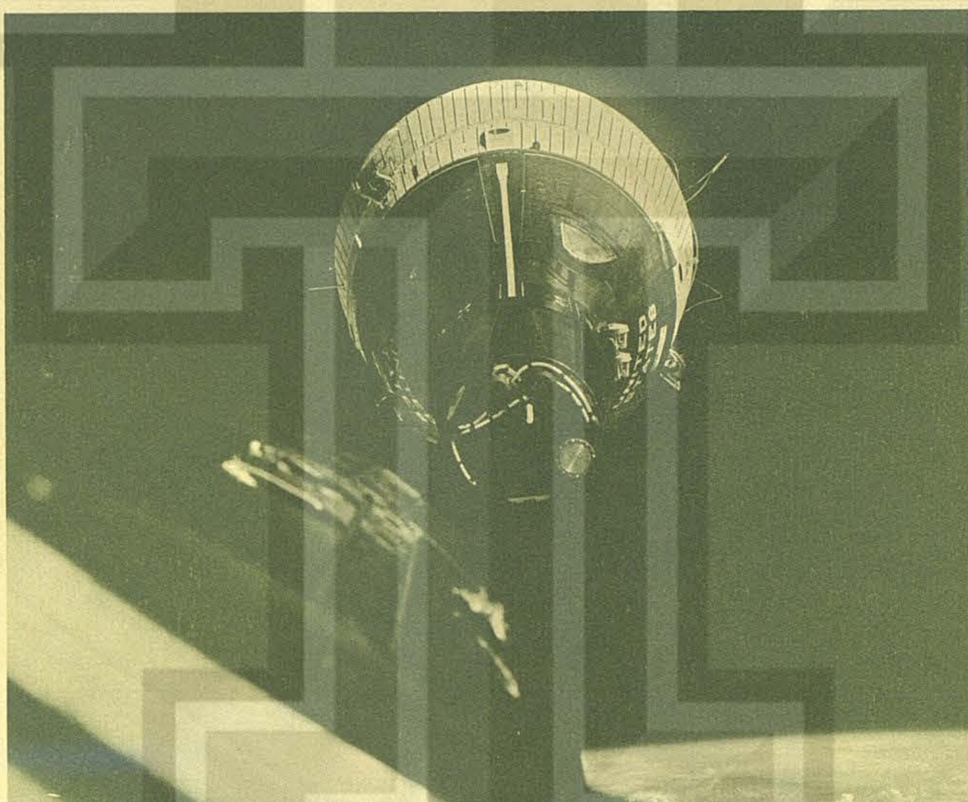
NASA SP-121

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*

®

NASA SP-121

GEMINI MIDPROGRAM CONFERENCE

INCLUDING EXPERIMENT RESULTS

MANNED SPACECRAFT CENTER

HOUSTON, TEXAS

FEBRUARY 23-25, 1966

Scientific and Technical Information Division 1966

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Washington, D.C.

For sale by the Superintendent of Documents, U.S. Government Printing Office, Washington, D.C. - 20402 - Price $2.75
Library of Congress Catalog Card Number 66–60089

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,

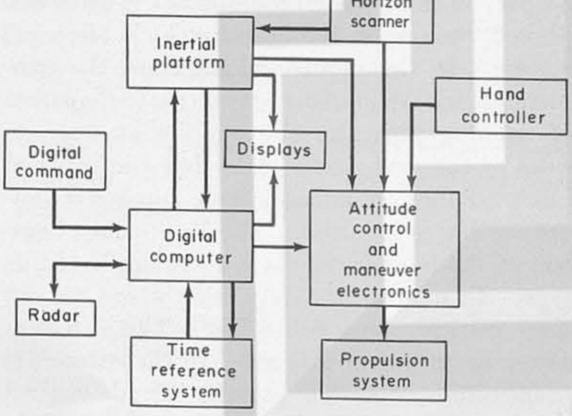


FIGURE 2–1.—Example of Gemini systems applicable to future programs and missions (guidance and control system shown).

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

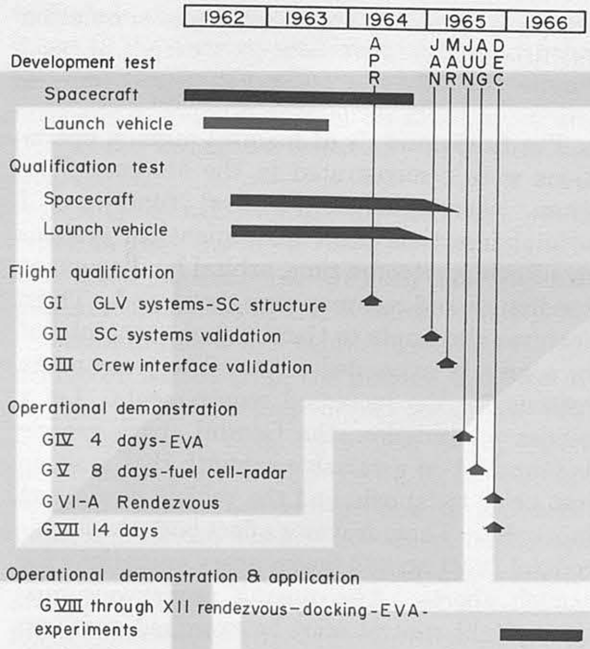


FIGURE 2-2.—Gemini test program.

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

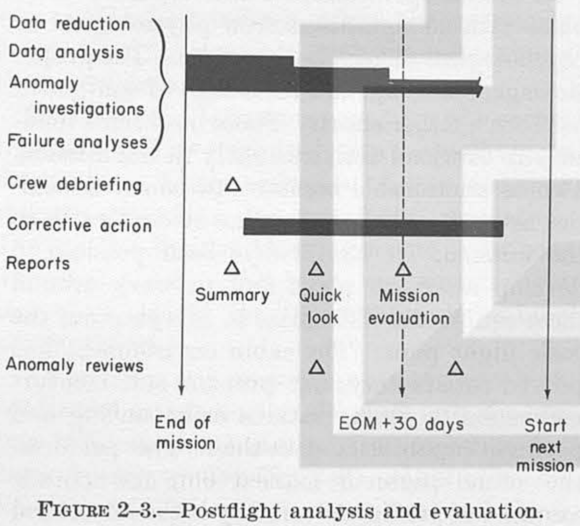


FIGURE 2–3.—Postflight analysis and evaluation.

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.

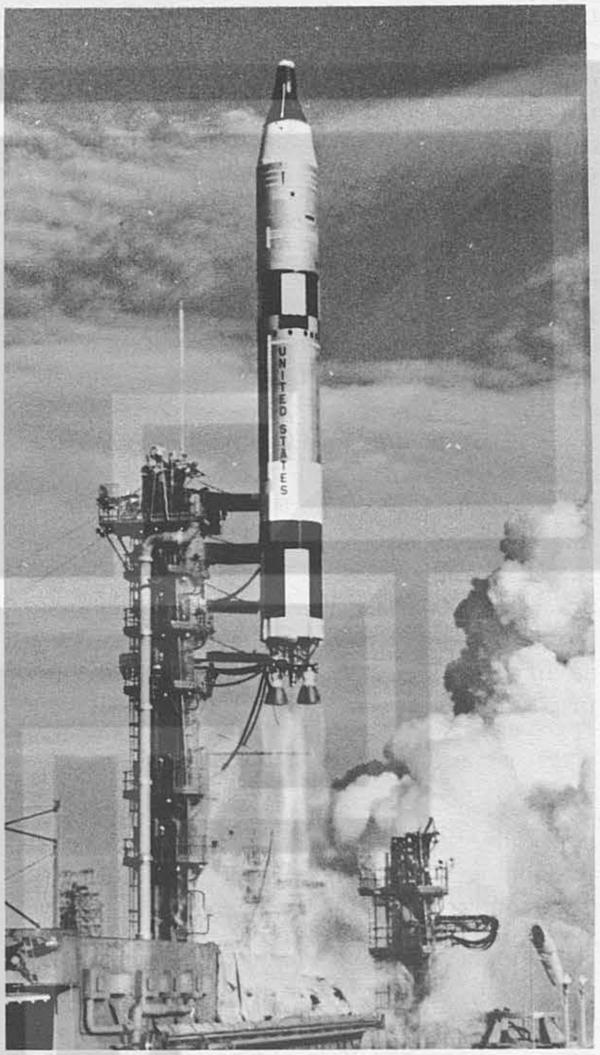


FIGURE 2–4.—Lift-off of Gemini space vehicle.

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

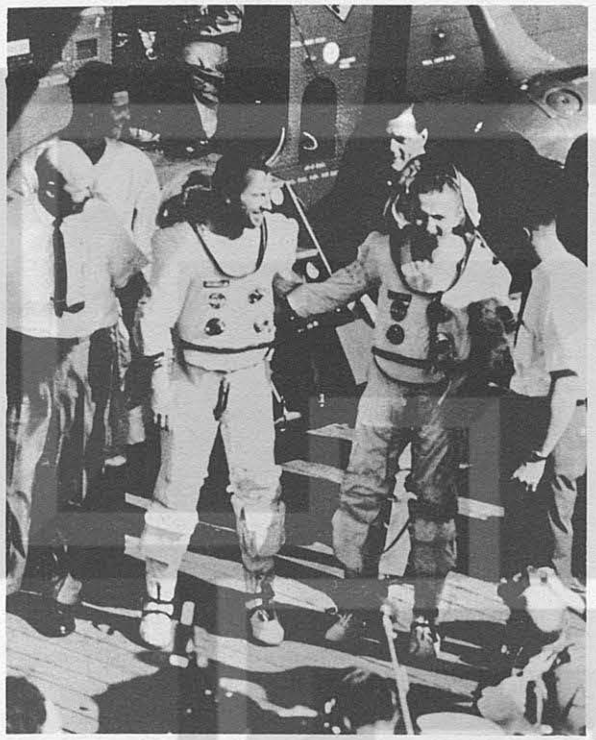


FIGURE 2-5.—Gemini VII flight crew onboard recovery ship.

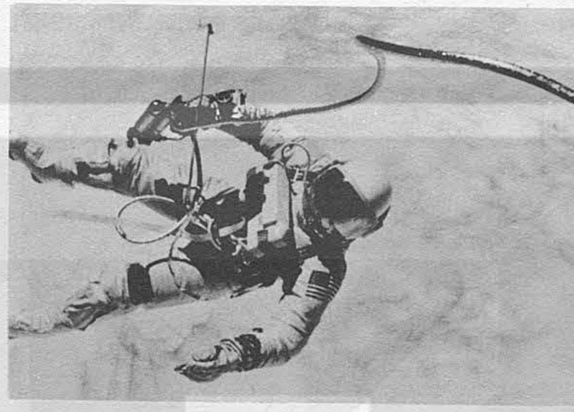


FIGURE 2-6.—Extravehicular activity during Gemini IV mission.

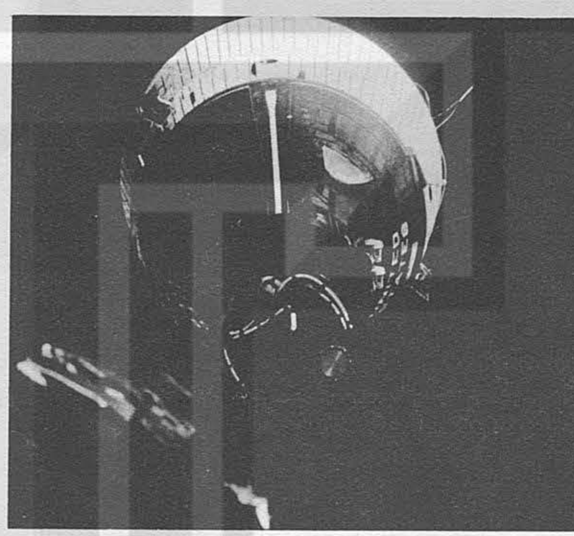


FIGURE 2-7.—Rendezvous during Gemini VI-A and VII missions.

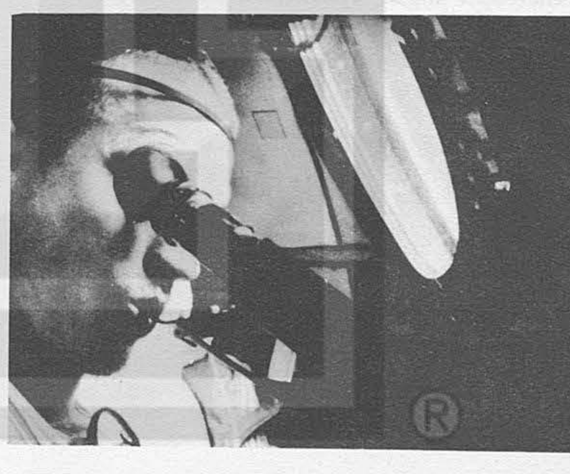


FIGURE 2-8.—Typical experiment activity.

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

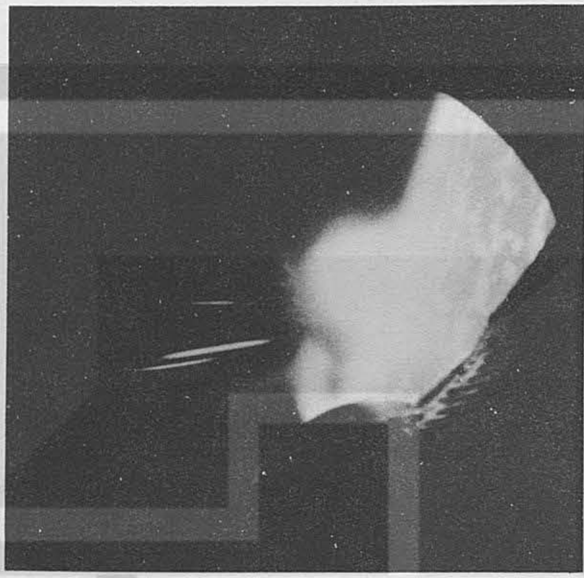


FIGURE 2-9.—View through spacecraft window during reentry.

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

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.

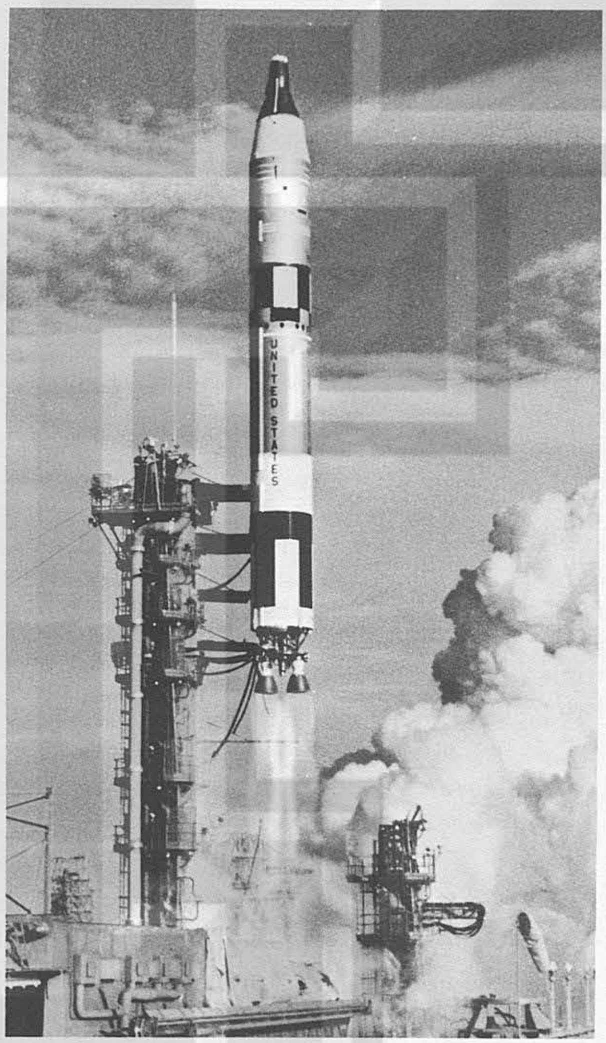


FIGURE 3–1.—Gemini space vehicle at lift-off.

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 liftoff. 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.

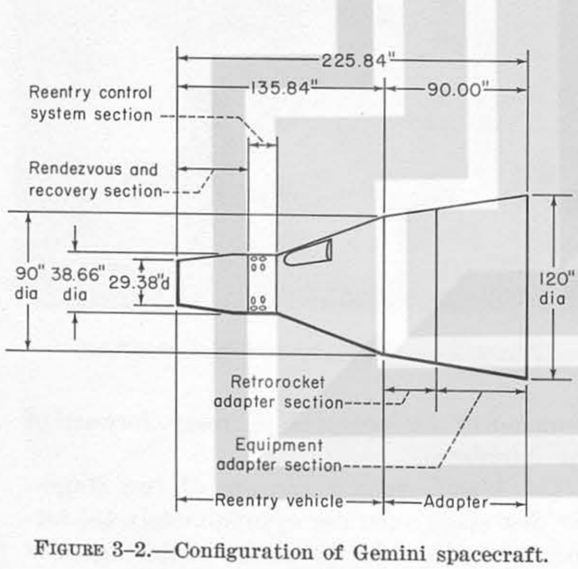


FIGURE 3–2.—Configuration of Gemini spacecraft.

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.

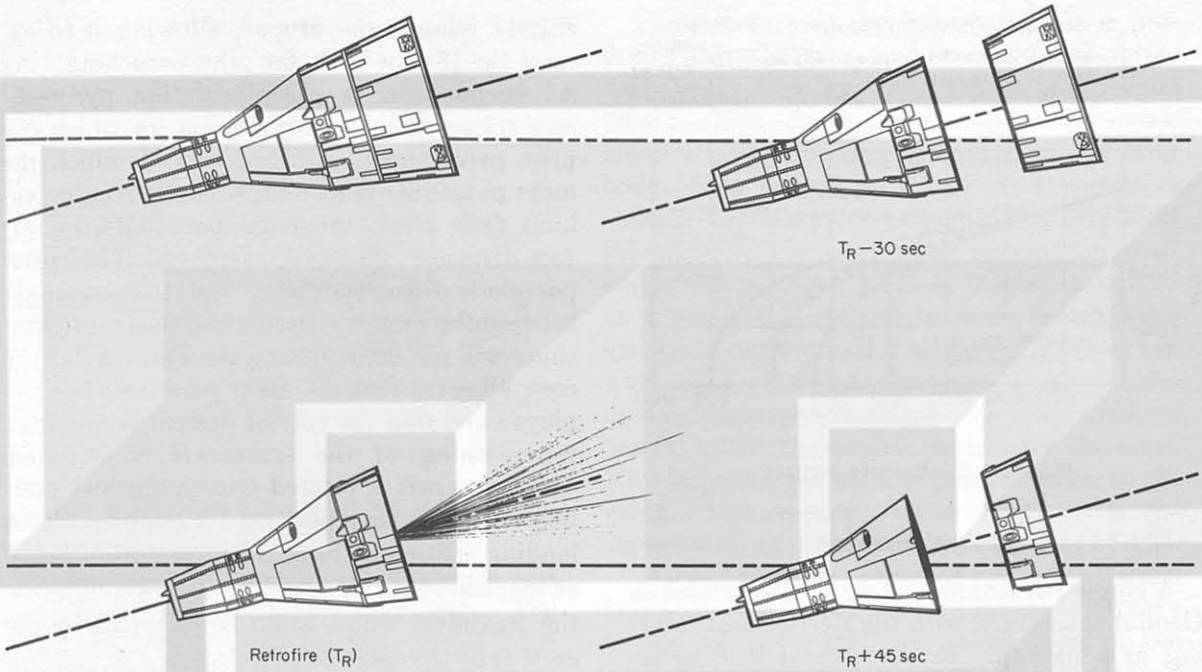


FIGURE 3-3.—Retrograde sequence.

(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.

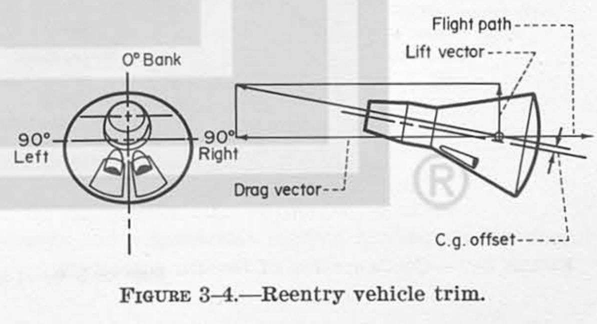


FIGURE 3-4.—Reentry vehicle trim.

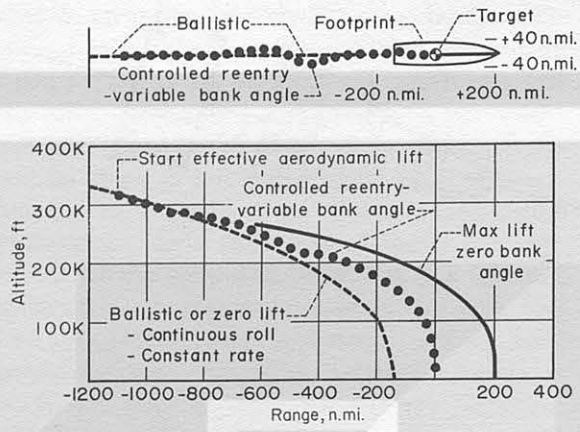


FIGURE 3-5.—Reentry control.

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-

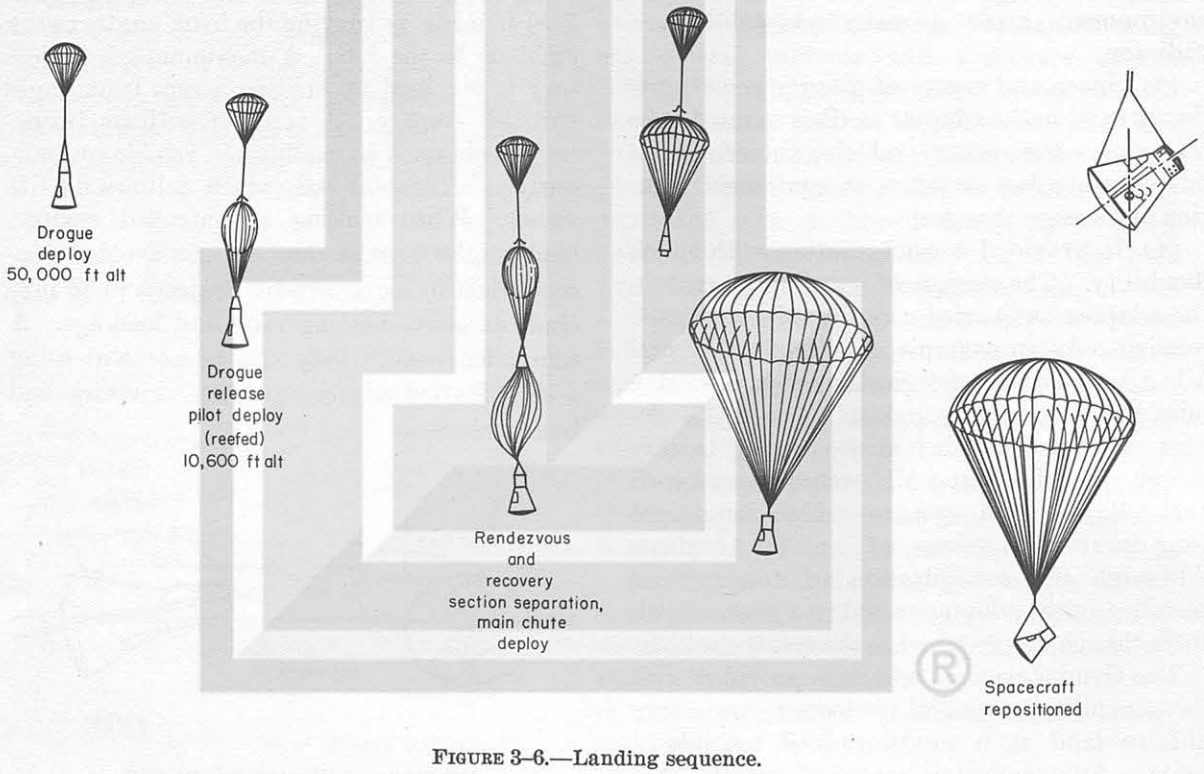


FIGURE 3-6.—Landing sequence.

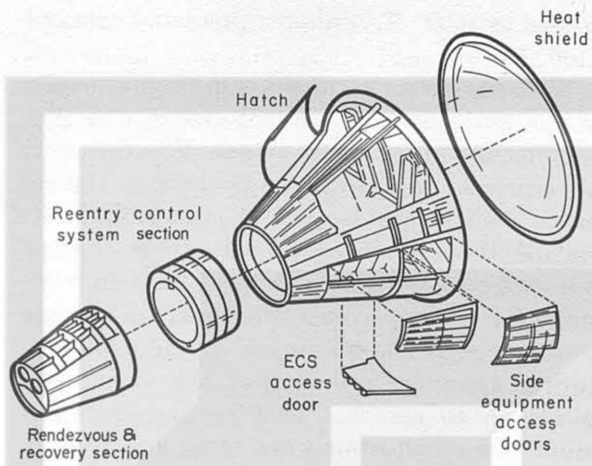


FIGURE 3-7.—Reentry vehicle structure.

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/ft^2/sec and a maximum total heat of 13 138 Btu/ft^2. 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

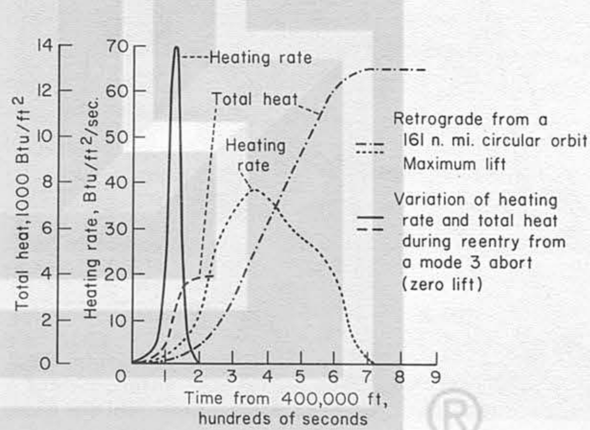


FIGURE 3-8.—Spacecraft reentry heating versus time.

shingles. Based on the Gemini II trajectory, the stagnation heating rate reached a calculated value of 71.8 Btu/ft²/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.

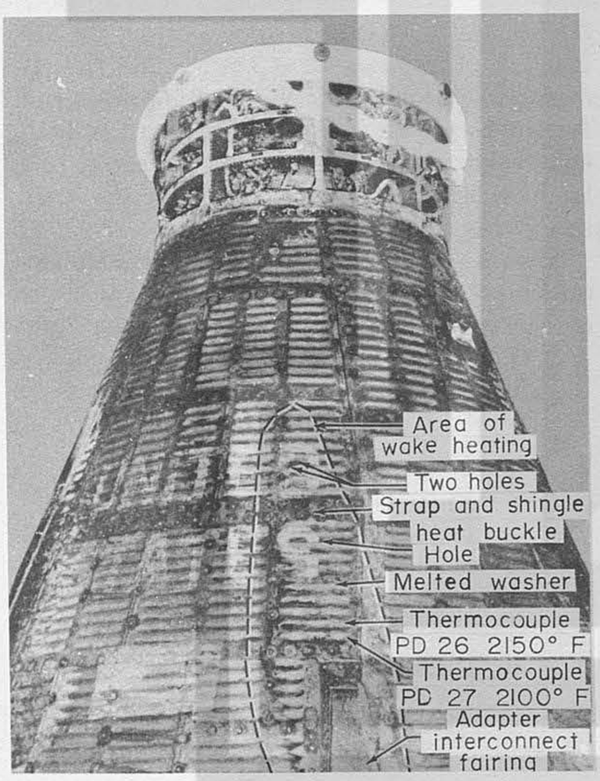


FIGURE 3–9.—Effects of reentry heating on the Gemini II spacecraft.

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.

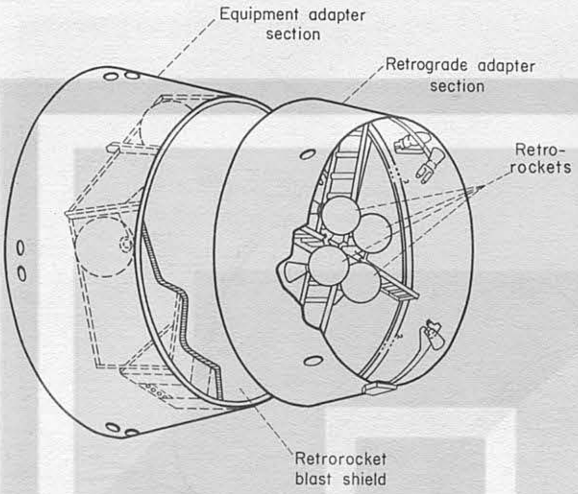


FIGURE 3-11.—Spacecraft adapter assembly.

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.

