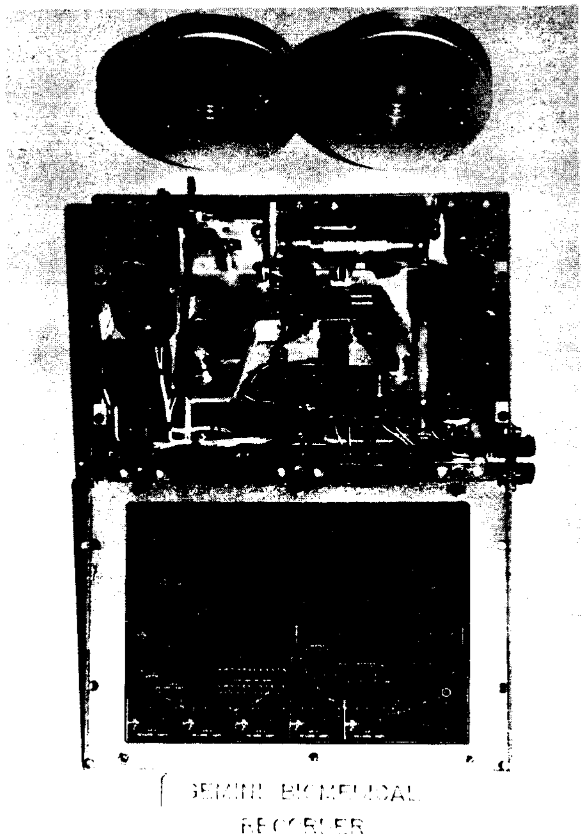

FIGURE 41–3.—Biomedical recorder.

Procedure

Experiment M–4 was accomplished in Gemini V by means of the instrumentation system described above.

The analog data from the biomedical tape recording were played back in real time, digitized, and then analyzed by computer techniques.

The playback protocol included the following periods: (1) Initial: continuous for 9 minutes, starting at 1 minute before lift-off until orbital insertion; and (2) Final: continuous from 5 minutes before reentry until touchdown. In addition, records approximately 1 minute in duration were obtained at hourly intervals for the first 24 hours of the mission and at 4-hour intervals for the remainder of the mission until 5 minutes before reentry.

The analog records of electrocardiogram and phonocardiogram were semiautomatically digitized with a Telecordex analog-to-digital converter. Digital readings were obtained at each of the following points: (1) at the onset of a QRS complex; (2) at the onset of the first heart sound; (3) at the onset of the second heart sound; and (4) at the onset of the next QRS complex. A computer program provided calculations of the duration of each RR interval, the duration of the mechanical systole (plus excitation time), the duration of diastole, the interval between the onset of QRS and the first heart sound (electromechanical delay), and the interval between the first and second heart sounds. The same program computed the means and standard deviations of these variables after each 15 consecutive beats.

Results and Discussion

Both astronauts had similar patterns of change in the duration of the cardiac cycle and of its several phases throughout the mission, but quantitative differences between the two subjects warrant separate discussions.

Results on the Command Pilot

Figure 41–4 indicates the serial plot of measurements throughout the mission. In the records that were obtained just before lift-off, the total duration of the cardiac cycle was 455 milliseconds (equivalent to a heart rate of 132 beats per minute). Electromechanical systole (mechanical systole plus excitation time) lasted 345 milliseconds; electromechanical delay (onset of QRS to first heart sound) was 100 milliseconds; and the interval between the onset of the first and second heart sound was 245 milliseconds. At lift-off, the duration of the cardiac cycle was 345 milliseconds (equivalent to a heart rate of 173 beats per minute). The cardiac cycle gradually increased in duration (cardiac deceleration) after orbital insertion, and a stabilization occurred at approximately 14 hours after lift-off. A significant shortening of the cardiac cycle, with shortening of systole and slight shortening of the electromechanical delay, occurred during a period of exercise at 9 hours 13 minutes after lift-off when the heart rate rose from a value of 75 to 125 per minute. Throughout the mission, there were wide fluctuations in the cardiac cycle (plot R of fig. 41–4), which seemed to correlate with concomitant changes in the duration of electromechanical systole (plot S of fig. 41–4) and the time interval between the first and second heart sounds (plot X of fig. 41–4). The electromechanical delay (time interval between the onset of QRS and of the first heart sound) remained relatively constant throughout the mission, although, as discussed later, the values were higher at lower heart rates. It is noteworthy that the electromechanical delay became slightly shorter approximately 12 hours before reentry, at which time the peak heart rate was recorded at 137 beats per minute. The duration of systole also became considerably shorter at this time.

Figure 41–5 reveals the fluctuations of the heart rate observed throughout the mission. From the tenth hour after lift-off to approximately 7 hours before reentry, Command Pilot Cooper had consistently low heart rates, with an overall average of approximately 68 beats per minute. The lowest values were recorded on the fourth and fifth days of the mission (50 beats per minute). It is interesting that the highest values of heart rate were recorded usually a few hours before midnight, eastern standard time. This was particularly evident during the last 3 days of the mission and suggests persistence of the circadian rhythmicity of heart rate based on the normal Cape Kennedy day-night cycle. Similar observations had been previously made in the command pilot of Gemini IV (ref. 1).

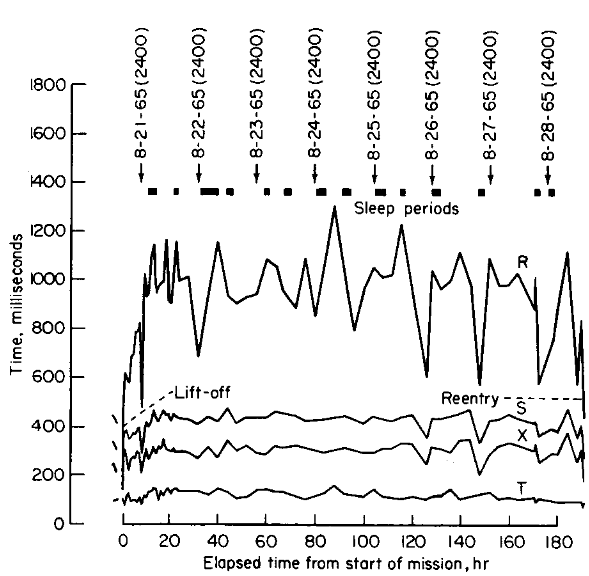

FIGURE 41–4.—Cardiac measurements for Gemini V command pilot.

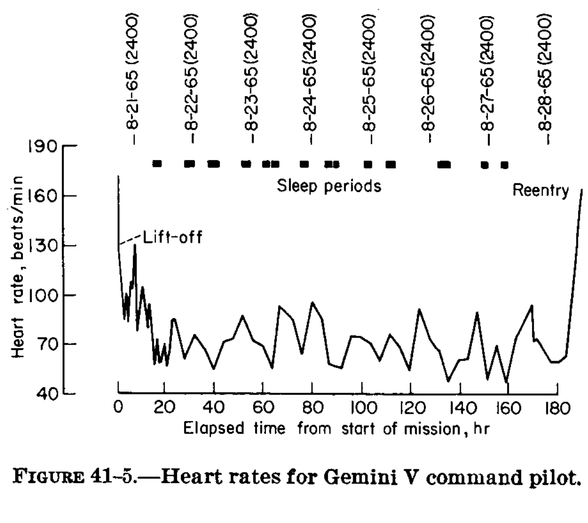

FIGURE 41–5.—Heart rates for Gemini V command pilot.

Figure 41–6 illustrates the correlation between heart rate and the duration of electromechanical systole and electromechanical delay. The average values for the duration of the cardiac cycle (R) at different time periods are plotted along the ordinate. The corresponding average values for the duration of electromechanical systole (S), for electromechanical delay (T), and for the time interval between the first and second heart sounds (X) are plotted along the abscissa. It is clear that in general the values of S, X, and T were longer when the total duration of the cardiac cycle was also longer (that is, when the heart rate was lower). It is remarkable that practically all the systolic values were longer in the case of the command pilot than those predicted for healthy subjects, using the regression equation proposed by Hegglin and Holzmann (ref. 2). Only at the time of lift-off and reentry were the values of S closer to the predicted norms.

Since it has been observed that cholinergic influences produce a relative prolongation of mechanical systole as well as a tendency toward lower heart rates, it may be concluded that Command Pilot Cooper had a preponderance of vagal tone throughout the mission. An increased vagal tone was suggested also by the marked respiratory sinus arrhythmia (respiration heart rate reflex) which was evident during periods of reduced activity and sleep.

Scant information is available on the relationship between electromechanical delay and heart rate. In general, the value of T remains almost

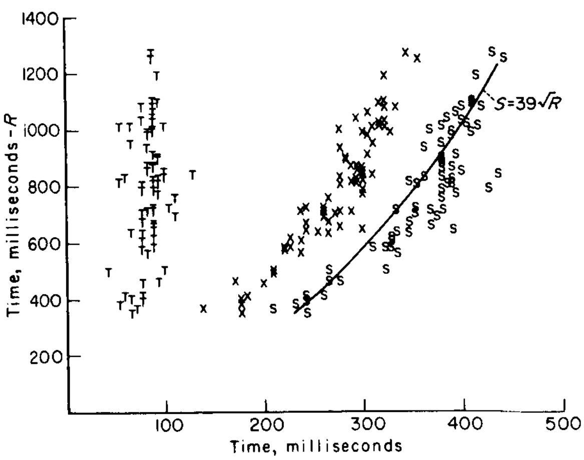

FIGURE 41–6.—Correlation of cardiac measurements for Gemini V command pilot.

constant at about 100 milliseconds when the heart rate varies between 60 and 120 per minute. The T values for the command pilot were greater, and the longest duration observed was 150 to 160 milliseconds during the fourth and fifth days of the mission. It must be emphasized, however, that the longest delays occurred at the lowest heart rates, suggesting that a preponderance of vagal tone also influenced the delay. It is likely that the stressful circumstances of lift-off and reentry accounted for the observed adrenergic effects on the heart. An increased heart rate and an absolute and relative shortening of mechanical systole and of electromechanical delay were the result of these adrenergic influences.

A prolongation of the electromechanical delay had been reported by Baevskii and Gazenko (ref. 3) during the flight of Cosmonaut Titov. The observations made of Astronaut Cooper suggest that increased vagal tone accounted for this prolongation, but since, in the case of Astronaut Cooper, manifestations of nausea or other untoward signs of vagal preponderance did not occur, we may conclude that the finding of prolonged electromechanical delay did not have any pathological significance, and was perhaps only a manifestation of superb physical conditioning.

Results on the Pilot

The responses observed in Pilot Conrad were similar to those observed in Command Pilot Cooper, but there were quantitative differences (fig. 41–7). The duration of Conrad's cardiac cycle just before lift-off averaged 460 milliseconds (equivalent to a heart rate of 130 beats per minute). The average duration of electromechanical systole was 305 milliseconds; that of electromechanical delay, 70 milliseconds; and that of the time interval between the first and second heart sounds, 235 milliseconds. At lift-off, the shortest cardiac cycle corresponded to a heart rate of 171 beats per minute. There was a gradual deceleration after insertion into orbit, and the values became stable at approximately 16 hours from the onset of the mission. Throughout the mission, the duration of the cardiac cycle varied considerably, with concomitant changes in the duration of systole (S) and of the time interval between the first and second heart sounds (X). The electromechanical delay (T) remained relatively constant, but there was a significant shortening that began approximately 20 hours before reentry. Low values for the duration of the cardiac cycle and its various components were observed at the time of reentry when the duration of the cardiac cycle was 365 milliseconds (equivalent to a heart rate of 164 beats a minute). At that time, mechanical systole reached its lowest value (220 milliseconds), and electromechanical delay was measured at 75 milliseconds.

The heart rate fluctuated throughout the mission, but in general the average values were somewhat higher than those of the command pilot (fig. 41–8). In addition to the peak values at lift-off and at reentry, there was also a high value shortly after the ninth hour when the flight schedule called for a period of physical exercise. At that time the heart rate peaked at 130 beats per minute. Circadian fluctuations of the heart rate were not so evident in the case of the pilot as compared with the command pilot, although peaks of heart rate were also recorded in the evening hours of the last 3 days of the mission.

In contrast to what was observed in the case of the command pilot, the values of the duration of electromechanical systole (S) for Pilot Conrad were closer to normal throughout the mission (fig. 41–9). Values of systole shorter than those predicted were measured at the time of reentry. A correlation between the electromechanical delay (T) and the duration of the cardiac cycle (R) was not as evident in the pilot as in the command pilot, but in general the lowest values were measured at the peak heart rates recorded at lift-off and at reentry. These findings suggest that vagal preponderance in Pilot Conrad was less prominent than that observed in the command pilot, and that adrenergic influences may have prevailed occasionally during the mission. These observations correlate well with findings of numerous extrasystoles during the first hours of the mission and at the time of reentry. Extrasystoles occurred at random throughout the mission but not so frequently as during lift-off and reentry.

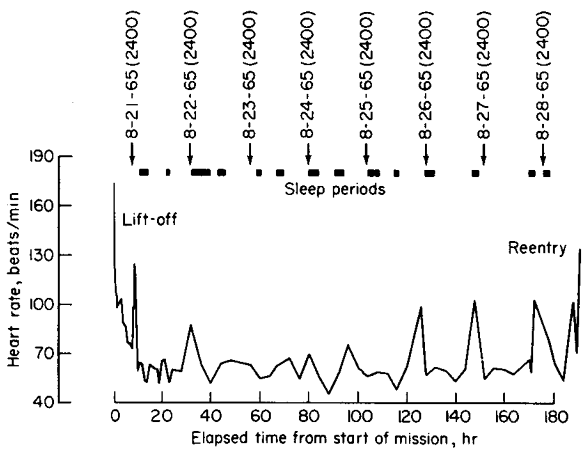

FIGURE 41–8.—Heart rates for Gemini V pilot.

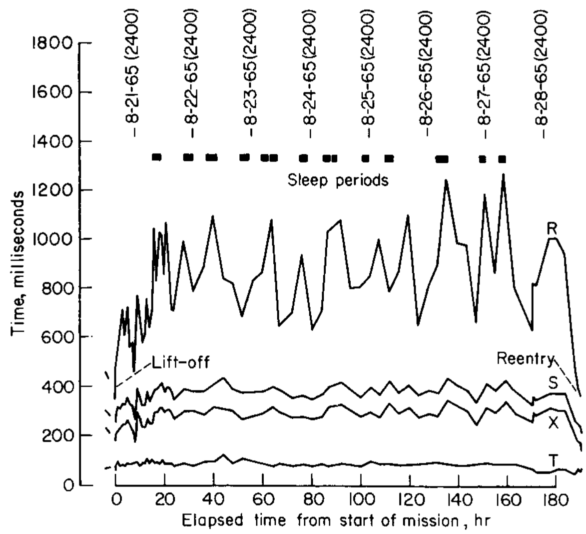

FIGURE 41–7.—Cardiac measurements for Gemini V pilot.

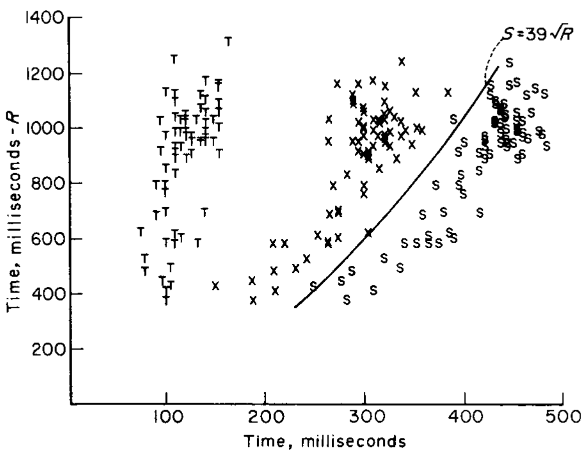

FIGURE 41–9.—Correlation of cardiac measurements for Gemini V pilot.