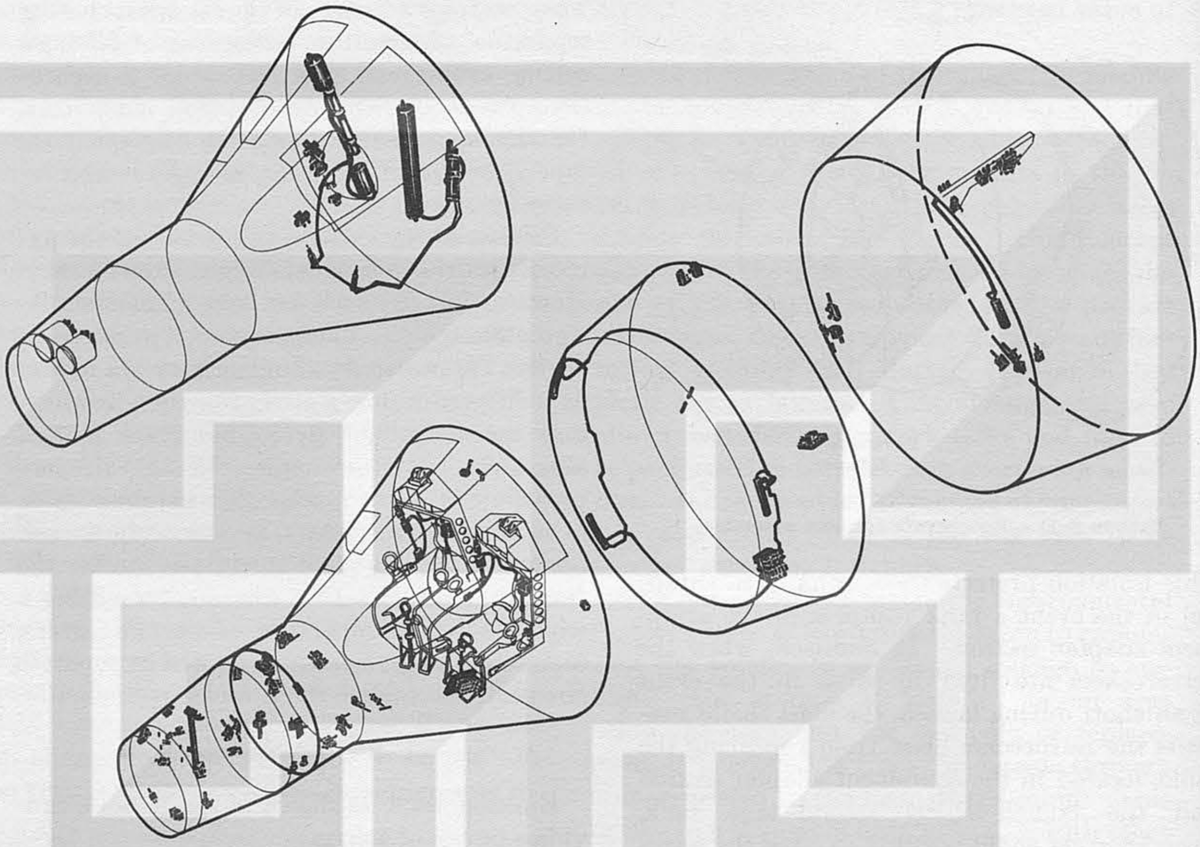


FIGURE 3-12.—Location of pyrotechnic devices in the spacecraft.

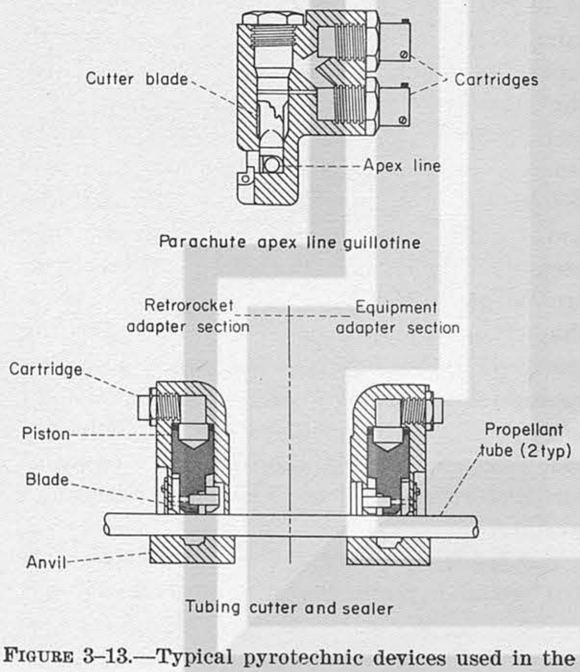


FIGURE 3-13.—Typical pyrotechnic devices used in the spacecraft.

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.

4. GUIDANCE, CONTROL, AND PROPULSION SYSTEMS

By Richard R. Carley, *Gemini Program Office, NASA Manned Spacecraft Center;* Norman Schulze, *Propulsion and Power Division, NASA Manned Spacecraft Center;* Benjamin R. Drone, *Gemini Program Office, NASA Manned Spacecraft Center;* David W. Camp, *Gemini Program Office, NASA Manned Spacecraft Center; and* John F. Hanaway, *Guidance and Control Division, NASA Manned Spacecraft Center*

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,

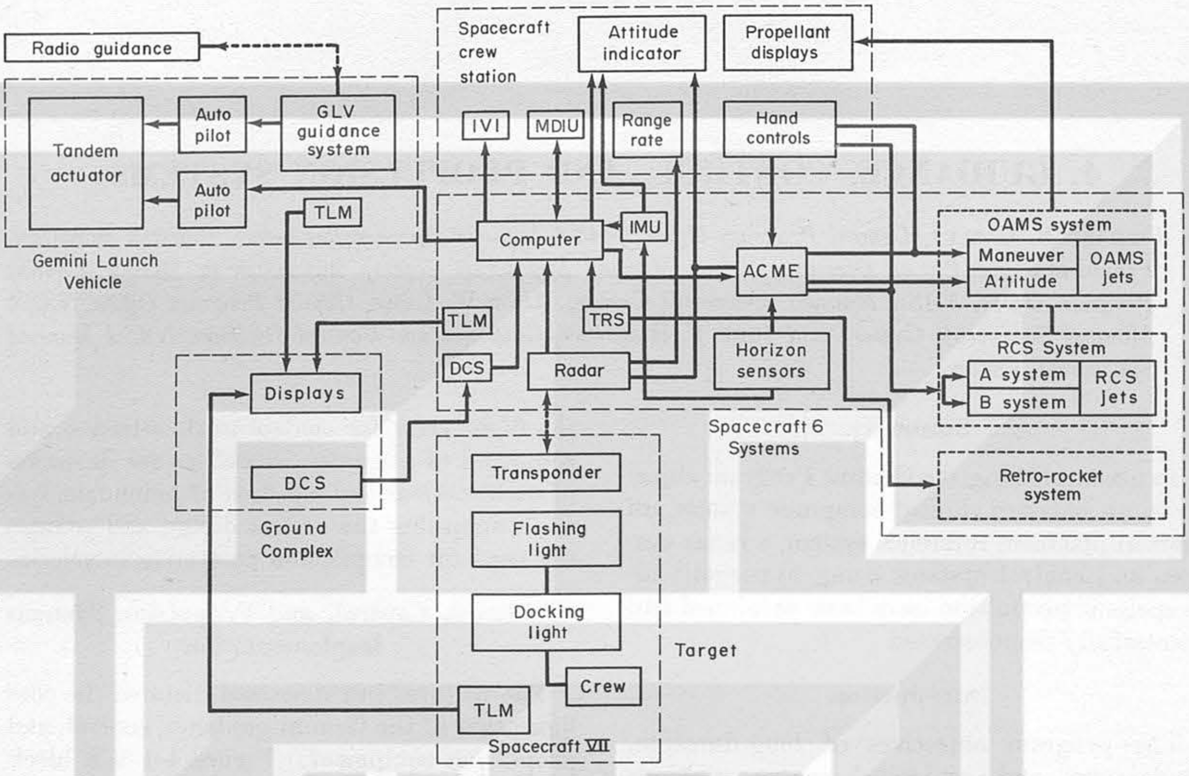


Figure 4–1.—Spacecraft guidance and control system.

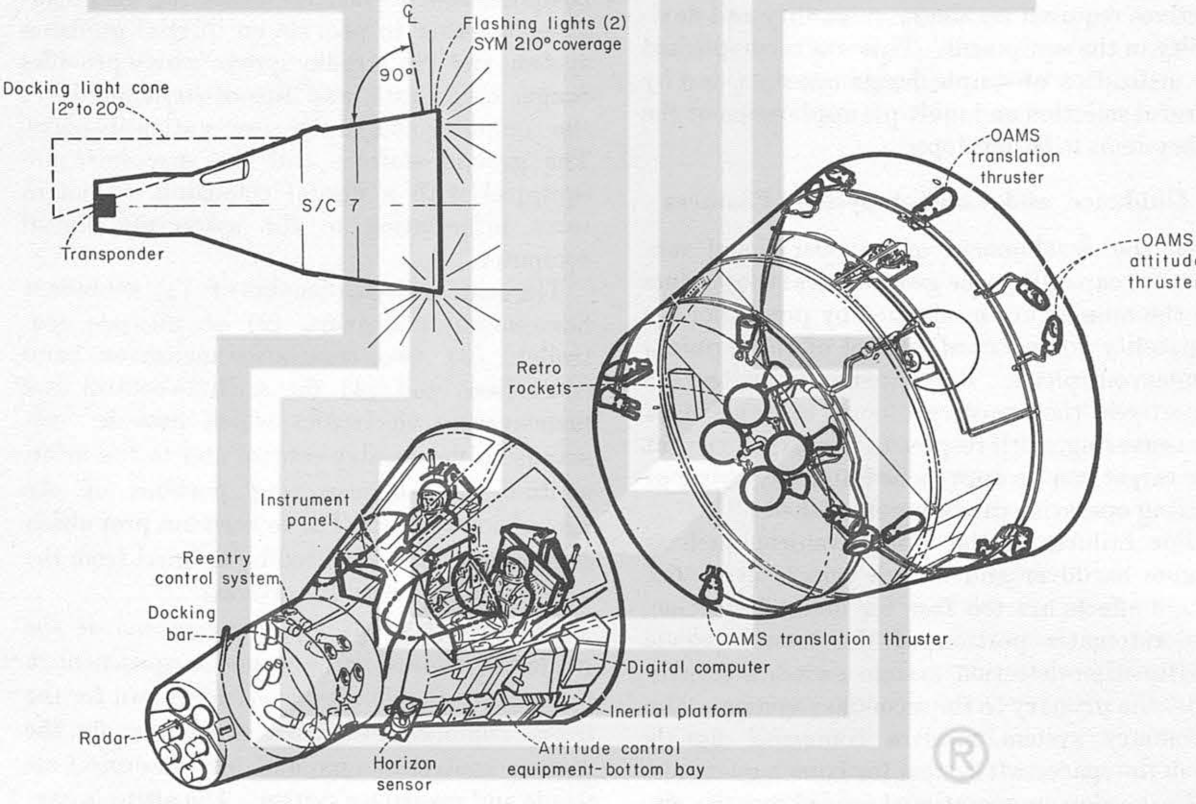


Figure 4–2.—Arrangement of guidance and control system components in the spacecraft.

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 back-up 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

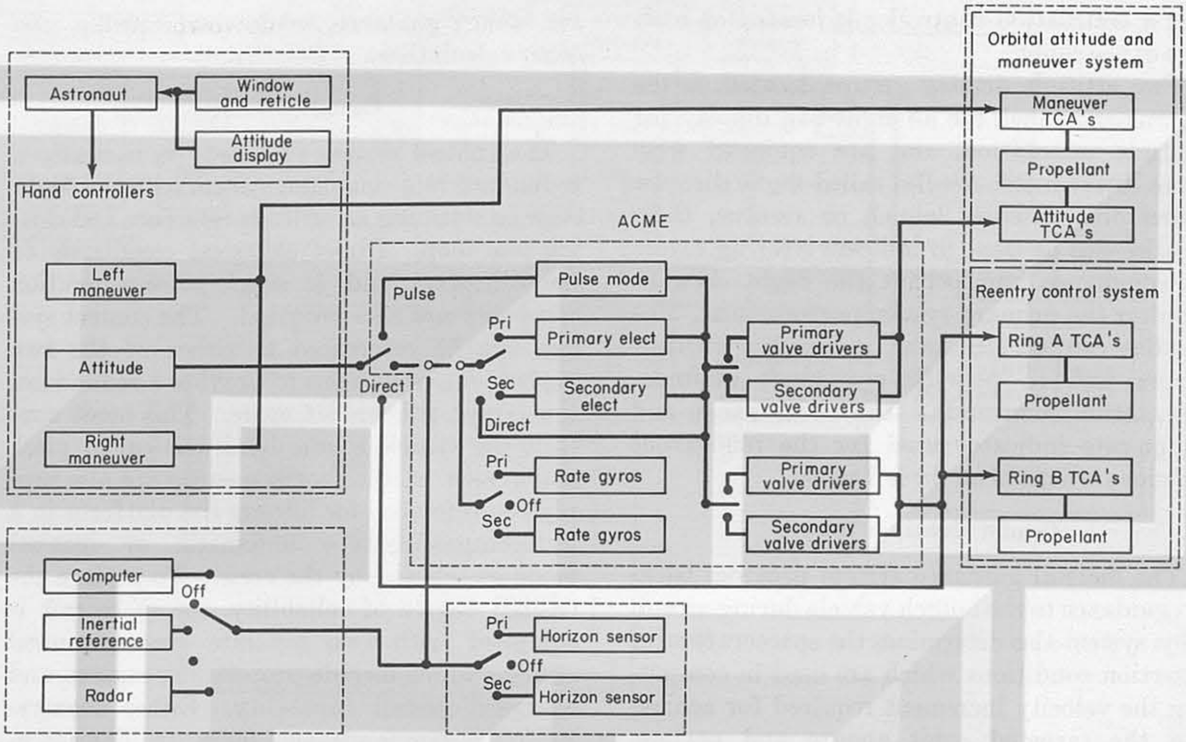
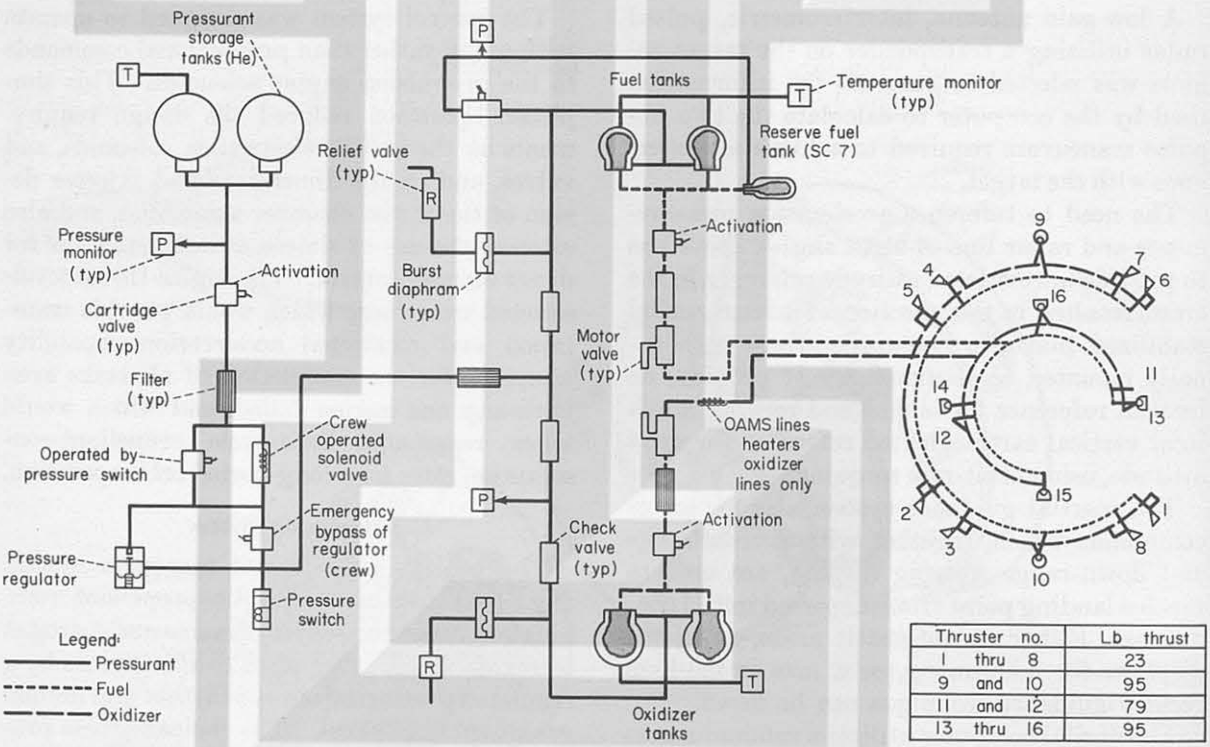


FIGURE 4-3.—Control system.



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			258
	2	79	180 000	710	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.

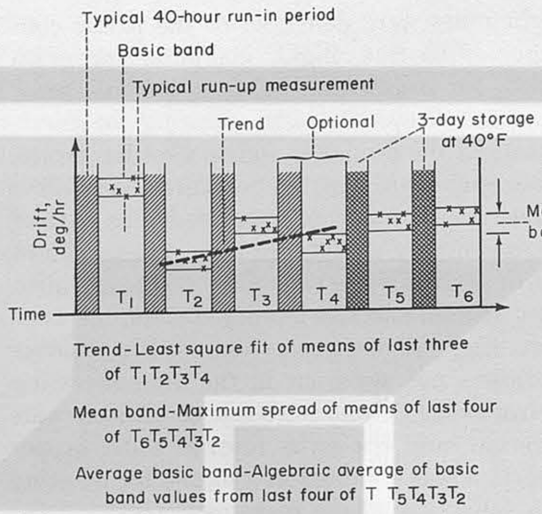


FIGURE 4–5.—Gyro test procedure.

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.

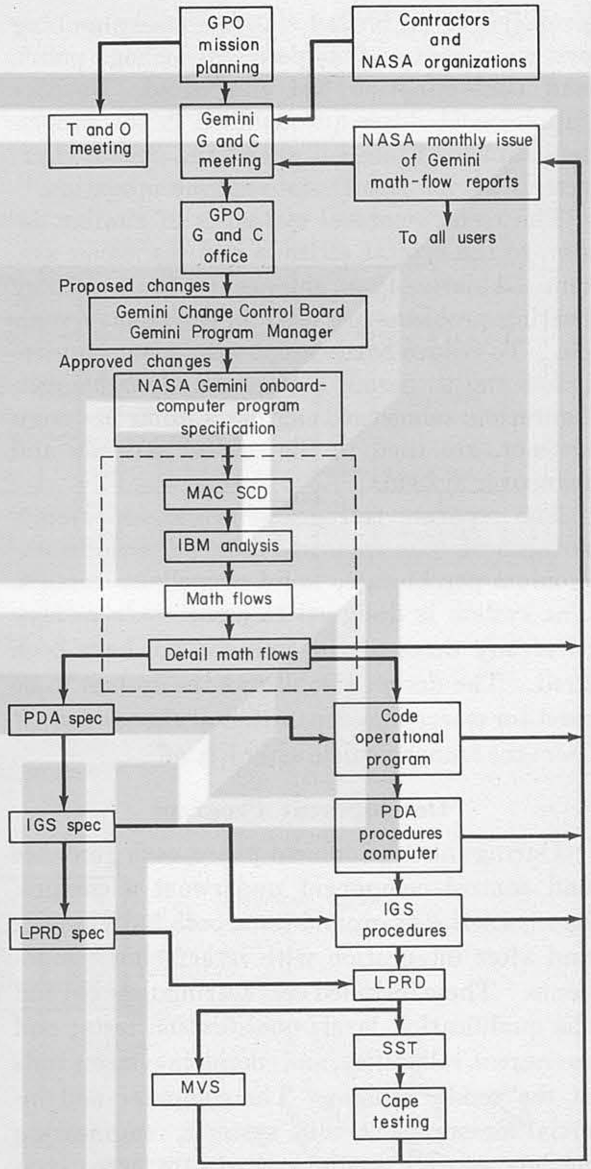


FIGURE 4–6.—Math flow control procedures and required intermediate goals.

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 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-

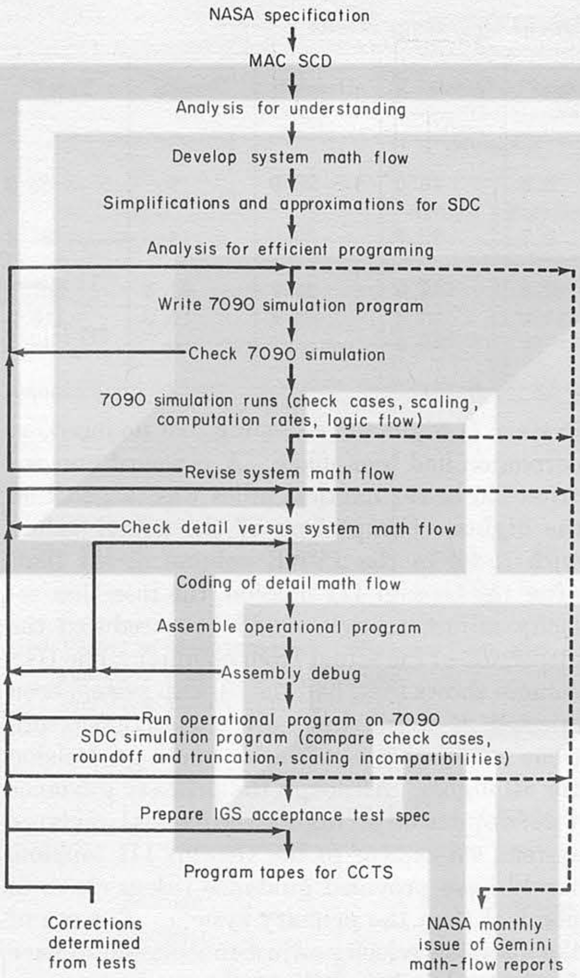


FIGURE 4-7.—Required intermediate goals in math flow development.

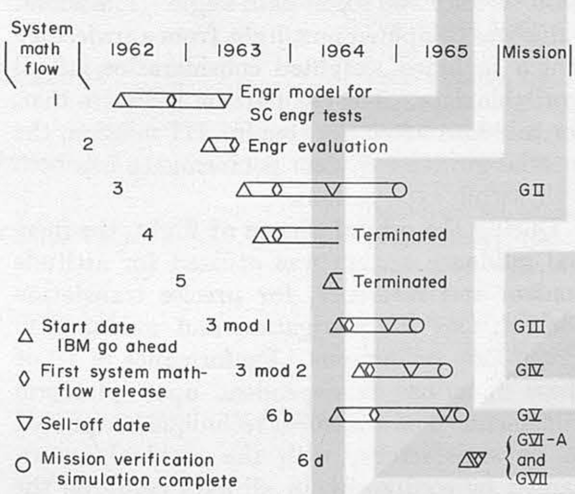


FIGURE 4-8.—Computer program development status chart.

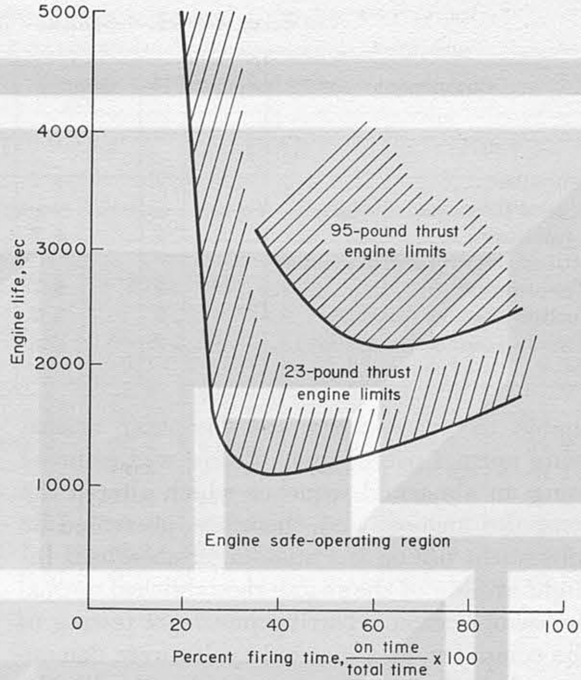


FIGURE 4-9.—Engine firing capability.

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.

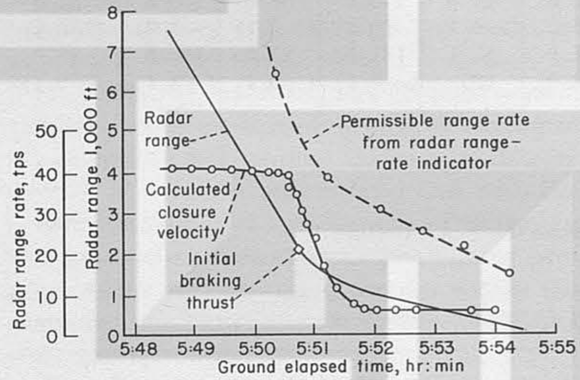


FIGURE 4–10.—Radar trajectory range comparison for Gemini VI–A and VII rendezvous.

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.

5. COMMUNICATIONS AND INSTRUMENTATION

By Clifford M. Jackson, *Gemini Program Office, NASA Manned Spacecraft Center;* Andrew Hoboken, *Office of Resident Manager, Gemini Program Office, McDonnell Aircraft Corp.;* John W. Goad, Jr., *Gemini Program Office, NASA Manned Spacecraft Center; and* Meredith W. Hamilton, *Instrumentation and Electronic Systems Division, NASA Manned Spacecraft Center*

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

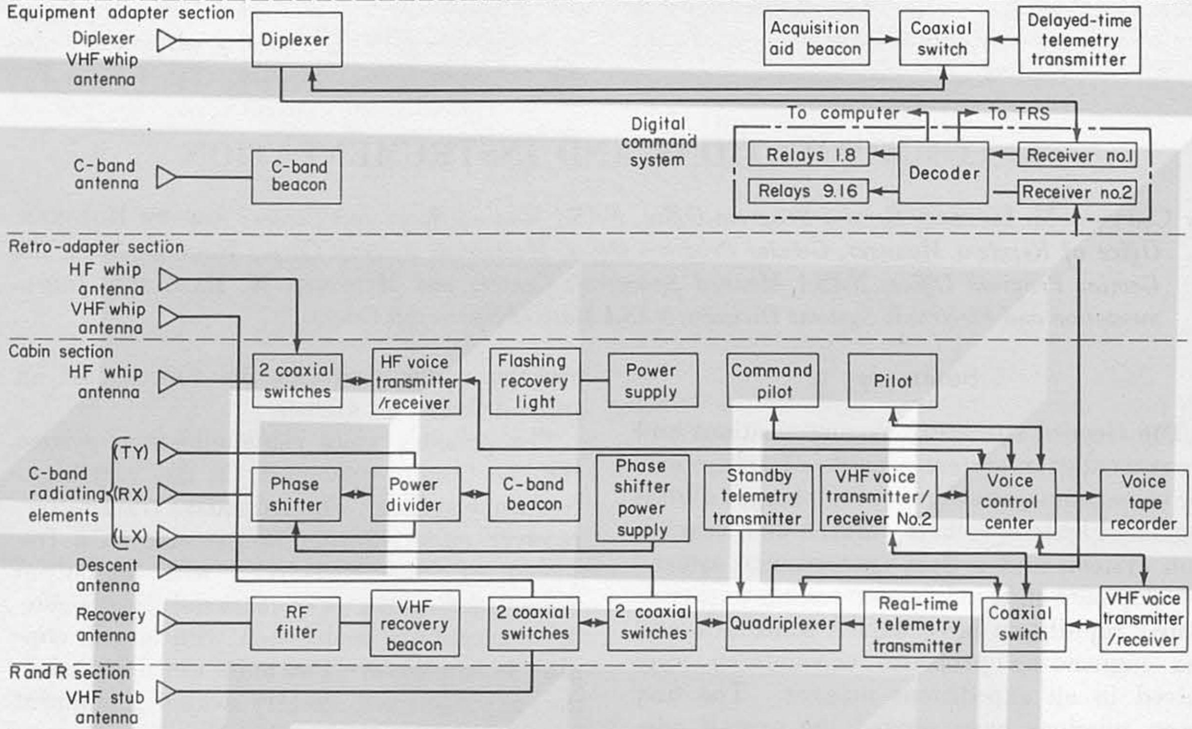


FIGURE 5-1.—Communications system.

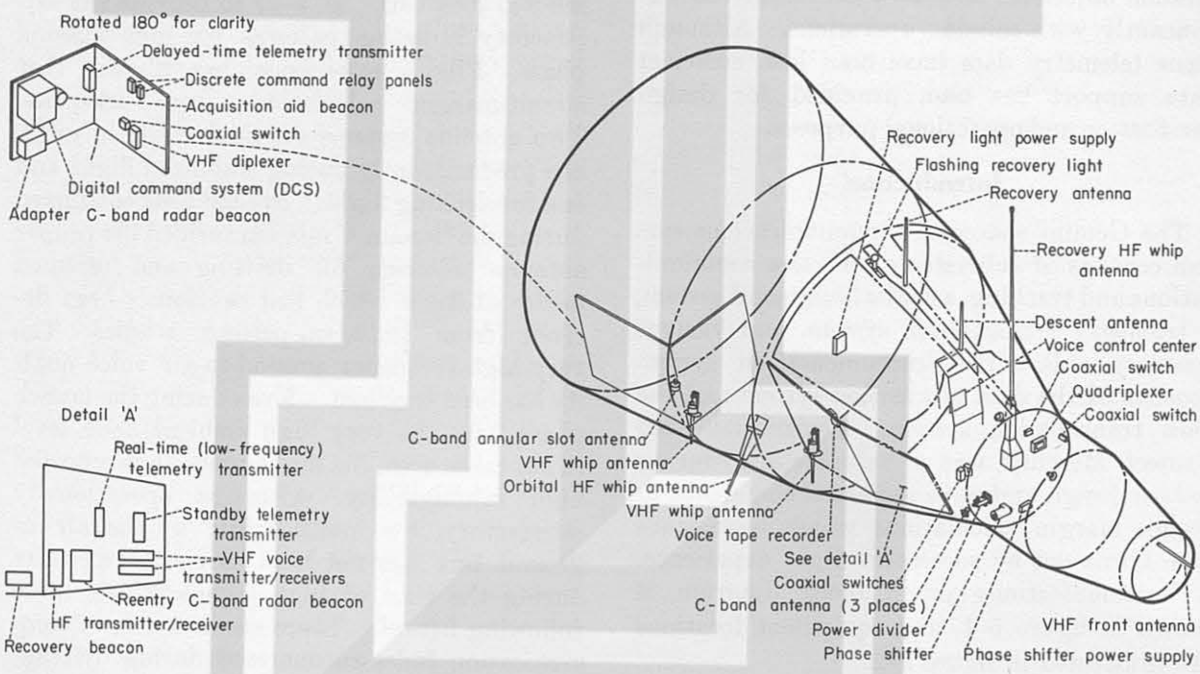


FIGURE 5-2.—Location of Gemini spacecraft communications equipment.

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.

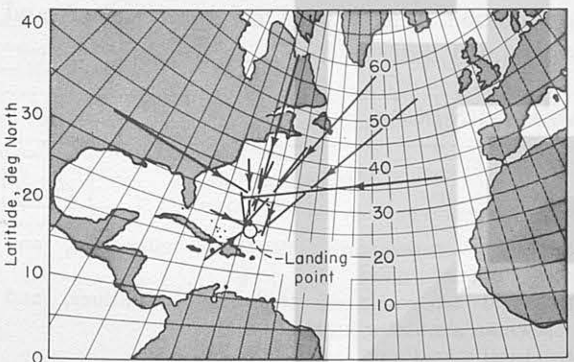


FIGURE 5–3.—HF–DF bearings to Gemini VI–A after landing.

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 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.

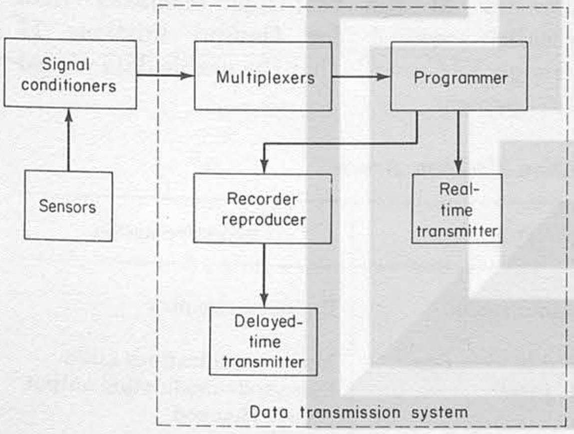


FIGURE 5–4.—Block diagram of the instrumentation system.

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)

6. ELECTRICAL POWER AND SEQUENTIAL SYSTEMS

By Percy Miglicco, *Gemini Program Office, Manned Spacecraft Center;* Robert Cohen, *Gemini Program Office, Manned Spacecraft Center; and* Jesse Deming, *Gemini Program Office, Manned Spacecraft Center*

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.

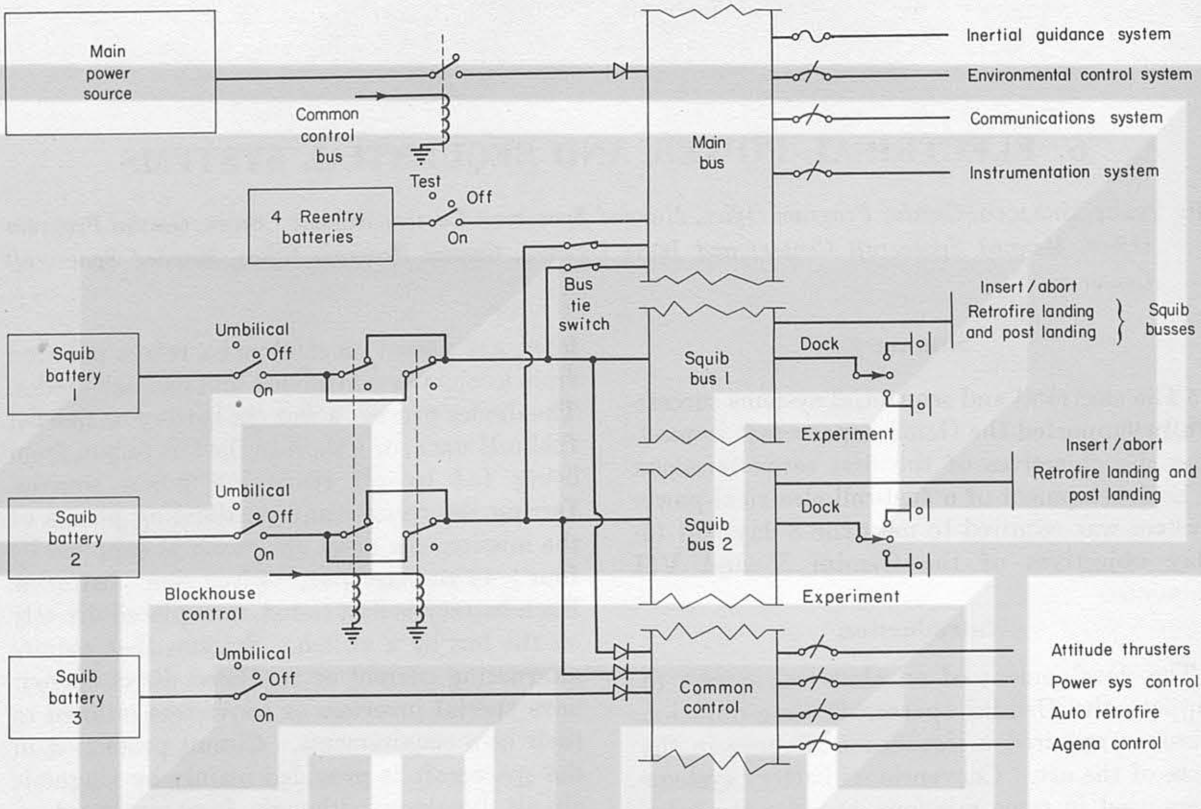


FIGURE 6-1.—Gemini electrical system.

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.

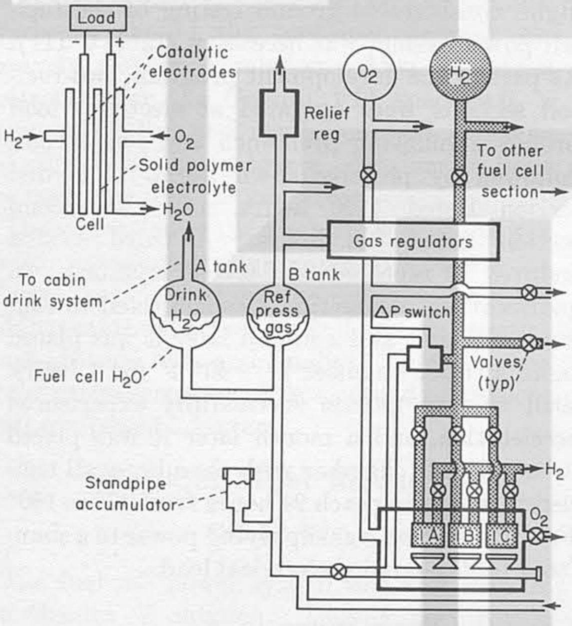


FIGURE 6-2.—Spacecraft 7 fuel-cell/RSS fluid schematic.

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

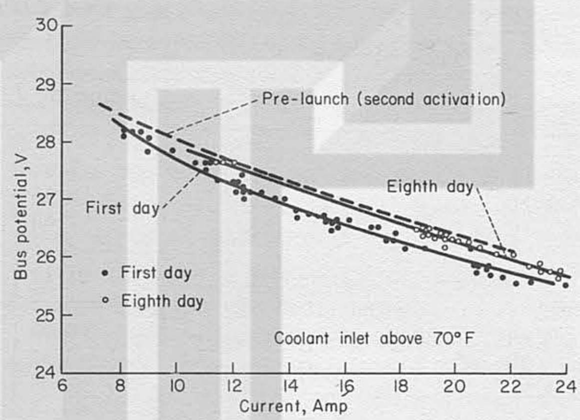


FIGURE 6–3.—Fuel-cell section 1 performance for the Gemini V mission.

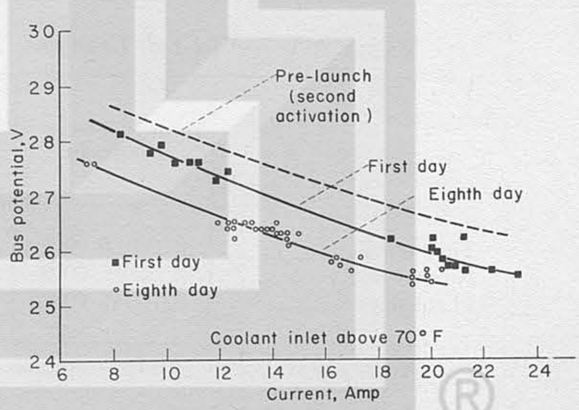


FIGURE 6–4.—Fuel-cell section 2 performance for the Gemini V mission.

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

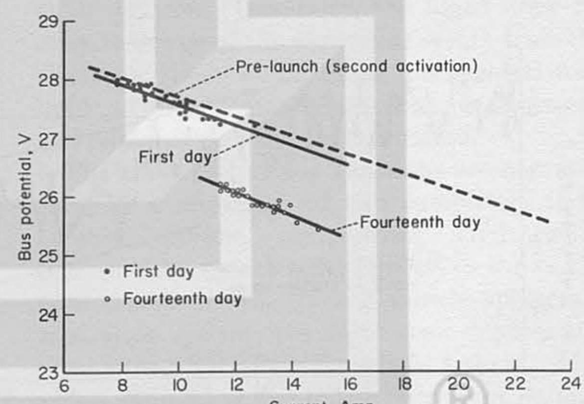


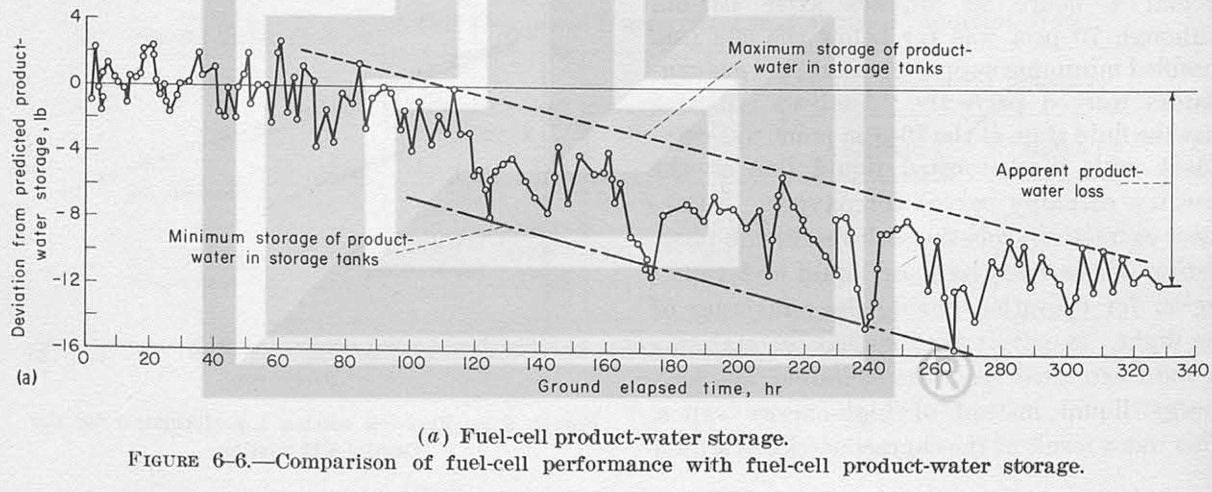
FIGURE 6–5.—Fuel-cell section 1 performance for the Gemini VII mission.

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.

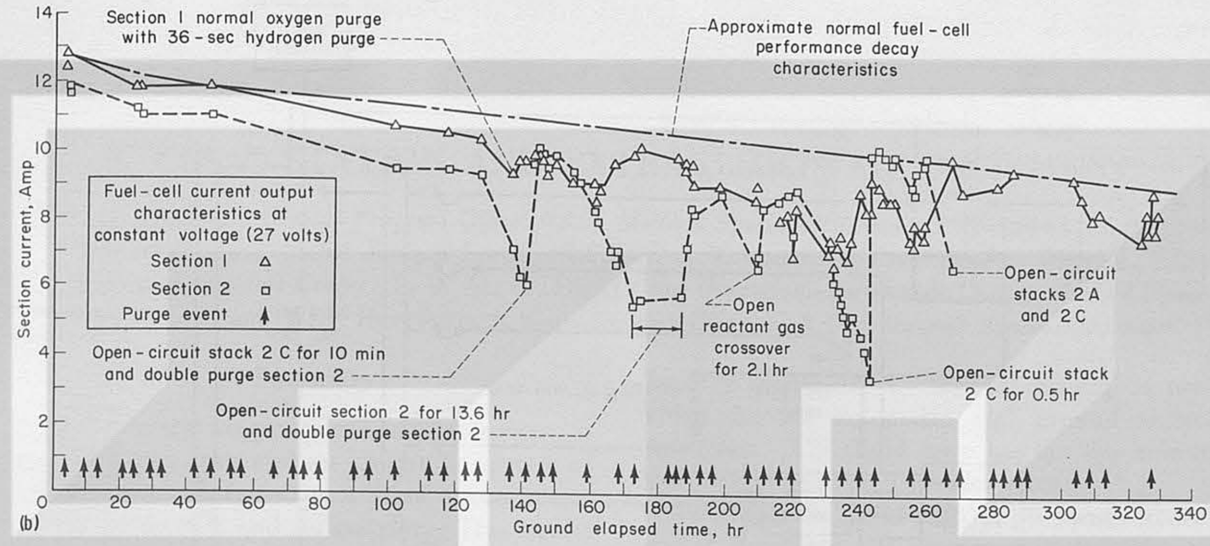
Figures 6–6(a) and 6–6(b) show the deviations in product-water storage with the performance of the fuel-cell sections as a function of time from lift-off. Between 100 and 265 hours after lift-off, a maximum storage fluctuation of 8 pounds occurred around the gradual storage reduction. The gradual storage 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.

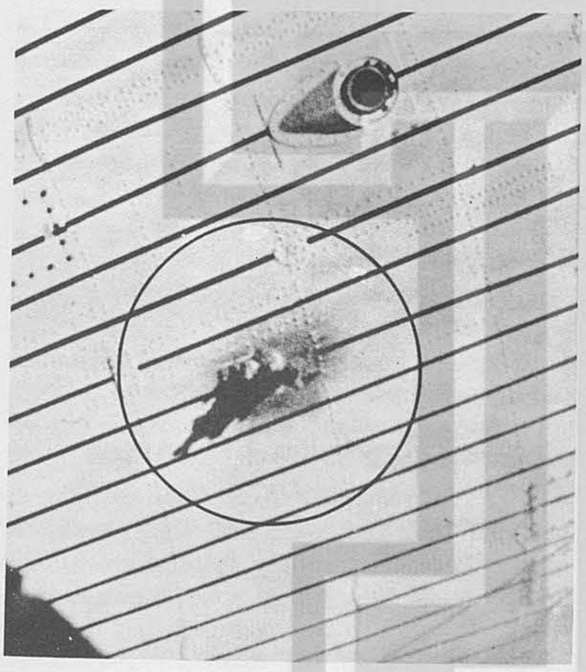


FIGURE 6–7.—Ice formation at hydrogen vent.

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

218–556 O—66——5

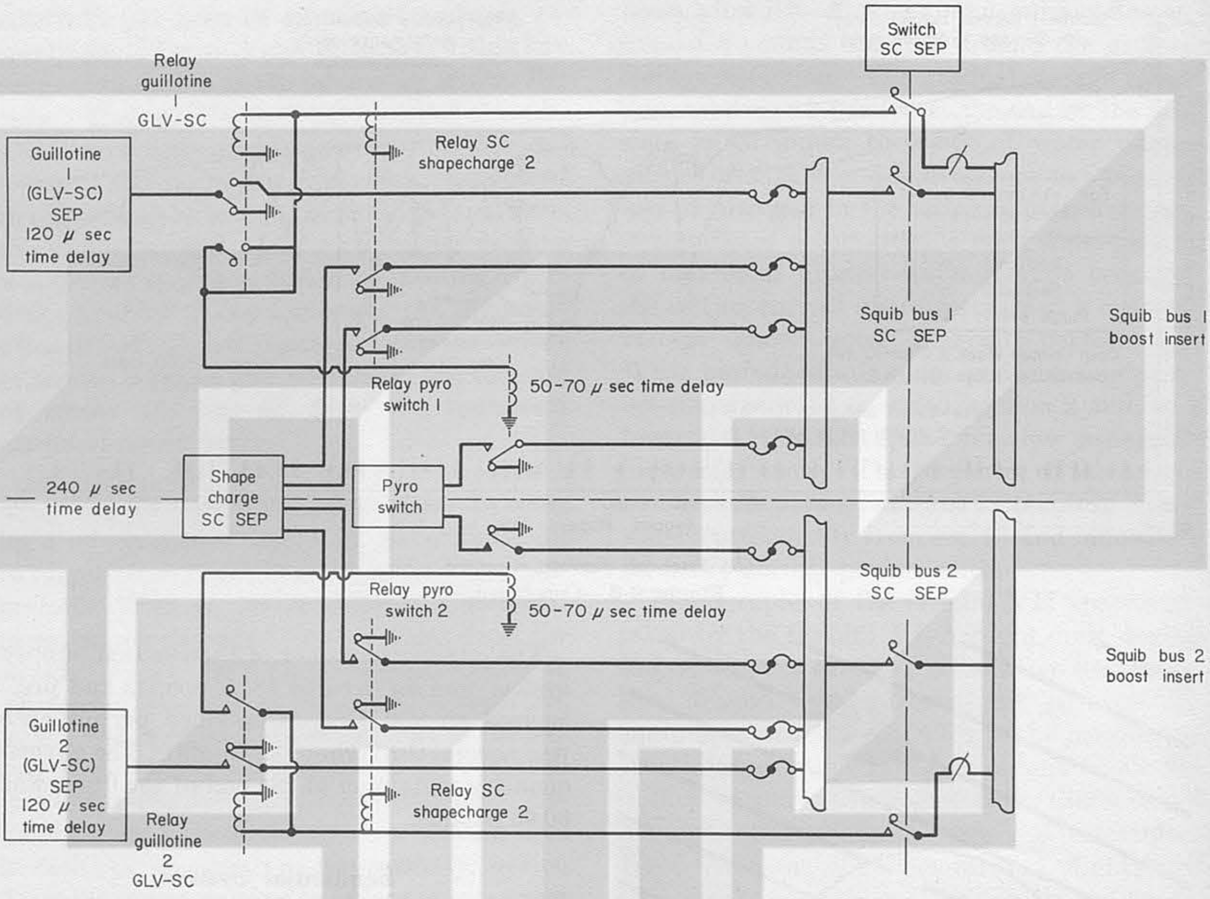


FIGURE 6-8.—Launch-vehicle–spacecraft separation circuitry.

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

By R. M. MACHELL, *Gemini Program Office, NASA Manned Spacecraft Center;* J. C. SHOWS, *Flight Crew Support Division, NASA Manned Spacecraft Center;* J. V. CORREALE, *Crew Systems Division, NASA Manned Spacecraft Center;* L. D. ALLEN, *Flight Crew Operations Directorate, NASA Manned Spacecraft Center; and* W. J. HUFFSTETLER, *Crew Systems Division, NASA Manned Spacecraft Center*

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.

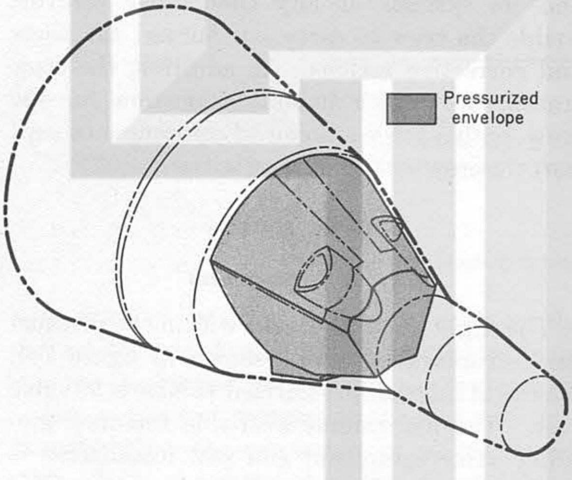


FIGURE 7–1.—Crew-station pressure vessel.

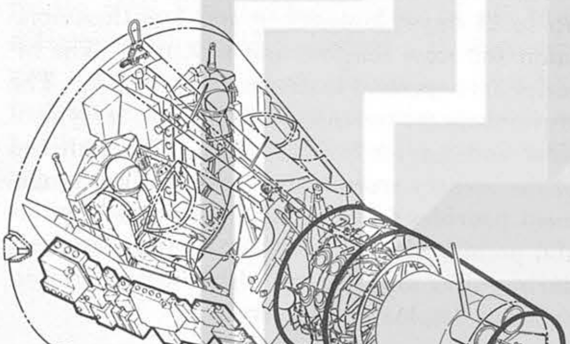


FIGURE 7–2.—Crew-station interior arrangement.

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.

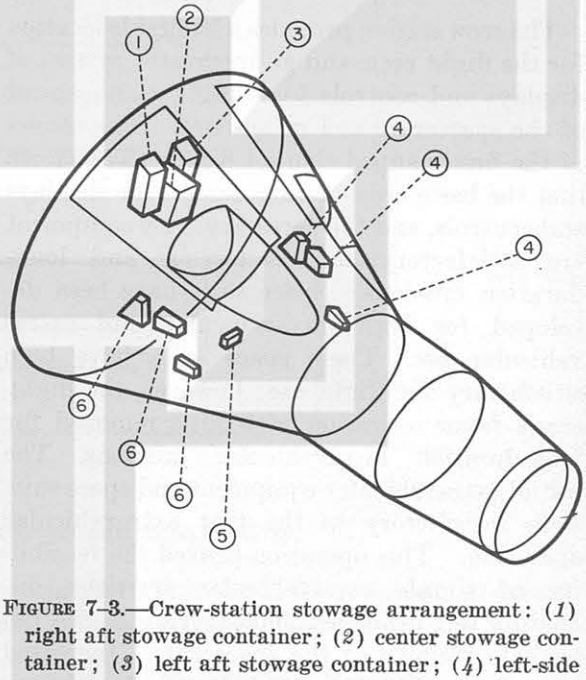


FIGURE 7–3.—Crew-station stowage arrangement: (*1*) right aft stowage container; (*2*) center stowage container; (*3*) left aft stowage container; (*4*) left-side stowage containers; (*5*) orbital utility pouch (under right instrument panel); (*6*) right-side stowage containers.

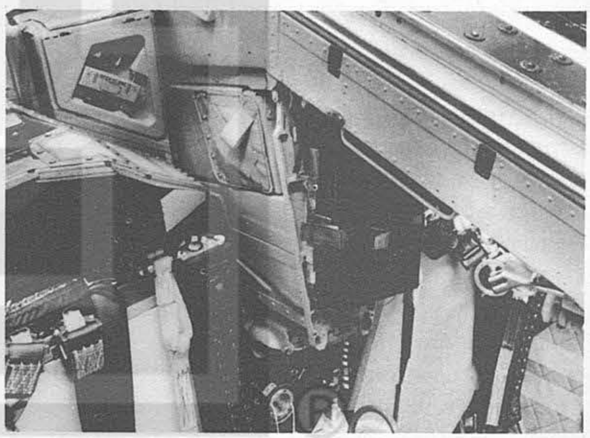


FIGURE 7–4.—Spacecraft center and right-aft stowage containers (viewed from right side looking aft).

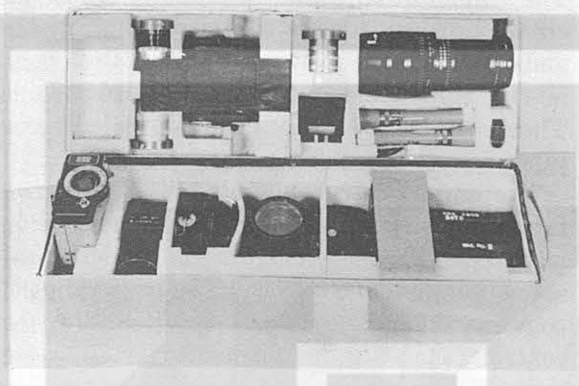


FIGURE 7-5.—Stowage of equipment in center stowage box.

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):

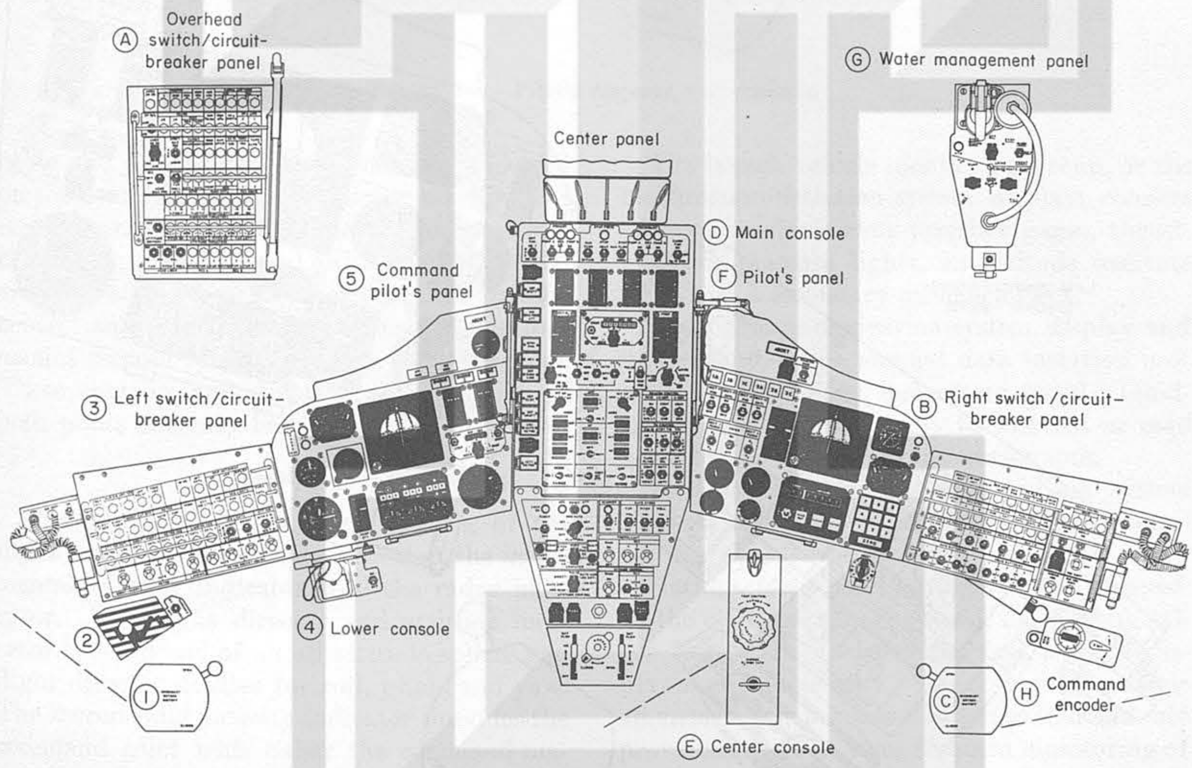


FIGURE 7-6.—Spacecraft instrument panel: (*1*) secondary oxygen shut-off (l.h.); (*2*) abort handle; (*3*) left switch/circuit-breaker panel; (*4*) lower console; (*5*) command pilot's panel; (*A*) overhead switch/circuit-breaker panel; (*B*) right switch/circuit-breaker panel; (*C*) secondary oxygen 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

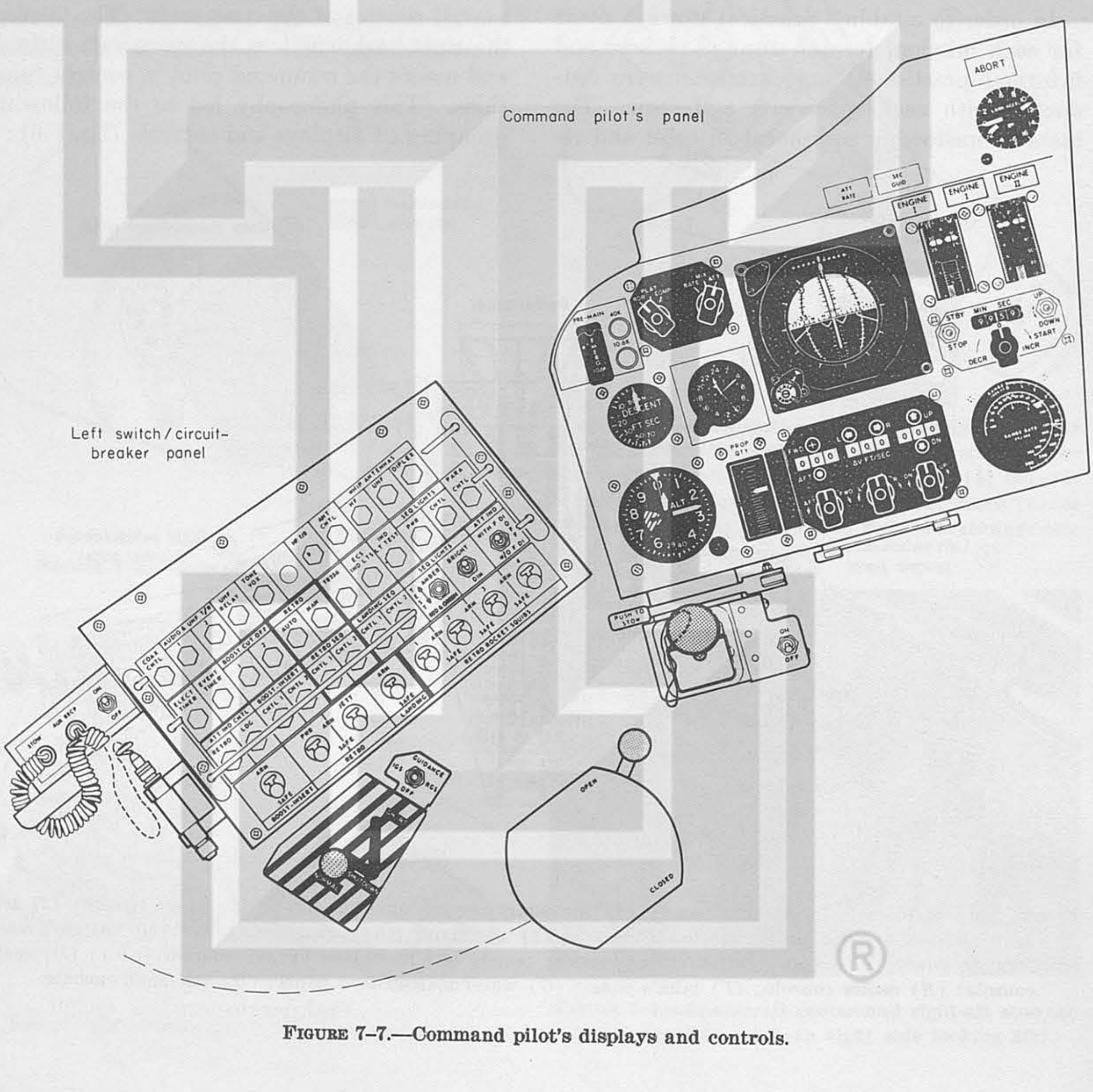


FIGURE 7-7.—Command pilot's displays and controls.