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42. EXPERIMENT M–5, BIOASSAYS OF BODY FLUIDS

By LAWRENCE F. DIETLEIN, M.D., *Assistant Chief for Medical Support, Crew Systems Division, NASA Manned Spacecraft Center, and* E. HARRIS, Ph. D., *Crew Systems Division, NASA Manned Spacecraft Center*

Objective

Medical Experiment M–5 is designed to obtain objective data concerning the effect of space flight on several of the systems of the human body. This experiment, as part of an overall evaluation, addresses itself to those areas where effects can be observed by alterations in the chemistries of body fluids.

Procedures

Inflight and postflight steroid and catecholamine values provide a means for assessing the extent of the stresses to which the crewman is subjected, and provide a measurement of the physiological cost to the crewman in maintaining a given level of performance during space flight.

To assess the effects of space flight upon the electrolyte and water metabolism of the crewman, plasma and urinary electrolytes and urine output values are determined along with the antidiuretic hormone (ADH) and the aldosterone.

The readily recoverable weight loss during flight may be related to water loss. Water loss, in turn, may be of urinary, sweat, or insensible origin. The fluid intake and urinary output, along with changes in the hormone and electrolyte concentrations, can be measured in the recovered samples. Plasma and urine samples are analyzed before flight to obtain baseline data. During flight, only the urine is sampled. To accomplish this and to obtain the total voided volumes, a urine-sampling and volume-measuring system is used (fig. 42–1). The system consists of a valve which introduces a fixed quantity of tritiated water into each voiding. A sample of approximately 75 milliliters of each voiding is taken after adding the isotope. Upon recovery, the total volume can be calculated by measuring the dilution of the tritium in the sample. Benzoic acid is used as the preservative.

Immediately upon recovery, the first postflight plasma sample is obtained. Samples are taken at 6, 24, and 72 hours after flight. Urine is collected continuously for 48 hours after flight. Each sample is frozen and returned to the Manned Spacecraft Center for analysis.

The following analyses are performed:

(1) Plasma/Serum
- a. 17-hydroxycorticosteroids
- b. Proteins
  - 1. Total
  - 2. Albumin/globulin ratio
  - 3. Electrophoretic pattern
- c. Antidiuretic hormone
- d. Hydroxyproline
- e. Electrolytes, the ions of sodium, potassium, calcium, chlorine, and phosphate
- f. Bilirubin
- g. Uric acid

(2) Urine
- a. Volume
- b. Specific gravity
- c. Osmolality
- d. pH
- e. 17-hydroxycorticosteroids (free and conjugated)
- f. Electrolytes, the ions of sodium, potassium, calcium, chlorine, and phosphate
- g. Catecholamines
  - 1. Epinephrine
  - 2. Norepinephrine
- h. Nitrogenous compounds
  - 1. Total nitrogen
  - 2. Urea nitrogen
  - 3. Alpha amino acid nitrogen
  - 4. Creatine and creatinine
  - 5. Hydroxyproline
- i. Antidiuretic hormone
- j. Aldosterone (preflight and postflight only)

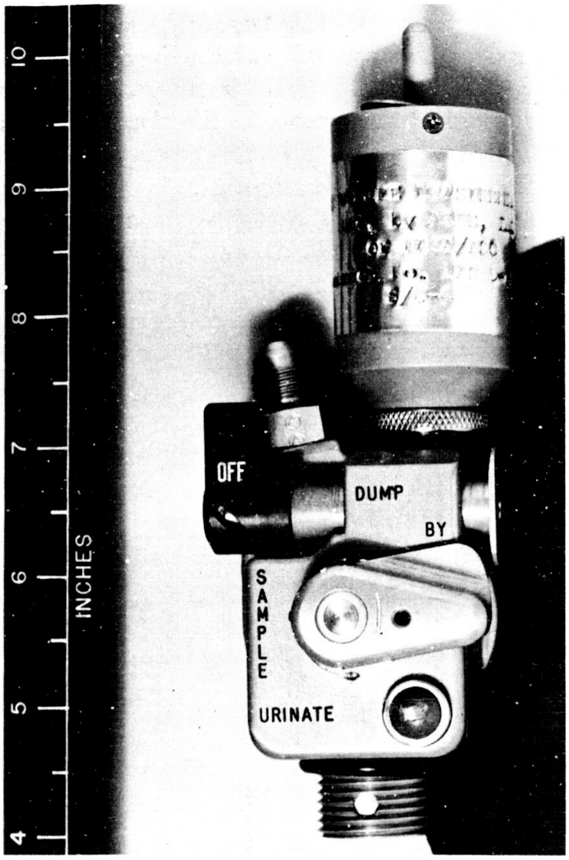

FIGURE 42-1.—Urine sampling and volume measuring system.

Results

Experiment M-5 was first scheduled for flight on Gemini VII. However, preflight and postflight plasma samples were obtained from the crewmen of Gemini IV through VI-A. No values out of the normal range were observed, nor were any trends evident in the Gemini IV through VI-A samples.

Analysis of the Gemini VII samples is still underway. The preflight and postflight plasma samples have been analyzed, and the results are presented in tables 42-I and 42-II. Electrophoretic patterns were normal. The values were all in the normal range, except for an anticipated increased 17-hydroxycorticosteroids in the first sample drawn following recovery. These returned to essentially preflight levels within 6 hours.

Hydroxyproline, which was determined because of its presence in collagen and its possible relationship to the decalcification process, did not change sufficiently to be interpreted in terms of bone density changes.

The drop in plasma uric acid immediately postflight must be examined further. A likely cause of the drop could be low purine intake. This possibility is being examined.

TABLE 42-I.—*Gemini VII Command Pilot Plasma Analysis*

[All dates 1965]

Components	Preflight		Postflight			
	Nov. 25	Dec. 2	Dec. 18 (1130 hr)	Dec. 18 (1820 hr)	Dec. 19	Dec. 21
Sodium, meq/liter	147	146	138	140	144	143
Potassium, meq/liter	4. 7	5. 4	4. 1	4. 7	4. 7	4. 9
Chlorine, meq/liter	103	103	100	102	103	106
Phosphate, mg, percent	3. 2	3. 7	4. 0	4. 2	3. 1	3. 6
Calcium, mg, percent	9. 0	9. 2	8. 6	9. 2	9. 0	9. 2
Urea nitrogen, mg, percent	19	16	16	20	25	18
Uric acid, mg, percent	6. 8	6. 6	4. 6	6. 0	5. 9	6. 0
Total protein, g, percent	7. 3	7. 4	6. 8	7. 6	7. 0	7. 1
Albumin, g, percent	4. 7	4. 9	4. 2	QNS	4. 5	4. 6
17-OH corticosteroids, micrograms per 100 ml	18. 8	----------	28. 3	16. 0	----------	----------
Hydroxyproline, micromilligrams per ml:						
Free	. 008	. 007	. 010	. 011	----------	----------
Bound	. 131	. 146	1. 51	. 185	----------	----------
Total	. 139	. 153	. 161	. 196	----------	----------

Plasma ADH was elevated enough for determination only in Pilot Lovell's first postflight plasma sample, although, as can be seen in tables 42–II and 42–IV, marked water retention was exhibited by both crewmembers immediately postflight. The water retention and the rapid weight gain after flight are consistent with the assumption that the weight lost during flight was the result of water loss.

Tables 42–III and 42–IV are comparisons of preflight and postflight 24-hour urine samples.

The retention of elecrolytes and water following reentry is consistent with the hypothesis that atrial and thoracic stretch receptors are of physiological importance in the change from a condition of 1 gravity to null gravity, and vice versa. A change from null gravity to an erect position in 1 gravity would result in a pooling of blood in the lower extremities and an apparent decrease in blood volume as experienced in

TABLE 42–II.—*Gemini VII Pilot Plasma Analysis*
[All dates 1965]

Components	Preflight		Postflight			
	Nov. 25	Dec. 2	Dec. 18 (1230 hr)	Dec. 18 (1800 hr)	Dec. 19	Dec. 21
Sodium, meq/liter	149	146	139	144	143	144
Potassium, meq/liter	4. 9	5. 1	4. 1	5. 0	5. 5	5. 0
Chlorine, meq/liter	104	103	97	101	100	104
Phosphate, mg, percent	3. 1	3. 3	3. 9	3. 9	3. 4	3. 4
Calcium, mg, percent	9. 6	9. 6	9. 2	9. 4	10. 0	9. 6
Urea nitrogen, mg, percent	23	22	21	28	27	24
Uric acid, mg, percent	6. 1	5. 8	3. 8	5. 3	5. 0	5. 0
Total protein, g, percent	7. 8	7. 8	7. 2	7. 9	8. 1	7. 2
Albumin, g, percent	4. 8	4. 7	4. 3	----------	----------	----------
17-OH corticosteroids, micrograms per 100 ml	13. 3	----------	26. 2	8. 9	----------	----------
Hydroxyproline, micromilligrams per ml:						
Free	. 017	. 010	. 010	. 005	----------	----------
Bound	. 161	. 167	. 182	. 187	----------	----------
Total	. 178	. 177	. 192	. 192	----------	----------

TABLE 42–III.—*Gemini VII Command Pilot Urinalysis*
[All dates 1965]

Components	Preflight		Postflight	
	Nov. 23	Dec. 1	Dec. 18	Dec. 21
Chlorine, meq	144	148	61	145
Calcium, mg	254	266	310	268
Uric acid, g	. 96	. 95	1. 20	1. 07
Total volume, ml	2920	3235	2160	3690
Sodium, meq	141	146	64	133
Potassium, meq	93. 0	79	73	106
Phosphate, g	1. 13	1. 16	1. 72	1. 12
17-hydroxycorticosteroids	6. 9	8. 76	13. 69	9. 28
Total nitrogen, g	19. 2	22. 6	30. 9	20. 5
Urea nitrogen, g	18. 1	18. 5	26. 6	18. 7
Hydroxyproline, mg	48. 74	37. 0	65. 4	39. 9
Creatinine, g	2. 11	2. 11	2. 86	1. 80

TABLE 42-IV.—*Gemini VII Pilot Urinalysis*

[All dates 1965]

Components	Preflight		Postflight	
	Nov. 23	Nov. 30	Dec. 18	Dec. 19
Chlorine, meq	177	139	40	45
Calcium, mg	182	126	115	207
Uric acid, g	.91	1.14	.45	.92
Total volume, ml	1912	1737	735	1405
Sodium, meq	162	145	35	58
Potassium, meq	76	93.0	44	58
Phosphate, g	1.12	1.27	.80	1.07
17-hydroxycorticosteroids	8.0	9.07	7.83	8.33
Total nitrogen, g	19.94	21.6	12.81	22.8
Urea nitrogen, g	17.19	17.06	11.75	21.51
Hydroxyproline, mg	39.39	43.1	31.8	37.4
Creatinine, g	2.27	2.25	1.75	2.16

the atria and thorax. This would produce an increased output of ADH and aldosterone, and a consequent water and electrolyte retention would occur. In null gravity, the increased volume of blood in the thorax and atria would produce a diuresis by a reversal of the above mechanism, and weight loss equivalent to the water loss would occur. Other mechanisms such as alterations or changes of water and electrolyte distributions in the various body compartments may also contribute to the observed results. Resolution of the mechanism still awaits results of the aldosterone and the inflight sample analyses.

Conclusions

Preflight and postflight urine and plasma samples from the Gemini VII crew were analyzed. Electrolyte and water retention observed immediately postflight are consistent with the assumption that the Gauer-Henry atrial reflex is responsive to a change from the weightless to the 1-gravity environment. Alterations in electrolyte and water distribution during flight may also be contributory.

Immediately postflight, plasma 17-hydroxycorticosteroid levels were elevated. Plasma uric acid was reduced. The cause of the reduction is unknown, but presumed to be dietary.

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43. EXPERIMENT M–6, BONE DEMINERALIZATION

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Summary

Experiment M–6 of this series of investigations on bone demineralization was designed to find the effect upon the human skeletal system of prolonged weightlessness and immobilization associated with confinement for a period of days in the Gemini spacecraft. This investigation was conducted both on the primary and backup crews of the 14-day Gemini VII mission, using the same method of radiographic bone densitometry as that employed in the Gemini IV and Gemini V studies. Radiographs were made preflight and postflight of the left foot in lateral projection and of the left hand in posterior-anterior projection of each crewman:

(1) At 10 days and at 3 days preflight and on the day of launch at Cape Kennedy.

(2) On the aircraft carrier U.S.S. *Wasp* immediately after recovery and again 24 hours later.

(3) At the Manned Spacecraft Center at 11 days and at 47 days following recovery.

In the laboratories of the Texas Woman's University Research Institute, sections of the os calcis, the talus, and hand phalanges 4–2 and 5–2 were evaluated for changes in skeletal mineralization. The method used was radiographic bone densitometry. The percentages of decrease in X-ray-equivalent calibration wedge mass found between radiographs made immediately preflight and postflight are shown in the table which follows.

Losses of this magnitude do not denote skeletal pathology, since all of the astronauts met or closely approached their preflight status before the respective studies closed.

The crewmen of Gemini VII, as seen in the table, experienced far lower losses in the os calcis than were found in the crews of Gemini IV and Gemini V. Losses in the finger were less than were found in the crewmen of these two previous flights, for whom bone densitometry measurements were made, although the differences were not so wide as in the case of the os calcis changes.

	Command pilot	Pilot
Conventional os calcis scanning section	−2. 91	−2. 84
Overall os calcis involving multiple traces over 60 percent of the bone	−2. 46	−2. 54
Section through the distal end of the talus	−7. 06	−4. 00
Multiple traces covering hand phalanx 4–2	−6. 55	−3. 82
Multiple traces covering hand phalanx 5–2	−6. 78	−7. 83
Greatest change in any section of the os calcis	−5. 17	−7. 66
Greatest change in hand phalanx 4–2	−9. 11	−8. 00
Greatest change in hand phalanx 5–2	−12. 07	−14. 86

The crewmen in the backup crew experienced only those changes in bone density found in healthy men pursuing their everyday activities.

The results of this study cannot be evaluated fully until further data are available, especially with respect to the difference in skeletal changes in the heel bone and the finger bone. Factors which probably contributed to the superior findings in the os calcis were these:

(1) The crewmembers of this mission ate a far higher proportion of the diet prepared for them than did those of Gemini IV and particularly of Gemini V.

(2) The crew had isometric and isotonic exercise for prespecified periods of time daily.

(3) An exerciser was used routinely.

(4) The crewmen slept for longer periods of time.

Methods

Densitometer Assembly

The instrumentation employed for the photometric evaluation of bone density from radiographs is a special analog computer consisting of a series of subassemblies, all designed to operate together as a completely integrated system. The basic units of the overall assembly, the theoretical aspects of the technique, and the history of the development of the method have been reported in references 1 through 4. Certain applications of the use of the bone densitometric employed in this study have been described in references 5 through 9.

Standard Radiographic Exposure Technique

Because different X-ray units were used at the three locations, the radiographs employed for densitometric measurements at different sites were standardized by three methods:

(1) An aluminum-alloy wedge exposed on the film adjacent to the bone was used.

(2) A roentgen meter to determine the calibrated kilovoltage which would produce identical beam qualities in each of the three X-ray units was used.

(3) A specially prepared phantom which was shaped like an os calcis and contained a standard quantity of ash enclosed in a tissue-simulating absorber to detect possible technique variations, was exposed at each testing site.

The X-ray machines were calibrated before each group of exposures by means of Victoreen roentgen meters in order to relate kilovoltage to X-ray transmittance in milliroentgens through a standard 2-millimeter aluminum filter under a specific X-ray intensity. Under the exposure conditions utilized, all units yielded a beam quality of 60 kilovolts, comparable with the central unit at the Texas Woman's University.

The X-ray film used in this investigation was Eastman Type AA film, which was exposed in cardboard holders.