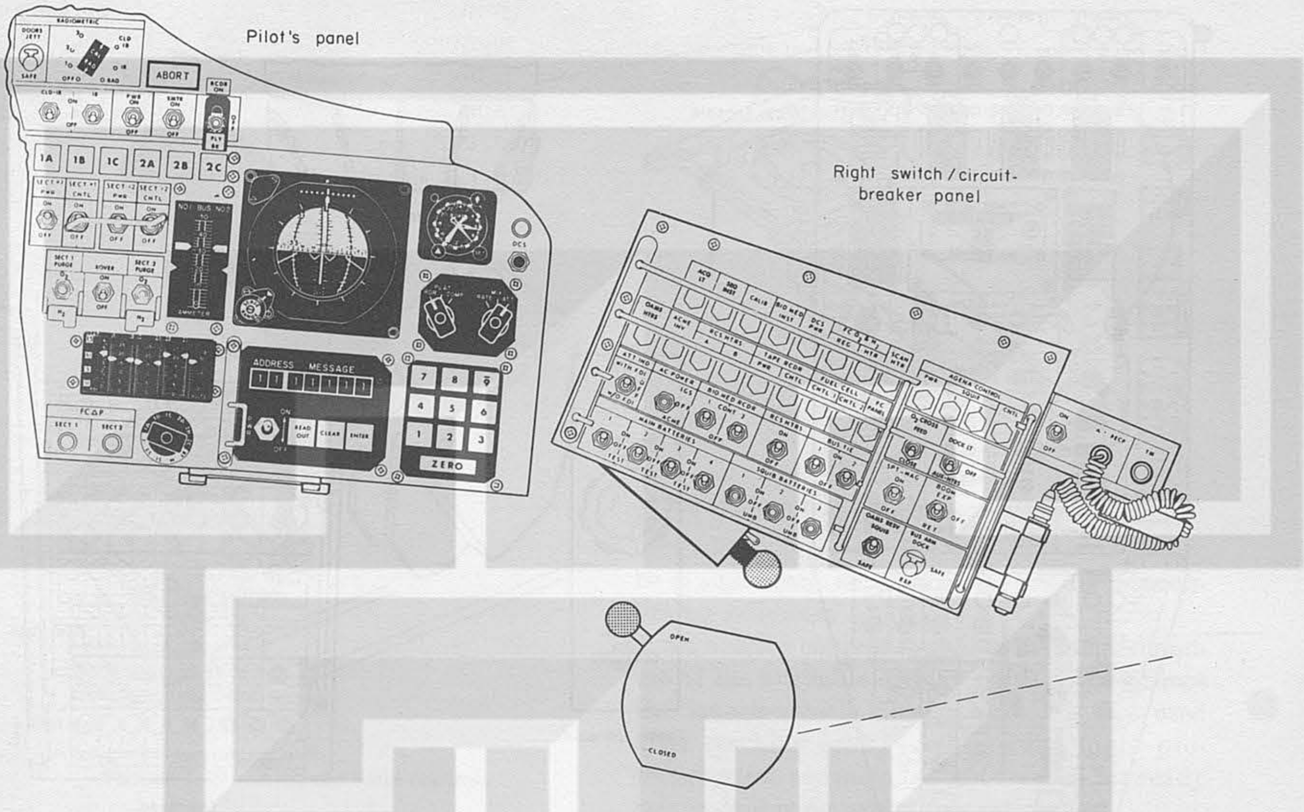


FIGURE 7-8.—Pilot's displays and controls.

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.

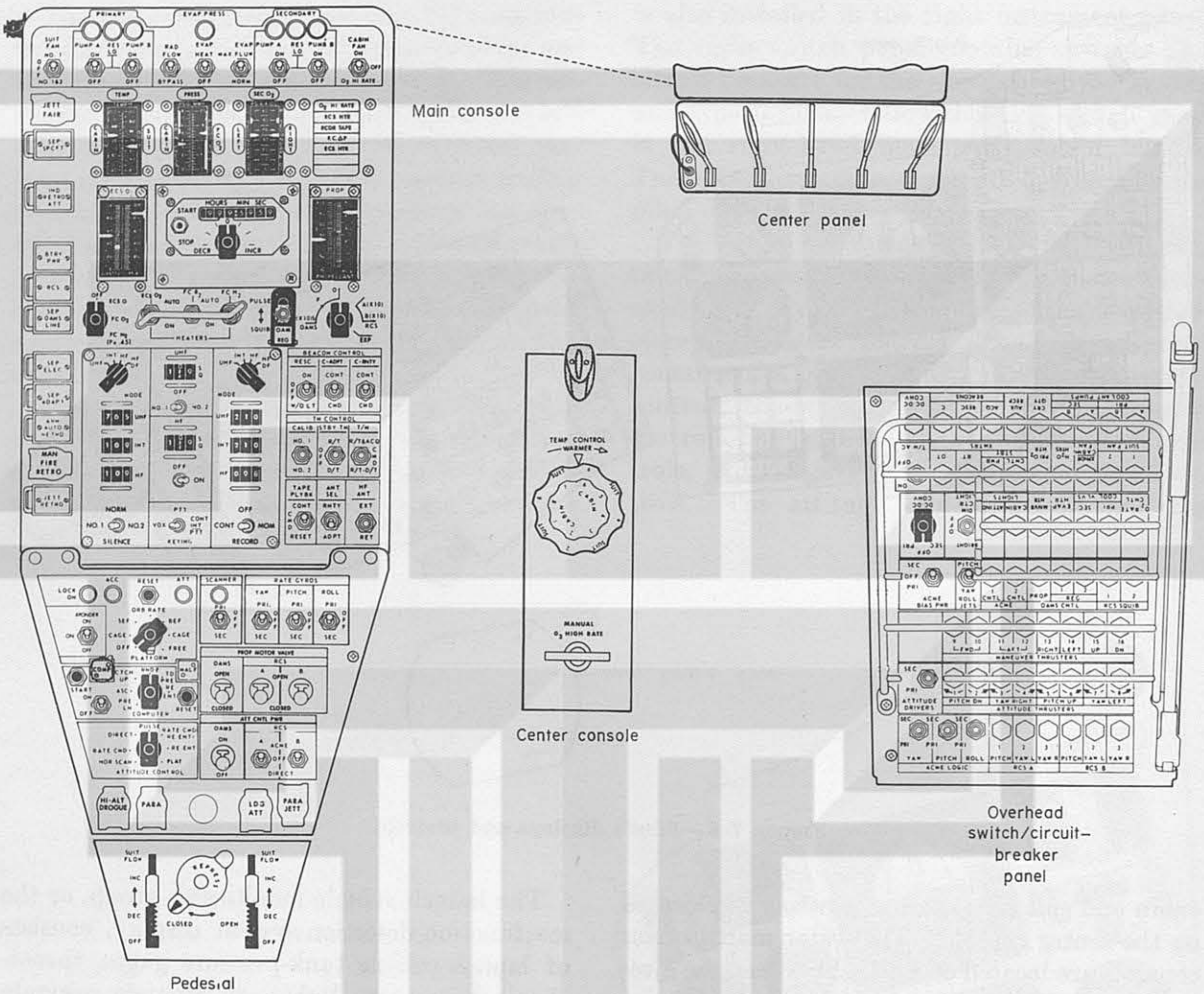


FIGURE 7-9.—Displays and controls used by both sides.

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

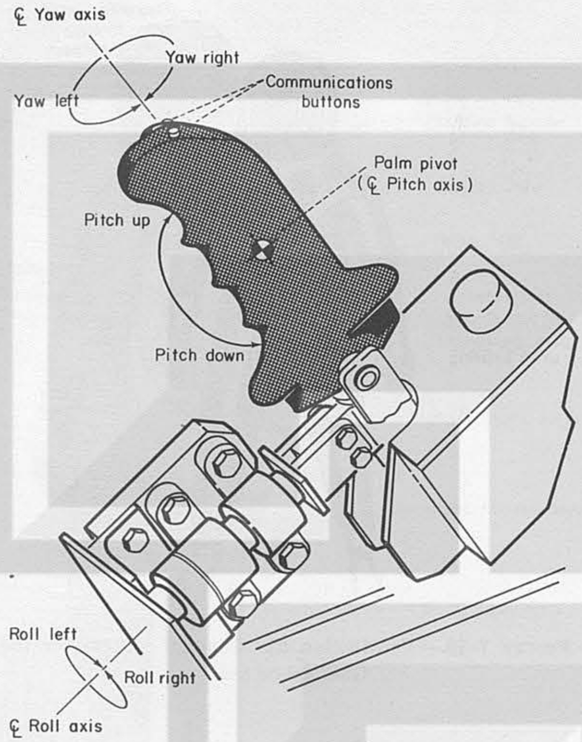


FIGURE 7-10.—Attitude hand control.

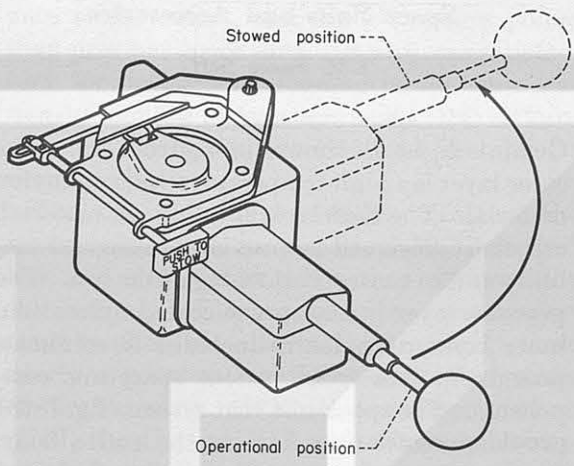


FIGURE 7-11.—Maneuver hand control.

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 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 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.

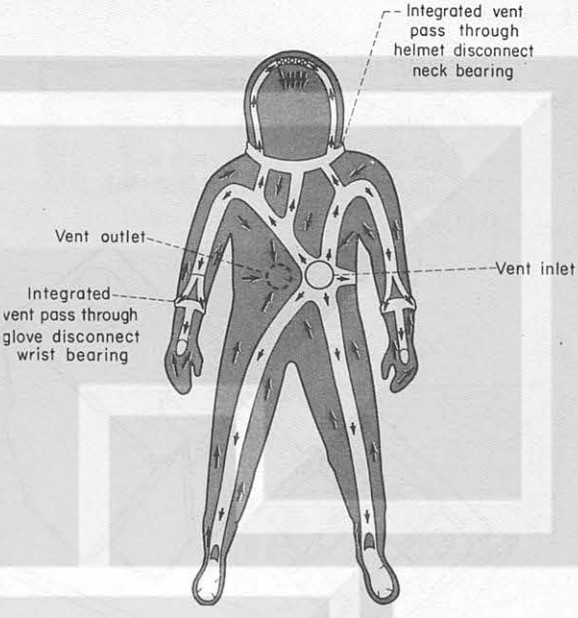


FIGURE 7–13.—Ventilation distribution system for the G3C space suit.

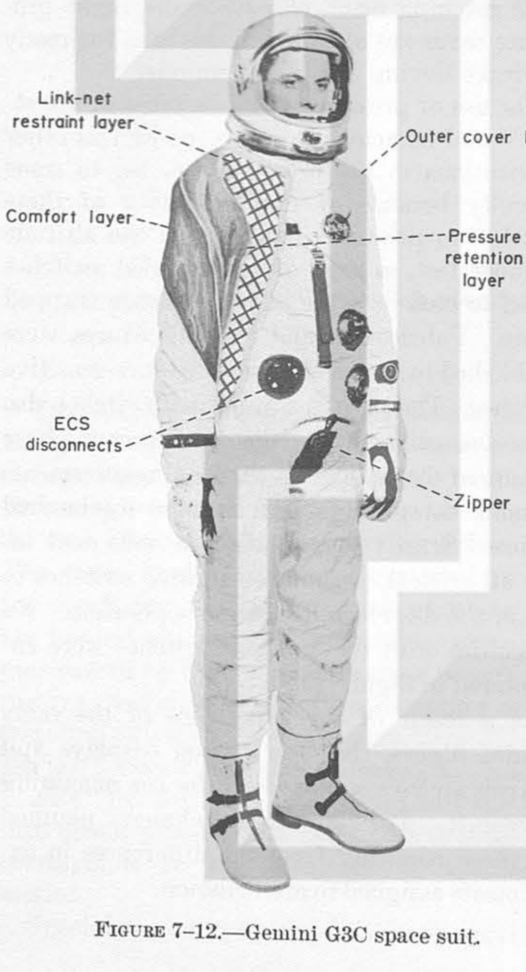


FIGURE 7–12.—Gemini G3C space suit.

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

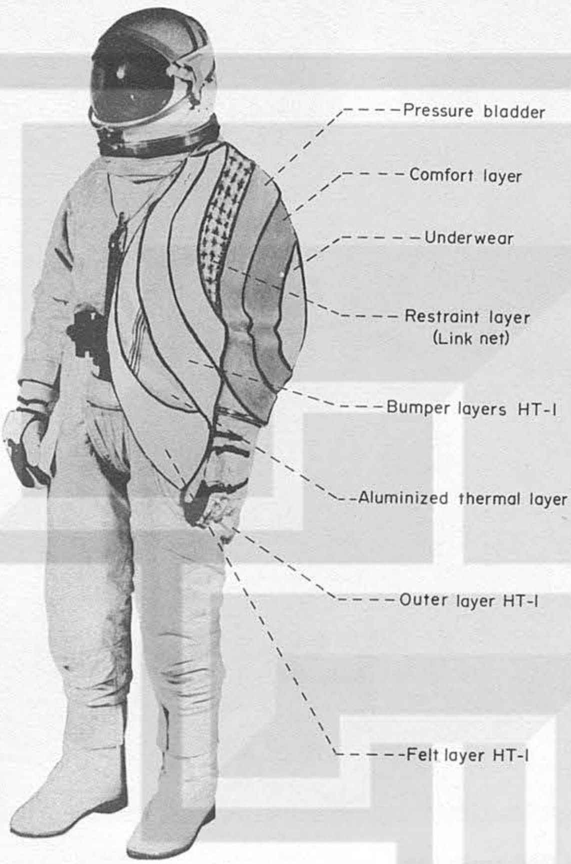


FIGURE 7-14.—Gemini extravehicular space suit.

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.

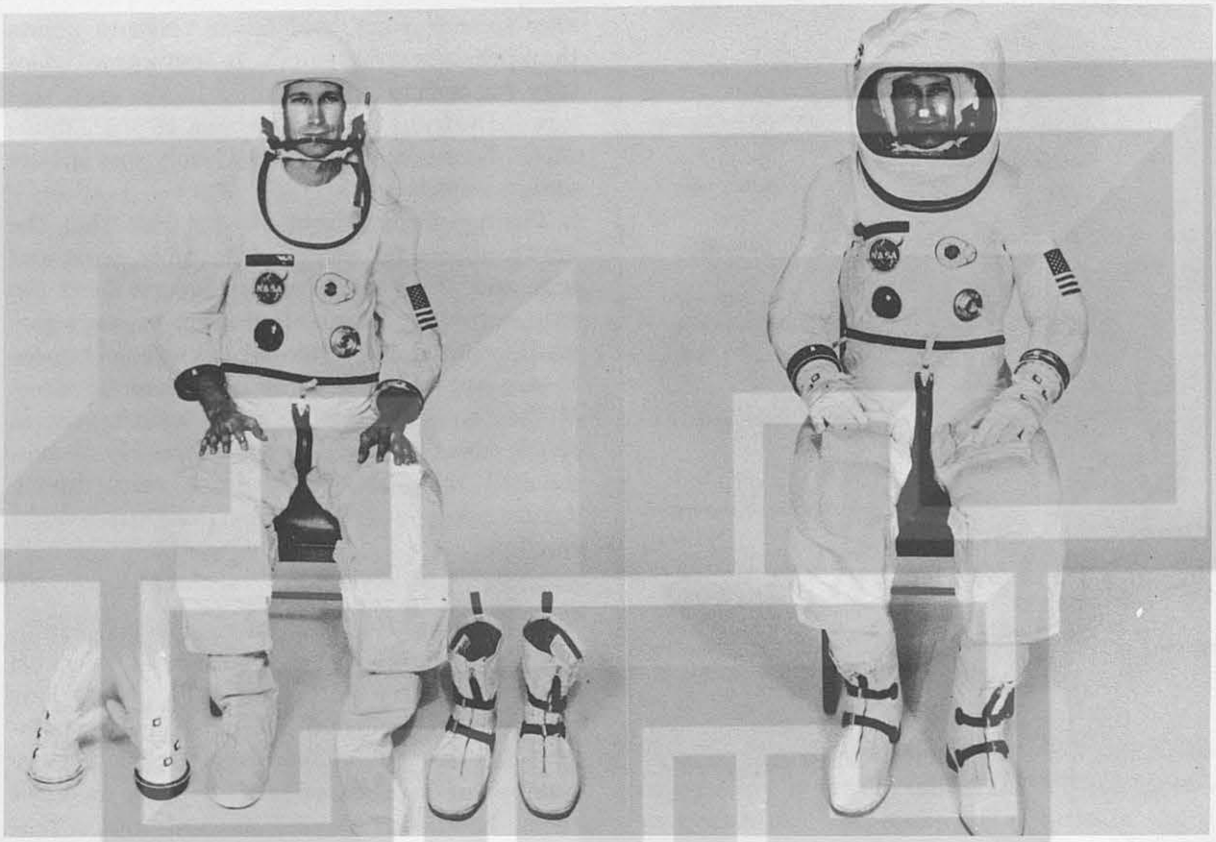


FIGURE 7-15.—Gemini G5C space suit.

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.

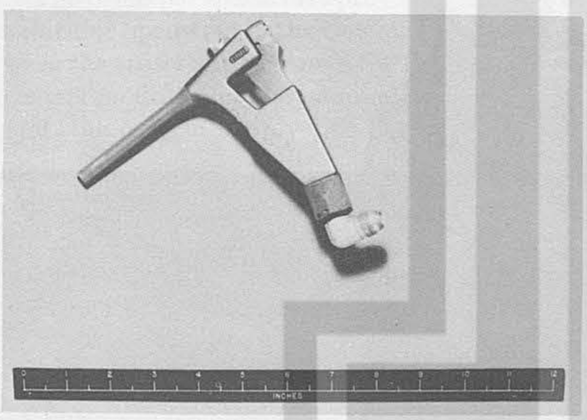


FIGURE 7-17.—Original Gemini water dispenser.

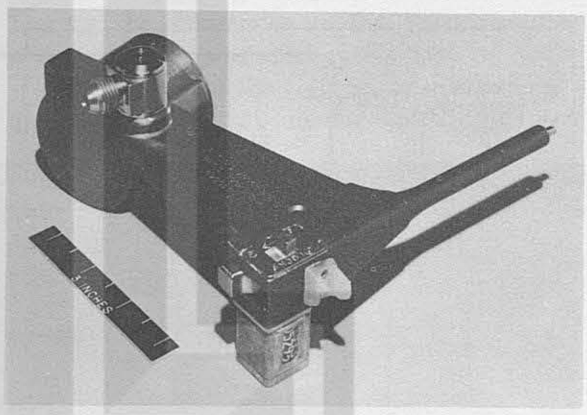


FIGURE 7-18.—Gemini water-metering device.

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

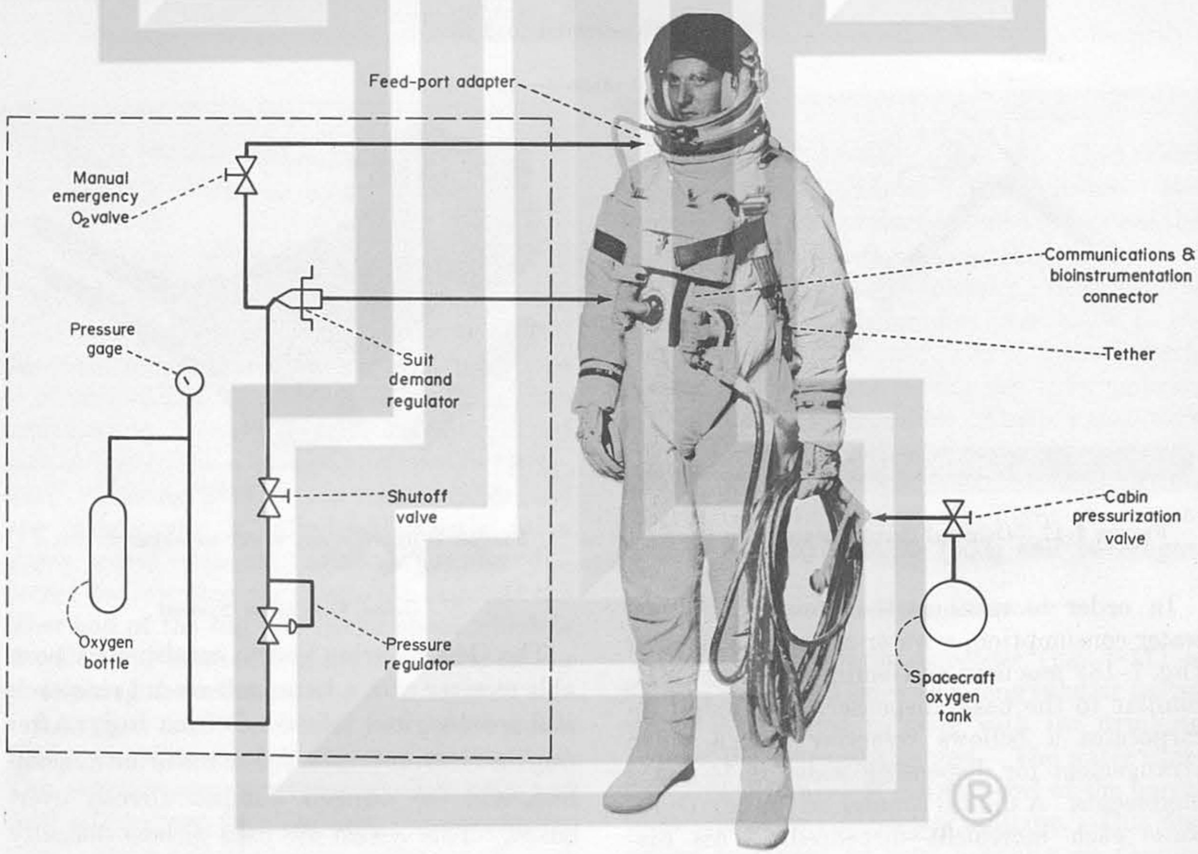


FIGURE 7–19.—Gemini IV extravehicular life-support system.

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

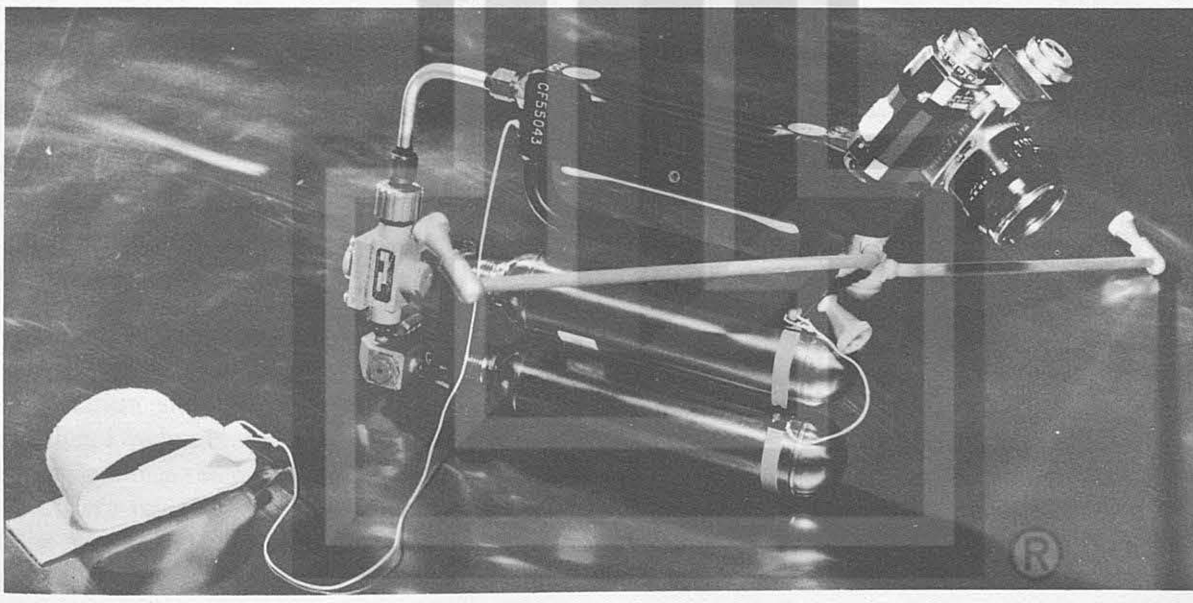


FIGURE 7–20.—Handheld maneuvering unit.

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.

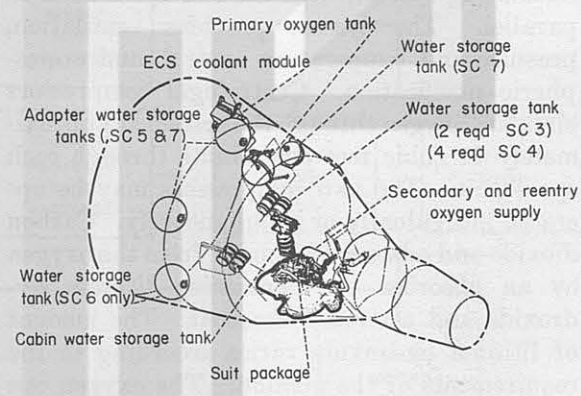


FIGURE 8–1.—Environmental control system.

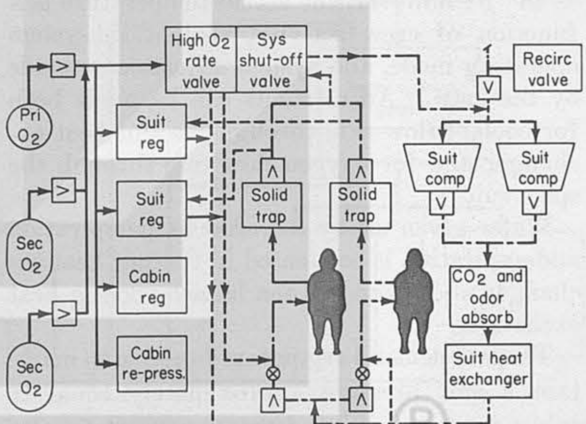


FIGURE 8–2.—Suit subsystem.

218-556 O—66——6

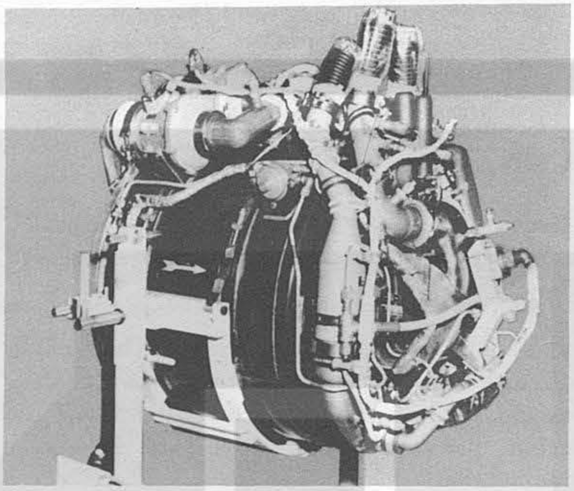


FIGURE 8-3.—Environmental control system suit subsystem module.

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

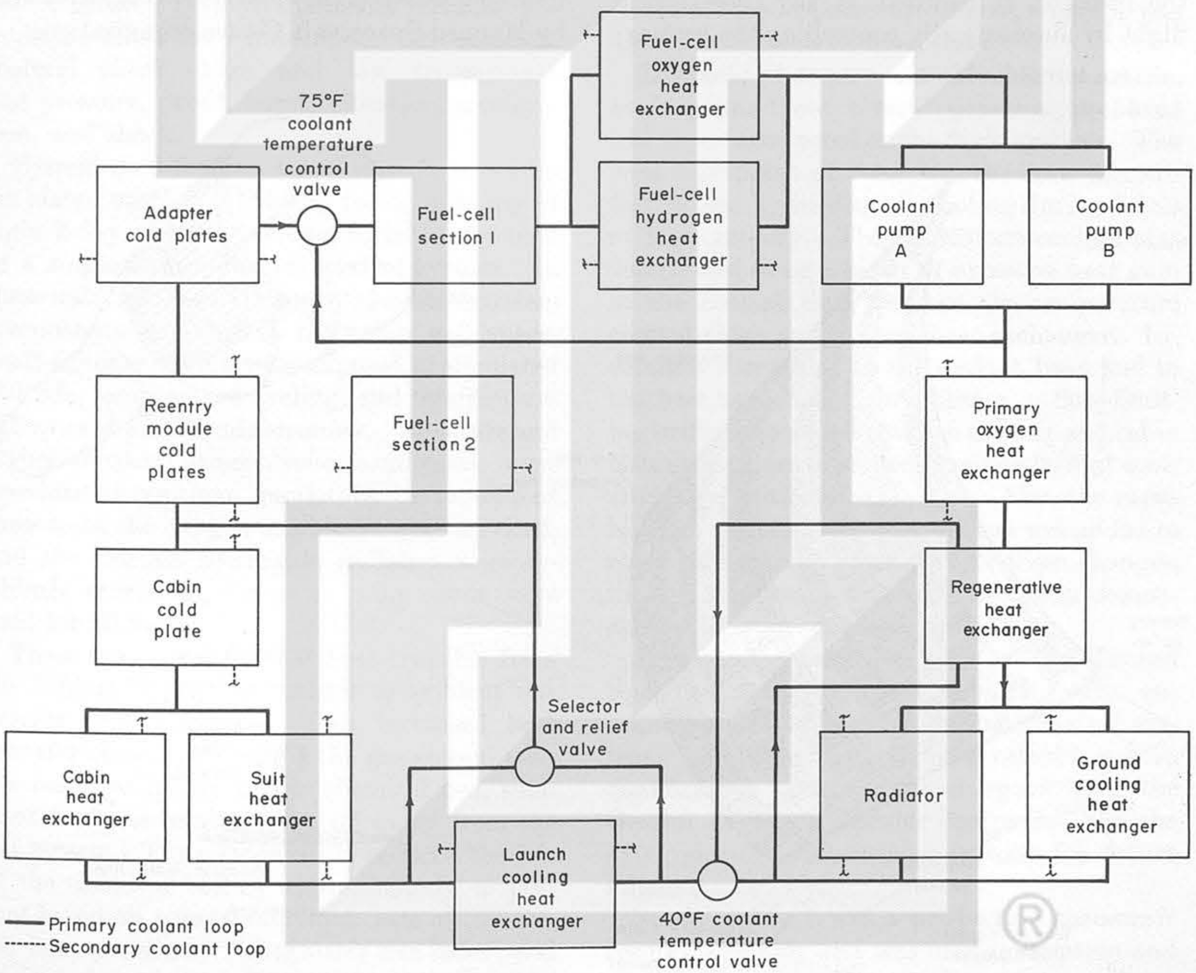


FIGURE 8-4.—Coolant subsystem.

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.

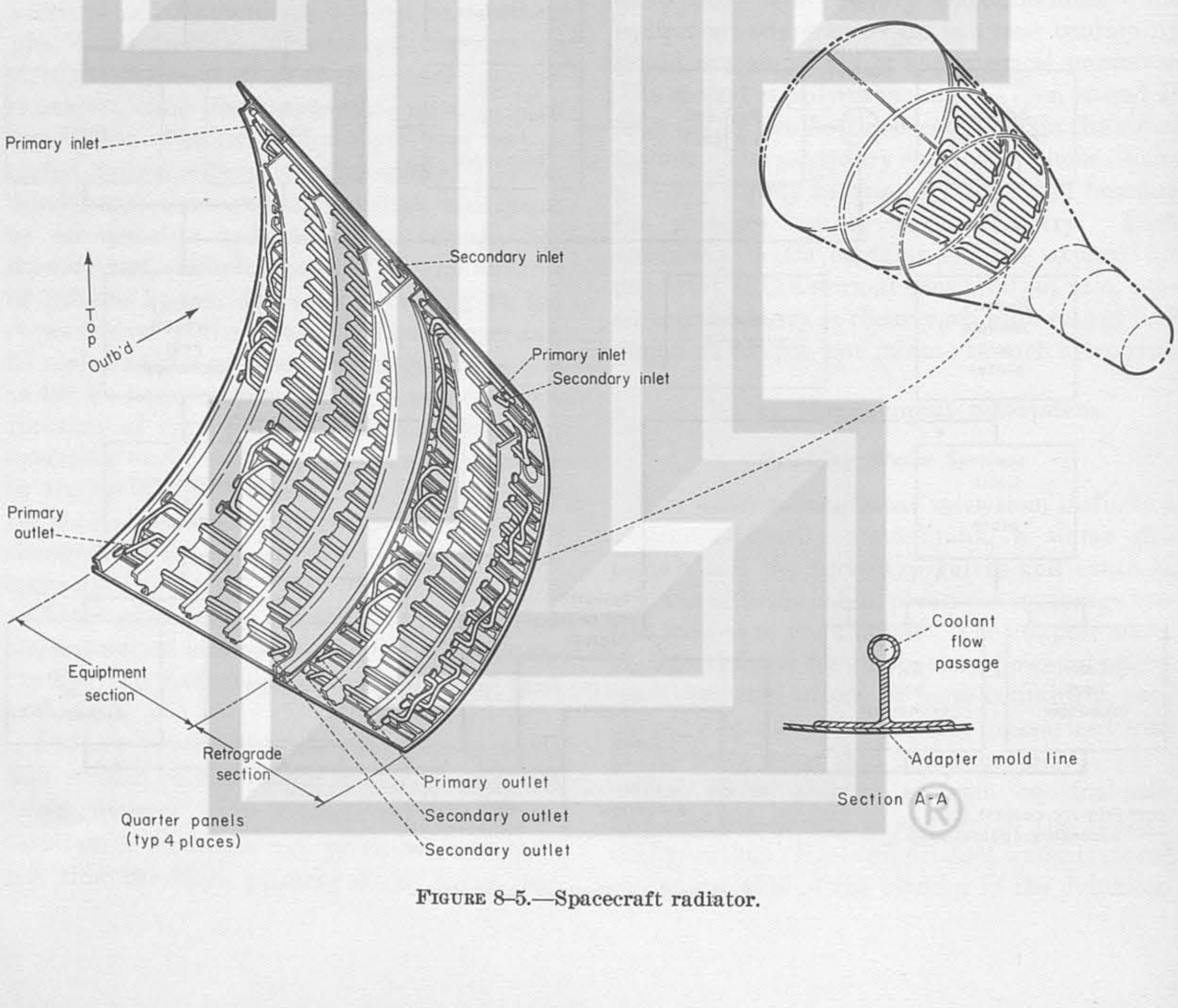


FIGURE 8–5.—Spacecraft radiator.

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

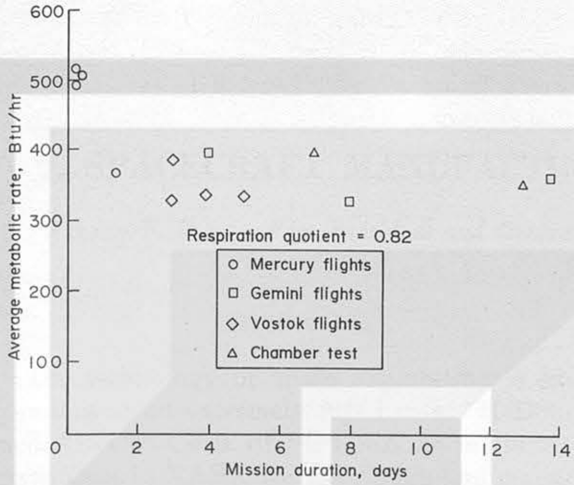


FIGURE 8–6.—Crewman metabolic rate.

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-

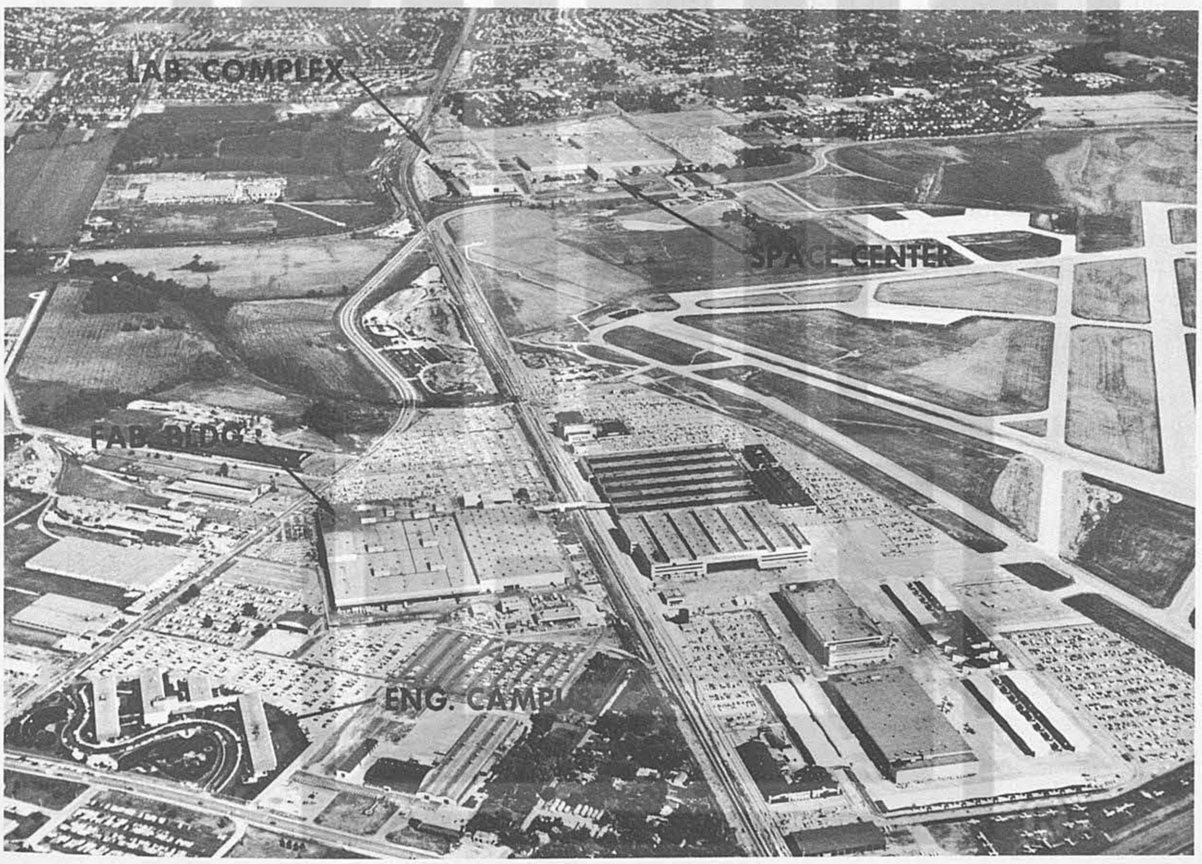


Figure 9–1.—McDonnell Aircraft Corp., St. Louis, Mo.

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

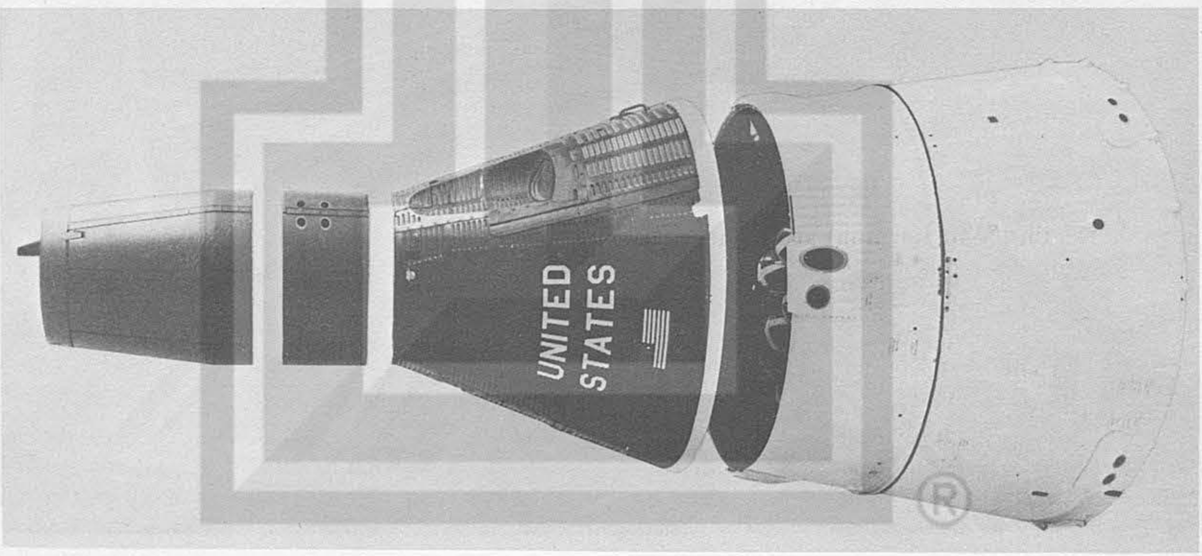


FIGURE 9-2.—Gemini spacecraft modular assembly.

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-

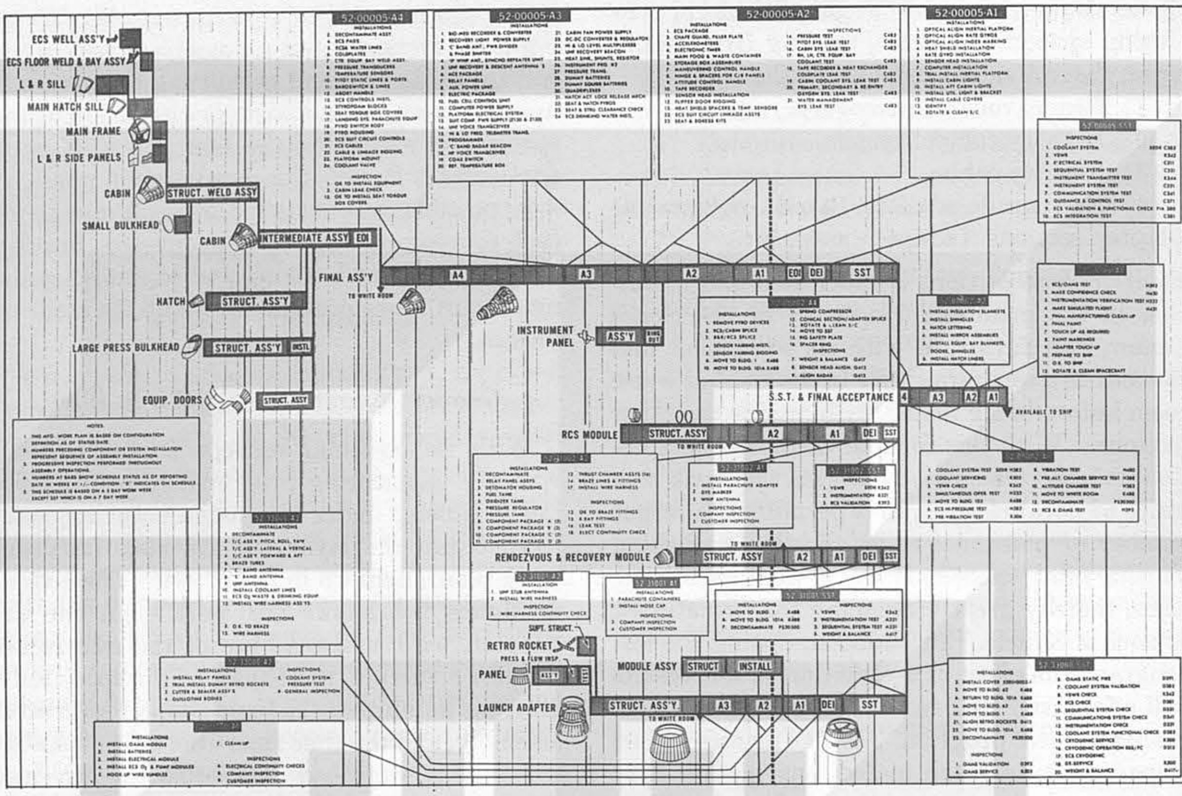


FIGURE 9-3.—Gemini spacecraft 4 manufacturing work plan.

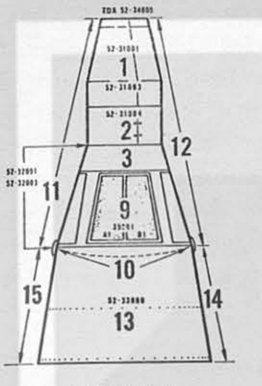
craft systems test at the module level. At this point in the manufacturing cycle, the three sections and the adapter assembly are assembled and the end-to-end spacecraft systems tests performed. From this manufacturing work plan, it can be seen that activities can be conducted on many zones of the spacecraft 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

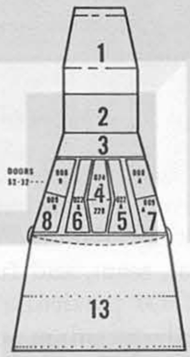
Manufacturing planners have the responsibility for determining the sequence in which individual installations are made. Obviously, this requires an evaluation of the time to make a particular installation and requires the assignment of the tasks to prevent delays due to interference between the production 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



TOP VIEW



BOTTOM VIEW

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
83700-723	MANUAL SHUT OFF VALVE	00005 A4	-8	00005 A2	+1	3
83700-277	ECS 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

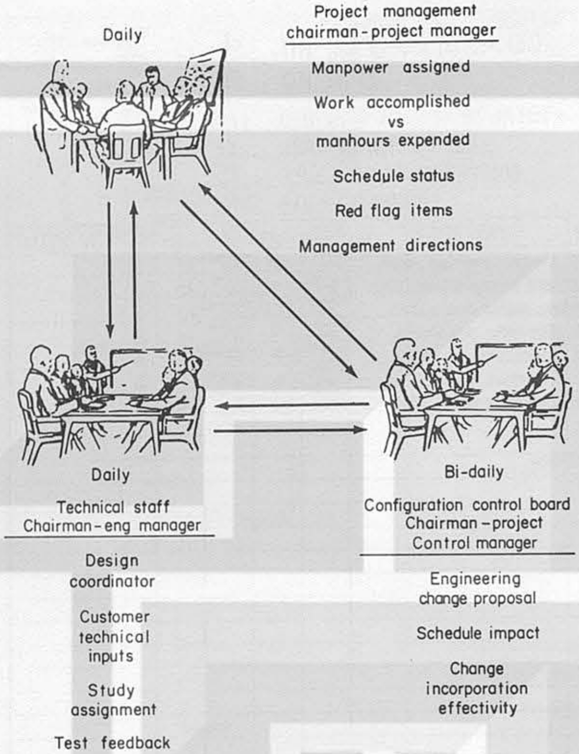


FIGURE 9–5.—Management control.

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-

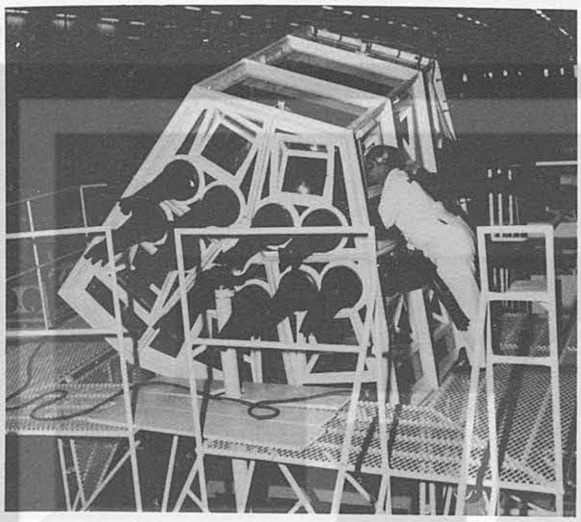


FIGURE 9–6.—Plexiglas welding boxes.

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-

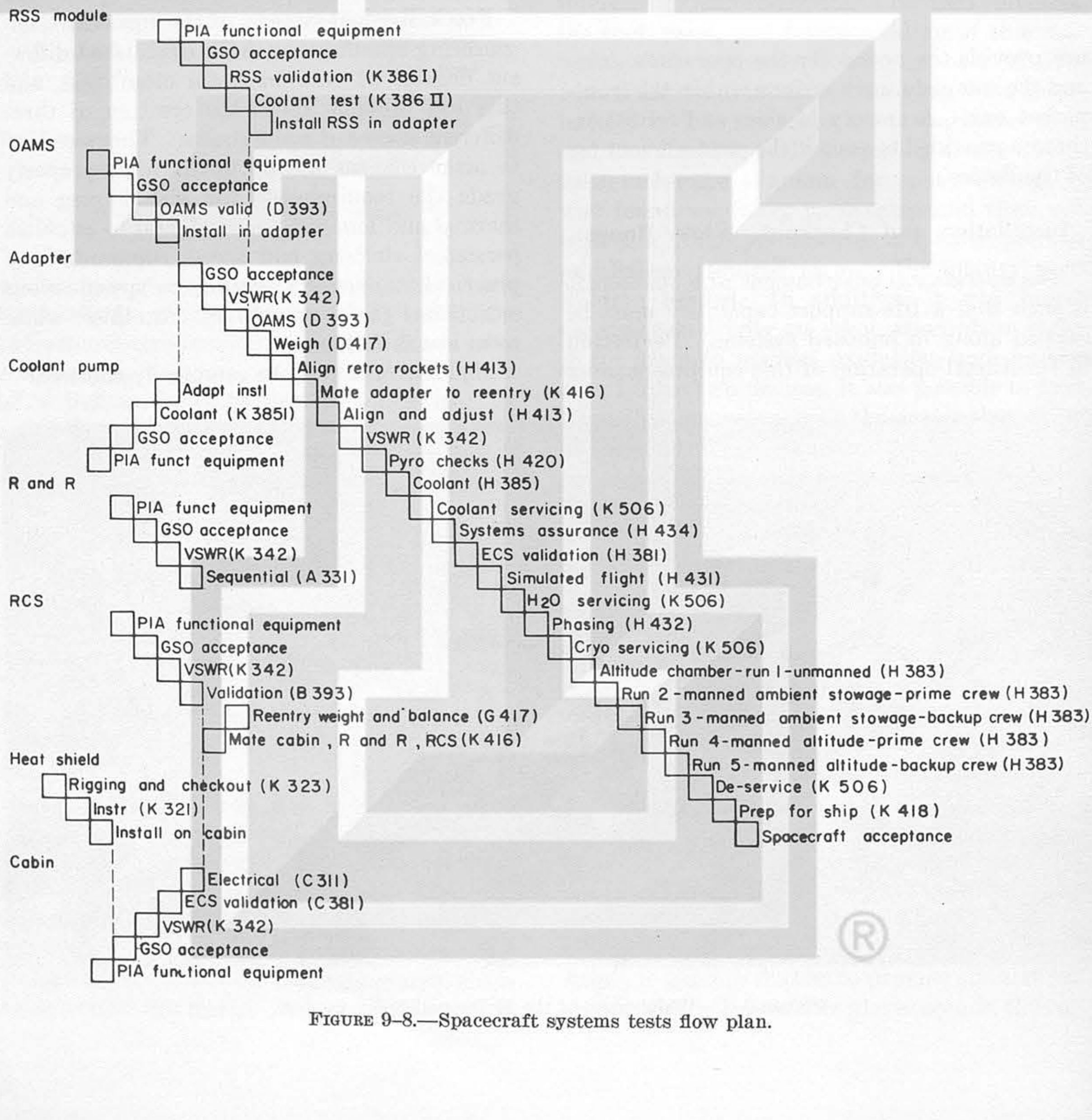


FIGURE 9–8.—Spacecraft systems tests flow plan.

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

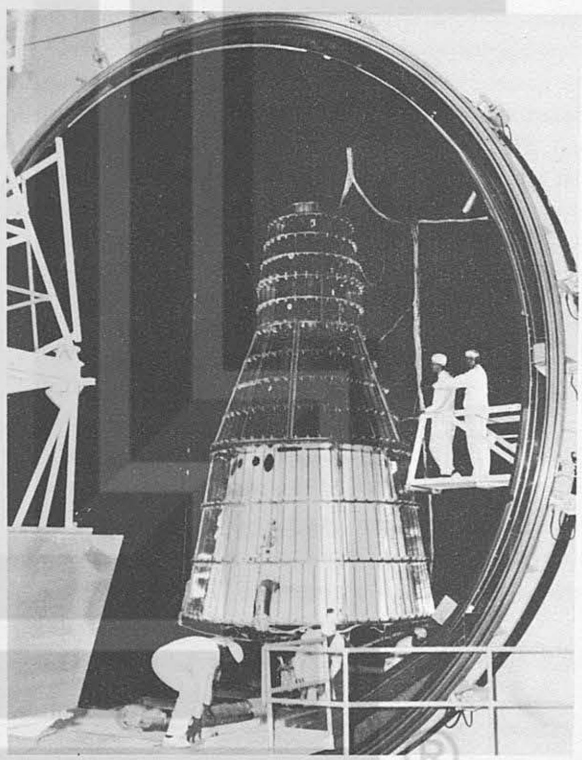


FIGURE 9–9.—McDonnell altitude chamber.

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

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

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

By WILLIAM H. DOUGLAS, *Deputy Manager, Office of Test Operations, Gemini Program Office, NASA Manned Spacecraft Center;* GREGORY P. MCINTOSH, *Gemini Program Office, NASA Manned Spacecraft Center; and* LEMUEL S. MENEAR, *Gemini Program Adviser, Flight Safety Office, NASA Manned Spacecraft Center*

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.

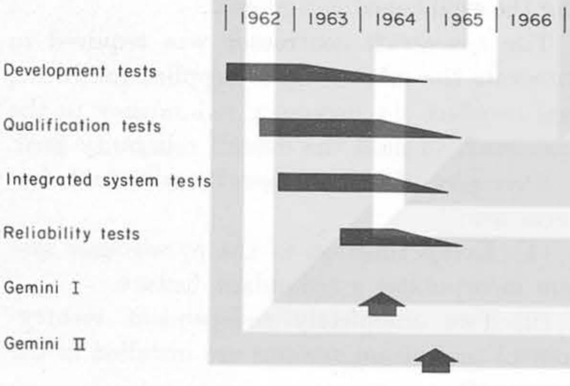


FIGURE 10–1.—Gemini test program.

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

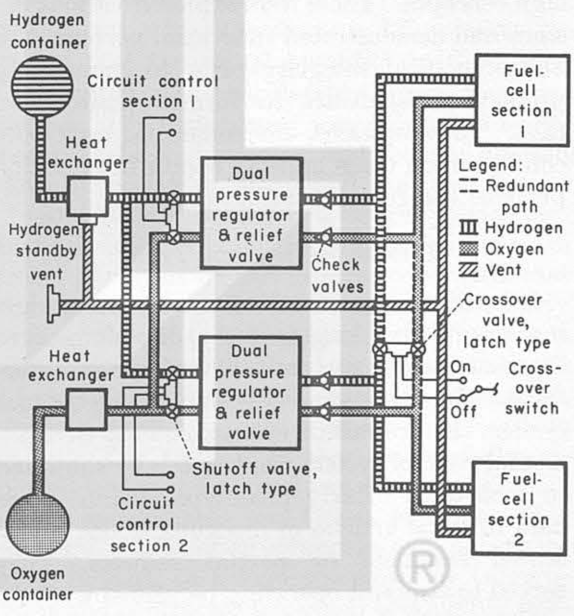


FIGURE 10–2.—Fuel-cell reactant supply system.

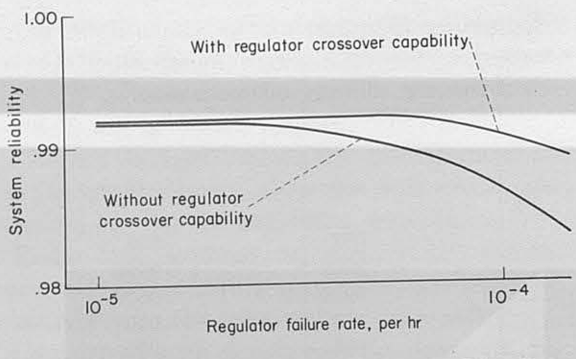


FIGURE 10-3.—Fuel-cell power system reliability for a 2-week mission.

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

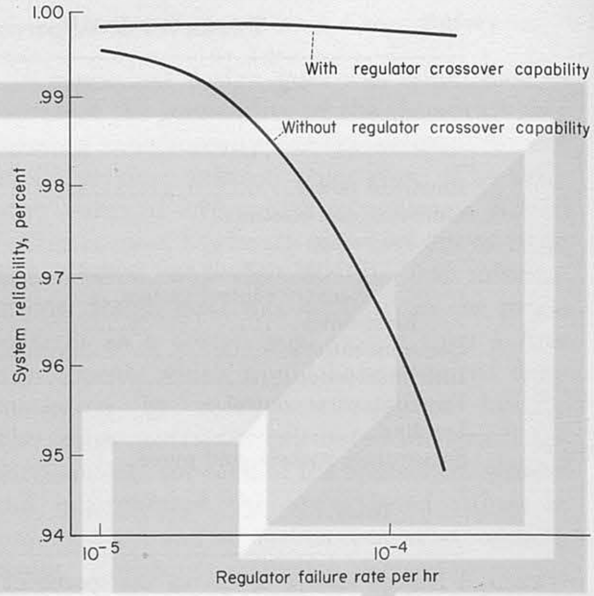


FIGURE 10-4.—Fuel-cell power system reliability for a 2-day mission.

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.

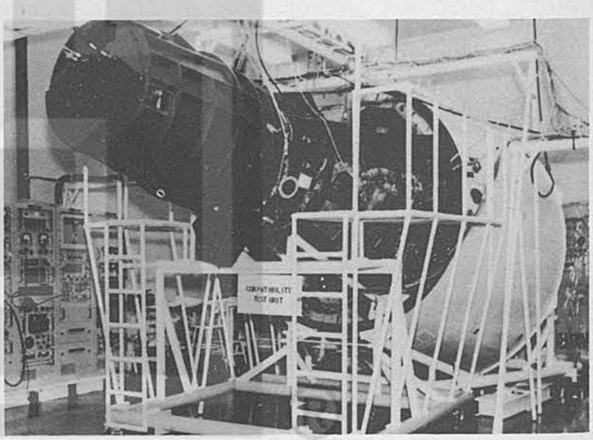


FIGURE 10-5.—Gemini compatibility test unit.

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.

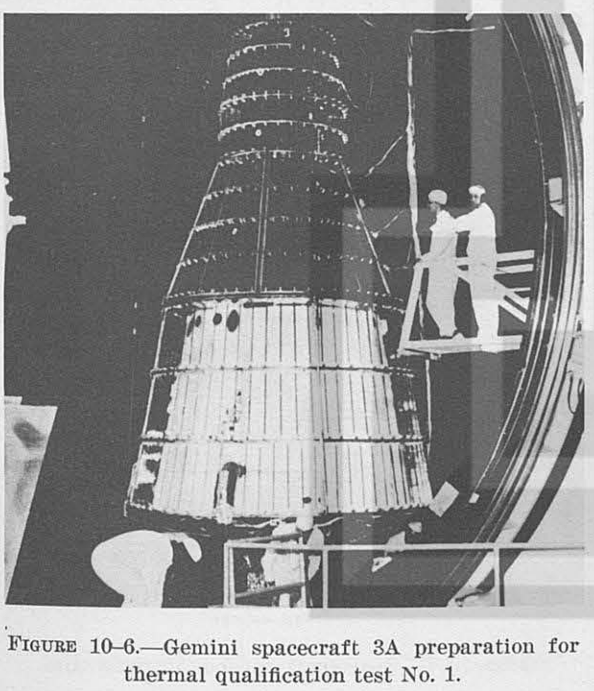


FIGURE 10-6.—Gemini spacecraft 3A preparation for thermal qualification test No. 1.

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-

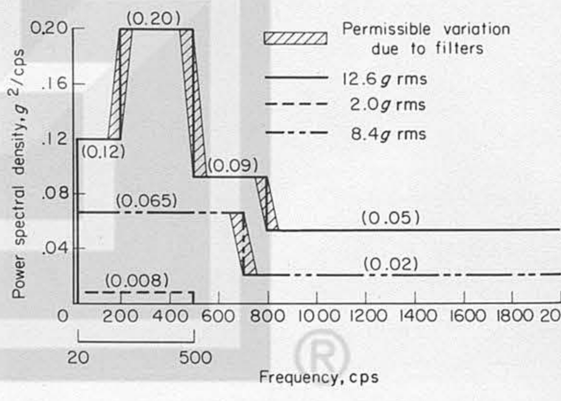


FIGURE 10–7.—Spacecraft random vibration test.

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	Environments required and status																										Test schedule				Similarity to part no.
				A	B	C	D	E	F	G	H	I	J	K	L	M	N	O	P	Q	R	S	T	U	V	W	X	Y	Z					
				Hi temp.	Lo temp.	Vib.	Shock	Temp. Alt.	Accel.	RFI	Humid (Opr)	Humid (N Op)	Acoustic N	Exp.	Rain	O_2 Atmos.	Low Pres.	Pres. Drop	Hi Pres.	Fungus	Salt Spray	S.W. Imm.	Struct.	Exp. Decom.	Temp. Cont.	Life	Proof Pr.	Ht. Trans.		Planned start	Actual start	Planned completion	Complete	
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

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.