Interpretation of the Term "X-Ray Absorbence"

As used in this report, the term "X-ray absorbence" by bone refers to the beam attenuation resulting from the hydroxyapatite and water-organic contents in their relative molecular weight concentrations, together with the overlying and underlying soft tissue. The results are reported in terms of wedge mass equivalency of the bone sites evaluated. Although changes in composition or thickness of the extrabone tissue could account for slight changes in total X-ray absorption, our tests have shown that, in the case of the os calcis, errors accountable to changes in soft tissue mass are slight.

Evaluation of Wedge Mass Equivalency in the Bones Evaluated

As noted, radiographs were made preflight and postflight of the left foot in lateral projection, and of the left hand in posterior-anterior projection of each crewman in the Gemini VII study. In previous investigations of bone mass changes before, during, and after orbital flight, the same radiographic exposures were made for the Gemini IV and the Gemini V crews.

In the Gemini IV study, the os calcis or heel bone was investigated, as was phalanx 5–2 of the left hand. In the Gemini V investigation, the same bones were examined, with the addition of phalanx 4–2, the distal end of the left radius, and the left talus. In the current study, the os calcis, the talus, and phalanges 5–2 and 4–2 of the left hand were included.

Central os calcis section.—This anatomical site was used in the M–6 Experiment in the Gemini IV and Gemini V flights and was repeated in the Gemini VII mission. The tracing path across the left os calcis in lateral projection runs diagonally between conspicuous posterior and anterior landmarks which, by superimposing successive radiographs, can be reproduced accurately in serial films of the same individual. This single path (1.3 millimeters in width) is known as the "conventional scan." (See fig. 43–1.)

Multiple parallel os calcis evaluations.—Approximately 60 percent of the total os calcis mass is evaluated in the parallel path system. After making the conventional scan, a series of parallel paths, 1.0 millimeter apart, were scanned, beginning 1 millimeter above the conventional path and continuing to the lowest

FIGURE 43–1.—Positive print of lateral foot radiograph showing location of the central section of the os calcis ("convention" section) which is evaluated for bone density changes, as well as the location of the section of the talus which is scanned.

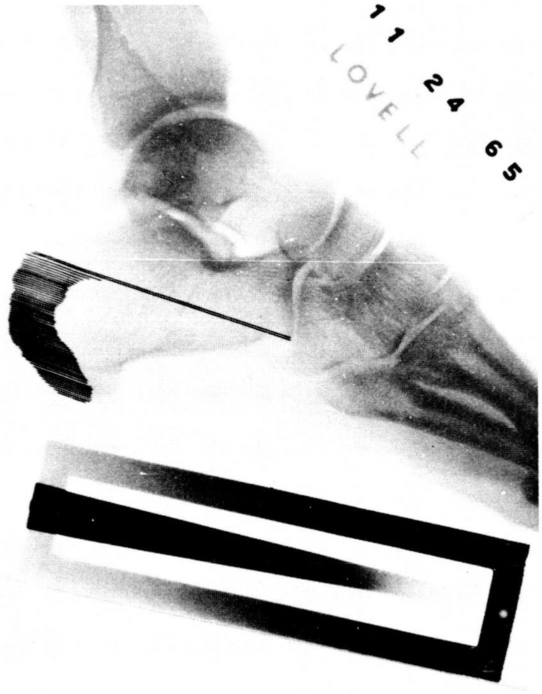

FIGURE 43–2.—Positive print of radiograph of os calcis showing location of the multiple sections which are evaluated. These scans are made entirely across the bone, parallel with the conventional section. They are 1 millimeter wide from the center of one scan to the center of the next scan, and hence they cover all of the 60 percent of this bone which is involved in this evaluation.

portion of the bone. The total number of paths scanned is, therefore, proportional to the size of the bone which, of course, has individual variations. For the command pilot, 38 paths were required to cover the os calcis portion examined, while 42 parallel scans were needed for the pilot. Figure 43–2 illustrates the alinement of parallel paths through the os calcis portion examined (every path is not shown in the illustration).

The talus.—A single scanning path was made through the talus of the left foot, originating at the interior surface and projecting anteriorly to the conspicuous landmark, shown in figure 43–1.

Sections of the phalanges 4–2 and 5–2.—The second phalanx of the fourth and the fifth fingers of the left hand was scanned by parallel cross-sectional paths 1 millimeter apart alined tangentially with the longitudinal axis and covering the entire bone area (fig. 43–3).

Results

X-Ray Absorption Changes in Central Os Calcis Section ("Conventional" Path)

The X-ray absorption values (in terms of calibration wedge equivalency) which were obtained from the central os calcis section throughout the Gemini VII mission are given in table 43–I and in figure 43–4. Based on a comparison of the calibration wedge equivalency of the immediate postflight radiograph with that made immediately before the launch, this central or "conventional" segment of the os calcis exhibited a change during the flight of only −2.91 percent for the command pilot and of −2.84 percent for the pilot.

It should be noted that there was an increase in bone mass of this anatomical site before the orbital flight and for 11 days after the flight in both crewmen. The postflight increase was more pronounced in the pilot. At the time the

FIGURE 43-3.—Positive print of hand radiograph in posterior-anterior projection, showing position of parallel traces on phalanges 5-2 and 4-2. The scans slightly overlap each other and cover the entire bone in each case.

last radiograph of the series was made, 70 days after the study had begun, the command pilot had leveled off in calibration wedge equivalency of this section of the os calcis at a value higher than any preflight result. The pilot, on the other hand, had a value in the last radiograph which was higher than that of any of his previous films except the next to the last measurement.

Table 43-II shows that the decrease in the overall sum of the sectional values obtained from the parallel scans made in the radiograph taken of the command pilot on the aircraft carrier immediately after his recovery was only −2.46 percent of the value made immediately before launch. The comparable change in values for the pilot was −2.54 percent. The table shows also that the greatest change during flight in bone mass in any of the multiple sections of the os calcis of the command pilot was −5.17 percent, while that of the pilot was −7.66 percent.

A graph of the sums of the calibration wedge equivalency values for the multiple os calcis sections for each of the preflight and postflight

TABLE 43-I.—*Bone Densitometric Values Obtained From Scanning the Central Section of the Os Calcis of Gemini VII Crewmen at Intervals Throughout the Preflight, Orbital Flight, and Postflight Periods*

[Based on integrator counts]

(a) Command pilot [a]

Film	Date	Integrator counts obtained during densitometric scanning of X-rays		
		Evaluation 1	Evaluation 2	Average, both evaluations
1	11/24/65	12 012	11 933	11 973
2	12/01/65	12 625	12 567	12 596
3	12/04/65	12 407	12 411	12 409
4	12/18/65	11 994	12 103	12 049
5	12/19/65	12 314	12 465	12 390
6	12/29/65	12 985	13 155	13 070
7	02/03/66	12 901	12 745	12 823

(b) Pilot [b]

Film	Date	Evaluation 1	Evaluation 2	Average, both evaluations
1	11/24/65	13 438	12 296	13 367
2	12/01/65	13 253	13 243	13 248
3	12/04/65	13 724	13 713	13 718. 5
4	12/18/65	13 306	13 351	13 328. 5
5	12/19/65	13 523	13 305	13 414
6	12/29/65	14 750	14 614	14 682
7	02/03/66	14 001	13 968	13 984

[a] Difference between immediate preflight and carrier postflight values=2.91 percent.

[b] Difference between immediate preflight and carrier postflight values=2.84 percent.

radiographs is shown for both crewmen in figure 43-5. A general similarity between the graph of the conventional trace and that of the overall os calcis sections for the serial radiographs of the pilot is seen in figures 43-4 and 43-5. The two graphs of the command pilot also bear some resemblance to each other.

Although there is some inconsistency in the magnitude of changes from section to section in the multiple scans of the os calcis, it is apparent that bone mass decreased somewhat more in the superior sections than in the inferior sections in both astronauts from the beginning to the close of the flight. The effect undoubtedly is attributable in major part to the greater pro-

TABLE 43-II.—*Comparison of Bone Changes During Flight in Total Os Calcis From Multiple Sections of the Os Calcis of the Crewmen in the Gemini VII Mission*

Position of tracing	Command pilot			Pilot		
	Integrator counts from densitometer 12/4/65 (average)	Integrator counts from densitometer 12/18/65 (average)	Percent change from 12/4 to 12/18/65	Integrator counts from densitometer 12/4/65 (average)	Integrator counts from densitometer 12/18/65 (average)	Percent change from 12/4 to 12/18/65
1 mm above	12 136	11 652	−3. 99	13 791	13 359	−3. 13
Conventional	12 409	12 049	−2. 91	13 719	13 329	−2. 84
1 mm below	11 468	11 124	−3. 00	12 592	12 239	−2. 81
2 mm below	11 229	10 836	−3. 50	11 937	11 689	−2. 08
3 mm below	10 988	10 648	−3. 09	11 838	11 550	−2. 43
4 mm below	10 956	10 628	−2. 99	11 928	11 465	−3. 88
5 mm below	10 726	10 418	−2. 87	11 613	11 306	−2. 64
6 mm below	10 460	10 142	−3. 04	11 314	11 186	−1. 13
7 mm below	10 332	9 934	−3. 85	11 214	11 013	−1. 79
8 mm below	10 238	9 709	−5. 17	11 122	10 898	−2. 01
9 mm below	9 978	9 597	−3. 82	10 799	10 591	−1. 93
10 mm below	9 690	9 415	−2. 84	10 630	10 275	−3. 34
11 mm below	9 630	9 248	−3. 97	10 394	10 046	−3. 35
12 mm below	9 294	8 964	−3. 55	10 126	9 890	−2. 33
13 mm below	8 968	8 690	−3. 10	9 790	9 562	−2. 33
14 mm below	8 694	8 568	−1. 45	9 536	9 276	−2. 73
15 mm below	8 557	8 381	−2. 06	9 280	9 186	−1. 01
16 mm below	8 090	7 996	−1. 53	9 056	8 866	−2. 10
17 mm below	7 795	7 578	−2. 78	8 979	8 586	−4. 38
18 mm below	7 570	7 451	−1. 57	8 960	8 274	−7. 66
19 mm below	7 470	7 328	−1. 90	8 222	7 892	−4. 01
20 mm below	7 403	7 268	−1. 82	7 452	7 432	−0. 27
21 mm below	7 295	7 209	−1. 18	7 331	7 290	−0. 56
22 mm below	7 221	7 184	−0. 51	7 241	7 168	−1. 01
23 mm be,ow	7 176	7 141	−0. 49	6 893	6 989	+1. 39
24 mm below	7 192	7 130	−0. 86	6 890	6 843	−0. 68
25 mm below	7 172	7 103	−0. 96	6 843	6 702	−2. 05
26 mm below	7 097	7 002	−1. 34	6 829	6 503	−4. 77
27 mm bleow	6 914	6 838	−1. 10	6 645	6 400	−3. 69
28 mm below	6 845	6 740	−1. 53	6 451	6 243	−3. 23
29 mm below	6 801	6 684	−1. 72	6 312	6 180	−2. 09
30 mm below	6 319	6 210	−1. 72	6 218	6 128	−1. 45
31 mm below	6 022	5 965	−0. 95	6 090	5 910	−2. 95
32 mm below	5 694	5 608	−1. 51	6 033	5 748	−4. 72
33 mm below	4 989	4 962	−0. 54	5 764	5 631	−2. 30
34 mm below	4 448	4 382	−1. 48	5 769	5 549	−3. 81
35 mm below	3 750	3 767	−1. 97	5 452	5 319	−2. 44
36 mm below	2 896	2 816	−2. 76	5 391	5 088	−5. 63
37 mm below	X	X	X	4 804	4 614	−3. 96
38 mm below	X	X	X	4 362	4 253	−2. 51
39 mm below	X	X	X	3 714	3 637	−2. 06
40 mm below	X	X	X	3 070	3 322	+8. 22
Total	311 912	304 244	X	352 394	343 427	X
Mean change	X	X	−2. 46	X	X	−2. 54

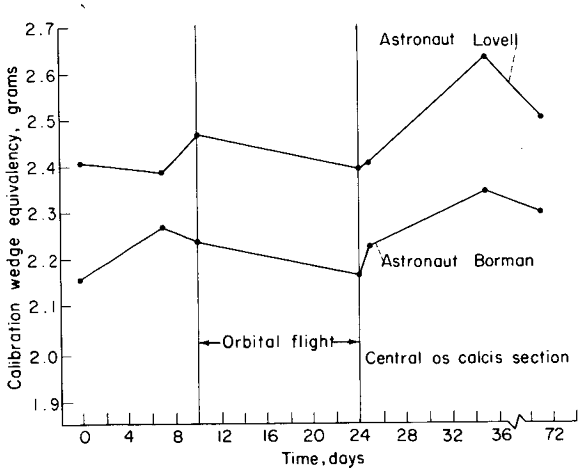

FIGURE 43-4.—Graph of the calibration wedge mass equivalency data on the "conventional" os calcis section which were evaluated for the Gemini VII flight crew.

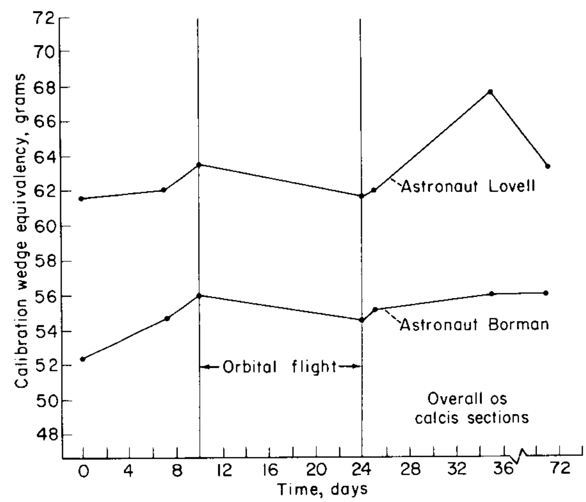

FIGURE 43-5.—Graph of the calibration wedge mass equivalency data on the total parallel sections of the os calcis which were evaluated for the Gemini VII flight crew.

portion of trabecular or cancelous tissue in the central and superior parts of this bone, with greater proportions of compact or cortical tissue in the distal sections.

Changes in the Talus

The calibration wedge mass equivalency at the talus scanning site obtained from the radiograph made immediately postflight was 7.06 percent lower than the final preflight value for the command pilot and 4.00 percent lower for the pilot. Prior to the flight the talus value first increased and then decreased for the command pilot, with a value at the time of launch which was slightly higher than the initial preflight level. The pilot showed a slight decrease in this site preflight. Both crewmen exhibited a marked increase for 11 days, after which there was a slight decrease, but with final values not markedly different from the initial levels. (See fig. 43-6.)

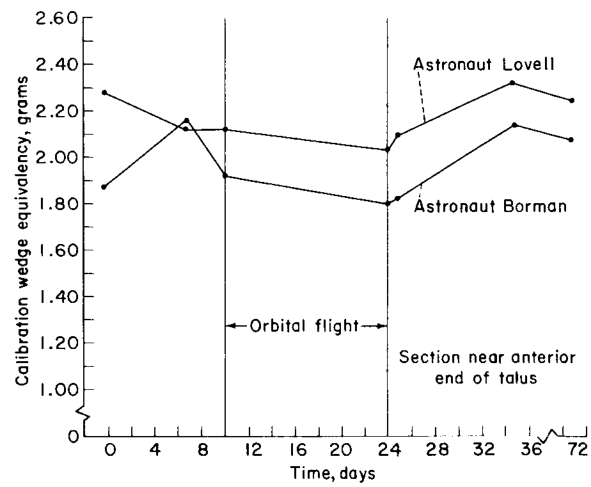

FIGURE 43-6.—Graph of the calibration wedge mass equivalency data on the section of the talus which were evaluated for the Gemini VII flight crew.