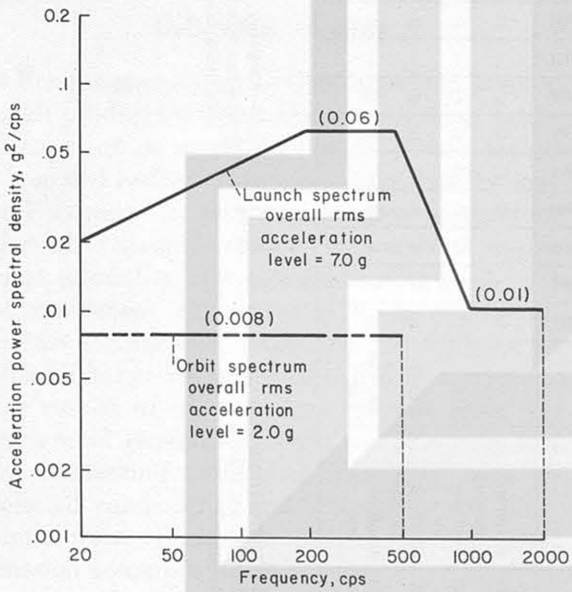


FIGURE 10-8.—Random vibration of test adapter blast-shield region.

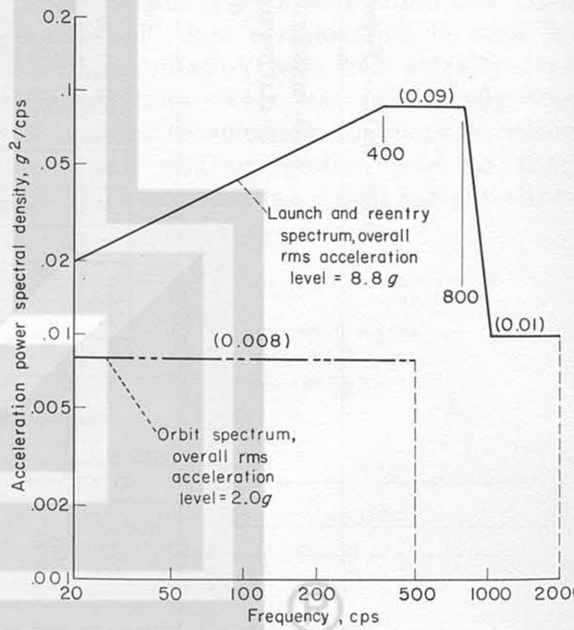


FIGURE 10-9.—Random vibration test of reentry assembly region.

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

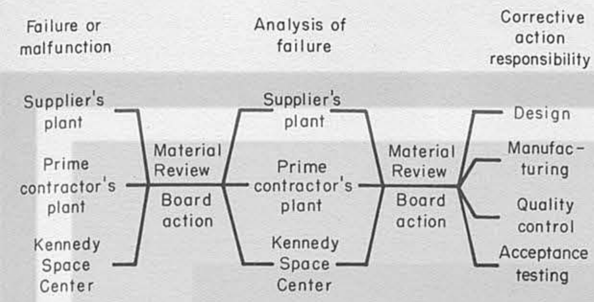


FIGURE 10-10.—Gemini corrective action flow schematic.

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.

B

LAUNCH VEHICLE

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.

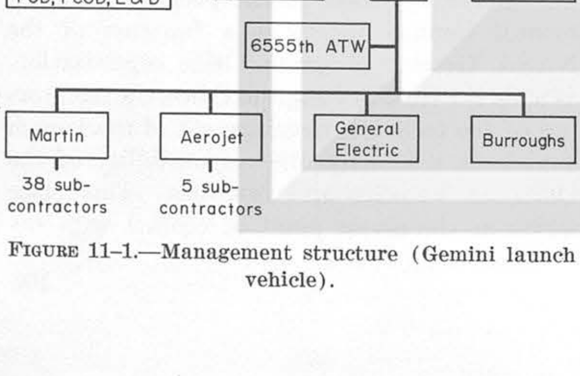


FIGURE 11-1.—Management structure (Gemini launch vehicle).

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

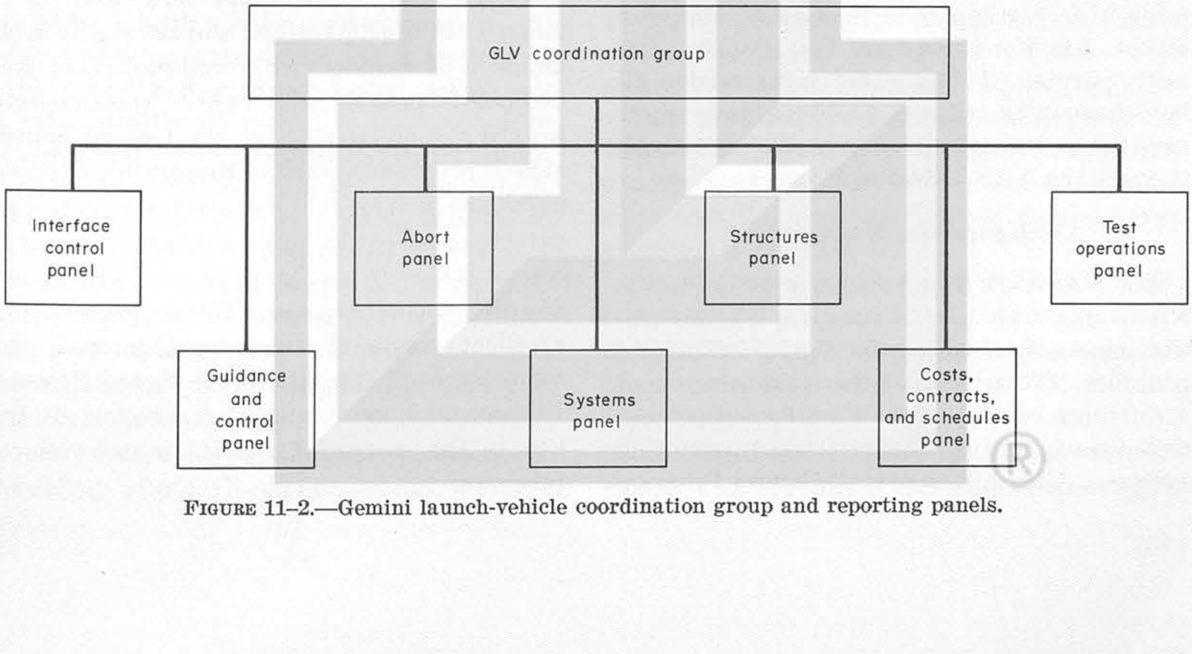


FIGURE 11–2.—Gemini launch-vehicle coordination group and reporting panels.

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.

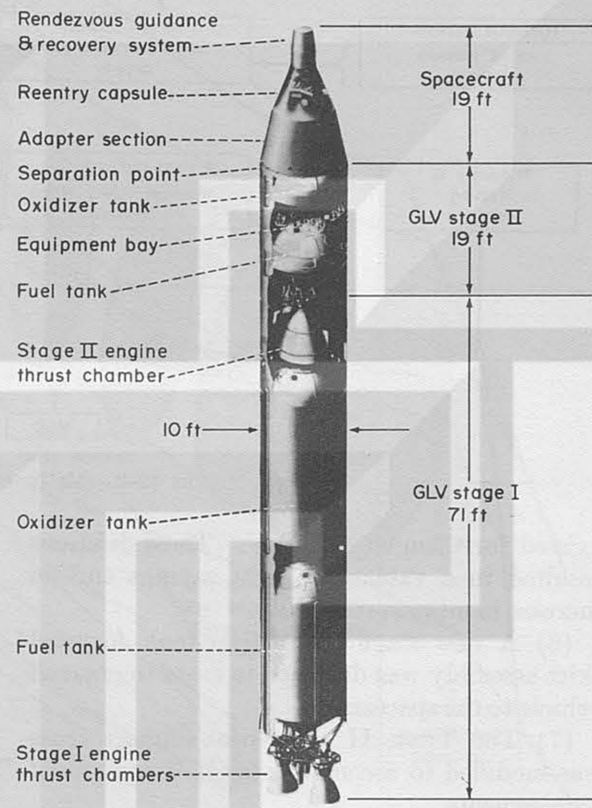


FIGURE 12–1.—Gemini launch vehicle.

(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

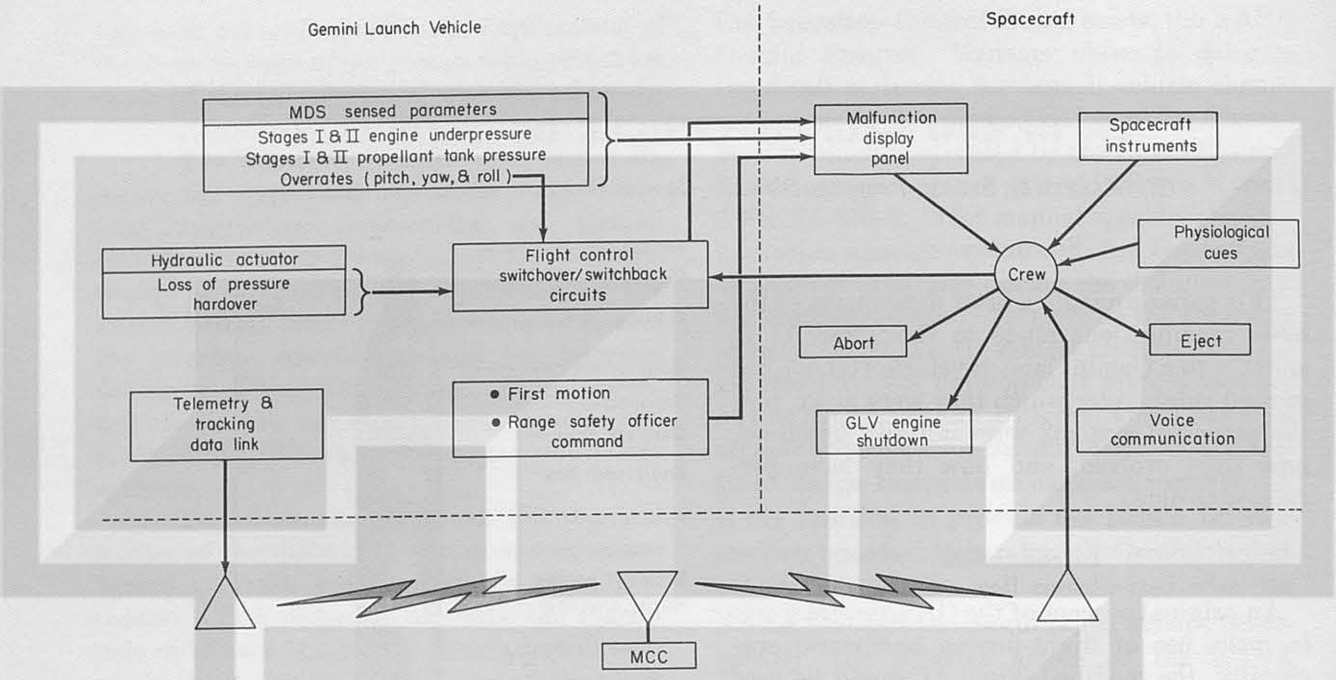


FIGURE 12–2.—Malfunction detection system.

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

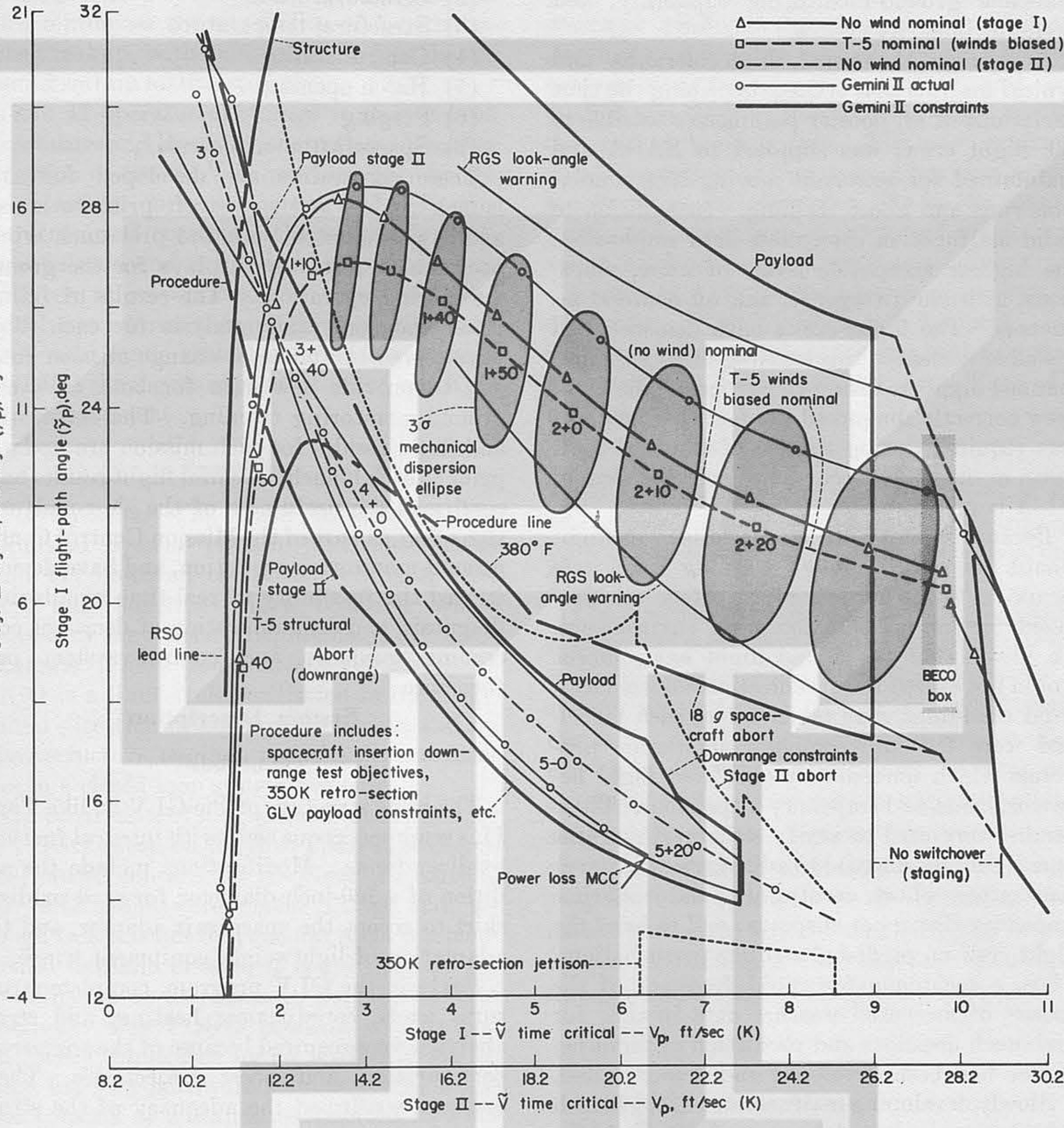


FIGURE 12–3.—Detailed failure-mode analysis.

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-

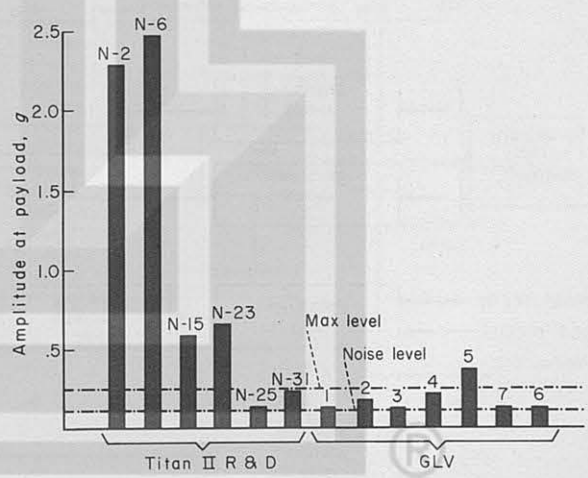


FIGURE 12–4.—History of POGO reduction.

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,

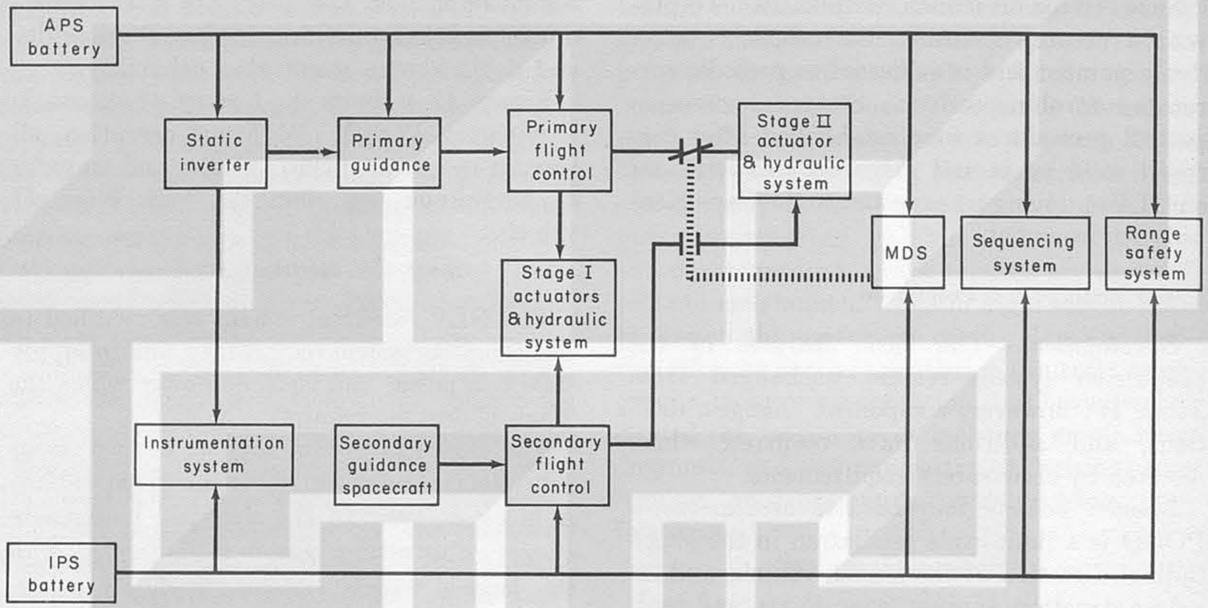


FIGURE 12-5.—Electrical power subsystem.

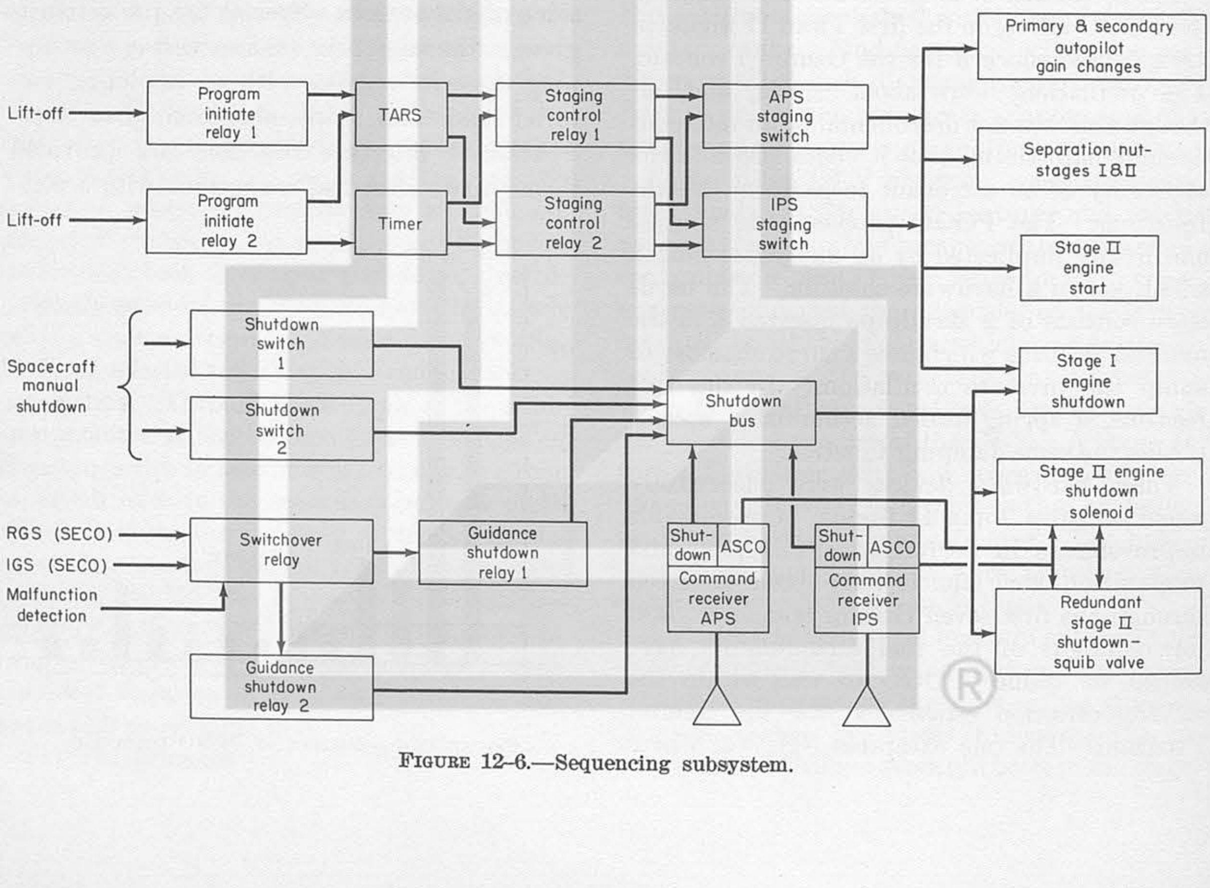


FIGURE 12-6.—Sequencing subsystem.

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.

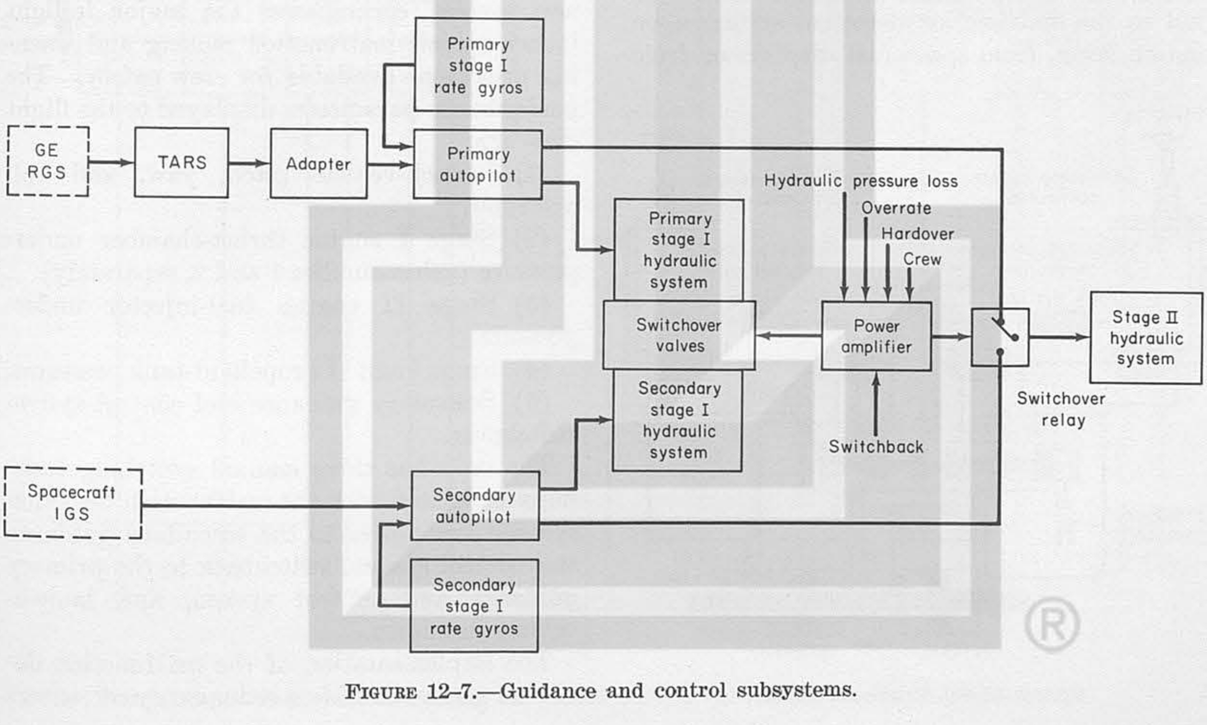


FIGURE 12–7.—Guidance and control subsystems.

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

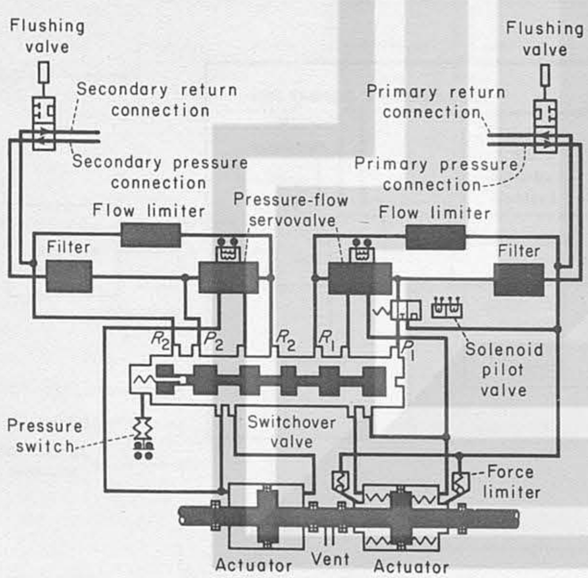


FIGURE 12–8.—Tandem actuator.

information sent by ground-monitoring personnel, that a primary system malfunction has occurred.

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

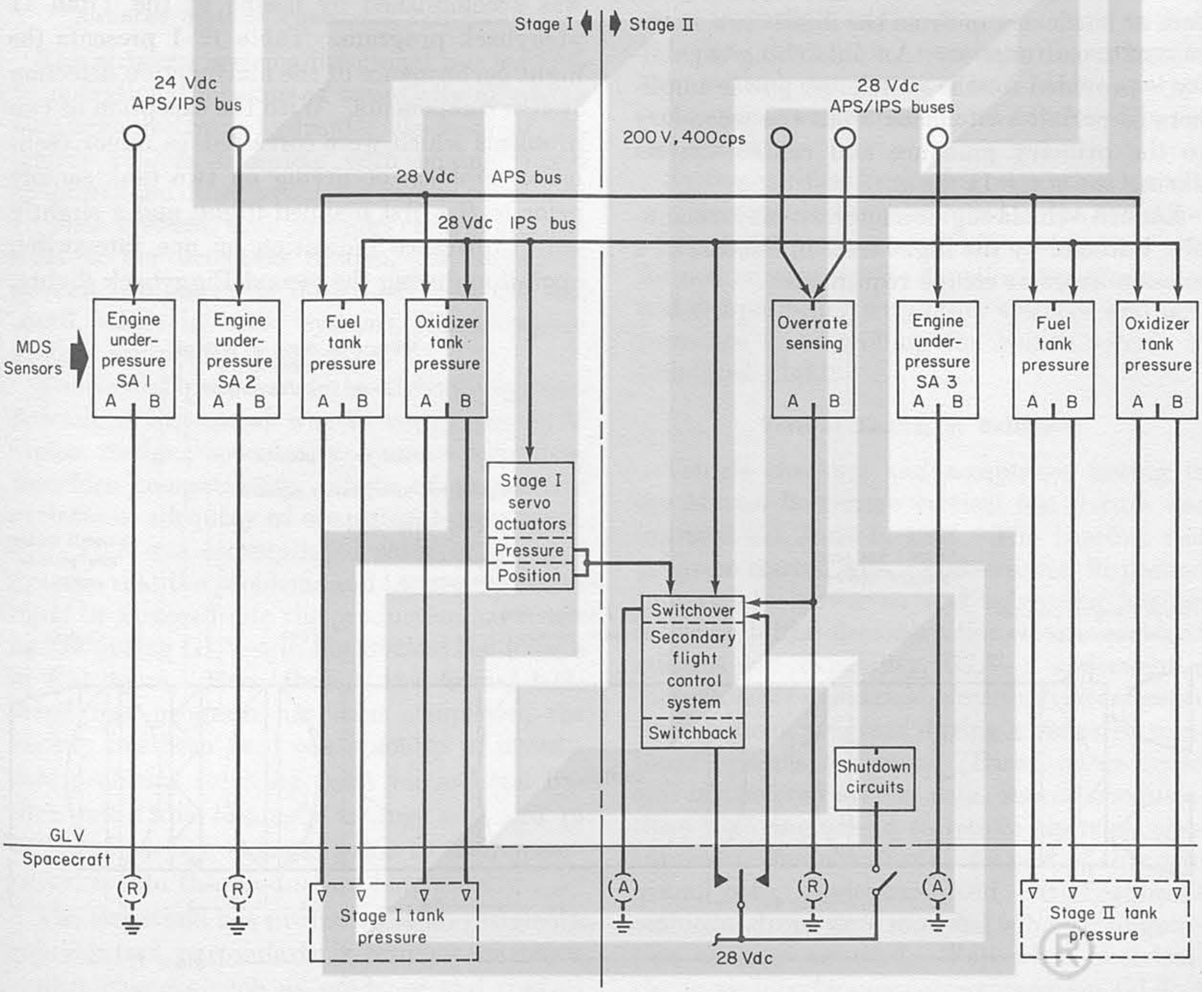


FIGURE 12–9.—Spacecraft monitoring of Gemini launch vehicle malfunction detection.

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),

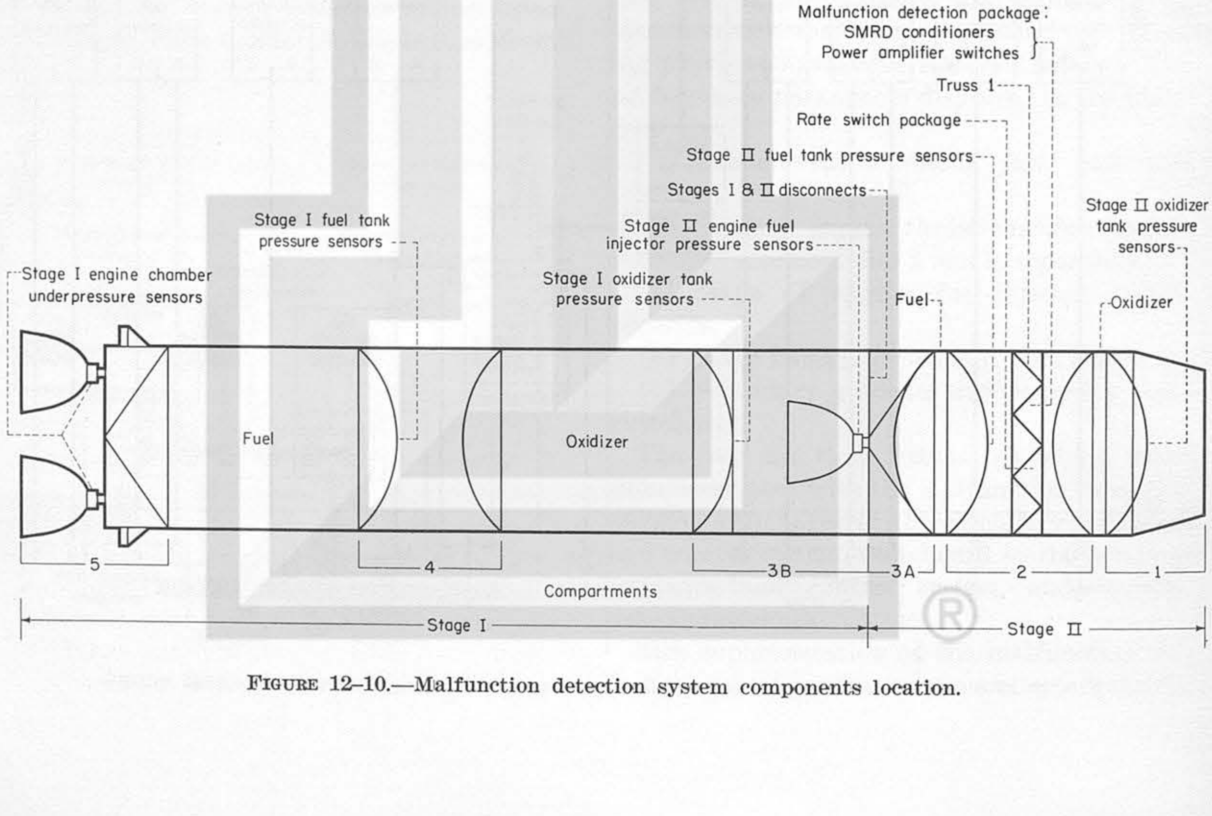


FIGURE 12–10.—Malfunction detection system components location.

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.

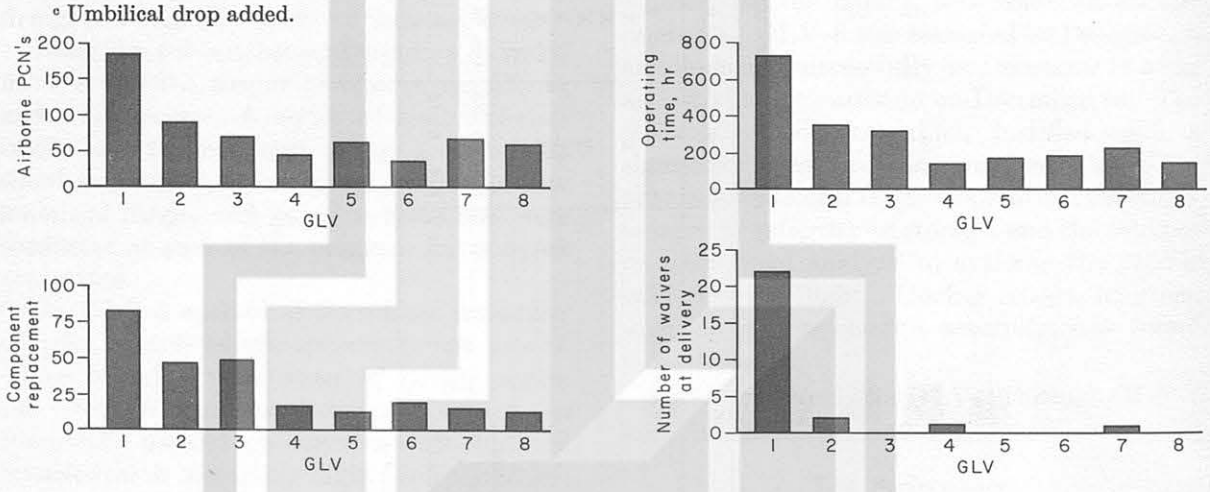


FIGURE 12-11.—Vertical test fixture performance.

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 reerection 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

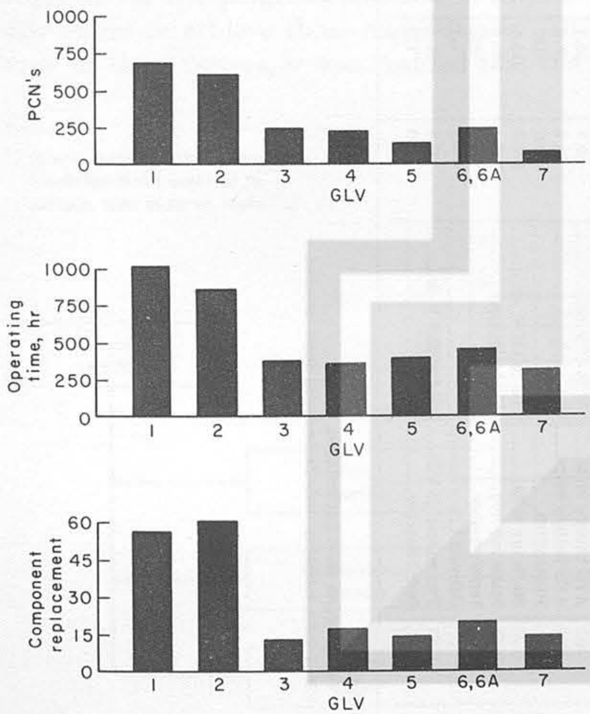


FIGURE 12–12.—Cape Kennedy testing performance.

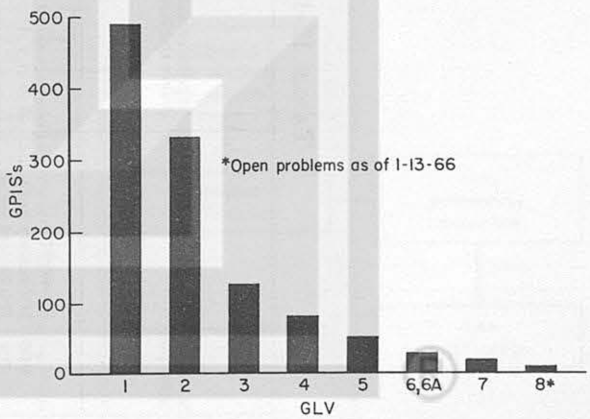


FIGURE 12–13.—Overall measure of test performance.

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: CSAT, Date 6-25-65, Test no. 011/012		ETR tests: 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.

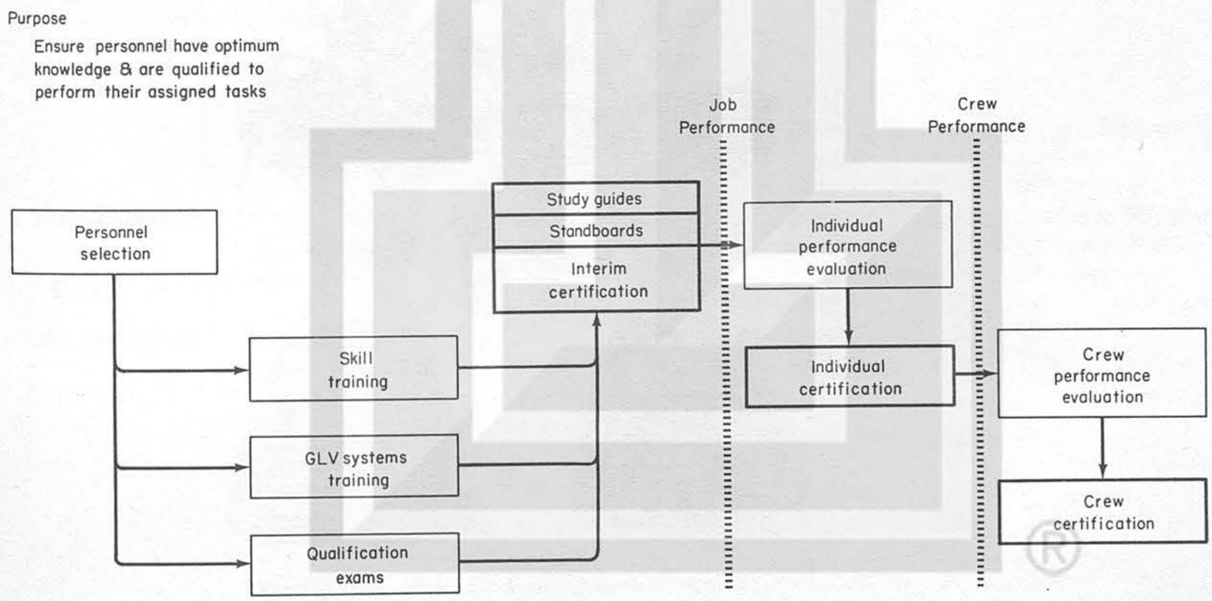


FIGURE 12–14.—Personnel training and certification.

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

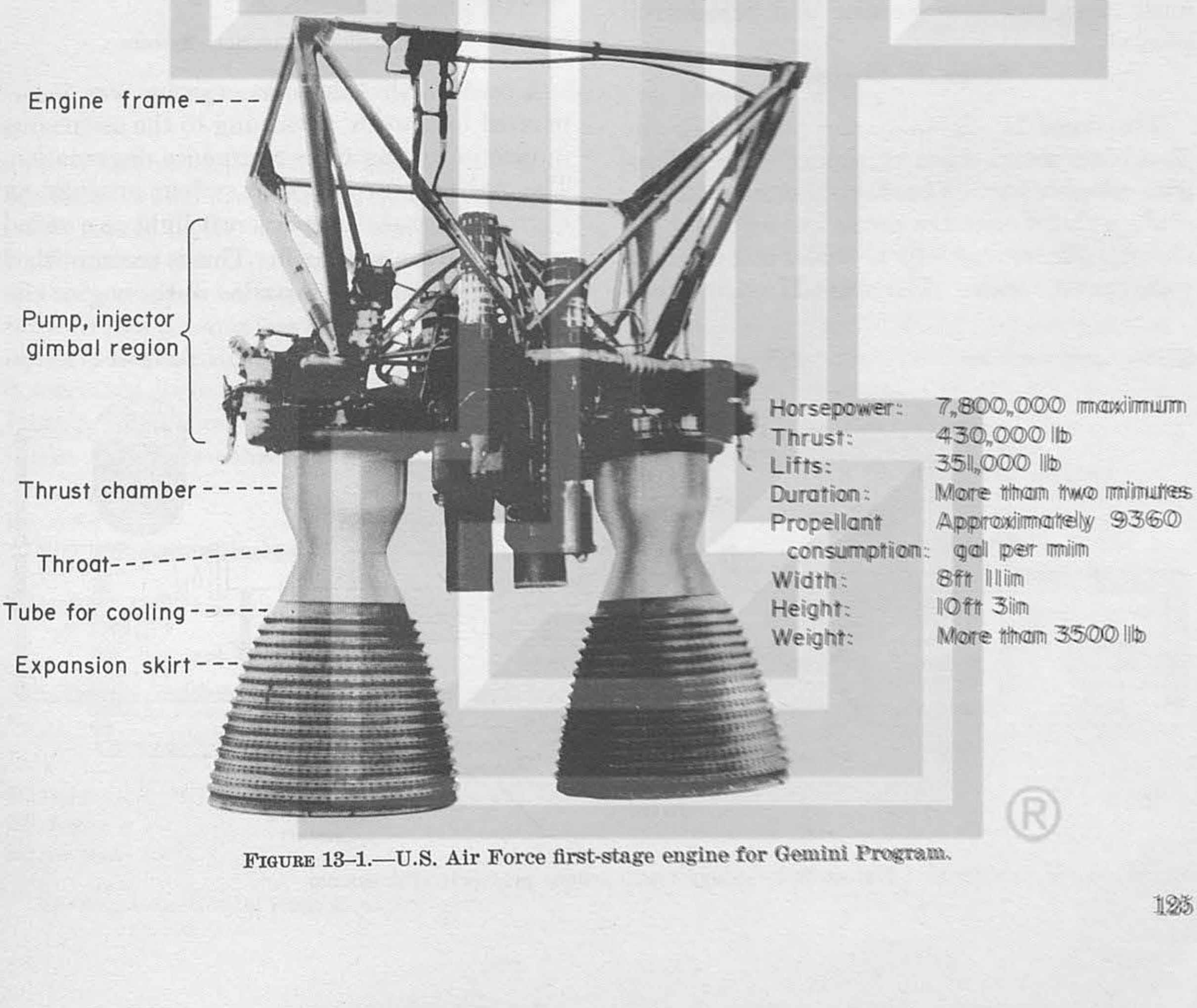


Figure 13-1.—U.S. Air Force first-stage engine for Gemini Program.

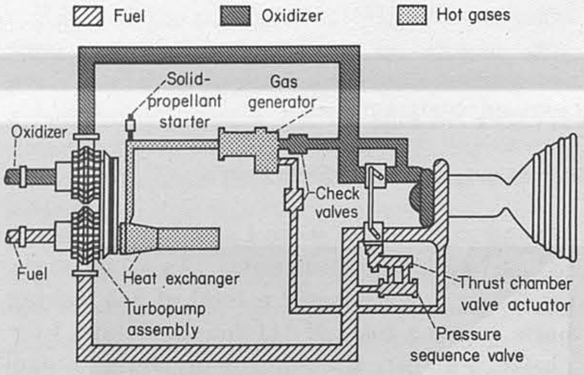


FIGURE 13-2.—Gemini stage 1 engine schematic.

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-

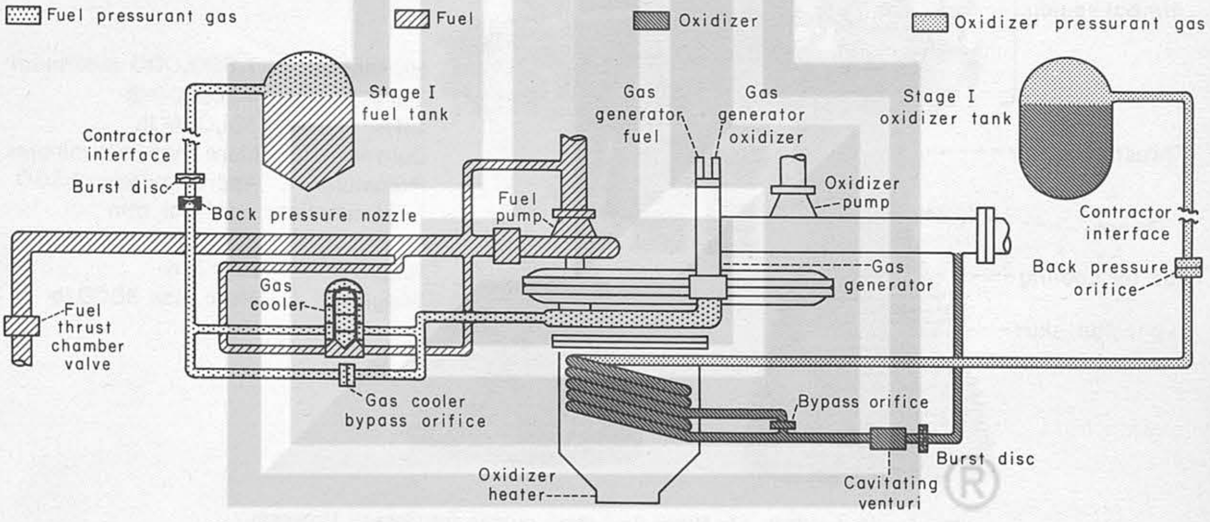


FIGURE 13-3.—Stage 1 autogenous pressurization system.

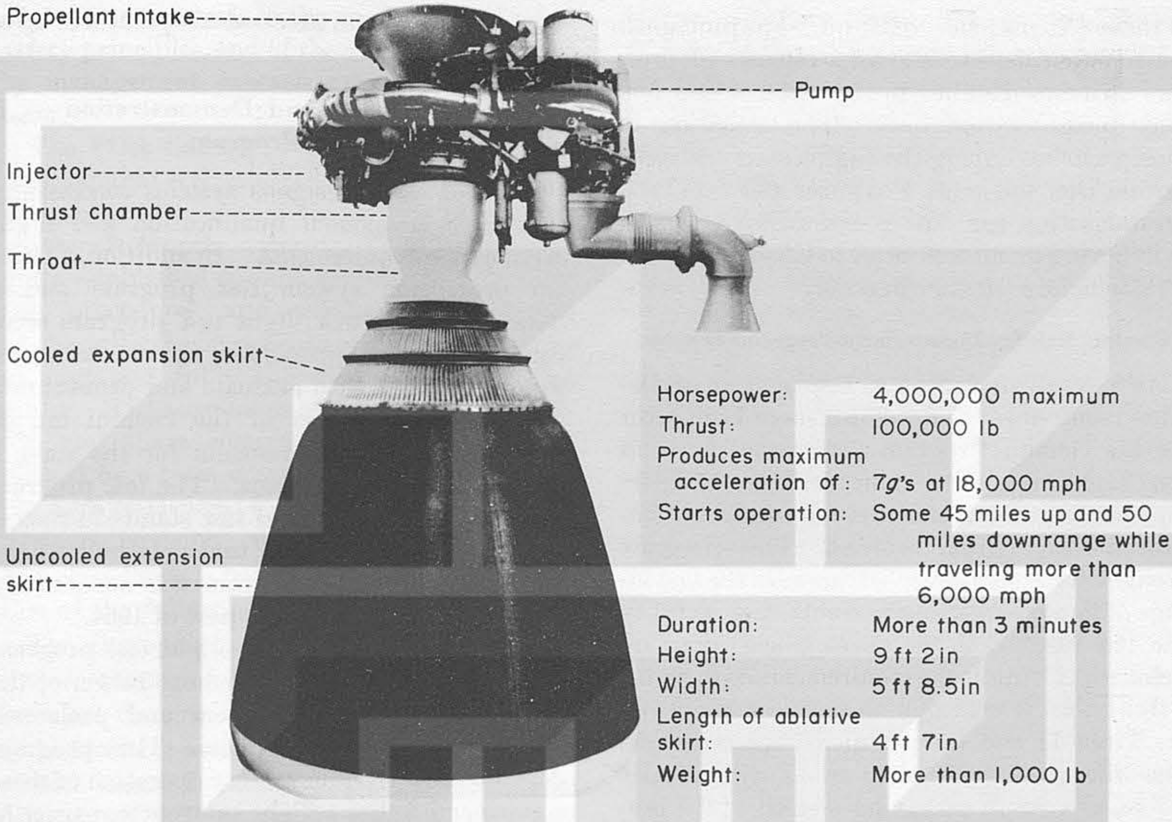


FIGURE 13-4.—U.S. Air Force second-stage space-start engine for Gemini Program.

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.

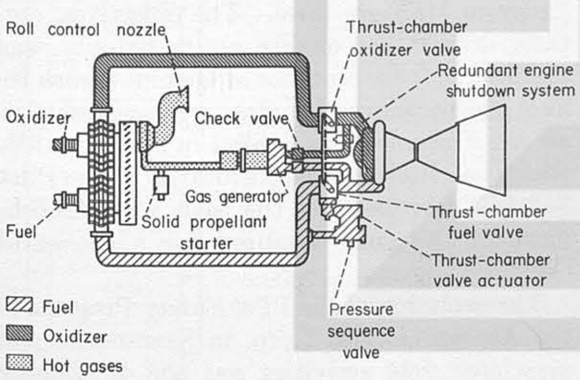


FIGURE 13-5.—Gemini stage II engine schematic.

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

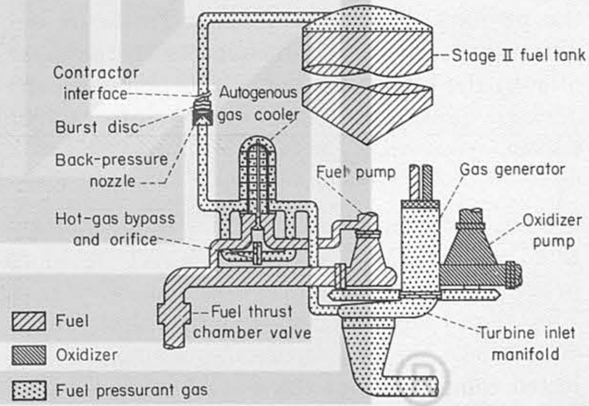


FIGURE 13-6.—Stage II autogenous pressurization system.

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.

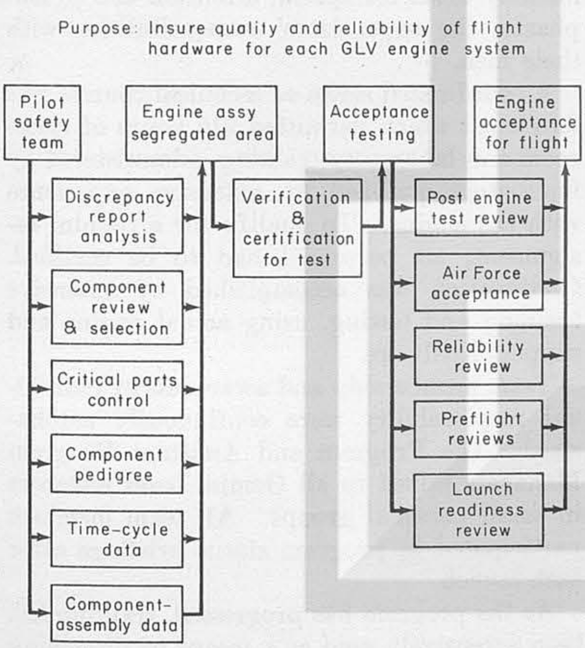


FIGURE 13–7.—Pilot Safety Program.

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.

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.

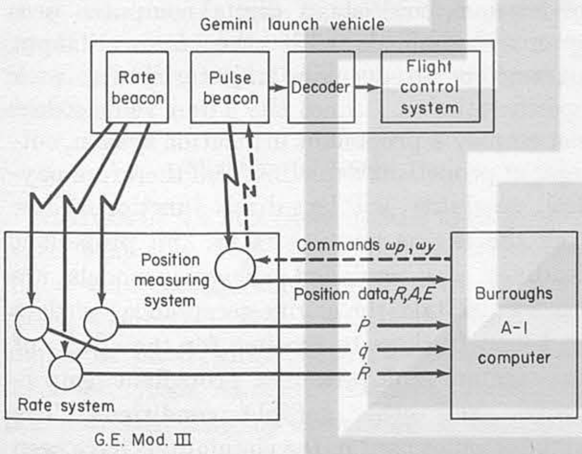


FIGURE 14–1.—Gemini launch vehicle guidance system.

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.

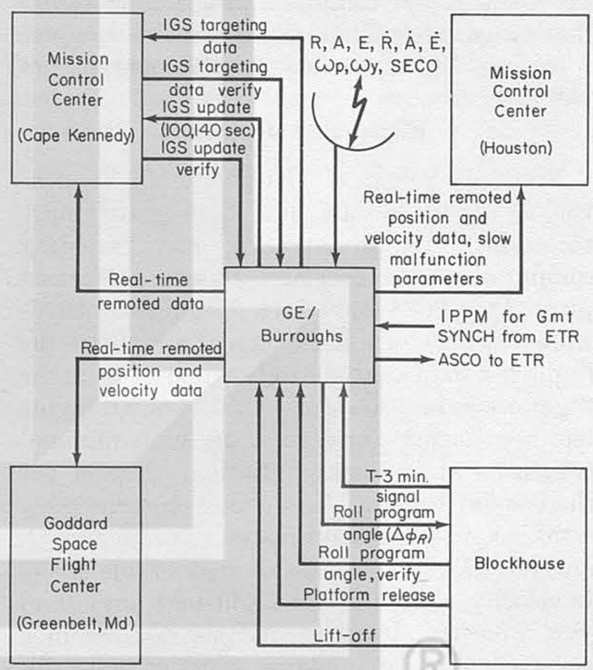


FIGURE 14–2.—RGS computer interfaces.

(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.

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

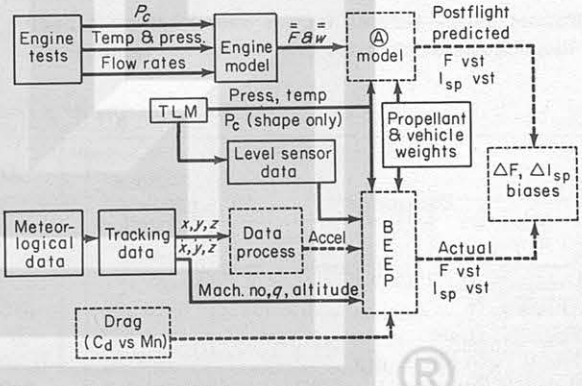


FIGURE 14–3.—Vehicle performance evaluation block diagram.

218-556 O—66——10

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.

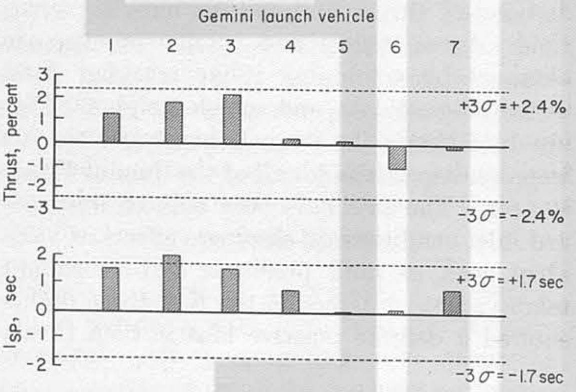


FIGURE 14–4.—Gemini launch vehicle stage I engine dispersions (normalized to standard inlet conditions).

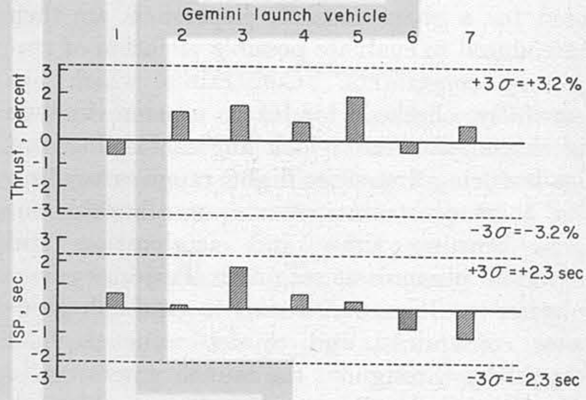


FIGURE 14–5.—Gemini launch vehicle stage II engine dispersions (normalized to standard inlet conditions).

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

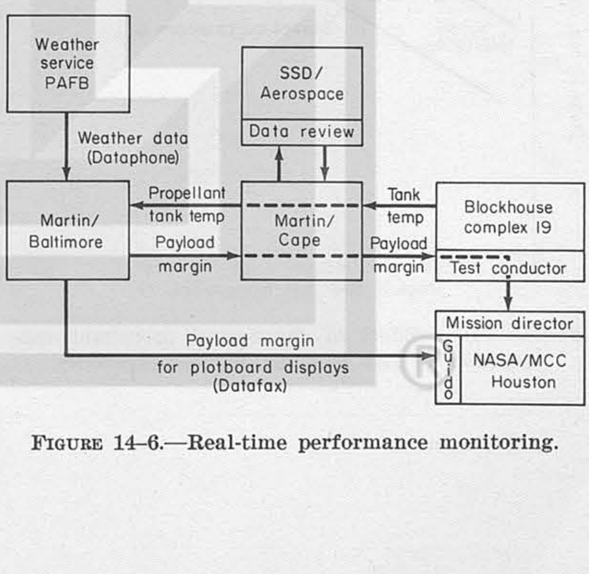


FIGURE 14–6.—Real-time performance monitoring.

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.

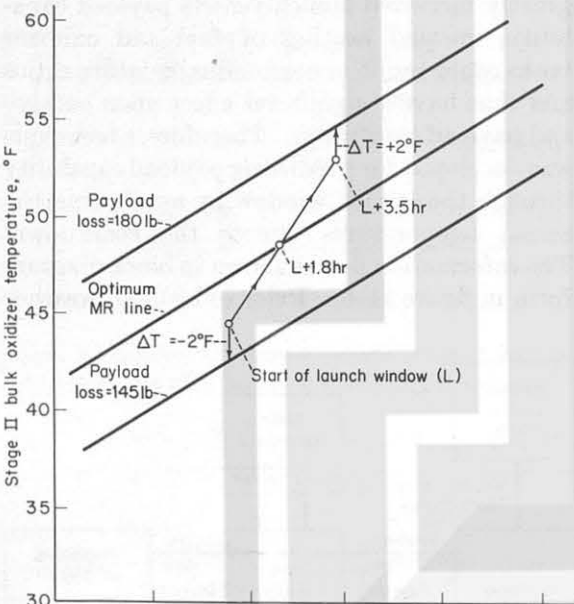


FIGURE 14-7.—Effect of differential propellant temperatures on GLV minimum payload capability.