Bone Mass Changes in Hand Phalanges 4-2 and 5-2

As in the case of the os calcis, multiple parallel scans were made across hand phalanges 4-2 and 5-2, with distances of 1 millimeter from the center of one scan to that of the next scan. In this matter, the entire area of each phalanx was evaluated in posterior-anterior projection. (See fig. 43-3 for the positions of the sections scanned.)

Phalanx 4-2.—From the time the radiograph was made immediately before launch until the one which was made 14 days later, immediately after recovery on the carrier, the command pilot sustained an overall change of −6.55 percent in the 25 scans required to cover phalanx 4-2. The change in this anatomical site for the pilot during the same period was −3.82 percent, with 25 scans required to cover this bone. The greatest change in any section of phalanx 4-2 was −9.11 percent for the command pilot and −8.00 percent for the pilot.

Figure 43–7 consists of graphs of the calibration wedge equivalency values for hand phalanges 4–2 for the serial radiographs of the two Gemini VII crewmen. The graph of the command pilot shows that the value for phalanx 4–2 was higher at the beginning of the orbital flight than the first preflight value, with a decline by the close of the flight. This was followed by a gradual increase after the flight.

The graph for phalanx 4–2 for the pilot shows a marked increase in X-ray absorbence during the first 7 preflight days, followed by a decrease during the last 4 preflight days. Following the decrease during the flight, there was a sharp and then a gradual postflight increase.

Phalanx 5–2.—From the beginning to the close of the orbital flight, the command pilot sustained an overall change of −6.78 percent in the 18 parallel sections of phalanx 5–2. In the 17 scans required to cover hand phalanx 5–2 of the pilot, an overall change of −7.83 percent in bone mass was found. The greatest change in this bone for the command pilot was −12.07 percent, and for the pilot, −14.86 percent. As in the case of the crewmen of Gemini V, the losses in phalanx 5–2 tended to be greater than that of phalanx 4–2.

Figure 43–8 shows graphically the overall changes in the bone mass of the sections of the hand phalanges of the crewmen throughout the study. The values for the command pilot did not experience as marked preflight and postflight changes as those for the pilot. The values for the pilot took a sharp upward trend during the first 7 days of the preflight period, followed by a decline during the next 3 days. The last preflight value, however, was higher than the initial level. After the decline in X-ray mass equivalency shown during the flight, there was a sharp increase during the first 24 hours after the flight, with a continued moderate increase through the next 11 days, followed by a final decrease. However, the value 47 days after the flight was higher than the initial value found when the study began.

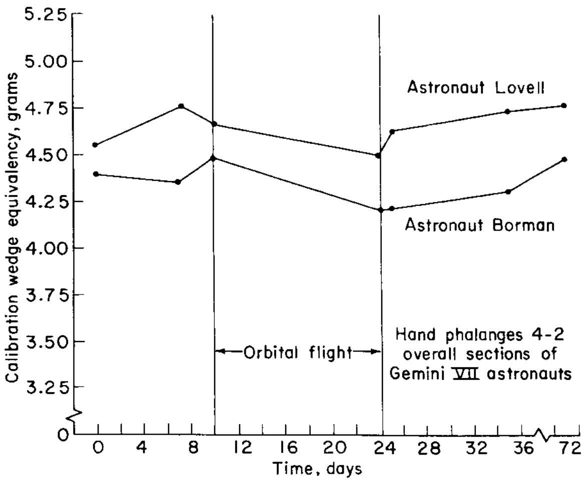

FIGURE 43–7.—Graph of the calibration wedge mass equivalency data on hand phalanx 4–2 for the Gemini VII flight crew.

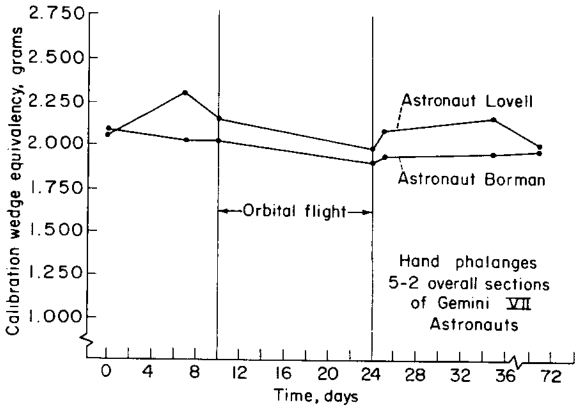

FIGURE 43–8.—Graph of the calibration wedge mass equivalency data on hand phalanx 5–2 for the Gemini VII flight crew.

Discussion

Comparison of Bone Density Changes in Crewmen of Gemini IV, Gemini V, and Gemini VII During Space Flight

It is interesting to note how the crewmembers of Gemini IV, Gemini V, and Gemini VII have compared with each other as to skeletal changes in three major anatomical sites with respect to changes in skeletal density during space flight. The bone mass changes in table 43–III (in terms of calibration wedge equivalency) have been found for the command pilot and the pilot in the "conventional" os calcis section, in the combined sections covering 60 percent of the os calcis, and in hand phalanges 5–2 and 4–2, both for the command pilot and the pilot for the three orbital flights.

TABLE 43-III.—*Comparison of Bone Density Changes in Crewmen of Gemini IV, Gemini V, and Gemini VII During Space Flight*

Position of anatomical site evaluated	Change in bone mass,[a] percent	
	Command pilot	Pilot
Conventional os calcis scan:		
Gemini IV	−7.80	−10.27
Gemini V	−15.10	−8.90
Gemini VII	−2.91	−2.84
Multiple os calcis scans:		
Gemini IV	−6.82	−9.25
Gemini V	−10.31	−8.90
Gemini VII	−2.46	−2.54
Hand phalanx 5–2 scans:		
Gemini IV	−11.85	−6.24
Gemini V	−23.20	−16.97
Gemini VII	−6.78	−7.83
Hand phalanx 4–2 scans:		
Gemini IV	([b])	([b])
Gemini V	−9.98	−11.37
Gemini VII	−6.55	−3.82

[a] Based on X-ray absorbency of calibration wedge.
[b] Not done on this flight.

Comparison of Bone Density Changes in the Gemini VII Crew With Bedrest Subjects on Similar Diets for 14 Days

On the basis of the tentative evaluation of food intake based on the residue removed from the spacecraft postflight, it is estimated that 1.00 gram of calcium was consumed by the Gemini VII crewmen during their orbital flight. On this basis, the os calcis and hand phalanx 5–2 were compared with subjects at supine bedrest for 14 days in the Texas Woman's University (TWU) bedrest units. Bedrest men on comparable diets lost slightly more in the os calcis and considerably less in phalanx 5–2 than did the crewmen on this mission, as seen by the data in table 43–IV.

Comparison of Bone Density Changes in Crew and in Backup Crew of Gemini VII

The backup crew of Gemini VII, which included Edward White and Michael Collins, had four radiographs made in connection with this mission on the following dates: November 24, 1965; December 1, 1965; January 3, 1966; and February 3, 1966.

TABLE 43–IV.—*Comparison of Bone Density Changes in the Gemini VII Crew With Bedrest Subjects on Similar Diets for 14 Days*

	Gemini VII crew		TWU bedrest subjects
	Command pilot	Pilot	
Mean calcium daily intake (estimated), grams	1.00	1.00	(1) 0.931 (2) 1.021 (3) 1.034 (4) 1.020 (5) 0.930
Change in conventional section of os calcis in bone mass (calibration wedge equivalency), percent	−2.91	−2.84	(1) −3.46 (2) −3.56 (3) −5.79 (4) −5.11 (5) −5.86
Change in bone mass of hand phalanx 5–2, percent	−6.78	−7.83	(1) −1.57 (2) −1.00 (3) −0.44 (4) −0.96 (5) −1.27

The spread from the highest to the lowest X-ray absorbency value in the os calcis for White was 2.5 percent covering a period of 3 months and 10 days. The spread for Collins was 3.2 percent over the same period. On comparable dates, not involving any aspect of the orbital flight, the spread in os calcis absorbency values was 6.6 percent for Frank Borman and 9.8 percent for James Lovell. This indicates that the maximum spread was less in the backup crew than in the flight crew.

No exact dietary records for the backup crew were kept during this period.

Conclusion

The Gemini VII flight crew activities were calculated in part to support a metabolic study. Hence, tasks not related to this objective were minimized, with the result that time could be spent on isometric and isotonic exercise, on exercise with a mechanical device, and on sleep. Also there was more time available for eating. By consuming a larger proportion of the diet provided for them, the crewmen not only increased the amount of calcium which they consumed, but also the quantity of total energy and of other essential nutrients. Furthermore, various foods supplied for this mission were provided with supplementary calcium.

The results of the study show decreased loss of X-ray density of the largest bone in the foot, but with far less dramatic results obtained with the hand. This would indicate to the authors the need for further attention to the development of exercise routines which would involve the hands and fingers.

Without reducing the emphasis on dietary calcium, a probable need also exists for further research in which other nutrients known to be related to skeletal status would serve as variables.

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44. EXPERIMENT M–7, CALCIUM AND NITROGEN BALANCE

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Introduction

The primary objective of Experiment M–7 was to obtain data on the effects of space flight of up to 14 days' duration on two of the largest metabolically active tissue masses of the human body, the bones and muscles, and thus on the functional integrity of the skeletal and muscular systems.

From prior ground-based studies on the effects of bedrest or immobilization on normal human subjects, it has been predicted that the confinement of the Gemini space vehicle, in association with the lack of physical stress and strain on muscles and bones due to weightlessness, would result in substantial losses of calcium, nitrogen, and related elements. Bedrest studies have shown, for example, that in 2 weeks of immobile rest, the amount of calcium excreted in the urine was doubled, and, over longer periods, substantial negative balances or losses of calcium, nitrogen, and other elements occurred. Significant losses in a space flight continuing over a period of several weeks theoretically could lead to a serious weakness of the bones and muscles.

By use of the metabolic-balance method, which involves precise control of the dietary intake and the collection and analysis of all excreta, it is possible to obtain a quantitative determination of the extent of change in the principal inorganic constituents of these systems, the degree of loss thereof being generally proportional to the degree of deterioration in function. Biomedical data on this problem using this quantitative method have not been obtained on previous American or Russian space flights. X-ray films taken before and after the Gemini IV and V flights indicated changes in the equivalent aluminum density of two bones, the heel, and a finger, but these findings cannot yet be equated with calcium losses from the whole skeleton.

Realistic consideration of this metabolic-balance study indicates that it was not, in any true sense, an experiment on the effects of weightlessness on body metabolism, but was rather an observation of biochemical changes occurring as a result of several complex, interrelated influences—principally weightlessness, confinement, moderate physical movement, slight hyperoxia, and low atmospheric pressure.

Because of the tremendous number of analyses to be carried out, specific analytical results are not available at the time of this preliminary report. However, an account can be given of the detailed and intricate protocol and of the generally successful accomplishment of a very difficult study.

Procedure

The general plan of a metabolic study requires continuous procurement of data during a control phase at normal activity on earth for as long a time as is feasible before flight. Complete inflight data and a postflight control phase are also required. In view of the numerous other requirements of the Gemini VII mission, the preflight control phase was limited to 9 days, beginning 14 days before launch. The postflight control phase was even more brief, lasting only 4 days.

The method employed in obtaining quantitative information on a metabolic system requires complete and continuous data on the dietary intake of each constituent under study and continuous collection of all urine and stool specimens before, during, and after the flight. Since under certain circumstances the skin may be an important avenue of excretion of various elements, particularly calcium, perspiration also had to be collected during representative periods before and after flight, and continuously during flight.

Dietary Intake

Not only must the content and composition of all food and water intakes be known, but, insofar as possible, the amounts must be kept as constant as possible. To the extent that the intake of each constituent can be kept constant from day to day and from control-to-experimental phase, the changes in the amounts of these constituents excreted can be safely attributed to the influences of the experiment itself—in this case, the flight. If the intake is not kept relatively constant, then changes in excretory levels will be difficult or impossible to interpret because of their change with the change in intake.

In this particular study, what was essentially necessary for diet control during the preflight and postflight control phases was establishment of metabolic kitchen facilities and techniques for food preparation, weighing, storage, cooking, and serving in the kitchen of the astronauts quarters in the Manned Space Operations Building at Cape Kennedy.

Standard metabolic-study techniques were used for minimizing variations from day to day in the composition of individual food items. All food items were weighed to a precision of 0.1 gram, and liquids were measured to less than 2 milliliters. A sample menu is shown in table 44–I. Variety was made possible by rotation of three daily menus. Table 44–II lists the actual composition (from diet tables) of the nitrogen and calcium consumed day by day during the preflight control phase. The extent to which the values varied from day to day, particularly during the first several days, is due to the fact that no time was available for a precontrol trial of the diets with the four crewmen in the control phase of the study, and also because there was need for adjustments during the study to fit the crewmembers' needs with respect to total calories and bulk. The extent to which the values remained constant from day to day was attributable not only to dietetic skill in menu planning under difficult circumstances, but also to the rapid understanding by the crewmembers of the principles and requirements of constant dietary intake in a metabolic study. The nearly constant diet control was also attempted for phosphorus, magnesium, potassium, sodium, fat, carbohydrate, and total calories.

TABLE 44–I.—*Menu 2 (Sample)*

Meal	Food [a]	Weight, grams
Breakfast	Eggs (2)	100
	Canadian bacon	50
	Bread (toast)	50
	Butter	70
	Puffed rice	20
	Grape jelly	25
	Orange juice	175
	Milk	340
	Coffee or tea	--------
Lunch	Baked ham	120
	Mashed potatoes	150
	Frozen baby lima beans	95
	Hot rolls	50
	Peach halves, canned	100
	Coffee or tea	--------
Dinner	Beef tenderloin steak	180
	Onions, Bermuda	30
	Baked Idaho potatoes	150
	Carrots, canned or frozen	100
	Hot rolls	50
	Lettuce	30
	Tomatoes, fresh sliced	75
	Mayonnaise	10
	Apricot halves	100
	Coffee or tea	--------
	Vanilla ice cream	150

[a] Salt: as desired; sugar: 10 grams.

An important point in overall dietary intake planning was the necessity to impose some degree of constancy of intake, particularly with respect to calcium, long before the control phase actually began, so that the excretory values during this relatively brief phase would not be merely a reflection of adjustment to a change in the customary level of intake. To provide this necessary element of control, the four crewmembers drank two glasses of milk daily for 5 months prior to the beginning of the study.

During the flight phase Edward White and Michael Collins dropped out of the study, while Frank Borman and James Lovell in the Gemini vehicle consumed the prepackaged, solid, bite-sized foods and the freeze-dried foods reconstituted with water which had been prepared on contract for the Crew Systems Division of NASA. Although the food items taken on

TABLE 44–II.—*Experiment M–7, Nitrogen and Calcium Dietary Intake*

[All data in grams per 24 hours]

(a) Preflight control days

Crewman	Element	12	11	10	9	8	7	6	5	4	Mean	Standard deviation
Frank Borman	Nitrogen	27.36	27.54	29.84	25.17	31.02	29.65	23.39	30.50	29.90	28.26	
	Calcium	.973	.982	.986	1.002	1.002	.986	1.000	.980	.999	.990	±0.011
James Lovell	Nitrogen	23.58	23.87	26.26	23.57	26.70	25.87	23.58	26.70	25.87	25.11	
	Calcium	.984	1.010	.992	1.001	.999	.958	.991	1.000	.958	.988	±.109
Edward White	Nitrogen	22.05	24.93	27.82	27.76	32.34	27.62	27.68	30.22	27.09	27.50	
	Calcium	.892	1.006	.985	.980	1.047	1.007	.972	.924	.977	.977	±.046
Michael Collins	Nitrogen	24.50	24.67	27.84	27.86	31.30	30.05	27.97	31.30	30.21	28.41	
	Calcium	.998	.998	.979	.981	1.001	.967	.997	1.001	1.003	.992	±.012

(b) Postflight control days

Crewman	Element	1	2	3	4	Mean	Standard deviation
Frank Borman	Nitrogen	24.54	31.01	22.00	23.66	25.30	
	Calcium	.941	1.045	.871	1.055	.978	±0.088
James Lovell	Nitrogen	22.42	26.08	26.45	24.00	24.74	
	Calcium	.953	.993	.901	1.071	.980	±.072

Gemini VII were generally similar to those on prior flights, certain foods—notably fruit drinks and puddings—were supplemented with calcium lactate in order to provide as closely as possible a mineral intake of the same level as was taken during the control phase. In addition, the flight food was packaged in specific meal-packs to be taken in a definite time sequence so that the day-to-day dietary intake would also remain as constant as possible under these difficult-to-control circumstances. For reasons which are not presently known, the crewmen did not follow the prescribed meal sequence; thus, when the inflight intake data from a combination of log information and diet analyses have been assembled, there will certainly be day-to-day fluctuations. It is possible that calcium fluctuations will turn out to be modest in view of the number of calcium-supplemented food items in nearly all the meals. In any case, since the crewmen consumed the various food items fairly consistently almost in their entirety, the intake of calcium and nitrogen for the block flight period will be closely similar to that of the control phase.

During the first day of the 4-day postflight control phase, the crewmen (onboard the carrier) consumed foods previously prepared at Cape Kennedy. They returned to their quarters at the Cape for the remaining 3 days, and ate the same diet as they did during the preflight control phase.

Collection of Specimens

Bottles, a commode adaptation of toilet seats, and a small refrigerator setup were used in the astronauts' quarters for the collection of all urine and stool specimens during the preflight and postflight control phases. This setup was similar to that used in hospital metabolic research wards. All specimens were labeled by the crewmembers with the initial of their last name, the date, and the time of passage. They were placed immediately in the refrigerator. Specimen collection stations were also set up at the Gemini Mission Simulator and at two other locations at Cape Kennedy. Specimens were picked up by the staff at regular intervals and returned to a laboratory in the Manned Space Operations Building where they were prepared for shipment to Cornell University for analysis.

On 2 days prior to the flight and on 2 days after the flight, perspiration collections were made separately for each crewman. The somewhat involved procedure included an initial washing of the subject's body with distilled water, the wearing of cotton long underwear for 24 hours, and a second body washing. The underwear was rinsed, and the water from this rinse, along with the water from the body washes, was collected and analyzed for minerals and electrolytes.

For the flight phase, collection of perspiration and its analysis were accomplished using the cotton undergarments, which were worn throughout the flight, and the distilled water from the skin wash performed shortly after arrival on the carrier.

Collection of urine and stool specimens during flight was a complex procedure in the weightless state, and it required development of special equipment. It was essential to have stool-specimen collection made with relative ease to assure that fecal material would be well formed. Apparently helpful in this process was the moderately-low-residue character of the metabolic diet which was continued until the morning of the launch. Stool specimens were wrapped securely (with preservative added) in plastic collection devices labeled with the crewman's name and the time. They were stowed in the locker for specimens.

Development of the urine collection device involved a great deal of effort and ingenuity, not merely because of the problem of collecting fluids in the weightless state but also because of lack of space for storage of the total volume of all specimens. It was necessary to devise a method of determining the volume of each voided specimen and then taking an aliquot for storage for later analysis. Several systems were tried, but the one used involved the introduction of a tracer quantity of tritium into an 800-milliliter plastic collection bag which received the urine voiding. After the tracer was well mixed with the full voiding, part was transferred to a 75-milliliter bag for storage and later analysis and the remainder was expelled from the spacecraft.

In actual experience the urine collection device worked well but with some leakage inconvenience at the point of connection between the subject and the device. The more serious problems were as follows:

(1) Since there was considerable concern about adequate stowage space and about whether the volume of each specimen saved could be controlled by the astronauts, one of the astronauts, during the early part of the flight, provided aliquots which were much too small.

(2) One sample bag broke.

(3) Four of the specimen bags were not labeled with either the crewman's name or the time.

Aside from the deficiencies noted above, most of the urine specimens were properly collected and labeled.

This brief summary barely hints at the considerable problems in planning and the tremendous detail involved in specimen collection, labeling, recording, and shipment. A 10-day full runthrough of the methods was conducted in September 1965 at the 6570th Aerospace Medical Research Laboratories, Wright-Patterson Air Force Base, Ohio. Members of the group involved in that exercise came to Cape Kennedy in November and December to assist in this study.

Analytical Problem

The principal reason that results are not yet available lies in the magnitude of the analytical problem in this study. Analyses are being done on specimens from a total of 76 man-days of study, involving approximately 300 urine specimens, 60 stool specimens, 14 perspiration samples, and an indefinite but large number of diet samples. Each of these specimens is being analyzed for nitrogen, calcium, phosphorus, magnesium, sodium, and potassium. In addition, the urine specimens are being analyzed for creatine, creatinine, sulfate, chloride, and hydroxyproline. Stool specimens are also being analyzed for fat. Added to the number of analyses to be accomplished and correlated, the problem is further complicated in the inflight phase by the irregular time periods from one voiding to the next. Because of this, some difficulty is anticipated in relating the analytical values to a regular 24-hour pattern.

Relationship to Other Experiments

A close working relationship was necessary between Experiments M–7 and M–5, the analysis of body fluids. Blood specimens were collected before and after flight as part of the M–5 protocol for serum calcium, phosphorus, and alkaline phosphatase. In bedrest studies involving extreme immobilization over several weeks, elevations in serum calcium have been noted. M–5 analyses of urine for electrolytes, corticosteroids, and catecholamines require urine collected in both Experiments M–5 and M–7, and aliquots of the urine specimens now at Cornell University are being sent to the Manned Spacecraft Center for the planned M–5 analyses.

Great interest will be focused on the correlation between the degree of apparent mineral loss from the os calcis and metacarpal bones in the M–6 Experiment and the total mineral loss from the whole skeleton, which will be indicated from the balance study. Since the skeleton varies considerably from bone to bone in the relative availability of calcium, the correlation between the two methods, if possible, will not be simple.

Interpretation and Significance of the Study

As indicated initially, during the space flight several influences in addition to weightlessness were present which could have had varying and conflicting influences on calcium metabolism. These included confinement, moderate physical movement, slight hyperoxia, and low atmospheric pressure. In interpreting the results, it may be necessary to deal with the possible interfering effects of the bungee exercise procedure (M–3 Experiment) for both astronauts and the M–1 alternating pneumatic cuff experiment for Lovell. The need is evident for careful selection of studies in future flights to assure as clear-cut answers as possible. In any case, there is a very important need for further ground-based studies to enable sorting out the kind and degree of effect of a number of the possible influences currently imposed on this experiment by various engineering constraints, such as low atmospheric pressure, high oxygen tension, confinement, and exercise. Regardless of these considerations, if significant changes in any of the various aspects of metabolism are found, they will serve as a basis for predicting what derangements of more serious degree are likely to occur on longer flights or in an orbiting laboratory, if well substantiated, effective protective procedures are not developed.

Conclusion

This preliminary report has attempted to describe the difficult and detailed planning, the rather prodigious management effort required by both the investigators and the NASA staff, and the tremendous and perceptive cooperation on the part of the crewmembers and their office that are required for completion of the calcium and nitrogen balance study. Considering the complexity of the study, it was conducted exceptionally well.

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45. EXPERIMENT M-8, INFLIGHT SLEEP ANALYSIS

By PETER KELLOWAY, Ph. D., *Chief, Neurophysics, Methodist Hospital, Texas Medical Center, Houston, Tex.*

Introduction

The necessity of monitoring the cardiovascular function during space flight has been recognized and implemented since the inception of the manned space-flight program. More recently, attention has been directed to the possibility of monitoring the brain function during space flight.

A cooperative research program at the Baylor University College of Medicine, at the University of California at Los Angeles Medical School, and at the Manned Spacecraft Center has been directed to the following practical and scientific questions:

(1) Can the electrical activity of the brain, as it is revealed in the electroencephalogram (EEG) recorded from the scalp, provide important and useful information concerning such factors as the sleep-wakefulness cycle, degree of alertness, and readiness to perform?

(2) Is it feasible and practical to record the EEG (brain waves), which is an electrical signal measured in microvolts, under the unique and difficult conditions which prevail during space flight?

The special conditions which exist during space flight consist of such factors as—

(a) Possible electrical interference from the many electrical devices near each other aboard the spacecraft.

(b) The necessity for recording during the routine activity of the subjects with attendant artifacts produced by muscle action, movements, sweating, skin resistance changes, and so forth.

(c) The requirement for miniaturization of the necessary instrumentation to a point sufficiently small and light in weight to justify its existence as part of the payload of the space vehicle.

(d) Provision of scalp electrodes and a method of attachment which would permit prolonged artifact-free recordings without producing significant discomfort or irritation to the scalp. (In clinical practice, electrodes are generally not required to remain in place for longer than 1.5 hours.)

(3) What are the minimal number of brain areas and, hence, of channels of electrical data which are necessary to provide EEG information adequate to identify and differentiate all levels of sleep and wakefulness?

(4) Can computer or other forms of automatic analysis be effectively employed to analyze the EEG data in order to yield the required information, thus avoiding the necessity of having EEG experts constantly at hand to read and analyze the records?

(5) Finally, can highly sophisticated techniques of computer analysis reveal important correlations between EEG activity and higher brain functions having to do with such states as vigilance and attention which are not evident on simple visual analysis of the EEG record?

These are the practical problems which are being studied. In addition, the following scientific questions are under investigation:

(1) Possible influences of weightlessness, and so forth, upon brain function and particularly upon the sleep-wakefulness cycle as evidenced by EEG changes.

(2) The application of computer analysis techniques to the analysis of the EEG under various controlled conditions; for example, sensory stimulation, heightened affective states, mental computation, as well as other similar factors.

Objectives

A major part of this research program has already been completed, but the present report is concerned only with the preflight and inflight data obtained in carrying out the specific experiment, Inflight Sleep Analysis, in connection with the Gemini VII flight.

The primary purpose of this experiment was to obtain objective and precise information concerning the number, duration, and depth of sleep periods of one of the members of the crew (Command Pilot Borman).

The importance of precise information concerning the sleep (hence, rest) of the crew, especially during prolonged flights, is obvious. The electroencephalogram is capable of providing this information, as the electrical activity of the brain undergoes clearly established and consistent variations with different levels of sleep. Using the EEG, it is possible to distinguish four levels of sleep ranging from drifting or drowsiness to profound sleep, and a special state sometimes called paradoxical sleep or the rapid eye movement stage of sleep, which is believed by many investigators to be important for the psychoaffective well-being of the individual.

Approach and Technique

Baseline Data

Baseline, multichannel EEG, and other psychophysiological data were recorded on Borman and the backup command pilot, White, at the Laboratory of Space Neurobiology at the Methodist Hospital during all stages of sleep and during the waking state. These recordings were used as a baseline for comparison with recordings made in the altitude chamber runs at St. Louis and finally with the inflight records.

Electrodes and Recording System

Preliminary studies of 200 control subjects, and specifically of White's and Borman's preflight EEG's, had shown that all of these stages of sleep could be differentiated and identified in records obtained from a single pair of electrodes placed on the scalp—one in the central, and one in the occipital region. It was also found that if these electrodes were placed in the midline of the head, the least possible artifact from muscle activity was attained. As weight and space limitations permitted only one more EEG recording channel, what was essentially a duplicate of the first electrode pair was used but displaced a few centimeters to the left of the midline. Such electrode placements reveal essentially the same information as the midline pair, but this choice was made (rather than obtaining data from another brain area) to provide for the possibility that one or more of the electrodes of one pair might be dislodged or become defective.

The recording system consisted of two miniature transistorized amplifiers, carried by the astronaut in pockets of his underwear, and a small magnetic tape recorder inside the spacecraft. The tape recorder, running at a very slow speed, was capable of recording 100 hours of data continuously.

Preflight Tests

Preliminary tests of the electrode system, amplifiers, and tape recorder under flight conditions were made first in the altitude chamber at McDonnell Aircraft Corp. and subsequently at the Manned Spacecraft Center.

Another dry-run test was made at Cape Kennedy the day before the flight, and recordings were made at the launch pad prior to lift-off.

All of these preflight runs yielded good recordings, clean of all artifact except that engendered by the movements of the subjects themselves.

Inflight Test

Recording of the EEG was to be continuous throughout the first 4 days of the Gemini VII flight. During these 4 days, the command pilot was to keep his helmet on unless marked discomfort or other factors necessitated its removal. The electrode system was, therefore, designed for a helmet-on arrangement.

Results

The events (as determined from the medical recorder data) from 15 minutes before lift-off to the time one of the second electrode pair was dislodged are shown graphically in figure 45–1. A total of 54 hours and 43 minutes of interpretable EEG data was obtained. Most of these data were of excellent quality from the viewpoint of visual interpretation.

EEG channel 1 became noisy after 25 hours and 50 minutes of flight (indicated by point B), and no interpretable data appeared in this channel after 28 hours and 50 minutes (indicated by point C). EEG channel 2 gave good, artifact-free data up to 43 hours and 55 minutes (point D), at which time it became intermittently noisy. No interpretable data were recorded after 54 hours and 28 minutes of flight (point E), at which time the electrodes for this channel were inadvertently dislodged. The sleep periods (shaded areas) will be discussed later in detail. The meals are indicated in the illustration because they represent periods of temporary interruption of the interpretability of the EEG data due to muscle and movement artifacts produced by rhythmic chewing (fig. 45–2).

As indicated in figure 45–1, 8 hours after lift-off, the command pilot closed his eyes and remained quiet for almost 2 hours—8:12:00 to 10:19:00 ground elapsed time (g.e.t.)—without showing signs of drowsiness or sleep. A portion of the record during this period is shown in figure 45–2.

Sleep is very easy to detect in the EEG records. Figures 45–3 and 45–4 show the distinc-

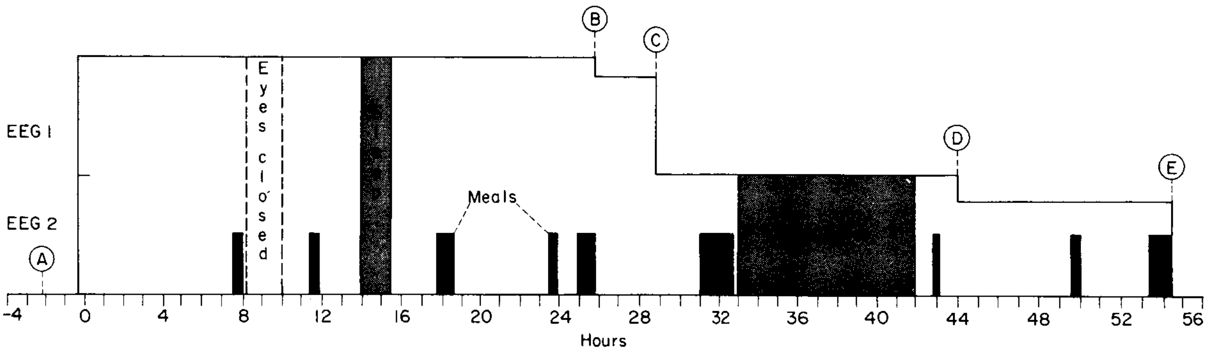

FIGURE 45–1.—EEG data flow.