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

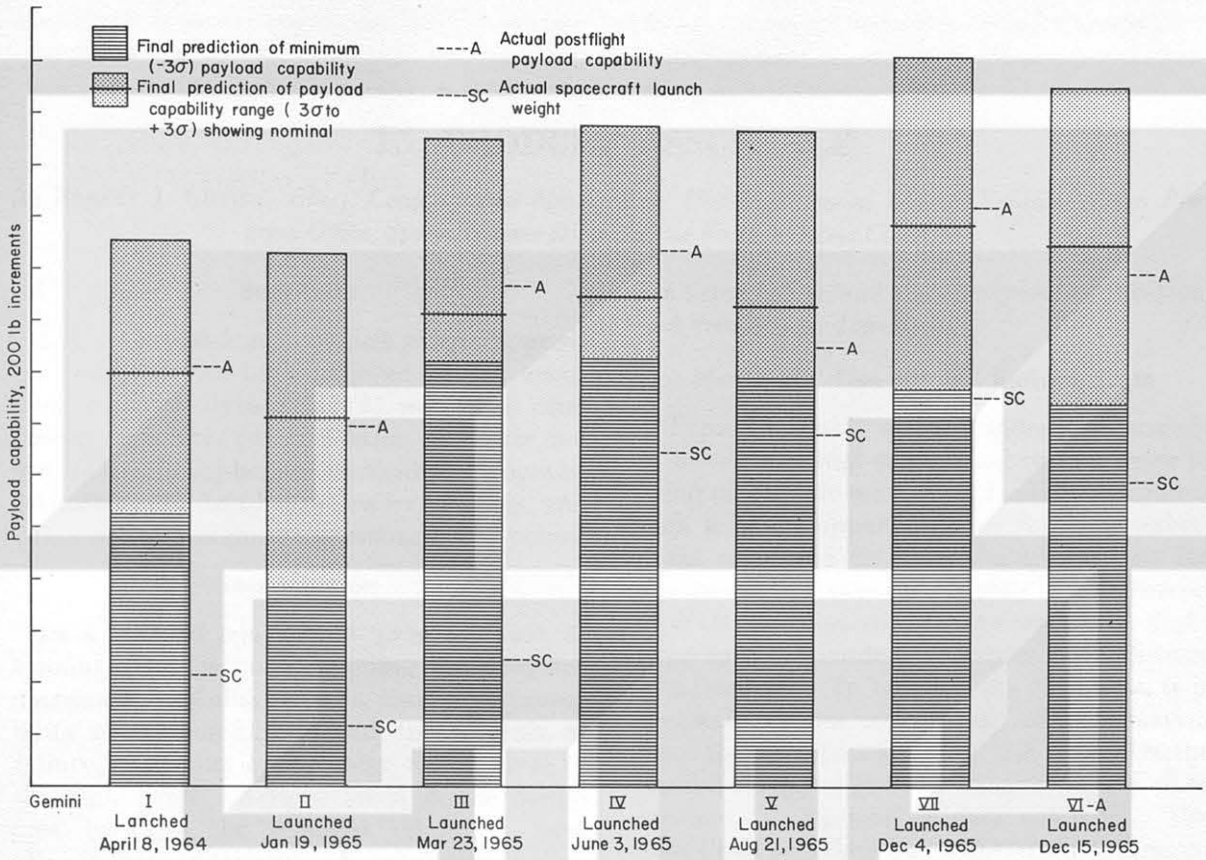


FIGURE 14-8.—Gemini launch vehicle performance history.

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.

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.

C

FLIGHT OPERATIONS

218–556 O—66——11

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.

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

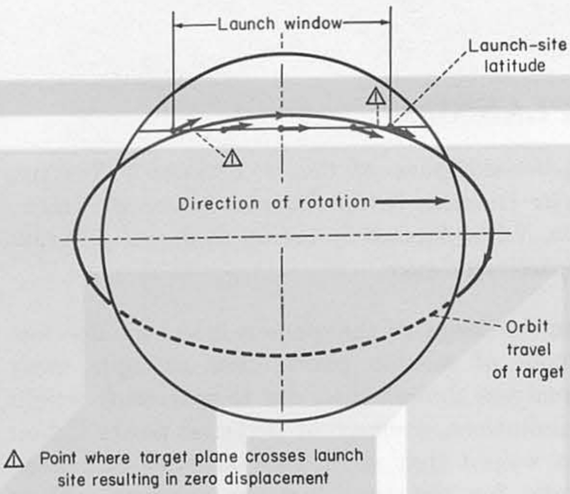


FIGURE 18-1.—Variable-azimuth launch technique.

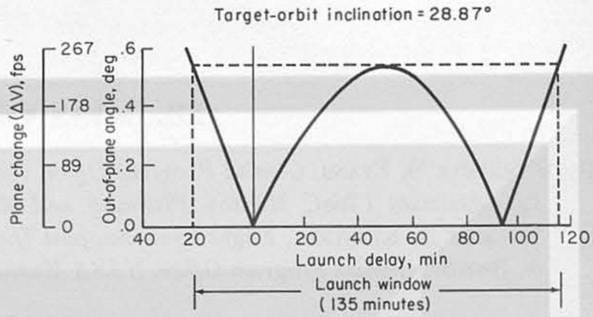


FIGURE 18-2.—Variable-azimuth launch window for target orbit inclination of 28.87°.

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 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-

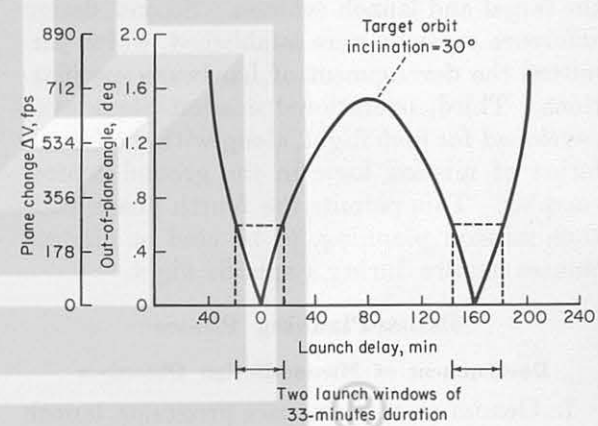


FIGURE 18-3.—Variable-azimuth launch windows for target orbit inclination of 30°.

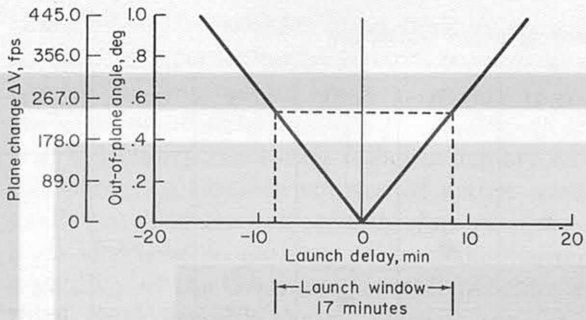


FIGURE 18-4.—Fixed-azimuth launch window for target orbit inclination of 28.87°.

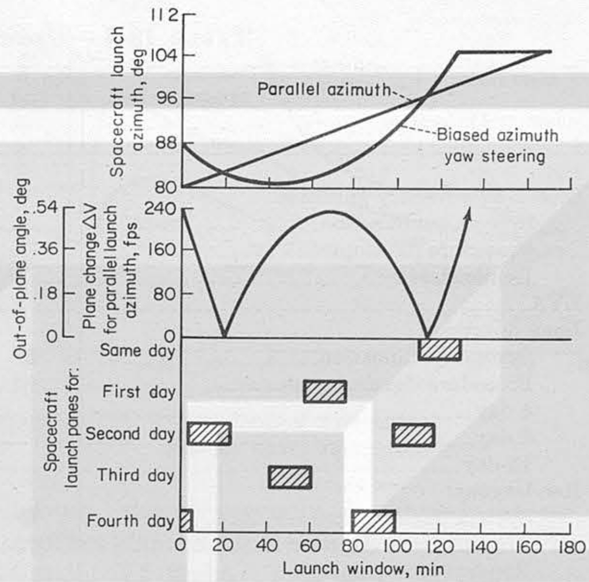


FIGURE 18-5.—Space-vehicle launch windows for rendezvous target orbit altitude of 161 nautical miles.

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 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

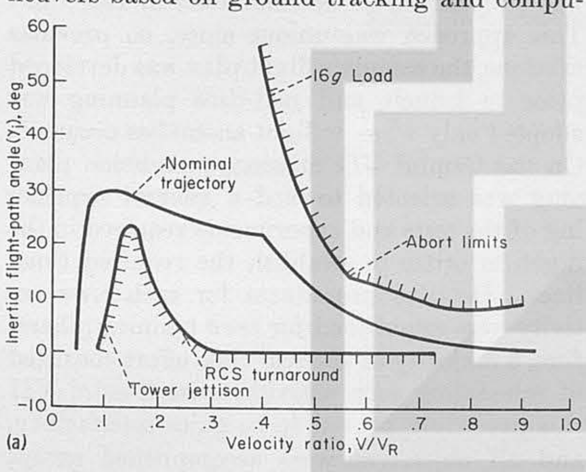
During Project Mercury, a major part of the trajectory-planning effort was spent in the development of the philosophy and techniques for monitoring the powered-flight trajectory, for determining when launch abort action was necessary, and for establishing go—no-go criteria for the acceptability of the 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.

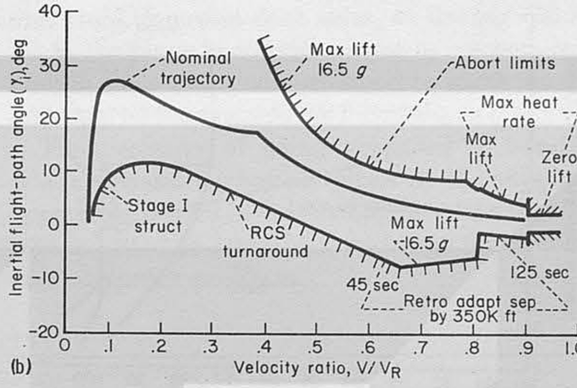
The capabilities of the Gemini spacecraft provide a tremendous amount of flexibility in the types of missions which can be designed. This flexibility has allowed modification of mission plans both before and during an actual flight. For example, during the Gemini V mission, problems with the spacecraft 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-



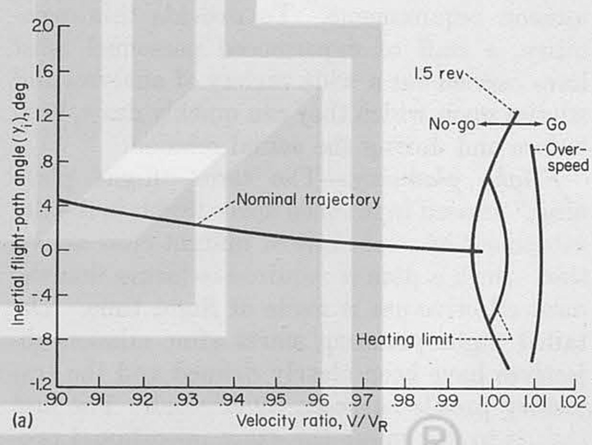
(*a*) Project Mercury.

FIGURE 18–6.—Abort limit lines launch trajectory monitoring.



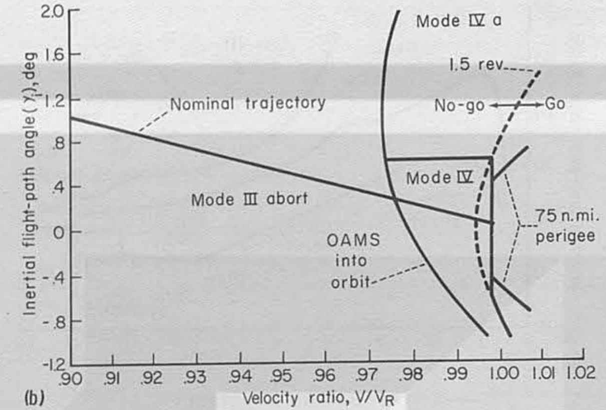
(*b*) Gemini Program.

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

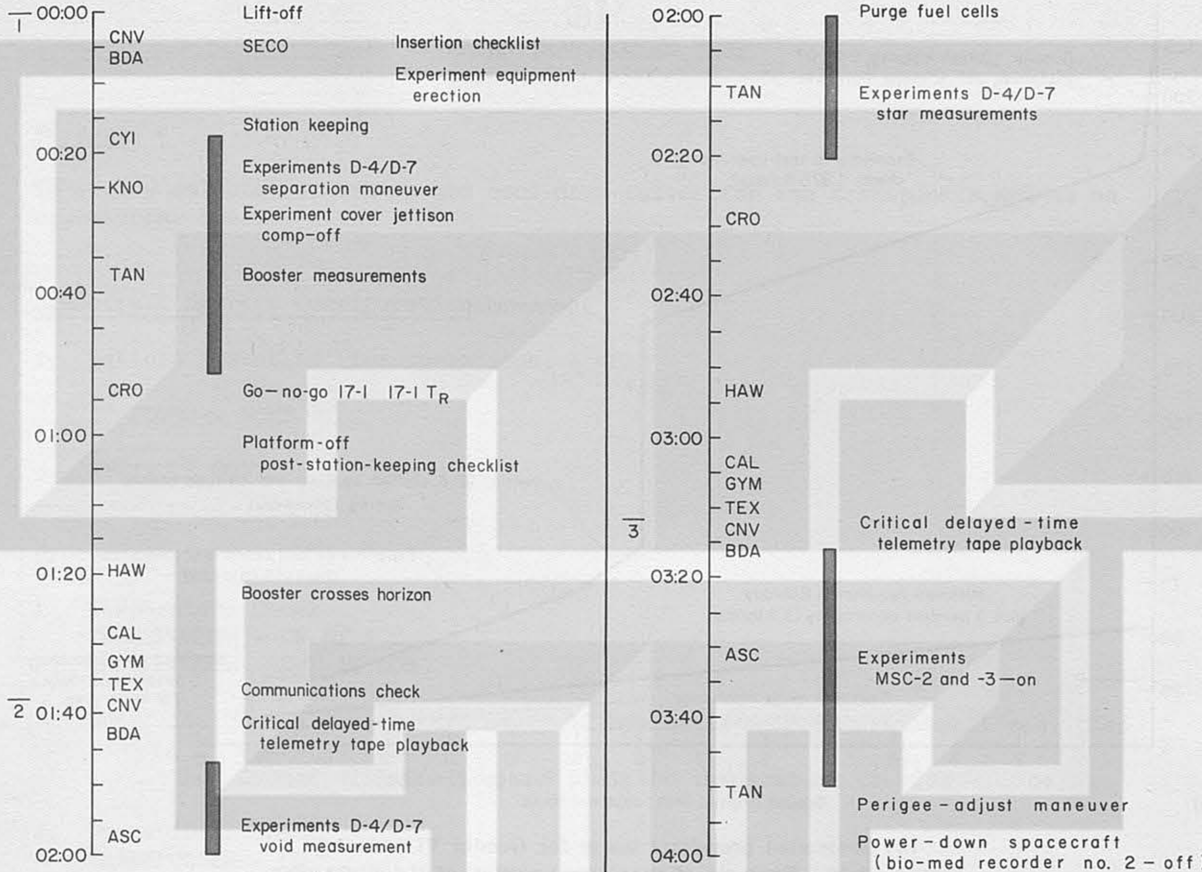
(*a*) Lift-off through first 50 seconds.
FIGURE 18-8.—Example of detailed flight plan.

(b)

TIME HR:MIN:SEC	COMP	PLAT	CNTL MODE	ACTION: COMMAND PILOT	ACTION: 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	

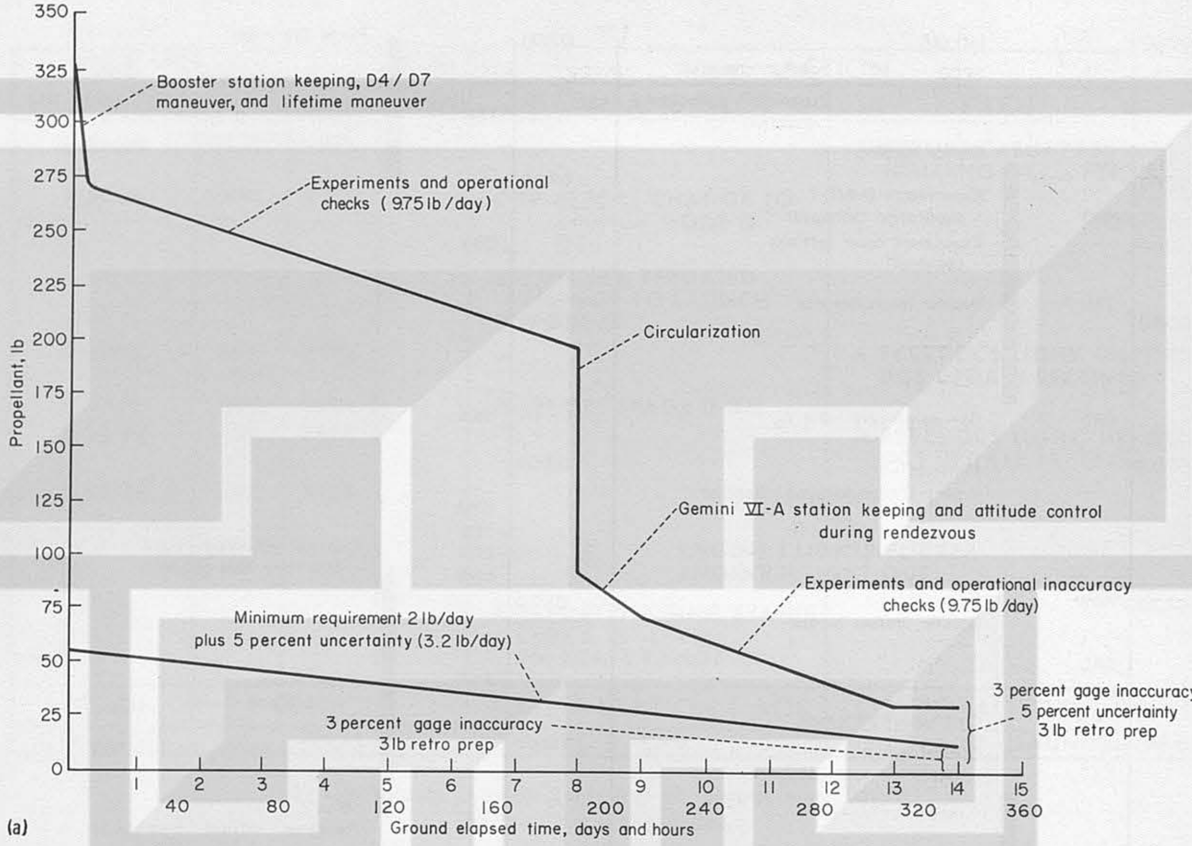
MISSION	EDITION	DATE	STATION	AOS	LOS	TOTAL	REV	PAGE
GEMINI VII	FINAL	11/15/65	CNV	0:00:00	0:06:57	6:57	LAUNCH	2

(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.

218-556 O—66——12

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

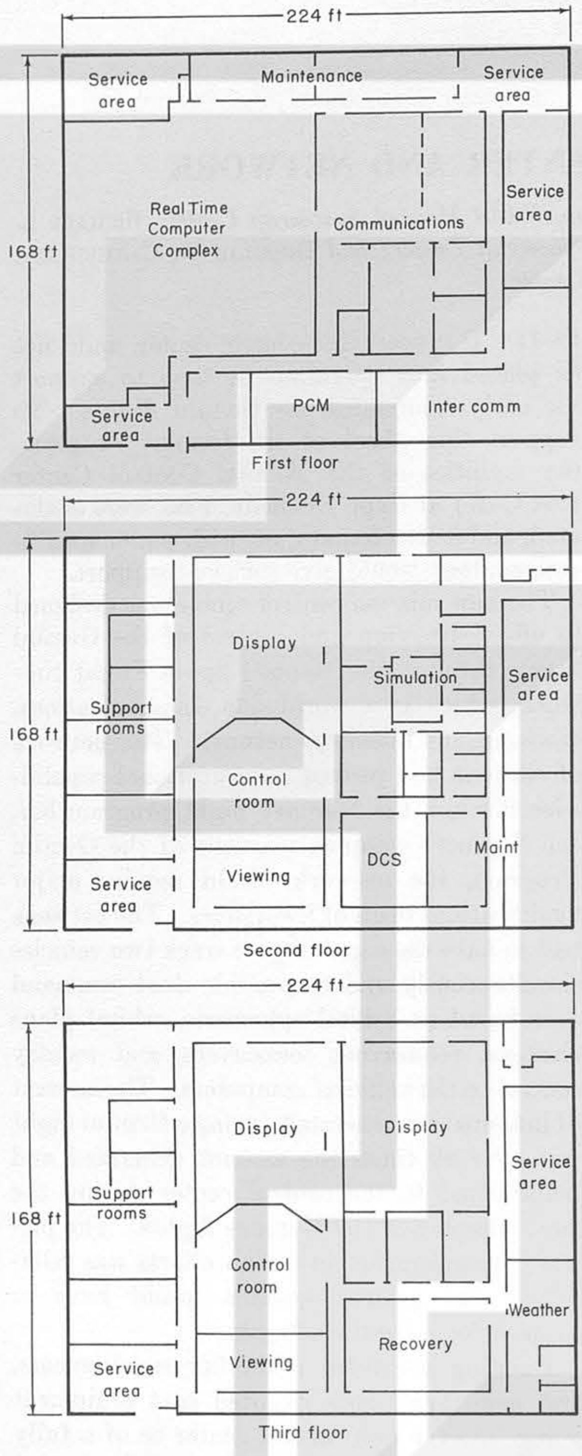


FIGURE 19-1.—Floor plan of Mission Control Center, Houston, Tex.

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

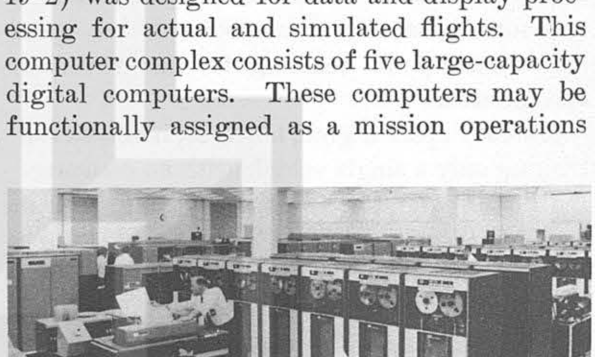


FIGURE 19-2.—Real Time Computer Complex, Houston, Tex.

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

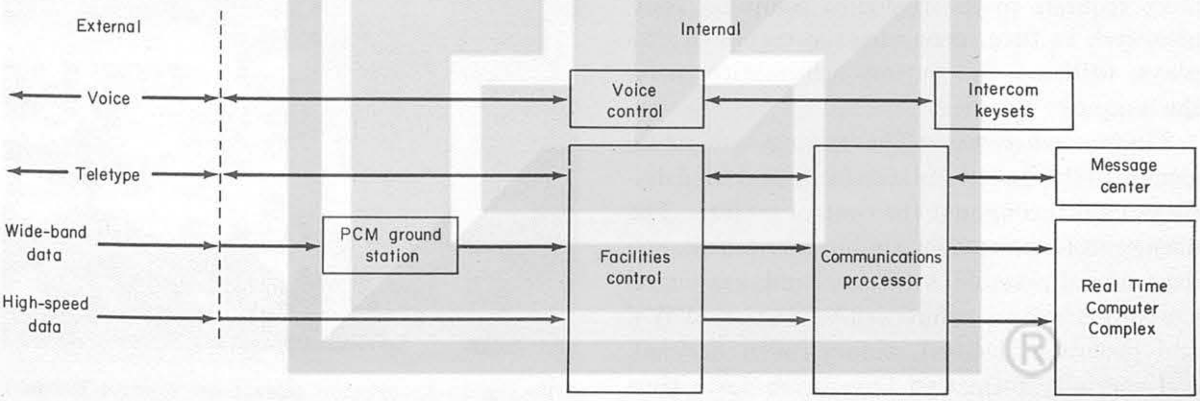


FIGURE 19–3.—MCC–H communications flow.

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.

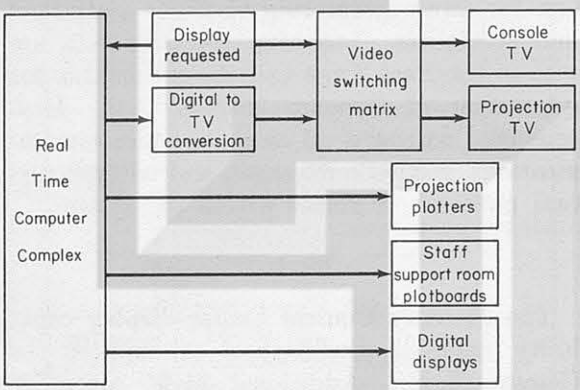


FIGURE 19-4.—MCC-H display/control subsystem.

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-

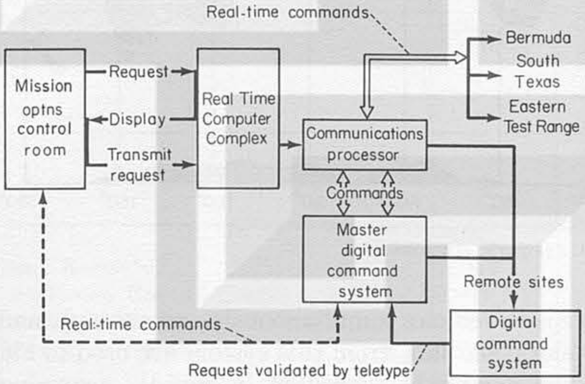


FIGURE 19–6.—Digital command system.

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.

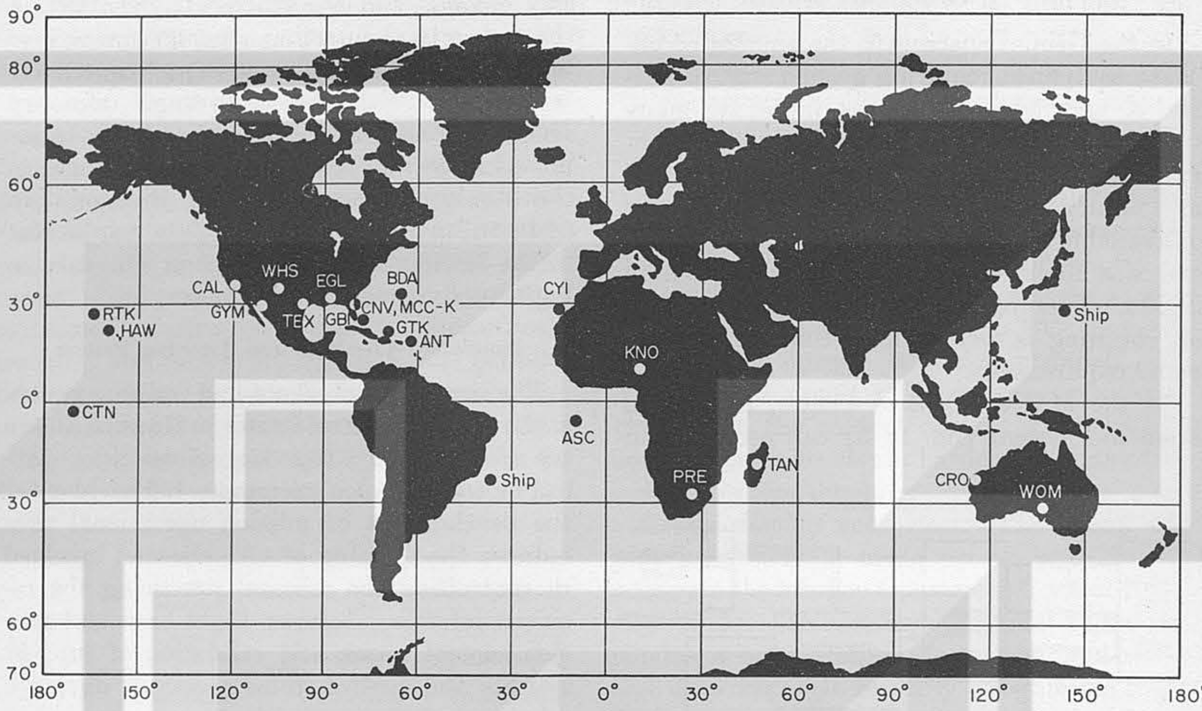


FIGURE 19-7.—Gemini network stations.

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.

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.

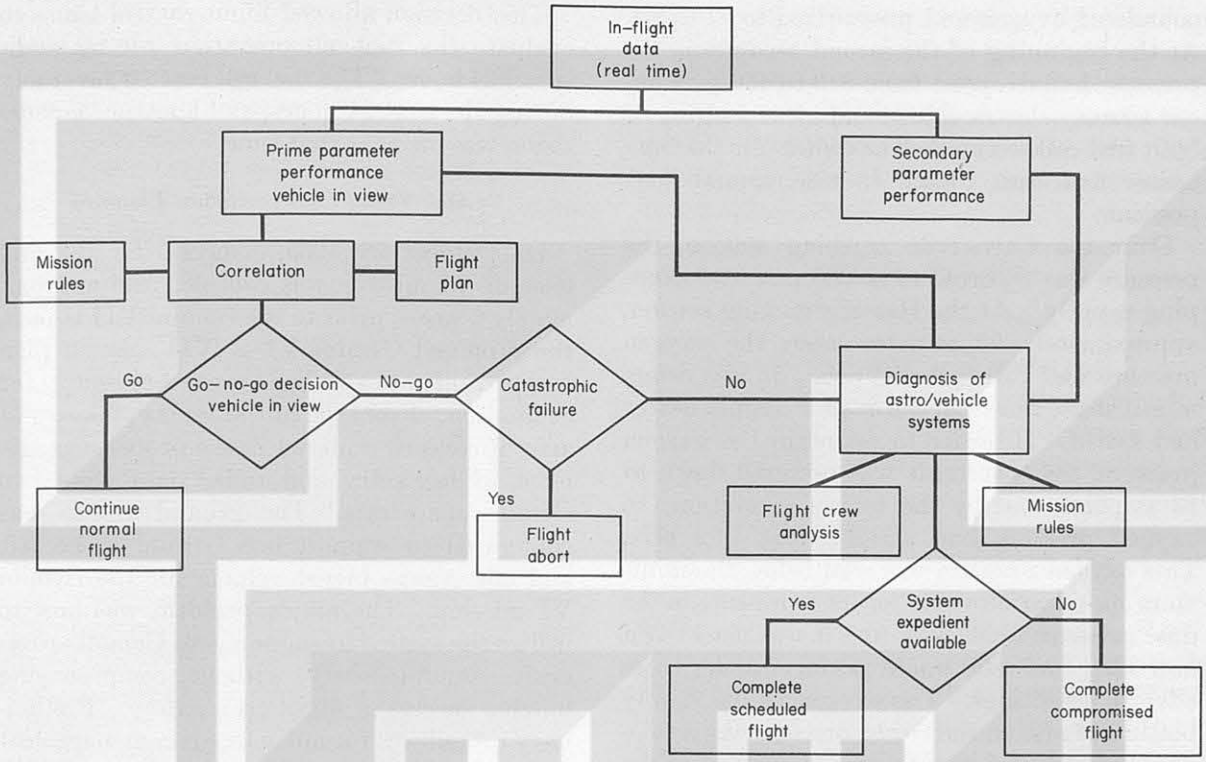


FIGURE 20–1.—The logic of flight-control decisions.

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