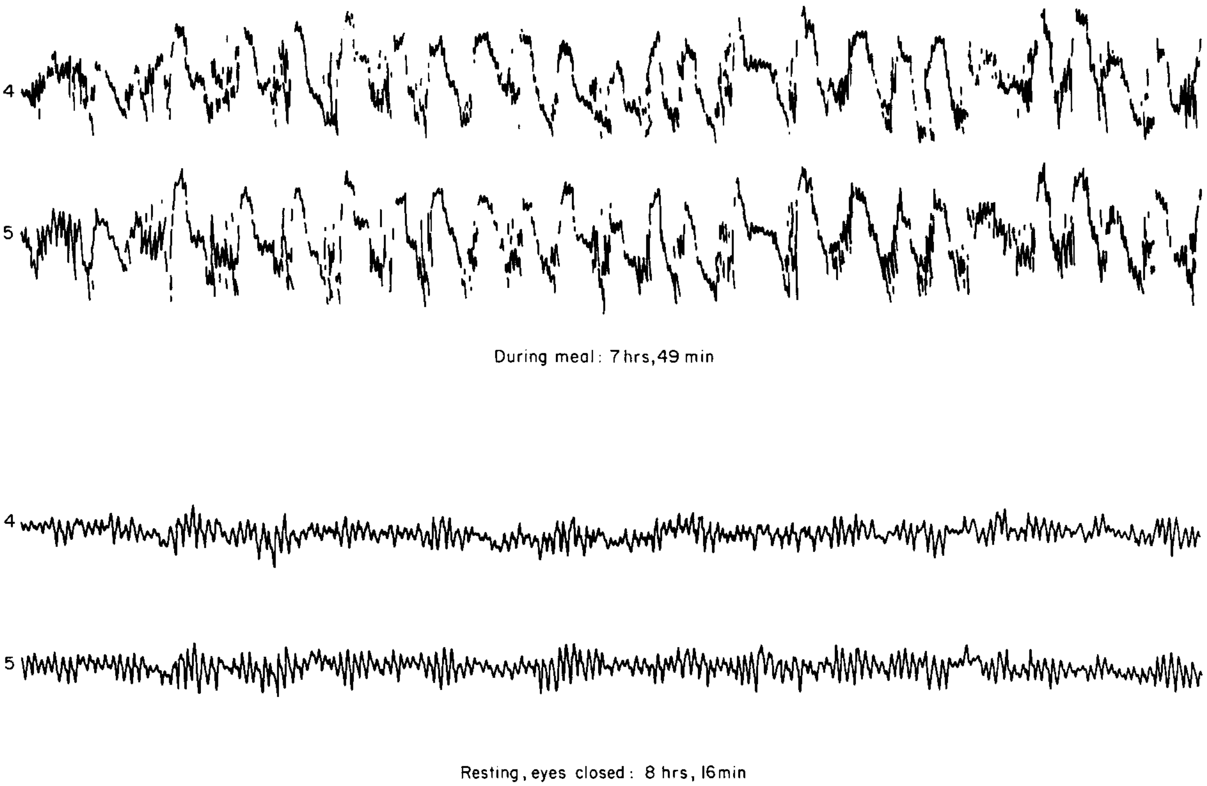

FIGURE 45–2.—EEG recordings taken during rhythmic chewing (upper) and during eyes-closed resting condition (lower).

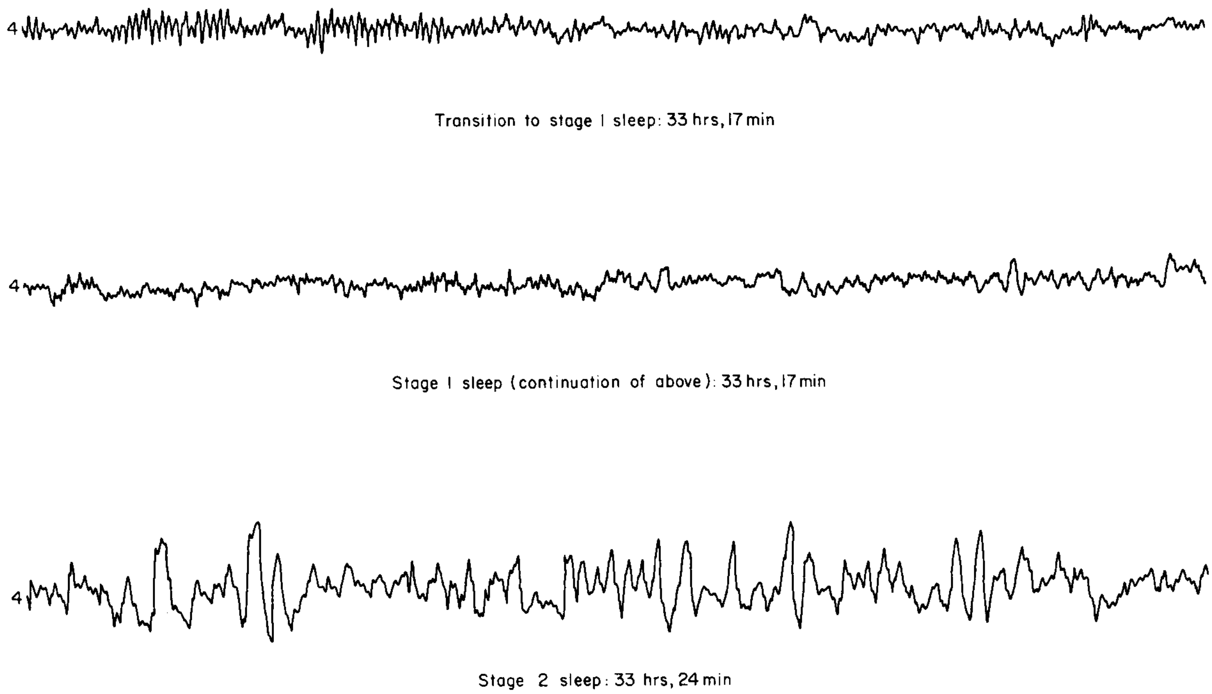

Figure 45-3.—EEG recordings showing progression from awake to light sleep.

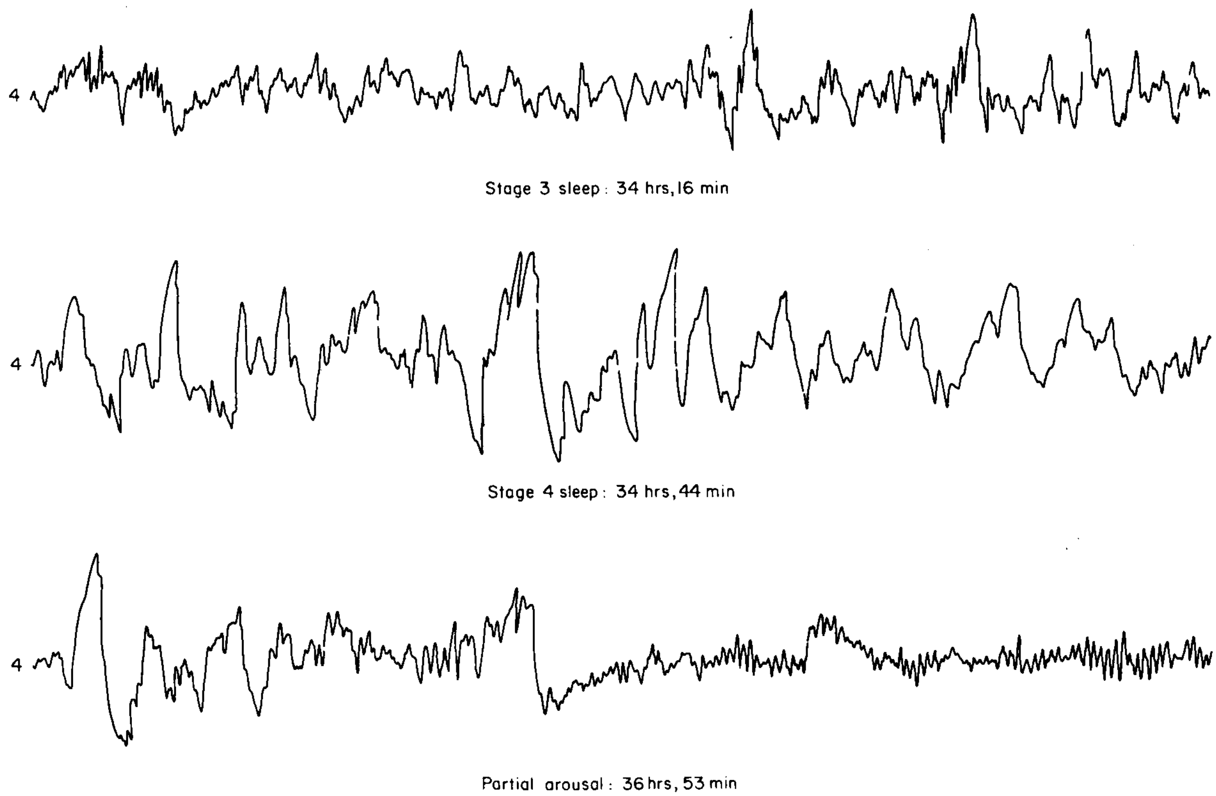

Figure 45-4.—Example of EEG recordings of moderate sleep (stage 3), deep sleep (stage 4), and partial arousal.

tive patterns found at each level of sleep. These illustrations were taken from the second sleep period during flight.

The total sleep periods are graphically represented in figure 45–5. For ease of representation, each period of sleep is divided into 1-minute epochs, and these are illustrated by the vertical lines. The length of this line represents the range of sleep level variation during the minute it represents.

The uppermost level on the vertical axis of the graph (EO) represents the eyes-open, alert-type EEG pattern. The next lower part of the vertical axis marks the eyes-closed, resting pattern (O). Each of the next successive points on the scale represents the four levels of sleep from light to deepest sleep. When, as often happened, more than one EEG stage of sleep occurred in a 1-minute epoch, the vertical line indicating stage of sleep is drawn to show the extent of the alterations of sleep level occurring during this time.

The horizontal axis of these graphs represents the flight time in hours and minutes, translated from the time code on the recording tape.

In addition to the two sleep periods during flight, a similar graphic representation is shown of the control or baseline sleep period made in the laboratory in September 1965. This is shown in order to compare the rate and character of the "falling-to-sleep" pattern, but it cannot be used to compare the cyclic alterations occurring in a full night's sleep because the subject was awakened after 2 hours and 45 minutes. The first part of the characteristic cyclic changes of level can, however, be seen.

The first inflight sleep period shown on the right side of the graph showed marked fluctuations between light sleep and arousal, with occasional brief episodes of stage 3 sleep for the first 80 minutes. At that time stage 4 sleep was reached, but in less than 15 minutes abrupt arousal and termination of sleep occurred.

On the second day, at 33 hours and 10 minutes after lift-off, the command pilot again closed his eyes and showed immediate evidence of drowsiness. Within 34 minutes he was in the deepest level of sleep (stage 4).

During this prolonged period of sleep, there were cyclic alterations in level similar to those which occur during a full night of sleep under normal conditions. Such cyclic changes are usually irregular and aperiodic, as shown in figure 45–6, which is taken from a normal control series studied by Dement and Kleitman. Generally, each successive swing toward deeper

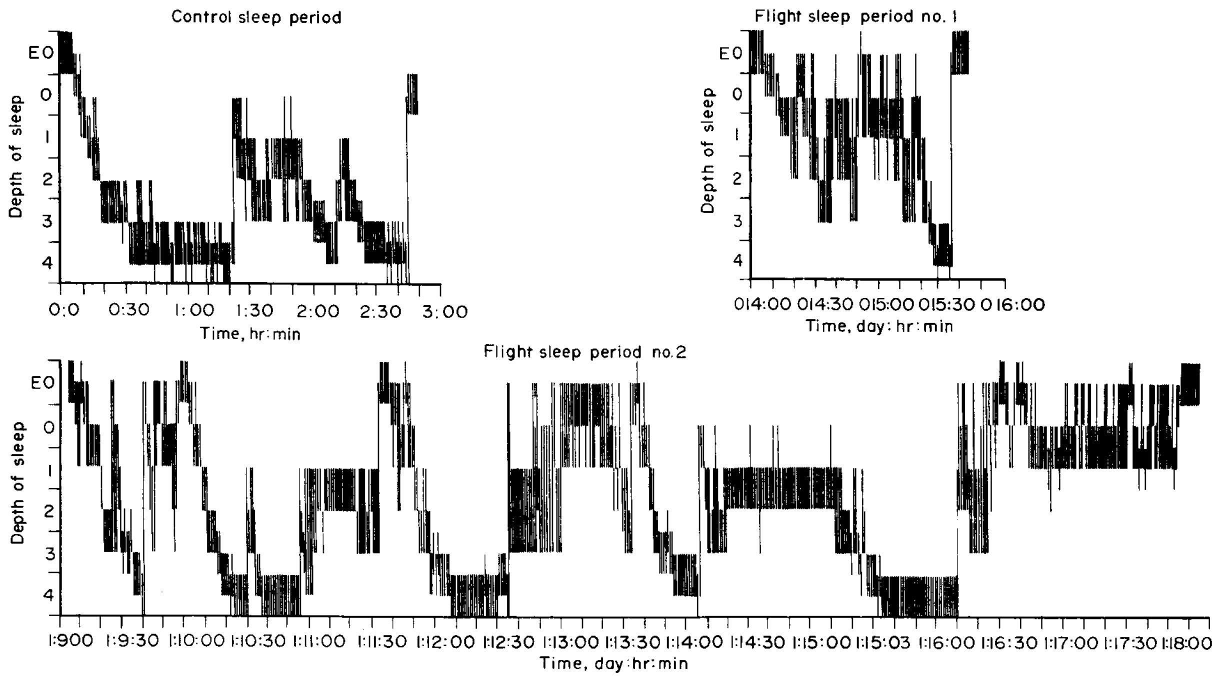

FIGURE 45–5.—Analysis of control sleep period and two flight sleep periods.

sleep, after the first period of stage 4 has been obtained, only reaches successively lighter levels; but, in Borman's second night of sleep, stage 4 was reached and maintained for 20 minutes or more at three different times after the first episode. It is interesting to speculate as to whether this increase in the number of stage 4 periods reflected an effect of deprivation of sleep during the first 24 hours.

After approximately 7 hours of sleep, a partial arousal from stage 4 sleep occurred, and, after a brief period (12 minutes) of fluctuating between stages 2 and 3, Borman remained in a state fluctuating between drowsiness and stage 1 sleep until finally fully roused about 1.5 hours later. Whether any periods of the so-called "paradoxical" sleep, rapid eye movement sleep, or dreaming sleep occurred during this oscitant period cannot be determined with certainty from our records because of the absence of eye movement records and because paradoxical sleep is generally very similar in its character to ordinary stage 1 sleep. However, two periods of a pattern which resemble an admixture of certain characteristics of stage 1 and stage 2 sleep, and which resemble some of the activity which this group and other investigators have observed in paradoxical sleep, were recorded for relatively long periods in the second day's sleep (at 11:05 G.m.t. and 14:20 G.m.t.). Typical examples of this activity (which consists of runs of 3 per second "saw-tooth" waves, runs of low-voltage theta and alpha activity, low-voltage beta activity without spindles, and occasional slow transients with a time course of about 1 second are shown in figure 45-7.

Conclusions

This experiment has clearly demonstrated the feasibility of recording the EEG during space flight. Refinement of technique and the development of more comfortable and efficient electrode systems will soon permit recording throughout prolonged space flights.

The precise information which the EEG can afford concerning the duration, depth, and number of sleep periods suggests that EEG monitoring should be considered for routine use in the prolonged space flights contemplated in the Apollo and other programs.

The importance of such information in the direction and execution of the flight, both to the medical monitors on the ground and to the crew, is evident.

In the meantime, EEG studies presently planned in the Gemini and Apollo programs, correlated in time with activity and events aboard the space vehicle, should provide important information for the formulation of future flight plans in relationship to scheduling of sleep periods.

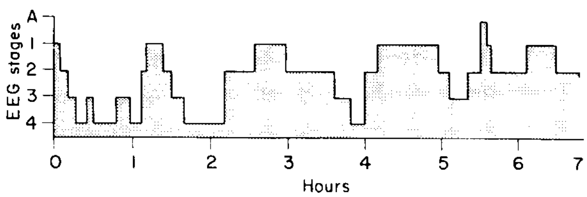

FIGURE 45-6.—Graph of cyclic variations during spontaneous sleep.

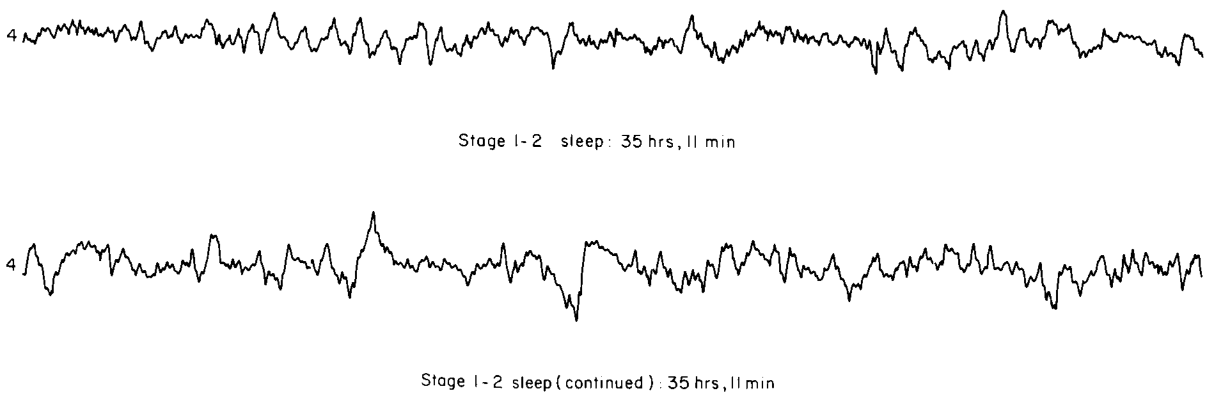

FIGURE 45-7.—Sample of EEG recording showing a mixture of stage 1 and stage 2 sleep (possibly representing "paradoxical" sleep phase).

The analysis of sleep by EEG is a very elementary exercise at the present state of the art. The possibility that monitoring electrical brain activity may yield important information concerning higher brain functions during flight has yet to be fully explored. It is to be hoped that the full exploration of the potentiality of electroencephalography as an analytic tool in brain function can be realized through the intense efforts catalyzed by the space program.

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46. EXPERIMENT M-9, HUMAN OTOLITH FUNCTION

By EARL MILLER, M.D., *U.S. Navy School of Aviation Medicine*

Objective

The purpose of the M-9 Experiment for the Gemini VII flight was identical to the experiment carried out in conjunction with the fifth flight of the Gemini series. In these flights, two kinds of information were sought:

(1) The ability of the astronauts to estimate horizontality with reference to the spacecraft in the absence of vision and primary gravitational cues.

(2) The possible effect of prolonged weightlessness on otolith function.

Preliminary results obtained during the Gemini V mission are contained in reference 1. In this report comparisons will be made among the results of the four pilots (A, B, C, D) involved in the Gemini V and VII missions.

Egocentric visual localization of the horizontal (EVLH) was the test chosen to measure "horizontality," inflight as well as preflight and postflight. It may best be described by means of an illustration (fig. 46-1). If an observer, while seated upright under ordinary conditions, regards a dim line of light in darkness, he is able to set a line in the dark to the horizontal with great accuracy (ref. 2). If, under proper conditions, he is exposed to a change in the gravitoinertial vertical with respect to himself, he is able to set the line approximately perpendicular to the changing direction of the mass acceleration (ref. 3). This indicates that in the absence of visual cues (the line itself is an inadequate cue), the ability of the observer to estimate the vertical and horizontal is due to the influence of primary and secondary gravitational cues. Persons with bilateral loss of the organs of equilibrium (otolith apparatus) are inaccurate in carrying out this task, indicating the important role of the otolith apparatus in signaling the upright. In weightlessness, primary gravitational cues are lost, and the otolith apparatus is physiologically deafferentated (ref. 4); that is to say, it has lost its normal stimulus. This creates a unique opportunity to investigate the role of secondary gravitational cues in orientation to the environment with which a person is in contact. The crewman in orbital flight is cued to his spacecraft, even with eyes closed, by virtue of tactile cues. Consequently, as a first step in exploring the loss of primary gravitational cues in space flight, it was deemed worthwhile to obtain serial EVLH measurements.

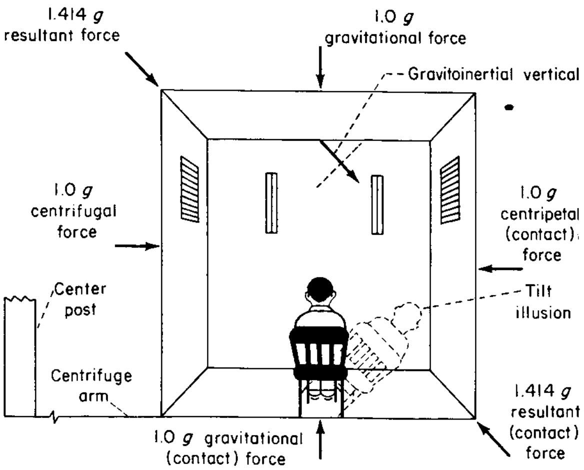

FIGURE 46-1.—Diagram illustrating egocentric visual localization of the horizontal in response to and in accordance with the direction of the active gravitational or gravitoinertial force.

Otolith function was measured by means of ocular counterrolling (ref. 5) during preflight and postflight periods. It depends on the observation that, when a person is tilted rightward or leftward, the eyes tend to rotate in the opposite sense. If proper technique is used (ref. 5), the amount of counterroll can be measured accurately. Persons with bilateral loss of otolith function either do not manifest counterrolling or the roll is minimal, possibly indicating a slight residual function (ref. 6). In its present form this test cannot be carried out in a small spacecraft; hence, the limitation exists for preflight and postflight measurements. The object of the test was to determine whether prolonged physiological deafferentation of the otolith apparatus had changed its sensitivity of response.

Apparatus and Procedure

The apparatus for measuring the EVLH of the spacecraft was incorporated into the onboard vision tester which was part of the S–8/D–13 Experiment. This incorporation was made to save weight and space and represented only a physical interface; in all other respects the two experiments were completely separate entities. The inflight vision tester is a binocular instrument (fig. 46–2) with an adjustable interpupillary distance (IPD) but without any focusing adjustment. The instrument device is held at the proper position, with the lines of sight coincident with the optic axes of the instrument, by means of a biteboard individually fitted to the subject. This insured that at each use the instrument was similarly located with respect to the subject's axes, if he had made the proper IPD adjustment. In this position the eyecups attached to the eyepieces of the instrument excluded all extraneous light from the visual field.

FIGURE 46–2.—Subject using vision tester with head brace attached to the instrument panel of the spacecraft.

Direct-current power regulated by the instrument was supplied by the spacecraft.

A headbrace, as shown in figure 46–2, was provided to connect the biteboard of the instrument to the map-board slot of the spacecraft and thereby eliminate any rolling movement or displacement of the zero target setting for horizontal with respect to the spacecraft; a limited amount of freedom around its pitch axis was permitted by the folding configuration of the brace as designed for storage purposes. This method of fixing the vision tester to the spacecraft was not used in the Gemini V mission, but a similar positioning of the instrument was achieved by having the subject sit erect in his seat with his head alined with the headrest.

The apparatus used represented a modification and miniaturization of a target device previously described (ref. 3). It consisted essentially of a collimated line of light in an otherwise dark field. This line could be rotated about its center by means of a knurled knob. A digit readout of line position was easily seen and was accurate within $\pm 0.25°$.

The device was monocular and fabricated in duplicate so that the astronaut in the left-hand seat used the right eye with the readout visible to the astronaut on his right; and vice versa with the other astronaut. The readout was adjusted so that horizontality to the apparatus was 76.6° for the astronaut on the left and 101.6° for the astronaut on the right. As in the Gemini V flight, the instrument's zero was represented by a value other than a zero of 180° to eliminate or reduce the possible influence of knowledge of the settings upon subsequent judgments.

The apparatus used for measuring ocular counterrolling (CR) is essentially a tilt device on which a camera system is mounted (ref. 7). The main supporting part of the CR device acts as a carrier for the stretcher-like section. This section contains Velcro straps and a saddle mount to secure the subject in a standing position within the device. It can be rotated laterally to $\pm 90°$ about the optic axis of the camera system and, when the subject is properly adjusted, about the visual axis of his right or left eye. A custom fitted biteboard was also used in CR testing to fix the subject's head with respect to the camera recording system.

The camera system used to photograph the natural iris landmarks includes a motor-driven

35-millimeter camera with bellows extension and an electronic flash unit. A console located at the base of the tilt device contains a bank of power packs which supply the electronic flash, a timer control mechanism, and controls for the flashing, round fixation light which surrounds the camera lens. A triaxial accelerometer unit which senses and relays signals of linear acceleration to a galvanometer recorder was mounted to the head portion of the device for shipboard use.

A test cubicle 12 feet by 16 feet by 10 feet (height) insulated against outside sounds, light, and temperature was constructed for carrying out the postflight tests of EVLH and CR onboard the recovery carrier.

Method

The preflight testing of CR and EVLH for both subjects was accomplished at Pensacola, Fla., and Cape Kennedy at 19 and 6 weeks, respectively, prior to the flight.

Immediately prior to the preflight and postflight testing of EVLH, one drop of 1 percent pilocarpine hydrochloride ophthalmic solution was instilled in the subject's eye which was opposite to the eye used for making visual orientation judgments. The subject was then placed in the CR tilt device, properly adjusted, and secured. The method of conducting the preflight and postflight EVLH test was as follows: The IPD of the vision tester was adjusted and the device was brought into its proper position by inserting the biteboard into the mouth of the subject. The experimenter initially offset the line target presented to one eye only (the other eye observed a completely dark field). By means of the knurled wheel, the subject rotated the target clockwise or counterclockwise until it appeared to be alined perpendicular to the gravitational vertical. This procedure was repeated in each test session until eight settings had been made in the upright position.

The method of testing EVLH in flight was as follows: Immediately after completion of the S-8/D-13 Experiment, and without removing the instrument from his face, the subject prepared for EVLH testing by occluding the left eyepiece (command pilot) or right eyepiece (pilot) by means of the ring of the eyepiece, and turning on the luminous target before the opposite eye. The target appearing against a completely dark background was initially offset at random by the observer pilot. The subject pilot's experimental task was to adjust the target until it appeared horizontal with respect to his immediate spacecraft environment. The subject, when satisfied with each setting, closed his eyes and removed his hand from the knurled ring. This served as a signal to the observer pilot to record the setting and offset the target. This procedure was repeated five times during each of the daily test sessions. The vision tester was then handed to the other pilot and the same sequence was carried out after completion of the visual acuity test. Finally, the readings for each pilot were tape recorded by voice. The subjects were instructed to apply the same amount of tension on their seat belts during the EVLH test in an attempt to keep the influence of secondary gravitational cues upon these judgments as constant as possible.

The preflight and postflight measurements of ocular CR were accomplished according to the standard procedure used at the U.S. Naval Aerospace Medical Institute. Following the EVLH test, the subject remained in the upright position in the tilt device. The vision tester and its biteboard were removed, and preparations were made for photographically recording the eye position associated with a given position of body tilt. The CR biteboard was inserted in the subject's mouth, and the position of his appropriate eye was adjusted so that it coincided with the optic axis of the camera system when he fixated the center of the flashing red ring of light. Six photographic recordings were made at this position; then the subject was slowly tilted in his lateral plane to each of four other positions ($\pm 25°$, $\pm 50°$) and the same photographic procedure was repeated.

The accelerometer system was used during the postflight EVLH and CR tests to record continuously the motions of the recovery ship around its roll, pitch, and yaw axes.

During the EVLH and CR tests, readings of blood pressure, pulse rate, and electrocardiogram were monitored by NASA Manned Spacecraft Center medical personnel. Postflight examinations were begun for pilot D and pilot C approximately 4.5 and 6 hours, respectively, following their recovery at sea.

Results

Ocular Counterrolling

Preflight.—Three separate preflight measurements of ocular CR (fig. 46–3) made on the same day indicated that basic otolithic function of pilot C and pilot D were well within the range of counterrolling response found among a random population of 100 normal subjects (represented in fig. 46–4 by the shaded area). This CR response of each member of Gemini VII crew is markedly different from that found for the crew pilots (CP, CN) but similar to other crewmen who have been tested (fig. 46–4).

Postflight.—As seen in figure 46–3, postflight measurements (solid line) revealed no significant change in the mean CR response from that manifested before the flight (broken line). The slight differences in the CR curves can be accounted for by the small rotary oscillations (physiological unrest) of the eye and the fact that an average of several recordings is used to define the position of the eyes associated with any given body tilt.

Egocentric Visual Localization of the Horizontal (EVLH)

Preflight and postflight.—The deviations from the instrument's zero of the pilot's discrete EVLH settings are summarized in figure 46–5. The judgments of each pilot in an upright body position as to the location of the horizontal under normal gravitational conditions were somewhat unstable prior to the flight. In approximately one-half the settings, deviations greater than 5° were recorded, and one setting of each pilot exceeded 10°. On the day of recovery, the pattern of response was similar to that of preflight in spite of the fact that the judgments were made under unstable, though relatively calm, sea conditions. The accelerometer tracings are being analyzed to determine the magnitude of linear and angular acceleration that occurred during the postflight test.

Inflight.—The EVLH judgment throughout the flight showed no trends with respect to longitudinal changes in the stability or absolute position of horizontal within the spacecraft. However, it should be noted that, on the initial day of testing, pilot C revealed somewhat more deviation on the average than during succeeding test sessions. In general, comparison of estimations of horizontality under weightless conditions were substantially more closely oriented to the immediate physical environment and more consistent than comparable EVLH settings under standard gravitational conditions.

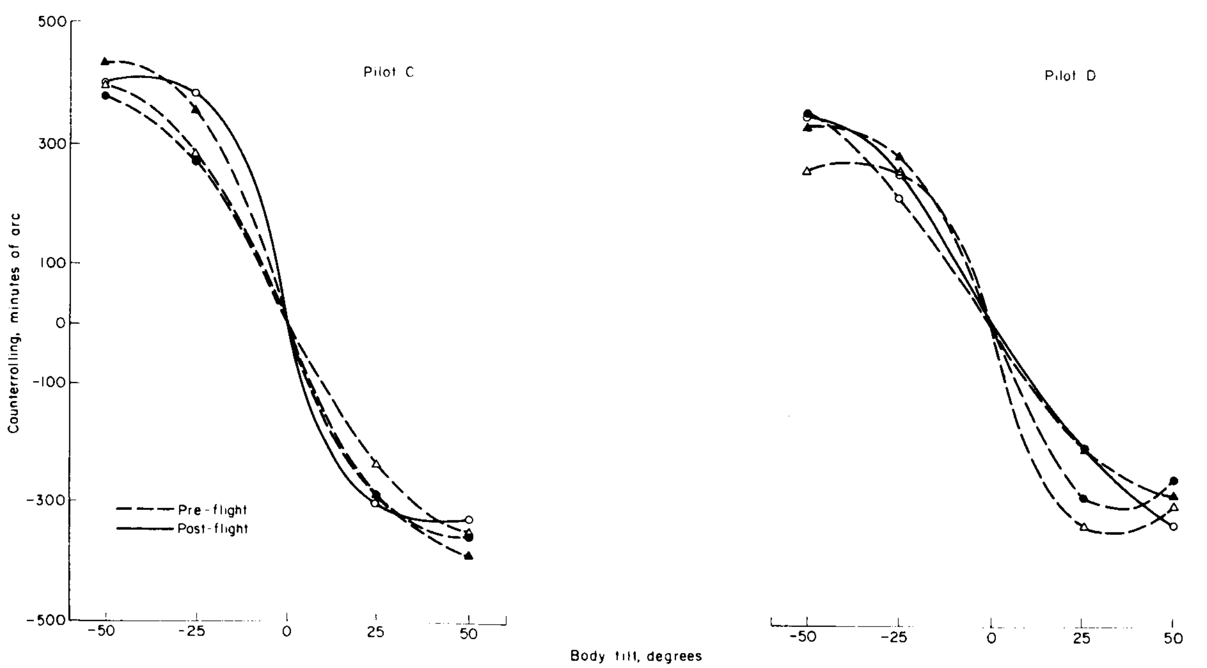

FIGURE 46–3.—Mean counterrolling response of each pilot subject.

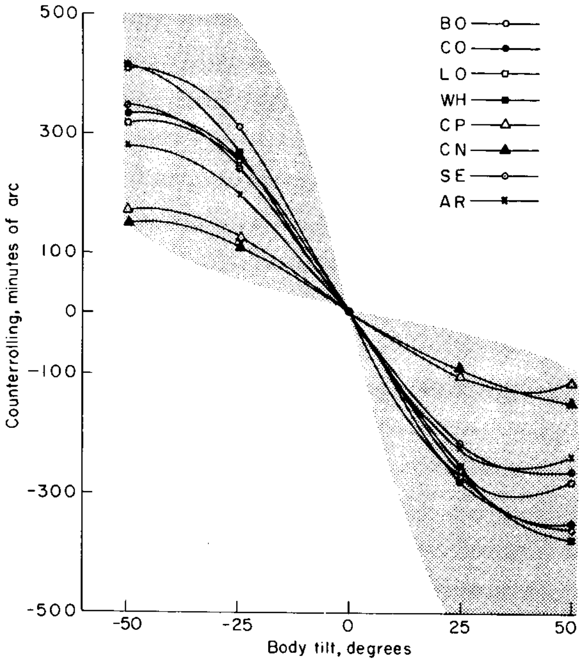

FIGURE 46-4.—Counterrolling response curves of eight astronauts (shaded area represents range of response of 100 randomly selected subjects).

Discussion

The completion of the M-9 Human Otolith Function Experiment carried out in conjunction with the Gemini V and VII flights has provided quantitative information concerning otolithic sensitivity and orientation of four subjects exposed to an orbiting spacecraft environment for prolonged periods of time.

Preflight counterrolling measurements revealed marked differences between the Gemini V and VII crews with regard to the magnitude of their basic response; however, after the flight, each pilot maintained his respective preflight level of response, which indicated that no significant change in otolithic sensitivity occurred as a result of the flight, or at least no change persisted long enough to be recorded several hours after recovery.

The EVLH data recorded for each subject confirmed the observation made repeatedly in parabolic flight experiments that a coordinate space sense exists even in weightlessness if contact cues are adequate; however, it was found that the apparent location of the horizontal within the spacecraft may not necessarily agree

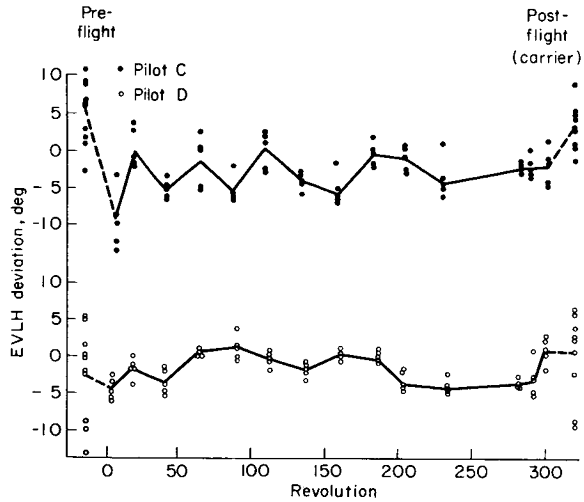

FIGURE 46-5.—Deviation from instrument's absolute zero of individual settings of EVLH.

with its physical correlate in the spacecraft (a line parallel to the vehicle's pitch axis). The data taken of pilot A, for example, revealed greater than 30° deviation from the absolute horizontal, indicating that with eyes closed the cues furnished by virtue of contact with the spacecraft did not allow correct perception of the cabin vertical. The uniformity of his settings throughout the flight suggested, furthermore, that "learning" did not occur in the absence of any knowledge of the accuracy of these estimates. With one possible exception already noted on pilot C in his first inflight test session, EVLH judgments were relatively accurate and more stable than under normal gravitational conditions. These data show that relatively accurate and consistent nonvisual orientation is possible throughout a prolonged period of weightless exposure so long as secondary cues are adequate. These same cues, however, may, in certain individuals, contribute to rather large errors in the perception of the principal coordinates of the spacecraft.

The potential influence of sensory cues on orientation is well known to the aviator who has experienced the "leans," that is, the tendency either to fly with one wing low, or, in straight and level flight using instruments, to feel inclined away from the "upright." This not uncommon illusion occurs in spite of the relative abundance of cues in this situation compared with those in a spacecraft. Further

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experimentation involving inflight serial EVLH measurements is planned in conjunction with the Apollo flights to increase the knowledge of the role of secondary cues in orientation, and the possible interindividual differences in their influence upon the crewman.

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APPENDIXES

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APPENDIX A

NASA CENTERS AND OTHER GOVERNMENT AGENCIES

This appendix contains a list of Government agencies participating in the Gemini Program.

NASA Headquarters, Washington, D.C., and the following NASA centers:
- Ames Research Center, Moffett Field, Calif.
- Electronics Research Center, Cambridge, Mass.
- Flight Research Center, Edwards, Calif.
- Goddard Space Flight Center, Greenbelt, Md.
- Kennedy Space Center, Cocoa Beach, Fla.
- Langley Research Center, Langley Station, Hampton, Va.
- Lewis Research Center, Cleveland, Ohio
- Manned Spacecraft Center, Houston, Tex.
- Marshall Space Flight Center, Huntsville, Ala.

Department of Defense, Washington, D.C.:
- Department of the Army
- Department of the Navy
- Department of the Air Force

Department of State, Washington, D.C.

Department of Commerce, Washington, D.C.

Department of the Interior, Washington, D.C.

Department of Health, Education, and Welfare, Washington, D.C.

Department of the Treasury, Washington, D.C.
- U.S. Coast Guard

Atomic Energy Commission, Washington, D.C.

Environmental Science Services Administration

U.S. Information Agency, Washington, D.C.

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APPENDIX B

CONTRACTORS, SUBCONTRACTORS, AND VENDORS

This appendix contains a listing of contractors, subcontractors, and vendors that have Gemini contracts totaling more than $100 000. It represents the best effort possible to obtain a complete listing; however, it is possible that some are missing, such as those supporting activities not directly concerned with Manned Spacecraft Center activities. These contractors, subcontractors, and vendors are recognized as a group.

Contractors

Acoustica Associates, Inc., Los Angeles, Calif.
Aerojet-General Corp., Downey, Calif.
Aerospace Corp., El Segundo, Calif.
Arde Portland, Inc., Paramus, N.J.
AVCO Corp., Stratford, Conn.
Burroughs Corp., Paoli, Pa.
Bechtel Corp., Los Angeles, Calif.
Bell Aerosystems Co., division of Bell Aerospace, Buffalo, N.Y.
CBS Labs Inc., Stamford, Conn.
Cook Electric Co., Skokie, Ill.
David Clark Co., Inc., Worcester, Mass.
Evans Construction Co., Houston, Tex.
Farrand Optical Co., Inc., Bronx, N.Y.
Federal Electric Corp., Paramus, N.J.
Garrett Corp., The, AiResearch Mfg. Co. Division, Los Angeles, Calif.
General Dynamics/Astronautics Division, San Diego, Calif.
General Dynamics/Convair Division, San Diego, Calif.
General Electric Co., Syracuse, N.Y.
General Electric Co., West Lynn, Mass.
General Precision, Inc., Binghamton, N.Y.
Honeywell, Inc., Minneapolis, Minn.
International Business Machines Corp., Owego, N.Y.
J. A. Maurer, Inc., Long Island City, N.Y.
Ling-Temco-Vought Aerospace Corp., Dallas, Tex.
Lockheed Missiles & Space Co., Sunnyvale, Calif.
Martin Co., Division of Martin-Marietta Corp., Baltimore, Md.
Martin Co., Division of Martin-Marietta Corp., Denver, Colo.
McDonnell Aircraft Corp., St. Louis, Mo.
Melpar, Inc., Falls Church, Va.
North American Aviation, Inc., Rocketdyne Division, Canoga Park, Calif.
Philco Corp., Philadelphia, Pa.
Philco Corp., WDL Division, Palo Alto, Calif.
Space Labs, Inc., Van Nuys, Calif.
TRW Systems, Inc., Redondo Beach, Calif.
Sperry Rand Corp., Sperry Phoenix Co. Division, Phoenix, Ariz.
Western Gear Corp., Pasadena, Calif.
Whirlpool Corp., St. Joseph, Mich.

Subcontractors and Vendors

ACF Industries, Inc., Paramus, N.J.
ACR Electronics Corp., New York, N.Y.
Advanced Communications, Inc., Chatsworth, Calif.
Advanced Technology Laboratories, Inc., Mountain View, Calif.
Aeronca Manufacturing Corp., Baltimore, Md.
Aeroquip Corp., Jackson, Mich.
American Machine & Foundry Co., Springdale, Conn.
American Radiator & Standard Sanitary Corp., Mountain View, Calif.
Astro Metallic, Inc., Chicago, Ill.
Autronics Corp., Pasadena, Calif.
Avionics Research Corp., West Hempstead, N.Y.
Barnes Engineering Co., Stamford, Conn.
Beech Aircraft Corp., Boulder, Colo.
Bell Aerosystems Co., Buffalo, N.Y.
Bendix Corp., Eatontown, N.J.
Brodie, Inc., San Leandro, Calif.
Brush Beryllium Co., Cleveland, Ohio

Brush Instrument Corp., Los Angeles, Calif.
Burtek, Inc., Tulsa, Okla.
Cadillac Gage Co., Costa Mesa, Calif.
Cannon Electric Co., Brentwood, Mo.
Cannon Electric Co., Phoenix, Ariz.
Calcor Space Facility, Whittier, Calif.
Captive Seal, Inc., Caldwell, N.J.
Central Technology Corp., Herrin, Ill.
Clevite Corp., Cleveland, Ohio
Clifton Precision Co., Clifton Heights, Pa.
Collins Radio Co., Cedar Rapids, Iowa
Computer Controls Corp., Framingham, Mass.
Comprehensive Designers, Inc., Philadelphia, Pa.
Consolidated Electrodynamics Corp., Monrovia, Calif.
Cosmodyne Corp., Hawthorne, Calif.
Custom Printing Co., Ferguson, Mo.
Day & Zimmerman, Inc., Los Angeles, Calif.
De Havilland Aircraft, Ltd., Downsview, Ontario, Canada
Douglas Aircraft Co., Inc., Tulsa, Okla., and Santa Monica, Calif.
Eagle-Picher Co., Joplin, Mo.
Edgerton, Germeshausen & Grier, Inc., Boston, Mass.
Electro-Mechanical Research, Inc., Sarasota, Fla.
Electronic Associates, Inc., Long Branch, N.J.
Emerson Electric Co., St. Louis, Mo.
Emertron Information and Control Division, Litton Systems, Inc., Newark, N.J.
Engineered Magnetics Division, Hawthorne, Calif.
Epsco, Inc., Westwood, Mass.
Explosive Technology, Inc., Santa Clara, Calif.
Fairchild Camera & Instrument Corp., El Cajon, Calif.
Fairchild Camera & Instrument Corp., Cable Division, Joplin, Mo.
Fairchild Controls, Inc., Division of Fairchild Camera & Instrument Corp., Hicksville, N.Y.
Fairchild Hiller Corp., Bayshore, N.Y.
Fairchild Stratos Corp., Long Island, N.Y.
Garrett Corp., The, AiResearch Manufacturing Co. Division, Los Angeles, Calif.
General Electric Co., West Lynn, Mass.
General Precision, Inc., Binghamton, N.Y.
General Precision Aerospace, Little Falls, N.Y.
Genistron, Inc., Bensenville, Ill.
Giannini Controls Corp., Duarte, Calif.
Goodyear Aerospace Corp., Akron, Ohio
Gulton Industries, Hawthorne, Calif.
Hamilton-Standard, Division of United Aircraft Corp., Windsor Locks, Conn.
Hexcel Products, Inc., Berkeley, Calif.
H. L. Yoh Co., Philadelphia, Pa.
Honeywell, Inc., Minneapolis, Minn.
Honeywell, Inc., St. Petersburg, Fla.
Hurletron Corp., Wheaton, Ill.
Hydra Electric Co., Burbank, Calif.
International Business Machines Corp., Owego, N.Y., and New York, N.Y.
Johns-Manville Corp., Manville, N.J.
Kaiser Aerospace & Electronics Corp., San Leandro, Calif.
Kinetics Corp., Solvana Beach, Calif.
Kirk Engineering Co., Philadelphia, Pa.
La Mesa Tool & Manufacturing Co., El Cajon, Calif.
Leach Corp., Compton, Calif.
Leach Relay Corp., Los Angeles, Calif.
Lear-Siegler, Inc., Grand Rapids, Mich.
Linde Co., Whiting, Ind.
Lion Research Corp., Cambridge, Mass.
MacGregor Manufacturing Co., Troy, Mich.
Moffett Tool and Machine Co., St. Louis, Mo.
Marotte Valve Corp., Boonton, N.J.
Meg Products, Inc., Seattle, Wash.
Missouri Research Laboratories, St. Louis, Mo.
Moog, Inc., Buffalo, N.Y.
Motorola, Inc., Scottsdale, Ariz.
National Waterlift Co., Kalamazoo, Mich.
North American Aviation, Inc., Canoga Park, Calif.
Northrop Corp., Van Nuys, Calif.
Northrop-Ventura Corp., Newberry Park, Calif.
Ordnance Associates, Inc., Pasadena, Calif.
Ordnance Engineering Associates, Inc., Des Plaines, Ill.
Palomara Scientific, Redmond, Wash.
Paragon Tool & Dye Engineering, Pacoima, Calif.
Pneumodynamics Corp., Kalamazoo, Mich.
Powertron, Inc., Plainsville, N.Y.
Pollak & Skan, Inc., Chicago, Ill.
Rader & Associates, Inc., Miami, Fla.
Radiation, Inc., Melbourne, Fla.
Raymond Engineering Laboratory, Middletown, Conn.
Reinhold Engineering Co., Santa Fe Springs, Calif.

Rocket Power, Inc., Mesa, Ariz.
Rome Cable Corp., Division of Alcoa, Rome, N.Y.
Rosemount Engineering Co., Minneapolis, Minn.
Servonics Instruments, Inc., Costa Mesa, Calif.
Space Corp., Dallas, Tex.
Sperry Rand Corp., Tampa, Fla.
Sperry Rand Corp., Torrance, Calif.
Speidel Co., Warwick, R.I.
Talley Industries, Mesa, Ariz.
Teledyne Systems Corp., Hawthorne, Calif.
Texas Instruments, Inc., Dallas, Tex.
Thiokol Chemical Corp., Danville, N.J.
Thiokol Chemical Corp., Elkton, Md.
Union Carbide Corp., Whiting, Ind.
Vickers, Inc., St. Louis, Mo.
Weber Aircraft Corp., Burbank, Calif.
Western Gear Corp., Lynwood, Calif.
Western Way, Inc., Van Nuys, Calif.
Westinghouse Electric Corp., Baltimore, Md.
Whiting-Turner, Baltimore, Md.
Wyle Laboratories, El Segundo, Calif.
Yardney Electric Corp., New York, N.Y.

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